The Wind and Beyond

A Documentary Journey into the History of Aerodynamics in America

Volume II

James R. Hansen, Editor

with Jeremy Kinney, D. Bryan Taylor, Molly Prickett, and J. Lawrence Lee

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The airplane ranks as one of history's most ingenious and phenomenal inventions. It has surely been one of the most world changing. How ideas about aerodynamics first came together and how the science and technology evolved to forge the airplane into the revolutionary machine that it became is the epic story told in this six-volume series, *The Wind and Beyond: A Documentary Journey through the History of Aerodynamics in America.*

Following up on Volume I's account of the invention of the airplane and the creation of the original aeronautical research establishment in the United States, Volume II explores the airplane design revolution of the 1920s and 1930s and the quest for improved airfoils. Subsequent volumes cover the aerodynamics of airships, flying boats, rotary-wing aircraft, breaking the sound barrier, and more.

In 2005, the Society for the History of Technology awarded its first annual Eugene S. Ferguson Prize for outstanding and original reference works to *The Wind and Beyond.* The citation read in part:

"*The Wind and Beyond* is remarkable in its breadth of vision. Its purview includes not just aerodynamical theories and research results, but also innovative airships and airship components as well as the institutions in which and through which aerodynamics developed… Each [chapter] essay is original in two ways. First, each is a first-rate piece of scholarship in its own right. Second, the very decision to include these narratives is significant: they comprise roughly 10 percent of the contents of the volume, but they make the other 90 percent both accessible and meaningful to the nonspecialist reader, simultaneously enhancing the value of and enlarging the potential audience for the whole volume….*The Wind and Beyond* will be a boon both to students and to established scholars in several ways. Like many similar collections, it provides one-stop access to documents that were previously scattered in many different places. Going beyond other similar collections, however, *The Wind and Beyond* makes the documents intellectually as well as physically accessible…The end result is an eminently readable reference work, one that is truly, as its title suggests, the beginning of a journey rather than the end."

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Vol. II

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The Wind and Beyond: A Documentary Journey into the History of Aerodynamics in America

*Volume II: **Reinventing the Airplane***

James R. Hansen, Editor

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Foreword

Airplane travel is surely one of the most significant technological achievements of the last century. The impact of the airplane goes far beyond the realm of the history of technology and touches upon virtually every aspect of society from economics to politics to engineering and science. While space exploration often claims more public glory than aeronautics research, many more individuals have been able to fly within the Earth's atmosphere than above it. Thus aeronautics and air travel have had an enormous practical impact on many more individuals.

The first two volumes in the *Wind and Beyond* series and the succeeding four now in preparation all cover the impact of aerodynamic development on the evolution of the airplane in America. As the six-volume series will ultimately demonstrate, just as the airplane is a defining technology of the twentieth century, aerodynamics has been the defining element of the airplane. The forthcoming volumes will proceed roughly in chronological order, covering such developments as the advent of commercial airliners, flying boats, rotary aircraft, supersonic flight, and hypersonic flight.

This series is designed as an aeronautics companion to the *Exploring the Unknown: Selected Documents in the History of the U.S. Civil Space Program* (NASA SP-4407) series of books. As with *Exploring the Unknown*, the documents collected during this research project were assembled from diverse public and private sources. A major repository of primary source materials relative to the history of the civil space program is the NASA Historical Reference Collection in the NASA History Division. Historical materials housed at NASA Field Centers, academic institutions, and Presidential libraries were other sources of documents considered for inclusion, as were papers in the archives of private individuals and corporations.

The format of this volume also is very similar to that of the *Exploring the Unknown* volumes. Each section in the present volume is introduced by an overview essay that is intended to introduce and complement the documents in the section and to place them in a chronological and substantive context. Each essay contains references to the documents in the section it introduces, and many also contain references to documents in other sections of the collection. These introductory essays are the responsibility of Dr. Hansen, the series author and chief editor, and the views and conclusions contained therein do not necessarily represent the opinions of either Auburn University or NASA.

The documents included in each section were chosen by Dr. Hansen's project team from a much longer list initially assembled by the research staff. The contents of this volume emphasize primary documents, including long-out-of-print essays and articles as well as material from the private recollections of important actors in shaping aerodynamic thinking in the United States and abroad. Some key legislation and policy statements are also included. As much as possible, the contents of these volumes comprise an integrated historical narrative, though Dr. Hansen's team encourages readers to supplement the account found herein with other sources that have already or will become available.

Please note that the chapters in this series are numbered sequentially. Thus the first chapter in this second volume is referred to as chapter three and so forth.

For the most part, the documents included in each section are arranged chronologically. Each document is assigned its own number in terms of the section in which it is placed. As a result, for example, the fifteenth document in the first chapter of this volume is designated "Document 3-15." Each document is accompanied by a headnote setting out its context and providing a background narrative. These headnotes also provide specific information and explanatory notes about people and events discussed. Many of the documents, as is the case with Document 3-15, involve document "strings," i.e., Document 3-15 (a-e). Such strings involve multiple documents—in this case, five of them (a through e) that have been grouped together because they relate to one another in a significant way. Together, they work to tell one documentary "story."

The editorial method that has been adopted seeks to preserve, as much as possible, the spelling, grammar, and language usage as they appear in the original documents. We have sometimes changed punctuation to enhance readability. We have used the designation [abridged] to note where sections of a document have not been included in this publication, and we have avoided including words and phrases that had been deleted in the original document unless they contribute to an understanding of what was going on in the mind of the writer in making the record. Marginal notations on the original documents are inserted into the text of the documents in brackets, each clearly marked as a marginal comment. Page numbers in the original document are noted in brackets internal to the document text. Copies of all documents in their original form are available for research by any interested person at the NASA History Division or Auburn University.

While the *Exploring the Unknown* series has been a good model in many ways, this volume indeed represents an expedition into uncharted waters. Dr. Hansen and his team have crafted a landmark work that will not only be an important reference work in the history of aeronautics, but interesting and informative reading as well. We hope you enjoy this useful book and the forthcoming volumes.

Dr. Steven J. Dick
NASA Chief Historian
Director, NASA History Division

Acknowledgments

This volume represents the collected efforts of many members of an outstanding team. At Auburn University, a number of individuals provided generous assistance to Dr. James R. Hansen's project team. Dr. Paul F. Parks, former University Provost, strongly encouraged and supported the project from its inception, as did Dr. Michael C. Moriarty, former Vice President for Research. To undertake his leadership of the project, Dr. Hansen gave up his job as Chair of the Department of History, something he would not have felt comfortable doing without being certain that the administration of his department would be in the capable hands of worthy successors—first, Dr. Larry Gerber, and then Dr. William F. Trimble. Both Gerber and Trimble gave hearty and vocal support to Auburn's NASA history project. A number of colleagues in aerospace history gave help to the project, including Distinguished University Professor Dr. W. David Lewis and Dr. Stephen L. McFarland. Dr. Roy V. Houchin, who earned a Ph.D. under Hansen, lent aid and comfort to the project team from his vantage point inside the U.S. Air Force. A number of Hansen's graduate students helped the project in various ways, notably Andrew Baird, and Kristen Starr, as did Dr. David Arnold, also of the USAF, and Dr. Amy E. Foster of the University of Central Florida, who earned their Ph.D.s in aerospace history during the time period when this project was being conducted.

A number of people in the NASA History Division deserve credit. Jane Odom, Colin Fries, and John Hargenrader helped track down documents from our Historical Reference Collection. Nadine Andreassen provided much valuable general assistance and helped with the distribution. Interns Rebecca Anderson, Giny Cheong, Jennifer Chu, and Caitlin Gallogly also helped out tremendously.

Also at NASA Headquarters, Tony Springer in the Aeronautics Research Mission Directorate served as an invaluable sounding board on technical aeronautics issues. Now at the National Air and Space Museum, former NASA Chief Historian Roger D. Launius is owed a special debt of gratitude for providing the initial impetus and guidance for this worthy project.

A talented group of professionals handled the production of this book. Heidi Pongratz at Maryland Composition oversaw the copyediting of this book. Tom Powers and Stanley Artis at NASA Headquarters acted as invaluable coordinating liaisons with the graphic design group at Stennis Space Center. At Stennis, Angela Lane handled the layout with skill and grace, Danny Nowlin did an expert job proofreading, and Sheilah Ware oversaw the production process. Headquarters printing specialist David Dixon expertly handled this last and crucial stage of production.

Introduction to Volume II

Reinventing the Airplane

The history of aeronautical technology concerns much more than just the nuts and bolts of airplanes and spacecraft, much more than just the history of propellers and wings, more than the history of landing gear and jet engines, more than the ornithology of P-51s and F-22s, or the genealogy of X-planes. The history of flight technology is just as much a story of people and ideas as are histories dealing with any other topic related to society and culture. Scholars who write about the history of aerospace technology have plenty to say about the research, design, building, maintaining, and utilizing of flight vehicles, but their studies are no less human, no less connected to social, political, or cultural forces because they deal with technical matters.

The history of aeronautical technology tells us a lot about our existence as a thinking, dreaming, planning, scheming, aspiring, and playful species. As aerospace industry analysts William D. Siuru and John D. Busick noted in relation to their study of the evolution of modern aircraft technology, humankind's journey through the ages has been eased and accelerated, but also *complicated* by our unique and irrepressible knack for technology and invention.[1] From the stone ax and clay pot to the electron microscope and Human Genome Project, our technological creations have been ingenious, phenomenal, and occasionally—for good and for ill—of world-shaking significance.

This is, by all means, true for the airplane, one of the most ingenious and phenomenal—if slow-to-come—inventions in history, and surely one of the most world-shaking. In how many ways has the flying machine changed society? As Antoine de Saint Exupéry wrote in 1939, it has "unveiled for us the true face of the Earth."[2] It has brought people together, changed our economy, added an unprecedented new dimension to warfare, affected everything from government, public administration, international relations, international policies, manufacturing, marketing, mining, cities, and real estate, to media, railroads, ocean shipping, agriculture, and forestry. It has affected population, the family, religion, health, recreation, education, crime, and even sex.[3]

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1 William D. Siuru, John D. Busick, *Future Flight: The Next Generation of Aircraft Technology*, 2nd ed. (Tab Books, 1994), p. 3.

2 Antoine de Saint Exupéry, *Wind, Sand, and Stars,* translated from French by Lewis Galatiere (New York, 1939), p. 97.

3 One of the most remarkable analyses of the overall impact of aviation on the world came right after World War II with William Fielding Ogburn's *The Social Effects of Aviation* (Boston, 1946).

It has not been all for the good. In the 90 years from the tragic death of Lt. Thomas Selfridge in Orville Wright's airplane at Fort Myers, Virginia, in 1908, to the use of commercial airliners by terrorists to attack New York and Washington, DC in September 2001, there has never been a time when aviation did not know terrible accidents. Aviation has also brought human conflict to new depths of destruction. Despite this fact, the flying machine has always inspired "great expectations"—perhaps too great, given that it is, after all, just one of our *many* machines. Orville Wright summed up our loftiest ambitions for aviation when he said that it had been his hope (and that of brother Wilbur) that they were giving the world "an invention which would make further wars practically impossible." Unfortunately, history proved them wrong, and it did not take long to do it. As much as we admire the "Bishop's Boys" for their dream of a benevolent instrument of global peace, we are equally astonished by how such extraordinarily clear and logical thinkers could have been so ordinarily naive about the forces in the world around them. Maybe someday their vision will be proved right, and the world will discover, as the Wrights did, that peace, like flight, requires not brute power, but control and balance.

Contrary to what many engineers, most technocrats, and the great majority of industrial entrepreneurs seem to believe; contrary to people who use the Internet to read the morning paper, or to golfers who cannot enjoy a round of golf without riding in an electric golf cart and swinging a $500 titanium-headed driver; contrary to what many people in modern consumer society seem to believe, *technology is not inherently good*. In the words of one of the founding fathers of the history of technology as a discipline, Melvin C. Kranzberg, "technology is neither good, nor bad, nor is it neutral." Kranzberg called this "The First Law of the History of Technology."[4]

By its very nature, no technology is absolutely "good"—and none is bad. But neither is technology ever *neutral*. Depending on how we design technology, and even more on how we *use* technology, it will affect us, and change us, in some way. Whether the effects and changes turn out to be good or bad, or both inseparably together, is not predestined in the inherent qualities of the technology itself but rather depends on the broader context and values within which we live our lives. The human consequences of the airplane have gone far beyond what the Wrights or anyone else imagined in 1903. If it had been invented at a different time, or if it had been introduced into a different context or under different circumstances, the invention of the airplane might have led to quite different results. In this case, as in others, "The river of history could have cut a different canyon."[5]

Kranzberg's first law reminds us to "compare short-term versus long-term

[4] Melvin C. Kranzberg's classic essay "Technology and History: 'Kranzberg's Laws'" (in *Technology and Culture* 27 [July 1986]: 544-560) offers penetrating and witty analysis of the interactions between technology and its social context.

[5] See George Will, "What Paths Would the Nation Have Taken Had Taylor Lived?" *Washington Post*, 20 June 1991. Will's article raises fascinating issues relevant to the historical "what-ifs" and "might-have-beens."

results, the utopian hopes versus the spotted actuality, the what-might-have-been against what actually happened, and the trade-offs among various 'goods' and possible 'bads.'" All of these comparisons can be made "by seeing how technology interacts in different ways with different values and institutions, indeed, with the entire sociocultural milieu."[6]

But Kranzberg's first law is not the only "law" apropos to consideration of the history of flight technology. Another basic insight comes not from historians, but from those who work in the aerospace industry. There is a saying in that industry: "Requirements push and technology pulls." What this means, in a nutshell, is that the requirements of new missions, or even the need to improve upon current jobs, often drives engineers and scientists to work on the leading edge of technology. They are being "pushed" by ever more demanding requirements to find solutions to problems through the invention of new ideas. Technology then "pulls" by attracting those responsible for finding a way to meet the requirements for the newest concepts germinating in university, government, and commercial laboratories. For the push and pull to work together effectively, it takes forward-thinking planning smart enough to envision a way to use the new technology successfully in the design of a brand new aircraft.

This sequence of developments—(1) requirements [or needs], (2) technology, and (3) concepts—has been, and still is, basic to the technological progress of most modern aircraft—and perhaps *all* military aircraft. "Requirements push and technology pulls" may be just a more complicated way of the old saying, "Necessity is the mother of invention." There is considerable common sense, and historical validity, to this aphorism, but it is also true that it is not always the case—or always that illuminating of what actually is going on. Sometimes "necessity is *not* the mother of invention," but rather "*invention* is the mother of necessity." This was, in fact, Kranzberg's second law of the history of technology—and it makes us think about aerospace technology in some very important ways.

Once the Wrights invented the airplane, all sorts of things really needed to happen. Over the course of the next 30 years, as this volume shows, the airplane was in a sense *reinvented* as the Wrights' achievement was completely rethought and reworked by emerging groups of professionals dedicated to the airplane's improvement and greater practicality. What Kranzberg's second law illuminates is that "Every technical innovation seems to require additional technical advances in order to make it fully effective."[7] In the case of the airplane, the invention quickly necessitated all sorts of auxiliary technologies: advanced structures and materials, new wing shapes, streamlined aerodynamics, retractable landing gear, efficient low-drag engine cowlings, variable-pitch propellers, and much more. But perhaps even more importantly, it also necessitated new social forms and organizations (e.g., military

[6] "Kranzberg's Laws," 547-548.

[7] "Kranzberg's Laws," 548-549.

air services, airlines, airports, government bureaus, research laboratories, engineering curricula, and much more) in order to make the airplane more fully practicable. "While it might be said that each of these other developments occurred in a response to a specific need," Kranzberg claimed, "it was the original invention that mothered the necessity."

It is important to underscore one last, essential point before moving into this second volume of *The Wind and Beyond.* Just because the history of technology involves technology, it does not mean that technical factors always take precedence. In the real world, "soft" and "mushy" things such as politics and culture, for example, what bankers think can make them money or what activists say may harm the environment, often override good technical or engineering logic. *And they should.* Some might say that is why an American SST has never flown. That is why in the history of the American space program, all the thoughtful and well-intentioned talk about "the next logical step" has almost never led to it. After launching a man into space via Project Mercury, NASA said that the next logical step was to establish a permanent manned presence in low earth orbit, but instead the country landed men on the moon. After going to the moon via Project Apollo, the next logical step was to build an earth-orbiting space station along with a space shuttle to service it, but instead, the Nixon Administration decided that the country could not afford both and could manage temporarily with just the shuttle, although the space station had always been the shuttle's main reason for existing. After the shuttle, surely the next logical step was to build a space station, but once again, the country found reasons to postpone building one.

Clearly, logic does not determine the history of technology, and technologically "sweet" solutions do not always triumph over political and social forces. Historical logic, if we even want to use that phrase, is not the logic of engineers and scientists; it is the logic of Lewis Carroll's *Through the Looking Glass*. In that all-too-real fantasy land, Tweedledee explains logic to Alice: "Contrariwise, if it was so, it might be; and if it were so, it would be; but as it isn't, it ain't. That's logic." Tweedledee's logic is the only kind the American space program has ever known, or probably ever will.

So, when stumbling across a book about the history of aerospace technology, a reader should not be put off because he might think the book, and author's brain, is simply full of engineering tables and equations. There is a lot of "soft and mushy stuff" there also. It is what makes our species human, an essential part of what makes us brilliant, and a large part of what drives us nuts. It is what makes the history of technology one of the most complex and fascinating subjects one can possibly study.

There may be a bigger message here as well. In 1998, Microsoft's Bill Gates said about the Wright brothers' invention in a speech he gave at *Time Magazine*'s 75th anniversary celebration of the airplane that, "We have to understand that engineering breakthroughs are not just mechanical or scientific, they are liberating forces that can continually improve people's lives."

Let us hope that the flying machine, in the 21st century, does "free" us, in *more positive* ways, than it has been able to do in the century just passed. There is no guarantee that it will. But like our dear Wright brothers gazing into their future that is our present, let us proceed into this new millennium with optimism that our globe's political environment will improve so that our future generations can enjoy our technical advances and not be destroyed by them. It is something in which the Wrights would want us not only to apply our best problem-solving and inventive skills, but also in which to invest our limitless capacity to hope and to trust.

Biographies of Volume II Contributors

James R. Hansen, Professor of History and Director of the Honors College at Auburn University, has written about aerospace history for the past 26 years. His newest book, *First Man: The Life of Neil A. Armstrong* (Simon & Schuster, 2005), offers insight into the life and times of the first man on the Moon, but also sheds new light on many of the aerospace events and personalities that shaped America in the second half of the 20th century. His two-volume study of NASA Langley Research Center—*Engineer in Charge* (NASA SP-4305, 1987) and *Spaceflight Revolution* (NASA SP-4308, 1995) earned significant critical acclaim. His other books include *From the Ground Up* (Smithsonian, 1988), *Enchanted Rendezvous* (NASA Monographs in Aerospace History #4, 1995), and *The Bird Is On The Wing* (Texas A&M University Press, 2003).

Jeremy R. Kinney is a curator in the Aeronautics Division, National Air and Space Museum, Smithsonian Institution, and holds a Ph.D. in the history of technology from Auburn, with a specialization in aerospace history. Prior to joining NASM, he served as the American Historical Association NASA Fellow in Aerospace History.

D. Bryan Taylor is a doctoral candidate in the history of technology at Auburn. He holds an M.A. in history from Brigham Young University.

Molly F. Prickett holds a B.A. degree in English from Auburn University.

J. Lawrence Lee holds B.M.E., M.A., and Ph.D. degrees from Auburn. He is an engineer-historian at the Historic American Engineering Record, a branch of the U.S. National Parks Service, and is former chair of the History and Heritage Committee of the American Society of Mechanical Engineering.

Chapter Three

The Design Revolution

Destination Document:

The DC-3 freed the airlines from complete dependency on government mail pay. It was the first airplane that could make money just by handling passengers. With previous aircraft, if you multiplied the number of seats by the fares being charged, you couldn't break even—not even with a 100 percent full load. Economically, the DC-3 let us expand and develop new routes where there was no mail pay.

C. R. Smith, President of American Airlines, ca. 1938, quoted by Robert J. Serling,* Eagle: The Story of American Airlines *(New York: St. Martins, 1985), p. 110.

On a Streamline to the DC-3

In 1935, the National Aeronautics Association (NAA) awarded its prestigious Collier Trophy to the Douglas Aircraft Company for its new family of twin-engine transport planes. The trophy, awarded annually, recognized the greatest achievement in American aviation. In doing so, the NAA acknowledged not only a superlative aircraft but also a revolution in the design of airplanes. In the three decades since the achievement of flight by the Wright brothers, the flying machine had evolved from a fragile contraption of wood, wire, and cloth into a sleek and sturdy

The Douglas DC-3 became the most popular and reliable propeller-driven airliner in aviation history and its appearance in the mid-1930s marked the culmination of the design revolution in American aircraft. (SI Negative No. ACC 1997-0033) form of modern public transportation. The Douglas transport honored by the 1935 Collier Trophy featured all-metal construction, an enclosed streamlined shape with cantilever monoplane wings, cowled radial engines, controllable pitch propellers, high-lift flaps, and retractable landing gear. One of the great airplanes of all time, the Douglas DC-3, represented not only what was to become one of the truly classic designs in aviation history but also the culmination of a creative process involving an almost total "reinvention" of the airplane.

The reinvention of the airplane flowed from many technological streams that converged in the years following World War I. Aviation technology had progressed in significant ways in the fifteen years after Kitty Hawk, but strut-and-wire-braced open-cockpit biplanes with fixed landing gear still pretty much reflected the state of the art. Aircraft designers worked from a very limited understanding of aerodynamics, and their products were primarily the result of empirical, even "cut-and-try," engineering. A few potential breakthroughs appeared during the war, but frail wooden biplanes covered with fabric, braced by wires, powered by heavy water-cooled engines, and driven by hand-carved wooden propellers still ruled the airways in 1918. The principles of aeronautical engineering had yet to be fully discovered, and only a few programs at major schools such as Massachusetts Institute of Technology (MIT) and the University of Michigan existed to find these ideas and teach them to students. Aircraft design remained a largely intuitive practice requiring bold speculation and daring, in both a financial and technological sense.

In terms of engineering, a number of unknowns endangered aircraft performance. Even those few professional aeronautical engineers who did exist did not know for sure how to reduce engine drag without degrading cooling. They did not know with certainty how to shape wings to increase lift or diminish the effects of turbulence. They did not know how and when flaps, ailerons, and other control surfaces worked best. In addition, they did not even know if it was worthwhile to retract landing gears—according to certain pundits, the added weight and structural complexity of a retractable undercarriage was not worth the saving in air resistance. Aeronautical engineers suspected that substantial increases in aerodynamic efficiency might follow on the heels of correct answers to just a few of these technical concerns, but they did not know exactly how, or even whether to try, to get at them.

In significant ways, the process of "reinventing the airplane" that unfolded in the 1920s and 1930s grew naturally out of these questions and suspicions. Institutionally, the process took shape and gained momentum as a growing body of professionals began to attack the problems obstructing the immediate progress of aviation, particularly those vexing the fledgling military air services and aircraft manufacturing and operating industries. Major new institutions played critical roles. In the United States, the National Advisory Committee for Aeronautics (NACA) and the research agenda of its Langley Memorial Aeronautical Laboratory was formative in defining what needed to be accomplished. Following its establishment in 1915, the NACA conducted research into basic aerodynamic, structural, and propulsion

problems. Solutions to these problems in the interwar period led in fundamental ways to the design and operation of advanced aircraft like the Douglas DC-3, which were safer, faster, higher-flying, and generally more versatile and dependable than any aircraft ever flown. New research and development tools, especially the wind tunnel, played a key role in the process of reinventing the airplane, as engineers worked to create the most effective streamlined shape possible.

One of the central problems addressed by the airplane design revolution of the 1920s and 1930s was aerodynamic drag reduction. In his history of aerodynamics, John D. Anderson has asserted that, "The major thrust in the age of the advanced propeller-driven airplane can be summarized in a word: *streamlining*.[1] Aircraft designers of the interwar period understood streamlining, in Anderson's words, as "adopting a form that is so shaped and so free of protuberances that it produces no eddies in the airflow over it." In essence, streamlining meant drag reduction, and drag reduction enabled aircraft to fly farther and carry heavier loads. Reduced drag would also allow aircraft to fly faster without expending more energy and fuel resources, meaning that the same size airplane could perform the same work with less energy cost. All the way back to the days of Sir George Cayley, aeronautical experts had been aware of the presence of aerodynamic drag (i.e., the resistance to the forward movement of body in a fluid like air). But because the sources and even the actual composition of aerodynamic drag were not well understood, few quantitative or otherwise nonintuitive methods were available to pioneering designers to lessen the effects of drag on aircraft performance.

As with most revolutions, the aircraft design revolution of the 1920s and 1930s drew upon earlier ideas not materialized in tangible ways during their own day. Without question, the reinvention of the airplane depended on this sort of heritage. It drew inspiration as well as practical lessons from what pioneers in theoretical aerodynamics discovered back in the late 19th century.

In key respects, it was the work of English engineer and automobile maker Frederick W. Lanchester that inaugurated the first great age of aerodynamic theory. Curiously, Lanchester's work was largely experimental and nonquantitative, involving a program of experiments

Frederick W. Lanchester's experimental and nonquantitative search for a "streamlined form" resulted in his 1907 pioneering book, *Aerodynamics.* (SI Negative No. 45374-C)

[1] John D. Anderson, Jr., *A History of Aerodynamics and Its Impact on Flying Machines* (Cambridge University Press, 1997), p. 319.

German physics professor Ludwig Prandtl pioneered the academic study of drag and the use of streamlining to improve aerodynamic efficiency overall in the 1920s and 1930s. (SI Negative No. 74-10601)

initiated in the early 1890s, with cambered airfoils. The principle objective of Lanchester's work was to define an efficient low-drag shape with smooth contours that offered no resistance to the passage of airwaves along its surface; he called this shape a "streamline form." Lanchester realized that irregularity in an airfoil's shape caused disturbances in an airflow and resulted in a significant amount of drag. Unfortunately, Lanchester's work did not receive the attention it deserved at the time, in part because his theories lacked the type of truly quantitative methodology that could have practically aided early aircraft designers. Document 3-1 provides excerpts from Lanchester's 1907 book, *Aerodynamics*, featuring his concept of streamlining.

Another formative aerodynamic thinker for the subsequent revolution in airplane design of the 1920s and 1930s was the influential German physics professor Ludwig Prandtl, much of whose work supported Lanchester's ideas about streamlining. The best understanding of aerodynamic drag at the turn of the century was based on what little was known about the performance of hydrodynamic forms. Yet both Lanchester and Prandtl realized that air and water were far different media—for one example, air created more frictional resistance than water. Addressing this issue at the Third International Congress of Mathematicians at Heidelberg in 1904, Prandtl observed that an ideal airflow moved in smooth parallel layers. But in real-world circumstances, such as the flow of air over a wing surface, this so-called laminar flow became turbulent, resulting in significant drag. What Prandtl reported in 1904—to a largely disinterested audience of purer-minded mathematicians—was that an airflow changed in dramatic fashion from zero to constant velocity in a very thin "transition layer" right next to the surface of the airfoil. In this extremely narrow zone, later to become known as the "boundary layer," transition from laminar to turbulent flow affected aircraft performance profoundly.[2], Prandtl's boundary-layer hypothesis of 1904 (excerpted as Document 3-2) offered major new opportunities for aircraft designers and aeronautical engineers. What

[2] The first known reference to the term "boundary layer," according to the *Oxford English Dictionary*, came in 1921, when Dutch fluid mechanics specialist J. M. Burgers wrote in his country's *Proceedings of the Royal Academy of Science* (Proc. K. Acad. Wetensch [Amsterdam]) that "we can calculate the distribution of the vorticity in the boundary layer, when we suppose the velocity outside the boundary later to be known." The second recorded use of the term, according to the *OED*, came in an article appearing in *Flight* magazine on 20 November 1924: "The deductions from the boundary layer theory gave a rather poor approximation to the truth."

Grover Loening (left), seen here with Orville Wright (right) in 1913, was one of the earliest aeronautical engineers employed by the United States government and advocated the streamline form as the ideal shape for an airplane. (SI Negative No. 80-5407)

he taught about airflow in the boundary layer near the surface of an aerodynamic body provided the first rational basis for calculating what came to be known as "skin friction drag." His theory also promoted vastly improved understanding of the dynamics of flow separation around an aerodynamic body, which was a major source of " form drag." Prandtl's students also learned that the best way to lessen the effects of "parasitic drag" (i.e., the combination of skin friction drag and form drag) was to delineate smoother contours for wings and other aerodynamic forms. In sum, Prandtl's work, like Lanchester's, not only highlighted the problem of drag, it underscored the practical advantages of streamlining.

While the work of Lanchester and Prandtl provided an essential theoretical basis for understanding drag, the concept of streamlining saw little practical application in the vast majority of aircraft designs emerging from World War I. Angular shapes and drag-producing protuberances were still ubiquitous features. Nevertheless, the idea of improving aircraft performance through streamlining did catch the attention of some people in the aeronautical community during the war. One such individual was Alexander Klemin, a civilian research working for the U.S. Army Air Service, who explored the idea of streamlining in a 1918 textbook entitled *Aeronautical Engineering and Airplane Design*, one of the first works of its kind published in the United States (see Document 3-3). A graduate of the aeronautical engineering program at MIT, Klemin came to head the Aeronautical Research Department at the U.S. Army Air Service's engineering facility at McCook Field, in Dayton, Ohio.

Aircraft such as the biplane, strut-and-wire braced Martin MB-1 Bomber represented the state of the art in aerodynamic design circa 1917. The designers of the aircraft, Laurence D. Bell (left), Glenn A. Martin (second from right), Donald W. Douglas (right), and James Kindleberger (not pictured), would use the advances of the design revolution to create vastly different aircraft. (Martin test pilot, Thomas E. Springer, is second from left.) (SI Negative No. 43067)

Equating "resistance" to "drag," he noted in his groundbreaking textbook that a streamline form was the ideal shape for an airplane. But major obstacles blocked the achievement of ideal streamlining, notably the realities of conventional design practice, which necessitated the incorporation of mammoth and strangely shaped engines, the bodies of pilots and passengers, plus messy structural items such as bracing wires and struts. Somehow, Klemin concluded, many of these obstacles would have to be overcome if an airplane 's aerodynamic effectiveness were ever to improve much.

After the war, more and more aviation advocates placed an emphasis on aircraft efficiency, culminating in 1922 when pioneering French aircraft manufacturer Louis Breguet issued a clarion call for the development of streamlined designs. In an address before the Royal Aeronautical Society in London entitled "Aerodynamic Efficiency and the Reduction of Aircraft Costs" (see Document 3-4), Breguet outlined the problem in both technical and economic terms, and he articulated various social incentives for the improvement of civilian aircraft in the postwar period. Breguet believed that an improvement in the lift-to-drag ratio, which he called the "fineness" of an aircraft design, would enhance an airplane's economic potential by

efficiently extending its range. To achieve an economic range through the improvement of the lift-to-drag ratio, Breguet called for the use of streamlining, specifically suggesting such measures as the use of retractable landing gear. He even defined a target "fineness" ratio for commercial aircraft—a ratio ultimately achieved just over a decade later in the design of the Douglas DC series.

Another factor that proved to play a critical role in spurring both the development and general acceptance of streamlined aircraft designs was performance at air races, which became enormously popular in Europe and America in the late 1920s and 1930s. At the international level, the greatest of these races, such as the Schneider Trophy competition, presented opportunities for benign competition between world powers; victories won and records set were perceived by all as signs of national technological prowess. Within a given country, races held similar significant meanings; for example, in the United States, the Army and Navy worked feverishly to come in first in the competitions, hoping through first-place finishes and world and national records to secure greater political support for the development of the air forces.

French aircraft builder Louis Breguet made one of the strongest cases for the social and economic benefits of aerodynamic streamlining in a 1922 speech delivered before the Royal Aeronautical Society entitled, "Aerodynamical Efficiency and the Reduction of Transport Costs." (SI Negative No. 77-542 or 78-13907)

The air races fostered experimentation and rapid technological development, much as automobile races did and still do. Both the Army and Navy used wind tunnels to refine their racers to get the utmost speed out of them. Many racing aircraft employed groundbreaking devices that would become part of the modern airplane; these included streamlined designs, cantilever wings, and retractable landing gear. In 1923, an American airplane, the U.S. Navy Curtiss CR3s, took first and second place in the Schneider Cup international seaplane race in Cowes, England, the winner at a top speed of 177.38 miles per hour (mph). The aircraft employed new technology, including flush wing radiators and a solid aluminum alloy propeller. This propeller incorporated aerodynamically advanced thin airfoil sections based on the designs of Dr. Sylvanus A. Reed. At the tips of the propeller blades, the airflow reached speeds approaching the speed of sound. But it was the sleek, streamlined fuselage of the aircraft that really captured everyone's eye. As contemporary observers noted, everything about the airplane spelled "speed" (see Document 3-5). Another Navy Curtiss racer, the R2C-1, won the 1923 Pulitzer Trophy Race in St. Louis, Missouri. In doing so, the slender little airplane set new world speed records of 243.8 mph for 100 km and 243.7 mph for 200 km over a closed circuit. No other aircraft flying at the time could boast cleaner aerodynamics. Curtiss designers got the airplane's zero-lift drag coefficient

The streamlined fuselage, flush wing radiators, and solid aluminum alloy Reed propeller of the navy Curtiss CR-3 racer highlighted the role of aerodynamics in high-speed aircraft design. Lt. David Rittenhouse (in cockpit) flew the CR-3 to victory in the 1923 Schneider Cup international seaplane race in Cowes, England. (SI Negative No. A-47217)

($C_{D,O}$)—considered by many experts as the best indicator of the aerodynamic cleanness or refinement of an aircraft.[3] In fact, it ranks among the lowest values in the entire history of aerodynamics for propeller-driven aircraft—lower than that enjoyed by four of the sleekest aircraft produced by the airplane design revolution of the 1930s: the Lockheed Vega (0.0278), Lockheed Orion (0.0210), Boeing 247D (0.0212), and Douglas DC-3 (0.0249). Only a few American prop-driven planes have ever achieved lower zero-lift drag coefficient. Noteworthy among these were the Beechcraft D17S four-place monoplane of 1939 (0.0182), North American P-51D fighter plane of 1944 (0.0163), and Beech Bonanza V-35 general aviation aircraft of 1970 (0.0192).

[3] For a clear explanation of the significance of the zero-lift drag coefficient, see Laurence K. Loftin, Jr., *Quest for Performance: The Evolution of Modern Aircraft* (Washington, DC: NASA SP-468, 1985), pp. 4-5, 158-160. Loftin defined this coefficient *Cd, o* as "a nondimensional number that relates the zero-lift drag of the aircraft, in pounds, to its size and the speed and altitude at which it is flying. Generally speaking, the smaller the value of the number, the more aerodynamically clean the aircraft." As useful as it is as a measure of aerodynamic refinement, the significance of the zero-lift drag coefficient is limited in application because it is based on wing area and because, for a given wing area, several different sizes of fuselage and tail may be employed. Thus, as Loftin made clear, "differences in zero-lift drag coefficients may be interpreted as a difference in aerodynamic refinement" when, in fact, the difference may be the product of differences in the ratio of "wetted" area (i.e., the area of the entire body that comes in direct contact with the airflow) to wing area (Loftin, pp. 158-159).

The challenges and glories of air racing definitely inspired streamlining, but in the minds of many, the association of streamlined shapes with racing implied that the technology of highly sleek and efficient aerodynamics belonged only to the realm of high-speed applications. But this was hardly the case. Advanced aerodynamic refinement in the form of streamlining was already beginning to play a critical role in the design of a number of revolutionary civil and commercial aircraft as well, a technological development that would help stimulate the first great age of aviation as an effective mode of mass public transportation.

One of the first nonmilitary or racing aircraft to demonstrate streamline design was the boldly innovative Lockheed Vega of 1926, designed by engineering genius John K. Northrop. Self-taught, highly creative, and remarkably proficient in transferring a speculative design from his "mind's eye" to reality, "Jack" Northrop stood ready at the leading edge of the streamlining movement in the United States, albeit from a direction quite different from that of building racers or being immersed academically in aerodynamic theory. His "feel" for aerodynamic refinement was more "aesthetic," in that it involved a heightened sensitivity for what looked beautiful and was in "good taste." Not that Northrop did not base his designs in engineering practicality. The key to the Vega's aerodynamic efficiency rested in the techniques he used in its construction. Lightweight, strong, and remarkably streamlined, the Vega represented a dramatic departure from previous designs in that it was a single-engine, high-wing cantilever monoplane with a semimonocoque (*monocoque* meaning "one shell") plywood stressed skin fuselage. The cantilever wing featured internal bracing. A major new feature incorporated in later Vega airplanes was a circular cowling surrounding the 450-horsepower Pratt & Whitney Wasp air-cooled engine. This cowling concept derived from the NACA's systematic identification of efficient cowling forms, ones that not only reduced drag but also at the same time, improved cooling of the engine. The only major nonstreamline element of the Vega's profile was its fixed landing gear, but in later versions, even that had fairings called "pants" around its wheels, which also helped to reduce drag. As mentioned previously, the airplane benefited from a very low zero-lift drag coefficient, which helped it to reach a maximum speed of 190 mph. Airlines used the plane to fly passengers (six at a time), and a number of pilots, including Amelia Earhart and Ruth Nichols, broke records in it. In a Vega named *Winnie Mae*, Wiley Post flew solo around the world in the summer of 1933, need-

Jack Northrop's innovative construction techniques facilitated his design of highly efficient and aesthetic streamline aircraft designs such as the Lockheed Vega and the Northrop Alpha. (SI Negative No. 75-5442)

The Lockheed Vega of 1926 represented a dramatic departure from previous designs with its monocoque construction, NACA cowling, and teardrop wheel pants. Wiley Post flew the Lockheed Vega 5-B "Winnie Mae" solo around the world in only seven and one-half days during the summer of 1933. (SI Negative No. A-47516)

ing only seven and one-half days. In Document 3-6, Northrop recorded some of his views on streamlining and its relationship to construction technology as embodied in the Lockheed Vega.

The development of streamlined aircraft did not come all at once. Perhaps more than any other single element, it awaited widespread recognition of the advantages of streamlining by some of its most influential leaders, if not the aeronautical community as a whole (see Document 3-7). The initial focus of the efforts to reinvent the airplane hinged on the independent development of specific airplane parts, what aviation historian Richard K. Smith referred to as "shelf items.[4] These included components such as airfoils, engine cowlings and nacelles, flaps, propellers, fillets, and retractable landing gear. Government researchers, engineers in industry, and lone inventors played a part in developing these innovations—for the most part, separately and individually. Designers and manufacturers would then incorporate and synthesize the new technology into final aircraft design as needed. Despite their "shelf-item" status, these innovations were not truly "stand-alone" technologies that could simply be put on the airplane; they had to be integrated into a total design. One important function performed by the NACA was full-scale wind tunnel testing of the aerodynamic advantages inherent to all these various components as they were located on new airplanes, which provided data tables and computations so designers could use the information for a variety of future designs. Richard K. Smith further

[4] Richard K. Smith, "Better: The Quest for Excellence," in *Milestones of Aviation*, ed. John T. Greenwood (New York: MacMillan, 1989), p. 240, 243-4.

observed that the leading developer of many of these "shelf items" was the NACA itself. In his view, "no other institution in the world contributed more to the definition of the modern airplane".[5] The improved components, along with the fundamental understanding of the mechanism of drag that resulted from NACA research, became widely disseminated throughout the aeronautical engineering community and the aircraft industry during this crucial transition period.

Without question, the NACA contributed greatly to virtually every aspect of airplane aerodynamics during the late 1920s and early 1930s. Some of the most important work was in the development of improved airfoils, a subject treated in detail in the next chapter. But the NACA team at Langley laboratory in Virginia also worked on many other areas of aircraft component improvement. For example, some of the first projects in the new Propeller Research Tunnel (PRT), which began operations in July 1927, concerned improvement of the aerodynamic qualities of engine installations, struts, and landing gear. The first important step in this work was to determine through systematic experimental parameter variation exactly how much drag a particular component produced.

The test program that initiated the NACA's move toward the widespread reduction of drag focused on engine cowlings. Aerodynamic tests on one of the Army's small Sperry Messenger airplanes in Langley's PRT in late 1927 revealed that the exposed cylinders of the air-cooled radial engine caused 17 percent of all drag plaguing the aircraft. The most obvious solution to this problem was simply to cover the engine with some sort of streamlined shroud or cowling. But such a covering, it was thought, would restrict airflow past the cylinders and cause the engine to overheat. With this in mind, cowlings were not used very often. But the drag problem grew into a more and more critical concern, so much so that at the NACA's first annual manufacturers' conference, held at Langley in May 1926, representatives of the aircraft industry and the U.S. Navy's Bureau of Aeronautics identified it as a national priority. The challenge for the NACA was to define a form of cowling that significantly improved aerodynamic efficiency without degrading cooling.[6]

Documents 3-8, 3-9, and 3-10 all relate to the NACA's cowling program, for which the NACA won its first Collier Trophy in 1929. The first of these documents is from the autobiography of Fred E. Weick, the Langley engineer who masterminded the development of the NACA's low-drag cowling. In it, he recalled in

[5] Smith, "Better," p. 240.

[6] For a complete analysis of the history of the NACA cowling program, see James R. Hansen, "Engineering Science and the Development of the NACA Low-Drag Engine Cowling," in *From Engineering Science to Big Science: The NACA and NASA Collier Trophy Research Project Winners,* ed., Pamela E. Mack (Washington DC: NASA SP-4219, 1998), pp. 1-27. This chapter is an updated and expanded version of Chapter 5 of Hansen's *Engineer in Charge: A History of the Langley Aeronautical Laboratory, 1917-1958* (Washington DC: NASA SP-4305, 1987), entitled "The Cowling Program: Experimental Impasse and Beyond," pp. 123-139.

The inclusion of NACA Cowling No. 10, seen here undergoing tests in the 20-foot tunnel in September 1928, into the design of the Martin B-10 bomber increased the airplane's maximum speed by 30 mph to 225 mph and reduced its landing speed significantly. (NASA Image No. 1974-L-02730)

The NACA won its first Collier Trophy in 1929 for its innovative work on low-drag cowlings. (NASA Image No. L-1990-04348)

The inclusion of the NACA low-drag cowling to the Texaco Lockheed Air Express allowed Frank Hawks to establish a new Los Angeles to New York nonstop record of 18 hours, 13 minutes in February 1929. (SI Negative No. A31250-G; Videodisc No. 2B-11966)

Hubert C. H. Townend of the British National Physical Laboratory developed a low-drag engine cowling in 1929. Known as the "Townend ring," the design reduced drag with little degradation of engine cooling and was widely adopted in Europe and America. (SI Negative No. 76-17366)

The alternative to the NACA cowling was the Townend ring, seen here mounted on a Boeing P-26 Peashooter perched in the Langley Full-Scale Tunnel for aerodynamic testing. (NASA Image No. L-09819)

The NACA worked to improve existing designs such as the Fokker Trimotor with the addition of experimental low-drag cowlings and streamline engine nacelles in 1929. (NASA Image No. L-03333)

extraordinary detail the careful experimental process by which the NACA arrived at its award-winning shape. The second involves a string of documents, the first of which relates to the stunning performance in February 1929 of a Lockheed Air Express, a derivative of the Lockheed Vega. Equipped with a NACA low-drag cowling that increased its speed from 157 to 177 mph, this aircraft, piloted by Frank Hawks, established a new Los Angeles to New York nonstop record (18 hours and 13 minutes). The third presents a trio of documents concerning the appearance of other new forms of engine cowlings that came to rival the NACA design.

During the process of developing better components for an aircraft, the work on one "shelf item" often led to study of different problems. This situation happened in regard to the NACA cowling, as research engineers found that the cowl did not benefit all aircraft equally: its effectiveness depended on the shape of the airplane behind it. While these complexities spurred on additional work related to cowlings, it also led to other lines of investigation. The original purpose of NACA's Langley Propeller Research Tunnel (PRT) was to develop improved propellers. But early tests in the PRT demonstrated that the machine could be used to study other problems, notably the overall effect of a propeller in combination with the placement of an engine and the placement of engine nacelles in relation to wings. These unintended lines of research led not only to data very significant to aircraft designers but to a few entirely new concepts. In studying the overall effect of the propeller in combination with the placement of the engine, for example, NACA researchers came to express their results in a new quantitative relationship they called "propulsive efficiency." This term reflected the overall propeller output in reference to its mounting on the aircraft. Propulsive efficiency calculations soon became a standard analytical tool in the design of propeller-driven airplanes and played a crucial role in the development of modern streamlined aircraft.

As Fred Weick recalled in our excerpt from his autobiography (Document 3-8), another one of these inadvertent research programs, one devoted to the proper placement of engine nacelles, originated in 1929 when the PRT team at Langley began testing cowling forms on a multiengine aircraft, the Fokker trimotor. When speed trials with the big transport proved extremely disappointing, Weick and his associates started to wonder how the position of the nacelles with respect to the wing might be affecting drag. This was a critical design revolution, especially for multiengine aircraft, as big commercial and military aircraft were bound to be. In the case of the Fokker (as well as the Ford) trimotor, the original design location of the wing engines was slightly below the wing's surface. As the air flowed back between the wing and nacelle, the expansion required was too great for the air to flow over the contour smoothly. The NACA Langley flight research division, in association with the PRT team, tried fairing in this space, but they achieved only a small improvement.

Eventually the NACA's systematic empirical approach yielded dividends. With the help of his assistants, Weick laid out a series of model tests in the PRT with NACA-cowled nacelles placed in 21 different positions with respect to the wing:

above it, below it, and within its leading edge. The resulting data on the nacelle's effect on the lift, drag, and "propulsive efficiency" of the big Fokker trimotor made it clear that the optimum location of the nacelle was directly in line with the wing and with the propeller fairly well ahead. Although their primary emphasis was on drag and improved cooling, the tests at Langley also confirmed that NACA cowling No. 10—the form that completely covered the engine but still managed to cool it by directing air to its hottest spots through ducts and baffles—could actually increase the lift of the airplane's wing in some cases, if the engine was situated in the optimum position.[7]

In transmitting this important information confidentially to the Army, Navy, and industry, the NACA helped build a several-months lead for American aircraft designers over rival European companies. After 1932, nearly all American transport and bombing airplanes—including the Martin B-10, Douglas DC-3, Boeing B-17, and many other famous aircraft of the era that followed—employed radial wing-mounted engines with the NACA-cowled nacelles located approximately in what Weick and his associates had identified as the optimum position. Without question, this combination led to an entire new generation of highly effective airplanes with which airlines for the first time could become financially self-supporting and no longer in need of government subsidies. Paradoxically, as Document 3-11 suggests, some leaders of the U.S. aircraft industry at the time did not fully credit NACA research for this essential contribution to what was quickly becoming a blossoming design revolution.

Another critical component in the reinvention of the airplane was the development of high-lift devices, primarily flaps. Essentially a section of the wing's trailing edge that could be hinged downward to increase the camber of the wing, a flap promised to boost lift and reduce the aircraft's stalling speed. It permitted a better angle of approach and lower speed on landing, important considerations not only for aircraft efficiency but also for safety. The earliest trailing edge flaps simply increased aircraft drag during landing and approach instead of boosting lift, but the desirable element was a structure that did both. In the 1930s, the introduction of high-lift devices such as flaps allowed engineers to design aircraft with higher wing loadings (i.e., the ratio of the gross weight of an airplane to the total planform area of its wing or wings). This was a new aerodynamic technology that proved crucial to the transition from biplanes to monoplanes.

7 We considered including the following NACA technical reports from 1932 in this chapter collection of documents, but, due to length, determined, it would suffice simply to reference them in a footnote: Donald H. Wood, "Tests of Nacelle-Propeller Combinations in Various Positions with Reference to Wings. Part I. Thick Wing—NACA Cowled Nacelle—Tractor Propeller," NACA *Technical Report 415* (Washington, 1932); and Donald H. Wood, "Tests of Nacelle-Propeller Combinations in Various Positions with Reference to Wings. II.—Thick Wing—Various Radial-Engine Cowlings—Tractor Propeller," NACA *Technical Report 436* (Washington, 1932).

It is perhaps curious to those unfamiliar with the history of aircraft technology to learn that it took so long for wing flaps to become commonplace. The basic idea for them had been around since the aileron device invented by Frenchman Henry Farman in 1908 to get around the Wright brothers patent for wing warping. Farman invented what came to be known as the aileron. Various inventors had created aileron devices prior to Farman, but they had all been separate rotatable surfaces placed in front of (not behind) a wing or sometimes, in the cases of biplanes, even between them. What distinguished Farman's aileron was that the surface was integrated directly into the wing as aircraft designers still incorporate them today. The modern definition of an aileron says that it is "a moveable control surface or device, one of a pair or set located in or attached to the wings on both sides of an airplane, the primary usefulness of which is controlling the airplane laterally or in roll by creating unequal or opposing lifting forces on opposite sides of the airplane.[8] But this definition applies to the word *aileron* only after Farman's innovation of 1908.

A few ideas for flaps sprouted during World War I, but not actually that many, since flying speeds were still too low to make such devices useful. As John D. Anderson explains in his history of aerodynamics, "the low wing loadings made flaps essentially redundant and most pilots rarely bothered to use them.[9] After the war, engineers in various countries pursued new concepts for high-lift mechanisms. In a case of nearly simultaneous invention occurring just before 1920, Dr. Gustav V. Lachmann in Germany and Frederick Handley Page in England invented the "wing slot." Lachmann's original design called for simply a long, spanwise slot located near the leading edge. His idea was to create a pressure differential between the lower and upper surfaces of the wing. This differential would impel a high-energy jet of air through the slot. This jet flowed tangentially over the top surface of the wing, along the way energizing the transition layer (i.e., boundary layer); delayed flow separation to much higher angles of attack; prevented stall; and increased lift in wind tunnel tests by a whopping 60 percent. Frederick Handley Page, who pooled patent rights with Lachmann in 1921 (Lachmann actually went to work for Page's company in 1929), went one step further. He combined a slot and a flap to create the slotted flap, a design that exposed a slot between the flap and the wing when deflected. In combination with the thick new airfoil sections then being designed (see the next chapter), the Page slotted flap reliably produced even greater lift. Still, airplane designers mostly stayed away from flaps. In the United States, this neglect included a new type of "split flap" invented in 1920 by Orville Wright and associate J. M. H. Jacobs in a small laboratory provided by the Army at McCook Field in Dayton. In the 1920s, it was hard to find any aircraft with flaps except for those designed by Page and Lachmann.

8 Frank Davis Adams, *Aeronautical Dictionary* (Washington DC: NASA, 1959), p. 7.

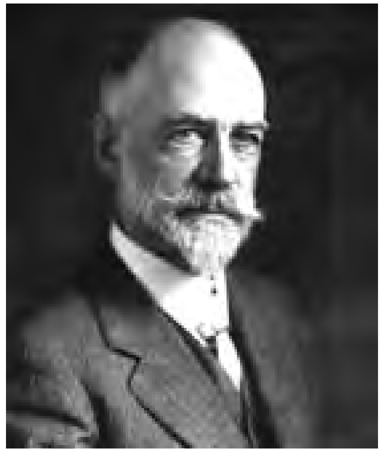
9 Anderson, *A History of Aerodynamics and Its Impact on Flying Machines*, p. 365.

It was not until the early 1930s that the idea of flaps started to entice many airplane designers—the result of the rapidly increasing speed of aircraft and their higher and higher wing loadings. The string of items comprising Document 3-12 traces the development of one of the most attractive and ultimately successful of these American high-lift designs, known as the "Fowler flap." Invented in 1924, this was an extensible trailing-edge flap that increased both the camber and wing area. Fowler flaps came to be used on several new aircraft from the 1930s on, notably planes built by Glenn L. Martin (in 1933, Martin hired Fowler to design flaps for him), as well as the Lockheed 14 twin-engine airliner of 1937. The Boeing B-29 bomber of World War II employed Fowler wing flaps (although they did not help much to improve the high stalling speed of 105 mph for the big plane); so, too, did later versions of the Lockheed P-38 "Lightning," for greater maneuverability. The wings of the Boeing B-17 employed very powerful Fowler flaps, designed specifically to keep the landing speed within acceptable limits.[10] The Germans even gave their Messerschmitt ME 262 jet fighter Fowler high-lift flaps, along the trailing edge of the wing (and in combination with full-span "slats"—that is, long narrow vanes or auxiliary airfoils—in the leading edges). After the war, Boeing's B-52 bomber augmented its lift via Fowler single-slotted flaps located at the trailing edge of the wing. So Fowler's invention of 1924 had a long shelf life.

Although the Fowler flap was only one of many different flaps designed and used after 1930, it is enlightening in Document 3-12 to follow how this one brand of flap disseminated through the American aeronautics community during the early 1930s. Similar studies could be made with the split flap, which was actually the type first tried on American aircraft. The Northrop Gamma used split flaps, as did the Douglas DC-1, in 1932, and the DC-3, in 1935. Split flaps ruled the roost early on—some suggest it happened in deference to Orville Wright, one of its inventors. About the time designers started to convert to Fowler flaps and other sorts of slotted flaps, the double-slotted flap appeared—on Italy's 1937 M-32 bomber and the Douglas A-26 bomber of 1941. From the DC-6 on, Douglas put double-slotted flaps on the wings of all its airliners. The triple-slotted flap made its first appearance with the Boeing 727 jet airliner in the early 1960s.

Perhaps an even more critical area of aerodynamic refinement necessary for the airplane design revolution of the interwar period involved advancing the state of the art in propeller performance. Again, the NACA played a fundamental role, conducting and supporting propeller research on a consistent basis from its inception as an organization in 1915. The committee's first *Annual Report* acknowledged the need "for more efficient air propellers, able to retain their efficiency over a variety of flight conditions." In the ensuing years, one of its strongest and most regularly funded programs focused on the improvement of propellers.

[10] Laurence K. Loftin. Jr., *Quest for Performance*, pp. 124, 132, and 139.



William F. Durand's long career in aeronautics included his pioneering, with Everett P. Lesley, a standard table of aerodynamic design coefficients for propellers from 1917 to 1926. (NASM Videodisc No. 2B-57570)

What NACA researchers most contributed was systematic correlation between airfoil theory, model propeller tests, and full-flight testing to find the best method of designing aerodynamically efficient propellers. Differences between the three sources of information about propellers showed that a significant gap still existed between empirical and theoretical knowledge. The first method, involving what was called "blade element theory," assumed that a propeller blade was a series of isolated airfoil sections that represented an ordinary wing as they traveled in a helical path. Blade element theory enabled engineers to design propellers that were 70 to 80 percent efficient. The multitalented Fred Weick, who authored a textbook on propellers for McGraw-Hill in 1931, said that blade-element theory enabled no more than a "cut-and-try process" when it came to designing new airfoils.

The second method involved testing model propellers in wind tunnels. The idea was to acquire reliable data from scale models that, through the law of similitude, could be applied to similar geometric bodies of full size. By the 1920s, this type of testing already had a distinguished history. From 1917 to 1926, professors William F. Durand and Everett P. Lesley of Stanford University conducted a program of "Experimental Research on Air Propellers" that provided a standard table of propeller design coefficients, which any designer could utilize.[11] Unfortunately, to make the data applicable for design, one had to carry out mathematical conversions that did not always work out right. It was thus vital to correlate experimental results with theory and with full-scale testing if the resulting propeller shape was to perform as expected.

The final method, full-scale testing, represented the final, ultimate check for propeller efficiency and design. NACA researchers in the 1920s found that the efficiencies of full-scale propellers were often six to ten percent greater than the efficiency of corresponding model propellers. A NACA report of 1925 attested that researchers "can never rely absolutely" upon model data until they verify it through full-scale flight tests. Testing the real thing in the actual flight environment, however, was expensive, time-consuming, potentially dangerous, and often inaccurate if researchers used poorly maintained testing equipment or inadequate techniques.

[11] See Walter G. Vincenti, Chap. 6, "Data for Design: The Air Propeller Tests of W. F. Durand and E. P. Lesley, 1916-1926," *What Engineers Know and How They Know It: Analytical Studies from Aeronautical History* (Baltimore and London: Johns Hopkins University Press, 1990), pp. 136-169.

Increased awareness of, and exposure to, the inadequacies of theory, wind tunnel experiment, and flight testing convinced the NACA that some form of "hybridization" of the three methods was necessary. In 1925, the NACA authorized the construction of the 20-foot PRT at Langley. In the following years, Fred Weick, Donald H. Wood, and other Langley researchers conducted various test programs that contributed to the fundamental development of the propeller—especially regarding the use of metal propellers. Their investigations found that by using blades that were thinner toward the tips, they could increase propeller efficiency. Propeller designers had already been aware that thin blade sections were ideal for high-speed applications because they did not suffer from what soon came to be known as "compressibility burble," which amounted to a sharp increase in drag at high speeds. But until propellers were manufactured in metal, this benefit could not be realized. Wooden propellers and their thick sections, necessary for structural integrity, suffered severe drag limitations at high speeds. To turn faster, propellers needed to be thinner, which required the strength of metal.

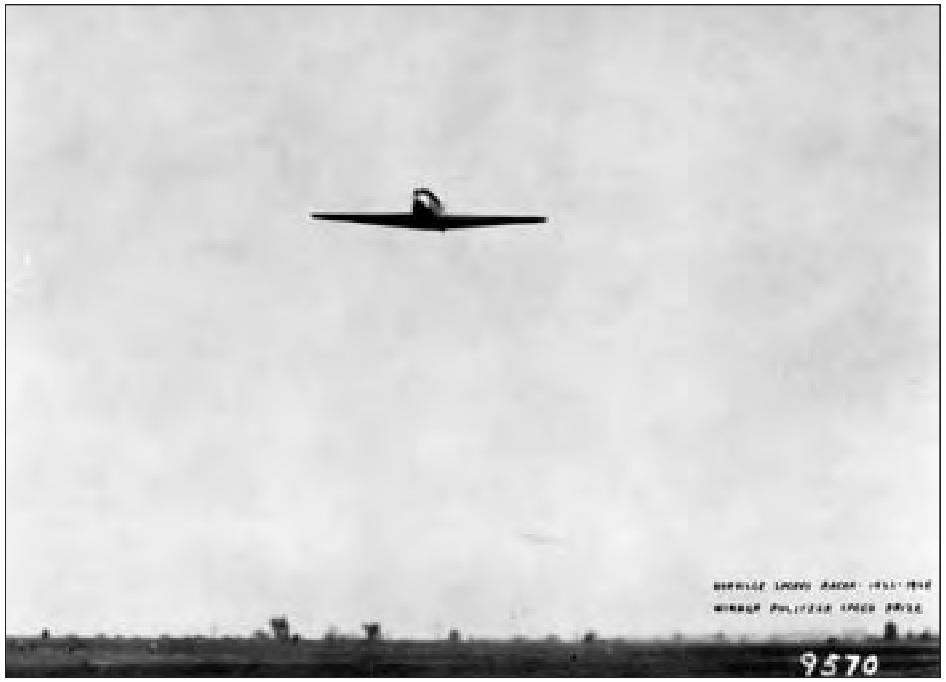
A significant breakthrough in propeller aerodynamics came in the form of the metal, multipiece, variable-pitch propeller—a technological development mostly associated with a research group, from the late 1910s into the late 1920s, under the leadership of Frank Caldwell in the Army's aircraft engineering division at McCook Field. During the first quarter of the century, aircraft propellers had all come with fixed pitch, meaning that the angle at which each propeller blade struck the air remained static. Caldwell's group at McCook recognized earlier that the ideal pitch of a propeller was not the same for different flight conditions. Low-pitch settings were more efficient for taking off and climbing, while high-pitch settings were better for cruising at altitude. With a fixed-pitch propeller, the setting of pitch involved an inherent compromise that somewhere sacrificed efficiency. The invention of the variable-pitch propeller bridged the gap between what could be considered the two major thrusts of the airplane design revolution of the interwar period: increased engine power and decreased drag.

Nothing illustrated the need for a variable-pitch propeller more than Charles A. Lindbergh's dramatic takeoff for Paris on 20 May 1927. Clearing telephone wires at the end of Long Island's Roosevelt Field by only 20 feet, Lindbergh's ground-adjustable propeller was set at a blade angle just below the optimum setting for cruise specifically to accommodate a better takeoff. Searching for every bit of economy for the 3,610-mile transatlantic flight, Lindbergh almost did not get off the ground. Document 3-13 features an April 1927 letter from Standard Steel Propeller Company to Ryan Airlines, a month before Lindbergh's takeoff, which dealt with the critical pitch setting for Lindbergh's historic *Spirit of St. Louis* Ryan-built airplane.

It was apparent to aeronautical engineers well before Lindbergh's takeoff that a method to adjust the pitch of the propellers automatically in flight was needed to improve aircraft performance. But experience proved it was not easy to design a mechanical system to accomplish this task reliably and as precisely as desired. Dr. H.

Lindbergh gambled that a ground-adjustable pitch propeller set just below the optimum setting for cruise would give him long-range efficiency while safely getting the Spirit of St. Louis off the ground at Roosevelt Field. The resultant dramatic takeoff for Paris on 20 May 1927 highlighted the need for variable-pitch propellers. (SI Negative No. A-4819A)

S. Hele-Shaw and T. E. Beacham patented a variable-pitch propeller mechanism in Great Britain in 1924, but it found little application, in part because the need for it was not as great as it was in America. The reason was topographical. Traversing North America by air meant surmounting obstacles like the Appalachian and Rocky Mountains. In 1929, McCook (later Wright) Field engineer Frank Caldwell, who had designed most of the detachable-blade metal propellers in use, resigned from the service to become chief engineer for Hamilton Standard Propeller Company. (He moved into this position just as United Aircraft bought out Standard Steel and joined the two companies, forming Hamilton Standard.) As head of Hamilton's production plant in Pittsburgh, Caldwell devoted himself to perfecting a controllable-pitch propeller, which he accomplished by 1933. Called "the gear shift of the air," Caldwell's device enabled pilots to adjust the pitch of their propeller blades automatically to match different flight conditions. This mechanism resulted in dramatic improvement in performance and won for Caldwell and Hamilton Standard the Collier Trophy in 1933 (see Document 3-14). Improvements in propellers con-



The Verville-Sperry R-3 racer, winner of the 1924 Pulitzer Race, was an important precursor for "modern" high-performance aircraft with its streamline design and retractable landing gear. (NASM Videodisc No. 1B-95330)

tinued, and by 1936, several new aircraft featured constant-speed propellers that could automatically adjust pitch. The NACA contributed key research data during this period relevant to the design and operation of controllable-pitch propellers.

Another essential shelf item developed during the design revolution of the interwar period, one meant to eliminate a significant source of drag, was retractable landing gear. As Frenchman Louis Breguet had declared in his 1922 Royal Aeronautical Society lecture on "Aerodynamical Efficiency and the Reduction of Transport Costs," the undercarriage should be made to disappear (refer to Document 3-4), but it took a while longer before much happened to eliminate it. Some aviation historians have written that the first American aircraft to incorporate retractable landing gear was the Boeing Monomail of 1930, but this is incorrect.[12] In the 1923 Pulitzer Trophy Race at Mitchell Field, Long Island, the Army entered a low-wing monoplane, an advanced version of the Verville-Sperry R-3, which incorporated retractable landing gear. Mechanical problems forced the R-3 to drop out of the race, which was won by a sleek Curtiss Racer flown by Lt. Alford J. Williams at the then-terrific speed of 245.3 mph— almost 40 mph faster than the speed the race was won at only a year

[12] See, for example, John D. Anderson, Jr., "Faster and Higher: The Quest for Speed and Power, " in *Milestones of Aviation* (New York: Hugh Lauter Levin, 1989), pp. 105-106. The Dayton-Wright RB-1 of the 1920s was also equipped with a landing gear that was semiretractable.

The highly successful Lockheed Orion utilized retractable landing gear and an aerodynamically refined shape to generate unprecedented overall performance. Nevertheless, its wooden construction and lack of a high-lift flap system and a variable-pitch propeller kept it from being a truly revolutionary airplane. (SI Negative No. 1996-0012; Videodisc No. 1B-14112)

before. The following year, 1924, the Verville-Sperry won the Pulitzer race, but at a speed of 215 mph, 30 mph slower than that achieved by Al Williams in his Curtiss Racer. Clearly, retractable landing gear represented a major aerodynamic advance, but it was not the only factor defining the performance of an aircraft.

In terms of integrating a number of critical new design features, including retractable landing gear, the Boeing Monomail stands out. This airplane was unprecedented in incorporating an all-metal structure, a smooth stressed skin, a cantilevered wing, and a Townend ring cowling. Unfortunately, it also possessed an ineffective propeller that cancelled out most of the benefits of the lower drag. Because its speed range was so extensive, the Monomail needed a variable-pitch propeller—a design then not yet available. Its fixed-pitch propeller could not match the plane's high speed and takeoff requirements.

In 1931, the Monomail was followed by a much more successful airplane employing retractable landing gear, the Lockheed Orion, a low-wing version of the pioneering Vega. (Also in 1931, the U.S. Navy procured its first aircraft with retractable gear, the Grumman XFF-1 two-seat fighter.) In comparison to the fixed-wing Lockheed Vega that came out just before, the advantages of the Orion's retractable gear were obvious: a maximum speed of 226 mph compared to 190 mph and a zero-lift drag coefficient of 0.0210 compared to 0.0278. These figures meant that the Orion could outperform any contemporary military aircraft.

The use of retractable landing gear was delayed for so long primarily due to a general belief in the industry that such gear was too heavy for practical use. Compared to what appeared to be subtle aerodynamic benefits, the difficulties of designing a reliable undercarriage that could be mechanically retracted into the body of an aircraft were glaringly obvious. More than any other single factor, it was the spectacular improvements realized with the Orion, which proved that the advantages far superseded what actually amounted to small increases in weight, that put retractable landing gear front and center of the shelf of new important new aircraft compo-

Fred E. Weick's W-1 home-built experimental airplane with steerable, tricycle gear precipitated a major change in how engineers designed modern aircraft. (NASA Image No. L-11151)

nents. As new streamline designs started flying in the 1930s, the same fixed landing gear that previously had accounted for only a small percentage of the overall drag of an airplane, usually in biplane form, now accounted for a significant portion of a monoplane's drag. As Laurence K. Loftin noted in *Quest for Performance*, "The configuration and design details of the Lockheed Orion represented an extremely high level of aerodynamic efficiency, a level that has seldom been exceeded in the years since 1931." Yet, as Loftin also pointed out, the Orion "lacked several features that later became an integral part of the propeller-driven aircraft in its final definitive form."[13] Like the Monomail, it also did not have a variable-pitch propeller, which meant its engine was not efficient over an entire range of flight conditions. This limitation became clearer and clearer to aeronautical engineers in the early 1930s, stimulating the development of the controllable-pitch propeller, without which the full aerodynamic potential of a high performance, low-drag aircraft simply could not be reached. The Orion also had only a simple trailing-edge flap, one designed to increase drag for approach and landing but not to increase maximum-lift coefficient. It really needed a high-lift system that increased maximum lift and reduced stalling speed. As suggested earlier, an advanced form of flaps was necessary—something that became standard equipment on high-performance aircraft by the late 1930s.

Aerodynamically speaking, "retracting" technology was the key to landing gear (Document 3-15). But before leaving the subject of this particular "shelf component," it is important to note the critical importance of developing *steerable*, tricycle

[13] Loftin, *Quest for Performance*, pp. 89-90.

landing gear. This was another development for which the NACA deserves considerable credit, and again, largely through the pioneering work of Fred Weick.

As with many other aeronautical engineers in the 1920s and 1930s, Weick dreamed of building a low-cost and simple-to-fly airplane that was so inherently safe and inexpensive to operate and maintain that air travel in it could, for certain purposes, compete with automobiles as a mode of private and family transports. Weick pursued his dream in earnest in the early 1930s, when he and a small group of colleagues from NACA Langley designed the "W-1" in a private venture. This home-built experimental airplane had several unique features, including an elevator with upward travel that was limited to the point where the airplane could not be forced into a spin. Another innovative feature of the W-1 was its coordinated control system. Weick's idea here was to reduce the number of controls from three to two by connecting the ailerons and rudder, thus eliminating the possibility of crossing these two controls and thereby simplifying the process of learning to fly.

In order to make it easier both to taxi and land the plane, Weick and his associates also equipped the W-1 with what was then an unconventional undercarriage: they moved the two main wheels a short distance behind the plane's center of gravity, took away the tail wheel, and added a nose wheel that could be steered. Such an arrangement, Weick thought, stood a very good chance of eliminating "ground looping," a serious problem even into the 1930s. All landplanes in production were still being equipped with tail skids—or, when paved runways came into use, with tail wheels. These gears had their main fixed-axis wheels located ahead of the center of gravity, were naturally unstable directionally, and thus tended to ground loop. Automobiles, bicycles, and motorcycles were stable in this regard because their main fixed-axis wheels were in back of the center of gravity. The conventional tail-wheel-type gears also nosed over easily, because the main wheels were just a little ahead of the center of gravity. Some of the early pusher airplanes, popular around 1910, had a single wheel well ahead of the center of gravity and two wheels in back of the center of gravity, which took care of the nosing-over difficulty reasonably well. But all wheels were on fixed axes and could be steered very well on the ground. All of these realizations led Weick to conclude that the undercarriage of the W-1 needed to be quite different. Weick dubbed his new steerable undercarriage a "tricycle gear," and it was a name that stuck. (In March 1932, Weick received a U.S. patent for his W-1 design with steerable tricycle gear; he received another patent for a two-control airplane design incorporating a tricycle landing gear in March 1938; see Document 3-16). The gear performed extremely well for the W-1; the results were published in NACA reports. When first learning of Weick's gear, some people in the industry said that it was just a reversion to the three-wheeled gears that had been used on pusher airplanes before World War I, but this was not the case. Airplanes during that era had fixed nose wheels, and operations took place in open grass fields, with a great deal of manpower available for handling the airplane. This meant that people in those days had not cared whether gear was stable in taxiing or not. There is no

The one-and-only Douglas DC-4E represented many of the aerodynamic innovations that evolved during the 1920s and 1930s. It was the first large transport to use tricycle gear and its design process helped originate NACA research in the establishment of stability and control specifications prepared to insure good flying qualities. (SI Negative No. A0193510)

question that Weick's tricycle gear was a new line of thought and a worthwhile improvement. At the NACA conference in 1935, Orville Wright had made exactly these points to Weick, which encouraged him greatly.

It was also in 1935 that engineers with the Douglas company showed the first interest in applying the tricycle gear to large airplanes. Dr. Arthur Raymond, chief engineer at Douglas, sent F. R. Cohlbaum and W. Bailey Oswald to Langley to find out about the tricycle gear on the W-1 and to ask questions about what it might do for larger airplanes. After their visit, Douglas, in cooperation with the Army, tried the tricycle gear on one of its Dolphin airplanes, which originally possessed a tail-wheel-type gear. The Dolphin was an amphibian flying boat, and the Douglas engineers simply moved the wheels of the main gear back a bit and put a castering nose wheel under the front of the hull. These tests confirmed the gear's advantages. In October 1935 TransContinental and Western Air (as TWA was known before 1950) also asked for information about the possible use of tricycle gear on a transport airplane. The NACA responded by pointing out the various advantages of the gear, plus the fact that with a twin-engine transport, one could extend the forward part of the fuselage sufficiently to support the nose gear well forward and also provide a satisfactorily long wheelbase.

Soon thereafter, three or four airlines met with Douglas in the hope of getting a transport that was larger than even the new DC-3 that had just appeared. Edward P. Warner, a former NACA chief physicist, editor of *Aviation* magazine, and professor

of aeronautics at MIT, served as a consultant to this group, and he visited Langley Field several times to review the possibilities of applying the tricycle gear to such a large transport.[14] The result of this activity was the one-and-only Douglas DC-4E, the first large transport to use the tricycle gear. Even then, the Douglas engineers made provisions for returning to a tail-wheel-type gear in case they were not entirely happy with the tricycle gear. During World War II, a modified version of the DC-4 was used in military activities as the C-54, complete with tricycle landing gear. The tricycle gear was also used on other military aircraft, including the Douglas A-20, Lockheed P-38, Bell P-39 and P-63, Consolidated B-24, North American B-25,

NACA engineers found that with the later addition of wing fillets and a NACA cowling, the diminutive McDonnell Doodlebug experienced greatly reduced buffeting and aerodynamic interference. (SI Negative No. A43437-C)

Martin B-26, and Boeing B-29. It eventually became the standard gear for most all military, airline, and general aviation airplanes. Even the NASA Space Shuttle, arguably the world's most sophisticated aircraft, benefits from the same type of tricycle gear with steerable and castered nose wheel that Weick designed back in the 1930s.[15] Again, this development did not relate directly to improved aerodynamics, but it shows how various new design features synergized in the airplane design revolution of the interwar years.

Another part of this synergy came in the form of wing fillets, which were concave fairings used to smooth the interior angle of the wing-fuselage juncture. With-

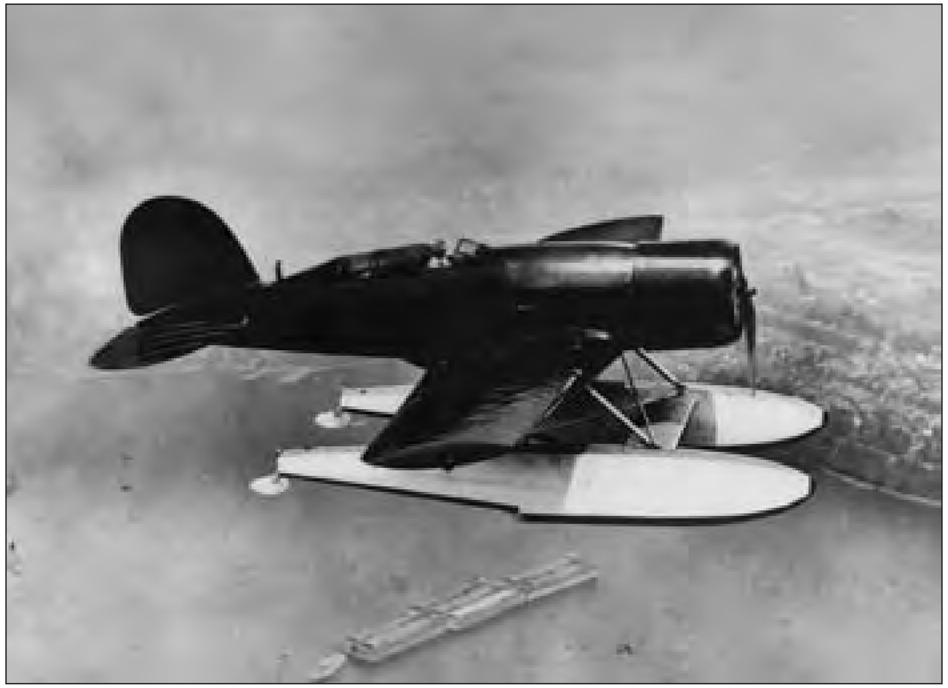
[14] For an account of Edward P. Warner's outstanding career in American aviation, see Roger E. Bilstein, "Edward Pearson Warner and the New Air Age," in *Aviation's Golden Age: Portraits from the 1920s and 1930s*, ed. William M. Leary (Iowa City IA: University of Iowa Press, 1989), pp. 113-126. In various ways, technically and politically, Warner made notable contributions to the "reinvention" of the airplane.

[15] For Weick's account of his development of the tricycle gear for the W-1 airplane, see Weick and Hansen, *From the Ground Up*, pp. 131-140.

out fillets, the intersection of wing and fuselage experienced airflow interference. A smooth "fairing" of the area where the wing joined the fuselage (i.e., fitting and shaping so as to make it smooth and streamlined) greatly improved the handling characteristics and efficiency of low-wing monoplanes. Developed at the Guggenheim Aeronautical Laboratory of the California Institute of Technology under the direction of Theodore von Kármán, the fillet soon became a standard feature of monoplane designs (see Document 3-17). Both the Boeing Monomail of 1930 and the Northrop Alpha of 1931 benefited from wing fillets, as would the culminating airplane of the design revolution, the Douglas DC-3. As for the Alpha, the advantages of wing fillets could not overcome its basic aerodynamic obstacles, notably a fixed landing gear and an open cockpit for the pilot. It is curious that so many new aircraft during this era combined anachronistic features with important new innovations. But such combinations of old and new are, in fact, typical of transitory times before all the features of a new technology come together and form a new design paradigm.

Equally typical of the transitory period is the appearance of a "mainstay design," one that maximized the existing state of the art but offered a sophisticated overall product that nonetheless augured important elements about the future. Such an aircraft was the ubiquitous Ford Trimotor, a 13- to 15-passenger transport that did the lion's share of work for America's budding airlines in the late 1920s and early 1930s. Although the Trimotor remained aerodynamically primitive in some respects, such as its fixed landing gear and uncowled engine nacelles, aerodynamic refinement, as Document 3-18 shows, had been one of the goals of Ford Motor Company's Aircraft Engineering Department when it started designing the Trimotor from 1925 to 1926. Henry Ford had built a mammoth reputation as a manufacturing and production genius in the automobile industry, and he wanted to do something similar

The Ford Trimotor 5-AT-B was a "mainstay design" for America's growing commercial airline industry, but its improperly placed engine nacelles below the wing, angular windscreen, exposed engines, ground-adjustable pitch propellers, fixed landing gear, and corrugated aluminum construction did not reflect the rapid advances in aerodynamic streamline design. (SI Negative No. 47937-J)



Aircraft such as the Lockheed Sirius "Tingmissartoq" flown by Charles and Anne Lindbergh represented the transitional nature of aerodynamic design with its advanced streamline shape combined with an open cockpit, ground-adjustable pitch propeller, and large floats. (SI Negative No. A-5172)

The Martin B-10 represented the revolution in combat airplane design, and in key respects, presaged the total package of a "reinvented" airplane. (SI Negative No. 78-1328)

The Boeing 247D integrated all the most desirable features emerging from the airplane design revolution, making it the first “modern airliner.” (SI Negative No. 75-12118)

with airplanes. So resolved, he put the resources of his entire company behind the building of a Ford transport. He even bought out the Stout Engineering Company, which had recent experience with producing an innovative, cantilever-wing transport design. Stout’s transport, which was also all metal, would soon become famous for running on the Stout-Ford airlines between Chicago, Detroit, and Cleveland. Ford would soon have William B. Stout himself at work, in his small engineering laboratory in Dearborn, Michigan, on the design of the Ford Trimotor.

Two other airplanes that represented a link between old and new were the previously mentioned Lockheed Vega, which appeared within one year of the Ford Trimotor, and the Lockheed Sirius of 1930. All three planes reflect the flux and indeterminacy of aerodynamic development of aircraft during the late 1920s, before all of the components of the airplane design revolution came together into a coherent whole. Synergy of all the necessary shelf items was not yet possible, because not all the shelf items had yet appeared, so to speak, “on the shelf.” But all three designs indicate that their designers were, without question, paying attention to reducing aerodynamic drag through streamlining. Harold Hicks replaced George Pruden as the head of Ford’s Aircraft Engineering Department (Henry Ford fired Pruden for a breach of company etiquette; he appeared in a newspaper photo as a representative of the Ford Company) and oversaw the development of the trimotors. Document 3-19 concerns the design of the Sirius airplane. Document 3-20 details yet another one of these transition types—one that stimulated a revolution in combat airplane

design and in key respects presaged the total package of a “reinvented” airplane—the Martin B-10 bomber.

But the aircraft that integrated all the most desirable features emerging from the airplane design revolution even more fully was the Boeing 247. Aviation historians consider this Boeing design, first flown in February 1933, to be the first “modern airliner.” The definitive version of the airplane, the model 247D, had the following features: all-metal, stressed-skin construction; cantilever wings; retractable landing gear; an efficiently cowled, lightweight radial engine; controllable-pitch propellers; and single-speed geared supercharger. Model 247D was also the first transport aircraft to employ rubber de-icing boots and to have a significant amount of instrumentation for blind flying. Aerodynamically, the design was quite far advanced, with split-type landing flaps and a fairly high wing loading. Its 0.0212 $C_{D,O}$ was rather low, and its 13.5 maximum lift-drag ratio was quite high; both numbers compared favorably with the performance of virtually all previous aircraft. Operating for the airlines, the Boeing 247 carried up to ten passengers comfortably, at a cruising speed of 185 mph at 7,500 feet. In service mainly with United Airlines(along with Boeing, its maker, part of the United Aircraft and Transport Corporation), the airplane established a new standard for commercial air travel. In doing so, it provoked other aircraft manufacturers, as well as the airline operators, to seek the design of their own more refined aircraft.

Jack Frye’s August 1932 letter to American aircraft manufacturers requesting a new all-metal airliner for TWA led to the creation of the Douglas DC-series aircraft. (SI Negative No. 75-5208)

One aviation executive who desperately sought a new airplane was Jack Frye, vice president for operations of Transcontinental and Western Airlines (TWA), a company that was precluded from buying the new Boeing 247 because of the corporate connection between Boeing and United Airlines. On 2 August 1932, Frye sent letters to Curtiss-Wright, Ford, Martin, Consolidated, and Douglas seeking proposals for a new all-metal trimotor design, one that he hoped could profitably replace TWA’s aging fleet of Fokker trimotors. That letter, reproduced in its entirety as Document 3-21, reflected Frye’s belief that trimotored aircraft were still the norm for airline operation. Interestingly, the TWA leader stipulated no other design details and only suggested a few of the desired performance characteristics. He left it up to the manufacturers to establish the formal design parameters.

Intrigued by the challenge of producing a design that might even turn out better than the Boeing 247, Donald W. Douglas and his design team in Santa Monica, California, began work on a 12-passenger twin-engine airliner incorporating all the latest advances in aircraft components and design. This included certain features that the Boeing 247 lacked, notably variable-pitch propellers and split flaps for higher wing loading. A Douglas delegation headed by a young Caltech graduate, Arthur E. Raymond, traveled to TWA's offices in New York City and won the contract for the new airplane. After ten months of development work, the first of the new Douglas DC series transports, the DC-1, took to the air in July 1933. The prototype all-metal DC-1 incorporated the most advanced innovations of the time. In addition to the newest shelf items, the Douglas design team utilized the latest in aeronautical design methods to fully integrate them into the overall design. This utilization included an unprecedented level of "research for design" conducted within a wind tunnel (see Document 3-22).

It was not just technological incentive that led to the DC-1's innovative design. Economic incentives further hastened technical innovation when Douglas engineers elongated the fuselage of the DC-1 prototype (see Document 3-23) to allow for two more passengers, bringing the capacity up to 14. The resultant DC-2 first flew on 11 May 1934 and became the first commercially available Douglas transport. Flying TWA's 18-hour "Sky Chief" transcontinental route from Newark to Los Angeles (with en route stops at Chicago, Kansas City, and Albuquerque) at speeds up to 175 mph, the DC-2 carried the same amount of passengers as the old trimotors at less cost and at faster speeds. The success of the DC-2 was not just national but international. It was also stunning. In 1934, a DC-2 flown by the Dutch airline KLM finished second in the 11,300-mile London-to-Melbourne MacRobertson Trophy Race—and amazingly did so with a full load of mail and three passengers. Quickly, KLM ordered a dozen DC-2s, followed by numerous orders from other European carriers. For the first time, an American airliner began to outcompete the Fokker transports, which had dominated European airline service for several years. Although a British plane had won the MacRobertson Race, the winner had been a special customized racer, the De Havilland D.H.88 Comet. British aviation leaders recognized that the real significance of the race lay in the American commercial airliner coming in second. The editor of London's *Saturday Review* declared soon after the race: "Britain has won the greatest air race in history, but she has yet to start on an even greater air race: a race in commercial and military supremacy." Sadly, no British airplane, not even the best machine in regular use with the Royal Air Force "at the present time is fast enough to have finished the race within a thousand miles of the American machines." (A Boeing 247 flown by Roscoe Turner and Clyde Pangborn finished third in the race, two and a half hours behind the DC-2.) "It is almost incredible, but it is true," the British newspaperman lamented.[16]

Dutch KLM's use of the DC-2 in the 1934 London-to-Melbourne MacRobertson Trophy Race announced to the world the growing ascendancy of American aeronautical technology. (SI Negative No. 523183)

In the design of the Douglas DC-2, the two main approaches that American engineering had been following for the purpose of streamlining aircraft designs (i.e., commercial efficiency and greater speed) converged very successfully, and American aviation benefited greatly from it, on a worldwide scale. The DC-2 clearly outperformed all other airliners. For the first time, American industry was producing a transport so superior to all others available that even European carriers committed to their own national aviation industries had no choice but to start flying the American planes. The significance of the DC-2 does not end there. As historian Richard K. Smith noted, the appearance of the DC-2 also created, for the first time, "a distinct division of labor between the design of military and civil aircraft." Before 1933, bombers were typically just converted airliners with bomb shackles, armor, and guns added. But the new streamlined forms dictated more specialized designs. Streamlining called for bombs to be carried internally, and because bombs took less space than a load of passengers, from this point on "cross-section designs of bombers and airline equipment would move in opposite directions."[17] In this sense, the reinvention of the airplane brought on greater specialization. Streamlining was a general objective, but the special forms of streamlining required by aircraft with significantly different missions led to one of the clearest bifurcations in aircraft design history. This was a critical element of change that is often missed when discussing the airplane design revolution of the interwar years.

As mentioned earlier, the DC-2 design incorporated the latest refinements enjoyed by the Boeing 247, plus a higher wing loading and split-type flaps. The Douglas Company found itself in the enviable position of trying to fill an over-

[16] Quoted in Terry Gwynn-Jones, "Farther: The Quest for Distance," in *Milestones of Aviation*, p. 67.

[17] Richard K. Smith, "Better: The Quest for Excellence," in *Milestones of Aviation*, p. 253.

The highly successful DC series of aircraft, such as the Douglas Skysleeper Transport designed for American airlines, garnered Douglas Aircraft the 1935 Collier Trophy. (NASM File No. AD-761192-75)

whelming backlog of orders for the plane. Designing an even bigger and more effective transport became a growing possibility when Cyrus Rowlett "C.R." Smith of American Airlines, the third U.S. transcontinental service (with TWA and United) wanted an aircraft to replace the fleet of aging Curtiss Condor biplanes that American had been using on its nighttime Pullman-style sleeper service. American's chief engineer, William Littlewood, determined that by widening the fuselage of the DC-2 by 26 inches and adding 10 feet to its wingspan, it could accommodate 14 sleeping berths. The revised airplane, known as the Douglas Skysleeper Transport, made its maiden flight on 17 December 1935, in celebration of the 32nd anniversary of the Wrights' flight at Kitty Hawk. Fitted with Wright R-1820 engines and outfitted for 21 daytime passengers (without the sleepers), the airplane flew as the DC-3. On 1 July 1936, President Franklin D. Roosevelt, on behalf of the National Aeronautic Association, presented the Robert J. Collier Trophy to the Douglas Company for the greatest achievement in American aviation in 1935, the refinement of the DC series aircraft. In his remarks, President Roosevelt said, "This airplane by reason of its high speed, economy, and quiet passenger comfort has been generally adopted by transport lines throughout the United States. Its merit has been further recognized by its adoption abroad, and its influence on foreign design is already apparent."[18]

Soon, the DC-3 became the most popular and reliable propeller-driven airliner in aviation history. Its appearance marked the culmination of the design revolution in American aircraft and, as C. R. Smith, president of American Airlines, noted in

[18] Quoted in Douglas J. Ingells, *The Plane that Changed the World: A Biography of the DC-3* (Fallbrook, CA: Aero Publishers, Inc.), p. 166.

our Destination Document, it was the first airplane that enabled the airlines to make money by solely carrying passengers. Technologically, it represented a completed synergy of virtually all the major innovations that had taken place in aircraft design since 1920. Aerodynamically, the airplane was extremely far advanced, producing a very low $C_{D,O}$ of 0.0249; this was 17 percent less efficient than the Boeing 247, which had an extremely low value of 0.0212. But this difference was because the DC-3 had a much larger fuselage—to accommodate three-abreast seating—that resulted in a larger ratio of wetted area (for wings and wing-like surfaces, the wetted area is related to the exposed planform area) to wing area. The DC-3 enjoyed a very high L/D, a ratio of 14.7. This L/D was higher than anything that came before—and better than virtually all that came after, certainly with propellers. (Only the Boeing B-29 bomber of 1944 and the Lockheed L.1049G Super Constellation airliner of the early 1950s enjoyed higher L/Ds, of 16.8 and 16.0, respectively.) One very noteworthy aerodynamic feature of the DC-3 was its sweptback wing. This was not engineered for the same reasons that wings of transonic and supersonic aircraft would later be given sweepback; rather, it had to do with positioning the airplane's aerodynamic center properly in respect to its center of gravity. To some extent, this had been done also with the DC-2. But as the design of the DC-3 evolved, it became evident to Douglas's engineers that the center of gravity was located farther toward the rear (or aft) than they expected. They mounted the outer panels of the DC-3 wing with greater sweepback, moving the aerodynamic center to the correct position. This change helped the airplane achieve a cruising speed of 185 mph (at 10,000 feet) when carrying a full load of 21 passengers. In other words, the DC-3 benefited from an even more streamlined shape than its predecessors, while increasing capacity by 50 percent. Thanks not only to streamlining but also to the entire process of reinventing the airplane, for the first time, the American airline industry had a passenger aircraft that could make a profit without a government airmail subsidy.

C.R. Smith's need for new aircraft for American Airlines resulted in the DC-3, which Smith declared as the first airliner to enable the airlines to make money by solely carrying passengers. (SI Negative No. 8094; Videodisc No. 2B-75566)

Without a doubt, 1934, 1935, and 1936 were banner years for both American and global aviation. From the point of view of aviation only, it has been argued that it was the aircraft developments associated with this period that, as much as anything else technological contributed directly by American industry, helped begin to lift the U.S. economy out of the Great Depression; aircraft developments certainly played a part in the general recovery that ultimately depended on involvement in the Second World War. By the mid-1930s, the streamlined shape defined largely by the American aeronautics community became standard in all new aircraft designs, making advance performance through design refinement a reality and the U.S. aircraft industry the world leader (see Document 3-24). As seen in this chapter and its documents, a myriad of new flight technologies contributed to this airplane design revolution. Although major developments in propulsion, structures and materials, flight instrumentation, and stability and control technology played fundamental roles in the reinvention of the airplane, the role of aerodynamic refinement was primary in many ways. The introduction of superior propellers with controllable pitch, cantilever wings, wing fillets, flaps, efficient engine cowlings, proper nacelle placement, and retractable landing gear all pushed the airplane to extraordinary new levels of sophistication and performance. So, too, did the definition of advanced airfoils, perhaps the most important single component contributing to the airplane design revolution. This was such a crucial topic for aerodynamicists, that it will be explored separately as the entire focus of the next chapter.

It took more than just the integration of various airplane components to cement the airplane design revolution within professional aeronautical practice and prepare aeronautical engineers for the future. Even after the reinvented airplane had been made essentially complete in the form of the DC-3, significant problems still affected total airplane performance. As Document 3-25 shows, one of these problems was aerodynamic stall, the potentially dangerous, even fatal, flight condition when an aircraft started to fly at an angle of attack greater than the angle of maximum lift, resulting in a loss of lift and an increase in drag. Stall had always been a concern for engineers and pilots, but in the early years of aviation, stalling speeds had remained low, in the range of 40 to 50 mph. This was due to the relatively poor flying characteristics of aircraft of that period, the fact they did not benefit from high-lift devices, and related directly to the short, unpaved fields then serving as airports. A pilot simply could not come in for a landing at a very high speed and expect to land safely.

The design revolution of the interwar years changed the descent and landing situation dramatically. The flying characteristics of airplanes improved greatly. High-lift devices became sophisticated and commonplace; runways became paved and quite long. Aircraft could manage landings at much higher speeds; but in doing so, they ran new risks of stalling not experienced earlier. The stalling speed of the DC-3 rose to over 60 mph; the Boeing B-17 to over 80; the North American P-51 to about 100; the Martin B-26F's stalling speed rose to over 120 mph. By the start of World War II, the stalling speeds of virtually all advanced aircraft had risen into the range of 80 to 100 mph; without the help of high-lift devices, stalling speeds would, in fact, have become much higher, further increasing risk. Increased speeds generally equated to progress in aviation, but landing at higher and higher speeds, even with the aid of high-lift devices, raised new issues that aeronautical engineers had to resolve.

The NACA had conducted many studies related to stall before the mid-1930s, but the operating problems of a large new airliner as revolutionary as the DC-3 sparked a new wave of concern. In September 1937, United Airlines loaned NACA Langley one of its DC-3 Mainliners in order to conduct stall tests (see Document 3-25). Out of this specific program came a more general investigation into stalling. Through the 1940s, the NACA worked on various stall-warning indicators and continued to look generally into stall phenomena. In the early 1940s, as readers will learn in a later document in this chapter, the NACA also played the key role in spelling out "stalling characteristics" as part of a larger program of establishing uniform "flying qualities" requirements for American aircraft.

In flying United Airlines' DC-3 Mainliner in late 1937, NACA Langley investigated more than the stalling problem; it also looked into the equally if not more dangerous problem of aircraft icing. As historian Glenn Bugos has explained, icing was a critical systems-wide problem for aircraft, then and now:

> Ice caused aircraft to crash by adding weight and preventing the pilot from climbing above the icing clouds, so that the aircraft gradually lost altitude and slammed into the ground . . . [I]ce accreted along the wing and tail leading edges disturbing lift and adding drag. Ice clogged the interstices of rudders and ailerons, preventing control and inducing buffeting. It changed the aerodynamic profile of the propeller, causing it to vibrate and exert less thrust per horsepower. It coated windshields, so the pilot flew blind. Ice made antenna wires oscillate and snap, and generated static that rendered useless most radio communication and navigation. It distorted pitot shapes, so that pilots got erroneous airspeed readings. And it clogged carburetors, suffocating the engine.

Within minutes, pilots could lose all of their critical systems, not just the engine, wings, control surfaces, indicators, and radio, but also their own sight as well.[19]

Icing was a problem to which the NACA committed considerable time and energy. A rash of aircraft accidents traced to icing problems attracted public attention in the late 1930s; a number of commercial operators, not just United Airlines, clamored for useful information on the subject. The NACA launched a comprehensive study of the icing problem that would last for many years. Researchers looked for mechanical, chemical, and thermal ways of breaking up ice before it formed dangerously on an airplane's vital surfaces. Innovative new tests looking for answers to icing began in flight and in wind tunnels. Various de-icing systems were tried, including a thermal (heat-directed) system for which the NACA and its

[19] Glenn Bugos, "Lewis Rodert, Epistemological Liaison and Thermal De-Icing at Ames," in *From Engineering Science to Big Science: The NACA and NASA Collier Trophy Research Project Winners*, ed. Pamela E. Mack (Washington, DC: NASA SP-4219, 1998), p. 32).

engineer Lewis Rodert won the Collier Trophy for 1946. Document 3-26 surveys the NACA's pioneering work on aircraft icing from the late 1930s to 1948. It was a research specialization that NASA would pursue with vigor.

Another generic problem that the NACA investigated (to the advantage of the airplane design revolution of the 1930s) centered on gust loads and gust alleviation. One event sparking early interest in this problem was the destruction of the ZR-1 Navy airship *Shenandoah*, which crashed in a thunderstorm near Ava, Ohio, on 3 September 1925, killing 14 of 43 on board. Engineers in the PRT section at Langley reacted to this tragedy by mounting some rudimentary equipment on the top of their tunnel building trying to test the magnitude of gusts found in the free air. Later, NACA researchers collected pressure distribution measurements of air loads in wind tunnels and in flight, but there was still no strong impetus to apply them to the design of specific aircraft. Not until the early 1930s did a coordinated NACA program of gust research begin that was interested in applying such data to aircraft design. A few special NACA committees then formed to oversee loads research. Eventually, Langley created a separate structures research division and, in 1938, built a structures research laboratory.

As this commitment materialized, it grew clearer and clearer to everyone that the stresses brought on by gusts had to be a more essential factor in the design of aircraft, especially civil airliners. Understanding gust loads lay somewhere about midway between aerodynamics and meteorology, and without question threatened the safe and comfortable operation of aircraft, with special concern for passenger transports. As an aerodynamic problem, the NACA first conducted gust loads research "on the premise that a gust acted almost like an instantaneous change of attack of the airplane encountering it."[20] Engineers who came to specialize in the problem, notably Langley's Richard V. Rhode (a 1925 University of Wisconsin graduate who would win the Wright Brothers Medal of 1935 for his gust loads research) later refined this concept by taking into consideration the distance over which a gust extended and the time it took for the wing flow to adjust to the new angle of attack. Rhode's concept of a "sharp edge" gust became "the backbone for all gust research."[21] Document 3-27 provides a string of documents from the 1930s concerning the emerging importance of this major new field of research.

Although not covered by any documents in this chapter, the NACA's study of gust loads for the purpose of articulating design requirements carried over into several other critical areas, some of which concerned military more than civilian aircraft. One of these areas involved a systematic exploration, both theoretical and experimental, into the dangers of aerodynamic flutter. Another concerned the loads

[20] Hartley A. Soulé, "Synopsis of the History of the Langley Research Center, 1915-1939," HQA HHN-40, 1966, p. 39.

[21] Soulé, "Synopsis," p. 39.

and stresses on combat aircraft while dive-bombing. In the 1930s, NACA researchers worked out charts showing the relationships between dive angle, speed, and the angle required for recovery. Using these charts, the Navy established design requirements for its dive-bombers around 1935. The NACA later conducted other flight dive tests meant to ensure that elevator control force would not hinder the acceleration a pilot needed to recover his aircraft from a dive. Through this research program, the U.S. aeronautics community came to a much better understanding of the distribution of loads between the various parts of the airplane—wing, fuselage, and tail surfaces—and how these individual parts were "loaded" by the wide range of maneuvers, some of them extreme undertakings by a high-performance aircraft.

As already suggested in reference to defining "stalling characteristics," another major NACA contribution of this period was its specifying of uniform "flying quality" requirements for American aircraft (see Document 3-28). As already emphasized, the airplane design revolution of the interwar period involved more than the development of aircraft component technology and its various integrations and synergies. It also involved broader intellectual developments—notably the translation of ill-defined problems, issues understood and handled previously in a largely qualitative and subjective manner, into much more objective problems defined and resolved on a stronger quantitative basis. In this respect, the reinvention of the airplane hinged on an epistemological reorientation within American engineering, one that changed not only what aeronautical engineers knew but how they knew it.

In retirement, as part of a productive second career as a historian of technology, former NACA/NASA research engineer Walter G. Vincenti delved deeply into this epistemological transition. Several chapters in his prizewinning 1990 book, *What Engineers Know and How They Know It: Analytical Studies from Aeronautical History,* looked into the significance of this change, how and why it happened, as well as its impact on the wider world of aviation. In particular, his chapter, "Establishment of Design Requirements: Flying Quality Specifications for American Aircraft, 1915-1943," analyzed the complex intellectual and social process by which U.S. aeronautical engineers in the 1920s and 1930s came to establish more objective specifications for aircraft design.

Aeronautical engineers as late as the start of World War II really did not know how to phrase, let alone categorize, their observations about the performance of an airplane. Everyone associated with aircraft development and operation wanted good flying and handling characteristics. They wanted machines that precisely, rapidly, and predictably obeyed a pilot's inputs—without unwanted dips or drops and without the pilot having to manhandle controls. Pilots, however, had a hard time putting what they wanted into words. For example, a pilot reported his machine as "tail heavy" or commented on the "lightness" of his controls, but nowhere were any of these terms defined concretely. Pilots described what "flying qualities" were most acceptable to them impressionistically, in their own words, to their own apparent satisfaction, but without a clear, quantitative basis. Whatever was actually tangible in the relationship between the "feel" of an aircraft and its actual nuts-and-bolts performance remained vague and imprecise.

Vincenti's chapter revealed the 25-year process it took, from 1918 to 1943, before "flying qualities" became well enough defined to be specified. As with so many elements of the airplane design revolution, what unfolds in Vincenti's story is the history of an idea: "The notion that specifications could usefully be written for something as subjectively perceived as flying qualities had itself to be realized intellectually and verified in the real world. It was not at all an obvious or obviously useful idea at the outset."[22] By the early 1940s, however, specifications did become defined, as did the whole concept of "flying qualities." NACA test pilot Melvin N. Gough defined that concept early in World War II as "the stability and control characteristics that have an important bearing on the safety of flight and on the pilot's impressions of the ease and precision with which the aircraft may be flown and maneuvered."[23]

The string of documents in Document 3-28 all relate to the research program undertaken by the NACA from the mid-1930s onward to establish precise "flying qualities." In essence what the NACA did, primarily through flight research of full-scale aircraft, was measure how aircraft responded to specific control inputs and then correlate them with pilots' opinions of the aircraft's behavior. The NACA investigators then related what they found to the engineering parameters that had been employed in designing the aircraft. The result was a long menu of "requirements" essentially falling into three main categories. The first group, "requirements for longitudinal stability and control," subdivided into:

(1) elevator control and takeoff;
(2) elevator control in steady flight;
(3) longitudinal trimming device;
(4) elevator control in accelerated flight;
(5) uncontrolled longitudinal motion;
(6) limits of trim due to power and flaps; and
(7) elevator control and spinning.

The second category, "requirements for lateral stability and control," subdivided into:

(1) aileron control characteristics;
(2) yaw due to ailerons;
(3) rudder and aileron trim devices;
(4) limits of rolling moment due to sideslip;

[22] Walter G. Vincenti, "Establishment of Design Requirements: Flying Quality Specifications for American Aircraft, 1918-1943," in *What Engineers Know and How They Know It: Analytical Studies from Aeronautical History* (Baltimore and London: The Johns Hopkins University Press, 1990), p. 52.

[23] Quoted in Loftin, *Quest for Performance*, p. 117.

(5) rudder control characteristics;
(6) yawing moment due to sideslip;
(7) crosswind force characteristics;
(8) pitching moments due to sideslip; and
(9) uncontrolled lateral and directional motion.

The third category, "stalling characteristics," has been discussed earlier in this chapter in relation to the Douglas DC-3. In his book chapter on the "Establishment of Design Requirements," Vincenti looked at the NACA's articulation of all the sub-divisions, with a specific focus on elevator control in steady flight, uncontrolled longitudinal motion, and elevator control in accelerated flight, all part of the "requirements for longitudinal stability and control."[24]

As Document 3-28 indicates, the immediate catalyst for the NACA's flying-quality research came in 1935 and resulted from a set of specifications prepared for the Douglas DC-4E. Edward P. Warner, the accomplished aeronautical researcher and former editor of *Aviation* who was then working as a consultant for the Douglas Aircraft Company, asked the NACA to help him specify the stability and control characteristics to be built into the DC-4 transport. In December 1935, the NACA Aerodynamics Committee, which was chaired by Warner, approved what became Research Authorization (RA) 509, "Preliminary Study of Control Requirements for Large Transport Airplanes." The purpose of this investigation was to determine "what specific qualities pilots desired, so that they could be numerically specified in future design competitions." A team of NACA Langley researchers under Hartley A. Soulé started this work in 1936 using a Stinson cabin monoplane. Langley instrumented the airplane so that its response characteristics, following known control inputs from the test pilot, could be measured, related to design parameters, and correlated with the pilot's qualitative evaluation of the ease and precision with which he maneuvered the plane. Soulé's team continued this effort using all 12 different airplanes that could be obtained for this purpose until 1941, when it was ready to specify its first set of uniform requirements for the flying qualities of airplanes.

The U.S. Army and Navy enthusiastically welcomed the NACA's product and started to revise the preliminary specifications to meet their immediate requirements. In 1942, both services asked the NACA to continue validating and upgrading handling requirements specifically for military aircraft. This request made many more airplanes available to the NACA as well as a broader view than would have been possible with just a few select commercial aircraft. By the end of World War II, the NACA had measured the stability and flying qualities of 60 different aircraft, and military and civil aircraft handling requirements had been standardized. This effort foreshadowed the extensive work that would be undertaken in the field over the next five decades, leading to the present uniform rating system. Unquestionably,

[24] Vincenti, "Establishment of Design Requirements," pp. 93-98.

the appearance of more objectively defined flying qualities proved to be a culminating development in the airplane design revolution, one that led to a more concrete understanding of aircraft operation, a sounder approach to aircraft design, and a much improved sense of the pilot-machine interface.

Not all revolutionary episodes turn out so neat and tidy, but one can say that the airplane design revolution ended with a literal "dusting off" of what had become of the modern airplane, at least in its sleek military forms. Within the context of NACA research, this final stage took the form of "drag cleanup," a program of systematic aerodynamic refinement involving dozens of aircraft that were to see action in World War II.

One might think that with the appearance of the revolutionary DC-3 that the aerodynamics of propeller-driven aircraft had become so refined that no further improvements could be made. An ultrasleek internally braced monoplane with retractable landing gear like the DC-3 could be expected to enjoy something akin to "ideal" aerodynamics, but in actual service, that hardly proved to be the case. Ideally, in aerodynamic terms, the drag of such an airplane—more specifically, its $C_{D,O}$, defined earlier—should be extremely low, "only slightly in excess of that which would be calculated with the use of the total wetted area of the airplane and a skin friction corresponding to a turbulent boundary layer."[25] But, as facts murder theory, real-world circumstances tended to tarnish and undermine this ideal. In actual service, not even aircraft like the DC-3 ever achieved their ideal drag coefficient. Roughness or unevenness on an aircraft's surface, even smashed insects on a wing, disturbed smooth airflow and increased drag. So, too, did antenna or guns projecting outside the smooth basic contour of an airplane. Even air unintentionally leaking through an aircraft structure influenced aerodynamic performance in adverse ways. Sometimes, an aircraft in actual service experienced twice as much drag as its calculated ideal (based on wetted area and a turbulent skin friction factor). This was true, for example, of the Messerschmitt 109 fighter, the archenemy of the Spitfire and the Hurricane during the Battle of Britain in 1940.

What this underscored was the absolute need for "detailed design" work if actual drag were to come anywhere close to ideal values. Everything about an airplane and its parts required meticulous attention. In the early 1930s, experience in the NACA's 30-by-60-foot Full-Scale Tunnel (FST) at Langley indicated that even the most minute external protuberances could seriously undermine aerodynamic performance, but it took a desire to obtain every last ounce of benefit from the airplane design revolution, plus the threat of war, before the U.S. aeronautics community, specifically the U.S. military, sought systematic "drag cleanup" for their aircraft.

What first prompted the NACA's systematic drag cleanup program was a loud cry for help from the U.S. Navy, which in 1938 was unhappy with the 250-mph flight test performance of its new experimental fighter, the Brewster XF2A Buffalo. The Bureau of Aeronautics wanted the NACA to look for " kinks" or "bugs" in the plane's general design and to determine, in only one week's time, what drag reduction could be expected from changes that might readily be incorporated in the event that the XF2A was put into production. The NACA quickly agreed, and even before a formal research authorization was transmitted to Langley laboratory, the NACA flew an XF2A to Langley Field for tests in the FST. The FST team at Langley acted quickly to satisfy the Navy's urgent request, and within a few weeks had discovered ways to increase the Buffalo's speed by 31 mph to 281, more than a 10 percent improvement in performance.

[25] Loftin, *Quest for Performance*, p. 108.

Word got around, and soon not just the Navy but the Army as well was sending all of its new prototypes to Langley for drag cleanup. Between April 1938 and November 1940, Langley gave 18 different military prototypes thorough goings-over in the FST to see if the airplanes could be improved in any way. On this list was the Grumman XF4F-2 Wildcat (June 1938), Vought-Sikorsky SB2U-1 Vindicator (August 1938), Curtiss P-36A Mohawk (August 1938), Curtiss XP-40 Kittyhawk (August 1938), Grumman XF4F-3 Wildcat (September 1939), Republic XP-47 Thunderbolt (May 1940), Chance Vought XF4U-1 Corsair (September 1940), and Consolidated XB-32 Dominator (November 1940). Through its systematic drag reduction investigations in Langley's FST, the NACA did its best to help industry realize some dramatic increases of speed in its production aircraft. This effort can be seen clearly in the contents of both Document 3-29, which concerns drag cleanup of the XF2A, XP-40, and XP-41, and Document 3-30, involving drag cleanup of the Bell XP-39 Airacobra. Although the history of the XP-39 Army fighter plane's top speed was complicated by a series of changes related to its engine, armor, and guns (which significantly increased the machine's overall weight, and thus made speed increases difficult), careful analysis confirms that if the XP-39 had not gone through drag testing, the production Airacobra's top speed of just under 370 mph would have been quite a bit lower, in the range of 25 to 30 mph.[26]

A great deal more drag cleanup work was started when a second NACA laboratory—the Ames Aeronautical Laboratory—opened for business at Moffett Field in Sunnyvale, California, near San Francisco, from 1940 to 1941. With so much of the aircraft industry located on the West Coast, it was quicker and easier to test the new planes at Ames rather than at Langley. Thus, in addition to a new family of high-speed wind tunnels, at Ames, the NACA constructed a 7- by 10-foot wind tunnel and a 40- by 80-foot wind tunnel that also began to serve as workhorse facilities in the comprehensive drag cleanup program.

In terms of the history of aerodynamics, compared to bold new inventions or ingenious new theories, the significance of the NACA's drag cleanup work might seem pedestrian. But this sort of systematic experimental investigation should not

[26] See James R. Hansen, *Engineer in Charge: A History of the Langley Aeronautical Laboratory, 1917-1958* (Washington, DC: NASA SP-4305, 1987), pp. 198-202.

be underestimated—not as a valuable type of engineering science, and certainly not in the context of what happened in World War II. By pointing out ways for aircraft to gain a few extra miles per hour, the NACA effort might often have made the difference in performance between Allied victory and defeat in the air. In measurable aerodynamic terms, the difference seems miniscule. In the case of the Seversky XP-41, for example, cleanup reduced the airplane's drag coefficient from 0.0275 to 0.0226, a difference of 0.0049. But for those who understood the impact of such small numbers, this type of difference was whopping. Realizing seemingly tiny differences such as this figure provided the coup de grace for the airplane design revolution, leading into World War II.

Improved aircraft manufacturing techniques also played a critical role in aerodynamic refinement, particularly through the introduction of flush riveting, which eliminated the dome-shaped rivet heads that protruded untidily above an aircraft's surface (see Document 3-31). Industry even worked to smooth the camouflage paint it used on combat aircraft, trying to minimize the surface roughness of the paint; this step involved water-sanding the airplane's surface with fine sandpaper and rubbing it down with powdered pumice. So important to the aerodynamic performance of an airplane were even the most minor details of design and fabrication that many aircraft manufacturers assembled comprehensive manuals for their employees by which to ensure an aircraft as "clean" as possible. Document 3-32 provides excerpts from one such manual put together by the Boeing Aircraft Company in 1940. Of course, under actual combat conditions, it was nearly impossible to maintain the aircraft in pristine conditions, meaning that the aerodynamic numbers achieved in wind tunnel tests were hardly ever achieved in reality. Still, without the original "cleaning up," the airplanes would have turned out to be that much "dirtier" in actual practice. The starting point of the design in terms of aerodynamic cleanness did make a significant difference.

Thus, it was an almost totally reinvented airplane that fought World War II, and no country did more to generate that revolution and then realize its full potential than the United States, which put the new form of airplane to enormous use in defeating the Axis powers. Of course, another revolution was already brewing, one based on a bold new type of engine, the turbojet. For it, the NACA and the rest of the American aeronautics community did not provide the visionaries; they surfaced in Europe, in the rebellious intellects of British engineer Frank Whittle and German engineer Hans von Ohain. All the while, American aeronautics kept its focus on ever-greater refinement of propeller-driven aircraft, a concentration that may have cost it a chance to explore the concept of jet propulsion more fully—along perhaps with some other revolutionary ideas in aeronautical science and technology such as rockets and cruise missiles. Some critics blamed (and still do blame) the NACA for tunnel vision; others defend the organization by saying that there was no way for it, politically, to get so far out ahead of its clients—and those clients were not interested in pursuing such radical alternative technologies so early. One NACA engineer

has written that "it would have been quite impossible in the prewar period to have any major support from the military, industry, or from Congress for research and development aimed at such radical concepts as the turbojet, the rocket engine, or transonic or supersonic aircraft," and another engineer has commented that "it is certain that if the NACA had had the foresight to do research on the turbine engine in the decade before World War II, the agency would have met with such technical ridicule and criticism about wasting the taxpayers' money that it would either have had to drop it or have been eliminated."[27]

Luckily, the "failure" of NACA researchers and other American engineers to anticipate the jet engine's potential as quickly as a few men in Germany and Great Britain made little difference in the practical outcome of World War II. A handful of jet planes flew in the war, such as the German Messerschmitt Me 262 and the British Gloster Meteor, but they really did not fly effectively enough, in great enough number, or in well enough synchronization with operating squadrons of propeller-driven aircraft, to make a critical difference. Before jet airplanes could reach their potential, they needed to go through the same sort of exhaustive, systems-wide development that prop planes experienced in the two decades between the wars. Jets would eventually go through that years-long, systematic process, as we will see documented in a later volume of this overall study. But by the end of the war, that process had barely started.

Propeller-driven aircraft would continue to be built, and continue to develop, for various purposes well after the appearance of successful jets, even to the present. In 1985, Laurence K. Loftin, Jr., who had then recently retired from NASA after a long career in aeronautical research, called the period from 1945 to 1980 one of "Design Maturity" (see Document 3-31). In terms of basic configuration, propeller-driven airplanes would not change much after the war—although some of them would benefit from the addition of the turboprop, advanced superchargers, and cabin pressurization. In terms of aerodynamics, however, their level of refinement would never surpass the best aircraft of World War II. Attempts would certainly be made to apply new ideas in aerodynamics and other areas, and many excellent propeller-driven airplanes would be built, both as transport aircraft, such as the Vickers Viscount 810 of the late 1940s and Lockheed Super Constellation and Lockheed C-130 turboprop cargo transport of the 1950s, as well as some remarkable general aviation aircraft, such as the Piper Cherokee, Cessna Skyhawk, Beech Bonanza, Cessna 310, and the Beech Super King Air. Much more functional and safe "crop dusters" and other agricultural airplanes would fly, such as the Piper Pawnee. Enthusiasts of "homebuilts" and sailplanes (which, of course, are propellerless) would seek ways

[27] John V. Becker, *High-Speed Frontier: Case Studies of Four NACA Programs, 1920-1950* (Washington, DC: NASA SP-445, 1980), p. 31, and Ira H. Abbott. "A Review and Commentary of a Thesis by Arthur L. Levine, Entitled 'A Study of the Major Policy Decisions of the National Advisory Committee for Aeronautics,' Dated 1963," NASA HQ History Office Archive, HHN-35, Apr. 1964, p. 135.

of improving aerodynamic performance. But generally speaking, the aerodynamics just did not improve that much. What Loftin observed in terms of "design trends" for propeller-driven aircraft since 1945 was clearly a kind of "steady state" that he predicted would continue indefinitely into the future.

When the last volume of *The Wind and Beyond* is published in a few years, readers will want to compare Loftin's assessment with that of Dennis Bushnell, also a NASA Langley aerodynamicist. Unlike Loftin, Bushnell will argue in very strong terms in "The Shape of Things to Come – Views of the Future of Aerodynamics," that aerodynamics has not settled in as a mature or plateau technology and that it has been intellectually constraining to consider it so. Although Bushnell's historical assessment of what has happened specifically in the design of propeller-driven aircraft will not be much different from Loftin's, his explanations for why a steady state has prevailed in the second half of the twentieth century will be. In Bushnell's view, the last word about the form of all aircraft, including those prop-driven, has not yet been written. A new wave of knowledge inspired by a "renaissance of aerodynamics" may change everyone's opinion about what has happened in the past and what could still happen in the future.

The Documents

Document 3-1

Excerpts from Frederick W. Lanchester, *Aerodynamics*: *Constituting the First Volume of a Complete Work on Aerial Flight* (New York: D. Van Nostrand Company, 1908), pp. 10-45.

The following excerpts from Frederick W. Lanchester's 1907 book, *Aerodynamics,* feature his pioneering concept of the "streamline," one of the central themes of the airplane design revolution that was to unfold in the era between the two world wars. As Lanchester described it in his book, his goal was to define an ideal streamline form (by this, he meant an efficient low-drag shape having ultrasmooth contours offering no resistance to the passage of air along its surface). All of the excerpts below are from chapter one of Lanchester's book, entitled "Fluid Resistance and its Associated Phenomena." The author reproduces the continuously running section of the book most concerned with streamlining.

Lanchester's ideas on drag and streamlining directly refuted the widely held but incorrect theory developed earlier by American scientist Samuel P. Langley that skin friction was a negligible factor in aircraft performance. Lanchester was also one of the first to identify and discuss the airflow vortices present at the tips of a wing in flight–a phenomenon that later came to be known as the source for "induced drag." Lanchester gave only preliminary consideration to the notion of induced drag (which actually proved to be quite significant at low speeds), conjecturing that it was simply a necessary part of the cost of gaining lift. The Englishman's focus was, rather, on the ideal streamlined shape, one that would minimize other sources of drag besides induced drag.

Document 3-1, Excerpts from Frederick W. Lanchester, Aerodynamics: Constituting the First Volume of a Complete Work on Aerial Flight *New York: D. Van Nostrand Company, 1908.*

§ 9. On Streamline Form.–When a body of fish-shaped or ichthyoid form travels in the direction of its axis through a frictionless fluid there is no disturbance left in its wake. Now we have seen that in any case the fluid as a whole receives no momentum, so that it is perhaps scarcely legitimate to argue that there is no resistance because there is no communication of momentum, although this is a common statement.[1] It is clear, however, that if there is no residuary disturbance there is no necessary expenditure of energy, and this equally implies that the resistance is nil.

The fluid in the vicinity of a streamline body is of necessity in a state of motion and contains energy, but this energy is conserved, and accompanies the body in its travels, just as in the case of the energy of a wave. It adds to the kinetic energy of the body in motion just as would an addition to its mass.

According to the mathematical theory of Euler and Lagrange, all bodies are of streamline form. This conclusion, which would otherwise constitute a reduction ad absurdum, is usually explained on the ground that the fluid of theory is inviscid, whereas real fluids possess viscosity. It is questionable whether this explanation alone is adequate.

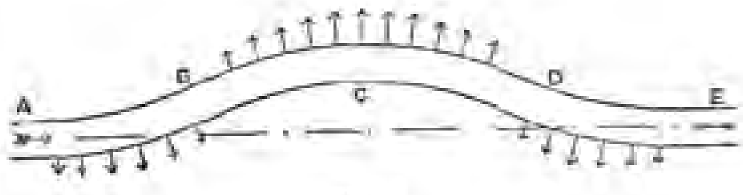


Fig. 2.

§ 10. Froude's Demonstration.–An explanation of the manner of the conservation of kinetic energy, in the case of a streamline body, has been given by the late Mr. A.V. Froude.

Referring to Fig. 2, A, B, C, D, E, represents a bent pipe, through which a fluid is supposed to flow, say in the direction of the lettering, the direction at A and at E being in the same straight line; it is assumed that the fluid is frictionless. Now so long as the bends in the pipe are sufficiently gradual, we know that they cause no sensible resistance to the motion of the fluid. We have excluded viscous resistance by hypothesis, and if the areas at the points A and E are equal there is no change in the kinetic energy. Moreover, the sectional area of the pipe between the points A and E may vary so long as the variations are gradual; change of pressure will accompany change of area on well-known hydrodynamic principles, but no net resistance is introduced; consequently the motion of the fluid through the pipe does not involve any energy expenditure whatever.

Let us now examine the forces exerted by the fluid on different portions of the pipe in its passage. The path of the particles of fluid in the length between the points A and B is such as denotes upward acceleration, and consequently the fluid here must be acted on by an upward force supplied by the walls of the pipe, and the reaction exerted by the fluid on the pipe is equal and opposite. A shorter way is to

[1] This somewhat academic objection would cease to apply if any means could be found to properly define the *idea* which undoubtedly is conveyed to the mind by the argument in question.

regard this reaction as the centrifugal component of the curvilinear path of the flow, and as such it may be indicated by arrows as in the figure.

By assuming the bends in the pipe to be equal and a uniform velocity throughout, it follows that these centrifugal components exactly balance one another, each to each, and the pipe has no unbalanced force tending to push it in one direction or the other. The argument may be found presented in this form in White's "Naval Architecture." The same net result follows, no matter what the exact form of the bends, or whether or not the velocity is uniform, provided the bends are smooth and the cross-section (and therefore the velocity) is the same at E as at A, for under these circumstances the pressure at A will be the same as at E, the applied forces thus being balanced, and there will be no momentum communicated by the fluid in its passage.

When a streamline body travels through a fluid the lines of flow may be regarded as passing round it as if conveyed by a number of pipes as in Fig. 2. It is convenient, and it in no way alters the problem, to look upon the body as stationary in an infinite stream of fluid (Fig. 3); we are then able to show clearly the lines of flow relatively to the surface of the body. Now let us take first the fluid stream that skirts the surface itself, and let us suppose this included between the walls of an imaginary pipe, then forces will be developed in a manner represented in Fig. 2, and these forces may be taken as acting on the surface of the body. It is not necessary to suppose that there is actual tension in the fluid, as might be imagined from Fig. 3, where the forces act outward from the body, this is obviated by the general hydrostatic pressure that obtains in the region; the forces as drawn are those supplied by the motion of the fluid, and can be looked upon as superposed on those due to the static pressure.

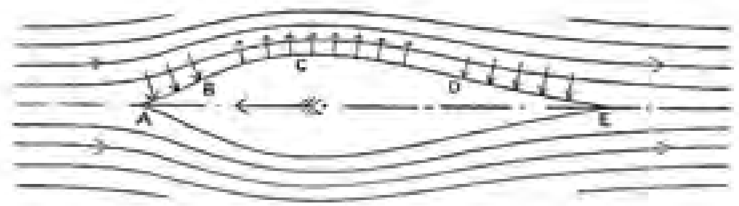


Fig. 3.

If, similarly, we deal with the next surrounding layer of fluid, we find that the pressure to which it gives rise acts to reinforce that of the layer underneath (i.e., nearer the body), and so on, just as in hydrostatics the pressure is continually increased by the addition of superincumbent layers of fluid, and thus we find that the body is subjected to increased pressure acting on its front and rear, and diminished pressure over its middle portion. Now it has been shown, in the case of the pipe, that the algebraic sum of all forces in the line of motion is zero, so that in the streamline body the sum of the forces produced by the pressure on its surfaces will be zero, that is to say, it will experience no resistance in its motion through the fluid.

It may be taken as corollary to the above, that in a viscous fluid the resistance of a body of streamline form will be represented approximately by the tangential resistance of its exposed area as determined for a flat plate of the same general proportions. This is the form of allowance suggested by Froude; a more elaborate and accurate method has been given by Rankine, in which allowance is made for the variation in the velocity of the fluid at different points on the surface of the body. Neither of these methods includes any allowance for viscous loss owing to the distortion of the fluid in the vicinity of the body.

§ 11. The Transference of Energy by the Body.–It is of interest to examine the question of the transference of energy through the streamline body itself from one part of the fluid to the other. For the purpose of reference the different portions of the body have been named as in Fig. 4, the head, the shoulder, the buttock, and the tail, the head and shoulder together being termed (as in naval architecture) the entrance, and the buttock and tail the run. The dividing line between the entrance and run is situated at the point of maximum section, and the dividing line between the head and shoulder on the one hand, and between the buttock and tail on the other, is the line on the surface of the body at which the pressure is that of the hydrostatic "head."

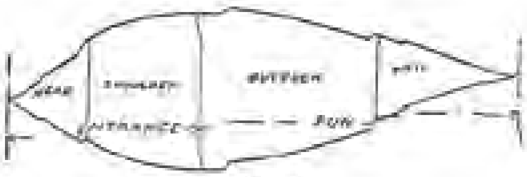


Fig. 4.

Now, as the body advances, the head, being subject to pressure in excess of that due to the hydrostatic "head," is therefore doing work on the fluid; that is to say, transmitting energy to the fluid; the shoulder also advancing towards the fluid is subject to pressure less than that due to hydrostatic head, and is consequently receiving energy from the fluid; the buttock, which is receding from the fluid, is also a region of minus pressure and so does work on the fluid; and lastly, the tail is receding under excess pressure and so receives energy. We thus see that there are two regions, the head and buttock, that give up energy continuously to the fluid, and two regions, the shoulder and tail, that continuously receive it back again. The condition of perfect streamline motion is that the enemy account shall balance.

§ 12. Need for Hydrostatic Pressure, Cavitation.–The motion impressed on the fluid by the pressure region of the head is compulsory, unless (as may happen in the case of a navigable balloon) deformation of the envelope can take place. The motion impressed by the shoulder, on the contrary, depends upon hydrostatic pressure, for otherwise there is no obligation on the part of the fluid to follow the surface of the body. Hydrostatic pressure is necessary to prevent the formation of a void. The pressure measured from the real zero must everywhere be positive, otherwise the fluid will become discontinuous and cease to follow the surface. This is a difficulty that has been actually experienced in connection with screw propellers, and termed cavitation.

§ 13. The Motion in the Fluid.–It has been shown that the head of a streamline form is surrounded by a region of increased pressure. Consequently the fluid as it approaches this region will have its velocity reduced, and the streamlines will widen out, as shown in Fig. 3 (see also Figs. 42, 44, 45, etc.). This behavior of the fluid illustrates a point of considerable importance, which is frequently overlooked. Whenever a body is moving in a fluid, its influence becomes sensible considerably in advance of the position it happens to occupy at any instant. The particles of fluid commence to adjust themselves to the impending change with just as much certainty as if the body acted directly on the distant particles through some independent agency, and when the body itself arrives on the scene the motion of the fluid is already conformable to its surfaces. There is no impact, as is the case with the Newtonian medium, and the pressure distribution is more often than not quite different from what might be predicted on the Newtonian basis.–This behavior of a fluid is due to its continuity.

It follows from elementary considerations that the fluid in the "amidships" region possesses a velocity greater than the general velocity of the fluid (the body, as before, being reckoned stationary). We know that at and about the region C, Fig. 3, the fluid has a less area through which to pass than at other points in the field of flow. It is in sum less than the normal area of the stream by the area of cross-section of the body at the point chosen. But the field of flow is made up of a vast number of tubes of flow, so that in general each tube of flow will be contracted to a greater or less extent, the area of section of the tubes being less at points where the area of the body section is greater. We know that a contraction in a tube of flow denotes an increase of velocity.

Thus on the whole the velocity of the fluid is augmented across any normal plane that intersects the body itself, but the increase of velocity is not in any sense uniform in its distribution. In fact, towards the extremities of the body, and in its immediate neighborhood, we have already seen that the motion of the fluid is actually slower than the general stream.

The motion of the fluid is examined from a quantitative point of view in a subsequent chapter (Chap. III.), where plottings are given of the hydrodynamic solution in certain cases.

§ 14. A Question of Relative Motion.–The motion of the fluid has so far been considered from the point of view of an observer fixed relatively to the body; it will be found instructive to examine the same motions from the standpoint of the fluid itself, that is to say, to treat the problem literally as a body moving through the fluid, instead of as a fluid in motion round a fixed body.

It is evident that the difference is merely one of relative motion. The problems are identical: we require to consider the motions as plotted on co-ordinates belonging to the fluid instead of co-ordinates fixed to the body itself. The relation of the streamlines (which we have so far discussed) to the paths of motion (which we now propose to examine) is analogous to that of the cycloid or trochoid to its generating circle.

§ 15. Displacement of the Fluid.–An unfamiliar effect of the passage of a body through a fluid is a permanent displacement of the fluid particles. This displacement may be readily demonstrated. If a mass of fluid be moved from any one part of an enclosure to any other part, the enclosure being supposed filled with fluid, there is a circulation of fluid from one side to the other during transit; and if we suppose it to be moved from one side to the other of an imaginary barrier surface, then an equal volume of fluid must cross the same barrier surface in the opposite direction. Now it is of no importance whether the thing we move be a volume of fluid or a solid body, so that when a streamline body passes from one side to the other of a surface composed of adjacent particles of fluid, that surface will undergo displacement in the reverse direction to that in which the body is moving, and the volume included between the positions occupied before and after transit will be equal to the volume of the body itself.

Moreover, since the actual transference of the fluid is clue to a circulation from the advancing to the receding side of the body, it will take place principally in the immediate vicinity of the body and less in regions more remote; it is, therefore, immaterial whether the fluid be contained within an enclosure or whether one or more of its confines be free surfaces, provided that continuity is maintained, and that the body is not in the vicinity of a free surface.

§ 16. Orbital Motion of the Fluid Particles.–Since the motion of the fluid results in a permanent displacement, the motion of a particle does not, strictly speaking, constitute an orbit. It is, however, convenient in cases such as the present to speak of the motion as orbital.

If we could follow the path of a particle along any streamline, and note its change of position relatively to an imaginary particle moving in the path and with the velocity of the undisturbed stream, we should have data for plotting the orbital motion corresponding to the particular streamline chosen. Thus we know that the amplitude of the orbit of any particle, measured at right angles to the direction of flight, is equal to that of the corresponding streamline.

We further know that, in general, the particles have a retrograde motion–that is, their final position is astern of their initial position–also that the maximum retrograde velocity is to be found in the region of maximum amplitude. Beyond this we know that the initial motion of any particle is in the same direction as of the body, and that this initial motion is greater for particles near the axis of flight than for those far away.

Let b, b, b, etc., Fig. 5, represent the final position of a series of particles originally situated in the plane a, a, a; then the orbits of these particles will originate on

the plane a, a, a, and terminate of the surface b, b, b, and the motion will be of the character shown.

The form of the surface b, b, b, will be different for different forms of body. It will evidently approach the plane a, a, a, asymptotically, and generally will tend to form a cusp pointing along the axis of flight. The development of this cusp is greatest in cases where the extreme entrance and run are of bluff form, as in the Rankine Oval, Fig. 42, where the point of the cusp is never reached, the surface approaching the axis of light asymptotically. In reckoning the displacement of the fluid (§ 15), the volume included in the cusped surface forward of the plane a, a, must be considered negative, since here the fluid is displaced in the same direction as the motion of the body.

§ 17. Orbital Motion and Displacement, Experimental Demonstration.–The displacement of the fluid and the form of the orbit can be roughly demonstrated by a simple smoke experiment. If a smoke cloud be viewed against a dark background during the passage of a body of streamline form in its vicinity, the retrograde movement of the air is clearly visible. So long as the surface of the body is not too close, the movement is clean and precise, and the general character of the orbit form can be clearly made out; it is found to be, so far as the eye can judge, in complete accord with the foregoing theory. The commencement and end of the orbit, where the motion should be in the same direction as the body, is most difficult to observe, though even this detail is visible if the orbit selected be sufficiently near to the axis of flight. The difficulty here is that the latter part of the orbit is generally lost in consequence of the "frictional wake,"[2] i.e., the current set up by viscous stress in the immediate neighborhood of the body in motion. In all actual fluids a wake current of this kind is set up, and the displacement surface b, b, b, Fig. 5, is obliterated in the neighborhood of its cusp by a region of turbulence.

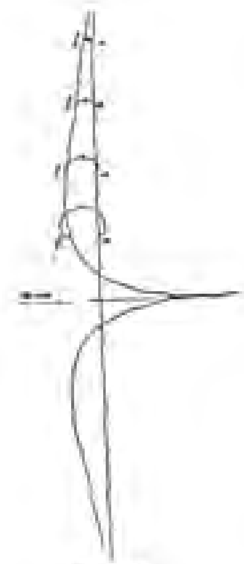


Fig. 5.

§ 18. Orbital Motion, Rankine's Investigation.–The form of the orbits of the fluid particles has been investigated theoretically for a certain class of body by Rankine (Phil. Trans., 1864).

Rankine closely studied the streamlines of a body of oval form, generated by a certain method from two foci (§ 77), and by calculation arrived at the equation to the orbit motion of the particles. The result gives a curve whose general appearance is given in Fig. 6 (actual plotting), in which the arrows represent the motion of the particle, the direction of motion of the body being from left to right.

[2] A term used in naval architecture.

Discussing the particular case in which the eccentricity of the oval vanishes, and the form merges into that of a circle, Rankine says,–"...The curvature of the orbit varies as the distance of the particle from a line parallel to the axis of X, and midway between that axis and the undisturbed position of the particle. This is the property of the looped or coiled elastic curve; therefore when the water-lines are cyclogenous the orbit of each particle of water forms one loop of an elastic curve." Further, he says–"The particle starts from a, is at first pushed forward, then deviates outwards and turns backwards, moving directly against the motion of the solid body as it passes the point of greatest breadth, as shown. The particle then turns inwards, and ends by following the body, coming to rest at b in advance of its original position."

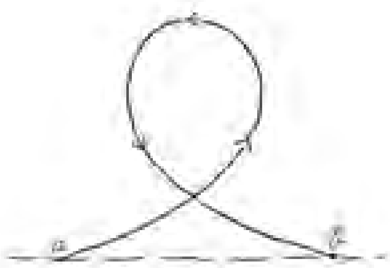


FIG. 6.

This orbit in some respects resembles that arrived at by the author, but differs in the one very important point that, whereas the author's method gives a retrograde displacement of the fluid as the net consequence of the passage of the body, Rankine's conclusion is exactly the contrary.

As the author's result is capable of experimental verification, it is evident that some subtle error must exist in Rankine's argument, the exact nature of which it is difficult to ascertain.

§ 19. Bodies of Imperfect Streamline Form.–In an actual fluid, bodies of other than streamline form experience resistance apart from that directly due to viscosity.

In the practical shaping of a streamline body it is found essential to avoid corners or sharp curves in the line of flow. Bodies in which duo precaution is not taken in this respect offer considerable resistance to motion, and the regions of abrupt curvature give rise to a discontinuity in the motion of the fluid. Thus Fig. 7 represents a double cone moving axially, and it will be noticed that the flow has not time to close in round the run, as it would do in a properly formed streamline body, but shoots past the sharp edge, as indicated in the figure. The region in the rear of the body, Z, is filled with fluid that does not partake of the general flow, and which is termed dead-water.

The resistance experienced by bodies of imperfect form is due to the work done on the fluid, which is not subsequently given back, as is the case with the streamline body. This resistance can be traced to two causes, namely, excess pressure on the surface in presentation and diminished pressure in the dead-water region. The former is of dynamic origin, the energy being expended in directly impressing motion on portions of the fluid; the latter is due to the entrainment or viscous drag experienced by the dead-water at the surface hounded by the live stream. It is generally believed that, in a fluid whose viscosity is negligible, the latter cause would be inoperative, the whole resistance being then due to the excess pressure region in front of the body, the dead-water or wake being at approximately the hydrostatic pressure of the fluid.

The surface separating the live stream and the dead-water constitutes a discontinuity, since the velocity of the fluid, considered as a function of its position in space, is discontinuous. This case is not one of a physical discontinuity, such as discussed in § 12, for the region on either side of the surface is filled with the same kind of fluid; it is rather a kinetic discontinuity, that is to say a discontinuity of motion.

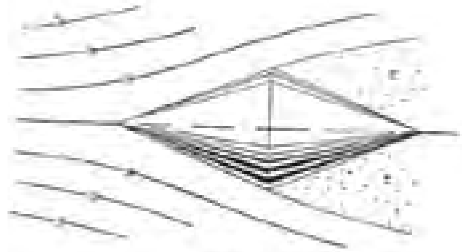


FIG. 7.

§ 20. The Doctrine of Kinetic Discontinuity.–The theory of kinetic discontinuity is of modern origin, having been introduced and developed by Kirchhoff, Helmholtz, and others, to account for the phenomenon of resistance in fluid motion. The analytical theory, based on the hypothesis of continuity, does not in general lead to results in harmony with experience. All bodies, according to the Eulerian theory, are of streamline form, provided that the hydrostatic pressure of the fluid is sufficient to prevent cavitation; we know that in practice this is not the case.

According to the teaching of Helmholtz and Kirchhoff, a kinetic discontinuity can be treated as if it were a physical discontinuity; that is to say, the contents of the dead-water region can be ignored; and this method of treatment is now generally recognized, although not universally so. The controversial aspect of the subject is discussed at length at the conclusion of Chap. III.

The principal objection to the theory of discontinuity is that in an inviscid fluid a surface of discontinuity involves rotation, and therefore, by a certain theorem of Lagrange, it is a condition that cannot be generated.[3] A further objection sometimes raised is that such a condition as that contemplated would be unstable, and that the surface of discontinuity, even if formed, would break up into a multitude of eddies. Whether this is the case or not in an inviscid fluid, it is certain that in a fluid possessed of viscosity a surface of discontinuity does commence to break up from the instant of its formation; but as this breaking up does not affect the problem in any important degree, the objection in the case of the inviscid fluid is probably also without weight.

In a real fluid a finite difference of velocity on opposite sides of any surface would betoken an infinite tangential force. Consequently the discontinuity becomes a stratum rather than a surface, and the stratum will either be a region in which a velocity gradient exists (§ 31), or it will become the seat of turbulent motion (§ 37), the latter in all probability.

[3] Chap. III. §§ 65–71.

The conception of the discontinuity as a surface and the method involving this conception are in no way affected by these considerations. The term surface of discontinuity may be looked upon as an abstraction of that which is essential in a somewhat complex phenomenon.

§ 21. Experimental Demonstration of Kinetic Discontinuity.–The reality and importance of the discontinuous type of motion can be demonstrated conclusively by experiment.

In Fig. 8, a, b, c, is a hollow spherical globe in which d is a tube arranged to project in the manner shown. An ordinary lamp globe and chimney will be found to answer the purpose the former having one of its apertures closed by a paper disc. The whole is carefully filled with smoke and then moved through the air in a direction from right to left, the relative direction of the air being indicated by the arrow.

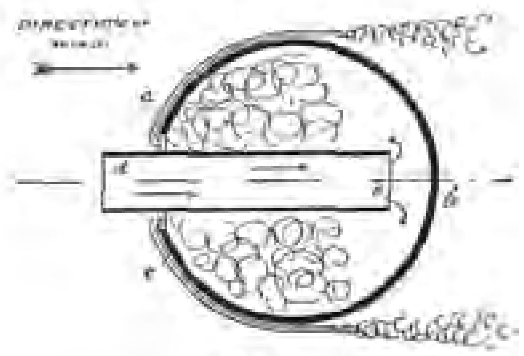
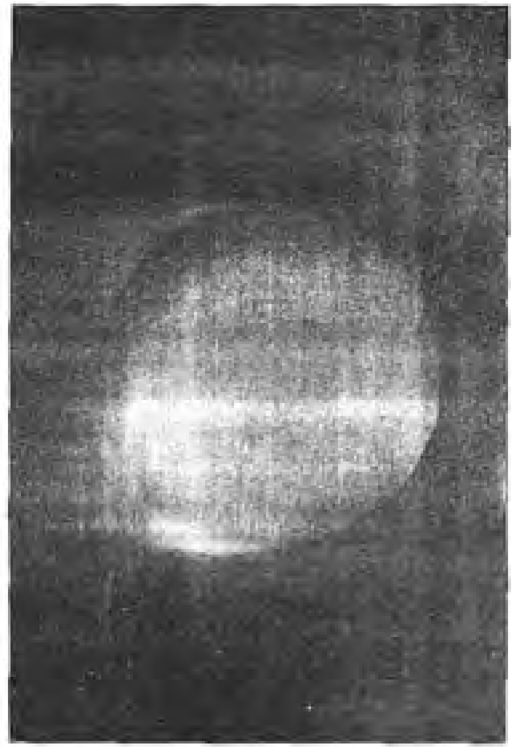


Fig. 8.

It will be found that the air will enter the tube and displace the smoke through the annular aperture. The issuing smoke follows the surface of the sphere in the most approved manner as far as the "equator," but then passes away at a tangent, the stratum discontinuity, the dead-water region, and the turbulent character of motion, being all clearly manifest. The discontinuity, as may have been anticipated, does not appear as a clean-cut surface; it is marked almost from the commencement, as indicated in the figure, by eddy motion; but when we remember that, according to the Fulerian theory, the lines of flow should carry the smoke along a symmetrical path to the opposite pole of the sphere, as in Fig. 45 (Chap. III.), the conclusion is plain.

The author has succeeded in photographing the flow round a cylinder in motion in a smoke-laden atmosphere (Fig. 9). In this example it may be noticed that the surface or stratum of discontinuity arises from a line some distance in front of the plane of maximum section; the difference in the behavior of a cylinder and sphere in this respect is clue to the fact that in the former case the lines of flow are cramped laterally, the motion being confined to two dimensions, whereas in the 1atter case, the motion being in three dimensions, the fluid can "get away" with greater facility. This difference is reflected in the lower coefficient of resistance found experimentally for the sphere than that ascertained for the cylinder. Thus in the experiments of Dines (§ 226) the pressure per square foot of maximum section on a 5/8-in. cylindrical rod was found to be more than double that on a 6-in. sphere, though doubtless the difference in size in the bodies compared may contribute something to the disparity.

The theory of discontinuity also receives support of the most convincing description from the experiments of Mutton, 1788, and Dines, 1889, by which it is shown that the pressure on a solid hemisphere, or a hemispherical cup (such as used on the Robinson anemometer), both in spherical presentation, does not differ from that on a complete sphere to an extent that experiment will disclose. This not only disposes of the streamline sphere of mathematical conception, but proves at the same time the approximate constancy of wake pressure under variation of rear body form. The same lesson is to be gleaned from experiments in the case of the hemisphere, cone, and circular plate (all in base presentation), whose resistance is found to be approximately equal (Fig. 17).



§ 22. Wake and Counterwake Currents.–Reference has already been made to the frictional wake current to which a streamline body gives rise owing to the viscous stress it exerts on the fluid in its neighborhood. With bodies of imperfect form there is, in addition to the frictional wake, a wake current constituted by the contents of the dead-water region, that is, the fluid contained within the surface of discontinuity.

The general motion of the wake current is in the same direction as the body itself, but, owing to the viscous drag exerted on it by the surrounding stream, this motion has superposed on it one of circulation, which probably results in the central portion of the wake traveling actually faster than the body[4] and the outer part slower, though Dines' experiments seem to point to the disturbance being of so complex a character that it is impossible to trace any clearly defined system.[5]

Now, since there can be no momentum communicated to the fluid in sum (§ 5), there must be surrounding the dead-water or wake current a counter-current in the opposite direction to that of the wake, that is, in the reverse direction to the

[4] Since writing this passage the author has observed this "overtaking" current photographed in Fig. 9. It may be faintly discerned in this Figure in the central region of the "dead-water."

[5] "On Wind Pressure upon an Included Surface," Proc. Royal Soc., 1980.

motion of the body; and this counterwake current is being continuously generated, just as the wake current itself, and contains momentum equal and opposite to that of the wake. When in a fluid possessing viscosity the wake and counterwake currents intermingle by virtue of the viscous connection between them, and become involved in a general turbulence, the plus and minus momenta mutually cancel, and the final condition of the fluid at all points is one of zero momentum.

We may regard the counterwake current as a survival of the motion which, we have shown, must exist in the neighborhood of the maximum section of a streamline body (§ 13) opposite in direction to its motion through the fluid. The failure of the stream to close in behind the body means that this motion will persist.

The mingling of the wake and counterwake may be regarded as a phenomenon quite apart from the initial disturbance, and the turbulence or otherwise of the wake does not materially add to or detract from the pressure on the front face of the body, but concerns merely the ultimate disposal of the energy left behind in the fluid.

No distinction is necessary between the frictional wake and the dead-water wake so far as the production of a counterwake current is concerned. The total wake current is the sum of the two, and the total counterwake is equal and opposite to the total wake.

§ 23. Streamline Motion in the Light of the Theory of Discontinuity.–The theory of kinetic discontinuity presents the subject of streamline motion in a new light, and enables us to formulate a true definition of streamline form. Thus–

A streamline body is one that in its motion through a fluid does not give rise to a surface of discontinuity.

In the previous discussion, § 9 et seq., no attempt has been made to delineate streamline form, that is to say (according to the present definition), the form of body that in its motion through a fluid will not give rise to discontinuity. It has been assumed that such a body is a possibility, and from the physical requirements of the case the general character of the body form has been taken for granted.

Under our definition, if, as in the mathematical (Eulerian) theory, we assume continuity as hypothesis, then all bodies must be streamline, which is the well-known consequence. If, on the other hand, as in the Newtonian medium, we assume discontinuity, then it is evident by our definition that streamline form can have no existence, which, again, is what we know to be the case. It remains for us to demonstrate, on the assumption of the properties of an ordinary fluid, the conditions which govern the existence or otherwise of discontinuity, and so control the form of a streamline body.

In order that streamline motion should be possible such motion must be a stable state, so that, if we suppose that by some means a surface of discontinuity be initiated, the conditions must be such that the form of motion so produced is unstable.

Let us suppose that we have (Fig. 10) a streamline body made in two halves, and that the rear half, or run, be temporarily removed; then a surface of discontinuity will be developed, as indicated in the figure. Let now the detached portion be replaced. Then the question arises, What are the changed conditions that will interfere with the permanence of the discontinuous system of flow, as depicted in the figure?

If, in the first place, the fluid be taken as inviscid, and if, for the purpose of argument, we assume that the system of flow indicated in the figure is possible in an inviscid fluid, then it is evident that when the run is replaced we shall not have disturbed the conditions of flow, for our operations have been confined to the dead water region, where the fluid is at rest relatively to the body. Consequently the discontinuous system of flow will persist. That is to say, under the supposed conditions streamline motion is either unstable or is at best a condition of neutral equilibrium. Let us next introduce viscosity as a factor. The conditions are now altered, for the fluid in the dead-water region is no longer motionless, but is in active circulation, and the introduction of the rear half of the body obstructs the free path of the fluid, so that, as the outer layers of the dead-water are carried away by the viscous drag, the fluid in the interior has difficulty in finding its way back to take its place. This difficulty is greatest in the region from which the discontinuity springs, where the dead-water runs off to a "feather edge," and it is evident that some point of attenuation is reached at which the return flow becomes impossible, and the fluid will be "pumped out" or ejected from the region forward of this point. This brings the discontinuity further aft on the body, where the process can be supposed repeated, so that eventually the whole dead-water has been pumped away, and streamline motion supervenes. It is evident that the process will not occur in stages, as above suggested, but will be continuous.

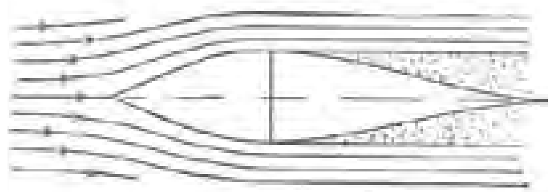


Fig. 10.

It might be supposed from the foregoing argument that the degree of curvature of the surface of the body would not be a matter of importance, as in any case the feather edge of the dead-water would be sufficiently fine to ensure the ejection of some small amount of the fluid, and this process by continuous repetition would eventually clear the wake of its contents. If the surface of the body were frictionless, doubtless this might be the case, but it is established that there is continuity between the surface of an immersed body and the surrounding fluid; that is to say, there is the same degree of viscous connection between the fluid and the surface as there is between one layer of the fluid and another. The consequence of this is that the dead-water never fines off entirely, but extends forward as a sort of sheath enveloping the whole surface of the body, and if the curvature at any point is too rapid, the ejection may not prove effective, and the discontinuity will persist. It is evident therefore that there will be some relation between the bluffness of form permissible and the viscosity of the fluid, and, other things being equal, the less the viscosity the finer will have to be the lines of the body. The theory evidently also points to the importance of smoothness of surface when the critical conditions are approached.

The subject is not yet exhausted. We know that the thickness of the stratum of fluid infected by skin friction increases with the distance from the "cut-water"; that is to say, the factor on which the curvature of the surface probably depends is relatively more important on the buttock than on the shoulder. Hence we may expect that the lines of entrance can with impunity be made less fine than the lines of the run.

Again, all forces due to the inertia of the fluid vary as the square of the velocity; those due to viscosity vary in the direct ratio of the velocity (§ 31). Therefore for different velocities the influence of viscosity predominates for low velocities, and that of inertia when the velocity is high. Consequently the form suited to high velocity will be that appropriate to low viscosity, and vice versa; that is to say, the higher the velocity the finer will be the lines required.

§ 24. Streamline Form in Practice.–The practical aspect of streamline form may be best studied from the bodies of fishes and birds, the lines of which have been gradually evolved by nature to meet the requirements of least resistance for motion through a fluid, water or air, as the case may be.

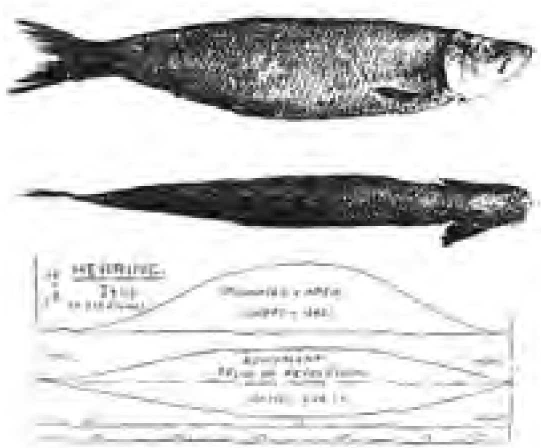


Fig. 11.

Since all animals have functions to perform other than mere locomotion, we find great diversity of detail, and we frequently meet with features whose existence is in no way connected with the present subject. We may readily recognize in these cases the exceptional development of certain organs or parts to meet the special requirements of a particular species, and by a sufficiently wide selection we can eliminate features that are not common, and so arrive at an appreciation of that which is essential. Thus the herring (Fig. 11), the trout (Fig. 12), or the salmon (Fig. 13) may be cited its typically fish shaped fish.

Beyond the lessons to be derived from these natural forms, there is very little practical information available. The lines of ships are governed by considerations foreign to the subject, the question of wave-malting, for example, being a matter of vital importance. The submarine has not yet reached its stage of development that would justify its form being taken as a fully evolved model; also, for obvious reasons,

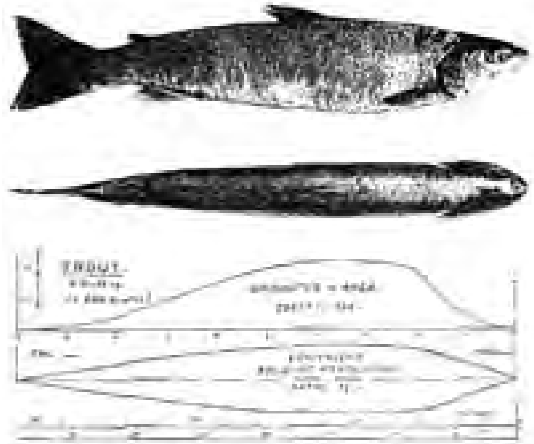
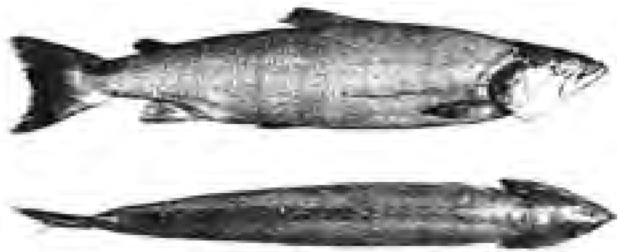


Fig. 12.

this type of vessel is one of which but little information has been published.

In Figs. 11 and 12 curves are given whose ordinates represent the area of cross-section at different points. This curve has been obtained by differentiating a displacement curve plotted from a series of immersion measurements. These measurements were made by a method of displacement, the fish, suspended tail downward, being lowered stage by stage into a vessel of water, measurements being made of the overflow.

The area curves have been further translated into the form of solids of revolution, which may be taken as the equivalent of the original form in each case. Some doubt exists as to the exact form in the region of the head, owing to the water entering the gills. The effect of this is very evident in the case of the trout (Fig. 12), where the form has been "made good" by a dotted line.



For the purpose of comparison outline elevations are given in Fig. 14 of three types of Whitehead torpedo. These are forms that have been developed by long experience, but the shape is largely dictated by special considerations. The bluff form of head, for example, in models A and C is adopted in order to bring the explosive charge into as close proximity as possible to the object attacked. It probably also gives a form that is more easily steered.

§ 25. Streamline Form.–Theory and Practice Compared.–Before a rigid comparison can be instituted between the theoretical results of § 23 and the actual forms found in nature considerable further information is required. We do not know with accuracy the speeds for which the different fish forms have been designed or are best adapted. We also lack knowledge on certain other important points. The present comparison must therefore be confined to generalities.

In the first place, we may take it that the conclusion as to the bluffer form being that suited to greater viscosity is fully borne out in practice, though the whole of the considerations bearing on this point are not here available. It is explained in Chap. II. that the viscosity divided by density (or kinetic viscosity) is the proper criterion in such a case as that under discussion, and on this basis air is far more viscous than water, so that we shall expect to find aerial forms bluffer in their lines than aqueous

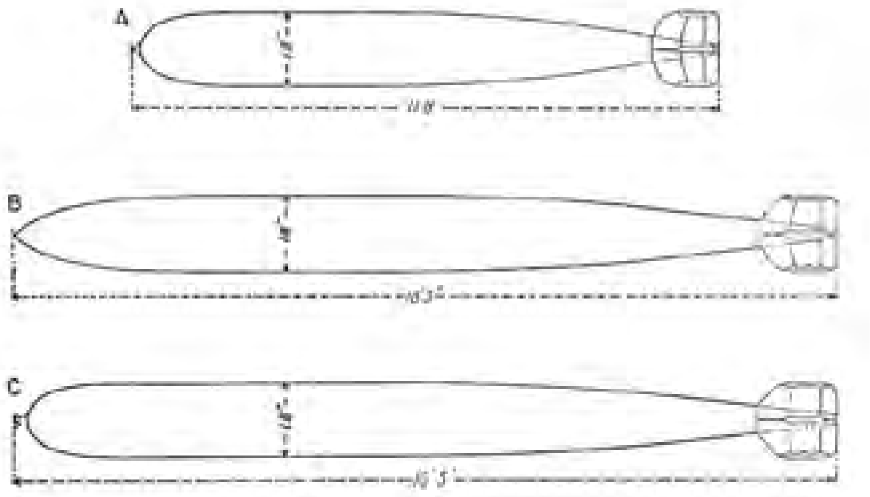


Fig. 14

forms. Taking the solid of revolution as the basis of comparison, we have in the case of the herring and the trout the length approximately seven times the maximum diameter. The general ratio found amongst bird forms is about three or four to one, the samples chosen for measurement being as far apart as the albatross and the common sparrow. Consequently we find that the theoretical conclusion receives substantial confirmation.

The relation of fineness to speed is not so easy of demonstration, owing to the absence of accurate data. It would, however, seem to be sufficiently obvious as a matter of general experience that our conclusions hold good. It is almost certain that in general the fish with the finer lines are the faster swimmers. If this conclusion be accepted, the viscosity relation of the preceding paragraph is emphasized, for there is no doubt that the average speed of flight is greatly in excess of any ordinary velocity attained by fish.

§ 26. Mutilation of the Streamline Form.–There are certain types of body that may be regarded as mutilations of the streamline form, and the consequences of such mutilation may now be examined.

If, in the case of a body propelled at a constant velocity, the entire run be removed, as in § 23, the consequence is a surface of discontinuity emanating from the periphery of section. Under these circumstances, if we neglect the influence of viscosity and the consequent loss of wake pressure, the work done appears wholly in the counterwake current, on the production of which energy is being continu-

ously expended. This performance of work is otherwise represented by a resistance to motion, being the difference between the excess pressure on the head and the diminished pressure on the shoulder, according to the principle explained in § 11. If now we restore the buttock, so that the mutilation is confined to the simple loss of the tail (Fig. 15), the diminished pressure on the buttock acts as a drag upon the body, and more work must be expended in propulsion. This additional energy will appear in the fluid as a radial component in the motion of the stream which does not exist if the whole run is removed. It is probable that some of this energy is restored by an increase in the pressure of the dead water due to the converging stream, but we have no means of making a quantitative computation.

An illustration of this principle may be cited in the type of hull employed in a modern racing launch. The stern is cut off square and clean, and may constitute the maximum immersed section. There would seem in fact to be no logical compromise between a boat with an ordinary well-proportioned entrance and run, and one in which the latter is sacrificed entirely. In such a form, when traveling at high speed the water quits the transom entirely, and consequently sacrifice is made of the hydrostatic pressure on the immersed transom area. The point at which the front half of a boat thus takes less power for its propulsion than the whole is probably about that speed at which the skin friction on the run (the after-half), if present, exceeds the hydrostatic pressure on the maximum immersed section. This does not, however, determine the point at which it pays to make the sacrifice, owing to the fact that for the same capacity the truncated form has to be that of a larger model. The rating rule also exerts an arbitrary influence. When, as is usual, the length is penalized, an additional inducement is offered for the designer to adopt the truncated type. When the truncated type of hull is adopted it is advantageous to employ shallow draught, for the hydrostatic pressure for a given displacement is less. This form is also partly dictated by considerations relating to propulsion.

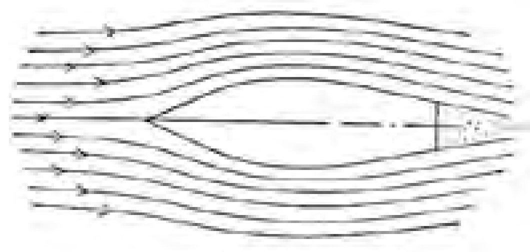


Fig. 15.

§ 27. Mutilation of the Streamline Form (continued).–In Fig. 16, A and B, the consequences of truncating the fore body, or entrance, of a streamline body are indicated diagrammatically. If, as in A, the mutilation be slight, the result may be merely a local disturbance of the lines of flow. A surface of discontinuity will probably arise, originating and terminating on the surface of the body in the manner shown. It is possible that if the streamline body be traveling at something approaching its critical velocity (at which even in its complete form it is on the point of giving rise to discontinuity), a minor mutilation such as here suggested might have more serious consequences.

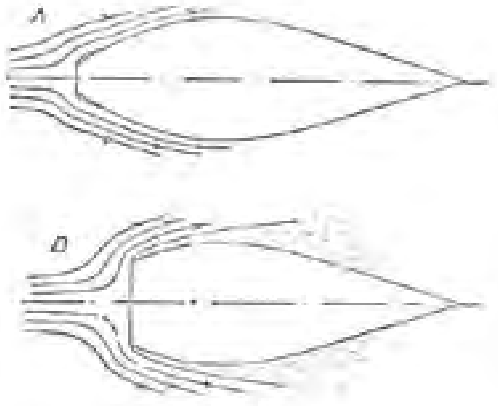


Fig. 16.

If the greater part of the entrance be removed, as shown at B, the surface of discontinuity generated quits the body for good, and the resistance becomes immediately as great as that of a normal plane of area and form equal to that of the section. This is in harmony with the experiments of Hutton and Dines, to which reference has already been made (Fig. 17), the three bodies shown being found to offer the same resistance within the limits of experimental error.

It is evident that the dictum of the late Mr. Froude, that it is "blunt tails rather than blunt noses that cause eddies" (and therefore involve a loss of power), is applicable only to bodies having already some approximation to streamline form. It is obviously useless to provide a nice sharp tail if previous attention has not been given to the shoulder and buttock lines. Mr. Froude probably meant that in a well-designed streamline form the tail should be finer in form than the head, a matter that up to his time had presumably been neglected.

The primary importance of easy shoulder lines has been long recognized as a fundamental feature in the design of projectiles. A full-sized section of a Metford 303 bullet, illustrating this point, is given in Fig. 18, and a streamline form of which it may be regarded as a "mutilation" is indicated by the dotted line.

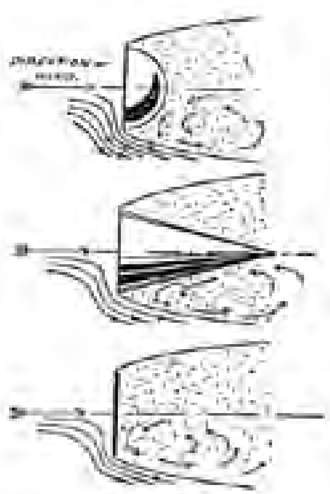


Fig. 17.

§ 28. Streamline Flow General.–Let us suppose an approximate streamline form to be built of bricks, and, in the first place, we will assume that the bricks are so small as to merely give rise to a superficial roughness. Then this roughness will add to the skin friction and will give rise to some local turbulence, but the general character of the flow system remains as before. We may go further and suppose the bricks so large as to form steps capable of giving rise to surfaces of discontinuity (Fig. 19). Then the resistance will be increased, and the layer of fluid next the body will be violently stirred up; but if we examine the fluid some distance away we shall still find it comparatively unaffected. If we now suppose the body to consist of a few large blocks, the depth of fluid affected by turbulence will be greater, but at a sufficient distance away we may still expect to find

lines of flow of characteristic streamline form. We may therefore generalize and say, All bodies passing through a fluid are surrounded by a streamline system of a flow of a greater of less degree of perfection depending upon the conformability or otherwise of the surface of surfaces of the body.

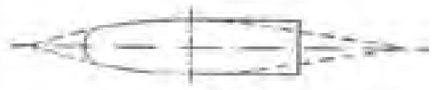


Fig. 18.

This proposition, if not sufficiently obvious from the considerations above given, may easily be demonstrated experimentally.

In the experiment described in § 17, the orbital motion of the particles of the fluid is demonstrated by the motion of an ichthyoid body in air irregularly charged with smoke. This orbital motion, with its consequent displacement, is quite characteristic, and if other shapes of the body be substituted for the streamline form, the motion can be observed much closer to the axis of flight than is the case for a sphere or other bluff form; also when the movement is complete nothing further happens. In the case of a sphere, the looked for movement duly takes place; but immediately after the whole of the fluid under observation is involved in a state of seething turbulence, where the wake and counterwake currents are mingling. If the point of observation is sufficiently remote, the orbital motion may be detected, even in the case of the normal plane, beyond the immediate reach of the wake turbulence.

§ 29. Displacement due to Fluid in Motion.–It has been shown (§ 15) that the fluid in the neighborhood of the path of flight of a streamline body undergoes displacement, and that the total displacement is equal to the volume of the body. It might be expected in the case of the normal plane, which possesses no volume, that the displacement would be nil, and such would doubtless be the case if the form of flow were that of the Eulerian theory.

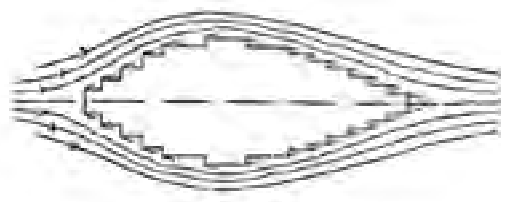


Fig. 19.

In actuality the normal plane, in common with bodies of bluff form, carries a quantity of fluid bodily in its wake, which from the present point of view becomes in effect part of the body, so that the displacement manifests itself just as if the plane were possessed of volume. This is characteristic of all bodies that give rise to discontinuous motion; the displacement is greater than the actual volume of the body. If there were no mingling of the wake and counterwake currents, the displacement would be infinite, for the counterwake current would persist indefinitely.

In the case of a streamline body, a certain amount of fluid is carried along with the body by viscosity, and this similarly increases the effective displacement volume.

It would appear from actual observation that, where the displacement is due to the attendant fluid, the outer streamlines have a motion closely resembling that produced by a streamline body, but that those nearer the axis of flight terminate in the turbulent wake; the commencement of the orbit is all that can be seen.

§ 30. Examples Illustrating Effects of Discontinuous Motion.–On the practical importance of the study of motion of the discontinuous type it is unnecessary to dwell. It is at present the only basis on which it is possible to account for the phenomenon of fluid resistance as experimentally known. Beyond this there are many examples and illustrations which are of especial interest, considered either as proofs of the theory itself or in relation to their actual consequence or utility.

A useful application of the principle is found in the screen employed on fast steamships to protect the navigating officer, and frequently the "watch," from the rush of air, without obstructing the field of vision. This is illustrated diagrammatically in Fig. 20, in which it will be seen that the live stream is carried clear over the sailor's head, the latter being protected by the surface of discontinuity. A similar device is frequently adopted in connection with the dashboard of a motor car.

Evidence of the most striking kind of the existence of a surface of discontinuity is sometimes met with in the growth of trees in the immediate vicinity of the edge of a cliff (Fig. 21). It may be seen that the form of the surface is clearly delineated, the tree top being cut away as though it might have been sheared off by a stroke of a mighty scythe.

Fig. 20.

An interesting example of an indirect effect of discontinuity is to be found in the effect of "cut" or "side" on the flight of a ball. Let a ball (Fig. 22) moving in the direction of the arrow A have a spin in the direction of the arrow B. Now where the direction of motion of the surface of the ball is the same as the relative motion of the fluid, as at D, the surface will assist the stream in ejecting the dead water, so that the discontinuity will be delayed, and will only make its appearance at a point some distance further aft than usual. On the other hand, on the side that is opposing the stream the surface of the ball will pump air in, and so assist the discontinuity, which will make its appearance prematurely. The net result of this is that the counterwake will have a lateral component (downwards in the figure), and, on the principle of the continuous communication of momentum, there will be a reaction on the ball in the opposite direction, that is to say upwards. A ball may therefore be sustained against gravity or be made to "soar" by receiving a spin in the direction shown, or, if the spin be about a vertical axis, the path of the ball will be a curve (in plan), such that the aerodynamic reaction will be balanced by centrifugal force.

Fig. 21.

The actual means by which the reaction acting on the ball comes about may be understood from either of two points of view. We may (Fig. 22) regard this reaction as the centrifugal effect of the air passing over the ball preponderating greatly over that of the fluid passing underneath, or if we anticipate a knowledge of hydrodynamic theory (Chap. III.), we know that the greater proximity of the lines of flow in the former region is alone sufficient to indicate diminished pressure. The lines as drawn in the figure are not plottings–there is no way known of plotting a field of flow of this degree of complexity–but they may be taken as a very fair representation of what the plotting would be if it could be effected.

The reason that the streamlines have been shown rising to meet the ball in its progress will be better understood in the light of Chaps. III. and IV. This detail is related to more advanced considerations than can be entered into at present.

A further interesting example is found in the aerial tourbillion[6] (Fig. 23), in which the rotor K is a stick of segmental section mounted to revolve freely about the axis L. The plane face of the rotor is set truly at right angles to the axis of rotation. If this apparatus be held in a current of air with the plane face fronting the wind, as, for instance, by holding it outside the window of a railway carriage in motion, the rotor evinces no tendency to go round in the one direction or the other. If, however, a considerable initial spin be imparted in either direction, the wind will suddenly get a bite, so to speak, and the rotor will gather speed and spin at an enormous rate, as if it were furnished with sails like a well-designed windmill.

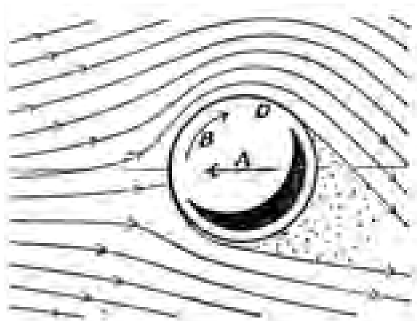


Fig. 22.

Referring to Fig. 24, we have at a the type of flow illustrated to which the blade of the rotor will give rise when its motion is normal to the air; b similarly indi-

[6] This interesting aerodynamic puzzle was first brought to the notice of the author by Mr. Henry Lea, consulting engineer, of Birmingham, who, it would appear, had it communicated to him by Mr. A. S. Dixon, who in turn had it show him when traveling in Italy by Mr. Patrick Alexander. The author has taken no steps to trace the matter further. The explanation here given is his own.

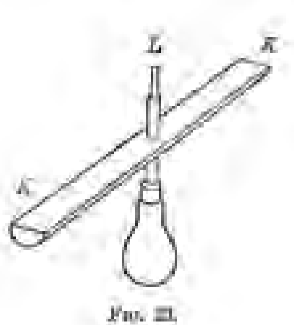


Fig. 23.

cates the form of flow when the rotor is going round slowly, not fast enough for the air to take hold. In both these figures we have the flow independent of the "rear body form," and the rotor behaves just as if it were a flat plate. Now, let us suppose that the rotor be given a sufficient initial spin to bring about the state of things represented at c.

The surface of discontinuity that ordinarily springs from the leading edge has got so close to the rear body of the rotor as to have ejected the "dead-water" on that side, and the resulting form of flow will be something like that illustrated in Fig. 25. Here the pressure on the left-hand side (as shown) will be that of the "dead-water," which is, as we know, somewhat less than that of hydrostatic head, while that on the right hand side will, owing to the centrifugal component of the stream, be very

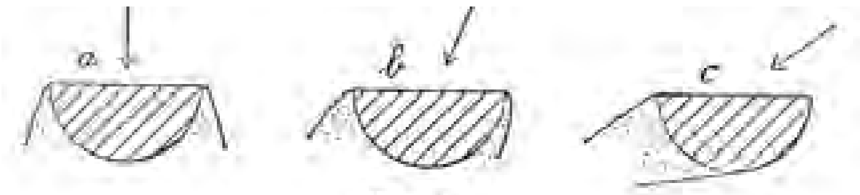


Fig. 24.

much lower; that is to say, the rotor will experience a force acting from left to right which is in the direction of the initial spin, so that the motion will be accelerated and will continue. The fact that the propelling force only comes into existence when the initial spin is sufficient to eject the dead water from the leading aide of the rotor blade fully explains the observed fact that a very considerable initial spin is necessary.

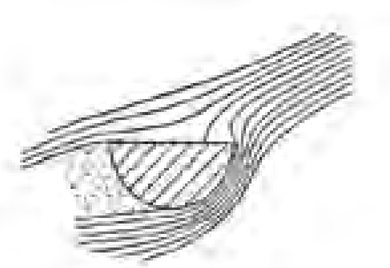


Fig. 25.

Document 3-2

Ludwig Prandtl, "Motion of Fluids with Very Little Viscosity," NACA Technical Memorandum No. 452 (Washington, DC, 1928), translation of "Über Flüssigkeitsbewegung bei sehr kleiner Reibung," in *Vier Abhandlungen zur Hydrodynamik und Aerodynamik,* ed. L. Prandtl and A. Betz (Göttingen: Selbstverlag der AVA, 1927, reissued 1944), pp. 1-8. The 1927 publication was a reprint of Prandtl's 1904 paper that had appeared in *Verhandlungen des III. Internationalen Mathematiker-Kongresses, Heidelberg, 1904,* ed. A. Krazer (Leipzig: Teubner, 1905), pp. 484-91.

This paper in which Germany's Dr. Ludwig Prandtl of the University of Göttingen introduced the concept of what came to be known as the "boundary layer" is, without question, one of the most classic expressions in the entire history of aerodynamics. Never before had anyone in fluid mechanics so embodied the ideal of binding theory and practice into a unified whole, one in which abstract theorems and experimental facts worked together for the purpose of fundamental applications.

In his 1982 book, *Bringing Aerodynamics to America,* historian Paul A. Hanle examined "The Achievement of Ludwig Prandtl" in detail. As Hanle explained, Prandtl's was an achievement crucial to "bringing aerodynamics to America" notably because two of his prize students, Max Munk and Theodore von Kármán, were to play critical roles after 1920 in the development of aeronautical research and development (R & D) capabilities in the United States. In Hanle's view, Prandtl's 1904 paper typified not only his own mode of research but also that of coming generations of scientists and engineers dedicated to aerodynamic R & D. Prandtl's arguments were "essentially physical," based on his "intuition reinforced by his confirming experiments." He gave roughly equal time to explaining his theory and to verifying it by reference to numerous drawings and photographs of flow taken from experiments in a water canal. The 1904 paper presented an approximate solution to the problem he posed, arrived at by numerical computation. Although highly technical, the piece could be easily surveyed. Even today, it deserves our attention for what it reveals about Prandtl's scientific turn of mind, an approach to solving problems that would influence American aerodynamics in many essential ways for years to come (*Bringing Aerodynamics to America* [Cambridge, MA, and London: The MIT Press, 1982], pp. 43-4.).

The central idea of Prandtl's paper rested in two sentences stated right near the beginning: "I have set myself the task of investigating systematically the motion of a fluid of which the internal resistance can be assumed very small. In fact, the resistance is supposed to be so small that it can be neglected wherever great velocity differences or cumulative effects of the resistance do not exist." As Hanle explained, what Prandtl called his "systematic investigation" in truth consisted of "treating no more than a few 'single questions' in broad outline" (Hanle, pp. 44-5). From this basis, it did not take him long to present the most important point of his paper: "By far, the most important question of this problem is the behavior of the fluid at the surfaces of the solid body." Sensing something physical about the airflow, Prandtl postulated the existence of a thin "transition layer" in which viscous effects were in fact extremely significant. Two insights in particular led him to hypothesize what came to be known as the boundary later; they were, in Hanle's words, that "the fluid must be immobile at the surface of the body" and that "the fluid must follow classical hydrodynamic streamlines only a short distance from the surface" (Hanle, p. 45).

In laying out a plan for solving problems involving the transition layer and presenting the necessary governing equations, Prandtl made the consequences of his hypothesis clear: "According to the above, the treatment of a certain process of flow therefore reduces to two parts in mutual interaction: one has on the one hand a free fluid, which can be treated as inviscid according to the vortex principles of Helmholtz; on the other hand, the transition layers at the fixed boundaries, whose motion is governed by the free fluid, but which for their part give the free motion its characteristic features by the emission of vortex layers."

Three other points should be made about this paper (all of them echoing Hanle) before sending the reader off to discover Prandtl's astuteness up close and personally. First, Prandtl gave absolutely no thought at this point in 1904 to the relevance of his hypothesis to the science of flight. He did not even become slightly interested in any theory of flight until 1906. So, "rather than his interest in flight leading Prandtl to the discovery of the boundary layer, cause and effect were likely reversed" (Hanle, p. 47).

Second, as suggested earlier, Prandlt's paper made little impression on the 80 or so people attending the Third International Congress of Mathematicians at Heidelberg. This helps to explain why it took as long as it did for his boundary-layer hypothesis to capture any widespread attention. Finally, it is remarkable, given its venerated place in the history of aerodynamics, that the 1904 paper was only eight pages long. Nevertheless, in those few pages, Prandtl laid the foundation for the modern theory of aerodynamic drag. In doing so, he provided in a sort of time capsule an important building block for the airplane design revolution of the 1920s and 1930s.

Document 3-2, Ludwig Prandtl, "Motion of Fluids with Very Little Viscosity,"
NACA Technical Memorandum No. 452 ***Washington, DC, 1928.***

TECHNICAL MEMORANDUMS

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

NO. 452

MOTION OF FLUIDS WITH VERY LITTLE VISCOSITY

By L. Prandtl

From "Vier Abhandlungen zur Hydrodynamik and Aerodynamik"

Gottingen, 1927

Washington
March, 1928

National Advisory Committee For Aeronautics Technical Memorandum No. 452 Motion Of Fluids With Very Little Viscosity.*

By L. Prandtl.

In classic hydrodynamics the motion of nonviscous fluids is chiefly discussed. For the motion of viscous fluids, we have the differential equation whose evaluation has been well confirmed by physical observations. As for solutions of this differential equation, we have, aside from unidimensional problems like those given by Lord Rayleigh (Proceedings of the London Mathematical Society, 11 page 57 = Papers I page 474 ff.), only the ones in which the inertia of the fluid is disregarded or plays no important role. The bidimensional and tridimensional problems, taking viscosity and inertia into account, still await solution. This is probably due to the troublesome properties of the differential equation. In the "Vector Symbolics" of Gibbs, ** this reads

$$\rho((\partial/v)/(\partial/t) + v_0\, \Delta v) + \Delta\, (V + p) = k\Delta^2 v$$

in which v is the velocity; ρ, the density; V, a function of the power; p, pressure; k, viscosity constant. There is also the continuity equation

$$\text{div } v = 0$$

for incompressible fluids, which alone will be here considered.

* "Ueber Flussigkeitsbewegung bei sehr kleiner Reibung." This paper was read before the Third International Congress of Mathematicians at Heidelberg in 1904. From "Vier Abhandlungen zur Hydrodynamik und Aerodynamik," pp. 1-8, Gottingen, 1927.

** aob scalar product, a x b vector product, Δ Hamilton differentiator

$$(\Delta = i\ (\partial/\partial x) + j\ (\partial/\partial y) + k\ (\partial/\partial z).$$

From the differential equation, it is easy to infer that, for sufficiently slow and also slowly changing motions, the factor ρ, in contrast with the other time, can be as small as desired, so that the effect of the inertia can here be disregarded with sufficient approximation. Conversely, with sufficiently rapid motion, the quadratic

term v o Δ v (change of velocity due to change of location) is large enough to let the viscosity effect appear quite subordinate. The latter almost always happens in cases of fluid motion occurring in technology. It is therefore logical simply to use here the equation for non-viscous fluids. It is known, however, that the solutions of this equation generally agree very poorly with experience. I will recall only the Dirichlet sphere, which, according to the theory, should move without friction.

I have now set myself the task to investigate systematically the laws of motion of a fluid whose viscosity is assumed to be very small. The viscosity is supposed to be so small that it can be disregarded wherever there are no great velocity differences nor accumulative effects. This plan has proved to be very fruitful, in that, on the one hand, it produces mathematical formulas, which enable a solution of the problems and, on the other hand, the agreement with observations promises to be very satisfactory. To mention one instance now: when, for example, in the steady motion around a sphere, there is a transition from the motion with viscosity to the limit of nonviscosity, then something quite different from the Dirichlet motion is produced. The latter is then only an initial condition, which is soon disturbed by the effect of an ever-so-small viscosity.

I will not take up the individual problems. The force on the unit area, due to the viscosity, is

$$K = k\, \Delta^2\, v$$

If the vortex is represented by w = ½ rot v, then K = −2 k rot w, according to a well-known vector analytical transformation, taking into consideration that div v = 0. From this it follows directly that, for w = 0, also K = 0, that is, that however great the viscosity, a vortexless flow is possible. If, however, this is not obtained in certain cases, it is due to the fact that turbulent fluid from the boundary is injected into the vortexless flow.

With a periodic or cyclic motion, the effect of viscosity, even when it is very small, can accumulate with time. For permanence, therefore, the work of K, that is, the line integral ∫ K o d s along every streamline with cyclic motions, must be zero for a full cycle.

$$\int K \text{ o d s} = (V_2 + p_2) - (V_1 + p_1).$$

A general formula for the distribution of the vortex can be derived from this with the aid of the Helmholtz vortex laws for bidimensional motions which have a flow function ψ (Cf. "Encyklopadie der mathematischen Wissenschaften," Vol. IV, 14, 7).

With steady flow we obtain *

$$-(dw/d\psi) = (V_2 + p_2) - (V_1 + p_1) \,/\, P$$
$$2\,k \int \text{v o d s}$$

With closed streamlines this becomes zero. Hence we obtain the simple result that, within a region of closed streamlines, the vortex assumes a constant value. For axially symmetrical motions with the flow in meridian planes, the vortex for closed streamlines is proportional to the radius w = cr. This gives a force K = 4 kc in the direction of the axis.

The most important aspect of the problem is the behavior of the fluid on the surface of the solid body. Sufficient account can be taken of the physical phenomena in the boundary layer between the fluid and the solid body by assuming that the fluid adheres to the surface and that, therefore, the velocity is either zero or equal to the velocity of the body. If, however, the viscosity is very slight and the path of the flow along the surface is not too long, then the velocity will have its normal value in immediate proximity to the surface. In the thin transition layer, the great velocity differences will then produce noticeable effects in spite of the small viscosity constants.

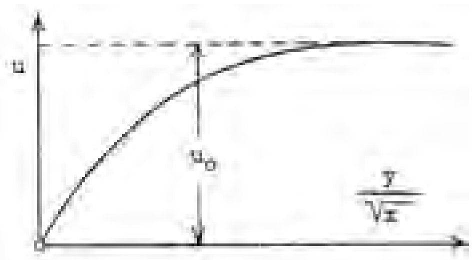


Fig.1.

This problem can be handled best by systematic omissions in the general differential equation. If k is taken as small in the second order, then the thickness of the transition layer will be small in the first order, like the normal components of the velocity. The lateral pressure differences can be disregarded, as likewise any curvature of the streamlines. The pressure distribution will be impressed on the transition layer by the free fluid.

For the problem which has thus far been discussed, we obtain in the steady condition (X-direction tangential, Y-direction normal, u and v the corresponding velocity components) the differential equation

$$\rho\ (u\ (\partial u/\partial x) + v\ (\partial u/\partial y) + dp/dx = k\ (\partial^2 u/\partial y^2)$$
and
$$\partial u/\partial x + \partial v/\partial y = 0.$$

* According to Helmholtz, the vortex of a particle is permanently proportional to its length in the direction of the vortex axis. Hence we have, with steady even flow on each streamline (ψ = const.), w constant, consequently w = fψ. Herewith

$$\int K\ o\ d\ s = 2k\int \text{rot}\ w\ o\ d\ s = 2k\ f'\ (\psi) \int \text{rot}\ \psi\ o\ ds = 2k\ f'\ (\psi) \int v\ o\ d\ s.$$

If, as usual, dp/dx is given throughout, as also the course of u for the initial cross section, then every numerical problem of this kind can be numerically solved, by obtaining the corresponding $\partial u/\partial x$ by squaring every u. Thus we can always make progress in the X-direction with the aid of one of the well-known approximation

methods (Cf. Kutta, "Zeitschrift fur Math. und Physik," Vol. 46, p. 435). One difficulty, however, consists in the various singularities developed on the solid surface. The simplest case of the conditions here considered is when the water flows along a flat thin plate. Here a reduction of the variables is possible and we can write u = f(y/√x). By the numerical solution of the resulting differential equation, we obtain for the drag the formula

$$R = 1.1 \ldots b\sqrt{(}\ k\ \rho\ l\ u_0^3$$

(b width, l length of plate, u_0 velocity of undisturbed water opposite plate). Figure 1 shows the course of u.

The most important practical result of these investigations is that, in certain cases, the flow separates from the surface at a point entirely determined by external conditions (Fig. 2). A fluid layer, which is set in rotation by the friction on the wall, is thus forced into the free fluid and, in accomplishing a complete transformation of the flow, plays the same role as the Helmholtz separation layers. A change in the viscosity constants k simply changes the thickness of the turbulent layer (proportional

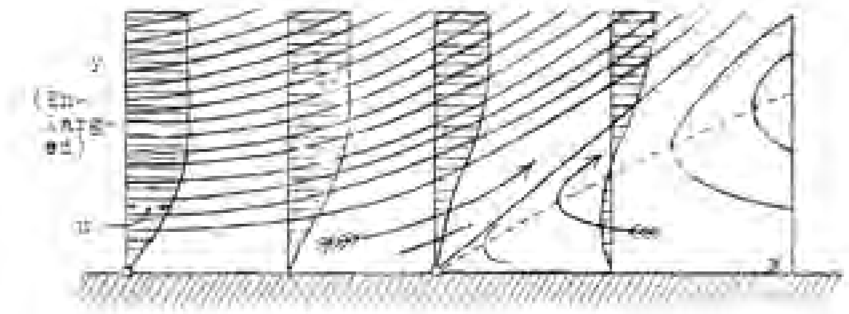


Fig.2

to the quantity √(kl/ρu)), everything else remaining unchanged. It is therefore possible to pass to the limit k = 0 and still retain the same flow figure.

As shown by closer consideration, the necessary condition for the separation of the flow is that there should be a pressure increase along the surface in the direction of the flow. The necessary magnitude of this pressure increase in definite cases can be determined only by the numerical evaluation of the problem which is yet to be undertaken. As a plausible reason for the separation of the flow, it may be stated that, with a pressure increase, the free fluid, its kinetic energy is partially converted into potential energy. The transition layers, however, have lost a large part of their kinetic energy and no longer possess enough energy to penetrate the region of higher pressure. They are therefore deflected laterally.

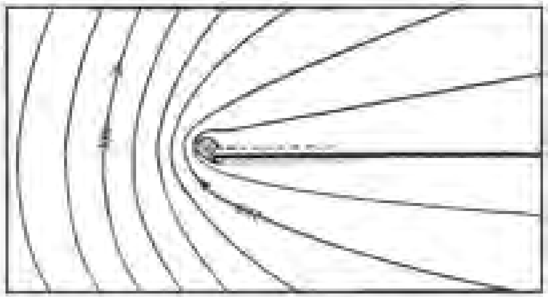


Fig.3

According to the preceding, the treatment of a given flow process is resolved into two components mutually related to one another. On the one hand, we have the free fluid, which can be treated as nonviscous according to the Helmholtz vortex laws, while, on the other hand, we have the transition layers on the solid boundaries, whose motion is determined by the free fluid, but which, in their turn, impart their characteristic impress to the free flow by the emission of turbulent layers.

I have attempted, in a few cases, to illustrate the process more clearly by diagrams of the streamlines, though no claim is made to quantitative accuracy. In so far as the flow is vortex-free, one can, in drawing, take advantage of the circumstance, that the streamlines form a quadratic system of curves with the lines of constant potential.

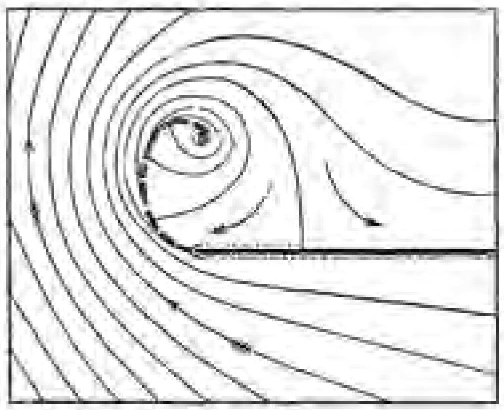


Fig.4

Figures 3-4 show, in two stages, the beginning of the flow around a wall projecting into the current. The vortex-free initial flow is rapidly transformed by a spiral separating layer. The vortex continually advances, leaving still water behind the finally stationary separating layer.

Figures 5-6 illustrate the analogous process with a cylinder. The fluid layers set in rotation by the friction are plainly indicated. Here also the separating layers extend into infinity. All these separating layers are labile. If a slight sinoidal disturbance is present, motions develop as shown in Figures 7-8. It is clearly seen how separate vortices are developed by the mutual interference of the flows. The vortex layer is rolled up inside these vortices, as shown in Figure 9. The lines of this figure are not streamlines, but such as were obtained by using a colored liquid.

I will now briefly describe experiments which I undertook for comparison with the theory. The experimental apparatus (Fig. 10) consists of a tank 1.5m (nearly 5 feet) long with an intermediate bottom. The water is set in motion by a paddle wheel and, after passing through the deflecting apparatus a and four sieves b, enters

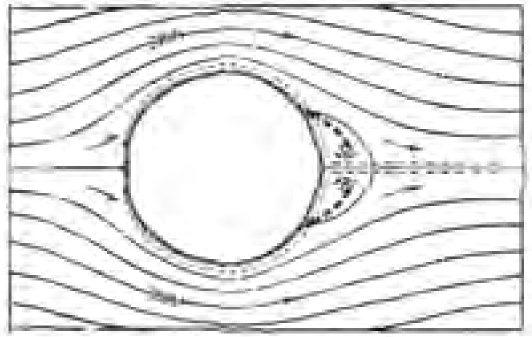


Fig.5

the upper channel comparatively free from vortices, the object to be tested being introduced at c. Fine scales of micaceous iron ore are suspended in the water. These scales indicate the nature of the flow, especially as regards the vortices, by the peculiarities of their reflection due to their orientation.

The accompanying photographs were obtained in this manner, the flow being from left to right. Nos. 1-4 show the flow past a wall projecting into the current. The separating or boundary layer, which passes off from the edge, is apparent. In No. 1 it is very small; in No. 2, concealed by strong disturbances; in No. 3, the vortex spreads over the whole picture; in No. 4, the permanent condition is shown. A disturbance is also evident above the wall. Since a higher pressure prevails in the corner, due to the obstruction of the water flow, even here the flow separates from the wall after awhile (Cf. Figs. 1-4). The various striae visible in the vortex-free portion of the flow (especially in Nos. 1-2) are due to the fact that, at the inception of the flow, the liquid was not entirely quiet. Nos. 5-6 show the flow around a curved obstacle or, from another viewpoint, through a continuously narrowing and then widening channel. No. 5 was taken shortly after the inception of the flow. One boundary layer has developed into a spiral, while the other has elongated and broken up into very regular vortices. On the convex side, near the right end, the beginning of the separation can be seen. No. 6 shows the permanent condition in which the flow begins to separate about at the narrowest cross section.

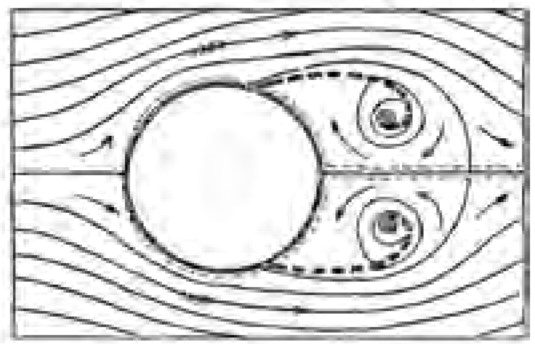


Fig.6

Nos. 7-10 show the flow around a cylindrical obstacle. No. 7 shows the beginning of the separation; Nos. 8-9, subsequent stages. Between the two vortices there is a line of water which belonged to the transition layer before the beginning of the separation. No. 10 shows the permanent condition. The wake of turbulent water behind the cylinder swings back and forth, whence the momentary unsymmetrical appearance. The cylinder has a slot along one of its generatrices. If this is placed

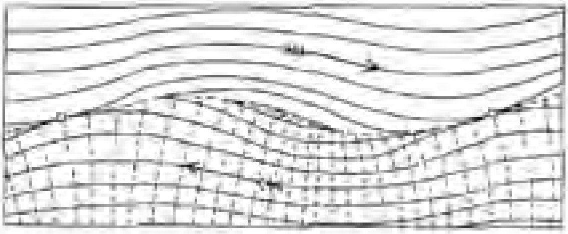


Fig.7

as shown in Nos. 11-12 and water is drawn out through a tube, the transition layer on one side can be intercepted. When this is missing, its effect, the separation, is eliminated. In No. 11, which corresponds, in point of time, to No. 9, there is seen only one vortex and the line. In No. 12 (permanent condition), the flow closely follows the surface of the cylinder till it reaches the slot, although only very little water enters the cylinder. A turbulent layer has developed instead on the flat wall of the tank (its first indication having appeared in No. 11). Since the velocity must diminish in the widening cross section and the pressure consequently increases ($\frac{1}{2}\rho v^2 + V + p =$ constant on every streamline), we have the conditions for the separation of the flow from the wall, so that even this striking phenomenon is explained by the theory presented.

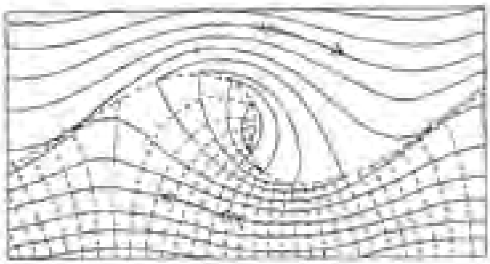


Fig.8

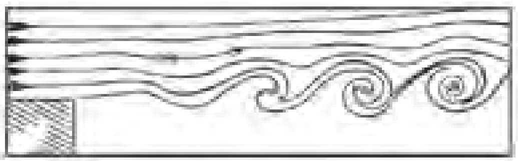


Fig.9

Translation by Dwight M. Miner, National Advisory Committee for Aeronautics.

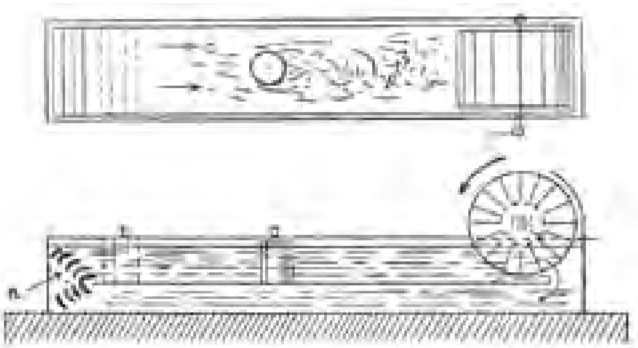


Fig.10

Document 3-3(a-b)

(a) Alexander Klemin, Chapter X: "Resistance of Various Airplane Parts," in *Aeronautical Engineering and Airplane Design* (New York: Gardner-Moffat, 1918), pp. 61-4.

(b) Grover C. Loening, *Military Aeroplanes: An Explanatory Consideration of their Characteristics, Performances, Construction, Maintenance, and Operation* (Boston: W.S. Best, 1918).

This pair of documents reproduces excerpts from two of the earliest aeronautical textbooks to appear in the United States, both in 1918. The author of the first text, Alexander Klemin, was a graduate of Massachusetts Institute of Technology's (MIT) aeronautical engineering program (he actually succeeded Jerome Hunsaker as the instructor in 1913) and the head of the U.S. Army Air Service's Aeronautical Research Department at McCook Field in Dayton, Ohio. The author of the second, Grover Loening (1888–1976), was an early aeronautical engineer and aviation manufacturer in the United States. He received an M.A. in aeronautics from Columbia in 1910, worked for Orville Wright, and served as chief aeronautical engineer of the Aviation Section of the Army Signal Corps at San Diego in 1914. Loening's book first appeared in 1915, but it was not fully recognized until three years later when he had the benefit of evaluating the rapid pace of technological development that took place during World War I. In 1917, he formed his own company, the Loening Aeronautical Engineering Corporation, and designed a strut-braced monoplane for the army and experimented with amphibious aircraft. Loening won the 1921 Collier for his Air Yacht, a five-seat monoplane boat, which evolved into the Loening Amphibian, said to be the first practical airplane of that type.

As might be expected, Klemin's text was the more theoretical of the two, indicative of his advanced education in aeronautics at MIT. Loening, on the other hand, wanted a book that would more practically benefit "aviators and students" and that featured the lessons and experiences of military aircraft design, manufacturing, and operation. In important ways, the two texts represent what in the history of American engineering education would come to be known as the "school culture" (Klemin) versus the "shop culture" (Loening). Shop culture, generally speaking, promoted a technical approach that could be applied directly to industrial work. In contrast, school culture placed more emphasis on basic studies in mathematics and science and on original research. To this day, some degree or another of tension between shop culture and school culture exists within most engineering communities and has shaped the personality of American engineering schools and professional organizations. Both approaches bear their type of fruit, although without question, the school culture approach has provided a much more solid basis for long-term fundamental development in aeronautics and most other highly technical fields.

One aspect that the two textbooks had in common was that they both reflected a state-of-the-art that had been largely driven by European developments in the years leading up to 1918. Much of the information provided in the texts was based on knowledge and experience gained during the war in Great Britain at its National Physical Laboratory in the 1910s, in France at Gustav Eiffel's laboratory, or gleaned from Germany's many aeronautical advances.

Document 3-3(a), Alexander Klemin, Chapter X: "Resistance of Various Airplane Parts".

Chapter X

RESISTANCE OF VARIOUS AIRPLANE PARTS

One of the most difficult problems in aeronautical design is the prediction of the total resistance of the machine. The wind tunnel test is a good check, but it is most important to assign resistance values to various parts and to tabulate them prior even to the construction of the model. In this chapter have been collected as far as possible all the data available for bodies, radiators, fittings, wheels, cables and wires and certain other miscellaneous objects.

AIRPLANE BODIES FROM THE AERODYNAMICAL POINT OF VIEW

If airplane bodies were designed from a purely aerodynamical point of view, they would follow dirigible practice and be of streamline form. There are, however, a number of structural requirements which have to be met, which preclude the employment of such forms. The body must enclose the power plant and the personnel, the length must be long enough to place the rudders well clear of the wash of the planes, the shape of the body must conform to structural requirements such as the use of four longitudinal girders, or a triangular form which has been found to be advantageous in steel construction.

No wind tunnel tests on bodies alone can determine exactly their resistance on an airplane, because the question is complicated by the position and form of the motor and the disposition of the tail surfaces. The propeller in a tractor machine also introduces three possible variations in drag coefficients: (1) when the propeller is pulling and there is a slip stream of velocity greater than the airplane velocity, (2) the resistance on a glide when the engine is shut down, but the propeller is revolving as an air motor, (3) when the propeller is not, revolving at all, the engine being held.

TRACTOR BODIES

In Table 1 is given a comparative table of resistance coefficients for area in normal presentation of a number of airplane bodies, and in Fig. 1 are shown sketches of the same bodies. Exact comparisons are impossible because some of the bodies are made for two men and others for one. Still quantitative conclusions can be drawn. The N. P. L. Model 5, more symmetrical than the B. E. 8, shows a distinct improvement over the latter which is somewhat discounted by the fact that the B. E. 8 carries two men unshielded. The B. F. 36, all almost perfect dirigible form, is markedly better than either of these two bodies.

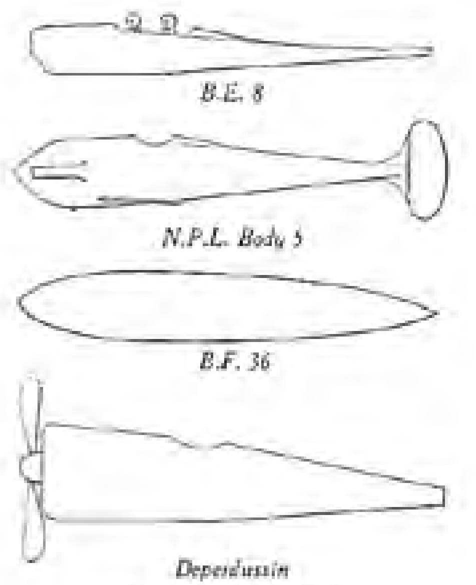


Fig. 1. Tractor Bodies.

The resistance of the body in an airplane is apparently a small quantity, but the figures given below do not represent the resistance of a body in full flight where it is increased by 40 percent, the propeller slip stream increasing the relative speed of the air by some 25 percent. Also, it must be remembered that with a best glide of 1 in 8, a 5-pound increase in resistance is practically equivalent to an added weight of 40 pounds. A blunt, square form of body such as is often seen in American practice may increase resistance even more, and better aerodynamical design of bodies seems a feature worth considering.

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COMPARATIVE TABLE FOR TRACTOR BODIES

PUSHER BODIES

A pusher body such as the Farman 3, illustrated in Fig. 2, gives a not much larger resistance than the tractor bodies, but when the head resistance of the uncovered outriggers is taken into account, it will probably be found that, pusher arrangements offer considerably more resistance than tractor bodies.

RADIATOR RESISTANCE

The only values available for this are the results of some tests at the Massachusetts Institute of Technology. These were carried out on portions of a radiator of the honeycomb type having sixteen 1/4-inch cells to each square inch of the surface normal to the wind. The tests were repeated on two sizes of radiator section, one 0.25 square feet and the other 0.111 square feet, and at various speeds. No important variation in the resistance coefficient was apparent and the average coefficient

TABLE 1.

COMPARATIVE TABLE FOR TRACTOR BODIES.

Designation.	Coefficient of resistance, K where $R = KAV^2$ (A = area in normal presentation in square feet V = miles per hour; R = drag in lbs.)	Length / maximum depth.	Resistance for a body of 5 square feet normal presentation at a speed of 60 m. p. h.
British R. E. 3 (with 2 men)	.000720	7.35	29.7
N. P. L. Model 5	.000420 (approx.)	5.50	12.6
British B. E. 9 (dirigible form)	.000235	5.75	7.4
Deperdussin (enclosing rotary motor)	.001215	5.6	35.1

TABLE 2.

COMPARATIVE TABLE FOR PUSHER BODIES.

Designation.	Coefficient of resistance K where $R = KAV^2$ (A = maximum area in normal presentation in square feet; V = miles per hour; R = drag in lbs.)	Length	Resistance for a body of 5 square feet normal presentation, at a speed of 60 m. p. h.
N. P. L. Model Body 3 (fairly symmetrical section)	.000271	2	7.8
Farman 3 (body in form of a boat, two men unshielded)	.000545	3.2	21.1

may be used for practical calculations. This has a value K_x = .000814 pounds per square foot of projected area per foot per second or .00173 pounds per square foot of projected area per mile per hour.

RESISTANCE OF FITTINGS

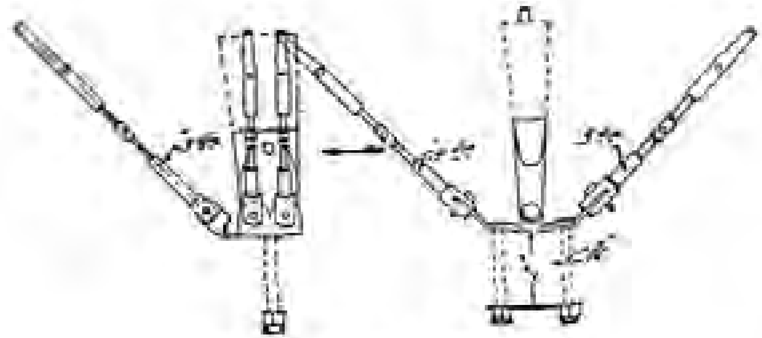
Fittings are so variable in design that it is impossible to give definite figures to meet every type of wing strut fitting. Tests were conducted at the Massachusetts Institute of Technology on the fittings of which dimension drawings are given in Fig. 3; the coefficients of resistance are $R = .00030\ V^2$ and $R = .00040\ V^2$ for the two types which at 60 miles an hour gives 1.07 and 1.44 respectively. Such figures will be at least approximately correct in design.

RESISTANCE OF AIRPLANE WHEELS

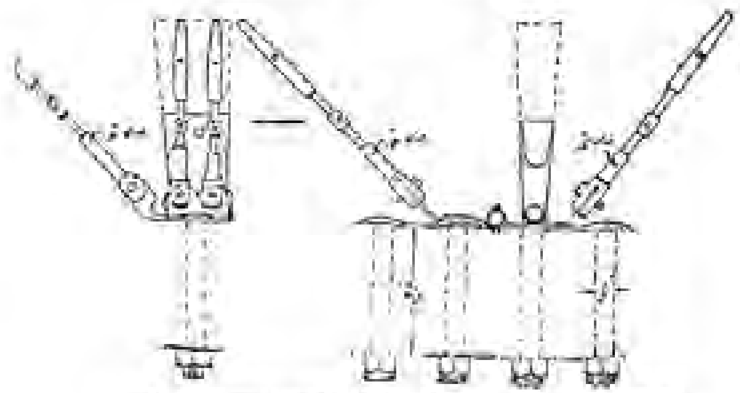
For a standard airplane wheel of about 26 X 4 inches in size, the drag found by the N. P. L. is about 1.7 pounds at 60 miles per hour. This again is sufficiently accurate for practical purposes. Eiffel has experimented with a number of wheels and has shown that no great variation need be expected front the above value. An important result from

FIG. 2. A PUSHER BODY

the French experiments was the fact that an uncovered wheel had a resistance of 50 percent more than a covered wheel of similar dimensions. This justifies the standard practice of covering the wheel in.



Outer Panel $R = .00030\ V^2$.



Inner Panel with Wing Hinge. $R = .00040\ V^2$.

FIG. 3. FITTINGS EMPLOYED IN TESTS FOR HEAD RESISTANCE AT MASSACHUSETTS INSTITUTE OF TECHNOLOGY

CLARK STRUT FITTINGS. Resistance includes fitting, five turnbuckles and bolts but not dotted portions as indicated on drawings. Resistance in pounds; Velocity in miles per hour.

RESISTANCE OF WIRES AND METHODS OF PLOTTING

A certain complication is necessary in the methods of plotting the results for the resistance of cables and wires. As we have seen from the diagram of Fig. 19, in Chapter 3, of the Course, the resistance of a wire or any cylindrical body is partly due to turbulence, partly due to skin friction. It cannot therefore be represented by such a simple expression as

$$R = KLD\,V^2\,F\,(VD).$$

We do not know what function *F (VD)* is exactly, nor how it varies with size and scale except from experimental results, and comparisons of resistance varying as $LD\,V^2$ can only be made between two cables if *VD* is a constant. If *K* is taken as a function of *VD*, the *R* may be written $R = KLD\,V^2$ but then *K* must be plotted against *VD* in analyzing experimental results. This is the only rational and scientific method.

An empirical method, however, is sometimes employed with fair accuracy of plotting the resistance of a wire whose length is equal to its diameter against V^2D^2. This has the advantage that the graph approximates very closely to a straight line, the slope of which is equal to *K*, thus giving an easy means of determining a mean value of *K*.

RESISTANCE OF STATIONARY SMOOTH WIRES

The most accurate researches have been carried out at the N. P. L. and their results are, shown in Fig. 4 plotted against *VD*. In the expression $K = \frac{R}{LDV^2}$, *R* is in pounds, *L* in feet, *D* in feet, and *V* in miles per hour. But in the abscissae, values of *VD*, *V'* is in feet per second, and *D* is feet, so as to give the correct scale and speed relationships which must be in the same units.

The accuracy of the curve at its lowest portion is doubtful, since the flow is apparently just changing its nature at that point, and {AQ3} successive observations under the same conditions may give quite different results. On modern machines of fairly high speed, however, the valves of *VD* nearly always exceed 0.35 and consequently do not lie on this section of the curve.

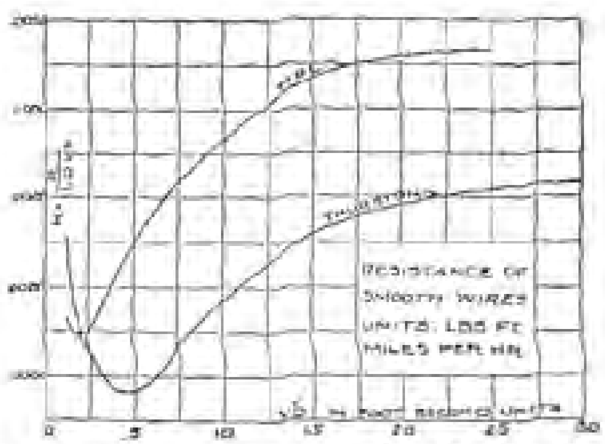


Fig. 4. Resistance of Smooth Wires Per Foot Run.

Similar tests were made by Mr. Thurston and M. Eiffel, and the values obtained by the former are plotted in the same figure.Thurston's experiments, however, were very much earlier, and Eiffel's covered a less range and were performed with less sensitive apparatus, so it is advisable to use the N. P. L. results.

RESISTANCE OF VIBRATING WIRES

When the question of resistance first began to arouse interest, it was popularly supposed that a vibrating wire had much greater resistance than a stationary one. This, however, is not the case. Research on this point at the N.P.L. failed to disclose any difference whatever, although the balance would have shown deviations as small as 3 percent, even for the extremely small forces under consideration. Mr. Thurston, on the other hand, concluded that vibration at the rate of 15 per second increased the resistance by about 5 percent for small wires and by a somewhat smaller percentage for those of larger diameter. In any case, the effect is unimportant.

RESISTANCE OF STRANDED WIRES

The air resistance of stranded wires was also investigated at the N.P.L., and was found to be about 20 percent greater than that for a smooth wire of the same diameter. This is only approximate, as the coefficient depends on the number of strands, type of lay, etc. It is also impossible to plot the values of *K* against *VD* for wire rope, as the *VD* law holds good only for objects which possess strict geometrical similarity, a thing which stranded wires of different sizes never do.

RESISTANCE OF WIRES PLACED BEHIND ONE ANOTHER

The manner in which resistance is affected by the close juxtaposition of two wires, one behind the other, is a point of great interest. Here, too, it is at present necessary to rely on Mr. Thurston, although we hope to be able soon to present the results of some more extensive and accurate tests on this matter.

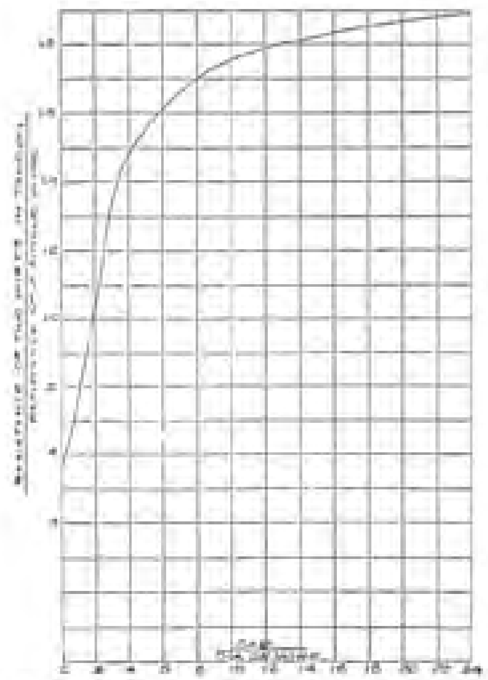


Fig. 5. Resistance of Wires in Tandem as a Ratio of the Resistance of a Single Wire.

Fig. 5 gives, in terms of the resistance of a single bar, the resistance of two bars or wires separated by various distances. It will be seen that two wires placed one behind the other and spaced from 5 to 9 diameters apart, as is usual in double-wiring a biplane cellule, have from 60 percent to 75 percent more resistance than a single wire. The force is, however, materially less than for the two wires placed side by side.

Eiffel has experimented on the resistance of inclined wires. As would be expected the resistance of a wire progressively decreases as its angle with the wind diminishes. Table 3 gives correcting values. This table has been omitted by the editor.

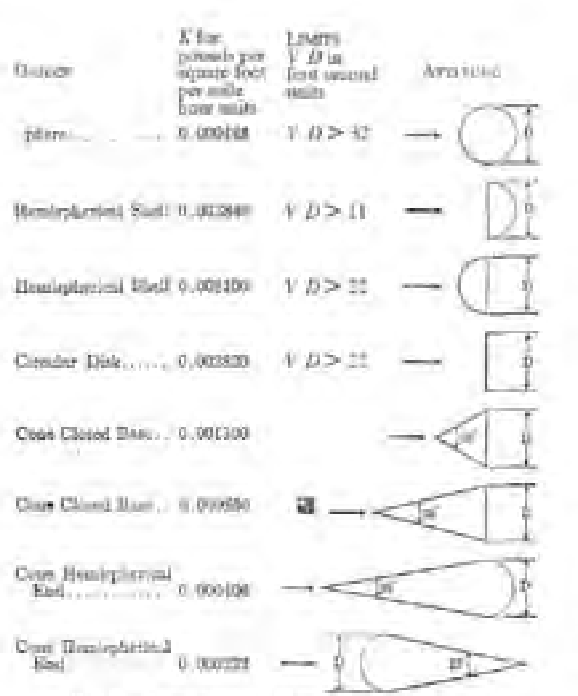


FIG. 6. RESISTANCE OF MISCELLANEOUS OBJECTS (AFTER EIFFEL)

SUGGESTIONS FOR STREAM-LINING WIRES

It has been suggested from time to time that wire resistance should be decreased by "stream-lining" or adding a triangular portion in back of the wire. From experiments by Ogilivie, however, it appears that a section made up of a semi-circle a triangle has a decidedly high resistance, and the gain from such a procedure wound be small.

Wires placed behind one another have also been covered in. The British Royal Aircraft Factory produces a very heavy R.A.F. wire in use on big machines which is stream-line in form. But the direction in which progress manifests itself at present is in the elimination of wires by certain modern trussing such as used in the recent Curtiss biplane.

RESISTANCE OF MISCELLANEOUS OBJECTS

This resistance of certain miscellaneous objects as deduced by Eiffel may sometimes be useful. The values for such objects within certain limits are illustrated in Fig. 6.

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British Report 1912-1913, page 116.
"La Resistance de l'Air et l'Aviation," Eiffel, 1914, page 250.

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"La Resistance de l'Air et l'Aviation," Eiffel, page 97.
"New Mechanical Engineers' Handbook," Section on Aeronautics, by J. C. Hunsaker.

Document 3-3(b), Grover C. Loening, **Military Aeroplanes: An Explanatory Consideration of their Characteristics, Performances, Construction, Maintenance, and Operation** *(Boston: W.S. Best, 1918).*

CHARACTERISTICS OF AIRFLOW.

Having defined air, the manner in which it flows may be considered. Air either flows smoothly past an object in **streamlines**—continuous filaments—or it breaks up into swirls and eddies, due to too abrupt a change in flow. The accompanying photographs of airflow illustrate this.

It is apparent that a spindle or fusiform shape, gently dividing the air at the front, and gradually permitting the filaments to close together at the rear, will give a smooth flow, which amounts to the same thing as a very low resistance. It is also evident that a flat surface creates very great disturbance, and consequently high resistance.

The curve of the streamlines, necessary to prevent disrupting them, may be computed for any speed, by applying fluid dynamics. But it must be kept in mind that a form of this kind gives its low resistance, only at one particular speed, since the path of flow is affected by the speed. It is unnecessary here to take up the determinations of these forms. If the streamlines flow smoothly past an object, and close up again without eddies, it follows that the only resistance experienced is frictional. There is hardly any shape, however, which does not create small eddy resistance.

Methods of measuring the resistance of the air that have been widely used, are the following:

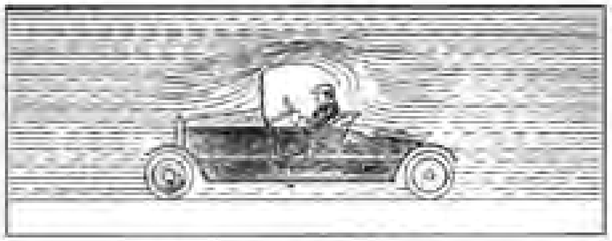
1. Dropping surfaces from a height and measuring time of drop and pressure, used by Newton, and Eiffel in his earliest experiments.

2. The whirling arm, used by Langley, and consisting of whirling the surface at the end of a large arm around a circle of large diameter and recording the resistance automatically.

3. The moving carriage, an automobile, trolley or car, as used in the experiments of the Duc de Guiche, Canovetti, and the Zossen Electric Railway tests.

4. By blowing or drawing air through a tunnel in which the object or a model of the object is placed. This method is the most modern and convenient, and permits of a uniformity of the air current, which cannot be obtained as easily in the open.

In wind tunnels the best practice is to draw the air in, through screens and channels that straighten it out, past the experimental chamber, and thence to the fan. Practically all the great Aerodynamic Laboratories use the wind tunnel method of experiment. The prominent ones are: the Eiffel laboratory in Paris, the National Physical Laboratory in England, and the tunnel at the Washington Navy Yard. The speed of the wind in the Eiffel laboratory can be brought up to almost 90 miles per hour (40 meters per second), and its size permits of testing many objects, such as struts, to full size and complete models of aeroplanes to one-tenth full size. Such a magnitude permits of exceedingly valuable determinations, and the work of the laboratories is daily being applied with entire success to full-sized aeroplanes, although the higher speeds of aeroplanes require considerable correction of wind tunnel results. This is particularly true in the measurement of pressures on wing sections at low angles.



THE AIR FLOW OVER A WINDSHIELD VISUALIZED

It must be borne in mind, therefore, that the air in a tunnel is confined and that all tunnel results are not perfectly adaptable to machines unless suitable corrections are applied.

Combining the results of all the laboratories, we may draw some general conclusions with regard to air resistance, as follows:

1. The resistance of an object in an airstream is proportional to the square of the velocity of the air.

In other words, if the velocity is doubled, it follows that the resistance will be increased four times, or if velocity is five times as great, the force on the same object would be twenty-five times as great. **This is merely an experimental fact.**

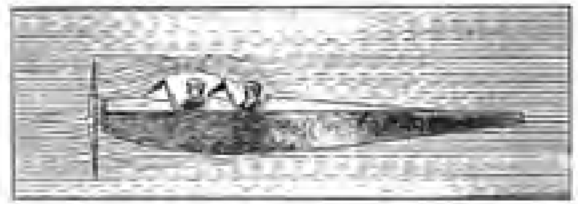
There are many ways of determining the manner in which the air flows past an object, such as noting the direction in which light silk threads are blown, or introducing smoke or particles into the air and photographing it. Ammonium Chloride is a very convenient smoke.

IMPORTANCE OF VISUALIZING THE AIR.

It is of great value in aeroplane work, to become accustomed to visualize the streamline flow of air, and ability to "see the air" often solves many problems of stability and reduction in resistance, without any recourse to mathematics or measurements. Besides this, there is offered in the study of airflow by photography, a field of investigation of great promise and absorbing interest.

It is a common experience that in a wind, at the front of a flat surface, there is a dead region of air, where no wind is felt. Photographs show this air cushion clearly.

In stability discussions, effect of following planes, interference and propeller stream action, priceless secrets would be revealed if the air could be followed in its every movement.



The air flow over the front wind shield throws the air into the rear upon a face – a feature that would have been corrected if the designer would have "seen the air."

DETERMINATION OF AIR RESISTANCE.

The nature of the action of air on objects has been considered, but we must know in addition with what force in pounds **P**, the air pushes on an object when it passes it at velocity **V**. We cannot refer to theory for this, satisfactorily, so we must obtain actual measurements of the air resistances on various objects.

2. Air Resistance Increases as the Object's Size Increases.

This experimental fact is also subject to modification, since, as the size of surface increases, the pressures are somewhat greater in proportion. But we can disregard this also without serious error.

3. The Air Density Influences the Air Resistance.

It has already been pointed out that heavy air (low altitude) has more resistance than light air (high altitude).

4. The Shape of an Object Controls its Air Resistance.

The beautiful streamline photographs have already discovered this for us, and show how easily and with what small resistance the air slides by a streamline shape.

Let us combine all this into a compact sentence called a formula, where **P** = the resistance in lbs., **S** = the area in sq. ft., **V** = the velocity through the air in miles per hour, **d** = the density of the air in lbs. per cubic foot, and finally describe the shape of the object in order to ascertain whether it is clumsy or streamline, by a numerical multiplier, which we will call **k**, and which we will define as a "**shape coefficient.**"

In others words, $\mathbf{P = k.\ d.}\ S\,V^2$.

But to simplify matters, since all of these shape coefficients for various shapes have had to be measured, and mostly at sea level, we can call **d** also a numerical factor, and combine it with **k**, so that **kd=K**, which is a number, always applicable to that particular shape, and represents **the coefficient at sea level**, in which we can use, for any size body of that shape, at any speed, in order to obtain **the resistance at sea level**. For many different shapes, all we need, therefore, is for someone to measure and tell us what the values of K are for different shapes. Just as a grocer will tell you that a piece of cheese weighs half a pound because he measures it, so has M. Eiffel told the aviation world that **K** for a wire is .0026.

Therefore $P = KSV^2$

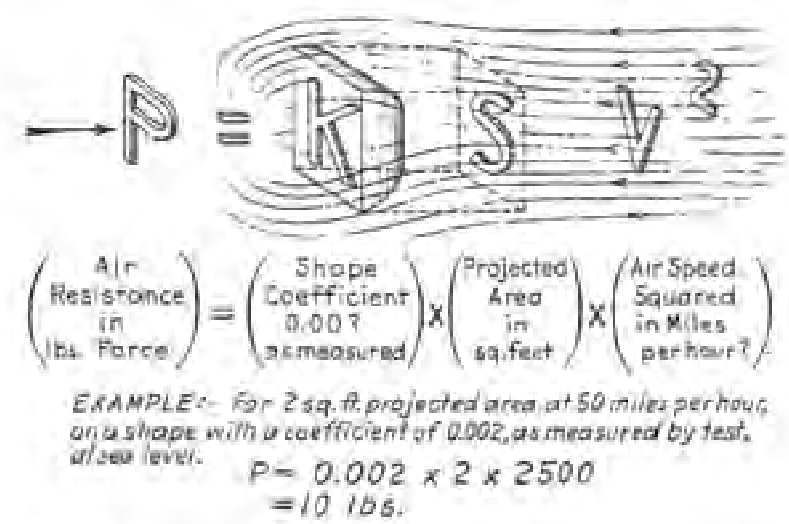
This formula applies to all air forces, whether they are resistances, as we are considering here, or lifting pressures as we will consider later—with this reservation, that the shape coefficient is either for resistance, or for lifting power, and one must always be careful to determine which it was meant to be measured for.

We may now proceed with the interesting study of what values of **K** have been found for various shapes, and not only will those be described by diagrams, but frequent numerical examples are given, of how to apply the data to determine values of air resistances. A study of this subject will give the student a most valuable insight into this branch of the science, and an appreciation of what shapes are "good" and "bad."

DEFINITIONS.

In Aerodynamical studies it has become customary in defining objects to use unfamiliar terms.

Aspect Ratio — is a term used to define the shape of a surface, and is the long span of the surface across the wind divided by the width.



Here is a picture of the fundamental and only necessary formula for practical aeroplane knowledge.

Fineness Ratio — is a term used to define the general shape of bodies, and is obtained by dividing the fore and aft length of the body by the greatest width across the wind.

Master Diameter — is the greatest width of a body across the wind.

Fairing — is used to denote the additional "tail" or filler used to make a poorly shaped body more streamline in form, thereby reducing its resistance.

Diametral plane —is the plane, passed through a body, facing the wind perpendicularly, and cutting through at the master-diameter.

Normal plane — is another expression for diametral plane, and merely refers to the maximum cross-sectional projection of the body. It also refers to a flat surface held normal (perpendicular) to the air current.

Equivalent Normal Plane — is the size of normal flat surface, that would give the same resistance as does the body referred to.

Flat Surfaces,
Normal to the Airstream.

SQUARE PLANES:

In square planes, normal to the air, the value of **K** is **.003** for surfaces up to two or three feet square, and **.0033** for very large surfaces like the sides of buildings.

It may be stated, therefore, for aeroplane usage, that **P**, the air resistance in lbs., of a square surface, **S** sq. ft., in area, at a velocity **V** miles per hour, is

$$P = .003\ S\ V^2$$

Thus, for a surface 2 feet square, at 70 miles an hour:

$$P = .003 \times 4 \times 4900$$
$$P = 58.8 \text{ pounds}$$

RECTANGLES:

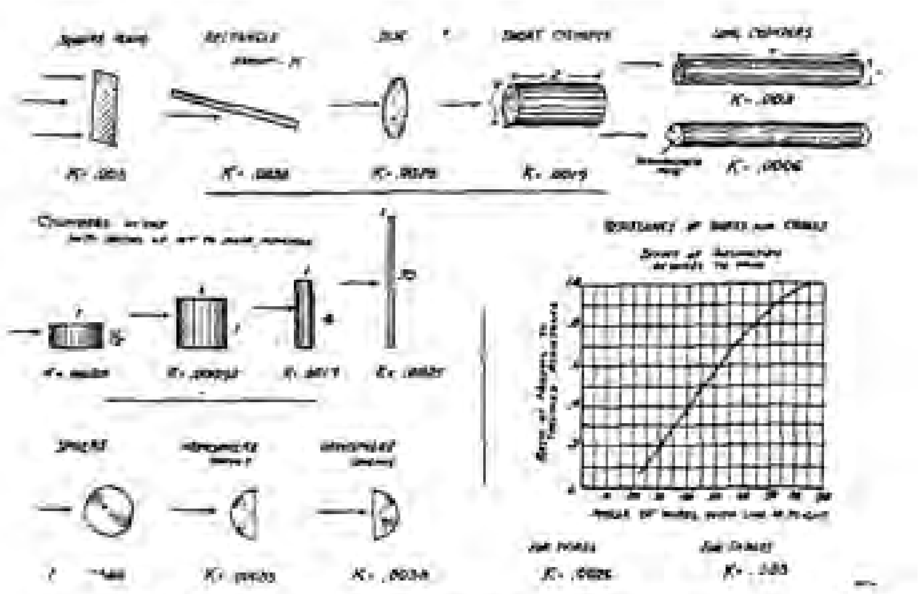
This aspect ratio of a square is one. Rectangles have aspect ratios above one, when presented normally to the air.

Up to an aspect of 5 or 6, **K** remains about **.003**.

An increase in the value of **K** is found for rectangles as the aspect ratio increases.

When the aspect ratio of the rectangles increases to 15, **K** becomes **.0035** and on further increasing the aspect ratio to 30, **K = .0038**.

A flat rectangle, perpendicular to the air current, with its dimension across the current, thirty times as large as its width, might be met with in rods, temporary struts, etc., and it is interesting to note how high the resistance would be.



The shape coefficients of various surfaces and bodies.

DISCS:

The shape of flat surfaces also affects their air resistance. Passing from a square plane to a round disc, reduces **K** to **.0028**, so that the air resistance of a disc 2 feet in diameter, at 60 miles per hour, is

$$P = KSV^2 = .0028 \times .7854 \times 4 \times 3600$$
$$P = 32 \text{ pounds}$$

In general rounded edges may be expected to reduce **K**, for flat surfaces.

Discs or flat rectangles, placed one in front of the other, interfere with each other and exhibit a most important phenomenon, shown on page 52.

CYLINDERS:

Passing from the disc to the cylinder, with the circular base facing the wind, the resistance is found to be less as the length of cylinder is increased, until the length becomes greater than 5 diameters, when the resistance is found to increase again. Some values of **K** are given on the chart on page 48.

Wires and cables are merely long cylinders. Extensive experiments have been conducted on them, and values of **K** found. For smooth wires **K** = **.0026**, whereas cables are found to have considerably higher resistance with **K** = **.003**.

Thus, a machine having 200 feet of 1/8 inch cable, giving a projected area of 200/96 = 2.08 sq. ft., will have an air resistance due to the cables at 80 miles an hour of

$$P = .003 \times 2.08 \times 6400$$
$$= 40 \text{ lbs.}$$

This high value immediately suggests the advisability of reduction of cable resistances. In double cables, it would prove beneficial to tape them together, so as to streamline each other. A graph is given showing the reduction in resistance due to inclining the wires i.e., staggered planes, on page 48.

SPHERES:

The resistance of the air on spheres presents a study of interest. The sphere is the simplest geometrical form, and, as a basic one, it should long ago have served as the unit form for air resistance. Lack of agreement in the experimental results of different laboratories was only cleared up when Eiffel discovered that an increase of speed of the air above 20 miles per hour caused a change of flow, due to the flattening out of vortices back of the sphere, which reduced the resistance considerably. And that above this speed, the nature of the air resistance remained constant. **K** = **.00044**, for a sphere, at speeds above 20 miles per hour, whereas at very low speeds **K** becomes **.001**. In having a smoother flow at the higher speeds, less lbs. of air are put in motion, which means that the resistance is less. This action of air, in tending to smoother flow with speed increase, is important to bear in mind.

STREAMLINE SHAPES:

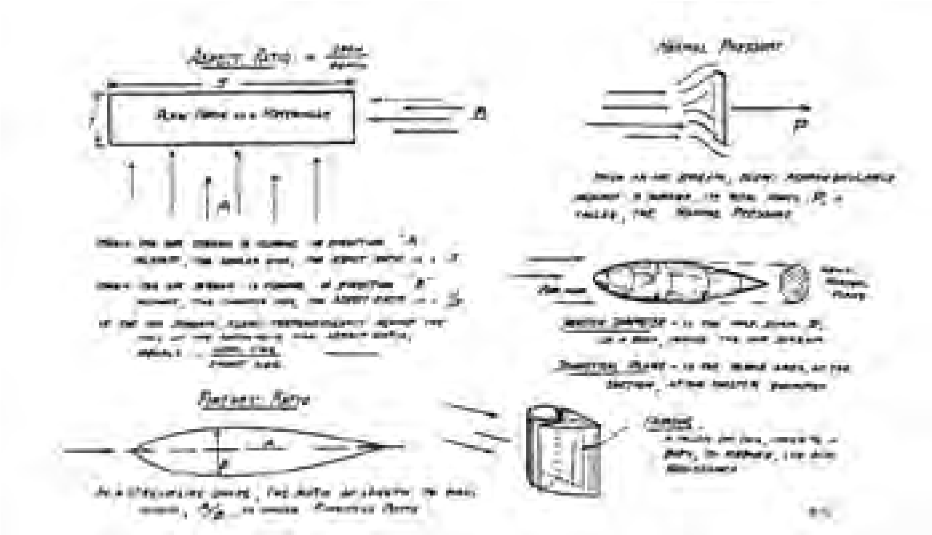
In this class may be included bodies of fusi-form or streamline form, shaped for least resistance. Their application to the design of tanks, fuselages, nacelles, hoods, etc., is of fundamental importance.

In a most interesting set of experiments, conducted by M. Eiffel, on streamline shapes, illustrated in the diagrams and chart on p. 52, the bodies consist of a nose, a cylindrical central portion, and a tail.

The results of the experiments show that:

1. The blunter the nose, the greater the resistance.
2. The shorter the central cylindrical portion is, for the same nose and tail, the lower the resistance.
3. The effect of shortening up the tail is not very great, although slightly increasing the resistance.

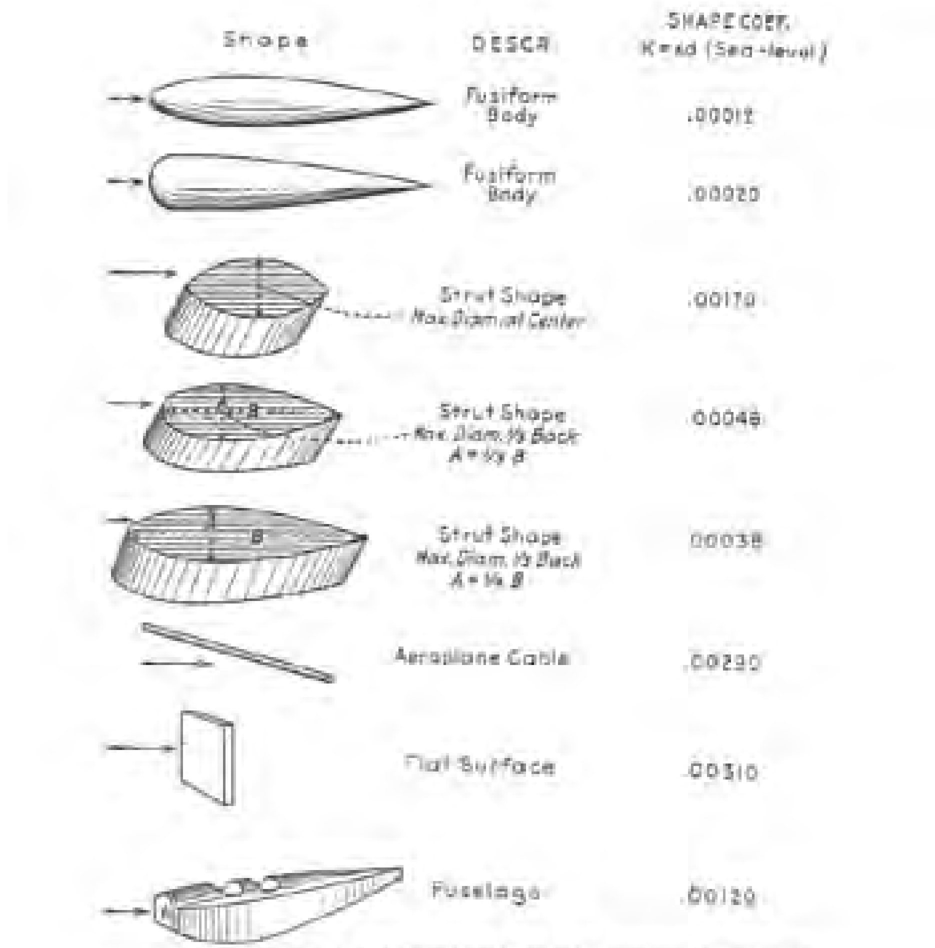
In each case, however, measurements made at speeds up to 90 miles an hour



showed that the resistance does not vary as V^2, the value of **K** becoming constantly less with speed increase. This is a very significant determination, and may be explained on the ground that, in bodies of this kind, the major part of the resistance at high speeds is frictional and therefore increases at much less than V^2. In addition the effect of velocity increase is to flatten out the flow and suppress eddies.

The values of K for these bodies are given.

The Goettingen Laboratory conducted extensive experiments on the best shapes



VALUES OF SHAPE COEFFICIENTS FOR SEVERAL SHAPES

for dirigible balloons which it is important to consider. The models tested measured 3.75 feet long and .62 feet in diameter, giving a fineness ratio of 6. The shapes in their order of least resistance and values of **K** for 25 m.p.h. are given. At higher speeds, still lower **Ks** would be expected.

The form No. 1, having the least resistance, is, perhaps, the best form that has ever been tested in a laboratory, and at high speeds would give a resistance about 1/25th of the normal pressure on its diametral plane. It is the form used in the Parseval non-rigid dirigibles.

It is interesting to note in studying low resistance bodies, how closely they resemble the shapes of fishes, and of birds, measurements of a fast swimming fish showing an almost exact resemblance to this Parseval shape.

As a general rule, the best streamline body is the one having a fineness ratio of 6 and with the master diameter about 40% back of the nose, both nose and tail being fairly well pointed.

STRUTS:

The application of fineness ratios, and shapes of least resistance, to improvement in the form of struts, has in many instances tremendously improved the performance of aeroplanes.

In addition to the form for least resistance, however, the weight of the struts and their strength are factors that must be considered in choosing the best shapes. We will confine ourselves here, however, to a study of the resistance of various shapes.

A group of strut sections are given and **K** for each one. It is to be noted that the effect of yawing is greatly to increase these resistances by presenting the strut sidewise to the air, and it will be necessary later to consider the amount of this increase.

Inclining the strut to the vertical, as in staggered planes, has the effect of increasing the length of section in the airstream, and, consequently, the resistance does not decrease for streamline shapes, while for blunter shapes, inclination reduces the resistance considerably.

In struts, as in bodies, an increase of velocity is accomplished by a reduction in the value of **K**, that is more noticeable the greater the fineness ratio, i.e., the longer the section of the strut. This is again due, probably, to the preponderance of friction in the total resistance.

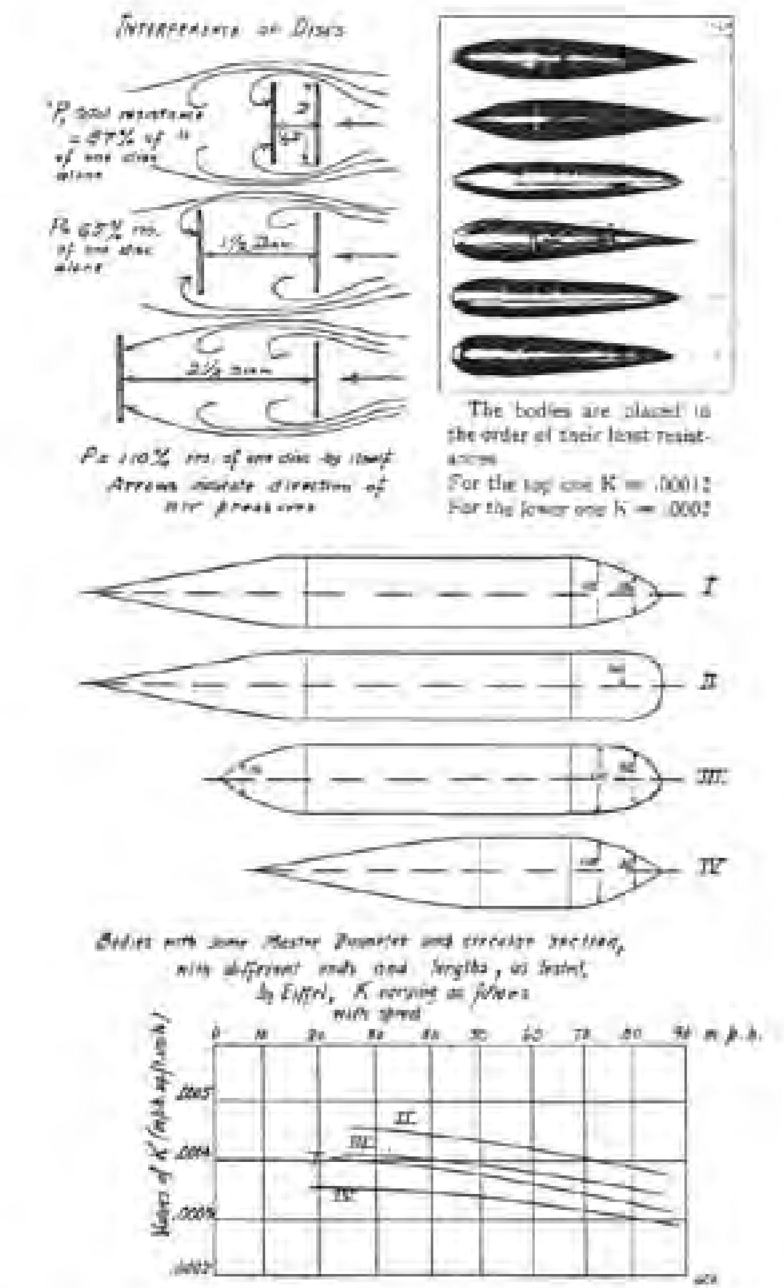
The results obtained in studies of strut resistance indicate the importance of having struts well made and of a uniform section. Just as in bodies, abrupt changes in contour must be avoided and attention paid to a smooth curve on either side of the central portion.

It is found, in general, that a fineness ratio of 5 to 1 is best for use, where a fin effect is desired, and where not,— the best fineness ratio is 3 to 1.

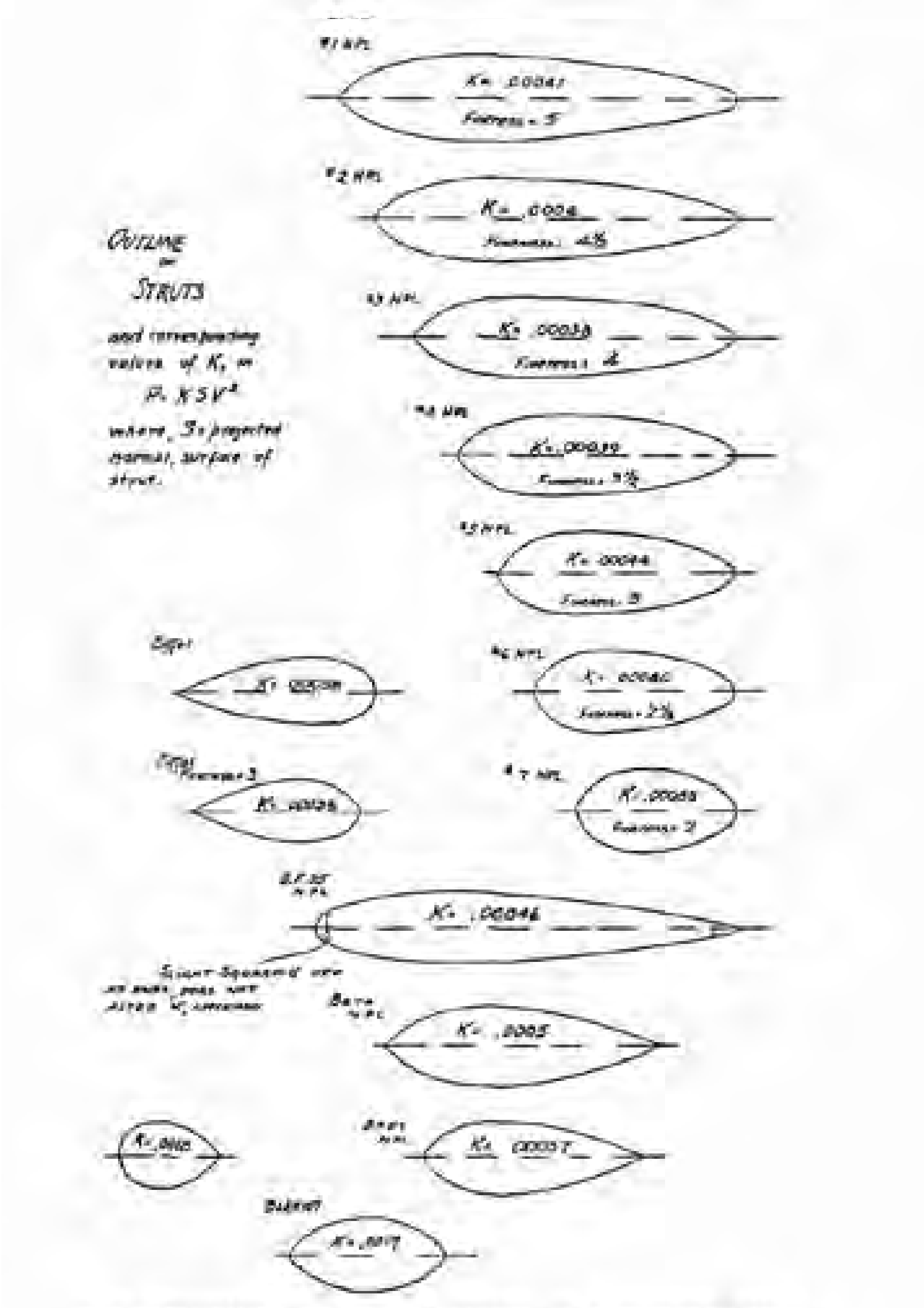
WHEELS:

The air resistance of chassis wheels is a considerable item in flight. Experiments have been conducted on various-sized wheels, and the results are as follows:

28 ½ inches diameter by 2 ½ inches tire, K = .0025
24 ½ inches diameter by 3 ½ inches tire, K = .00265
21 ½ inches diameter by 3 ½ inches tire, K = .0018
18 ½ inches diameter by 2 ½ inches tire, K = .0021



TOP LEFT — INTERFERENCE OF FOLLOWING DISCS — TOP RIGHT, THE BODIES TESTED AT GOETTINGEN — BELOW, BODIES TESTED BY EIFFEL



THE RESISTANCE OF SEVERAL STRUTS OF DIFFERENT SHAPE

When the wheels are covered in, it is found in almost every case that the resistance is halved, so that for the 24 inch X 13 inch wheel, when covered in, K = .00133. An average **K** for wheels would be **.002**.

As an example, it is desired to determine the resistance of two 26 inch X 4 inch wheels at 80 m.p.h.

The projected surface = 1.4 sq. ft.
$\therefore$ P = .002 X 1.4 X 6400
= 18 lbs.

If the wheels were covered in at this high speed, about 9 lbs. would be saved in resistance; this would permit of carrying about 60 lbs. more load on an efficient machine, or would add 10 gallons more fuel.

SUMMARY:

The data given in this chapter enables the air resistance of various shaped bodies to be computed for any speed **V** and any size surface **S**, where **S** in the maximum cross-sectional projection of the body, perpendicular to the airstream. It is merely necessary to supply the numerical values of **K**, **S** (in sq. ft.), and **V** (in m.p.h.), in the formula.

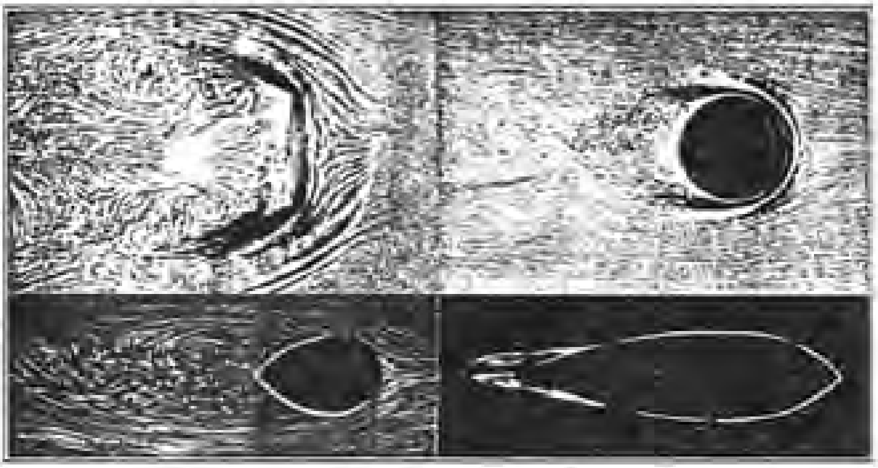
$$\mathbf{P = KSV^2}$$

Where **K = kd, d** being the density of the air, and the values here being correct only for sea level and therefore largely comparative.

It is well again to recall that the propeller of an aeroplane must give a pull or push great enough to overcome:

1. The resistance to motion of the struts, wires, body, wheels, fittings, skids, gas tanks, etc., called Structural Resistance.
2. The dynamic resistance of the wings and rudders, called the **Drift** and generated by the same pressure that gives the **Lift**.

In this chapter the first has been considered. And a study of the second may now be taken up.



THE FLOW OF AIR. UPPER LEFT, A FLAT SURFACE—UPPER RIGHT, A SPHERE—LOWER LEFT AND RIGHT, STRUTS OF DIFFERENT FINENESS RATIO.

Document 3-4

Louis Breguet, "Aerodynamical Efficiency and the Reduction of Air Transport Costs," *Aeronautical Journal* (August 1922): 307-320.

In a speech delivered before the Royal Aeronautical Society in 1922, French aircraft builder Louis Breguet made one of the strongest cases for the social and economic benefits of aerodynamic streamlining. The British *Aeronautical Journal* subsequently published his talk, which is included in its entirety below. The theme of his talk, as indicated in its title, was how improved aerodynamic efficiency could reduce the cost of air transport and thereby breathe life into civil and commercial aviation. To achieve this goal, engineers would need first and foremost to improve an airplane's "fineness ratio," by which he meant its lift-to-drag ratio. Improving this ratio would extend the range of aircraft and make their overall operation more efficient. The way to achieve this improvement was through the use of retractable landing gear and other forms of aerodynamic streamlining. Breguet even set a target for the fineness ratio of commercial aircraft, one that would be hit barely a decade later by the Douglas DC-series aircraft.

The aircraft that Breguet built after 1922 crept closer and closer to such levels of aerodynamic efficiency. The most remarkable of his designs, the Breguet XIX, which was refined from a 1921 military biplane, "enabled France's globe-trotting pilots to start an international craze for nonstop long-distance flying" (Terry Gwynn-Jones, "Farther," in *Milestones of Aviation* [Hugh Lauter Levin Associates, Inc., 1989], p. 46). Two of these flyers, Dieudonné Costes and his navigator Joseph Le Brix, in a Breguet XIX, made the first nonstop crossing of the south Atlantic in October 1927, five months after Lindbergh's historic flight thousands of miles to the north. A year earlier, in 1926, Breguet XIXs broke the nonstop flight record three different times; in the last, pilot Peltier D'Oisy flew a Breguet nearly 6,300 miles from Paris to Peking in just a little over two and one-half days.

Document 3-4, Louis Breguet, "Aerodynamical Efficiency and the Reduction of Air Transport Costs,"* Aeronautical Journal *(August 1922): 307-320.

PROCEEDINGS.
TWELFTH MEETING, 57TH SESSION.

A meeting of the Society was held in the rooms of the Royal Society of Arts, Adelphi, London, on Thursday, April 6th, 1922, the Chairman (Lieut.-Colonel M. O'Gorman) in the chair.

The CHAIRMAN said that the members were to hear M. Louis Breguet, whose name they were too well acquainted with to need any introduction from him. The members ought to know, however, that M. Breguet had been decorated in 1911 and subsequently promoted to the rank of officer of the Legion of Honour, in France, for the services he had rendered during the war, and moreover, that he was the President of what corresponded, in France, to the Society of British Aircraft Constructors, namely, the Chambre Syndicale des Industries Atsronautiques. (Applause.)

The CHAIRMAN then called upon M. Breguet to read his paper on "Aerodynamical Efficiency and the Reduction of Air Transport Costs."

AERODYNAMICAL EFFICIENCY AND THE REDUCTION OF AIR TRANSPORT COSTS.

The possibility for aeroplanes used in air transport services to be made to pay has, so far, been very often questioned, considering the present cost price per ton-mile flown.

I intend to demonstrate that one can, from now, predict an important reduction in air freight rates, not only through greater safety of flight mainly resulting from a longer life of aeroplanes and engines, but also through the betterment of several coefficients which characterise the aerodynamical qualities of aeroplanes.

The study of these coefficients can be made either by laboratory tests on models or by full-scale tests in flight which, if judiciously interpreted, should yield information sufficiently accurate for practical purposes.

May I remind you that the principles of aerodynamics lie in the following formulae:—

$$Rx = (Kx + o/S)\ SV2\ \ldots\ (I)\{$$

Wherein:—

Rz is the total drag of the whole aeroplane along the direction of motion.

Kz is a coefficient relating to R= and dependent upon the value of the angle of incidence in flight.

a- is the projected area, normally to the direction of motion, representing the passive resistances of the aeroplane.

S is the projected area of the wings on a plane parallel to the direction of motion.

V is the speed of the aeroplane along the trajectory.

$$R, = K,S\ V\text{-}\quad (2).$$

wherein :—

R, is the lift of the aeroplane, normal to the direction of motion.

K, is a- coefficient relating to R, and dependent upon the value of the angle of incidence in flight.

$$R= = (K= + (T/S)\ SV2$$

Dividing equation No. (r) by equation No. (2), which gives the ratio of drag to lift, we have:

$$R./R, = (Ks + a\text{-}IS)IK9 = \tan\ (3)$$

That expression tan 0 is usually tailed the "fineness" of the aeroplane, and every aeroplane is characterised by a certain minimum value of tan ¢ which can be represented by tan 0. The smaller tan 0m is the better will be the aerodynamical qualities of the aeroplane.

That fineness is used when calculating the longest distance which a given aeroplane can fly in a "no wind" atmosphere.

$$L = (622\ pl\ m\ \tan ¢m)\ \log PI\ (P - p)\quad (4)$$

wherein :—

L is the said longest distance in kilometres.

p is the efficiency of the propeller.

m is the "fuel and oil" consumption of the engine per horse-power and per hour in kilogs.

P is the total weight in kilogs. of the aeroplane and cargoes at the start. p is the weight in kilogs. of the "fuel and oil" consumed by the engine during the flight and which was included in P.

One at once realises the very great importance of the fineness tan ¢m which in that formula is the only term depending upon the aerodynamical qualities of the aeroplane.

For certain given values of L, p, -in and P, a reduction of tan ¢m causes a reduction in the "fuel and oil" consumption, and consequently increases the weight of paying freight which could be carried.

The conclusion is that one must bring to the minimum the value of tan ¢m. It can be obtained by choosing the best possible profile for the wings, the best designs for the body, empennage, etc. Moreover, the undercarriage should be made to disappear inside the body or the wings when the aeroplane is in flight, etc.

Another interesting coefficient appears when one calculates the power spent in horizontal flight is given by the formula :—

$$W = R_X V = (K + \sigma/S)\ SV^3$$

Where

W is the power in kilogrammetres
R_X is the drag in kilogs as calculated by formula (I)
S is the wing area in square metres
V is the speed in metres per second

On another side, the value of the lift, as given by formula (2), is equal to the total weight P of the aeroplane in horizontal flight. We thus have :

$$P = R_y = K_y SV^2$$

Eliminating the speed V between the two equations (5) and (6) we come to another formula giving the power W:

$$W = P\ \sqrt{}(P/S)\ (K_X + \sigma/S/(K_y)^{3/2}$$

We thus see that the necessary power is in direct proportion to the value of the term

$$(K_X + \sigma/S)/(K_y)^{3/2}$$

I have represented the term by the Greek letter S and call it the coefficient of power.

For a given aeroplane, the value of % will reach a minimum for a certain angle of incidence in flight, and to that particular value of S will correspond the minimum of power required, and that is why I propose to call that special value S,n the *coefficient of minimum power*.

That coefficient of minimum power is used to calculate the minimum power necessary to maintain in horizontal flight a given aeroplane, and the height of ceiling chosen will fix the nominal power to be provided for the engine. One can therefore see that, should it be possible to reduce in the proportion of two to one, for instance, the value of Sm, then an engine of half the nominal power would be sufficient to reach the same ceiling.

Since I have now clearly shown the way these several coefficients affect the aerodynamical qualities of aeroplanes, I can explain to you how one can design in the future, starting from existing aeroplanes, new ones which should reduce by more

than one half the present rates of aerial freight, and how also, when bringing other improvements which I will specify later on, one can expect to bring down the rates of aerial freight to the value of those now charged in France for first class railway passengers.

An aeroplane of high standard quality now has :—

A fineness equal to	... 0.12
Its coefficient of minimum power is equal to...	... 0.55
Its propeller efficiency is	... 0.73

Fuel and oil consumption of its engine at an altitude of 2,000 metres is 290 grammes per horse-power and per hour.

We will, moreover, admit the total weight of that aeroplane to be such as to allow it to climb to 4,500 metres within one and a half hours if the engines are run at full power,* and we will consider an aerial line of 800 kilometres non-stop flights.

If such an aeroplane has a wing surface of 100 square metres, for instance, and a power plant of 600 horse-power, its dead weight will be equal to 2,200 kilogs., and it will be able to carry a total load of 2,100 kilogs. Its total weight will thus be equal to 4,300 kilogs.

Its flying speed at full power at an altitude of 2,000 metres will reach 175 kilometres an hour, and its commercial speed at the same altitude can be reckoned as to be equal to 150 or 155 kilometres an hour.

The weight of fuel and oil necessary in a still atmosphere for a given flight is deduced from the formula (4), which, in the present case, gives the weight as being 568 kilogs.

But practical flying has proved that one must expect to have to fly against a fifty kilometres an hour wind, and therefore it is necessary to have at least a fifty percent. safety margin in fuel and oil. In other words, we shall have to carry in the case under consideration a total weight of fuel and oil equal to 568 x 1.5 = 850 kilogs, whilst the average consumption will only be of about 1.1 times the calculated quantity of 568 kilogs., that is to say, 625 kilogs.f

The crew (pilot and second pilot-mechanic) together represent a weight of about 180 kilogs., and another 70 kilogs. are necessary for various instruments and T.S.F. apparata.

The total weight of fuel and oil, crew and instruments thus reaches eleven hundred kilogs., leaving a clear margin of one thousand kilogs. for passengers and paying cargo.

The present running costs of an aerial line of 800 kilometres long, with the above aeroplanes, are as follows :—

	Francs.
Petrol (820 litres at francs 1.82 for Boo kilom. flown) per kilom.	1.87
Oil (50 litres at francs 3 for Boo kilom. flown)	o. ig

Crew (pilot at francs 0.20 per kilom., second pilot-mechanic
at francs 0.15)* 0.35
• Sinking fund for aeroplane and engine 6.50
Upkeep of aeroplane and engine and aeroport expenses, etc. ... 4.50
General expenses of the company ... ... 4..00
Francs 17.41

Should an aeroplane always fly with full cargo load and without incident or interruption, the cost price per ton and per kilometre would thus be about 17.40 francs. But one has to reckon with trips made with smaller paying loads, or incidentally interrupted, and therefore it is wise to calculate the cost price per ton on flights with half cargo loads as an average, which brings the present cost price per ton and per kilometre to 35 francs. That is about the price at which are now working the best aerial lines. A few companies work on still higher prices, either because the average cargo load is less than one half the maximum, or because they fly over particularly awkward grounds. They then reach such prices as 50 francs or £I 12.s od. per ton-mile.

Should such prices be considered as irreducible they would be practically prohibitive and little chance would be left to commercial aviation, since one would have to charge from eleven to twelve hundred francs per passenger from Paris to London or vice versa, without any profit to the company.

How can these figures be reduced in the near future? We will consider:—

1. —Advantages to be drawn solely from the betterment of aerodynamical qualities of the aeroplanes and of the thermal efficiency of engines.

2.-Advantages resulting from such improvements in the engines and in mechanical parts of aeroplanes as would bring reductions in the provision for sinking funds, upkeep of machines and general expenses.

1.-Advantages to be drawn solely from the betterment of aerodynamical qualities of the aeroplanes and of the thermal efficiency of engines.

Taking, for the sake of demonstration, the same size of aeroplane as already considered, that is to say

Wing area: 100 square metres.
Total weight: 4,300 kilogs. at the start.
Ceiling: 4,500 metres.
And if we suppose that we can bring down
Its fineness to ... ... o.o65 instead of 0.12
Its coefficient of minimum power to ... 0.28 0-55
Its propeller efficiency to ... ... 0.7750.73 The fuel and oil consumption of its
engines per horse-power and per
hour to ... ... 215 ego grams

then such an aeroplane will not require more than 288 horse-power to reach the same ceiling of 4,500 metres instead of 600 h.p.

It is evident that the reduction in power is only due to the betterment of the coefficient of minimum power and of the efficiency of the propeller.

One can thus spare 312 horse-power, which means a saving of 468 kilogs. of dead weight (at the rate of one and a half kilogs. per horse-power for the engine and its appliances).

The quantity of fuel and oil to be carried can be deduced from the formula (4), as has already been done for the first type of aeroplane considered. This gives

p = 230 kilogs.

The weight it is wise to carry up will be taken equal to

230 x 1.5 = 345 kilogs.

and the average actual consumption will be about

230 x 1.1 = 255 kilogs.

The weight saved on fuel and oil with the second type of aeroplane will thus be practically

850 - 345 = 505 kilogs.

and the excess of useful load is thus increased to a total of

468 + 505 = 973 kilogs.

The commercial efficiency of that type of aeroplane will thus be practically double that of the first type, since it will be able to carry 1,973 kilogs. of passengers or paying freight, instead of 1,000 kilogs.

At the same time that aeroplane will cost about 25 percent less than the first, since its nominal power plant will only be half the size of the first, and the value of the nominal power plant now represents about half the price of an aeroplane.

The running cost of the new type of machine will then be as follows:—
Francs.
Petrol (335 litres at francs 1.82 for 8oo kilometres) per kilom. 0.765
Oil (20 litres at francs 3 for 8oo kilometres) ... ... 0.075 Crew ... ... 0.35 Sinking fund (75 per cent. of francs 6.50, since the cost price
of the aeroplane has been found reduced in that proportion) ... ... 4.90 Upkeep of aeroplane and engine and aeroport expenses ... 4.50 General expenses of the company ... ... 4.00 R
Francs 14.59

As the paying load has been raised to 1,973 kilogs., the cost price per ton and per kilometre becomes equal to 7.40 francs, or—if calculated on a half cargo load basis—14.80 francs instead of the present cost price of 35 francs. London to Paris passenger fares can then be brought down to some 450 or 500 francs without profit for the company. Although very high, these last figures are more encouraging and nearly workable.

2.-Advantages resulting from such improvements in the engines and in mechanical parts of aeroplanes as could bring reduction in the provisions for sinking funds, upkeeping and general expenses.

When one can rely on an average life of one thousand hours for aeroplanes and engines instead of the present 200 or 250, much better prices will be obtained. Supposing that. such an aeroplane be bought for the same price as the last one mentioned, but calculating on 1,000 hours life, the sinking fund only calls for 1.10 francs, whilst the upkeep can be reasonably considered as half as expensive since the aeroplanes and engines will be of much better quality and strength. General expenses will also be considerably reduced through the much larger turnover that the companies will then be able to secure.

In such conditions the running costs would be as follows: Petrol ... ... per kilometre, francs 0.765 Oil ... ... 0.075 Crew 0.35

Sinking fund r.r0

Upkeep 2.25

General expenses (estimated) ... ... r.oo

Francs 5-54

That is to say, 5.55 francs for 1,973 kilogs. carried, or 2.82 francs per ton and per kilometre, or 5.65 francs if calculated on the usual half cargo load basis. Then the Paris to London fare will become about 190 francs, or say £4, the present price of the third class railway fare.

When that time comes then the aerial transport companies will be flourishing paying concerns, and no more subsidies will be required from the State, because the time saved in travelling by air from Paris to London, for instance, will certainly be worth a fare of say £6, showing £2 profit on each passenger.

It is worth noting in the last cost price we have given that fuel and oil, although intrinsically very expensive, amount to only fifteen percent of the total cost price, whilst the crew draws 6.4 percent.

On the other side, sinking fund, upkeep and general expenses, which, on the present cost of 35 francs, represent 86 percent of the total cost price, have grown to 92 percent with the second example and come down to 78 percent in the last considered circumstances.

May I therefore state—in opposition to the sayings and writings of certain experts who do not know much about aerial transports—that it is not the cost of fuel or crew which makes aeroplanes expensive machines, but only their present short life, with resulting consequences. But we must expect and hope to see large and strong aeroplanes of the future become as safe as motor cars, steamships and railways, and when that time comes, then we will be able to apply to sinking funds the same coefficient as is now used for steamships, that is to say, about an average life of twenty-five years!

From the example I have taken for the purpose of demonstration of an aeroplane of one hundred square metres wing area and 4,300 kilogs. total weight, you must not infer that I expect the improvements of aerodynamicaI qualities (with the consequences I drew from it) to be realised by such small machines. I believe one can build very good aeroplanes of four to five tons total weight; but I think it would be much nearer to reality to talk of 500 square metres wing surface, twenty to forty tons total weight and fifteen hundred to four thousand horse-power. Such large machines* will most probably have very thick wings. By thick, I mean about six feet, and within these wings will be provided cabins, saloons and every comfort for passengers. It is worth noting that if a large increase of the size of an aeroplane does not improve materially its commercial efficiency, it has nevertheless the great advantage of allowing much more room for passengers and freight, and that can be easily understood when one remembers that whilst the weight, the power, the capacity in cargo and the cost price of an aeroplane vary as the square of its lineal dimensions, the volume to be reserved for passengers will vary as the cube of the same dimensions. In other words, if the weight, surface and power have been increased one hundred times, then the cubic capacity will be one thousand times larger.* Passengers, for instance, will proportionately dispose of ten cubic metres each instead of one cubic metre in the first case, and that shows one of the most important advantages to be drawn from the manufacture of very large machines.

It is not exaggeration to talk of the days when the price of aerial transport will come down to two francs per ton and per kilometre, since an average life of two or three years—or say two thousand hours of flight—for aeroplanes and engines would suffice to bring that result, even with petrol and oil at their present very high prices. It is practically certain that aeroplanes of the future will not burn petrol but rattier heavy oils or other cheap fuels, the cost price of which should be about one-fifth that of petrol.

Now first class passenger fares in France are calculated on the basis of 21.15 francs per one hundred kilometres. Supposing that a passenger with his hand luggage weighs an average of ninety kilogs., we find that the ton-kilometre (passenger) runs to 2.35 francs. That shows that the future prices of aerial transports will be of the same order as the present first class passenger fares in France.

More striking still is the comparison with the fares on steamship lines, since first class passenger nowadays pays about six francs per ton-kilometre, whilst state cabins are charged at the rate of ten francs.

In conclusion, aerial transports are very expensive for the present because they are not yet out of the experimental stage and that the sinking funds, the upkeep and general expenses are very heavy; but one can reasonably say that within ten years these costs should be reduced in the proportion of seven to one, and within twenty years roughly in the proportion of fifteen to one.

Moreover, one must not forget that "time is money," and therefore a saving of four or five days on London to Cairo, for instance, will be of tremendous value to business men. The fares charged will then be of comparatively small importance to them as long as safety and speed are secured, and as aeroplanes will be unbeatable as regards speed, aerial transports are bound to wipe out all other systems of international communications.

We must therefore work hard and steadily, with full confidence in the future of aviation.

DISCUSSION.

The CHAIRMAN said they had listened with interest to M. Breguet's display of an optimism which was well worthy of their respect. He had indicated the steps by which they might compete on equal terms with railways for certain classes of traffic, and if they did not arrive at the whole result they might look step by step at the various factors and see which of them they had failed to bring up to the high level which had been foreseen. He had suggested that the fineness of aircraft could be improved by 50 percent. He had shown the hope that fuel consumption might be reduced, and had discussed similar factors, showing that even if these factors of improvement could not all be obtained *in toto*, he had shown that the sum of the advances obtainable make a fundamental and logical improvement in the whole aeronautical situation. M. Breguet's optimism was legitimate if the moneys spent on research were not reduced. Each specialist could then tackle, in his own sphere, some one of the problems that M. Breguet had exposed, and in such measure as they secured improvement, these would become operative together, and we should approach the competition, on an equal basis of charge, of first class railway fare and travel by aeroplane, with the enormous advantage to the aeroplane traveller that he had saved so much in time. With regard to the author's prophecy of the 40-ton monoplane, he did not suppose that an engineer of the author's distinction would have put that forward entirely at random without having thought it out as a thing thoroughly worth considering as a commercial possibility. The author had shown that the real incoming of aerial transport was not going to be achieved by prophetic letters to the Press nor yet by meetings of the Civil Aviation Advisory Board at the Air Ministry, unless the latter proved themselves competent to understand the technical nature of the problem and secured a sufficient addition of funds for the specialised research which civil transport needed. He agreed that civil transport was the basis for full factories and full factories the basis of military aeronautical operations. Nothing else but the scientific work of men of intelligence and intellect in the laboratory and in the air would solve the technical problems; and nothing but a solution of the scientific problems could make air transport self-supporting. Everyone would agree that that was the solution to be obtained. It was deplorable to realise that the authorities seemed so little capable of realising this, that they allowed the technical and research expenditure to be cut down with a view to national economy; he felt very strongly that this was simply national extravagance. It was killing that which alone could economically keep the aircraft factories in being in the state necessary to provide the fighting machines for aerial defence—Britain's, and indeed any country's, first line of defence.

Mr. F. HANDLEY PAGE said it was very interesting to have M. Breguet to address a meeting of the Royal Aeronautical Society. He had been one of the great pioneers of aviation, and one who had devoted himself not merely to the type of aircraft that they saw flying today, but he believed some of his earliest efforts were in the direction of the helicopter, which, if it had not produced practical results, had at least produced a very lengthy correspondence from the inventors. In having M. Breguet at the meeting the Society was greatly honoured. He (the speaker) was an optimist; his optimism was not based merely on the feeling that the success of air transport depended solely on the research laboratory, which the President had emphasised. It had been very much borne in upon him during the last few days that even with the latest and best of up-to date machines they did not necessarily achieve an air service. An air service was dependent, quite apart from equipment, on the ground organisation that dealt with it. If we were able to make machines that we could run at one-third, or, one-fifth, or even one-fifteenth of the present-day cost, that would not make possible a flight to Paris in thick fog and low clouds, and it would not enable the pilot to know whether he could start now or in an hour's time unless the meteorological service was very good. It was the improvement of the adjuncts to an air service, quite apart from the machine itself, or the skill of the mechanics who looked after it, or the pilots who flew it, that would make development possible. "If it were possible to fly present-day machines in all weathers strictly to scheduled time, and for them to return strictly to scheduled time, no matter what the weather was, costs would be very considerably reduced as well as charges to the public. He had looked with great interest at the formulae in the paper and he did not know whether there was a solution to any of those equations that gave one the value of profits. It seemed to him that if they could determine that accurately by means of an equation it would be an excellent thing for mathematical treatment. As a distinguished mathematician had said, " What comes out of the mathematical mill depends on what you put into it," but there were so many variables in arriving at a result by mathematical methods that it was necessary to study the component factors entering into it before they could arrive at a figure for the profits they were likely to get. He laid stress on the particular item of the ground. Organisation. He was very much interested to see, in the latter part of the paper, M. Breguet's visualisation of the very large aeroplane. The difficulty of structure weight had always seemed to him to be rather insuperable if they went beyond a certain size. In the four-engined machine made by his firm, which weighed 30,000 lbs., it seemed that they were gradually approaching the point where materials had been used to the best possible advantage, and there seemed little possibility of improving the use of those materials and getting over the theoretical disadvantage that was attendant upon increased span. He did not know whether M. Breguet, with his great engineering skill, had thought of any method by which that disability might be overcome. Without a doubt, were it possible to obtain a scale effect with larger sized machines, by which they could fly with greater loadings per square foot and thus decrease the span, such an improvement might be possible. In that direction it was rather interesting to know that in nature some of the bigger birds which, he was assured by Dr. Hankin, flew at the same speed, carried a greater load per square foot, although their landing speed was the same. It would appear that nature had got over structural difficulties by introducing a scale effect. Whether it was possible to utilise that in a very much bigger way in the construction of large machines he did not know. Perhaps M. Breguet could say something about it. He again thanked M. Breguet for his interesting paper, and congratulated him on having read it in English.

Captain GOODMAN CROUCH congratulated M. Breguet on his very interesting and rather optimistic paper, and the Society on having the opportunity of listening to one of the earliest pioneer aviators of France, who was also one of her greatest engineers.

He recalled that he had had the honour of being associated with M. Breguet in aviation some 16 years ago, when he (M. Breguet) put into the air a type of aeroplane which was called a double monoplane, since it was so unlike the usual box kite form of biplane then known. People looked at this curious beast and asked what it was, since with biplane wings it had a fuselage and was almost entirely of metal construction. One had only to consider the machines of a few years later to realise that M. Breguet was a true prophet, for it would be remembered that he alone at that time was the designer working on those lines.

With regard to the air transport costs quoted by M. Breguet, the list of prices shown for running costs of an aerial line of 800 km. length included an item of Frs.6.50 for sinking fund out of a total of Frs. 17.41. This, in his opinion, was a remarkably high percentage. It indicated that in the hypothetical case M. Breguet had taken he had chosen a machine presumably with a far shorter life than that of present-day aircraft, and he believed this item could be considerably reduced. This appeared to be borne out by the length of life assumed by M. Breguet when considering the advantages resulting from such improvements in engines and in mechanical parts, as could bring reduction in sinking fund and upkeep charges. Under this head M. Breguet had assumed the life of present-day machines to be 200 or 250 hours, and predicted the possibility of improving this to 1,000 hours.

Captain Crouch ventured to suggest that even during the war the life of 250 flying hours had been reached, and even surpassed by some of the heavy bombers, and the figure of 1,000 hours was hardly a prophecy since a total of approximately 800 hours had already been reached by a machine on the London-Paris Service.

With regard to the advantages hoped for from the betterment of aerodynamic qualities, M. Breguet's fineness coefficient indicated a figure for *L/D* of approximately 18~. This appeared to be optimistic, but he dare not say much in criticism of the point, since M. Breguet had already shown himself to be a true prophet. Recent aerodynamic tests on a model of a Woyevodski type had given a maximum *L/D* of about 12, and that, as far as he knew, was the highest figure yet reached for a complete model.

M. Breguet's remarks concerning future large machines were teeming with interest, but he felt that M. Breguet was extraordinarily optimistic in assuming that the increase, in volume available for passengers would be so much greater than the increase in weight, power and cost.

Finally, he again thanked M. Breguet for the lecture, which from a technical point of view gave so much food for thought.

Mr. W. O. MANNING said he was sure everybody appreciated the honour M. Breguet had conferred upon them by reading his exceedingly interesting paper. Those who had followed aviation from the early days would remember the large, series of aeroplanes known as the Breguet type, and the brilliant engineering design that invariably characterised them. He endorsed what Captain Goodman Crouch had said to the extent that, although he was not connected with commercial aviation, he certainly expected that the life of aeroplanes and engines today in commercial work was considerably longer than 200 hours. He was not an engine builder and could, perhaps, speak with less bias on that particular point, but he knew he could introduce M. Breguet to one or two English engine builders who could beat that performance. It would give them a great deal of pleasure when they remembered that in the early days of aviation practically the whole of English aviation was dependent upon French engines if they were able now to return the compliment. M. Breguet had done a very great service to aviation by pointing out the enormous importance of improvement in aerodynamical efficiency. One tried to improve the efficiency of the machines, but one did not appreciate, until it had been pointed out so clearly, what a very important matter aerodynamical efficiency was, not only in the saving of engine power, but in increasing the useful load and in cheapening the cost of the machine. With regard to M. Breguet's large machine, he was, of course, up against the dimensional law that the weight went up considerably faster than the area and that there is a definite limit of size for machines of present-day construction and design. But it by no means followed that the limit was the same for other types. It was possible that a very large monoplane, such as that referred to by the Lecturer, presumably with highlift wings, and with the passengers, fuel, engines, etc., distributed along the wings, might be capable of being constructed for a reasonable weight. He again thanked M. Breguet for his interesting paper.

Colonel W. D. BEATTY joined with previous speakers in congratulating M. Breguet upon having read his paper in English. There was one point which had struck him, and that was in regard to the symbols used. The efficiency factor he had assumed corresponded to our English *L/D*. Although there were exceptional cases where the technical experts in the two countries did speak each other's languages, the majority did not, but it was obviously desirable that they should all think in the same mathematical language. With regard to that he was glad to, say that preliminary steps had already been taken, in conjunction with the French Air Ministry, in order to get down to the same basis in England and France. While he did not wish to plunge into the argument as to the possibility or otherwise of the very large machine, he felt rather glad that there seemed to be such hope for the future. We still had a lot to do in the way of building up the traffic necessary to make it commercially practicable.

Mr. O. T. GNOSSPELIUS, after expressing his interest in the Paper, congratulated M. Breguet upon his courage in assuming that what we called *L/D*, or efficiency of the machine, could be improved, because he was quite sure, by his own experiments, that it could be done. They knew that certain experiments had been done at the N.P.L. and other places, and they got certain results, but if one made experiments one's self one got quite a different outlook. He was sure M. Breguet was quite right in saying he could get something like 15 to 1 *L/D*, because he himself had made pieces of wood in the shape of bodies, wings and tails which gave that effect, so that he did not see why the complete machine should not do the same. The trouble was that we did not know much about aerodynamics, and did not know the proper shape to make. It seemed to him that work on that line, was very essential, because 151bs. per h.p. was not practicable; we wanted to turn it into 30. In present machines the figure was more or less 151bs., and he was very glad that M. Breguet had brought forward this sort of figure for efficiency.

Mr. A. P. THURSTON said that the lecture impressed one very much indeed, that famous men, like science, were international. We (in England) more or less regarded M. Breguet as one of ourselves, and felt it a very great honour to receive a lecture from him. He was a pioneer of many things, but it was not realised that over 1,000 of M. Breguet's all-metal machines were actually used in the fighting line during the war. That was a very considerable achievement.

M. Breguet had brought out very clearly that to increase the efficiency of a machine it was necessary to reduce the "useless surface" (surface invisible) to the minimum amount, and the ingenuity of our designers must be utilised in taking, off all extraneous corners and everything which caused waste by increasing that surface. But there was a point which was not, perhaps, brought out quite so clearly, and that was that the efficiency in carrying weight per distance could be increased by actually increasing the speed of the machine and decreasing the lifting area. The great difficulty in this connection was that of landing speed. But there were ways of doing it which would enable them to get a higher speed still in the air with smaller surface, and yet maintain slow landing speeds. In other words, it was possible, as suggested by Mr. Handley Page, to get something of a scale effect by taking advantage—he was not at liberty to say how—of certain properties of the air. He did think that it would be possible to increase the present efficiency of our machines in order to get a greater weight mileage for a certain expenditure either of money or of fuel, and in that way increase the possibilities of commercial aviation. He had once taken the trouble to go through the figures, taking a line to India. Assuming there were a large number of machines, taking the cost of maintenance of grounds, and petrol at 2s. per gallon, assuming the organisation was so perfect that each machine could be flown for 10 hours a day, and each machine would last on an average two years, on that basis he had made out that it was quite possible to maintain a good dividend and charge passengers at the rate of 3d. per mile.

With regard to M. Breguet's remarks about engines, he would like to endorse what was said by Mr. Manning. It was only a few days ago, at Croydon, that a Napier Lion was pointed out which had been running continuously for 450 hours without being taken down. It was possible for British engines to do better still, and attain running efficiencies which would be considerably better than the 220 hours of the French engines.

He also endorsed M. Breguet's remarks as to aviation being the greatest future means of international communication, and in conclusion took the opportunity to thank M. Breguet for the courtesy which he had extended to him (the speaker) when he went through his works.

The CHAIRMAN translated some of M. Breguet's remarks, in which he explained that he was once of Mr. Handley Page's opinion that six tons was the limit at which the weight grew so fast that the area could not be expected to bear it profitably. He had proposed a complete departure in the type of construction, which, although it had not yet actually been put into being, he thought would get rid of the difficulty of the relation of weight to wing surface. He did not claim as an invention at all, but he had made calculations by which, using the thick wing type of machine, burying the engine and load in the wings, and distributing them carefully along the wings, on a 50-ton machine, he hoped to be able to arrive at much the same wing loading and power loading as would be obtained on a smaller craft of 2 or 3 tons. The wings would be 7ft. thick.

A hearty vote of thanks to M. Breguet concluded the proceedings.

NOTES ON M. BREGUET'S PAPER.

Contributed by Captain W. H. SAYERS: I think it is most important and most encouraging to hear so very high an authority as M. Breguet expressing the opinion that very great improvements in the aerodynamic qualities of the aeroplane are not merely desirable, but are also possible. The data given in his Paper are conclusive proof—if proof be needed—of the value of any great improvement in the "fineness"—or in usual English terms—the *L/D* ratio of aeroplanes.

The figures as to costs, etc., given in the Paper relate to French practice and are not directly applicable to British conditions. In certain respects I think British aircraft constructors may rightly claim that they can improve on those figures both aerodynamically and in the equally important matters of the durability and longevity of their aircraft. Such criticisms do not substantially affect the justice of the author's general conclusions, with which I am in entire agreement.

I think, however, that the time has now come to question what have hitherto been regarded as the fundamental bases of aeroplane design-bases which are apparently accepted by M. Breguet for they are implied in his two equations Nos. 1 and 2.

The assumptions are that an aeroplane may be regarded as a heterogeneous assortment of surfaces and bodies, and that the forces on each of the component surfaces or bodies, taken separately, may be added up and will then represent the total of the forces on the aeroplane. Every aeroplane designer knows that these assumptions are in fact inaccurate. No isolated body of good form can be cut in two and have its resistance determined by summing the resistance or the forces on the parts.

That the resistance and lift of present aeroplanes can be determined with reasonable accuracy by this method is, I submit, evidence that present-day aircraft are aerodynamically merely a collection of uncoordinated components. They will continue in this state for so long as designers allow themselves to be limited by a basis of design which is entirely empirical and seriously misleading. It is not true that a given wing has definite lift and drag coefficients at definite angles of attack which are independent of the body, tail and wing bracing structure to which the wing is attached. It is equally not true that the body, tail and other organs which form the complement to wings have force and resistance coefficients which are independent of the wing.

Because existing aeroplanes are so bad that they behave nearly as though these false assumptions were actually true, the designer comes to believe in them. Because he has come to believe in them, he has also come to believe that it is not possible very greatly to improve the aerodynamic efficiency of existing types of aircraft.

I am personally firmly convinced if designers can only be persuaded to forget all about the itemised resistances of the components with which they at present deal, and will regard a projected aeroplane as a single aerodynamic body, and will design it with an eye to its lines as a whole—just as they would design, say, an airship body—that eye will very speedily be found possible to design a complete aeroplane having a *L/D* ratio of 20/1 or over, or—in Nil. Breguet's terms—with a "fineness" of .05 or less.

TRANSLATION OF LETTER FROM MONSIEUR LOUIS BREGUET, DATED 28TH APRIL, 1922, TO THE ROYAL AERONAUTICAL SOCIETY.

Gentlemen, —I return herewith draft copy of the report of the meeting of the 6th instant, and regret that I have been unable, owing to absence, to reply earlier, as promised, to the various speakers who took part in the discussion. I very much appreciate their remarks, for which I thank them.

Replying in the first place to Mr. Handley Page, I agree with him that the Paper had in view only a part of the big problem of the future of aerial transport, but it is quite certain that while it is necessary to have good aircraft, it is equally indispensable to have as perfect a ground organisation as possible, otherwise the use of aircraft will be very uncertain, however perfect they themselves may be.

For instance, let us imagine modern navigation carried out as is actually the case with superb boats, but lacking any organisation of ports, routes, provision of buoys, lighthouses, wireless telegraphy, meteorological service, current charges, etc. The result would be practically negligible and its existence very precarious.

I did not raise this question at the meeting as I am of opinion that if from, the present time until the fact is accomplished, sufficient funds are available, it ascertain that excellent aerial ports, which are more easily and much more cheaply established than seaports, will rapidly come into being.

It is foreseen, however, that the aircraft destined to maintain the big international transport service will have to be of the amphibian type, and they will thus be able to utilise sea routes and have at their disposal the whole existing organisation in the big seaports throughout the world.

From carefully carried out experiments made in various laboratories I am able to state definitely that with the very good coefficients I quoted for "fineness" (indicating *L/D* efficiency), propeller efficiency, and minimum h.p. will certainly be realised in the near future.

I would like particularly to reply to Mr. Handley Page on the subject of large aircraft of the future. I can share his opinion that if aircraft are built on a larger scale whilst remaining geometrically similar, the ratio of wing structure weight to total weight will increase, since, all other things being equal, the weight of the wing structure, i.e., spars, ribs, interplane struts, bracing, etc., increases as the surface to power 3/2. For this reason I was for some time led to think that very large aircraft would not give results of interest, and that no comparison was possible between boats and aircraft, for even given an assured advantage to be gained by increasing the tonnage of ships, rather the opposite would obtain in increasing the size of aircraft. At the beginning of last year, however, I started to give serious thought to a type of monoplane with wings of a section deep enough to house engines, tanks and passengers. Further, by suitably distributing the loads on the wings it is easy to imagine an aircraft in which the dead weight, instead of increasing as the power 3/2 of the area, will only increase directly proportional to the area, and under these circumstances there were no longer any disadvantages in increasing the size of the machines.

I sketched on the blackboard after the discussion how I envisaged such an aircraft. It would comprise three fuselages, one in the centre for the crew, controls, instruments, etc., while the other two on the right and left respectively of the first mentioned, and far enough apart, would be used as hulls or floats. These would contain first class cabins. Finally, the tanks and the float would be distributed inside the wings. This distribution in weights in large aircraft would obviously give them a considerable moment of inertia, but as there would be no need for "stunting," their large lateral moment of inertia would mean a high degree of stability in the air.

With big monoplanes the reduction of parasite resistance can be pushed as far as one likes, and it is to be hoped that results similar to those of plain wings furnished with the tail unit can eventually be reached. It follows that these large machines would eventually have characteristics comparable with those of large birds, and the coefficient which would be applicable to them in that case will certainly be better than those I quoted in the lecture.

To Captain Goodman Crouch I admit that certain British engines have a much longer life than 250 hours, but I ought to say that the average life of the engines used by French companies does not reach this figure.

I am entirely in agreement with Captain Goodman Crouch that in a few years aeroplanes will obtain an average life of 1,000 hours, and for this reason I have every hope of seeing engines reach 2,000 hours.

With regard to Mr. Thurston's remarks, may I say that actually there were more than 6,000 metal machines, and not 1,000, put into service during the war which are still being used.

Finally, I would like to confirm what was the basic point of my paper, i.e., that technical research should be pushed forward without relaxation so that improvements already envisaged may be realised with as little delay as possible as well as those for which as yet one hardly dares to hope.

Yours, etc.,
(Signed) LOUIS BREGUET.

Document 3-5

Captain W. H. Sayers, "The Lesson of the Schneider Cup Race," *Aviation* 15 (5 November 1923): 577-580.

It was clear to many informed observers of the American victory in the Schneider Trophy competition of 1923 held in England that the key to the American success lay in how the designers of the winning Navy Curtiss CR3 aircraft had found ways to limit the amount of what was at the time generally called aerodynamic "resistance," meaning drag. In this short commentary that first appeared in the British magazine *The Aeroplane*, British RAF captain W. H. Sayers emphasized the "very complete way in which the whole detail design [of the Navy Curtiss racer] has been studied throughout with the idea of keeping resistance down to the absolute minimum." Primarily a critique admonishing the British Air Ministry for not providing the type of funds the U.S. government had been giving to winning such international air competitions, Sayers's editorial reported that the American emphasis on "detail design" for the purpose of reducing drag had made all the difference in the recent Schneider Cup race. The Americans had made every effort to make their airplane sleeker and faster. As a result, they had produced an integrated system designed for one purpose: speed.

Aviation historians still debate how influential the air races actually were on the aircraft design revolution of the late 1920s and 1930s and the more sophisticated civil and commercial aircraft that emerged from it. But there is no doubt that the U.S. government's discontinuation of formal sponsorship of international racing efforts left a void and somewhat slowed the pace of experimenting with cutting-edge aeronautical technology.

Document 3-5, Captain W. H. Sayers, "The Lesson of the Schneider Cup Race,"* Aviation *15 (5 November 1923): 577-580.

THE LESSON OF THE SCHNEIDER CUP RACE

"America Won Because it Entered the Most Perfect Examples Of Racing Aircraft Yet Seen in Europe"

Under the above title W. H. Sayers, (Captain, late Royal Air Force) contributed a very interesting article to our English contemporary The Aeroplane. It is reproduced here not only because of its evident technical value but also because of the clean sportsmanlike manner in which it appreciates the victory of our entries in the Schneider Cup race. – EDITOR.

The American team won the Schneider Cup because they entered for it what are undoubtedly the most perfect examples of racing aircraft that have yet been seen in Europe. It is entirely beside the point to suggest that they won the race on machines that would not have stood up to the Navigability Tests in rough weather.

They stood up to the tests in the weather that actually befell them, and their behavior on the water was of a nature which somewhat upset one's preconceived ideas as to their seaworthiness. Their floats are one believes distinctly heavy, they are certainly extremely strong.

THE CLEANEST THINGS EVER SEEN

As for the rest of the machines they are the cleanest things one has ever seen. The type 29 racing Nieuport biplane was one imagined as clean as a biplane could be. The Curtiss CR3 is a little cleaner in the matter of minor details such as wire fittings, etc., it has no radiator other than the surfaces of the upper plane, and despite the fact that it has an engine of some 500 bhp as against the Nieuport's 320 bhp, that engine is stowed away in a fuselage which is certainly of no greater--and looks to be of distinctly less--cross-sectional area than that of the Nieuport.

So far as the general arrangement of the machine is concerned the Curtiss machines show no surprising features. They use a normal type of thin wing--one believes it to be a "Sloan" section--which looks not unlike a RAF15. There is nothing abnormal about the wing wiring or strutting. The gap/chord ratio is not far from the usual value of 1.

The fuselage is an exceedingly pretty veneer--built affair-made one believes in two halves--and the tail unit has no striking peculiarities.

The floats are of a type which is now practically standardized in America for all float seaplanes. They are of the long type, designed to be stable fore and aft without a tail float, have one shallow step close under the C.G. of the machine, with a very long and fine tail behind it. The bottoms have a fairly steep Vee throughout, and the section above the bottom is practically semicircular throughout.

In the matter of air resistance they are undoubtedly good. On the water--as already remarked--they are surprisingly amenable to the wishes of the U.S. Navy's undoubtedly able pilots.

But it is not the general design of the major components, or the lay-out of the whole machine which accounts for the surprising performance which the Curtiss machines undoubtedly possess. This is due to the very complete way in which the whole detail design has been studied throughout with the idea of keeping resistance down to the absolute minimum.

THE CURTISS ENGINE

The Curtiss D12 engine must be considered to be a most important factor in the whole. It is evidently an engine of the very highest class--but more than this it is definitely a racing engine such as does not exist at present outside of America.

In making this statement one does not refer specifically to its low weight per hp. or its obviously excellent running qualities. The engine is astonishing--considering its great output--on account of its extremely small frontal area and of the absence of any projecting gadgets for which it would be necessary to provide bulges in the cowling.

There are other engines which can cheerfully face comparison with the Curtiss on a weight per horsepower and a reliability basis. One does not know of one of anything like the power which would go into the Curtiss racing fuselage with absolutely nothing sticking out except the airscrew boss and the stub ends of the exhaust pipes. Personally one has no doubt that a big body with nothing whatever sticking out is better for speed work than is a smaller one with an assortment of minor bulges to accommodate odd excrescences which will not go inside the main lines.

The Curtis body is small--and it has not one single bump on it from nose to tail. And it is pretty certain that a small clean body is better than a large clean one when one is trying to push it along at 200 mi./hr. or so.

For those who contemplate the designing of a machine to beat the Americans in the next race for the Schneider Cup it may be as well to remark that it is alleged that the new Wright T3 engine of 700 hp. has an even better frontal-area-to-power ratio than the Curtiss D12, and the reports that the Pulitzer Trophy machines with this engine have achieved speeds of within a mile or two of 250 to the hour seems to be entirely trustworthy.

No English engine maker has been able to afford to design an engine specially for racing purposes, and to that extent we are necessarily handicapped over this matter of body size.

THE REED AIRSCREW

Another factor contributing toward the success of the American machines is the Reed duralumin airscrew. Those used in the race were 8 ft. 9 in. diameter and they were directly driven at 2,300 rpm, giving a tip speed of 1,054 ft. per sec., which was near as no matter the velocity of sound. According to the R.A.E. the efficiency of an airscrew with a tip speed equal to the speed of sound is as nearly as can be zero.

The Reed airscrew at this tip speed is obviously fairly efficient--and it allows the Americans to dispense with a reduction gear on their racing engines. They can thus use a lighter engine--a smaller diameter airscrew,--and keep their undercarriage struts of reasonable length and still have water clearance. All these little things count. A special article on the Reed airscrew will be published as soon as possible.

The engines, radiators, and airscrews are the most striking features of the American machines. According to our authorities to attempt to run an airscrew of this diameter at this speed is to invite complete failure. The Americans have shown that our authorities are wrong on this point, and it is now up to us to find out how it is done.

The wing radiators are pretty certainly a very important feature. To all intents and purposes using this type of cooling system is equivalent to getting one's cooling for nothing whereas with a normal radiator the power absorbed in pushing the radiator of a 500 hp. engine at 180 mi./hr. would not, be much less than 5 hp. There is no novelty about the idea of using wing or body surface for cooling. There are obvious practical difficulties, and for some purposes some disadvantages. There is no disadvantage, for racing purposes at any rate, which can counterbalance an increase of at least 10 percent in the horsepower available for propulsion.

DETAILED DESIGN

These three main items--all concerned with the power plant--would by themselves give the American team ample excuses for having defeated us in the race. Beyond them however there is still something to be learned. The designer of the Curtiss racer was not content with general cleanness of design. He has seen to it that no unconsidered trifle was forgotten in the original lay-out and was afterward permitted to cause a minor irregularity of outline. As a matter of fact there are one or two small fittings for bracing wires on the underside of the top wing which look as though they might have been more carefully kept out of the draught. They are confined to the undersurface, where they have the least effect. It is extremely difficult to make a satisfactory wire terminal inside the wing surface--doubtless the wing radiators add to the difficulty--and there is precious little of these fittings anyway.

Otherwise the machines are as nearly perfect in this respect as they could be. The ailerons--on the bottom wing only--are recessed into the wing so that one could not put a strip of thin paper through the gap. Rudder to fin and elevator to tailplane hinge gaps are covered with rubber sheeting so that there is no gap in fact, and there is not a projecting control lever or wire anywhere on the machines.

The gasoline tanks are in the floats. The hand holes which give access to the filler caps, and those for inspecting the float interiors are recessed slightly into the surface. After these tanks were filled they were covered neatly by a square patch of doped on fabric, varnished as perfectly as the rest of the float and giving an unbroken surface.

The undercarriage struts go clean into the floats--apparently they are built in before the float is planked--and there are no fittings disfiguring their exposed surface. And so on throughout.

HIGH CLASS WORKMANSHIP

Not only is the workmanship and exterior finish of a very high class but it is obvious that the designer has taken as seriously the polishing up the last detail of his design as he took over the general lay-out. In fact he must have taken considerably more.

The merits of the American racers have been dealt with at this considerable length in order to make it quite clear how the Americans have achieved their success. It has been very generally asserted that the explanation is simply that the American Government was willing to spend as much money as might prove to be necessary

in order to win the Schneider Cup, whereas the British Aircraft Industry had no money to spare to defend it.

THE REAL COST OF WINNING

This explanation is accurate enough in its way. It is not however the whole explanation. In the first place it has to be remembered that what has been spent directly for the purpose of financing the Schneider Cup raid is a very small proportion of what has had to be spent to make that raid possible. The Curtiss D12 engine is a direct descendant of the Curtiss-Kirkham engine produced just after the war. Between that engine and the present there are at least three quite distinct varieties of Curtiss 12-cylinder high-performance engine each one an advance toward the ideal racing engine. No English manufacturer has been able to carry on the intensive development of high performance engines to this extent since the war although the engine work actually accomplished in this country under much more severe limitations on expenditure has possibly been of greater practical utility than that accomplished in America.

Over and above engines it may be recollected that since the War the American Aircraft Industry has produced a new batch of special racing machines every year--first of all for in attempt on the Gordon-Bennett trophy and since then for their own Pulitzer Trophy races. Last year alone America built more racing machines than Great Britain has built in her whole history. Thus the actual expenditure which has made it possible to produce the Schneider Cup Racers is extremely large--and almost the whole of it has been provided by the Government of the United States.

But it may be doubted whether the United States despite its large expenditure on this special object has an Aircraft Industry which is financially in a position to overwhelm the French Aircraft Industry in the production of racing aircraft. The Americans beat England--and still more thoroughly France--in the Schneider Cup race because regarded as racing machines pure and simple their Curtiss CR3s were as nearly perfect in every detail as they could be made.

WHAT IS NEEDED TO REGAIN THE CUP

The British Aircraft Industry can produce machines at least the equal of the CR3s. It needs for this purpose a fair amount of money--which at present it does not possess--but it need not expend a tithe of the money which has been spent on this purpose in America during the past five years. For there is not the least doubt that so far as general technical ability is concerned the British Aircraft Industry is ahead of the American, and that much that America has learned in the process of developing racing aircraft we have learned from our general experience.

But we have to learn that racing machines cannot safely be improvised at the last minute. The perfection of detail, the avoidance of every unnecessary compromise are essentials in the design of a successful racing airplane.

In aircraft built for either war or commerce design throughout is a matter of compromise between the conflicting requirements of aerodynamic efficiency practical utility, and cost, and British designers who have been brought up strictly to design for practical utility seem sometimes inclined to under-estimate the aerodynamic importance of detail design on the resistance of the complete machine.

Taking everything into consideration one has no doubt whatever that Great Britain can bring the Schneider Cup back from America--if not next year at least the year following. If we are to do it next year we have to start work at once, and some money has to be raised quickly.

INTER-GOVERNMENTAL SPORTS?

But on the whole the question of whether we are to try for the Cup again seems to be a matter for the Air Ministry to settle up with the Treasury. One had an idea that the Schneider Cup was instituted as a sporting Trophy. Since the advent of the mechanically-propelled vehicle one has had to modify one's ideas on the subject of the connection between sport and commerce far as motor car and airplane racing is concerned.

But the advent of the U. S. Government as a direct competitor in a "sporting" event is somewhat of a startling innovation. Obviously if the American Government has decided that the development of racing aircraft will lead to technical developments of real utility it is perfectly entitled to finance the design and building of such craft and to enter them for International events. And certainly none can suggest that in any detail the American team at Cowes and their auxiliaries have shown themselves to be anything other than sportsmen in the highest sense of the term. But it is pretty obvious that no private individual or corporation can afford to compete in such a test with the Government of the richest Power in the World.

Personally one believes that it would be all to the good were the precedent set by the U. S. Government in this matter followed by others. Inter-Governmental sporting events should have the happiest results on International relations if they were to become normal events but one cannot expect the British Aircraft Industry to compete on level terms with the American Government.

If the Air Ministry agrees with America as to the importance of racing as a method of developing the technique of aircraft design, then it is up to it to accept the perfectly open and honest challenge which America has offered, and to enter a team for the Schneider Cup and the Pulitzer Trophy races either next year or the year following.

THE "SEA LION"

The general characteristics of the "Sea Lion" are already pretty well known, because the machine is actually that which won the Schneider Cup last year at Venice. But certain modifications of detail have been made. The nose of the hull has been modified--the marked "ram" of recent Supermarine hulls has been considerably reduced by bringing the chines in more gradually which probably makes the hull a bit dirtier on the water but correspondingly reduces the air resistance.

Fairing pieces have been added behind the side of the main step, and wing-tip floats of an elliptical cross section, of reduced buoyancy-- but fitted with bow hydrovanes to give dynamic support on the water have been fitted.

The engine cowling has been cleaned up appreciably and a smaller radiator of the "long tube" type has been fitted. Altogether the effectiveness of the modifications is shown by the increase of some 10 mi./hr. in the average speed attained this year as compared to the last race for the Cup.

In fact the "Sea Lion" did astonishingly well--the American racers ought to have beaten her by more than 20 mi./hr. if their superior cleanliness of line had been all gain, and the result actually achieved would seem to indicate that the flying boat type has intrinsic aerodynamic advantages as against the twin-float seaplane.

Personally one believes that the "Sea Lion" with wing radiators and a totally cowled-in "Lion" could give the CR3s a very good race indeed without further alteration. Unfortunately we shall not again have CR3s with which to compete.

THE CAMS

The two CAMS boats are of very similar design--the only important difference being that the type 36--designed for last year's race is a tractor with the pilot behind the wings, whereas the 38 has a pusher airscrew with the pilot ahead. Both are extremely clean and taking-looking boats. The type 33 is alleged to be some 10 mi./hr. the faster of the two.

The hulls are thoroughly well-made boat-built jobs, roughly rectangular in section forward, with a domed upper surface and the concave bottom which is favored by Signor Conflenti and by Mr. Short. They have but one step, the tail of the hull rising fairly steeply behind it, and the fin and tail plane are built up as part of the hull.

The wings are of the normal single-bay biplane type, but with a smaller chord on the lower wing. The engine nacelle is built up into the upper center section and is very thoroughly streamlined--the airscrew in both tractor and pusher types being fitted with a large spinner.

The engine mounting is used as the central support of the upper wing, thus securing a minimum of strutting at this position.

Both boats are exceptionally clean and given engines of the same class as those against which they had to compete should have made a very good show.

One believes that in regard to seaworthiness these two boats are distinctly good. They make a lot of spray getting off, and if they do not come away cleanly they fall back with a somewhat startling thud. But they probably unstick better in a bit of a sea than they do in a flat calm. They are--so far as one can judge--rather better rough-water craft than the American float machines, and closer approaches to pure racing machines than is the "Sea Lion." They were however outclassed in the matter of engine and in addition their luck was right out.

THE LATHAM

The big Latham boat with Lorraine engines in tandem was the foul-weather hope of the French team. Her flight to Cowes in a full vale was a very fine performance indeed, but one cannot feel very greatly enthused over the design. There is no doubt that for really rough seas one needs a heavily-built seaplane with plenty of engine power. The Latham had these qualifications and perhaps with the limited choice of engines available in France it may be regarded as a very fairly satisfactory attempt at a powerful and reasonably fast seagoing flying boat.

THE BLACKBURN PELLET

The fate of the Blackburn "Pellet" is very greatly to be regretted. All those of the Blackburn Staff and of Mr. Saunders' staff who worked so hard to get the machine ready in time for the race deserve every sympathy with the very bad fortune that befell their efforts.

One sympathizes with them the more because in many respects the "Pellet" was of a distinctly taking design, and because one feels sure that had there been a reasonable opportunity of getting the machine tried out before the race it might have put up a really good performance. .

As it was the machine had no chance. It was designed to make use of an existing hull which in fact proved to be defective in strength and unpleasant on the water under conditions quite different from those for which it was originally designed. Nevertheless one believes that it could have been made to get off and one is certain that in the air it would have been very fast.

But it was an effort to improvise a racing machine--and since the Americans have taken a hand in the game improvisations have a precious poor chance in International Air Races. Mr. Kenworthy's lack of seaplane experience spoiled what little chance remained.

The hull itself was of the Linton Hope type in general design, and of very clean lines. It was however intended for a much less powerful machine than the "Pellet" and was pretty certainly overloaded on the water.

On this hull was mounted a bottom wing of small chord and span, and above it--supported on a steel tube cabane and outwardly-raking N struts a top wing of considerably larger dimensions.

The Napier "Lion" was mounted above the top wing over the central cabane, and drove a tractor Leitner-Watts steel airscrew. Originally tubular radiators recessed into the lower surface of the top wing were fitted. These were not satisfactory, and a single Lamblin, mounted below the engine was later fitted instead.

The machine was undoubtedly distinctly tricky to handle on the water. A hull of relatively small beam--a high C.G., a narrow base for the wing-tip floats, and the torque of the Napier "Lion" made her very prone to immerse a wing-tip on opening up the engine. The hull obviously tended to porpoise badly at low speeds, and this combination of qualities needed a lot of handling.

Document 3-6

John K. Northrop, quoted in Garry R. Pape, *Northrop Flying Wings: A History of Jack Northrop's Visionary Aircraft* (Atglen, Penn.: Schiffer, 1995).

The following comments by Jack Northrop, quoted in a 1995 book detailing his aircraft designs, reflect his affinity for streamlining, but from the viewpoint of his extraordinarily innovative approach to aircraft construction technology. In this fashion, Northrop's abilities to integrate a new type of structure accentuated the aerodynamic capabilities of new airplane design. First tried on the diminutive Lockheed S-1 sport plane of 1919, Northrop's technique of forming a plywood fuselage between a concrete mold and an inflatable rubber bag enabled a manufacturer to build an exceptionally streamlined airplane, and to do so rather economically. As for the wing, Northrop looked for inspiration to Dutch aircraft builder Anthony Fokker's cantilever designs. He went with cantilever even though Lockheed president, Allan Loughead, was against it, because he felt that potential buyers would feel the plane was unsafe without an appropriate amount of bracing struts and wires. A highly intuitive thinker, Northrop reasoned that the streamlined combination of cantilever wing and monocoque fuselage, both without drag-producing struts and wires, could operate at less power and do a better job than aircraft that weighed much more. His gamble proved highly effective, and the new airplane, the Vega, became one of the breakthrough designs of all time.

Document 3-6, John K. Northrop, quoted in Garry R. Pape, **Northrop Flying Wings: A History of Jack Northrop's Visionary Aircraft** *(Atglen, Penn.: Schiffer, 1995)*

NORTHROP FLYING WINGS
QUOTES FROM JACK NORTHROP'S FIRST COMPANY:

A steady program of development and refinement has been underway for the past twenty years until we have at present [1930] a number of carefully designed and comparatively efficient planes embodying streamline fuselages, carefully cowled engines, and 'clean' landing gears with superfluous struts, wires and fittings suppressed to an absolute minimum. It seems quite apparent that our best designs are close to the limit of practical efficiency; yet we find that their maximum over-all L/D (lift-drag) ratio is only about 10, whereas the L/D ratio of the active supporting surfaces of an airplane is normally double this amount.

An analysis of the items adding to parasite drag in the normal design shows that landing gear, power plant, fuselage, interference and bracing, and control surfaces are the major contributors to parasite power loss; the item of control surfaces being by far the smallest. Individual examination of the various units shows that nearly all possible improvement has been made in existing designs.

"We didn't dare to go the whole way and eliminate the tail."

It was obvious that the ideal flying wing was impractical or impossible to construct except in very large sizes. The airplane cannot justify its existence unless it has capacity to carry a comparatively bulky cargo. With conventional wing thicknesses it was impossible, even with very large wing taper, to build a flying wing in sizes of less than 150 or 200 foot span.

The whole objective was to build as clean an airplane as we could possibly conceive in those days. The average airplane had struts or wires or fuselage forms that weren't as smooth or streamlined—with as low a drag as possible. It was pretty obvious, it seemed to me, that a full cantilever wing neatly faired to the fuselage on a perfectly streamlined fuselage would take less power to do the job than some other types. So it was a breakthrough in that we went wholeheartedly into, for the time and at the time, to conceive as clean an airplane as we could. Fortunately, the work in Santa Barbara some years previously had already developed the technique for building a fuselage. It was then just the necessity of designing a full cantilever wing.

Document 3-7

B. Melville Jones, "The Streamline Aeroplane," *Aeronautical Journal* 33 (1929): 358-385.

In 1929, Great Britain's leading aerodynamicist, Cambridge professor Dr. B. Melville Jones, presented what in retrospect must be regarded as the single strongest statement made in favor of the advantages of reducing drag through streamlining that has ever been made in the history of aviation. The paper he read before the Royal Aeronautical Society, entitled "The Streamline Airplane," outlined conclusively how streamlining directly reduced the total drag of an airplane. Jones stressed that such drag reduction would result in higher cruising speed and lower fuel consumption and would consequently increase range and payload. The Cambridge don pointed out the inefficiency of previous aircraft designs, and he cited reduction of form drag as the most important obstacle to increased aerodynamic efficiency.

The aeronautical community received Jones's landmark speech with tremendous enthusiasm, and their reaction to his talk indicated the first widespread conversion of an entire aeronautical community to the benefits of streamlining.

Document 3-7, B. Melville Jones, "The Streamline Aeroplane," **Aeronautical Journal** *33 (1929): 358-385.*

During the past two years Professor Jones has been turning his attention to quantitative work on control. He has recently completed a first stage in a research showing by instrumental records exactly what happens to all the controls and to the aeroplane itself for certain specified movements made by the pilot. The flights were carried out largely by Professor Jones in an aeroplane specially loaned by the Air Ministry to the Cambridge University Air Squadron; the results of this pioneer research work provided a basis for similar experiments on other aeroplanes to be carried out at the R.A.E. and at [Martingham].

About a year ago Professor Jones first drew the attention of the A.R.C. to the great step forward that was possible in the improvement of aeroplane performance by comparing the resistances of an actual aeroplane with those theoretically possible. Much further evidence has been added since that date with the consequence that the A.R.C. initiated a series of researches on the interference of one part of an aeroplane on another. The first results of the interference research, carried out by a sub-committee under his chairmanship, are now to hand and have justified the prediction made by Professor Jones as to the large improvement that was possible.

Professor Jones has been an independent member of the A.R.C. since its appointment in May, 1920, and has assisted in a number of further matters to which that committee has paid attention since that date.

THE STREAMLINE AEROPLANE

BY B. MELVILL JONES, A.F.C., M.A., F.R.A$_E$.S.

Ever since I first began to study Aeronautics I have been annoyed by the vast gap which has existed between the power actually expended on mechanical flight and the power ultimately necessary for flight in a correctly shaped aeroplane. Every year, during my summer holiday, this annoyance is aggravated by contemplating the effortless flight of the sea birds and the correlated phenomenon of the beauty and grace of their forms.

We all possess a more or less clear ideal of what an aeroplane should look like; a kind of albatross with one or two pairs of wings—depending on whether we live in Germany or Britain. In our more sanguine moments we even—like Alice and the cat—see the wings without the albatross. But progress towards this ideal, so far as the general purposes craft is concerned is, we must all admit, painfully slow. It has seemed to me that a contributory factor to the slowness of this evolution has been the lack of any generally understood and easily visualized estimate of what could be achieved were the difficulties in the way of realizing the ideal form overcome. There is a natural tendency to decide on one day that the gain—say 20 percent on the total drag, or 7 percent on the speed—to be had by spending endless trouble on improving the undercarriage design, is not worth the trouble; on the next day to come to a similar conclusion about the drag of the engine cooling apparatus; on the next day about the wires, struts and minor excrescences; and on the next about the pilot's view; omitting to notice that if all the improvements were made at once the total gain would not be some insignificant percentage of the whole, but might reduce power consumption to a small fraction of its original value and so extend the range and usefulness of the aeroplane into realms which would otherwise be unattainable.

Considerations such as these led me some time ago to examine the possibility of devising simple formulae for the power required by the ideal streamlined aeroplane, and this lecture is a result of that examination. The very simple formula which I shall discuss is not precise—the present state of aerodynamical knowledge does not allow precision in this matter. It is not possible, for example, to say that lower power expenditures than my estimate might not occur. I am not so bold as to prophesy any limit to progress, but I think that I have made out a case that the power expended on the ideally streamlined aeroplane would not be greater than my estimate. Since this is of the order of one third the power expended on modern passenger carrying aeroplanes, the estimate should suffice, until such time as it more nearly represents actual performance; by which time we may hope that aerodynamic science will have progressed to a stage at which more accurate estimates can be made.

The formula at which I arrive is so simple and obvious that many will consider that I have wasted breath, ink and paper in putting it before you. That may be, but the difficulty lies in the choice of the quantities to be neglected and the argument here bristles with points where the scientist can pick holes. If you had had the job of

putting a theoretically leaky argument such as this before a body of scientists, who are of necessity more concerned with detecting fundamental errors in reasoning than in devising engineering approximations, you would understand why I have gone to some trouble to expose, as precisely as I am able, the assumptions upon which the final conclusion is reached.

My lecture therefore deals with the possible reduction of aerodynamic head resistance, or drag, and hence of the power required for flight. At the outset it is perhaps worth noticing that such reductions will not greatly reduce the maximum power necessary for a given service, since maximum power is required for getting off from the aerodrome—an operation in which weight is the predominating factor. Reduction of drag will, however, enable an aeroplane of a given power loading, either to cruise at a higher speed or with a lower petrol consumption. This again will result in increased range or paying load, both factors of the first importance in aeronautical development.

Before the power required to overcome drag can be interpreted in terms of engine power, the airscrew efficiency must be taken into account. Efficiencies as high as 75 percent are practicable on present-day craft, and efficiencies higher than say 85 to 90 percent are unlikely to be achieved in the near future; hence the possible power economies to be obtained from further airscrew development—e.g., through the use of the variable pitch screw—through by no means negligible, are unlikely to be very great. I shall not deal with the airscrew problem further, except in so far as the body-airscrew interference may influence my main argument.

Thanks to the combined efforts of Lanchester and Prandtl we can now isolate from the whole power that part—the power to overcome induced drag—which is expended in supporting the weight on a wing of finite span. This induced power, as it may be called, depends primarily on "*span loading*" that is weight/(span)^2. For a biplane of reasonable proportions the induced power per 1,000 lbs. weight can be expressed as

$$2.80^1\varpi/\sigma V_m$$

where V_m is the speed, with one hundred miles per hour as the velocity unit.
σ is the ratio of the air density to that of standard air, and
ϖ is the span loading in pounds per square foot.

The brake horse-power corresponding to the induced power is obtained by dividing by the airscrew efficiency. Assuming an airscrew efficiency of 75 percent, an average span loading of say 2.2 lbs. per sq. ft., and making σ equal to unity, we get the following table for the b.h.p. required for the induced power in a normal biplane.

<u>B.h.p. per 1000lbs. weight</u>

Speed m.p.h.	For Induced Drag	Total normally supplied	(Ratio) Total Power / Induced Power
90	9.1	35	25%
100	8.2	45	18%
120	6.8	75	9%
150	5.	5120	5%

Although the "induced power" is an important item in the power account at the lower cruising speeds, it is not the predominating factor at speeds above 90 m.p.h. Moreover, since it is clear from theory that no notable reduction in induced power is to be obtained without using much larger wing spans, I shall not in this lecture examine the problem of reducing it still further.

Since no great power reductions are to be obtained either by improving airscrew efficiency or by reducing "induced power," it remains to examine the possibility of reducing that part of the head resistance which arises merely because the aeroplane is being dragged through the air, without reference to the fact that is must support its own weight. Using terms in common use, this part of the whole drag is described as the sum of the wing profile drag and the parasitic drags. The power required to overcome it is seen in the table above to lie between 75 and 95 percent (according to speed) of the total power applied to the modern general purposes aeroplane. Any serious reduction in this item will therefore have an important influence on the total head resistance.

We all realize that the way to reduce this item in the power account is to attend very carefully to "*streamlining*." It is the main object of this lecture to examine how far this item can be reduced by perfect streamlining, and this incidentally will involve a preliminary explanation of what I mean by an ideally streamline aeroplane.

We recall first a proposition relating to that mythical and much abused substance, "the perfectly inviscid incompressible fluid." Theoretically when this hypothetical fluid streams steadily past a body, such as an aeroplane, the forces exerted on the surface of the body are everywhere normal to the surface; no tangential or skin friction forces can arise. The total reaction on the body, the resultant of all the pressures acting normally to the surface, is such that the "*drag*," or down-wind component, will be related to the "*lift*," or cross-wind component, according to the Prandtl theory of induced drag. In other words, Prandtl's induced drag is the sum of the down-wind components of all the normal pressures which would be set up by that steady flow of the hypothetical fluid which gives the required lift. It follows as a particular case of this proposition that the steady flow of the inviscid fluid which gives no lift also gives no drag.

In aeronautical nomenclature a "*streamline body*" is one about which the flow of a real fluid, such as air, approximates closely to a steady flow of the hypothetical inviscid fluid, except in a very thin layer called the "*boundary layer*," surrounding the

exposed surfaces. In such a flow the pressures normal to the solid surfaces are very closely equal to the pressures of the corresponding flow of the hypothetical fluid, but the action of the real fluid in the thin boundary layer causes additional tangential surface forces, or skin frictions, which are not present in the hypothetical flow.

An ideally streamline aeroplane may therefore be defined as one which:—

(a) Generates a flow identical, except in a very thin "boundary layer," with the flow of an inviscid fluid.

(b) Experiences a pressure distribution identical with that due theoretically to the inviscid fluid;

and therefore

(c) Experiences a drag which is the sum of the induced drag and the tangential or skin friction forces resolved in the down-wind direction.

Like all ideals, the ideally streamline aeroplane cannot exist; the boundary layer must have some thickness, so that the flow cannot be exactly the same as one in which there is no boundary layer. If the "*Reynolds' number*" is small, or in other words, if the effective viscosity of the fluid is high, the boundary layer becomes so thick as to make the above definition meaningless, but with the very high Reynolds' numbers, typical of aeronautical practice, the approximation for a good streamline body is very close.

It is, of course, well known that, unless bodies are carefully shaped, they do not necessarily generate streamline flow but shed streams of eddies from various parts of their surface. The generation of these eddies, which are continually being carried away in the air stream, requires the expenditure of power additional to that required to overcome induced drag and skin friction. The power absorbed by these eddies may be, and often is, many times greater than the sum of the powers absorbed by skin friction and induced drag. The drag of a real aeroplane therefore exceeds the sum of the induced drag and skin friction drag by an amount which is a measure of defective streamlining.

Having arrived at the conclusion that the drag of the ideally streamline aeroplane is the sum of the induced and skin friction drags, the next step is to estimate the magnitude of these drags. Methods of estimating the former are well known and an approximate formula is given on page 359. No corresponding theory yet exists for the estimation of skin friction on curved surfaces such as those of the wings and body of an aeroplane. We are therefore forced to adopt some empirical method of investigation. The simplest experiments on skin friction are those upon a thin flat plate edgewise to the wind, for here the resultant, in the down-wind direction, of the surface pressures is necessarily zero. The best experiments, of which I am aware, upon the drag of such a plate at high Reynolds' numbers were made at Göttingen some years ago. They lead to the formula

$$k_F = 0.019\, R^{-.15}$$

where k_F, the skin friction coefficient, stands for (skin friction)/$\rho V^2 E$ and R stands for $\rho V l/\mu$ where

V stands for the relative velocity of air and plate.

E stands for the total exposed area of both sides of the plate.

l stands for the length of the plate in the wind direction.

ρ and μ are the density and viscosity of the air.

R is thus the Reynolds' number involving the dimension parallel to the wind direction.

The experimental range was approximately from $R = 2 \times 10^5$ to $R = 10^7$.

This expression (1) relates to conditions in which the "boundary layer" is turbulent over the greater part of the plate, that is to say, the air very close to the surface of the plate is eddying—as it were, rolling on the surface. Burgers, of Delft, has shown that near the front of the plate the boundary layer is not turbulent, the separate layers of fluid near the surface of the plate slip over each other smoothly. When the Reynolds' number is small this smooth region extends over the whole plate and the skin friction drag has, in these circumstances, been shown to agree closely with the theoretical expression

$$k_F = 0.66\, R^{-1/2}$$

Thus at very low Reynolds' numbers the drag coefficient is as equation (2), but at high numbers as equation (1). In Figs. 1 and 2 I have plotted curves representing the expressions (1) and (2), above, on the assumption of smooth and turbulent boundary layers respectively. The upper curve in each figure represents expression (1) and the lower expression (2). At medium Reynolds' numbers, where the boundary layer is smooth over the front part of the plate and turbulent over the rear part, the average drag coefficient for the whole plate will lie between the upper and lower curves of Figs. 1 and 2; its exact value depending upon the distance from the leading edge at which the breakdown from laminar to turbulent flow occurs. The curve followed by the drag coefficient in this intermediate condition may be described as a "*transition*" curve, between the lower and upper curves of Figs 1 and 2. I have lightly dotted in roughly calculated transition curves, on the assumption that the critical change from smooth to turbulent boundary layer occurs when the Reynolds' number formed from the distance of the breakdown point behind the leading edge has certain arbitrarily selected values, e.g., 10^5 and 5×10^5.

Burgers has found that for the flat plate the value of the Reynolds' number at this point of breakdown depends on the steadiness of the air in the tunnel before reaching the plate. When the flow was as steady as he could make it the change occurred when $R = 5 \times 10^5$. When the flow was deliberately made unsteady, by bringing the wind tunnel honeycomb nearer to the plate or by placing wire gauze across the tunnel in front of the plate, the break up of the boundary layer occurred

FIG. 1.
COMPARISON BETWEEN PROFILE DRAG OF WELL-KNOWN WINGS AND THE SKIN-FRICTION ON A FLAT PLATE

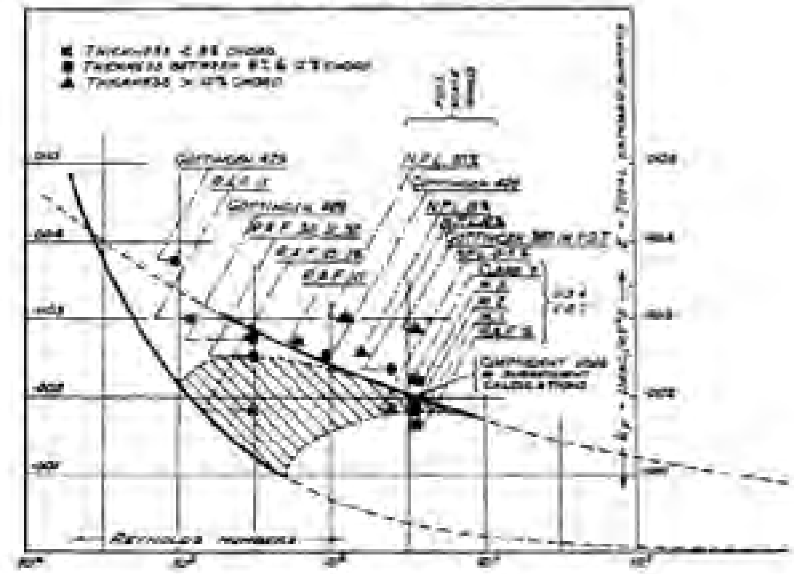
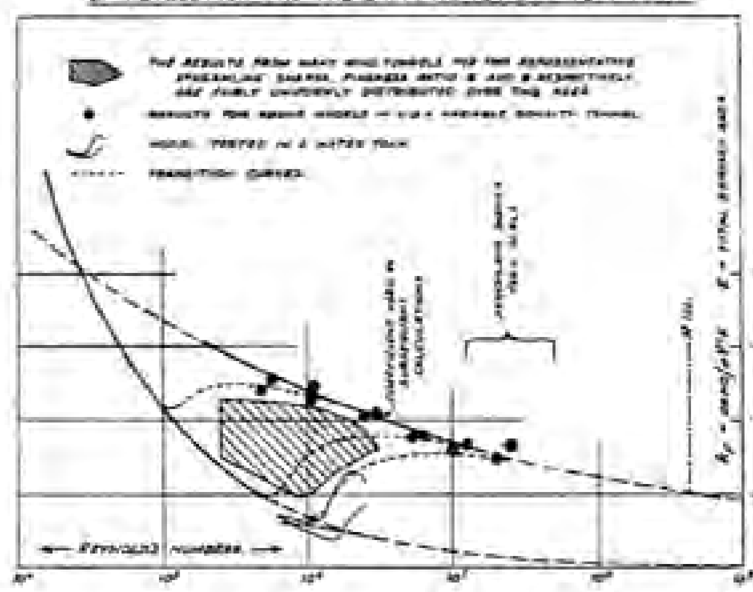


FIG. 2.
COMPARISON BETWEEN THE DRAG OF STREAMLINE SOLIDS OF REVOLUTION AND THE SKIN FRICTION ON A FLAT PLATE



earlier, $R = 10^5$ approximately. Thus in the former case the drag coefficient of the flat plate should be expected to follow the lower curve until R has increased to 5 X 10^5 and then follow the transition curve which starts at this point, to the upper curve. In the latter case the change would take place along a transition curve which leaves the lower curve at $R = 10^5$. If it is assumed that the point of breakdown is uncertain between the above limits, the drag of the thin flat plate would be expected to follow the lower curve below $R = 10^5$ and the upper curve above $R = 5$ X 10^6, but it might be anywhere within the shaded area when R lies between these two values. If the wind tunnel experiment were made on the plate between these two limits of R, the drag would be expected to vary erratically from tunnel to tunnel according to the turbulence of the air in the tunnel.

Our precise knowledge of the drags of very good streamline bodies other than the flat plate is confined at present almost entirely to the results of wind tunnel tests on solids of revolution, such as bare airship hulls, and on isolated wings. The former when moving axially exert no lift, and consequently their drag, if they conform to my definition of "streamline," should be entirely due to skin friction; the latter, when they exert lift should, if streamline, experience induced drag as well as skin friction drag. My first concern, when starting the investigation upon which this lecture is based, was to determine to what extent the drag of the better known streamline solids of revolution and the profile drag (total drag minus induced drag) of the better known wings, could be estimated from the known skin friction on a flat plate. To do this I have expressed the profile drags of certain wing sections and the drags of certain streamline solids of revolution in the form

$$k_F = (\text{drag})/\rho V^2 E$$

where E is the total exposed area. These coefficients are plotted in Figs. 1 and 2, where they can be directly compared with the curves showing similar coefficients for a thin flat plate, as explained above. The Reynolds' number for these wings and streamline solids is of the form

$$R = \rho V l/\mu$$

where l is the maximum dimension of the body parallel to the wind—the "length" of the streamline solid and the "chord" of the wing.

Comparison in Figs. 1 and 2 between the experimental points and curves for the solid shapes and the curves for the flat plate at the same value of R, amounts to comparison between the drags of the solid shapes and of a thin flat plate of the same total exposed area, having a length parallel to the wind equal to the maximum length in this direction of the solid shape. I shall in future describe such a thin flat plate as the "*equivalent plate*."

Figs. 1 and 2 are, I think, striking and significant. In Fig. 1 the points clearly tend to cluster around the upper curve, appropriate to the turbulent boundary layer, but one set of points, those for thin wing sections R.A.F. 25 and 26 at values of R in the neighborhood of 3 X 10^5, show a tendency to fall towards the lower curve. We may summarize this diagram by the statements that:—

When the thickness of the wing is less than 8 percent of the chord the profile drag is less than that of the equivalent plate. This includes nearly all wings used in Britain before some two or three years ago.

Up to a thickness of 12.5 percent the profile drag is within some 10 percent of that of the equivalent plate. This includes the great majority of wings used in Britain at the present time.

Thicker wings, up to 15 percent or 20 percent, apparently have higher profile drag coefficients—between, say, 15 percent and 50 percent greater than the equivalent plate, according to shape and place of test.

Turn now to Fig. 2. The bulk of the available information here results from extensive trials upon two models of bare airship hulls which were tested in a number of different wind tunnels in the various aerodynamic laboratories of the world, with the primary object of comparing the tunnels. The result was to show that wind tunnel experiments on very low drag models, such as these, are uncertain and depend to a great extent on the local conditions, such as turbulence, etc., in the tunnel. Speaking broadly, it may be said that the various wind tunnels gave experimental results for these models lying anywhere within the shaded area; you can take your choice as to which you are prepared to believe. There are clear signs, however, that as the Reynolds' number rises towards the extreme limit of atmospheric tunnels, the uncertainty becomes less and the tunnels tend to agree in placing the coefficient slightly below that of the equivalent plate. This tendency is somewhat faintly reflected in the shape of the shaded area.

The black circular dots relate to individual experimental results for these same two models in the American variable density wind tunnel. These points, which are carried to much higher Reynolds' numbers than were obtainable in atmospheric pressure wind tunnels, fit the upper or turbulent boundary layer curve with remarkable accuracy, even in the region of the lower Reynolds' numbers, where the majority of the atmospheric pressure tunnels give lower coefficients. The explanation of this is probably that the variable density tunnel employs a fine honeycomb, very close in front of the model, which should result, according to Burgers' experiments, in an exceptionally early break up of the boundary layer from laminar to turbulent flow. Thus under conditions in which, for a flat plate, the boundary layer would be turbulent over the majority of the plate, the drag of two different airship models is found to be closely equal over a very large range of Reynolds' numbers to that of the equivalent flat plate with turbulent layer.

The thin continuous lines near the bottom of the figure, in the neighborhood of

$$R = 10^6,$$

relate to an experiment carried out on a different model in the experimental naval tank of the National Physical Laboratory. Here the fluid—water—was very still and the model was pushed from behind instead of being supported by wires attached at various points on its surface. External disturbances were thus reduced to a minimum. This treatment is seen to result in the coefficient falling very close to the lower curve appropriate to a flat plate with *laminar* layer. The experimental curves, however, begin suddenly to rise at Reynolds' numbers slightly greater than 10^6, in exactly the manner which would be expected if the layer at the rear part of the model were beginning to become turbulent at the higher Reynolds' numbers. That is to say, the curves rise approximately along the theoretical transition curves, which here have a slightly different shape from those in Fig. 1, owing to the tapered tails of the models. This same model was tested with the same method of support in one of the National Physical Laboratory 7 ft. wind tunnels, and produced a curve almost coincident with the lower of the skin friction curves in this figure. It was not, however, practicable in the wind tunnel to reach a Reynolds' number much above 10^6 and no signs of the rise in coefficient was observed. These experiments, both in air and water, were repeated with a thin string loop slipped on to the nose of the model, to form a slight obstruction which, from previous experiments of a similar nature, would be expected to cause the boundary layer behind it to become turbulent. With this addition the drag coefficients both in air and water increased some fivefold and approximated closely to that for the equivalent flat plate with turbulent boundary layer. Thus, in circumstances where the boundary layer, from analogy with the flat plate, may be expected to remain laminar to a relatively high Reynolds' number, the drag of the model is very closely equal to that of the equivalent flat plate with *laminar* boundary, and the rise in coefficient, when it does occur, takes place in exactly the manner which would be expected on a flat plate with tapered rear. When, however, a slight change is made to the surface, which would be expected to break up the boundary layer over the greater part of the surface, the drag approximates closely to that of the equivalent flat plate with *turbulent* boundary layer.

I have examined the drags of other streamline solids of revolution in the same way as above, and I find that so long as the fineness ratio is not less than 4 the coefficients all fall between the upper and lower curves of Fig. 2.

Reviewing the evidence relating to streamline solids of revolution, it seems difficult to resist the rather surprising conclusion that their drag can be estimated with considerable accuracy from the drag of what I have called the equivalent flat plate. If the experiments recorded in Fig. 2 had been conducted upon thin flat plates of the same superficial area as the models and of the same length in the direction of motion as the overall length of the models, they cold not have given results more in conformity with the known behavior of the boundary layer on a flat plate, than the results actually recorded.

We do not yet know to what extent it may be possible to increase Reynolds' numbers whilst retaining a laminar boundary layer, but it seems probably that in the majority of instances the layer will be mainly turbulent before the Reynolds' number appropriate to aeroplane bodies in flight—say, between 2 and 5 times 10^7—is reached.

For the purposes of the present paper it is sufficient to note that the upper or turbulent layer curve appears to give an upper limit to the drag coefficient of good streamline shapes, whether they be in the form of wings or solids of revolution, provided that the fineness ratios are not less than 8 in the case of wings and 4 in the case of solids of revolution.

The information displayed in Figs. 1 and 2 relates to simple shapes such as isolated wings and solids of revolution. There is as yet little information available relating to good streamline shapes in combinations, such as the wings and bodies of aeroplanes, or to the distortion and interference to which a shape can be subjected without becoming unstremline . Such information as is available suggests that combinations, or minor distortions, can be made without appreciable increase of drag, provided we know how to make them. It is therefore reasonable to suppose, after examining Figs. 1 and 2, that complex bodies such as aeroplanes could, given sufficient knowledge and structural ability, be made to have a drag no greater than the sum of the induced drag and the drag of the equivalent flat plate.

If the drag of the streamline aeroplane is simply the sum of the induced and skin friction drags, the power required to tow it would be obtained by multiplying these drags by the forward velocity V. Let I and F be the powers required to overcome induced and skin friction drags respectively. If the streamline aeroplane is towed by an airscrew of efficiency η, effectively isolated from it so that there is no interference, then the b.h.p. required to drive the screw will be

$$(I + F)/\eta$$

How are we to regard interference between screw and body, when such exists? Interference will alter torque and thrust reactions between screw and engine. The change in torque may call for a redesign of the screw to make it balance engine torque at suitable engine revolutions. The change of thrust is of no interest, except in relation to the size of thrust bearing required, for it may be largely neutralized by change of body drag. Neither change is of direct interest from the present point of view, which is to examine any possible changes in b.h.p. due to interference, assuming always that *the screw is properly designed for the engine*. With this assumption it is not difficult to show that, to a first approximation, the b.h.p. required to propel the *streamline* aeroplane should not be seriously influenced by interference, provided that the interference does not cause the flow to cease being streamline.

We arrive at this conclusion by considering the disturbances left in the air behind the streamline aeroplane. They may be divided into:—

1. The airscrew slipstream.
2. The induced vortices.

3. The small scale turbulence, and possibly temperature rise, due to skin friction on the exposed parts of the airscrew blades and on the aeroplane.

Since, by hypothesis, the flow around the streamline aeroplane is substantially that of an inviscid fluid, except in the very thin boundary layer where the skin friction is applied, the *whole energy* expended—the b.h.p.—is included in the above three items.

The energy in the slipstream is mainly due to the axial velocity of the stream; it is dependent upon the effective diameter of the stream and the net thrust of the screw and body behind it. If, as we may reasonably suppose, these are unaltered by the interference, item (I) will remain substantially unaltered.

The energy *I* in the induced vortices will not be altered by interference, unless the interference appreciably alters the lift distribution across the wing span. If considered necessary, the alteration could be estimated and *I* taken to apply to the induced power associated with the actual lift distribution in the presence of interference.

The energy expended on skin friction on the airscrew blades will not be seriously affected by any reasonable interference.

The energy *F* expended on skin friction on the aeroplane will be affected only on those surfaces which are exposed to the slipstream. This effect will always be small; at high speeds, where *F* is an important item, the slipstream velocity is low, whilst when climbing *F* is but a small part of the power expenditure. To a first order, therefore, this item in the power account will also be unaffected by interference, but if higher accuracy is required *F* must be taken to relate to the estimated skin friction with allowance for increased velocity over surfaces exposed to the slipstream.

I do not maintain that the above arguments are precise, but if they are examined carefully, with due regard to the quantities involved in practicable aircraft, I believe that they will be found sufficient.

We are now in a position to state the main conclusion of the lecture, which is as follows:—

The brake horse-power required to propel the streamline aeroplane horizontally may be estimated as

$$(F + I)/\eta$$

where η is the efficiency to be expected from an isolated screw performing the service required.

F is the power expended in skin friction, which may for the present be taken to be that required to propel the equivalent flat plate.

I is the induced power.

As a first approximation *F* and *I* may be estimated for the aeroplane isolated from the screw. As a second approximation *F* can be altered to allow approximately for the increased velocity over surfaces exposed to the slipstream and *I* to allow for alterations in lift distribution across the wing span due to the slipstream. In the estimates which follow the first approximation is sufficient for my purpose and I have not proceeded beyond it.

When the streamline aeroplane is climbing, the above expression for brake horse-power must include another term *C*, equal to the product of weight and rate of climb. Thus

$$\text{b.h.p.} = (F + I + C)\eta$$

It remains to give numerical expression to the symbols *I*, *F*, *C* and η.

The estimation of *I* presents no difficulty. A simple formula $I = 2.80\ \varpi/\sigma V_m$ horse-power per thousand pounds weight with span loading ϖ in pounds per sq. ft. and V_m measured in hundred-mile-per-hour units has already been given. This formula applies strictly to a biplane or normal gap and aspect ratio and of rectangular plan form, but is easily modified (see footnote on p. 4) to apply to a monoplane or biplane of any desired gap or aspect ratio.

The skin friction power *F* depends upon the skin friction coefficient

k_F = (skin friction drag)/$\rho V^2 E$

and before *F* is estimated some value for k_F must be adopted. The question arises as to what value is to be given to k_F. If it is to be taken from the value for the equivalent flat plate, i.e., from the upper curve of Figs. 1 or 2, its value will depend on the Reynolds' number, and this raises the further question as to how the Reynolds' number for the aeroplane is to be estimated. The Reynolds' numbers of full scale aeroplane wings, based on the chord as a measure of size lie between 4 X 10^6 and 10^7 and the corresponding value of k_F for a thin plate (see Fig. I) lies between 0.0020 and 0.0017. The Reynolds' number of the bodies of full scale aeroplanes, based on their length as a measure of size, lie between 2 X 10^7 and 5 X 10^7, and the corresponding value of k_F (see Fig. 2) lies between 0.00155 and 0.00135. The logical proceeding would be to estimate the wing skin friction drag from Reynolds' number appropriate to the body. But for simplicity and so as to be on the safe side, I propose, for my present purpose, to take an over-all figure for k_F of 0.0020. This should leave us a little in hand to allow us to use moderately thick wing sections—say, thickness up to 12 percent of the chord—in our ideally streamline aeroplane.

Using this figure we have

$F = 0.002\ \rho V^3 E$ foot pounds (using any consistent units).
$= (27\ \sigma V_m{}^3 E)/W$ horse-power per 1000 lbs. weight.

where V_m, is in hundreds of miles per hour units, *E* is the exposed surface in square feet, and *W* is the weight in pounds.

Finally, the climb power *C*, per 1,000 lbs. weight, is easily shown to be given by $C = 0.030\ V_C$ where V_C is the rate of climb in feet per minute.

These three simple expressions for I, F and C, allow an approximate and probably conservative estimate to be made of the power that would be required by an aeroplane in any given circumstance, provided that the flow around it were everywhere streamline and no unnecessary eddies were generated. The difference between the actual power expended and the power so estimated must be regarded as being expended in the generation of unnecessary eddies which might be avoided by more careful attention to external form.

I have thought it of interest to prepare a few curves showing the performance, in level speed and climb, of a streamline biplane in air of standard density near ground level. For this purpose I have taken $E = 3.2 \times S$ where S is the conventional wing area, so that $F = 86\,\sigma V_m^{\,3}/\omega$ where ω is the wing loading in pounds per sq. ft.

I arrived at this value 3.2 by computing E roughly for a number of well-known aeroplanes and finding that E/S ranged between 3.0 and 3.5 with 3.2 as an average value. In drawing the curves I have taken σ to be unity and η, the screw efficiency, to be 75 percent for level flight and 70 percent climbing.

Fig. 3 shows four curves giving b.h.p. per 1,000 lbs. weight, for level flight at various speeds for, the following conditions:—

wing loading (ω) 10 and 15 lbs. per sq. ft.
span loading (ϖ) 2 and 2 ½ lbs. per sq. ft.

The individual points, with the names of well-known aeroplanes written against them, refer to the b.h.p. at top speed as given in the 1927 edition of Jane's "All the World's Aircraft." The small circles relate to aeroplanes for which ω is approximately 10 and the squares to those for which ω is in the neighborhood of 15.

The vertical distance between the points and the appropriate curve represents the power being expended on what, from the present point of view, much be described as unnecessary turbulence. Horizontal distances from the curves represent the increased velocity which might presumably be obtained for the same power expenditure in the absence of unnecessary turbulence.

Apparently large commercial aeroplanes, such as the Argosy, the W.10 and the Hercules, would, were they ideally streamline, either fly at the present top speed for one-third the present power, or alternatively travel some sixty miles per hour faster for the same power. Two-thirds of the power used by these aeroplanes is being expended on the generation of turbulence which, from the aerodynamic viewpoint, is unnecessary.

Fig. 4 is constructed on the same lines as Fig. 3 except that the abscissae represent rate of climb and the three continuous lines show the expenditure of horsepower required for a streamline aeroplane, divided up into the three parts: weight-lifting power, induced power and skin friction power. Here the span loading is 2.2 lbs. per sq. ft., which is a good average figure for modern aeroplanes, whilst the wing loading is taken as 10. The airscrew efficiency is taken to be 70 percent.

FIG. 3.
COMPARISON BETWEEN REAL AND STREAMLINE AEROPLANE
FIGURES FROM 1927 JANE. TOP SPEED.

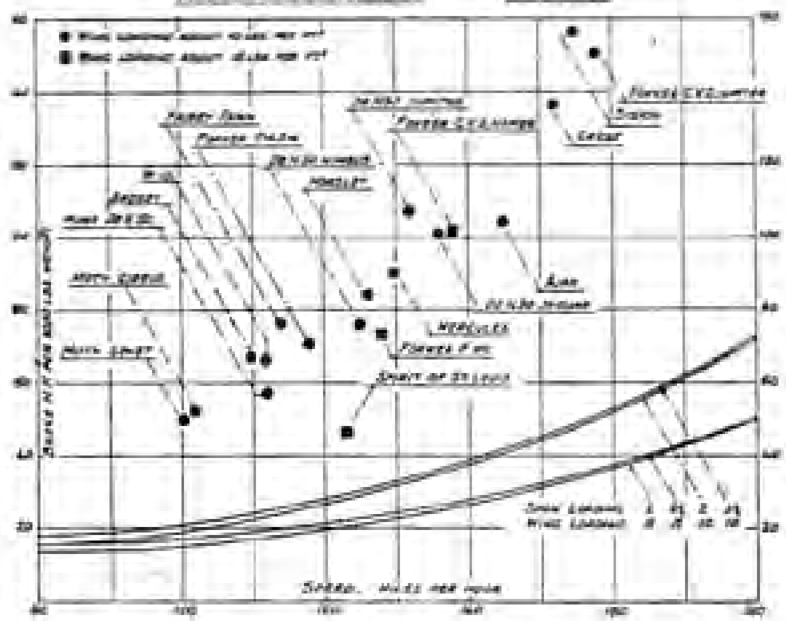
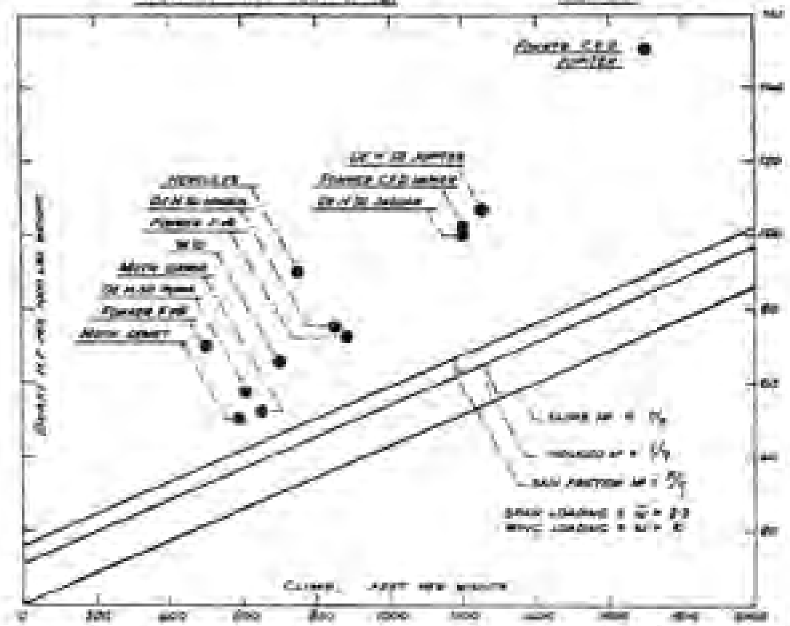


FIG. 4.
COMPARISON BETWEEN REAL AND STREAMLINE AEROPLANE
FIGURES FROM 1927 JANE. CLIMB.



Though improvements in power consumption for climbing are not so great as for top speed, they are still appreciable. The very high ratio between the power apparently expended in turbulence in some cases and the power lost in skin friction, suggests very high turbulence losses, due probably to severe interference between the slipstream, the wing and the body in these circumstances. Possibly the assumed airscrew efficiency of 70 percent is too high for climbing conditions on modern commercial machines, and this may account for some of the heavy losses indicated in the diagram.

RECAPITULATION

The lecture consists of a search for a simple and reasonably sound way of estimating the power which would be required to drive a perfectly streamline aeroplane. The formula finally reached is very simple and easily visualized; the difficulty lies in defending the approximations by which it is reached.

There are two main steps in the argument: the first is that the power required to tow the ideally streamline aeroplane should be merely the sum of the induced and skin friction powers. This idea suggests the use of a drag coefficient, based on the total exposed area, to estimate that part of the power which is not absorbed in the easily calculable induced drag and in climbing.

A comparison of such drag coefficients for a series of simple streamline shapes, with the skin friction coefficient of a thin flat plate, with turbulent flow in the boundary layer, suggests that the latter provided a convenient and safe estimate for the drag coefficient of any good streamline body.

Finally, a figure of 0.002 is shown to give a conservative estimate of the coefficient for wings and bodies at the Reynolds' numbers of full scale flight.

The second step in the argument is that in the streamline aeroplane, interference between the airscrew and the aeroplane, though it may affect the design of the screw suitable to a given engine, should not greatly affect the over-all power absorbed; hence, when estimating the brake horse-power required for the streamline aeroplane, the screw efficiency may be assumed to be that of an isolated screw.

These two conclusions, if accepted, enable the power required by the streamline aeroplane to be very easily calculated. It is submitted that this should always be done in order to keep before the designer and other people concerned the price that is being paid for defective streamlining.

It is not suggested that it is easy to design a streamline aeroplane which will be also a practicable machine, but the immense saving in power, and therefore in fuel consumption, which would apparently follow such a step, forces me to the belief that design will evolve steadily in this direction and that the ultimate aeroplane will be as well streamlined on the whole of its external surfaces as, say, the bottom of a racing yacht or the externals of an albatross. I am fortified in this belief by surveying the animal kingdom. Those birds and fishes which depend on speed for their existence have long since solved the problem. The compromise with structural difficulty was no doubt as difficult for them as for us, but it has ended in the complete triumph of the external form. We can only hope that it will not take us so long to reach this point as, if we are to believe the comparative anatomists, it took them.

DISCUSSION

The PRESIDENT: Professor Melvill Jones had drawn the attention of the Aeronautical Research Committee a year or so ago to the matters dealt with in the paper; as a result, experiments were carried out, which had in substance proved his contentions. He had set up a valuable standard, and when one considered how far present-day aircraft, both military and civil, were below that new standard, one could hardly cavil about its supposed inaccuracies. He had stated in the paper that his conclusions were based upon leaky theoretical arguments, but one felt that when the resistances of present-day aircraft had been reduced by about 50 percent it would be time enough to consider those leaky theoretical arguments with a view to their revision.

Major F. M. GREEN: He thanked Professor Melvill Jones for having set up not merely an ideal, but an ideal which could be expressed in figures. It meant much more to a designer to be able to say that his aeroplane represented a definite advance towards a definite ideal, than to be able to say merely that it was a little better than its predecessor, because in the latter instance he did not know how much further he had to go. Designers were now aware that a great deal of progress must be made, and the paper would stimulate them to make better aeroplanes. The influence of figures was very great, and designers would be able to express the efficiency of their aeroplenes as a percentage of Professor Melvill Jones' ideal, instead of saying merely that a particular aeroplane was fairly well streamlined or was fairly efficient. With regard to the overall figure of 0.002 for the skin friction drag of planes, he said that was half what was at one time called the minimum profile drag (0.004). The figure would vary, however, with different angles, and with greater angles the figure might be greater. With regard to turbulence, that in the air occurred on a very much greater scale than in a wind tunnel, and he asked whether the results obtained with an aeroplane on a day when there was considerable turbulence would be appreciably different from those obtained on a calm day.

Mr. R. McKINNON WOOD, referring to the use of the wind tunnel, said that for many years it had been thought that the ideal condition of test was that the turbulence of the air in the tunnel should be the minimum attainable, but in the last year or so it had dawned on some investigators—it had dawned on him a few months ago—that possibly the wind tunnel which was regarded as the worst was really the best; because, if there were sufficient turbulence in the air the results obtained would be as indicated by the upper curve exhibited by Professor Melvill Jones, and if one extrapolated along such a curve one should make a fairly good prediction of the full-scale value. If the results lay below that curve—somewhere in the indefinite area—one could not say in the least what the full-scale value might be. He hoped that the paper would be of great service by focusing the attention of designers of aircraft more on aerodynamic considerations. He would not reproach designers for giving, as he considered they did, too much attention to weight, to mechanical, structural and other aspects of their work and too little to the aerodynamic aspect. The reason could be found very simply; one could determine the weight of a machine easily and precisely, but it was more difficult to determine drag and to determine what improvements were effected by small alterations. A question which was continually before those engaged in full-scale test was what improvement was effected by replacing, say, a radiator by a different radiator. The gain in speed effected thereby was probably half a mile or one mile per hour, but the man who was making the tests knew that variations in engine power, up and down currents, and other factors might affect speed to a greater extent than would the changing of the radiator. He believed that this work of Professor Melvill Jones, in that is showed how much better a performance we could look forward to, would have the effect of focusing the attention of designers more and more on the problem of reducing drag.

Mr. A. FAGE: He would like to mention two series of experiments with which he had been connected, and which had a direct bearing on Professor Jones' paper. It was well known that on a large circular cylinder there was a certain range of Reynolds' number (VD/v) over which the character of the flow underwent a marked change, and which was accompanied by a large drop in the drag coefficient. He had had occasion to explore the boundary layers for these flows and to determine the breakaway points. Denoting the circular distance from the stagnation point to the breakaway point by "l," and the velocity just outside the boundary layer at the breakaway point by "V," he had found that the range of Reynolds' number (Vl/v), over which the marked change of flow occurred was from 1.1×10^5 to 4.4×10^5. This was exceedingly interesting, because, as mentioned by Professor Jones, Professor Burgers had found that the critical range of (Vl/v) for a flat plate (l is the distance from the leading edge) over which the flow along a flat plate could break down was about the same order, namely, 10^5 to 5.0×10^5. There was, therefore, from this point of view, a resemblance between the flows in the boundary layers along a flat plate and around a cylinder, and we should expect intermediate bodies such as airship forms and aerofoils to exhibit similar tendencies; Professor Jones had shown that they did. Another point of interest arose from some experiments made to measure the minimum drags (at zero incidence) of a family of symmetrical Joukowski sections of different thickness. If one plotted drag coefficient against thickness one was able to predict the drag coefficient for a Joukowski aerofoil of zero thickness (i.e., a flat plate). The values of Reynolds' number at which the predictions were made were in the critical region, and as would be expected the predicted values of drag coefficient lay between those for laminar and turbulent flows along a flat plate; further, these predicted values agreed fairly closely with Geber's values for a flat plate at the same Reynolds' number. These results gave additional support to Professor Jones' conclusions. A great merit of the paper was that attention was focused on the large amount of power frittered away in driving modern aeroplanes, and the paper should prove a great stimulus to all workers in aeronautics.

Mr. C. C. WALKER: Those concerned with aviation were particularly interested to know what chances there were of recovering some of the lost power and how much of the loss was inevitable. When Professor Jones had first published some of his results, he (Mr. Walker) had made calculations in respect of a number of aeroplanes, taking, instead of approximations, the actual "wetted" surface of each aeroplane and the available h.p. (after multiplying the b.h.p. by the propeller efficiency) and had found that an ordinary commercial aeroplane did attain from 60 to 67 percent of the streamline speed. Both the best and the worst examples among his own company's machines happened to be monoplanes. In a little racing machine which did 197 miles per hour with 133 h.p., a little over 80 percent of the streamline speed was attained. That meant that what had to be recovered amounted to the equivalent of three-quarters of a square foot of flat plate presented normally to the wind, and that had to include the undercarriage, all the bracing, the cooling of the engine, the cockpit, and the effects of the slipstream on the body and wings. The machine must have an undercarriage, and at present it must have some form of cockpit; cooling resistance might be reduced somewhat, but there seemed to be very little margin for recovery in a machine of that sort. In the case of a commercial aeroplane, which must have all sorts of other excrescences, it was difficult to foresee how far one could go towards eliminating the resistance. Such a machine had a radial engine sometimes on the front of the body, the cabin had to be ventilated by means of structures like ships' ventilators, sometimes there was a starting engine mounted outside the fuselage, and it did not matter very much what one did in the way of fairing after that. It would be interesting to know what Professor Jones thought about the amount that should be saved. (Laughter.) Referring to the problem of induced drag h.p., he said that Lanchester, in 1914, had shown how to work it out, and from his book, which was published in 1908, one found that he had put it forward to the Physical Society in 1894. At that time he had called it "aerodynamic drag," but it appeared to be the same as "induced drag." He (Mr. Walker) did not know whether or not Prandtl had worked it out earlier than that, but the point was that if "aerodynamic drag" were the same as "induced drag," and nothing more than a change of name was involved, it was rather a pity that Lanchester should be deprived of the credit due to him.

(*Communicated*): There is an aspect of Professor Jones' criterion of performance which is not free from objection. Presumably, by encasing the various necessary excrescences in large streamline boxes, or by so increasing the fuselage in size as to embrace what otherwise would not be embraced, credit could be taken for the extra surface, and a high streamline efficiency shown.

It could then quite easily happen that of two aeroplanes doing the same job, the slower would show a better figure of merit on this basis than the faster.

In one of Professor Jones' earlier papers, he placed on the credit side the area of the floats of a seaplane—as no credit can be taken for the wheels of the corresponding land machine—the situation mentioned above may be said to have already arisen.

The fact remains that the great virtue of this method is its absolute nature.

For purely comparative purposes, it is perhaps preferable to give the highest figure of merit to that aeroplane which carries a given paying load per h.p. fastest in relation to its landing speed or to use some similar basis. The "streamline efficiency" will then show to what extent this machine falls short of the ideal, but to aim directly and only at this in design would not necessarily produce the best aeroplane.

Mr. W. L. COWLEY: It would appear from his remarks that the wind tunnels had given us fictitious results in the past, but he would like to hear the lecturer explain the fact that, in many cases, excellent agreement had been obtained between model results and full-scale. As far as his personal knowledge went, designers had the greatest faith in wind tunnel work. The host of results upon various wing, strut, body and float forms tested in the past as well as on complete models, airscrews, controls, stability, etc., were made use of daily and no marked discrepancies with full-scale, except in occasional cases such as R.A.F. 19, had been reported. It would be strange if these agreements were purely accidental and still more so if they were but imaginary, and that this fact had escaped the great army of workers that had been engaged upon the subject.

Just before attending the lecture he calculated the drag coefficients of an aerofoil, a body and a gloat that were tested some time ago in connection with a certain machine. The points fell almost exactly upon the turbulent boundary layer curve given by Professor Jones. Now Professor Jones assumed that the drag curves for the airship forms ultimately merged into this turbulent boundary layer curve. If it be assumed that the drag curves for the parts mentioned should also follow this curve, the drag coefficient for the full-scale machine should have been approximately 0.0014 instead of about 0.0022 as the model scale effect curves indicated. The full-scale result appeared to be more consistent with the latter than with the former result. It may be argued that the resistance measured were only partially skin friction. If that was so one would expect the difference between 0.0022 and 0.0014 to be form resistance and the curves of drag coefficients of the models against log Vl/v to follow a transition curve rising up rapidly to a turbulent boundary layer curve parallel to the one given, but having the ordinates greater by an amount equal to this difference. In the tunnel tests mentioned, the variation of drag coefficient with log Vl/v however, was very slight. During recent years tests have been made at the National Physical Laboratory upon several models of complete machines, ranging from single-seater single-engined machines to large three-engined machines. Parts were tested alone and in combination and the whole work has been carried out in direct touch with the designers of the machines concerned. In no case has there been the slightest cause to suspect the results. It should be noted that the scale effect was considered so small that the results at the highest values of Vl/v in the tunnel

tests could almost be applied directly to full-scale without correction. Further, the parts were of fairly fine form and the minimum drag coefficients were low. In one machine the model results indicated a higher maximum lift than was expected, but the performance of the machine was found to agree remarkably well with all the model test results.

Reverting to the skin friction curves given by Professor Jones, it would be noticed that the phenomenon of double flow and the transitional region was similar to flow in pipes, but in the latter it was usual to make in the expression log Vl/v equal to the diameter of the pipe and not to a down wind dimension. Now it was implied by Professor Jones that surface area at the rear of the body was affected differently from that near the front on account of the accumulated disturbance produced in the boundary layer by the action of the surface coming before; and the average drag per unit area of the surface of a good streamline form depends only upon the length of the body when the speed, density and viscosity were constant. Now the resistance of a parallel pipe was all skin friction and its drag coefficient appeared to be of the same order as that for the flat plate. One would expect, therefore, similar conditions to prevail. Thus, if for the flat plate, the boundary layer became turbulent at a distance l from the leading edge, then it would be expected that turbulence commenced in a pipe at the same distance from the entry. In other words, turbulence in a pipe commenced at a certain distance from the end irrespective of the diameter, so that, for two pipers a and b using the same fluid, the distances and speeds were related by

$$V_a l_a = V_b l_b.$$

The dimensional theory, however, demanded that, if the conditions were similar, a breakdown in flow would occur at corresponding points when

$$V_a D_a = V_b D_b$$

where D is diameter. In other words, the diameter is all important. It followed, therefore, that the main body of the fluid affected the resistance as well as the surface fluid.

The apparent agreement between the experimental pressure distribution round a body and the pressure distribution calculated from the inviscid fluid theory gave no support to the contention that the integral effect on drag of the flow in the main part of the fluid was negligible. After all the integral of the discrepancies between the pressure distributions, when resolved, was equal to the whole of the drag. One could only conclude that the inviscid fluid theory, in its present form, might be good enough for calculating certain effects, but was useless for any work involving profile drag.

In conclusion, he said that although he regarded the lecture as very interesting and instructive, he thought it was a little dangerous to act upon any deduction made from it at the present stage of our knowledge. It did not justify as yet any loss of faith in wind tunnel work or any change in wind tunnel technique, although he agreed that further research on the lines given by the lecturer might be necessary.

Mr. PYE: Referring to Professor Jones' curves showing the b.h.p. per 1,000 lbs. weight in respect of certain machines for level flight at various speeds, he suggested that one or two spots representing the performance of machines which competed in the Schneider Trophy contest might be added, because the drag had been very materially reduced in the case of those machines. If such information were given, one would be able to make extraordinarily interesting comparisons, because the drag of those machines must be very much nearer to Professor Jones' theoretical limit than it was for any of the machines he had shown.

Dr. DOUGLAS: No doubt in cases where the full-scale flow was turbulent increase of turbulence produced a similar effect to that obtained by increasing Reynolds' number, and a better idea of full-scale value would be obtained if models were tested in a very turbulent tunnel, as suggested by Mr. McKinnon Wood. It was quite possible, however, that on objects, such as struts, the full-scale flow might actually be laminar.

Professor Jones had shown that existing aeroplanes have very much more drag than they need have. Machines were defective both in form and in surface. The latter defect was particularly noticeable on the Schneider Trophy winner displayed in London, and he had been shocked to notice the rivet heads and other excrescences on a machine designed to have the lowest possible drag. The components of a machine of this type have drags little greater than that given by thin plates of similar surface area, and when this standard had been reached it must pay to keep the surface clean. He hoped that Professor Jones' remarks would lead to a better appreciation of the need for reducing drag.

Mr. IRVING: With regard to the point raised by Dr. Douglas, Professor Jones, in arriving at his overall figure for the full-scale drag coefficient, had assumed that one traveled along the upper curve. It was just conceivable, however, that with certain shapes one might possibly keep to the lower curve. The departures from the lower curve had in most cases occurred in wind tunnel tests in which the general flow was turbulent, although in one case, in which tests were made by towing a model in still water, there was also a departure from the lower curve in the direction of the upper curve. Mr. Irving also referred to some tests carried out at the National Physical Laboratory in 1919, in which Mr. Ower and he had measured the skin friction of flat plates of various thicknesses on the balance in the wind tunnel, and extrapolated from different thicknesses down to no thickness. The values of the skin friction coefficient obtained for different sizes of plate at different wind speeds were plotted against *VL*; for the lower values of *VL* the points came actually on the lower curve, but as *VL* increased there was a gradual departure, and at the higher values of *VL* the curve crossed the region between Professor Jones' two curves and almost touched the upper curve. He mentioned this fact because he understood Professor

Jones' lower curve was based on measurements of flow near flat planes and not on overall measurements of the skin friction on a balance.

Mr. CAPON: On a previous occasion when Professor Jones had lectured to the Society, he had uttered the dictum that no research could be considered to be competed until it had been reduced to a rule of thumb. One might add that no research should be begun until there was a definite formula to work to, and the great value of Professor Jones' work was that he did provide such a formula. It had been known for a long time that great savings in drag could be effected; probably the reason why so little had been done was that there were a great many small contributions, and nobody felt really convinced about any of them. For example, it had been known for several years that a saving in drag on models could be effected by cowling the cylinders of radial engines. Why were radial engines not cowled? Probably because designers quite reasonably thought that there might be a scale effect on drag or cooling, or both. Again, we really did not know how much the sum of these contributions would amount to. Professor Jones had given a figure of merit to indicate what could be expected, and although there might be some quibbles as to what exactly that figure amounted to—questions such as airscrew interference, and so on, were involved—it was nevertheless a sufficiently definite figure. Professor Jones was to be congratulated, not only upon having produced it, but on having met so successfully, as it appeared, the objections raised by the scientists based on the uncertainty as to the skin friction coefficient to be adopted. His investigations in this direction had apparently led to results of great interest in other connections.

Mr. Scott HALL: It seemed to him that Professor Jones' assumptions with regard to the working of the screw in close proximity to the body were of a very grave nature, for in effect he had assumed that the loss due to increased drag of the body in the slipstream behind the screw was balanced entirely by the extra efficiency with which that screw worked, due to the body behind it. He would have liked to have heard more from Professor Jones on that matter. With regard to the curves showing the performance of various aircraft, he said that the point relating to the "Spirit of St. Louis" was situated a considerable distance from those relating to other machines. The "Spirit of St. Louis" was the only machine dealt with which had no open cockpit, nor openings of any kind, and the fact seemed to indicate that the openings on a streamline body were of very serious practical importance, more so even than had been hitherto supposed.

Mr. BRAMSON: He could not resist the temptation to point out one analogy; in all probability Professor Jones' treatment of the subject of the streamline aeroplane would be regarded as holding the same position in relation to aerodynamics as, for instance, the Carnot cycle held in relation to thermo-dynamics. Professor Jones had put forward an ideal beyond which, without the discovery of new principles, one could not hope to get, and he had given designers a standard by which the qualities of aeroplanes could be judged, a standard which has hitherto been lacking. He asked Professor Jones if he could give, there and then, an actual numerical example of what an aeroplane of reasonable weight, horse-power, and so forth, must do to achieve the ideal. With regard to the curious fact mentioned by one of the speakers, that in the course of experiments he had found that the lift coefficient began to drop at much lower values at low values of Reynolds' number than at high values, Mr. Bramson asked if it were not possible that that was due to the departure from the boundary layer occurring sooner at low speeds, due to the relatively greater effect of the "adverse pressure gradient" near the trailing edge, than at high speeds. With regard to circulation, he said he had heard it stated that the "equivalent circulation" round an aerofoil would never start in the ideal fluid. He asked if that were so, and whether the idea of a circulation around an aerofoil was at all a sensible one. It was difficult to see how, if there were anything which could properly be called circulation, it could go round a sharp trailing edge, for instance, in any intelligible manner.

George H. DOWTY (*contributed*): Following Mr. North's recent lecture, it is possible to compare the ideal aeroplane with present day standards and appreciate the vast improvements which have yet to be made. The lecturer stated that progress towards the ideal streamline aeroplane has been slow and points out that the tendency to belittle minor gains is a wrong policy. The design of racing machines has produced a very clean type of aircraft, and here the designer is prepared to sacrifice practically everything in producing an ideal aerodynamic form. When speed is not the ruling factor, the question of maximum aerodynamic efficiency often takes second place to other considerations and justification is claimed for this because the increase in performance is small. If the existing gap between present designs and the streamline aeroplane is to be bridged then improvements, however small, cannot be neglected. One of the chief items of parasitic drag, and one to which Professor Melvill Jones has drawn attention in R. & M. No. 1115, is the undercarriage. The writer has been engaged on work in this direction for some considerable time, and during the last three or four years has confined his attention to methods for reducing the drag of this unit. Investigations have led to the same conclusions as those given by Mr. North, that the retractable undercarriage is not desirable because of the complicated retracting mechanism and the extra weight involved. Mr. North suggested the use of mechanical launching gear, and from his remarks, it is understood, a form of ski landing gear. This form of undercarriage is open to strong objections because it limits the operation of aircraft to stations equipped with launching gear and such machines would be at a great disadvantage in the case of a forced landing. There is also the question of ground maneuverability and it does not seem reasonable to expect any sacrifice in this direction.

There is fortunately some scope for cleaning up existing forms of undercarriages, and it is on this basis that the writer has been working. The shock-absorbing member is usually located in the slipstream and it will be found that this member gives a resistance at least five times as great as a steel streamline tube carrying the same load. Further, the existing types of connections are very poor, the strut sections being broken at their points of juncture with the body. Such a condition must cause mutilation of the air flow at these points and give rise to additional drag (now termed interference). In order that these disadvantages can be obviated, the writer has designed several types of landing gears where the structure is rigid and the shock-absorbing mechanism lies within or adjacent to the wheel. The wheel with internal springing consists of an oil dashpot shock absorber, compression rubber springing and wheel brakes. This design of wheel has been in existence for two years, but it has only recently been favorably reviewed by wheel manufacturers. The Curtiss Company, of America, have produced a similar type of wheel, and they consider this undercarriage an attractive proposition. The shock-absorbing capacity of these wheels is better than that usually obtained with an oleo leg, because there are no limitations due to angular movements of the structural members. Conservative detail estimates show that the undercarriage drag can be reduced by 25 percent and the weight by 15 percent.

G. T. R. HILL (*contributed*): This paper sets up for the first time a standard by which the performance of any aeroplane may be judged. Considering how far all designers are below that ideal performance, or "Jones performance," as it should now be called, serious objection cannot yet be brought against the standard on account of its being built up with the aid of what Professor Jones calls "a leaky theoretical argument." When the present drag of our military and commercial aeroplanes has been halved, it will be time to set the standard more securely on its base with the aid of aerodynamic knowledge which will then be available.

There appears no doubt that the rate of progress towards the streamline aeroplane is greater than that achieved by Nature in bygone ages in the design of birds and fishes, yet any elation which may be felt over this will be considerably moderated by turning back some eighteen years to 1911, to the remarkable designs of M. Nieuport. There is a good description of these little aeroplanes in the old *Aero* of June, 1911; one of them which flew in the Gordon-Bennett Race of that year was fitted with an engine of only 28 h.p. nominally and with a wing area of about 170 square feet, it is reported to have attained a speed of 75 m.p.h. Looking at the photograph of the aeroplane it might be asked how could its resistance be brought down to the modern standard? It could be said at once that streamline wires could be fitted instead of round cables, and having said this, nearly all has been said.

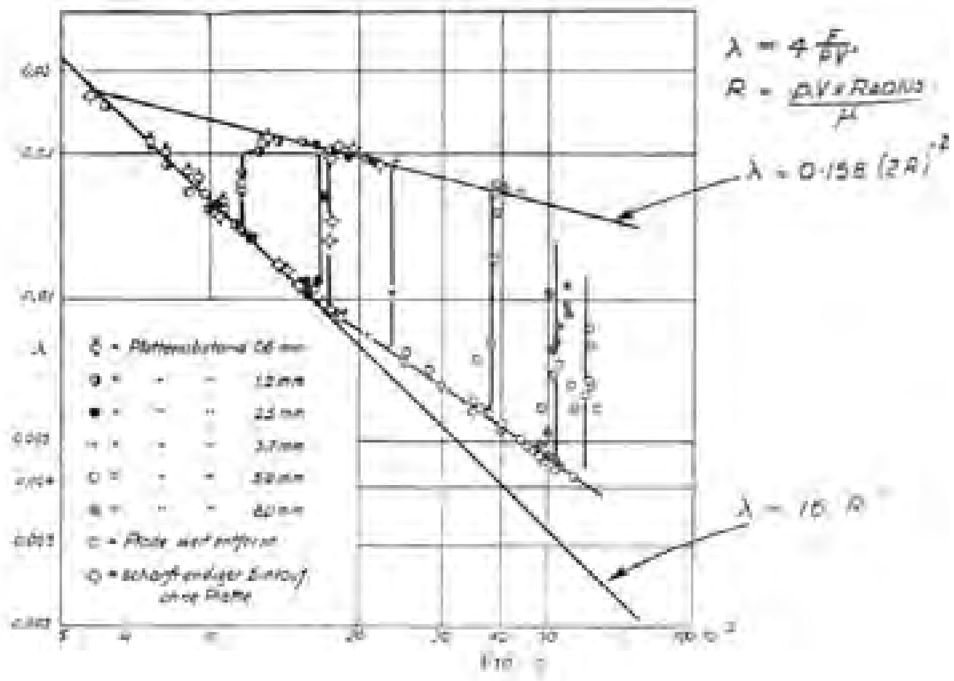
Turning to the future, the fact cannot be ignored that as the resistance due to some parts of the aeroplane is reduced, the resistance of those parts left unaltered calls more and more loudly for attention. For this reason, it is difficult to foresee clearly the future of the air-cooled engine, in which perhaps four-fifths of the resistance is due to the turbulent air flow around the cylinders and one-fifth to the skin friction which necessarily accompanies the cooling effect required. Already the air-cooled engine is seen to be at a disadvantage in high speed racing aircraft, and in the usual course of events, characteristics of the racing craft of today are exhibited by the general purposes craft of tomorrow. How long will that tomorrow be in coming?

The "Jones ideal performance" stands out as one of the beacons towards which the designer must steer his painful way through the fog of detail work which necessarily surrounds him.

S. J. DAVIES and C. M. WHYTE: (*contributed*): The writers would like to thank Professor Melvill Jones for the excellent way in which he made a highly specialized subject both intelligible and interesting to those who, like themselves, are not conversant with the subject. They were particularly struck by the number of points arising which threw light upon matters which had been obscure to them in an allied subject. It is possible that the following remarks, based on the study of flow through pipes, may in their turn be of interest to aeronautical engineers.

One of the most striking points brought out by the lecturer was that the majority of wind tunnel tests lie within the transfer region between laminar and turbulent flow. This fact must add greatly to the difficulty of interpreting the results, and it is remarkable that reliable figures have been obtained at all. For many years the similar transfer region in connection with pipe flow also gave rise to difficulty and led to much confusion of thought. And in fact it is only within the last few years that any certainty has been reached. The curves from flow in pipes are similar to those given in the lecture and the results of the former may help in interpreting the latter. The work of L. Schiller has played a large part in demonstrating that the amount of turbulence in the fluid as it enters the pipe determines the value of the Reynolds' number at which the transfer takes place. Some of his results, which do not appear to have been published in this country, are shown in Fig. 5. The wide difference between the values of the Reynolds' numbers of the figure and those mentioned in the lecture are due to the fact that in the figure the Reynolds' number is that formed by the linear dimension of the cross section of the stream, while in the lecture the overall length of the body is used. The value of $F/\rho v^2$ in the figure is arrived at by the method used in the lecture, but it should be noted that it does not include the resistance at or near the entrance to the pipe. The various curves were obtained with increasing entrant disturbances, and show that the greater the turbulence at the inlet the smaller is the Reynolds' number at which the transfer takes place. There is, however, a limiting value, and no amount of disturbance will cause the transfer to take place below R = 1160. The relevant conclusion to be drawn from this is that, unless a high degree of turbulence is provided by artificial means, the results will neither be free from the influence of the unknown state of the fluid at entry, nor be consistent among themselves. With artificial turbulence it is possible to obtain points on the turbulent line for flows only slightly in excess of R = 1160. Further tests of Schiller show also that a moderate degree of roughness of the pipe does not influence the transfer.

The vertical transfer curves shown in the figure were obtained only when a length of pipe at least equal to 130 diameters is interposed between the entrance and the testing length. Under other conditions the transfer curve is less definite and becomes rounded in form and very similar to the dotted curve shown by the



lecturer. The value of the Reynolds' number at transfer, however, remains approximately unaltered. The lecturer's method of calculating the form and position of the transfer curve is interesting, but he does not indicate the grounds for his implied assumption that the Reynolds' number, formed by the distance from the leading edge to the breakdown point, is constant during the transfer period. At first sight it might be expected to decrease progressively as the transfer curve is traced out in an upward direction.

The lecturer gave the impression that curvature tends to cause the boundary layer to become turbulent sooner. It would be interesting to know if there is any experimental evidence supporting this view. Curvature in pipe flow, provided that it is slight, appears to be without influence, but, as is shown by some tests about to be published, flow through a curved pipe, of which the radius of curvature is 15 times the pipe radius, is considerably more stable than in a straight pipe, and the transfer does not take place until about four times the Reynolds' number at which it would occur in a straight pipe.

The necessity for carefully regulated conditions when determining the resistance of complicated bodies is well illustrated by tests of a series of very rough pipes carried out under somewhat indefinite entry conditions. These tests show certain striking anomalies. Two pipes, one definitely rougher than the other, were reversed in their resistances when tested over a certain range of Reynolds' number near the transfer region. This behavior was characteristic, in this region, of the majority of the pipes tested, and persisted long after the flow had ceased to be streamline. The conclusion may be drawn that no reliance can be placed on the relative magnitudes of the observed resistances of two models of a complicated form unless the comparison is made at Reynolds' numbers many times as great as those of the transfer region, or, alternatively, unless the state of the fluid is completely known.

In connection with pipe flow the writers have found logarithmic plotting along both axes to be more convenient than the usual method of plotting $F/\rho v^2$ on log. *R*. Equations (1) and (2) or the lecture would then become straight lines sloping downwards. Plotted as they are, there is a false impression given, owing to the upward concave form of the curves, that they are tending to some constant value of $F/\rho v^2$ at high values of *R*. There is a further advantage of logarithmic plotting in that inaccuracies of test data of equal percentage are consistently shown at all parts of the diagram by equal displacements, so that undue importance is not attached to isolated tests which happen to fall on parts of the diagram where the scale may be relatively close.

Mr. SIMMONS (*communicated*): Professor Jones has tonight expounded a theory based on the assumption that the skin friction resistance of bodies of fine form follows one law at low values of Reynolds' number and another law at high values. The underlying basis of the theory recalls the methods employed by naval architects for predicting the resistance of ships, founded on the skin friction data of flat plates. He would like to call attention to some results published some years ago by Mr. Baker which he thought pertinent to the present discussion, and lent support to the new theory. These were shown graphically in Fig. 6. It would be observed that below a Reynolds' number of about 4 X 10^6 large discrepancies existed between individual observations made at the same value of VL/v; although at higher numbers the results were more generally consistent. Commenting on the measurements made in air and water on similar models, he thought it was significant that the drag coefficient measured in the turbulent flow of the wind tunnel was higher than the figure obtained in the water experiments. If the initial turbulence in the free stream was a factor which influenced the flow in the boundary and thereby the drag of the model, then its effect appeared to be most marked below

$$VL/v = 4 \text{ X } 10^6.$$

He was, however, a little doubtful whether a stable regime of flow independent of initial disturbances was completely established at this value of Reynolds' number, since the Gottingen results exhibited by the lecturer, and shown by the dotted curve were in excess of the figures derived from tests on the largest planks. Finally, he referred to the similarity existing between the results for bodies of fine form in the transitional region and those obtained with pipes; and he thought it must be more than coincidence that the turbulence curve for flat plates could also be taken to represent the resistance of pipes at high velocities.

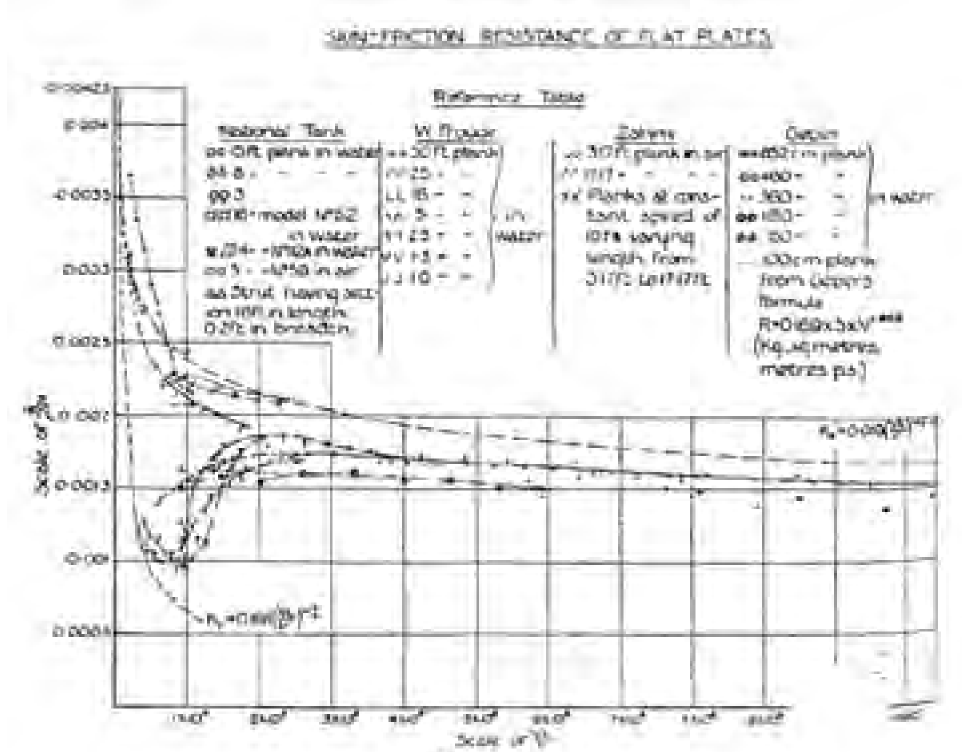


FIG. 6.

REPLY TO DISCUSSION

The discussion has been long and interesting, and I wish first to thank all those who have taken part in it.

In reply to *Major F. M. Green*, the turbulence in the wind tunnel which causes early break up of the boundary layer is of a peculiar kind with a very small scale. I imagine that the scale of the turbulence in the open air is too large to have any influence on the boundary layer and I should not therefore expect differences on different days due to this cause. This opinion is not based on definite evidence. The figure given for skin friction drag coefficient is half the minimum profile drag coefficient, because the area involved in the former is that of both upper and lower surfaces of the planes, whereas the area involved in the latter is one surface only.

With reference to *Mr. R. McKinnon Wood's* remarks, I agree that it is probably that wind tunnels which have been deliberately given a small scale turbulence will be used in the future, but this matter has not yet been clearly thought out so far as I am aware.

Mr. Fage's statement that the Reynolds' number at which the boundary layer breaks up on a circular cylinder is of the same order as that for a flat plate, is most interesting. The effect of curvature on the point of break up of the layer is a matter which requires much further research. I shall be interested to see his results for thin Joukowski sections when they become available.

Mr. Walker's estimate that the speed of his aeroplanes lies between 60 and 67 percent of the streamline speed agrees fairly well with my rougher estimates. Since power varies approximately as V^3, the realized speed of 70 percent of the streamline speed corresponds to an expenditure of about three times the streamline power for a given speed. I am obliged to Mr. Walker for emphasizing the fact that the ratio of actual to streamline power must not be regarded as an overall figure of excellence for the aeroplane. I had a warning to this effect in an early draft of the paper, but inadvertently omitted it in the final text. What the factor does show is how much power is being expended through defective streamlining. I agree with his remarks about Mr. Lanchester and the induced drag theory; all the same, it was Prandtl who threw it into its present practical form.

Mr. W. L. Cowley finds it hard to reconcile my lecture with the available results which have been given by wind tunnels in the past. I do not understand his difficulty. The majority of results which have been used extensively have not been for perfect streamline bodies such as I am discussing. Also it must be remembered that the majority of full-scale results refer only to total brake horse-power, and that scale effects on the various parts may be confused with variation in airscrew efficiency or interference between airscrew and body. Only in a few instances, where research methods have been applied on the full-scale, do we know the drag independent of the airscrew, and even then the whole drag and not that of the parts separately.

There is one way in which apparent agreement can be obtained in spite of the existence of large scale effects; thus, the airship R.101 when tested on the model at a Reynolds' number of 2 X 10^6 gave k_F = 0.0010. This figure was almost exactly that which we should now predict for the full-scale with a wholly turbulent boundary layer at a Reynolds' number of 3 X 10^8. The apparent agreement here is entirely fortuitous, for at intermediate Reynolds' numbers the coefficient is much higher, as would be expected from the theory which I have advanced.

In Mr. Cowley's remarks upon pipes he appears to have forgotten that in a long pipe the boundary layer extends to the middle of the pipe. There is a relationship between the flow in pipes and the flow for a flat plate, but it is not so simple as Mr. Cowley appears to assume; it has been worked out by Professors Prandtl and Von Kármán (See my reply to Messrs. S. J. Davis and C. M. White for references).

I cannot understand Mr. Cowley's statement that the pressure distributions when resolved are equal to the whole drag; so far as I am aware this is incorrect for streamline shapes.

So far from the conclusions of my paper leading to a loss of faith in wind tunnel work, they will, I hope, increase its usefulness through removing some of the anomalies which have been well known for some years.

In answer to *Mr. Pye*, I have not included the Schneider Cup racers because, being seaplanes with floats, they are not comparable with the other aeroplanes in Fig. 3. It would be interesting to see them worked out accurately by someone in the Air Ministry who has the facts at his fingers' ends.

I agree with Mr. Douglas as to the importance of a clean external surface, but it is just possible that small projections may have less influence on the full-scale, where the boundary layer is probably turbulent in any case, than on the wind tunnel model, where the projections may convert a laminar into a turbulent layer.

Mr. Irving's suggestion that the boundary layer may continue laminar to higher Reynolds' numbers in respect of some shapes is very significant. We had already given some thought to this matter at Cambridge, but I left it out of this paper as being too speculative. This, I hope, will be the subject of future research.

His remarks about his experiments on skin friction in 1919, in which the skin friction coefficient fell on the curve for laminar boundary layer, are most interesting and I should like the reference. Dr. Stanton has also done experiments on the skin friction on the thin rings similar to napkin rings, and has found the same agreement. There is no doubt that when the layer is laminar the agreement of the skin friction with Blasius' solution is thoroughly established.

Mr. Capon has clearly emphasized the main reason for the lecture, which was to show that an estimated sum of all conceivable drag reductions is so large as to be worth going for, even if the separate contributions may in themselves appear small.

Mr. Scott Hall has doubts about my screw efficiency theory. I am not surprised at this, as it is at present the weakest part of the paper and is as yet unsupported by crucial experiment. The main conclusions of the paper, however, stand without it. The assumption certainly is, as he says, that the interference of screw on body balances the effect of body on screw, except for the increased power loss due to skin friction in the slipstream. Stated in this way, the assumption is certainly startling, but if my argument is carefully followed I do not see how the conclusion can be avoided. It is to be emphasized, however, that the argument applies only to ideally streamlined machines as defined in the paper.

In reply to *Mr. Branson*, the suggestions in this paper cannot be considered as on the same plane as the Carnot theory of heat engines, but the practical outcome is similar—the provision of an ideal towards which to work. Whereas, however, the Carnot cycle is a precise theorem, my paper is more in the nature of an exercise in approximations.

I agree with *Mr. Dowty* that the landing gear is probably the greatest difficulty in the way of perfect streamlining, and I am interested to hear of his efforts to tackle the problem.

In answer to *Captain Geoffrey Hill*, the air-cooled engine is at present a distinct stumbling block to progress in streamlining, but I am not sure that it will always be so. Recent experiments in cylinder cowling are most encouraging and it should be remembered that, since the temperature of the cylinder is high, scientifically devised air-cooling systems, such as we have not yet got, may give rise to a relatively low resistance.

In reply to *Messrs. S. J. Davies* and *C. M. Whyte*, the relation between resistances to flow in pipes and the Reynolds' number formed from the pipe diameter was

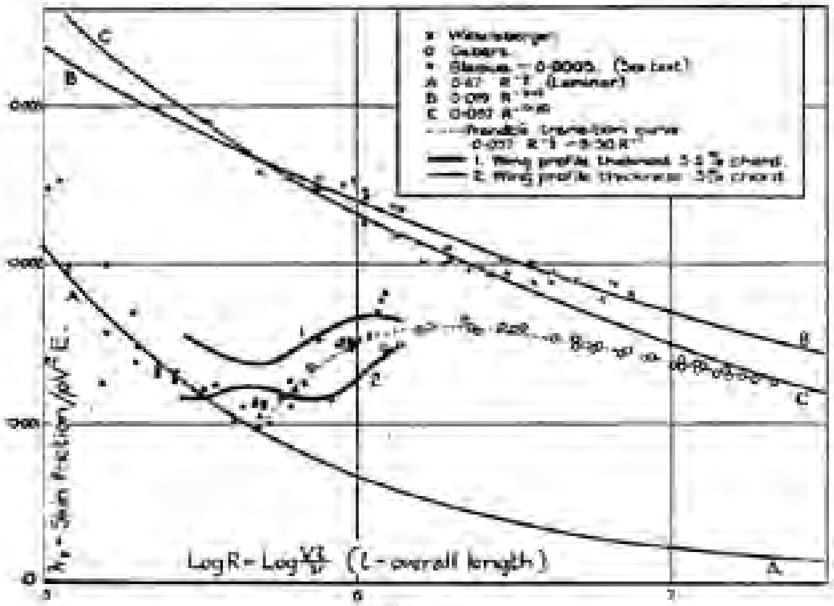


Fig. 7.

Prof. Prandtl's Transition Curve for Skin Friction on a Flat Plate.

Re-plotted from Reports 1 and 7, Ergebnisse der Aerodynamischen Versuchsanstalt Zu Göttingen III. Lieferung.

thoroughly worked out by Stanton and Pannel for both water and air, and a curve showing the relation of skin friction per unit area/ρV^2 to the Reynolds' number, for a very wide range including the critical region, was published in 1914.

The comparison between flow in pipes and the boundary layer on a flat plate has been thoroughly examined by Von Kármán, of Aachen (Abhandlunger aus dem Aerodynamischen Institut an der Technischen Hochschule, Aachen 1921) and by Prandtl (Ergebnisse der Aerodynamischen Versuchsanstalt zu Gottingen, 1927, Vol III). Both these papers have been translated into English, the former by the A.R.C. as Report T. 2219 and the latter by the Air Ministry. In these papers a law, k_F varies as $(l\upsilon/v)^{-0.20}$ for flat plates with turbulent boundary layer is, with certain plausible assumptions, deduced from the law k_F varies as $(l\upsilon/v)^{-0.25}$ which is found to hold experimentally for pipes above the critical Reynolds' number. Fig. 7 is a composite diagram taken from two reports in the Gottingen publication mentioned above and roughly re-plotted to conform with Figs. 1 and 2. The curve $k_F = 0.037(l\upsilon/v)^{-0.2}$ adopted by Prandtl, though it fits Gebers' points at the higher Reynolds' numbers, falls lower than the results from Froude and the N.P.L. given in Fig. 7. Hence for practical purposes I prefer his original formula

$k_F = 0.019(l\upsilon/v)^{-0.15}$, which gives a conservative estimate of skin friction drag for flat plates over the whole known range.

I do not understand the writer's difficulties with the transition curves. The only

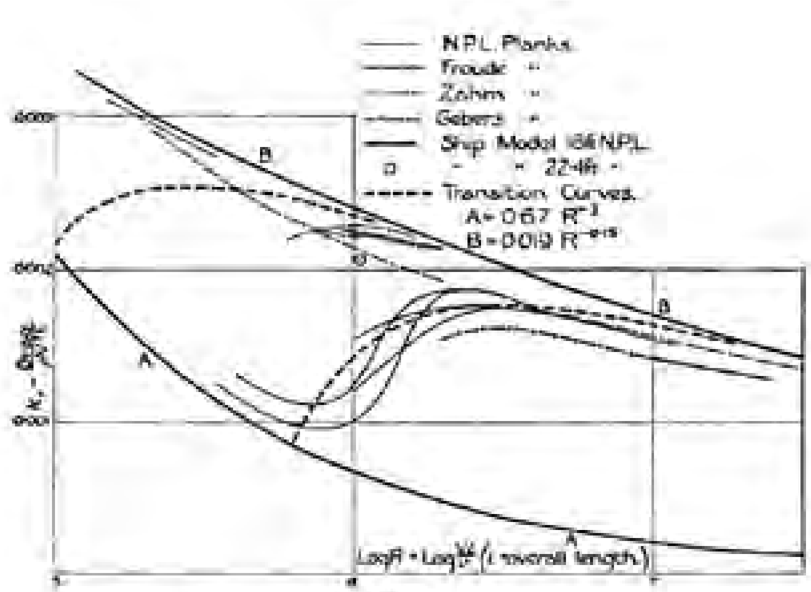


Fig. 8

Skin Friction Experiments in N.P.L. Naval Tank.

N.P.L. Collected Researches, Vol. XIII, 1916, pp. 103-139, from Plate I Re-plotted.

assumption regarding the point of breakdown from laminar to turbulent flow is that the point is uninfluenced by the portions of the plate behind it. This assumption must surely be correct for the flat plate; whether or not it is correct for the curved surfaces is shown by comparing the curves for k_F against R for the various streamline bodies with the theoretical transition curves for the flat plate.

To avoid unduly lengthening the paper, I did not give my methods of calculating the transition curves. The difficulty lies in deciding to what extent the laminar layer on the front part of the plate will influence the turbulent layer behind it. If it is assumed not to influence the turbulent layer at all, an upper limit is given for the coefficient of the average drag of the whole plate. If, on the other hand, it is assumed that the turbulent part in the rear behaves as though the whole layer were turbulent, a lower limit is given to the average coefficient. In forming the transition curves in Figs. 1 and 2 I worked out curves on both the above assumptions and took the mean between them. Prandtl, on the other hand, worked on the second assumption only in producing his transition reproduced curve in Fig. 7. The difference between my method and Prandtl's is not great, and I do not know which is nearer the truth.

The question of the effect of curvature on the stability of the boundary layer is still obscure, and will, I hope, shortly be the subject of experimental investigation.

I am aware of the advantage of complete logarithmic plotting, but I thought that on the balance the system which I adopted would be more easily explained to members of the audience who might not be very familiar with logarithmic plotting. The use of logarithms in connection with Reynold's numbers is easily explained as a convenient device for compressing a large range of numbers into a small space.

I am much obliged to *Mr. Simmons* for drawing attention to the available data contained in the figure accompanying his remarks. My attention had been drawn to the existence of this data by Sir Richard Glazebrook a day or two before the lecture, but I had not had an opportunity to consider it. Since the lecture I have secured Mr. Baker's original paper and have re-plotted the information on the same plan as Figs. 1 and 2 though on a larger scale, and the resulting diagram is reproduced herewith (Fig. 8). It is evident, both from Mr. Simmons' figure and from mine, that the boundary layer in the N.P.L. tank experiments was breaking up at about R = 6 X 10^5, and it is clear from my figure that at high Reynolds' numbers the curves of Froude and the N.P.L., which continue up to 4 X 10^7, lie within some 5 percent of the equation $k_F = 0.019(lv/\nu)^{-0.15}$. These latter data bring our knowledge of the skin friction coefficient of a flat plate up to a Reynolds' number of 4 X 10^7, that is to say, well into the region of Reynolds' numbers represented by full-scale aeroplane wings and bodies. They show that the formula which I used in my lecture for the skin friction is on the safe side in this region.

Document 3-8

Excerpts from Fred E. Weick and James R. Hansen, *From the Ground Up: The Autobiography of an Aeronautical Engineer* (Washington and London: Smithsonian Institution Press, 1988), pp. 49-61, 66-68, 72.

The organizing thinker and team leader of the NACA's original cowling program at Langley was Fred E. Weick, one of the most remarkable aeronautical engineers in the history of American aeronautics. Born near Chicago in 1899, Weick (pronounced Wyke) developed an avid interest in aviation by the age of 12, attending air meets at nearby Cicero Field and entering model airplane competitions. Upon graduation from the University of Illinois in 1922, he began his professional career as a draftsman with the original U.S. Air Mail Service. After a short stay with the Yackey Aircraft Company (during which time he worked in a converted beer hall in Maywood, Illinois, transforming war-surplus Breguet biplanes into "Yackey Transports"), he started a job with the U.S. Navy Bureau of Aeronautics in Washington, D.C., where, within a matter of months, the NACA's director of research, George W. Lewis (1882-1948), personally recruited him for important work to be done at Langley, some 120 miles to the southeast. (The NACA's Washington office was located in an adjacent wing of the Navy Building, thus facilitating close relations between the NACA and the navy.) Weick arrived at Langley in November 1925 just in time to take over the design and construction of the new Propeller Research Tunnel (PRT)—the job Lewis had specifically asked him to do.

The following is a series of excerpts from Fred Weick's 1988 autobiography in which he recalled the construction of the PRT and the origins of the cowling research program more than 60 years earlier. Readers will find that Weick's recall of these events from long ago was amazing. Perhaps even more amazing was that Weick was such a marvelously clear thinker when it came to technology and how his approach to everything related to engineering was strictly rational. One of Weick's friends and colleagues at NACA Langley in the 1930s (Weick worked at Langley from late 1925 to 1929, then moved to a job with Hamilton Aero Manufacturing Company in Milwaukee, Wisconsin, maker of adjustable aluminum-alloy propellers and steel hubs for both military and commercial aircraft, and then returned to work at Langley from 1930 to 1936) was the distinguished aerodynamicist Robert T. Jones, who became one of the fathers of the swept wing. In the foreword to Weick's autobiography, Jones paid tribute to this former NACA associate specifically in terms of his role in "reinventing the airplane." "Working with Fred," Jones wrote, "I had the feeling that the airplane was being reinvented, as it indeed was. Never mind that the general form and arrangement of the airplane had been well established for many years, Fred felt strongly that every function of the airplane needed to be studied from a logical point of view, without prejudice, so that designers and operators could make whatever changes were necessary to improve those functions" (p. viii).

Following his pioneering work on the NACA cowling, Weick went on to make many other significant contributions to the advancement of aeronautical technology, including development of the steerable tricycle landing gear, the conventional gear used today—even for the Space Shuttle. His most widely recognized achievement, the Ercoupe, has been the favorite airplane of thousands of private flyers since its first production model came out in 1940. And his revolutionary AG-1 and Piper Pawnee set life-saving standards of lasting benefit to both the agricultural airplane and general aviation industries. His autobiography from which these excerpts are taken tells his entire life story in fascinating detail, from his days with the barnstormers, through his navy and NACA years, to his many years in manufacturing for the Engineering Research Corporation (ERCO) and Piper.

Document 3-8, Excerpts from Fred E. Weick and James R. Hansen,* From the Ground Up: The Autobiography of an Aeronautical Engineer *(Washington and London: Smithsonian Institution Press, 1988), pp. 49-61, 66-68, 72.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS, LANGLEY FIELD

The distance by air from Washington, D.C., to Hampton, Virginia, the town nearest Langley Field, is only about 120 miles, but by road through Richmond it is about 175 miles. In the 1920s these roads were surfaced with gravel and often badly rutted; smooth ribbons of concrete were not to be found in rural Virginia. In our Model T roadster, packed to overflowing, it took us all day to make the trip.

The engineer in charge of the Langley Memorial Aeronautical Laboratory at that time, a Californian by the name of Leigh M. Griffith, appeared unhappy with the idea that I had been placed under him from above; in fact, Griffith must have been generally unhappy with his situation at Langley, for he left within the month. He was replaced by Henry J. E. Reid, an electrical engineer who had been in charge of the laboratory's instrumentation. Reid remained the engineer-in-charge until he retired from the National Aeronautics and Space Administration (NASA) in 1960. The lab had a flight research division headed by test pilot Thomas Carroll, a power plant division headed by Carlton Kemper, and two wind tunnel sections, one, the 5-foot or atmospheric wind tunnel (AWT) section headed by Elliott G. Reid, and, the other, the variable-density tunnel (VDT) section headed by George J. Higgins. There was also an instrument shop, model shop, technical service department, and a clerical and property office headed by Edward R. ("Ray") Sharp. The new 20-foot propeller research tunnel (PRT) was being constructed under the supervision of Elton W. Miller, a mechanical engineer who had previously been in charge of the

construction of the variable-density tunnel. I was placed under Miller until the tunnel was ready for operation.

By the time I started at Langley, the outer shell of the new tunnel had been completed but work on the entrance and exit cones and guide vanes was still going on. The tunnel, which had been laid out by Dr. Max Munk in Washington, was of the open-throat type then most suitable for testing propellers. My first job was to design and get constructed a balance arrangement that measured the aerodynamic forces on the model and the model's reaction to them. This balance had to support an airplane fuselage, complete with engine and propeller, 25 feet above the floor in the center of the tunnel's 20-foot-in-diameter airstream. All of the pertinent forces, such as drag, thrust, and moments, were to be measured down below by four small and simple beam scales.

Since 1921 Dr. Munk had been holed up in a little office at NACA headquarters in Washington, where he had been turning out excellent theoretical work. Munk had studied under Ludwig Prandtl at the University of Gottingen in Germany and had been brought to this country by the NACA in 1921. His entry into this country required two presidential orders: one to get a former enemy into the country, and another to get him a job in the government. And I guessed this helped him to appreciate his importance.

Without question, Munk was a genius, and, without question, he was a difficult person to work with. In early 1926 he decided on his own that, since Langley laboratory was where all the real action was taking place, that was where he should be. NACA headquarters must have agreed, because it made him the lab's chief of aerodynamics; this put him in charge of the flight research division and the two wind tunnel sections. My boss, Elton Miller, now reported to Munk, and all of my work ultimately had to be approved by him. I had known Munk in Washington and had great respect for his abilities. On the other hand, I did not want my balance design turned down at the last minute; so I had taken the pain to take each detail of design, mostly on cross-section paper, up to Munk to get his approval, and I got his initials on every single one of them. This, I thought, would certainly assure his final approval.

The movable parts of the balance supporting the airplane were supported by a structural steel framework about 12-feet high, 12-feet wide, and 16-feet long. In place of adjustable cables, steel angles ¼ by 2 ½ by 2 ½ inches provided the diagonal bracing.

A couple of days before we expected to try out the balance using a little Sperry Messenger airplane with its 60-horsepower engine running, Munk made an unannounced visit to the PRT building. Just as he walked into the bare-walled 50-foot cubicle that housed the test section, a loud horn squawked, calling someone to the telephone. This sent Dr. Munk into a tantrum, and I immediately had one of my mechanics disconnect the horn. Before he had entirely calmed down, he walked over toward the balance structure and put his hands on the long diagonal braces.

These were fairly flexible, and he found he could move them back and forth a bit. Visualizing the entire structure vibrating to the point of failure and the whole airplane and balance crashing to the ground, the perturbed Munk ordered me to tear down the balance entirely and to design a new foundation and framework for it. He then turned and went back to his office a couple of blocks away.

Naturally I, too, was perturbed. Munk, after all, had approved every detail of my balance design. Not knowing what to do, I waited for some time to give him an opportunity to cool down. Then I went to his office and, as calmly as I could manage, mentioned that I thought the natural frequencies of the long diagonal members would be so low that vibrations would not be incited by the more rapid impulses from the engine and propeller. But mainly I suggested that, inasmuch as all the parts were made and ready to be put up, why not wait a couple of days before tearing it down and make a careful trial using the Sperry Messenger, starting at low speed, gradually increasing it, before dismantling the apparatus. Munk finally agreed, but demanded to be present when the test was made.

I did not like that idea of his presence one iota. To start the engine, the Messenger's propeller had to be cranked by hand from a balloon ladder that was put up in front of the propeller 25-feet above the floor. (A balloon ladder was like a fireman's ladder but its base was attached to a pair of weighted wheels, which permitted it to be "leaned" out into space. At its base there was a "protractor" that told you how far it could be angled without tipping over.) This sweaty business often took some time. It was not the kind of operation I wanted the excitable Munk to watch. Moreover, since no one else in the PRT section had ever started an airplane engine by turning the propeller, I was the one who was going to have to do it.

I brought my problem to Elton Miller, my boss, and to Henry Reid, the engineer-in-charge. Together, we decided that the only thing to do was to make an end-run around Munk and check out the tunnel balance system in his absence. This was easily done, as Munk worked on theoretical problems in his room at a Hampton boarding house every afternoon. We set up the test run and after a bit got the engine started without any difficulty. We then experimented with it until we could start it easily and felt ready for the final trial.

The problem of convincing Munk remained. We could not simply tell him about the successful test, so we agreed to arrange another "first test" for Munk to witness. Engineer-in-charge Reid escorted Munk to the tunnel the next morning. I casually said, "Good morning," clambered up the ladder, and pulled through the Messenger's prop. Luckily, the engine started on the first try. We then moved the ladder away, ran the engine through its entire range with no vibration difficulty, and then shut it down. Now, I wondered, what sort of explosion will we have? I needn't have worried. Munk walked toward me with his hand outstretched and congratulated me on the success of the operation. Everything had turned out all right. The balance system of the PRT operated satisfactorily with engines of up to 400 horsepower into the late 1930s, when it was replaced by a new and better one.

In 1926 Dr. Munk gave a number of lectures on theoretical aerodynamics to a select group of young Langley engineers. I was very happy to learn these things from him. Ever since graduation from the University of Illinois, I had thought about taking some graduate courses in aeronautical engineering. While working for Tony Yackey, I had read in a magazine article about the graduate courses in aeronautics offered at Massachusetts Institute of Technology. I had written MIT for information and had received a letter back from Professor Edward P. Warner, who had been Langley's first chief physicist in 1919 and would later become assistant secretary of the navy for aeronautics, editor of the magazine *Aviation*, and finally president of the International Civil Aviation Organization (ICAO), which continues to coordinate the rules and regulations for aeronautical activities throughout the nations of the world. I had hoped still to find a way to work in some graduate courses even after reporting to work at the Bureau of Aeronautics. But Dr. Lewis had talked me out of the idea on the basis that formal aeronautical engineering education was inferior to what I could learn if I went to work for the NACA at Langley. I guess he was probably right in regard to the aeronautical courses per se, but on occasion, in later years, I sorely missed the extra mathematics and physics that would have been obtained in school.

As mentioned earlier, the power plant for the new PRT consisted of two 1,000-horsepower, six-cylinder in-line diesel engines taken from a T-2 submarine. These engines were located end-to-end with crankshafts connected to a large sheave or pulley between them. This sheave carried forty-four Tex-rope V-belts to a similar sheave on the shaft of the propeller fan that drove the air through the tunnel. The shaft of the propeller fan was 25 feet above the ground, and the two sheaves were 55 feet apart center to center. Because we were concerned that some destructive vibrations might occur in the crankshaft-sheave assembly, we decided that a theoretical analysis of the torsional oscillations should be made, with Dr. Munk outlining the problem and a new man, Dr. Paul Hemke, to work out the solution. As a junior engineer, my assignment was to give the measurements and sizes that I would get from the drawings of the engines and sheaves.

I had no difficulty giving them the measurements, but Dr. Hemke was never able to get the gist of the torsional pendulum problem as described by Munk. This went on for some time with no results being obtained. Finally, I looked into my mechanical engineers' handbook and into a couple of textbooks and found that considerable work had been done on the problem and that the solution was not too difficult. I made the computation myself, coming out with a natural frequency of 312 RPMs. Later on, after the tunnel was in operation, some men came down from the navy shipyard in Brooklyn with equipment to measure the torsional oscillations; they found exactly the same natural frequency as I had computed. Hitting it exactly, of course, was a matter of luck, but it helped give me a good reputation, whether I deserved it or not. The success put me in good with Munk, but unfortunately Dr. Hemke was never able to work satisfactorily with him. A short time later he left the NACA. Hemke later joined the faculty of the U.S. Naval Academy, after holding a prestigious Guggenheim Fellowship for research under B. Melville Jones at Cambridge.

Another problem I helped to solve was the design of the 28-foot propeller fan that was to circulate the air in the propeller research tunnel. This fan needed to have eight blades of normal width. The exact energy ratio of the tunnel was not known in advance, so I desired to have blades that could be adjusted so that the pitch could be set exactly right after trial runs. Aluminum-alloy blades therefore seemed the best choice, but the blades we wanted were too large to be forged in the manner of the aluminum-alloy propeller blades then being manufactured. Fortunately, the propeller was to turn at only 375 revolutions per minute, which meant that the stresses would be very low in comparison even with airplane propellers having large diameters. This gave me the idea that a cast aluminum alloy might be used successfully, which it was.

I arranged with the Aluminum Company of America to cast the blades in their plant at Cleveland, Ohio. Before the large blades were cast, however, the company made two blades for a small ten-foot model that I then took to McCook Field in Dayton, where they were tested by Army Air Service engineers on their propeller whirl rig. This test showed the blades to be sufficiently strong.

During the period that the blades were being manufactured, I made a number of trips to Cleveland. On one occasion, when I had an afternoon with nothing special to do, I visited the Martin aircraft plant in the city's southern suburbs. I went into the door to the main office and told a young lady at the desk that I was an engineer from the NACA at Langley Field and that I'd like to visit the plant. She ushered me into the office of Glenn L. Martin himself, and he spent a couple of hours showing me around. How different from the stilted, bureaucratized conditions existing today in an aircraft factory! Of course, the Martin plant was small then, with only a few hundred employees. Most of his production went to the Navy Department; in fact, while at the Bureau of Aeronautics, I had designed a couple of the propellers used on his airplanes. Because many of his models were seaplanes and nearby Lake Erie was frozen solid in the winter months, Martin was then looking around to find a place farther south where he could manufacture and fly them away directly from the factory all year round. On this account he asked about the conditions around Hampton and Newport News, a neighborhood that he thought might be quite suitable. I told what I could about the area, and later he made some overtures in this direction. But, as I remember it, the local people at Newport News were not interested. Martin finally moved his factory to Middle River, Maryland, near Baltimore, where local authorities gave him a very good deal.

The NACA held its first annual manufacturers' conference at Langley in May 1926. The meeting was attended by representatives from the military air services, Department of Commerce, and aeronautical manufacturing industry. The morning was spent touring the various laboratories and learning about the research work

that was going on in them. The propeller research tunnel was about finished, but Ted Myers, who was in charge of the tunnel's power plant, had not been able to get the diesel engines to run. However, we had the regular starting arrangement by which we turned the engines over by a blast of compressed air until they would start running as diesels. At the demonstration that morning, we ran the tunnel on the compressed air for about one minute; the little Sperry Messenger was up in the test section with its engine running also. In the afternoon, the conference was held at the military officers' club a few blocks away, and suggestions for possible new research were invited. One of the suggestions that was made concerned the cowling of radial air-cooled engines.

When I had started work at the Bureau of Aeronautics, almost all of the army and navy airplanes had had water-cooled engines. The navy, however, was interested in developing radial air-cooled engines. This work had been carried on under the direction of Comdr. Eugene E. Wilson of the bureau's power plant division and had been conducted mostly with the Wright engines designed by Charles Lawrance. The radial engines with their short crankshafts and crankcases and no radiators or water-cooling systems were lighter than the water-cooled engines. But the finned cylinders were cooled simply by projecting them into the airstream, and this caused a high drag. An attempt had been made to reduce the drag by putting propeller spinners over the hubs and cowling the crankcase and lower portions of the cylinders, but the outer ends of the cylinders still extended into the airstream.

During the morning session of the NACA conference, everyone had witnessed the operation of the Sperry Messenger airplane, with its radial air-cooled Lawrance engine running, in the propeller research tunnel. At the afternoon meeting, several people mentioned that tests should be made in the PRT to see how much the cowling could be extended outward without interfering too much with the cooling of the engine. Both the drag and propeller efficiency should be determined, we all agreed, as well as the cooling. During the ensuing months, I laid out a program for these cowling tests.

While studying propellers at the Bureau of Aeronautics, I learned from the propeller work carried out by William F. Durand and Everett P. Lesley at Stanford University the advantages of using a systematic series of independent variables in experimental research. I recognized that the range of variables should extend, if possible, on both sides well beyond the area of greatest interest. One extreme of this series was obviously making use of the bare engine with no cowling at all. The other extreme was to enclose the engine completely. This option had not been anticipated but looked enticing. An engine nacelle would then start with the best airship shape available, and the air could be brought in smoothly at the center of the nose. But how could one get the air out again in a smooth and efficient manner? Elliott G. Reid, who was in charge of the atmospheric wind tunnel at the time, had been making tests on Handley Page wing slots, and he helped me to design an annular exit slot. Together, these forms eventually became the NACA's low-drag cowling.

After I had completed the outline of a tentative cowling test program, the NACA sent it to the military air services and to various manufacturers that had shown interest at the May 1926 conference, and it was approved by all of them. Fortunately, getting their okay took some time, because the propeller research tunnel was at this point in no sense ready to operate.

After establishing that the tunnel was operating satisfactorily, we carried out several series of propeller tests and cowling tests at the same time. Among other things, this enabled us to obtain the effect of propeller-body interference on each cowling design. The various propeller tests were mostly covered in the following NACA reports: TR 306, "Full-Scale Wind-Tunnel Tests of a Series of Metal Propellers on a VE-7 Airplane" (July 13,1928); TR 338, "The Effect of Reduction Gearing on Propeller-Body Interference as Shown by Full-Scale Wind-Tunnel Tests" (March 20, 1929); TR 339, "Full-Scale Wind-Tunnel Tests with a Series of Propellers of Different Diameters on a Single Fuselage" (March 12,1929); TR 340, "Full-Scale Wind-Tunnel Tests on Several Metal Propellers Having Different Blade Forms" (March 18,1929); and TR 350, "Working Charts for the Selection of Aluminum Alloy Propellers of a Standard Form to Operate with Various Aircraft Engines and Bodies" (March 25, 1929).[1]

The goal that we had set for ourselves in the cowling program was a cowled engine that would be cooled as well as one with no cowling whatsoever. This program proceeded easily enough until the complete cowling, covering the entire engine, was first tried. At this point, some of the cylinder temperatures proved to be much too high. After several modifications to the cooling air inlet and exit forms, and the use of internal guide vanes or baffles, we finally obtained satisfactory cooling with a complete cowling. Don Wood was in charge of the actual operation of the testing, and the first of these modifications was made while I was away on a vacation. When I got back, it was obvious to me that the boys were on to something, and from that time on we all worked very hard on the program.

The results of this first portion of cowling tests were so remarkable that we decided that the NACA should make them known to industry at once. In November 1928 I wrote up Technical Note 301, "Drag and Cooling with Various Forms of Cowling for a 'Whirlwind' Engine in a Cabin Fuselage," which the NACA published immediately. The summary of the report was as follows:

The National Advisory Committee for Aeronautics has undertaken an investigation in the 20-foot Propeller Research Tunnel at Langley Field on the cowling of radial air-cooled engines. A portion of the investigation has been completed in which several forms and degrees of cowling were tested on a Wright Whirlwind J-5 engine mounted in the nose of a cabin fuselage. The cowlings varied from the one extreme of an entirely exposed engine to the other in which the engine was entirely enclosed. Cooling tests were made and each cowling modified if necessary until the engine cooled approximately as satisfactorily as when it was entirely exposed. Drag tests were then made with each form of cowling and the effect of the cowling on the propulsive efficiency determined with a metal propeller. The propulsive efficiency was found to be practically the same with all forms of cowling. The drag of the cabin fuselage with uncowled engine was found to be more than three times as great as the drag of the fuselage with the engine removed and nose rounded. The conventional forms of cowling in which at least the tops of the cylinder heads and valve gear are exposed, reduced the drag somewhat, but the cowling entirely covering the engine reduced it 2.6 times as much as the best conventional one. The decrease in drag due to the use of spinners proved to be almost negligible.

I concluded this summary by arguing that use of the form completely covering the engine was "entirely practical" under service conditions, but also by warning that "it must be carefully designed to cool properly."

Having completed the initial round of wind-tunnel tests, we then borrowed a Curtiss Hawk AT-5A airplane from the Army Air Service at Langley Field already fitted with the Wright Whirlwind J-5 engine, and applied the new cowling for flight research. These tests showed that the airplane's speed increased from 118 to 137 miles per hour with the new cowling, an increase of 19 MPH. The results of the instrumented flight tests had a little scatter, and we could have been justified in claiming that the increase in speed was 20 MPH instead of 19, but I wanted to be conservative. I didn't want people to expect too much from this cowling, so we called it 19.

The second part of the cowling program covered tests with several forms of cowling, including individual fairings behind and individual hoods over the cylinders, and a smaller version of the new complete cowling, all mounted in a smaller, open-cockpit fuselage. We also performed drag tests with a conventional engine nacelle and with a nacelle having the new complete design. Though the individual fairings and hoods proved ineffective in reducing drag, we found that the reduction with the complete cowling over that with the conventional cowling was in fact over twice as great with smaller bodies as with the larger cabin fuselage. Data from the AT-5A flight tests confirmed this conclusion.

The first public acclaim of the cowling came in February 1929 when Frank Hawks established a new Los Angeles-to-New York nonstop record (18 hours, 13 minutes) flying a Lockheed Air Express equipped with an NACA low-drag cowling that increased the aircraft's maximum speed from 157 to 177 MPH. The day after the feat, the NACA received the following telegram:

Cooling carefully checked and OK. Record impossible without new cowling. All credit due NACA for painstaking and accurate research. [Signed] Gerry Vultee, Chief Engineer, Lockheed Aircraft Co.

Some time later, the NACA gave me a photographic copy of this telegram, along with a picture of the cowled airplane.

A few weeks before Hawks's record-breaking flight, I had attended the New York Air Show in Madison Square Garden. At the show Chance Vought told me that Germany's Claude Dornier would like to talk to me about the possibility of putting the NACA cowling on the twelve uncowled radial air-cooled engines of his giant DO-X flying boat—which, at the time, was the largest airplane in the world. These twelve engines (British-made Jupiters rated at 550 horsepower each) were mounted back-to-back in six nacelles, each with one tractor propeller and one pusher propeller. Cowling the pushers would, of course, constitute an entirely new problem.

Before finishing this story, it should be mentioned that a great effort was then being made in a number of countries to develop aircraft suitable for airline use across the oceans, particularly the North Atlantic. The aircraft were of two main types: rigid airships of the Zeppelin type and large flying boats like Dornier's. The wing of the DO-X projected from the top of an ample hull with a span of 157 feet and a chord of 30 feet. Lateral stability on the water was obtained by the use of sponsons, or short and stubby winglike structures that projected from the bottom of the hull on each side. Constructed in the late 1920s at Altenrein, Switzerland, on Lake Constance and near Friederichshafen, Germany, the DO-X could accommodate sixty-six passengers comfortably over a range of 700 to 900 miles, but could not lift any kind of payload over transatlantic distances, the minimum such distance being roughly 2,000 miles. On one flight from adjoining Lake Constance, though, one hundred seventy people were crowded into the airplane (I've heard that nine were stowaways), making quite a record at that time.

By this time at Langley lab, we had mounted the low-drag cowling on all three engines of a Fokker trimotor airplane. The comparative speed trials proved extremely disappointing. Separate tests on the individual nacelles showed that cowling the Fokker's nose engine gave approximately the improved performance we expected. Cowling the wing nacelles, however, gave no improvement in performance at all. This was strange, because the wind-tunnel tests had already demonstrated convincingly that one could obtain much greater improvement with a cowled nacelle than with a cowled engine in front of a large fuselage.

Some of us started to wonder how the position of the nacelle with respect to the wing might affect drag. In the case of the Fokker (as well as the Ford) trimotor, the wing engines were mounted slightly below the surface of the wing. The upper surface of the fully cowled nacelle then came very close to the under surface of the wing. As the air flowed back between the wing and nacelle, and the distance between them increased toward the rear of the nacelle, the expansion required was too great for the air to follow smoothly. We tried fairing-in this space, but achieved only a small improvement.

As a result of these experiences I laid out a series of model tests in the propeller research tunnel in which an NACA-cowled nacelle with a power-driven propeller was placed in a number of different positions with respect to the wing. Where it appeared pertinent, extra fairing was put between them. These tests were run by Don Wood and his crew after I had left the NACA to work for Hamilton Standard.

The resulting data on the effect of the nacelle on the lift, drag, and propulsive

efficiency of the Fokker airplane made it clear that the optimum location of the nacelle was directly in line with the wing, with the propeller well ahead of the wing's leading edge. This position had the least overall projected area, and I suppose the result might have been expected. With the complete cowling, the radial engine in this position spoiled the maximum-lift coefficient of the wing. With the cowling, and the smooth airflow that resulted from it, the maximum-lift coefficient was actually increased. After this important information was transmitted confidentially to the army, navy, and industry, most all of the transport and bombing airplanes employed radial wing-mounted engines with the NACA cowled nacelles located approximately in the optimum position.

Document 3-9(a-b)

(a) Western Union telegram, Jerry Vultee, Lockheed Aircraft Co., Burbank, CA, to NACA, "Attention Lieutenant Tom Carroll," 5 Feb. 1929, copy in NACA Research Authorization file 215, NASA Langley Historical Archives.

(b) Fred E. Weick, "The New NACA Low-Drag Cowling," *Aviation* 25 (17 November 1928): 1556-1557, 1586, 1588, 1590.

National attention became focused on the success of the NACA's low-drag engine cowling in February 1929 when celebrated pilot Frank Hawks set a new nonstop speed record from Los Angeles to New York in a Lockheed Air Express equipped with a NACA cowl that increased its top speed from 157 to 177 miles per hour. Gerald "Jerry" Vultee, chief engineer with the Lockheed Aircraft Company, sent a telegram to NACA Langley's chief research pilot, Thomas Carroll, crediting the NACA for the flight's success; this telegram can be found as the first document below.

Not only did the low-drag cowling provide unprecedented performance for new aircraft like the Air Express, it was also a factor in generating public notoriety for how much U.S. aviation was advancing generally. It also bolstered the reputation of the young NACA. Amid a burst of publicity—some of it exaggerated—about the benefits of the NACA cowling, the National Aeronautics Association announced in January 1930 that the NACA cowl had won the Collier Trophy for the greatest achievement in American aviation in 1929.

The second document provides NACA Langley engineer Fred E. Weick's article on the low-drag engine cowling published in *Aviation* magazine in November 1928. This was still early in the NACA's cowling program. Work on advanced forms of cowlings continued throughout the 1930s.

Document 3-9(a), Western Union telegram, Jerry Vultee, Lockheed Aircraft Co., Burbank, CA, to NACA, "Attention Lieutenant Tom Carroll," 5 Feb. 1929, copy in NACA Research Authorization file 215, NASA Langley Historical Archives.

WESTERN UNION

FEB 5, 1929

To: NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

Street and No.: LANGLEY MEMORIAL AERO LABORATORY LANGLEY FIELD VIR

Place:

ATTENTION LIEUTENANT TOM CARROLL LOCKHEED AIR EXPRESS PILOT: FRANK HAWKS WHICH JUST ESTABLISHED NEW LOS ANGELES TO NEW YORK NONSTOP RECORD OF EIGHTEEN HOURS THIRTEEN MINUTES IS EQUIPPED WITH N.A.C.A. COWLING WHICH INCREASED ITS SPEED TWENTY MILES PER HOUR FROM 157 TO 177 STOP COOLING CAREFULLY CHECKED AND OK STOP RECORD IMPOSSIBLE WITHOUT NEW COWLING ALL CREDIT DUE N.A.C.A. FOR PAINSTAKING AND ACCURATE RESEARCH AND GENEROUS POLICY STOP KINDEST PERSONAL REGARDS STOP LETTER WITH DATA AND PHOTOS FOLLOWS.

JERRY VULTEE
LOCKHEED AIRCRAFT CO.

Document 3-9(b), Fred E. Weick, "The New NACA Low-Drag Cowling,"* Aviation *25 (17 November 1928): 1556-1557, 1586, 1588, 1590.

STATIC radial air-cooled engines are claimed by some engineers to possess several advantages over engines of other types, among which are low weight per horsepower, the small number of parts required, and the consequent reliability and low cost of manufacture. They are now widely used and are still gaining favor, both in commercial and military aeronautics. They have had, however, one outstanding disadvantage--their extremely high air resistance, caused by the large frontal area and very poor aerodynamic form. The high drag has been a serious handicap in the field of high speed pursuit planes and single seater fighters, but the low weight of the air-cooled radial has some advantages.

Aircraft designers have attempted to reduce the drag due to the air-cooled radial engine by enclosing varying amounts of it within the nose of the fuselage or nacelle. There has been a wide difference of opinion regarding the best forms and amounts of cowling, and its effect on the drag and the cooling of the engine. Some designers have cowled in almost the entire engine leaving only the cylinder heads and valve gear exposed, while others have been content to leave the engine entirely exposed. Practically no accurate or reliable information has been available on the subject, for until very recently there has been no practical way of obtaining it.

The National Advisory Committee for Aeronautics has made a practice for each of the last few years of inviting the aircraft manufacturers and their representatives to spend a day at the Committee's laboratory at Langley Field, in order that they may become more familiar with the work and facilities, and make suggestions regarding research which would be helpful to them. At the meeting held on May 24, 1927, the manufacturers were practically unanimous in urging that an investigation be made in the new 20 ft. propeller research wind tunnel, then just being completed, on the cowling of radial air-cooled engines. The new tunnel, in which an air speed of 110 m.p.h. can be obtained, is ideal for the purpose. An actual full size airplane can be accommodated except for the wing tips, and the engines can be run with its propeller as in flight. The propeller thrust, airplane drag, and propulsive efficiency can be measured under flight conditions but with laboratory accuracy, so that small differences due to slight changes in cowling are brought out, and thermocouples can be used to obtain the temperatures at a large number of points on the cylinders so that the effect of the cowling on the cooling can be studied.

In working out a program for the cowling tests, it was thought desirable to include not only all of the main conventional forms of cowling, but also to have them arranged in series with various degrees or amounts of cowling. At one extreme of the series, the engine was left entirely exposed except for the rear of the crankcase where it was fitted to a fuselage. For the other extreme, it seemed logical to enclose the entire engine. This involved problems in design, for it was of course desired to have the drag as low as possible, but still have the engine cool satisfactorily. It was easy to design a form enclosing the engine and having a low drag, but there was no information available as to the best means of cooling the engine. It was decided to make the outside of the complete cowling of circular cross-section and smooth form, without individual cylinder fairing such as have been used on a few experimental airplanes, particularly in England. This was done partly for the sake of simplicity and partly because in the case of most American radial engines, and especially the Wright Whirlwind J-5 which was used in this investigation, there is so little room between adjacent cylinders that individual hoods are impractical. It was necessary, of course, with the cowling entirely covering the engine, to separate the air used for cooling the cylinders from the general flow about the body, guide it past the cylinders, and then return it to the outside air again. If the separation and the return could be made smoothly, it seemed likely that a large decrease in drag could be obtained.

It was decided that the cooling air for all of the cylinders could be taken in most satisfactorily at the center of the nose where the air pressure on the body, when in motion, is greatest. This allowed a simple symmetrical design and a smooth separation of the cooling air from the general flow. Regarding the matter of returning the used cooling air to the general flow again, suggestions were obtained from the staff in charge of the five-foot atmospheric wind tunnel of the laboratory, who had done some work on wing slots and boundary layer control. As a result, the new cowling was designed with an annular slot extending entirely around the circumference of the body a short distance behind the engine. In section the slot was similar to some wing slots in which the air passing through is directed tangentially along the surface. Rough comparative tests on two small model fuselages were then made in the six-inch wind tunnel, and these indicated that a substantial reduction in drag and a reasonable flow of air through the nose and out of the slot would be obtained. The flow of cooling air is helped by the fact that the outside air pressure at the nose is the highest and that at the slot about the lowest found over the entire body, when the body is in motion with respect to the air.

A program of tests was drawn up including both extremes of cowling and several conventional intermediate steps, both with and without spinners and on a cabin type and an open cockpit type fuselage. This program was then submitted to the manufacturers for criticisms and suggestions, several of which were adopted. As the tests were finally made, the cooling with each form of cowling was first investigated and compared with that for the uncowled engine. Then the cowling was modified, if necessary, and retested until the cooling was approximately as satisfactory as for the entirely exposed engine. After that, tests were made on the drag and the propulsive efficiency.

The portion of the investigation involving the cabin fuselage has been completed, and tests are now being made with the smaller open cockpit fuselage.

One of the first findings of the tests was the enormous amount of drag due to the uncowled engine. The drag of the bare cabin fuselage with the engine removed

and nose rounded was more than tripled by the addition of the engine, and that of the open cockpit fuselage was increased nearly five times. In fact, the drag of the open cockpit fuselage was about 10 percent. greater than that of the cabin fuselage with the same engine cowling even though the open cockpit fuselage had only half the cross-sectional area of the other. The larger body behind the engine evidently has a beneficial effect on the drag, and the drag of a small fuselage without engine is insignificant compared with that when the engine is added. The conditions with a wing engine nacelle would probably be at least as bad as with a small fuselage. These facts showed plainly that there was an opportunity for great improvement, especially in the cases of small fuselages and engine nacelles.

The next outstanding development of the tests was that the conventional cowlings, in which only the central portion of the engine is covered, had but slight effect in reducing the drag. Even in the extreme case, in which a large spinner was used and the entire engine was covered in except for the tops of the cylinder heads and valve gear, the reduction in drag was comparatively small. Spinners as conventionally used in front of the projecting cylinders of radial engines were found to have a negligible effect on performance.

When the new N. A. C. A. cowling completely enclosing the engine was first tested, an exceptionally low drag was obtained, but the engine ran much too hot. The cowling was modified and retested many times before the engine cooled properly, but finally the cooling was approximately as satisfactory as with the uncowled engine. To accomplish this, it had been necessary to enlarge the hole in the nose, provide cut-outs over the magnetos, enlarge the slot, and guide the cooling air past the hottest parts of the cylinders. In this connection, there is still room for improvement in the effective use of the cooling air, with a possible further gain in performance.

The reduction in drag of the cabin fuselage with the final cowling, which cooled satisfactorily with the engine completely enclosed, was 2.6 times as great as with the best conventional type with a large spinner. A still greater difference is to be expected in the case of a small fuselage or a nacelle, the tests on which are now underway.

Rough calculations show that the power required to drive an average commercial Whirlwind engined cabin airplane at its maximum horizontal speed would be reduced by from 15 to 20 per cent by the use of the new N. A. C. A cowling in place of the best present conventional cowling. For three-engined transports with two wing engines the decrease in power required under similar conditions would be from 20 to 25 percent, and for machines with small open cockpit fuselages as much as 20 to 30 percent.

If the full engine power were used with the N. A. C. A. cowling, the maximum horizontal speed would be increased by from five to ten miles per hour for the average cabin airplane, somewhat more for the three-engined machines, and as much as 20 m.p.h. for small open cockpit planes such as single-seater fighters.

Inasmuch as the drag is less with the N. A. C. A. cowling, the power available for climbing is greater, and the rate of climb and the ceiling will be improved. The fuel consumption will also be reduced, the amount depending on the speed of flight. If the same cruising speed is maintained, as with the old cowling, the decrease in fuel consumption will be approximately proportional to the decrease in power required. On the other hand, if, as is more likely the case, the same engine power will be used in order to cruise at an increased speed, the number of miles obtained per gallon will be increased in proportion to the increase in speed. An increase in range will also be obtained, which is proportional to the increase in the miles flown per gallon.

It is believed that the improved performance with the new cowling makes the air-cooled radial equal to or better than the water-cooled engine in the matter of drag except in the case of pure racing planes on which wing radiators may be used.

The results of the wind tunnel tests were so promising that it was decided to check them in flight. The Army kindly loaned an AT-5A (Curtiss Hawk with Whirlwind engine for advanced training purposes) for the tests. The new N. A. C. A complete cowling was then adapted to the plane by the Flight Operations Section of the laboratory, which had also constructed all of the cowling for the wind tunnel investigation, and the tests were carried out by the chief test pilot, Thomas Carroll, and his two assistants, Messrs. McAvoy and Christopher. The best of the standard Army AT-5A planes was taken from the line and a direct comparison was made between it and the one with the complete cowling, both being flown at the same time. Each test point was checked several times by each of the three pilots.

The plane with the new N. A. C. A. cowling had a maximum horizontal speed at sea level of 137 m.p.h. as compared with 118 m.p.h. for the standard AT-5A, both being attained with the same engine revolutions. This represents a gain of 19 m.p.h due to the new cowling. The reduction of engine drag is probably directly responsible for about 13 or 14 mi. of this gain, the rest being due to the decrease in induced drag and the increase in propeller efficiency at the higher speed.

All of the pilots reported the plane with the new cowling smoother to fly and better in answering the controls than the standard plane. No doubt this can be attributed to the smoother airflow over the fuselage and inner portion of the tail surfaces.

The pilots also reported that with this airplane the new cowling did not alter the range of vision in any useful field. The complete cowling, of course, cuts off whatever view can be obtained between the cylinders. Incidentally, there seems to be a great difference of opinion regarding the usefulness of the vision which can be obtained between the cylinders, some pilots maintaining that it is essential in the case of some planes, such as fighters, and others believing that it is never used to an appreciable extent. It undoubtedly depends to some extent on the contour of the engine and the amount of space between the cylinders. If the complete cowling comes into general use, as these first tests seem to warrant, there may be a tendency toward the development of more compact engines having a smaller overall diameter.

Other engine developments which would improve the effectiveness of this type

of cowling are the placing of all accessories, especially the magnetos, in the rear (this is of course now being done in many power plants), and the provision of greater distance between the plane of the cylinders and the propeller so that a better shaped nose can be had.

The new N. A. C. A. cowling, being simple and smooth in form, is easily constructed. The nose piece or hood used in both the wind tunnel and flight tests was built into a complete ring which was inherently stiff and strong without bracing. It could be easily removed, but it was first necessary to remove the propeller. To avoid this in practice, it would probably be advisable to make the nose piece in two or three quickly detachable sections. When the nose piece is removed the small cowling over crankcase is similar to conventional types, and most parts of the engine requiring frequent attention can be easily reached. No difficulties of maintenance occurred during the many hours of wind tunnel running or in flight tests.

In manufacture, the complete cowling would probably cost no more than a conventional cowling without spinner, except for the nose piece. Since a spinner is not required, it would cost little if any more than the conventional types using a large spinner. The weight of the nose ring used on the AT-5A, part of which was made of 1-16 in. thick aluminum for convenience in working, was 27 lbs.

In the test cowlings, the engine exhaust was directed out of the slot by means of individual stacks on each cylinder. The conventional ring type exhaust collectors could be used behind the cylinders if desired, or if the engine exhausted at the front, the exhaust ring could be made the front part of the nose piece. This latter would provide a very convenient means of support for the rest the nose piece. The complete cowling is well adapted for the use of shutters, which could be made to reduce flow of cooling air over the entire engine if desired. It would seem that this would improve the operation of cooled engines very appreciably in cold climates.

One point regarding the application of this cowling is worth mentioning. The many modifications, which were necessary before proper cooling was obtained with the cowling, show that it must be carefully designed. It is possible that eventually the engine manufacturers will find it advisable to furnish the engines complete with cowling, thus ensuring proper cooling conditions for their products. In that case the exhaust system could no doubt be very neatly incorporated in the cowling.

In appearance, the N. A. C. A cowling is reminiscent of the hoods enclosing the old rotary engines of the war period. It gives the fuselage-engine combination longer and smoother lines with, even on a small fuselage such as that of the AT-5A, are not unpleasant to the eye.

In conclusion, it would seem from the test made to date that a very substantial increase in high speed and all-round performance can be obtained on practically all radial engined aircraft by the use of the new N. A. C. A complete cowling.

Document 3-10(a-c)

(a) H. C. H. Townend, "Reduction of Drag of Radial Engine by Attachment of Rings of Aerofoil Section," British Aeronautical Research Committee *Research and Memoranda 1267* (1929).

(b) United Aircraft and Transport Corporation Technical Advisory Committee, Meeting Minutes, December 1929, Boeing Historical Archives, Seattle, Washington, pp. 213-256.

(c) "The Curtiss Anti-Drag Ring," *Curtiss-Wright Review 1* (December 1930): 16.

The NACA cowling was not the only method of reducing the drag of air-cooled engines. In 1929, Hubert C.H. Townend of the British National Physical Laboratory also developed a low-drag engine cowling. Known as the "Townend ring," this design also reduced drag with little degradation of engine cooling and was widely adopted in Europe and America. Because it completely enclosed the engine, many people in the aviation industry believed that the NACA cowling was inevitably detrimental to engine cooling; therefore, they intuitively favored Townend's design. As indicated in the second document below, members of the United Aircraft and Transport Technical Advisory Committee (curiously of which Fred Weick was briefly a member) reflected this prejudice. Several Boeing aircraft, such as the P-26 "Peashooter" of the early 1930s, utilized the ring cowling. Other manufacturers employed similar structures. One company, Curtiss-Wright, developed its own form of Townend ring, which it called the "Curtiss Anti-Drag Ring," the subject of the third document below.

The history of the NACA cowling-Townend ring rivalry has yet to be written. In the beginning, neither the British National Physical Laboratory nor the American NACA appear to have been aware of the other's cowling work. The NPL published the results of its ring research just before the NACA's cowling reports appeared. To impress American manufacturers with the value of its cowling, the NACA placed its design into some direct competition with the Townend ring. For example, it did so with the Martin B-10 when it competed with the Boeing B-9 for a large army contract in 1932 (see Document 3-19). The overall competitive situation between the cowl and the ring fed the fire of what became a transatlantic dispute and resulted in a long series of patent suits.

Document 3-10(a), H. C. H. Townend, "Reduction of Drag of Radial Engine by Attachment of Rings of Aerofoil Section," British Aeronautical Research Committee* Research and Memoranda *1267 (1929).

AERONAUTICAL RESEARCH COMMITTEE

Report for the year 1929-30

Brigadier-General The Rt. Hon. The Lord Thomson,
P.C., C.B.E., D.S.O., *p.s.c.*, Secretary of State for Air.

June, 1930

My Lord,

The Aeronautical Research Committee beg to submit their report for the year 1929-30.

The Committee wish to draw attention to the stead expansion of activities in aeronautics throughout the world and to the consequent increase in the number of problems awaiting solution which have been brought to their attention in connection with developments within the British Empire.

The speed and size of aircraft continually increase and the number of uses to which aircraft are put grows steadily. New flying boats and aeroplanes, fitted with a number of engines, are projected. The speed of aircraft, especially for service and racing purposes, has increased greatly; the race for the Schneider Trophy in 1929 and the speed record made later over the three kilometer course by officers of the Royal Air Force indicate the extent of this advance. The new airships, R.100 and R.101, have completed their trials during the year and as a consequence of experience with them, the question of larger ships, bringing with it new problems both of design and performance, has naturally come forward. Research requirements demand not only the solution of these new problems but also experiments on larger models at higher values of the Reynolds number, i.e., at higher speeds or in air under pressure; small models do not always reproduce the actual aircraft in sufficient detail. For work at higher values of the Reynolds number a compressed air tunnel is under construction at the National Physical Laboratory, while for models requiring engine details the erection at the Royal Aircraft Establishment of a large wind tunnel with an air jet 24 ft. in diameter has been recommended. In addition, a scheme has been put forward for replacing the obsolescent 7 ft. tunnel No. 1 at the N.P.L., by two open jet tunnels of 8 ft. diameter housed in the existing building. This modernization will materially accelerate the progress of work in the N.P.L. program.

AERODYNAMICS.

When an aircraft is in motion, the disturbance produced in the air, together with the reaction up on the machine itself and its consequent performance all depend ultimately on the nature of the flow over its surface. This "boundary flow" is therefore of great practical importance, and detailed investigations of its characteristics have occupied the attention of the Committee during the year. Already in the previous year, Professor B. M. Jones had drawn attention, in R.&M. 1199, to the large difference existing between the drag of an aeroplane and that estimated from ideal conditions. The coefficient of frictional resistance to the motion of a flat surface through the air is known from experiment to depend on the Reynolds number at which the experiment is made. In the ideal conditions it should be possible to calculate the resistance to an aeroplane from a knowledge of its surface area and of this coefficient of friction, on the assumption that its value does not depend on the curvature of the surface. An enquiry has been started on the curvature of the surface. An enquiry has been started to investigate how far this assumption is justified.

The frictional coefficient depends also on the nature of the flow over the surface. This may be "laminar," in which case the air moves in smooth curves following more or less the shape of the surface, or it may be "turbulent" when a series of eddies is formed. For turbulent flow the coefficient considerably exceeds that found when the flow is laminar. In experiments in a wind tunnel both kinds of flow are observed; the flow over the forward part of a good streamline model is laminar, over the rear it is generally turbulent. As an aid to exploration near the surface of a body, an instrument has been designed which depends on the rate of cooling of a fine electrically heated wire and responds readily to fluctuations in the airflow. When placed near the surface of a model, this instrument can be arranged to indicate by audible means the nature of the flow and the points, if any, at which a change from laminar to turbulent flow takes place.

In some earlier investigations of the same problem, a small pitot tube was moved near to and parallel to the surface of the model. Some difficulties which attended this method have been overcome and the hot wire and pitot tube methods are now in good agreement. In addition to these two methods, the smoke trail from titanium tetrachloride, a chemical very suitable for this purpose, has made details of the flow visible. The first experiments with the chemical have given a good indication of the places on the models where the changes from laminar to turbulent motion occur but the technique of its use must be regarded as still in its infancy.

In addition to the study of the flow past a streamline body, a Joukowski type of aerofoil has been selected for experiments on resistance, for pressure plotting and for measurements of pitot head at distances as small as 2 or 3 thousandths of an inch from the surface. A report on this work is being prepared and will be published at an early date; it has a special interest since the form employed in this type of aerofoil is one for which the characteristics of the flow can be determined theoretically, so that comparison between theory and experiment will be possible.

The flow over airship bodies and aerofoils is generally of a stable nature, whereas the nature of the flow past a circular cylinder is known to vary rapidly at certain speeds. As another means therefore of attack on "critical flow," the experimental conditions for a cylinder have been changed by setting up artificial turbulence in the airstream and by the alteration of surface roughness. These changes produced orderly responses in the characteristics of the flow and it appears that even in the range of the Reynolds number in which there is a large change in the resistance coefficient these characteristics are quite definite.

The above experiments were all made at normal wind tunnel speeds. In other work carried out at very high speeds, pressure distribution has been measured round a Joukowski section. Up to a speed of half the velocity of sound the force coefficients vary by only a small amount from those found at low speeds but there is an appreciable divergence, when the speed reaches about 0.6 times the velocity of sound. On other wing sections closer agreement with results obtained in a low speed wind tunnel was obtained as regards lift, but as regards drag there was a marked difference even at velocities only half that of sound.

A theoretical study allowing for the compressibility of air has also been made of the conditions under which the flow at high speeds past curved surfaces may exceed the velocity of sound close to a convex surface, while remaining below that speed at some distance from the surface. There appears in the mathematical solution presented to be no indication of any discontinuity of the flow; moreover, speeds just greater than that of sound are attained without the formation of compressibility waves. This work explains why the experiments in the electrolytic tank at Cambridge mentioned in last year's report, were not successful at speeds above a certain value. Some confirmation of this theoretical work has been obtained in the small high speed jet at the N.P.L., but the matter is not yet completely cleared up.

INTERFERENCE.

While much time has thus been spent on studying flow near the surface, the effects due to the interference of one part of an aircraft on another have not been neglected. Attention had previously been drawn to the fact that aeroplane bodies and wings differing only in the method in which the wings were attached, gave widely different drags. The early experiments to elucidate this matter, described in last year's report, were purposely made upon a streamline body thicker and shorter than the average aeroplane body, with the object of accentuating the effects to be measured. Selected experiments have been repeated upon another body of proportions more nearly similar to those of the bodies of aircraft and the researches are being extended to include an analysis of the separate forces on wings and bodies. Moreover, a series of experiments on the interference of an engine nacelle and a wing has been commenced in the largest N.P.L. (Duplex) wind tunnel; an airscrew and an aeroplane body will be added later. The effect of changes in the relative position of the body and wing for the cantilever monoplane type of construction is being investigated at the R.A.E.

A cognate research on airscrew body interference, for both tractor and pusher screws using ideally streamline bodies, is well advanced. The experiments on the tractor screws are complete and those on pusher screws are in hand. This will provide a basis for further interference experiments upon airscrew-body-wing combinations.

Concurrently with the general investigations on interference and, indeed, arising out of them, a valuable device known as the Townend Ring has been developed, whose use leads to a substantial reduction of the drag of aeroplane bodies fitted with radial engines (*see* Illustration No. 2). The effect of the ring is to direct the air along the body and lessen the amount of turbulence; the higher the lift coefficient of the cross section of the ring the more efficient is the Townend Ring in decreasing total resistance to forward motion. Moreover, the cooling of the engine is not adversely affected. An account of the experimental work on this Ring has been issued in the Reports and Memoranda Series; since its publication, the results obtained with models have been confirmed by a number of full scale trials. In one case the top speed of a certain aeroplane was increased by eight miles per hour.

The evolution of the Townend Ring affords an example of the valuable practical results that may follow from simple experiments originally intended to throw light on fundamental points. The success achieved in this case has encouraged the Committee to continue their policy of investigating first the simplest cases of body and wing combinations in endeavoring to establish the main principles on which interference depends.

PERFORMANCE.

The variation of the range of an aeroplane with speed and height is of considerable practical interest. Experiments made at the A. & A.E.E., Martlesham, showed that the maximum range of a particular aeroplane increased substantially with height whereas theory indicates that a change in altitude should have little or no effect.

REDUCTION OF DRAG OF RADIAL ENGINES BY THE ATTACHMENT OF RINGS OF AEROFOIL SECTION, INCLUDING INTERFERENCE EXPERIMENTS OF AN ALLIED NATURE, WITH SOME FURTHER APPLICATIONS.

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Summary. Introductory. Some experiments are described in which large, and frequently negative, interference effects are found to be produced by certain objects of streamline or aerofoil form upon the drag of (1) a model of airship form (U.721), (2) model aeroplane bodies having radial engines in the nose, and (3) models in which turbulence is produced by grooves or sharp edges. The evidence provided by the tests shows that the effects produced in cases (2) and (3) are due, not to shielding or fairing of the obstructions, but to the addition of small aerofoils to the models in such a way as to control the airflow in the neighborhood of the obstructions, chiefly by governing its local direction.

In case (1) there were no obstructions on the body, and the results are concerned simply with the interferences between the body (of airship form) and rings of streamline section surrounding it or of streamline struts and aerofoils near it. The effects on the body were very marked, its drag falling to zero when surrounded by a ring of as much as three times its own maximum diameter situated near the plane of the nose. For struts or rings in closer proximity to the surface, the body drag was negative in direction and three or four times its normal magnitude. There is at present no satisfactory explanation to be offered for some of these latter results, through it is hoped that work at present proceeding or projected will help to throw some light on them. In one case of this kind the combined drag of body and ring, when suitably disposed to one another, was less than the sum of the drags of each in free air although the presence of the body might have been expected to increase the ring drag above its minimum value, by making the direction of the local airstream oblique to the ring section.

Applications. The chief result of the experiments has been the development of a method of reducing the drag of radial engines, which consists in placing a ring of aerofoil section round the nose in front of the engine and partly overlapping it. The aerofoil section adopted for the ring may vary widely from a cambered plate to a thick symmetrical section such as could be used for an exhaust pipe or silencer.

The magnitude of the reduction obtained with a given ring increases with the number of cylinders in the engine, at least up to nine. With 14 cylinders in two rows the reduction is practically the same as with 9. In the best case tested (with 9 cylinders) a reduction of drag was obtained equal to 60 percent of that of body and engine only (i.e., $R/R_o = 0.40$).

Full scale tests indicate that the cooling of the engine is usually unimpaired and in some cases improved. Considerable silencing may also be effected when the ring is used as an exhaust collector. The ceiling may be increased (*see* 13). The ring interferes but little with the accessibility of the engine, and involves no modification in the body whatever; in fact, it is probably that in most cases the spinner and in some cases even the crank case cowling may be discarded without appreciably affecting the drag.

The majority of these full scale tests have not been strict comparisons with the model tests reported here, although based on them; exceptions to this statement will be noted in the text.

Further developments. Full scale tests are in hand at the R.A.E. in which measurements are to be made of the effect of two typical rings on the performance of a Bristol Bulldog aeroplane (Jupiter engine).

The rings selected for test on this machine are the full scale equivalents of Ring J and polygonal Ring P_3 shown in Fig. 25 of this report with a slight modification to the angle between the chord and the body axis in the latter case.

The influence on the cooling of the Engine is also to be measured.

The experiments described in this report were commenced in December, 1927. The results include those given in the two earlier reports cited. The report is divided into two parts:

Part I. Interference on the drag of a model airship form (U. 721) caused by the presence near the nose of struts and aerofoils, and of rings of streamline section coaxial with the body, in various positions relative to the nose.

Part II. The reduction of the drag of aeroplane bodies fitted with radial engines in the nose by the addition of rings of aerofoil or streamline section. Some miscellaneous results are also given in this Part.

The two parts will be treated separately but the results of Part II will be presented in greater detail in view of their immediate application to the problem of the drag of the air-cooled radial engine.

PART 1
INTERFERENCE OF STRUTS AND RINGS OF AEROFOIL SECTION ON A MODEL AIRSHIP FORM (U.721).

1. *Introduction.—Origin.*—In the course of a test connected with a proposed investigation into the problem of airscrew body interference it was observed that the drag of a streamline body mounted in a wind tunnel was very considerably affected by the presence of a ring of streamline section surrounding it but independently supported from the tunnel floor. Although the diameter of the ring with which this observation was made was three times the maximum diameter of the body, and its cross section relatively small, the observed effects were large, the changes in drag being much greater than the total undisturbed drag. In view of this it was considered worthwhile to conduct a few experiments aside from the original program to investigate the effect, particularly as the drag was in some circumstances actually reduced—in one case to zero. With this object the following tests were made.

2. *Description of Tests*. (a) *Tests with Rings near a streamline body*.—The body upon which the experiments were made was a model of the airship form U.721 of maximum diameter 6 ½ in. and fineness ratio 4.622, having the low drag coefficient of

$$R/\rho V^2(\text{vol})^{2/3} = 0.007 \text{ approx., } (R/\rho V^2 d^2 = 0.0117)$$

The body was supported on wires from the roof of the (4 ft.) tunnel and from the main bottom balance in the usual way; the ring was mounted on a spindle attached to a large flat metal plate 3/16 in. thick which could be traversed along the tunnel floor. The axes of tunnel, body and ring were coincident.

In the initial experiment the diameter of the ring was 18 in. at its leading and trailing edges, and the chord of its (symmetrical) section 2 in. long. It was expected that the interference of so large a ring would be small; in fact the original tests was made with the object of verifying that it would be so. To accentuate its interference, therefore, the ring was made with a thick section only roughly faired (shown at γ, Fig. 1).

The results obtained with the first ring showed an unexpectedly large interference effect on the drag, a noticeable feature being that when placed in the neighborhood of the nose of the body the drag was *reduced*. The ring was then reversed, that is, placed trailing edge foremost, to make the section somewhat "worse". As this appeared to make the effect of the ring everywhere less, an attempt was made to improve the section by fairing it with plasticine, on the assumption that by so doing the effect would be correspondingly increased. This was found to be the case; so much so in fact that the minimum drag (which occurred when the nose of the body and the leading edge of the ring were coplanar) was reduced to zero (Fig. 1). As these changes were consequent solely upon a change in the section it was decided to test a ring having the section of a good strut of fineness ratio 2.67, but of only ¾ in. thickness (referred to as section α, *see* Fig. 1). The effect of this ring was somewhat smaller, no doubt due to its reduced thickness, but it exhibited the same general characteristics. For the sake of completeness three further rings were tested of decreasing cross sectional area, of the following sections:

(a) rectangular, 1/8 in. X 1½ in.
(b)circular, of diameter 5/16 in.
(c) circular, of diameter 1/8 in.

The results are shown in Fig. 1 where the drag of the body (in lb. at 60 ft./sec.) is plotted against the axial position, *x* (inches) of the ring relative to the body ($x = 0$ when leading edge of the ring is in plane of nose of body, and is positive when ring is entirely aft of this plane). It will be seen that the last 3 rings produce no perceptible *reduction* in drag anywhere, but that when amidships even the ring of 1/8 in. wire still has an appreciable effect.

The effects of these large rings (18 in. diameter), were observed for all axial positions of the ring relative to the nose of the body. Subsequent smaller rings were only tested in the neighborhood of the nose, where a reduction in the body drag occurs.

In view of the magnitude of the effects observed with rings of such large diameter, particularly when the sections were of streamline form, it seemed possible that interesting results might be obtained, and possible some insight into the cause of the phenomenon by varying

(a) the diameter of the ring;
(b) the orientation of the section of the ring (i.e., its incidence relative to the local streamlines);
(c) the size of the section.

However, to carry out such tests on rings would have been quite out of the question on account of the large number that would have been required, so it was decided to open out the ring, as it were, into a straight strut, which might logically be expected to give results of the same general nature as those obtained with the ring. A few rough preliminary tests showed that this was the case, although it was evident that when close to the body, the rapid change in the incidence of the strut along its length due to the great curvature of the streamlines near the body, as well as the rapid change in the distance of an element from the body surface, would preclude a strict comparison.

The substitution of struts for rings enabled (a) and (b) above to be varied with ease whilst (c) presented no difficulty, and the great resulting experimental simplification was considered to justify the strut tests apart from any interest they might have in themselves. A description of the strut tests is given in section 2.3 (b).

The strut tests suggested that the lowest drags would be obtained when the chord of the ring section (all the sections tested were symmetrical) was roughly parallel to the local direction of the streamlines. The nose of U.721 is of almost exactly the same shape as a 4 to 1 spheroid, and the streamlines and their directions near the nose, were calculated from the appropriate formulae for a spheroid given in Lamb's Hydrodynamics. The results are shown in Table 13 and Fig. 11.

Two rings (A and B) of section α (Fig. 1) were then made with [the] chord lines along the local streamlines when in position $x = 0$, [as] defined previously. These are shown in Fig. 2. The drag of [the] wings themselves was also measured by mounting them on a spindle in the bottom balance and fixing the body on a stand which could be traversed along the tunnel. The forces on rings and body separately for different relative positions are shown in Fig. 3. They are relatively unimportant however, as a result of adding the appropriate curves for body and ring shows that the overall drag is always more than twice that of the body only, but is, generally, considerably less than that of the ring alone, which is perhaps not surprising in view of the angle of the chord line for these rings. More interest attaches to the dotted curves which show the drag of the ring when seven radial cylinders are attached to the body. The modification in the case of Ring A is very marked; this feature is of practical importance and will be referred to in Part II.

2.1 *Effect of Rings on a Body having an annular excrescence.*—After the above tests were made a ring of rubber tubing (0.2 in. outside diameter) was nailed to the body in contact with its surface in the plane $x = 3.8$ in. The effect of the rings A and B on the drag of the body so modified was then measured. The results are not reproduced but may be expressed generally by remarking that the shape of the body drag curve (Fig. 3) with Ring B present is almost unaltered but the curve is raised bodily by about 0.055 lb. at 60 ft./sec. while the drag of the body with ring absent is raised by about 0.075 lb. The effect of ring B is therefore somewhat greater when the excrescence is attached than when it is not. Ring A on the other hand has a smaller effect when the excrescence is attached, the curve being raised to a greater extent than the drag of the body only and being somewhat flattened. The overall drag was not measured.

2.2 *Results obtained with Rings*.—With all rings tested the overall drag is greater than that of the body only, but is frequently less than that of the ring alone, and is almost always less (in the most favorable position) than the sum of the separate drags. This latter fact would be expected in the case of the smaller rings on account of the angle at which the chord line is set. There would also be a local change of pressure between the ring and the body giving rise to axial forces superimposed on the drag, properly so called, and in consequence the forces on ring and body taken separately have little significance. In any case the drag of the ring with the body absent would be high on account of the inclination of the chord to the free stream. It is, however, somewhat surprising that the same effect should also be observed in the case of the 18 in. diameter ring (section α), for in this case the section is already in the position of minimum drag when the body is *absent*, since the chord line is then parallel to the wind. It may be noticed that here the drag of each is reduced simultaneously, implying a reduction in the total turbulence apart from any pressure reactions. The introduction of the body would slightly increase the air speed at the ring, and so tend to increase its drag. No explanation of this effect is offered unless it is to be found in the fact that the ring section is somewhat critical or had its minimum drag at a finite angle of incidence. It would be worthwhile making some further search for the cause of this effect.

2.3 *Description of Tests*. (*b*) *Tests with Struts and Aerofoils*.—The struts under test were mounted on vertical spindles roughly coaxial with their centers of pressure, and were disposed in pairs, symmetrically on either side of the nose of the body. They were capable of being yawed about the axes of the spindles, which passed through holes in the floor of the tunnel. The upper ends of the struts were braced by wires to the walls of the tunnel. Attention was mainly concentrated on those positions of the struts which produced a reduction of body drag. The forces of the struts were not measured.

In Fig. 4 the sections and positions of the struts when at zero yaw, are shown in relation to the nose of U.721, together with the axis of yaw in each case.

It was found that the *change* in drag produced by two struts was almost exactly double that produced by one alone (*see* H and 2H, Figs. 4 and 5) when each strut was only 2 in. from the body axis; most of the tests were therefore made with one strut only.

The strut of section α (A to F, Fig. 4) was 3 ft. long; all the rest were 11 in. Aerofoils G, M and N were of Gottingen 429 section; aerofoils H, K and L were (approximately) R.A.F. 30 section.

A few figures bearing on the above remarks are collected in the following table:—

TABLE

Comparison of Changes in Drag (ΔR) due to Rings on U.721 with Analogous Tests in 2 Dimensions.

Centre lines of cylinders 4·0 in. behind nose.

V = 60 f/s.

Position of Ring Section.* x	y	θ degrees.	No. of Cylinders.	ΔR. lb.	ΔR, Interpolated for 8 cylinders. lb.
Ring A.					
0·20	3·65	8	7	0·109	0·140
0·30	3·65	8	9	0·178	
Analogue of Ring A in 2 dimensions. (a) *With Dummy Ends.*					
No reduction in drag obtained; results critical.					
(b) *With End Plates.*					
0·30	3·70	14	8	0·119	0·119
Ring E.					
0·65	3·50	11	7	0·109	0·155
			9	0·183	
Analogue of Ring E. (a) *With Dummy Ends.*					
0·75	3·60	17	8	0·153†	0·153 †interpolated between 2 values of y.
(b) *With End Plates.*					
0·75	2·60	14	8	0·204	0·204
		17	8	0·188	0·188
Ring G.					
0·25	3·80	8·5	7	0·164	0·201
			9	0·239	
Analogue of Ring G. (a) *With Dummy Ends.*					
0·25	4·00	14	8	0·162	0·182
(b) *With End Plates.*					
0·25	4·00	14	8	0·212	0·212
0·25	3·80	18	8	0·284	0·284

* x = distance of plane of trailing edge of section in front of centre lines of cylinders (ins.).
y = distance of trailing edge from axis of body (ins.).
θ = angle between body axis and chord of section (degrees).

Note.—The rings are not necessarily in the best positions, but in the positions corresponding to the strut results.

2.4 *Results obtained with Struts*.—The results of the foregoing strut experiments are shown in Figs. 5 (*a*) and (*b*), where the drag of the body (lb. at 60 ft./sec.) is plotted against angle of yaw of strut relative to the tunnel axis. The angle of yaw is positive when the leading edge of the strut is nearer the axis of the body than the trailing edge.

It was thought unnecessary to tabulate the large number of readings taken, as the curves were well defined including the critical region near—5° yaw.

The following points may be noticed:

(1) the reduction in drag becomes less as the aerofoil is moved downstream from the plane of the nose.

(2) Large reductions may occur even when the aerofoil is some distance from the body laterally, see A, Fig. 4.

(3) Aerofoils C and G, though of widely different sections and slightly different lengths of chord, produce, in the same position, nearly the same effect.

In addition to the above a few tests were made with a pair of aerofoils of Gottingen 429 section 11 in. long similar to M, Fig. 4, mounted on a stand in such a way that they could be set at any angles of yaw and could be traversed together along the tunnel over a [range] of *x* near the nose.

The axes of yaw, which were situated at ¼ chord from the leading edge of the aerofoils, were spaced 3.35 in. apart transversely. The drag of the body was measured for angles of yaw of both aerofoils of 15°, 25°, 35° and 45° over a range of *x* which included minimum drag.

The results are shown plotted in Fig. 6. The chief points of interest are as follows:

(1) The reduction in drag continues to increase up to quite large angles of yaw.

(2) For a given angle of yaw, the reduction in drag continues to increase as x increases until the aerofoils are quite close to the surface of the body. Minimum drag occurs when the minimum clearance is of the order of 3/16 in. On closer approach the drag becomes positive before actual contact occurs between aerofoils and body. There is therefore a stable relative position, in which the aerofoils, though not touching the body are "towing" it against the wind.

PART II

THE REDUCTION OF THE DRAG OF RADIAL ENGINES

3. *Introduction*.—In paragraph 2.2 (Part 1) tests were referred to in which the interference between a ring and body was negative, i.e., the combined drag was lower than the sum of the separate drags. This effect is not uncommon when the interfering bodies have a high drag, although in such cases the explanation is usually that one of the bodies tends to shield the other. In fact, ordinary fairing is an extreme example of this. In the case of the 18 in. diameter ring of section α already referred to (2.2) the two interfering objects not only did not shield one another but were low drag forms to start with, and yet their mutual interference was negative.

In seeking for some practical example in which advantage might be taken of this effect it was recalled that many aeroplanes powered with radial engines are provided with annular exhaust pipes running round the nose of the machine just in front of or behind the cylinder heads. In normal circumstances, such exhaust rings are made of circular or elliptical cross section and usually add quite considerably to the already high drag of the radial engine. Even when made of streamline section they increase the drag *as normally fitted*.

There was however, in this possible application one serious point of difference from the previous experiments which it was thought might vitiate the application of the results to the case when the radial engine was present, and that was the influence of the cylinders on the air flow; but as the ring had been found to be still effective when an annular excrescence was attached to the nose of the streamline body in the position which would be occupied by a radial engine (see 2.1) it was considered probable that this case also would be amenable to similar treatment, by a suitable modification of an existing exhaust pipe or by the addition of such a pipe when not already fitted. With this objective in view it was decided to make a few preliminary tests.

The results of these tests were very encouraging, and showed at once that appreciable reductions in drag were obtainable. They also disclosed the mechanism by which the ring produced its effect (in the case of the radial engine) and this information was used to design other rings giving much better results.

In order to render the description of the tests more intelligible it may be as well at this point before describing them to anticipate the results, for the purpose of giving an indication of the action of the ring.

3.1 *Function of the Ring*.—The distribution of streamlines around the nose of a streamline body, when it is not fitted with a radial engine is such that the flow continually converges towards the surface of the body as it proceeds from the place of the nose to that of the maximum cross section. When the cylinders of a radial engine are placed in their usual position in the nose they disturb this ideal condition and produce a violent deflection of the airstream away from the surface, this being so not only in the case of air which directly strikes the cylinders, but also of the air which passes between them. This effect can be observed by discharging a jet of smoke in front of the gap between two cylinders.

If by some means this divergence of the streamlines could be obviated it might be supposed that the inevitable disturbance produced by the cylinders themselves would be prevented from expanding to form a large turbulent wake, with a consequent reduction in drag. Now the attachment of a streamline ring is a method of effecting this, since it consists in placing immediately in front of the cylinders a curved aerofoil disposed in such a way with respect to the cylinders and the local direction of the flow as modified by them that the aerofoil section, working at a fairly high lift coefficient, secures control of the air before it is thrown outwards by the cylinders, by exerting a "downwash" directed towards the body axis which prevents the disturbance caused by the engine from expanding radially to produce a large wake. From smoke observations (visual) with and without a ring it was evident that the ring reduced the radial thickness of the wake behind the cylinders by something like 40 percent.

Some further visual experiments were made on a very small (dimensional) model in a water channel, in which the changes in flow were made visible by a jet of dye discharged in front of the model. Photographs of the flow are shown in Fig. 29. In each photograph the jet is in the same position. In the first (*a*) the flow past the body only is shown. In (*b*) two exposures were made on the same plate, the first a repetition of (*a*) and the second with the cylinders in position; the line of ink shows the flow between two cylinders. In (*c*) there are also two exposures on one plate; the first is a repeat of (*b*) with cylinders, and the second is the flow with an aerofoil attached in addition, to represent a ring.

4. *Description of Preliminary Experiments*.—In searching for a typical aeroplane which should have a body similar in shape to the model (U.721) already used for the experiments of Part I, an inspection was made of numerous photographs appearing in "Aviation." It was found that the U.S. Navy "Apache" biplane fitted with a "Wasp" radial engine had a fuselage the shape of which in side elevation was almost identical with that of U.721 up to the maximum diameter. From a few rough measurements of one such photograph seven model cylinders were made and attached to U.721 in the appropriate position. The cylinders were made to fit the surface of the model closely and each was held on by one central pin so that it could be removed and replaced easily. The shape of the cylinders and their situation in the nose are shown in Figs. 2, 13a and 14. They were very simple in outline and were fitted with two dummy cylindrical valves apiece.

An initial attempt to discover the most suitable angle for the ring section relative to the body axis under the new conditions was made by using 4 cylinders only, a pair on each side of the body, each pair being spaced as though forming part of a radial engine. Behind each pair was fixed a segment of a ring of symmetrical streamline section (Section β, Fig. 2). Each segment was attached by two pins (diameter 0.034 in.) to the body, so that by bending the pins the angle of the chord to the body axis could be varied to some extent. The segments were set to various angles and the drag was observed and plotted. It was found that there was a minimum which was appreciably lower than the drag of the body and cylinders alone. When placed just behind the cylinders the best angle of the chord was found to be 3° and when placed in front the best angle was 14°. These angles apply, of course, only to the axial and *radial* positions of the ring section shown as C and D, Fig. 2

Similar tests were made with segments of circular cross-section to imitate the more usual type of exhaust pipe. From photographs, the appropriate diameter was judged to be about 0.58 in. These tests showed that the drag was very considerably increased. The streamline section β was designed to have the same cross sectional

area as that of a 0.58 in. diameter circle. The clearance between the segments and the body was about ¼ in. to 3/8 in.

Two complete rings (C and D, Fig. 2) based on the above tests were then made and attached to the body with 7 radial cylinders in the same plane as before. The results agreed closely with expectations based on the segment tests, as did also tests made with rings of 0.58 in. diameter circular cross-section.

Several rings (A, B, C and D) were then tested for drag in the presence of the body (with 7 cylinders), and also the body and cylinders in the presence of the rings. The results (Table 7) are shown in Figs. 7, 8 and 9. The curves for ring and body are added graphically, the sum being shown by the dotted curves, while the points with tails are overall observations made with the rings actually attached to the body. The drag of the rings has been corrected for spindle effect which was measured by means of a dummy and in which there may be slight uncertainty. The important (overall) observations are not subject to this uncertainty. They have been corrected throughout for static pressure drop, and drag of wires and spindle, but errors in these corrections do not affect the comparisons. The forces have not been reduced to coefficients but are shown in lb. at 60 ft. per second wind speed. Scales of R/R_0 (where R is drag of body and cylinders with ring attached, and R_0 = drag of body and cylinders only) are drawn on the right hand sides of the figures.

In addition to the ring of 0.58 in. diameter circular section, rings of circular section were also tested having the same frontal area as that of rings C and D, i.e., of 0.3 in. radial depth, in the same relative positions as C and D respectively. These results are plotted at $x = -3$ and at $x = 0.1$ in Fig. 7.

Some further tests subsequently made on this model with other rings are described later together with the results of more extended experiments.

5. *Description of Extended Experiments*.—As the initial experiments were sufficiently promising to warrant a more thorough investigation, experiments were instituted on larger models in which the details of the engine cylinders were represented more accurately.

Since the circulation around the aerofoil section of the ring associated with the lift would produce a velocity tending to oppose the flow between the cylinders which might be important in connection with the cooling, a few observations of the change of air speed between the cylinders were made with a small pitot tube with various rings in position.

One model was fitted with an airscrew, which enabled measurements to be made of changes in net thrust, torque and efficiency due to the ring.

The models tested were:

(1)U.721 model described already (*see* A, Fig. 13).

(2)¼ scale model of Short Crusader Seaplane (9-cylinder engine) with and without wings and floats (D), Fig. 13 and Fig. 28.

(3)¼ scale model (modified) of body of Bristol Bulldog with 9-cylinder engine, with and without airscrew (C), Fig. 13 and Fig. 27 (*a*) and (*b*).

(4)1/5 scale model of Siskin aeroplane body with Jaguar engine (14 cylinders in 2 rows) (E), Fig. 13 and Fig. 27 (*c*).

(5)1/6 scale model of body with monoplane wings and Lynx engine (7 cylinders) (B), Fig. 13.

5.1 *Two-dimensional Experiments*.—Before commencing the above tests, it was deemed worthwhile to obtain, if possible, some general qualitative information about the effect of varying the chord and section of the ring and the angle of the chord to the body axis, by an adaptation of the "strut" method described in Part I (*b*), in which not only the ring but the body also would be reduced to 2 dimensions. For this purpose a model was constructed of exactly the same sectional shape and size as that of U.721, Fig. 13A, but of approximately 2-dimensional form. A sketch of this, as mounted in a 4-ft. tunnel is shown in Fig. 10. The "span" of that part on which the drag was measured was 10 in., extended by a further 10 in. at each end by dummies attached to the roof and floor of the tunnel and separated from the ends of the model proper by air gaps [1/14] in. wide. It was considered unnecessary to carry the dummies as far as the roof and floor; the upper dummy was slung on radius rods and was capable of being swung out of position to allow of easy angular adjustment of the aerofoils by which the ring was represented. These aerofoils were attached to the model by two flat brass strips provided with a row of holes ¼ in. apart through which screws passed into the ends of the aerofoils. The angle of an aerofoil was therefore adjustable about an axis whose position was itself adjustable by varying the position of the brass strips in the body and by selecting suitable holes in the strips.

The cylinders were represented by the same simple forms as were used on U.721. They were fixed in two rows of four cylinders each, one row on each side of the body in the same position relative to the nose as for U.721. They were separated laterally by a distance equal to their mean circumferential separation in the U.721 case with 7 cylinders.

In order to preserve the interference between the cylinders each row was extended beyond the air gaps by two cylinders at either side and at each end, fixed to the dummies. The aerofoils however were not extended in this manner.

5.2 *Relation between Tests in 2 and 3 Dimensions*.—It was realized that the best angles for the aerofoil sections would not be the same in the 2 and 3-dimensional cases, and some idea of the difference was sought by calculating the slopes of the streamlines at points in the vicinity of the aerofoils for each case on the assumption that the flow near the nose of the body would not be seriously modified by the tail being tapered instead of elliptical. The formulae used were derived from those given in Lamb's Hydrodynamics relating to the stream functions ψ for the elliptical cylinder and ellipsoid respectively.

To illustrate the change in angle the streamlines are shown in Fig. 11 for the two cases—for equal increments of ψ in the case of 2 dimensions, and in 3 dimensions for such values of ψ as make streamlines intersect those for 2 dimensions at points in an axial plane 2 in. behind the nose. The difference in slope between the 2 and 3-dimensional cases is given in Table 13 for various positions near the nose. In the important region the slope in 2 dimensions is about 6° steeper than it is in 3 dimensions. An inspection of the observations given in the tables will show that in cases where the ring and aerofoils correspond closely in other respects the value of θ for minimum drag in 2 dimensions is about 8 or 9 degrees greater than in 3 dimensions. A rigorous test would have required 3 or 4 rings differing only in the angle θ, and this was not considered worthwhile.

5.3 *Drag in 2 dimensions*.—The drag of the 2-dimensional model with and without cylinders was of course very much higher than that of U.721, and therefore the value of R/R_0 has very little significance in this case. It was found however that the *change* in drag (ΔR) obtained by the addition of a pair of aerofoils at their best position corresponded somewhat closely with the change produced in U.721 by the addition of the corresponding ring, the agreement being better than might have been anticipated in view of the inevitable dissimilarity of the two models. This encouraged some confidence in applying conclusions drawn from 2-dimenstional tests to rings with respect to the effects of chord, section, variation in the angle θ and to a lesser degree the value of θ.

A few figures bearing on the above remarks are collected in the following table:

TABLE

Comparison of Changes in Drag (ΔR) due to Rings on U.721 with Analogous Tests in 2 Dimensions.

Center lines of cylinders 4.0 in. behind nose.

V = 60 f/s.

The 2-dimensional analogue of Ring A was tested to determine whether such tests would bear out the results already obtained with this ring on U.721. The result $\Delta R = 0.119$ was considered to be sufficiently close to the value $\Delta R = 0.140$ obtained with U.721 to justify the method; other sections were therefore tested in 2 dimensions and from the results Rings E and G were designed. The subsequent tests with these rings bore out the 2-dimensional results quite closely.

With regard to the angle θ later tests suggest that this should be somewhat smaller (by 3 or 4 degrees) than that obtained by applying to the angle observed in 2 dimensions the theoretical correction previously referred to.

A few further tests were made in 2 dimensions with 2 rows of cylinders, one behind the other, to imitate the "Jaguar" type of engine. In this case the front row consisted of the same 4 cylinders per side as were used before in the same positions, but in addition 3 cylinders per side were added behind the spaces between cylinders in the front rows. Four cylinders only were attached to the dummies, one opposite each end of the front rows of cylinders on the model.

Similar results were obtained with this arrangement as with a single row of cylinders.

6. *Tests of Aeroplane Models*. (*See* Fig. 13.)—6.1. *Crusader (Scale)*.—After the preliminary work on the U.721 model already described, more detailed experiments, the results of which will be discussed together later, were made on scale models of bodies of existing aeroplanes (Fig. 13) in which the details of the engine were accurately represented.

The first such model used was the ¼ scale model of the Short "Crusader" high speed seaplane fitted with a 9 cylinder radial engine ("Mercury"), a model which had been previously tested in connection with the Schneider Trophy Seaplane Race. A photograph of this model is shown in Fig. 28 and a sketch in Fig. 13(D). The nose is shown on a larger scale in Fig. 15. The following tests were made on this model:

(1)Body and engine only, with normal cylinder fairings, cockpit, and pilot's fairings.

(2)Complete machine with wings, floats and tail unit, with normal fairings.

(3)Tests similar to (1) and (2) but with the individual cylinders fitted with dummy helmets. *See* Fig. 13D.

(4)All three tests were made with and without various rings, and in addition to measurements of forces, a few observations were made of velocity between the cylinders.

A visual study by means of smoke was also made of the effect produced by the ring on *the* wake behind the cylinders.

The results (R/R_0) obtained in test (1) with Ring A, agreed closely with the corresponding results for U.721 with 9 cylinders. In fact it became clear as the experiments proceeded that the magnitude of the reduction in the drag obtainable was not sensitive to the exact shape of the body, although the actual *design* of the ring for any particular case was dependent on the shape of the nose and disposition of the engine in relation to it.

6.2 *Aeroplane Body with "Lynx" Engine (1/6 Scale)*.—This was a model of an aeroplane body of circular cross-section fitted with monoplane wings of 5:1 aspect ratio springing from the center of the body. A detailed drawing of the nose and engine of this model is shown in Fig. 14, superimposed upon a sketch of U.721 and cylinders, with which the general shape and size corresponded fairly closely. The nose cap (A, Fig. 14) was removable, the normal outline being shown by the dotted lines, but for the purposes of the present tests it was considered fairer to substitute the nose cap A as representing the form of cowling usually fitted to such engines.

The drag in this condition is taken as standard (= R_0). (For body and engine only it is 7 percent less than that of the shape shown dotted.)

Some previous experiments had been made on this model at the R.A.E. with helmets fitted to the individual cylinders which were designed to give sufficient cooling (*see* Fig. 13B). It was therefore possible to compare the relative merits of helmets and rings for the case of 7 cylinders as well as for 9 (i.e., on the Crusader).

The tests made were:

(1)Body and engine only *

(2)Body and engine and wings *

(3)Same as 1 and 2 but with helmets instead of rings.

* with and without rings.

6.3 *Bristol Bulldog (modified), ¼ Scale*.—The body of this model was a solid of revolution in which the shape from the nose to just behind the engine was the same as the Bristol Bulldog. The outline of the rest of the body was a rough mean between the outlines in plan and elevation of the Bristol Bulldog.

The engine was represented by the same cylinders as used in the Crusader, but the Mercury valve rockers were removed and their places taken by dummy rocker hats of the type used on the Jupiter engine. The cylinders were so adjusted radially that the outside diameter was equal to that of the Jupiter.

Fig. 15 shows the engine in position on the Bulldog model. Superimposed is a sketch of the nose of the Crusader with the position of the Mercury valve rockers shown dotted.

6.4 *Airscrew Tests*.—In the case of the Bulldog, tests were made with the airscrew running. The screw used was a 2-blader which happened to be available and of approximately the correct diameter. The pitch was somewhat low (geometrical P/D approximately equal to 0.66), but this was considered unimportant as it was only desired to ascertain whether the screw would alter, to any appreciable extent, the character of the results obtained with various rings.

The screw tests consisted of the measurement of net thrust and torque. The apparatus used, shown in Fig. 16, was similar to the standard arrangement so far as thrust was concerned, which was measured on a bottom balance. The torque balance was somewhat modified however. The motor was carried at the rear on a cross beam A fixed to the body. At the front it was hung directly from the torque balance B on the roof by means of 2 parallel stranded cables C, attached to the torque arms on the motor casing. The torque arms were completely enclosed within the body. To prevent lateral motion of the motor relative to the body, the nose of the motor was constrained between 2 vertical strips of metal (not shown) attached to the inside of the nose of the body. A slight clearance was allowed between the strips and the motor to avoid torsional constraint. Lateral motion of the body was prevented by the V-wires (E) by which it was slung. These were arranged so that, if produced, they would intersect on the airscrew axis, in order to prevent any lateral forces on the motor from causing the body to roll. Current was led to the motor through mercury cups, M, at the tail of the body and through springs near A (not shown).

In all tests the normal working range of V/*n*D was covered. The motor was controlled on torque, and rotational speed was measured with a chronograph. The rotational speed (*n*) was kept nearly constant over the whole range of V/*n*D at about 43 revs. per second, for most of the tests, and the tunnel speed was varied from 80 f.p.s. to 35 f.p.s.

The chronograph was found to be more satisfactory than a stopwatch for measuring rotational speed. The extra accuracy on *n*, which affects the results virtually in proportion to n^3, was considered worth having; with the chronograph the thrust and torque curves were found to be noticeably smoother than with a stopwatch.

The following tests were made with the modified Bulldog model:

(1)Body and engine only with and without airscrew, with various rings.

(2)Body without engine, with and without airscrew.

(3)Velocity between cylinders. Rough measurements only for comparison between different rings.

6.5 *Siskin Aeroplane Body with Jaguar Engine (1/5 Scale)*.—Through the courtesy of Messrs. Sir W. G. Armstrong Whitworth Aircraft, Ltd., who loaned this model, it was possible to carry out experiments with rings on an engine having two rows of cylinders, the cylinders in the rear row standing opposite to the gaps between the cylinders in the front row. Viewed from the front this engine presents almost a solid disc to the airflow. In spite of this however it was found that a ring was practically as effective in reducing the drag of this engine as of engines having a single row of cylinders.

Details of the model are shown in Fig. 17. The upper half of the figure shows the ungeared Jaguar without rocker hats, whilst the lower half shows the geared model, which was fitted with rocker hats. Two typical rings are also shown in the figure. As this engine was fitted with an exhaust manifold behind the cylinders, no tests were made of this model with rings of the type of Ring A, Fig. 2.

The following tests were made:

(1) Body (with windscreen) and engine only (ungeared), with several rings.

(2) Check tests as (1) on the model with geared engine and rocker hats.

(3) Test with and without spinner and exhaust manifold.

6.6 *Scale Effect on the Drag of Engine Cylinders*.—The drag coefficient of an infinite cylinder is known to undergo a rapid drop in the neighborhood of a certain value of Reynolds number, V*d*/*v*, lying between the model and full scale values for cylinders of the diameter of normal engine cylinders.

It had been suggested by Mr. Relf in Ae. Techl. 399 that though an engine cylinder was very different from an infinite cylinder yet it might be reasonably expected to exhibit some decrease in drag coefficient near the same value of Reynolds number.

As the application of a ring to an engine partakes of the nature of a curve for the turbulence caused by the cylinders it was concluded that any serious difference in the degree of turbulence produced by the cylinders on model and full scale would show itself by a corresponding scale effect on the improvement in drag obtained by adding the ring, which would thus be less effective on full scale than on a model.

A few experiments were therefore made to determine the nature and extent of the scale effect near the critical value of Reynolds number on a very short cylinder projecting from a place surface. The results are quoted below.

Three cases were tested:

(1)8 in. diameter cylinder 10½ in. long in center of 4 ft. tunnel.

8 in. diameter cylinder 10½ in. long with false floor nearly in contact with one end of cylinder.

Document 3-10(b), United Aircraft and Transport Corporation Technical Advisory Committee, Meeting Minutes, December 1929, Boeing Historical Archives, Seattle, Washington, pp. 213-256.

PROCEEDINGS OF TUESDAY AFTERNOON, DECEMBER 3, 1929

The meeting convened at 1:30 o'clock P.M. pursuant to noon adjournment.

CHAIRMAN MEAD: Well, while we are waiting on Colyer and Humphries to come in, Mac has given us a sheet of airplane characteristics and I know Chatfield and the rest of us have been worrying for some time for a way to get this in a standardized form that will be acceptable to everyone. It might be a good idea to pass that around to see if we can get constructive comments.

MR. MONTEITH: Instead of that table of performances, I would rather see curves. It really makes two sheets out of it, but you really stop to plot this thing up in curves anyhow.

CHAIRMAN MEAD: That reminds me,—it may work everybody if we have these days pretty well filled up, but if you can get your gang together in the matter of performance testing of course, I have it down for Friday but we may all of talk by Friday we won't give it the proper consideration,—and see if we can't light on some system that looks reasonable to try out, then we can modify it in six months, but here is Northrop testing down at his place and we are testing in Hartford, and we hope Mac is going to be testing sometime, and it will be nice to get down to earth if we could.

MR. MONTEITH: Well there is not much trouble in the speeds if we get a decent speed course, which we do not have at the present time,—but practically everybody whom I have run into at all in the industry is making his climbs with an altimeter, which is the bunk. I think something we need every place is a good barograph with a calibration apparatus to go with it if we are going to get any decent climb data at all.

CHAIRMAN MEAD: Last night we had a supplementary meeting over at the Hotel and Mr. Sikorsky was putting up a method of measuring landing speed, and that is another thing we are worrying about.

MR. MONTEITH: I just had a letter from Reid down at Stanford. He wants us to loan him some planes: He has a method of testing landing speeds down there by photographic methods. He paints one wheel half white and half black and measures it with his photographic instruments.

CHAIRMAN MEAD: It would be nice if we could get an instrument [that is] not too expensive, but also simple and durable we could use on any of our ships either land or water. Mr. Sikorsky was telling us about a different method that might be worth while if he would explain that again.

MR. SIKORSKY: Certainly, I would be glad to. You see, here is the size we use (indicating on blackboard). This is a box about actual size. This part here is a piece of glass. There is an arm here inside with a visible hand here, and at a place somewhere about here,—of course there is an opening in the top to fix this up,—now here, there is sticking a support on which a disc of twenty-five centimeters diameter is fixed. How, there is existing very reliable data of the pressure on a disc of any dimension may be applied, but it is very easy to graduate it and fairly reliable information is available on the pressure at every speed.

Now the whole thing is to adjust the load here to such an extent as to show your desirable speed. Now, of course, all you can get here would be whether your landing is, say better than 54 or whether it is worse. In other words you have to check it a couple of times and sometimes you have to adjust the load a couple of times until you get the actual thing, but we make a multitude of tests and I would say that I have the full conviction that the thing is accurate within one mile. For instance, when we tested several, or tested the same airplane with several loads then the increase of landing speed went simply mathematical in accord with what it should do.

The thing we tested was the ship actually landing. In this case, of course, a second person must make the observations or it can be tested by the pilot himself. In this case, it is very easy to keep the thing just bouncing back and forth, or even to keep it down. Of course, here it is to keep it down which shows the speed is below a certain limit to which the instrument is adjusted.

The thing is very simply and cheap to construct.

McCARTHY: How about the effect of the angle line on the pressure?

MR. SIKORSKY: Here is the thing: The slight differences, even a good few degrees,—as far as I remember two or three or five degrees make a very little difference for the disc, therefore, of course, you put the disc, to be normal, to the wind and [approximate] your landing speed. Now there the advantage is, you need no correction whatsoever, if you do it [at a] thousand foot altitude the thing will show you,—the correct speed actually may be 63 miles per hour but it will show you 64, just so the corrections are mathematically taken care of.

CHAIRMAN MEAD: The thing seems to me we are interested in landing speed, otherwise we do not appreciate the top speed. If the landing speed is high, the top speed is not so interesting.

Well, it looks as though we have done about all we could with this matter of transportation. Of course, we would like to analyze this matter of maintenance as we said this morning, and probably we can in the near future in view of what you are doing.

The comparisons of performance for this six months with the last six months we can perhaps make graphically for you when we get the data, and of course you should appreciate that Chatfield and some few people in Hartford are now assigned to help us on this sort of work so as not to make too much of a burden for the individual units. If you can furnish the information, we can sometimes put it in shape and save you some work, but to gain anything out of this data of course it has to be in comparative shape all the time and I think we will find that we are going to be interested in fewer items than we have discussed this morning that we can substantiate those figures in the report so we don't need to take the time to discuss them each time, and we can show graphically on a chart prepared ahead some of these things in a very few minutes.

Well, the next item I have down for discussion is that of power distribution, believing that it was one way to get at the possible improvements in plane performance.

Now, Chatfield had better explain to us his story I think he sent around, and obviously, you can criticize considerably because we have not very much data to work with.

MR. CHATFIELD: I suppose the main objection to this is that based on a small machine, a Sperry Messenger which has only three cylinders, and in order to get it comparable to faster machines, the speed had been stepped up to a speed much higher than this airplane actually had.

The results are obtained by a sort of combination or the NACA reports, the technical report 304 which gave the flight polars for a complete airplane with four complete sets of wings, and technical report 271 which gave the driver of the component exclusive of the wings. The results are nothing more than a combination and reconciliation of these two, and I suppose I might put the percents on the board as being the easiest way of showing them.

(Mr. Chatfield puts tabulated list on blackboard)

This of course, also involves allowance for power lost in propeller so the total figure is not a distribution of drag but a distribution of brake horsepower, and it is break horsepower accounted for in these various ways.

That, I think shows it reasonably well. Of course the propeller is based obviously in propeller efficiency and in this case would be 2 percent.

Of course one surprising thing which probably is exaggerated as I said, in a very low portion which is accounted for by induced drag which might be extreme but you might say only 1.2 percent of brake horsepower is going to hold, the rest of it is all absorbed by various resistances and losses, but as mentioned before I put this account of having jacked up the speed. The induced is normally low and I imagine would be probably twice at any rate, or perhaps somewhat more than that, but still would give the result, the induced drag is a relatively small part of the total.

Of course the most conspicuous item is immediate-interplane bracing and interference which in a sense is the stuff you can't find somewhere else.

MR. MONTEITH: Pardon me, but in that particular ship do you recall that the interplane bracing was made of struts?

MR. CHATFIELD: Yes, the interplace bracing was struts in the particular machine, so possibly that again is somewhat higher than normal.

I might say the only justification for picking this particular airplane was because it was one from which we could get the data; I don't think there is any other argument to be advanced in favor of this set of figures.

Again, due to the engine this addition is undoubtedly abnormally low because it was a three cylinder engine and we don't have many three cylinder engines, and as Mr. Weick has also pointed out, if you went up to nine cylinders not only would the total engine drag be multiplied by the ratio of the number of cylinders, but the drag per cylinder would probably be higher as well on account of the interference.

MR. MONTEITH: Well, there is another feature there, Mr. Chatfield, too that in that particular airplane that was the first air-cooled engine we had you know, and the idea was to put all the cylinders out in the air.

CHATFIELD: It is conventional cowling of the old style with the cowling coming pretty well down to the bottom of the cylinder barrel and not up toward the head as is most common today. Of course, this was all several years ago.

Apparently what goes on inside the engine disc does not make much difference as regards resistance, isn't that the general conclusion?

MR. SHORT: That is what I judge from our tests all right.

CHAIRMAN MEAD: How much improvement do we get?

MR. McCARTHY: You get one mile more an hour high speed with the monoplane than you did the biplane.

CHAIRMAN MEAD: You must have a lot of miscellaneous in this 33.8, otherwise there ought to be a big gain with any monoplane.

CHAIRMAN MEAD: Have we any data in the family which would be useful or can we get it anywhere else which would bring this story any more up to date?

MR. CHATFIELD: Mr. McCarthy has one much more advanced than this.

MR. MONTEITH: I might state that in that particular model we made for the NACA, I was instrumental in building this set of wings for the test. We had one set arranged with internal bracing so we could put the struts in or take them out, but they let Doctor Munk get hold of it finally and he held it so long we never did get the complete test. We kept trying to find out what that interplane bracing interference amounted to on the ship but we never got the data.

MR. SIKORSKY: I would like to have the data concerning this picture. The trouble of course about discussing this thing will be that there would be one equation with quite a considerable amount of uniform factors,—for example, I think the picture here and this figure (33.8) certainly is very correct and probably is much greater, and yet the bracing and interference that is shown here would permit you to save considerable on some other items. Here the induced drag stays as a low item, but if you take a heavily loaded airplane,—if you take a big one, if you consider the case that you are flying with one unit out of commission, in other words, fairly slow speed, in most cases with more multi-engined ships it would be something like 75 or 80 miles per hour, then the figure of induced drag becomes a more considerable one, and to take care of it you need span, in fact nothing can be replaced without span.

In order to give the value of span, I may, for instance, give you the following very accurate information: We built an S-36 amphibian with a gross weight of approximately 6,500 pounds, with lots of bad features dynamically, and this ship was powered with two Wright motors of 200 horsepower each. With one engine dead it showed a loss of altitude of 400 feet per minute, in other words, an enormous lot of altitude.

This ship had 62 feet span. Now we cut the middle part of the wings, added 5 feet on each side and transposed it into [a] 72 feet span airplane, and with the same load, in other words, with greater gross weight than we had, we only showed a loss of 100 feet per minute instead of 400 feet loss previously.

It should be taken for granted that with all other conditions being equal, especially in a big ship, you can afford a bigger span if you use outside bracing than if you will use cantilever sings. This is the thing which to a very large extent permits all of us to profit in the externally braced airplanes, and this makes for the fact, as a general rule,--I am again speaking mainly of the big ships, the clean cantilever monoplane did not show loss of speed or anything more in general than the braced external biplane.

Again, I say this is true for very huge units. In very small ones, it seems logical as the little ships must be given very big preferences.

Now, I hope Mr. Mead will forgive me for the following statement: This is, I believe that what we actually have to experience in dragging across the air the radial engine, of course, the engine does it by itself, we want to get more out of it,—more according to our form of what would be shown by this figure. For example, a very accurate wind tunnel test I made for a very huge engine, the engine nacelle was a little over 5 feet, about 5 ½ feet in diameter. The Jupiter motor was very well streamlined, or rather cowled in so far that the cylinders were sticking [not] much here (indicating). As a matter of fact, if you would use the cross section, the engine would occupy a very inferior part of the circle.

Now in this case, the drag was distributed as follows: This whole business, one-third, and the engine cylinder heads, two-thirds.

Now, again a very large amount of tests, which were entirely reliable and accurate because we have several dozens of tests on our airplane cylinders; they show in general the following things, that the same cylinder which gives us an L.D. of 11 will give us the L.D. 18, so here we have a considerable amount of power going in because in this case it is already the difference of the whole airplane plus the induced drive and everything. Now, however, the windtunnel test showed a considerable improvement due to the cowling or a similar combination. For example, the cylinder which showed L.D. 11—the best lift drag could be pulled to 14 ½ only by specification of the NACA cowlings, or rather similar cowlings because we cannot say on the model what kind of cowlings we put in and how good they would build the actual airplane, so I believe that a very important thing and a very big gain for us would result from a very careful study of these very cowlings. That is where we can save a considerable amount of power.

Again, I think this is more true for exposed engines, but probably it is largely correct also for single engine airplanes, and I believe here in this close corporation with the help of the United facilities, we may be enabled to make a considerable step ahead because this here is quite a big item in which we can save lots of power.

Now, I believe maybe this may be an item which I say is general for all of us, we may expect to use the radial engines on probably most of the designs because they are so much more superior and convenient, and the operating cost is less, and people in those cases want them. The [airline] Pan-American, for instance, simply do not want to talk or to hear about anything except the radial air-cooled engine. That is the situation right now, so it seems a thorough study of all these things may be very justifiable and will do us all some good.

MR. NORTHROP: One thing might be mentioned in what Mr. Sikorsky had to say relative to interplane bracing, and interference with a large span. I have noticed in the high speed fairly small span planes I have had experience with the take-off and climb are not nearly as good in proportion to the rest of the performance as high speed, in other words at points where the induced drag would normally we shall say, the ship is highly efficient over a biplane arrangement, but in take-off and climb it is no better and in some cases, distinctly inferior to biplane arrangement.

CHAIRMAN MEAD: Of course, what interests us most is what sort of improvement we could make in those figures. I wish they were more representative of what one of our present air engines does, then we would know much better about what we are working with.

MR. McCARTHY: Let's put up some figures which will compare with this.

CHAIRMAN MEAD: Have you some there, Mac?

MR. McCARTHY: Yes, I have about five or six different airplanes here.

CHAIRMAN MEAD: Can't we take an average and set them up on the board?

MR. McCARTHY: I can put down about three. (Marking figures on the blackboard).

MR. SIKORSKY: Some of these are water-cooled jobs and some are air-cooled. This one is an XO3-1, that is a two-seater observation plane with a geared 1570 motor on it, and estimated high speed of 182 miles an hour. The total wing drag, I did not separate them, about 35.6.

Interplane struts, wires and interference is only 8.3; fuselage is 14. 1.

The radiator 6.

CHAIRMAN MEAD: Including the engine?

MR. McCARTHY: Yes.

Tail surface 8.7. Landing gear 11.2. Propeller loss 16.1.

Now we designed the same job as a monoplane and at a high speed of 183 miles an hour on the same kind of paper and using the same slide rule the wing drag came out 37.8.

This first (indicating) was really a sesquiplane, the other a monoplane.

Interplane bracing about the same as before. Fuselage 12.2, and radiator 6, tail 8.8, landing gear about the same as before, and we have just as good a propeller.

Now here is an O2U-2, a biplane with the famous Wasp engine, and this is calculated performance and agrees with the actual performance within a mile, 28.9, bracing and interference 15.4, fuselage 23.6.

MR. SIKORSKY: These data are at full speed?

MR. McCARTHY: Yes. If you took it at low speed, your wing drag would go up somewhat more. Now it is interesting that the total of this and this (indicating) agree fairly well with the total drag of the body. The O2U-2 has a large round body with the gas tank in the system, and these two models (indicating) were made with small narrow bodies with the gas tank in the body, so the projected area of these jobs was much less than the O2U-2.

CHAIRMAN MEAD: That seems to bring again the old problem of air-cooled versus water-cooled drag. I wonder if you can get anywhere with a check on these figures on the basis of what a machine does with retractable landing gear?

MR. MONTEITH: That landing gear on the O2U-2 looks awful small. What size wings did you have?

MR. McCARTHY: 30 by 5. The Army test made on a Falcon, wind tunnel test on a model and removing the chassis completely they reduced the drag 14 percent, so that would make the drag on this basis—

MR. MONTEITH: Let's have those figures on the Falcon, put up on the board.

CHAIRMAN MEAD: Have you anything, Monty, we could put up against this as a monoplane with the wheels up?

MR. MONTEITH: We have the overall test with the model 200. The tests are not very accurate with gear in and gear out. We do not figure any performance on it though because we have to have the body with just the blunt nose on it. We were supposed to have the NACA cowling you see, so we simply extended the lines back, so we have not any idea what the body drag is going to be.

MR. McCARTHY: On the Falcon, I have nothing nearly so complete at this. I have the drag, the loss to the radiator is 4 ½ percent for the total drag. You would have to divide that by power efficiency, I suppose, that would make 82 percent,—that would make the radiator 3.7 percent; the landing gear is 11.5. Those are the only two comparable figures I have.

They tried three or four places for the radiator and found the normal position under the body, the old funnel position was the best of all; that was even the best with this Heindrich type.

CHAIRMAN MEAD: I wonder if it is safe to take those and try and build up an average set of figures, and then from those figures what improvement anybody thought they could get for various reasons? That is, propellers could be improved, and engine cowling? Mr. Weick seems to have a lot of dope in his head from his tests of what we might be able to do.

MR. MONTEITH: The thing I am interested in right now is, are we going to be able to use NACA cowling on the geared—

CHAIRMAN MEAD: Well, Ford does.

MR. WEICK: I wouldn't say he did yet. I spent some time there and made some tests on that particular ship. He did not have what you would call NACA cowling on it in the first place, and he didn't know what it did in the second place.

CHAIRMAN MEAD: I know the boys finally got it so it would cool passibly, it does not cool too well; and I was going to suggest it seems to me the McCook Field type cowling would be ever so much better.

MR. WEICK: Well, I think myself a certain amount of development work has to be done on every installation, especially the hard ones like taking a Hornet for instance, and in the extreme case, taking a geared Hornet on a fairly slow ship and cooling it, and I do not think—that is, I think the general idea is to just say NACA cowling is the particular stuff outlined in the report there, but I do not think that is what should be stuck to at all necessarily, I think the air should be guided and used as efficiently as possible especially in the cases where cooling is hard, and in the case of a geared Hornet for instance I think without any doubt in the World it could be made to both cool quite satisfactorily and have a reasonable reduction in drag, but I do not think you can expect to do it the first time you try; I think you have got to work on that to the point [there] with your reduction in drag; you can use a comparatively small amount of air but use it in the right place and get effective cooling, too.

CHAIRMAN MEAD: I am going on the basis that Tillinghast says the cock with Breen's cowling will operate on the ground with the wheels blocked with full throttle for indefinite periods—half an hour at a time—without overheating the engine. Well, that is better than we have often heard of with NACA cowling? I don't mean that as criticism, but it just happens to work that way that the air seems to be better used.

CHAIRMAN MEAD: He has some control, of course, which the other has not.

MR. MONTEITH: On the other hand; the XP-12 which we built here was just exactly like Breen's cowling only we did not pull the skirt of the cowling in, we faired the job out to give it outside diameter of the engine, and it apparently worked all right out here but when he got it back East he had trouble with it overheating. Breen was out here and we laid the the whole thing out with him standing right over us.

CHAIRMAN MEAD: I think it all goes to show that we don't know as much about this job as we ought to know.

MR. MONTEITH: If we use this on the mail line it has to work darn well, not reasonably well. I am wondering [if] it is going to be worthwhile to put it on that ship?

CHAIRMAN MEAD: I think Short can tell some of his experiences, we saw that he had been using the tin shears in Wichita.

MR. SHORT: Now, ours unfortunately thus far have only covered the J-6 300 (horsepower). I hope before the week is over I will get a report on the Wasp.

We started out the first one following very closely to the rule of the NACA cowling, and the changes which we made however on the J-6 was to put the collector ring in the rim of the cowling as the one furnished by the Wright Company is not suitable for strictly NACA cowling.

With that first one and with the propeller the Hamilton Company furnished us we did the admirable high speed of 133.

CHAIRMAN MEAD: Was that in direct comparison with the ship before?

MR. SHORT: I don't know what the ship would do before, but with some slight alternations in the cowling, and with a more suitable propeller on the same ship we did 143.

CHAIRMAN MEAD: In other words, you lose speed with the coowling?

MR. SHORT: With the version of the NACA cowling we had at first.

MR. HAMILTON: You improved the cowling?

MR. SHORT: We made an improvement on the cowling, namely,—it may be interesting to know how we tried it in here (indicating on the blackboard)

MR. HAMILTON: Is that with the same propeller you got the improvement, or with a different propeller?

MR. SHORT: With a different propeller. With that token it is hard to differentiate even though we do know after the first experiment that the first thought on the NACA cowling and the exhaust collector ring was not ideal by any means.
The collector ring came in here (indicating) [as] an attempt to get that effect; that being the outer foil of the outer wrapper, and this was to develop the supposedly air foil appearance on the wrapper.
On cross section this is a collector ring, and that one,—the collector ring was getting quite hot, we were not getting the air through there apparently, and with the propeller and that combination of cowling on the nose we got the 133.

We then changed it to this effect (indicating), bridging over here, and leaving a small gap in there and using this whole portion right in here as a cockpit heater; a hot air heater for the carburetor, and with that 1219 propeller design we did on our course, as I said, 142.7. We figured that was near enough to 143 to swing over.

On our third attempt,—the inner wrapper coming off so fashion (indicating), we have discontinued it from the rest of the cylinders back to a point ahead of that one (indicating). We had a number of experiments and our Organization went through quite a session on it, and we saw no change of speed by removing that inner portion, and saved considerable time on construction of course, and then for maintenance it was ideal.

Now on the first Wasp job we are using some of Captain Breen's ideas, together with the NACA cowling, using a fairly well cowled up nose, a single outer wrapper that is on the Lockheed version like Breen's, toning it in but using—

CHAIRMAN MEAD: (Interrupting) You don't know how much of this increase was due to propeller?

MR. SHORT: No, we have not the opportunity now to try that out we have changed the cowling. We tried it with this portion out (indicating) and got about 130 miles an hour with that arrangement, just using the collector ring setting out here in the open and no outer wrapper.

I think we are safe in saying that on our first ship the NACA cowling did no good at all. The people who came up from Langley Field to see the job at Washington said we did well to get any increase, and we heartily agreed with them after we played with it ourselves a bit, but it brings out Weick's statement that everyone is a separate problem itself, and invariably I think it will happen that the first cut at the cowling is a disappointment.

MR. McCARTHY: How did the one work out that you had done at Langley Field last summer where you had the thing fairing [from] the fuselage?

MR. SHORT: The NACA people reported not so well. We have not gotten a real final report from them on it. That was faired in around a conventional rectangular fuselage with no attempt at fairing in the sides, exhausting through gills on the sides. On our Wasp combination, it points to a very logical arrangement, and as I say, I hope I can tell you what it will do before the week is over. It is a combination of Breen's and NACA, using a nose spinner,—Mr. Mead advised us very timely that we would not get very much engine control over that because the shutter covers only the very lower portion of the cylinder barrels and probably would not control the temperature, but it is getting the air in the inlet and that is the problem if we cowl it up too high.

We did not want to use Breen's Venetian shutter effect in there, the maintenance must be high on that, so we tried to steer clear of that.

MR. NORTHROP: Has anyone any experience in controlling cowling temperatures, that is in taking care of too cold weather by decreasing the gap in the rear of the cowling?

MR. SHORT: We are trying that now with a slipper ring on the end in fact, it will not only control the temperature but it will control your speed, and we thought of making a remote control to the cockpit because you will take off, as Colyer experiences in the mail run, take off from a warm field in Omaha and hit some bad weather before you get to Cheyenne, and you have to keep the motor warm, and you have to pay the penalty with speed so we had a cam arrangement worked out on the outer ring so as you would rotate this outer skirt it would creep back and forth.

MR. NORTHROP: That was a cylindrical skirt in that case?

MR. SHORT: Yes. Well, we made all of ours a perfect circle.

MR. MONTEITH: George, after following up what we said this morning, I would like to point out two things: This discussion of NACA cowling is fine but as far as any NACA cowling itself is concerned it is not only heavier than the normal cowling but also increases the maintenance troubles because of the two sets of cowling to take off.

CHAIRMAN MEAD: Of course, the first thing is to get any gain out of it, Monty, if we can.

MR. MONTEITH: We actually got a gain on P-12. Of course, we had that up to 183 miles an hour with the engine turning 2170.

MR. SIKORSKY: We had a very reliable information concerning the use of the NACA cowling with a sort of ring around the cylinder heads of our amphibians.

In order to have the data as accurate as possible, we made it as follows: We gauged accurately the speed of the ship with the thing on, then we quickly dismounted it and then on the same day, the same pilot, the same engines, checked the speed again with the first test, and we found the following data: The ship actually gained more than five miles per hour at top speed, at least as much in the cruising speed, the difference in the cruising speed appeared to be fairly distinctly visible and very much in favor of the NACA cowling, and of course it was measured with the same amount of cruise.

Now, there was no difference in climb, in other words, I would say that the ship did not lose any in climb, or lost so little that it was impossible to judge it.

Now, it seems that a little negative difference was in the landing speed which was possibly a little bit worse with the cowling than it was without it.

Now, this data was rather accurate and certainly was encouraging from the standpoint of using any kind of an arrangement of that sort, because the one which we used was just simply a first guess which we made, and we tried to be very much on the safe side in order not to overheat the engines, so it was only a tiny ring something like fifteen inches wide just covering the tops of the engine cylinders, and these were the results: The engine did not show at all any signs of overheating, so the cooling was satisfactory, and I believe quite a lot may be gained in this field and it seems to be very advantageous to go on further in some manner of search.

As I said, the data of the wind tunnel test we [did] with wind motor amphibian was still considerably more encouraging because the models with full cowling showed improvement of 3 ½ points, in other words 14 ½ total of drag instead of 18.

CHAIRMAN MEAD: Monty, have you any idea from flying a tri-motored plane with the various combinations what the ultimate situation might be? As far as I understood it the outboard motors do not give you any gain, it that correct?

MR. MONTEITH: That is correct, the center motor gave a gain of 3.1, but the three sets of NACA cowling added about 120 pounds to the ship, that is almost one passenger.

CHAIRMAN MEAD: I wonder if Weick knows from Langley Field whether they have arrived at any conclusions about the relation of cowling and motors to the wings? Isn't that the reason it don't work on Monty's ship?

MR. WEICK: That is the reason that I ascribe to it, that is you have an interference effect there between nacelles and the wings, and so far in just about every case I know of, which is three actual full scale tests, also some wind tunnel tests, the interference between a cowled nacelle and a wing nearby has been very great, in fact, great enough to usually nullify any possible gain due to the use of the cowling on the engines.

The first experience we had with that of NACA was that we cowled the three engines of a Fokker with whirlwind,—tri-motor whirlwind and we got a gain of about 3½ to 4 miles an hour from the nose cowling, and again got no gain whatever from the wing nacelles, and we did not know of course, exactly what to ascribe it to but immediately thought of the possible interference because our wind tunnel test in the twenty foot tunnel had shown just the opposite; it had shown that you get more gain from cowling and small nacelle than from cowling and engine in front of a large fuselage, and we took two methods then of going on to the problem, one was to put strings—small strings a couple of inches long all over the surface on the actual Fokker and on all the surface of the nacelle and the bottom of the wing, and we found that just behind the maximum diameter of the nacelle where the nacelle started to grow small again there was great turbulence in the first place. The nacelle was only about two or three inches from the surface of the wing and the air apparently would not follow through that and down around the nacelle, and most of the strings in that portion actually pointed forward,—going along over 100 miles an hour. It seemed very queer but it showed the condition of the flow.

Then, in the wind tunnel tests we took, in order to get as near as possible to the full schedule figures, we took a section of the Fokker wing and we built up a nacelle just like the Fokker nacelle as near as we could do it, and with a small imitation of a motor. We got about the same drag coefficient for that nacelle as we got for the complete nacelle, so we knew that part was just about right, and we found with the cowled nacelle we had just as much drag when it was next to the wing as we did with the uncowled nacelle, and then we did two things; first we started fairing that into the wing to see whether if by proper fairing we could reduce that interference, and we found in that particular case we could reduce it very materially, and by putting on a large fairing we increased the actual front projected area and appreciably we could reduce the drag very much so that we got almost the entire gain that we had anticipated we might get neglecting interference entirely.

CHAIRMAN MEAD: That is simply from the cowling into the wing?

MR. SHORT: Simply fairing from the nacelle which was already covered with NACA cowling clear to the wing without changing the location of it, and incidentally at that that type of fairing was put on the ship and a total of increase of twelve miles an hour was finally obtained. Also, the nacelle was moved in the model but it was too much of a job to do that full scale, but on the model, the nacelle was moved to several positions above and below the wing and the only one which was moved was the nacelle with the complete NACA cowling, and it was found that by moving it far enough down you could get away from fairing it, but it had to be an appreciable distance below the wing before the interference would be noticeable.

The best results were obtained by putting the nacelle exactly in front of the wing (indicating on the blackboard) with the top of the nacelle parallel to the top of the wing. [It] will just take a small space here (illustrating). Supposing that is the wing (indicating), and the normal nacelle came in about like this, and when it was in that position we got no gain whatsoever, because the air would not flow through

here, and having large fillets all around, we got an appreciable gain, but by moving this up so that the nacelle came about in this position (indicating) we got less drag than the sum of the wing drag and the nacelle drag when each was isolated, that is, the total drag of this was less than the sum of the wing drag of the wing and the nacelle when they were not together, so that it showed that we could get an increase in performance by changing the location of the nacelle and down the projected area of the entire works if you had a smooth enough cowling. Of course, if you had an exposed engine in that position you would spoil the lift over a good portion of your wing, and the Ford people with their first tri-motors ran into that problem, they had their f-5 engines right in front of the wing, and they had such high landing speed they had to remove them. They took it at the time, it was due to the fact that the propeller was in front of the wing.

MR. McCARTHY: (Interrupting) You think if they cowled those in they would be all right?

MR. SHORT: Yes, because we did not get any increase in lift by doing that, none whatever. We only did it at low angles of [attack]. I do not know what would happen at high angles of [attack].

MR. HAMILTON: High angles would be interesting as far as your landing speed is concerned, and that is where Ford had trouble.

MR. WEICK: There are no tests giving reliable data in this Country. There have been a few small tests made in Germany, but they also did not give just exactly what you want. Right at the present time,—in fact the last thing I did when I left NACA was to outline a series of tests they are now working on and not in the tunnel yet, which consists of tests on a combination of a wing and a propeller in a nacelle, and the nacelle has forty different positions. Approximately forty different positions with respect to the wing using the propeller as a pusher that includes tests with the propeller entirely above the wing and entirely below the wing and three different positions before and aft, so that it is right as near the wing as it could get clearance, and then some positions fore and aft, and in some cases like that where it is built right into the wing and other cases where the nacelle is suspended above or below the wings.

CHAIRMAN MEAD: Monty, are you going to have NACA cowling on the 200?

MR. MONTEITH: That is what we had planned originally, yes.

CHAIRMAN MEAD: Based on what we know now, the trouble with NACA cowling is heating, isn't it?

MR. MONTEITH: I am still worried about that geared engine though.

CHAIRMAN MEAD: You mean in the 200 you are going to use geared engine?

MR. WEICK: With the cowled nacelle, which is the only thing we had there as I remember it, if you got down close to one engine diameter you could get away from the interference, but that is more than you can ordinarily get in an airplane.

MR. MONTEITH: You mean the space to be one inch in diameter?

MR. WEICK: Supposing the diameter of that engine would be say 45 inches below the wing, that is what I mean.

MR. MONTEITH: On that question of Mr. Egtvedt's here a moment ago, Doctor Raam made some tests four years ago on propellers in front of large wings. The wings were much larger than the ones we are using however.

MR. WEICK: Yes. Well, his tests, I think, were very good and they show one thing too, they show in a case like that, or in any case where the propeller is in front of the wing, you have to put that appreciably in front of the wing or you get a very noticeable loss, and from his tests I got an idea on an ordinary arrangement, we will say a nine-foot propeller, which was what we had on the Fokker that we were working with at the time, if we had put the nacelles up into the wing such as that and then put the engines far enough ahead so that the propeller was approximately 18 inches ahead of the leading edge of the wing instead of just I think about five or six inches as they were in the actual installation, there would have been an appreciable gain all the way through, and it looks to me off-hand as if that is the best condition considering high speed. Now, what that particular condition would do to the lift at high angles or landing speed I do not know.

MR. HAMILTON: There is a Russian reproduction of the Ford tri-motor which I saw in London, where they have the engines out from the wing about better than two feet and it is not cowled. It has a Lynx engine and they claim they have tried the position of the Ford and also this position and the landing speed is no greater in this position than with the Ford,—than with the low wing position. However, their top speed is considerably improved. It seems to bear out the information due to the fact they are I should say almost 30 inches ahead of the leading edge.

CHAIRMAN MEAD: Well, it strikes me one thing we would like to know as soon as possible is the results of these tests in NACA laboratories, with the propeller turnings.

MR. WEICK: It is true, but you won't get those for some time.

CHAIRMAN MEAD: I wondered if there is any chance of you Mr. Chatfield; somebody who knows them well, to get in there ahead of the rest of the gang?

MR. WEICK: Well, there is that possibility—

MR. HAMILTON: (Interrupting) They tell me, just with that idea in mind—

MR. WEICK: (Resuming) Actually, I just did go down and got all the dope they have done on high tip speed so far. Those tests were started by me and they just finished them up after I left and I had no difficulty whatever in getting that stuff because I know the fellows down there and because also I was in on the start of the thing and they realized that I had something to do with them down there.

Now it so happened that I was in on the start of this particular thing and will have no difficulty whatever getting that information by going there, but I can't get it by getting them to write it to me.

CHAIRMAN MEAD: Then if we carry out our present scheme of getting such information to Chatfield he can broadcast it through the news letter and everyone will get it about, as quickly as possible.

MR. WEICK: Only, any such information which comes out before it is published by NACA should be kept within the family because otherwise they, of course, would not let us have anything further if they discovered that we were broadcasting it. It would get them in bad.

CHAIRMAN MEAD: You would not get any more?

MR. WEICK: Yes, that it is, sure.

CHAIRMAN MEAD: Then, as far as I can see at the moment we can't help Monty very much except as to additional sealing in the engine. I would like to know in what regard whether the cylinders themselves are warm, or whether you are just guessing by the oil temperature?

MR. MONTEITH: The cylinders themselves are warm on the transport, up as high as 580.

CHAIRMAN MEAD: How about the barrels?

MR. MONTEITH: The barrels are running in proportion; down around 470.

CHAIRMAN MEAD: I should think Breen's scheme of blocking in between the cylinders may improve that somewhat?

MR. WEICK: I think there is no doubt in the World that some scheme of blocking in between the cylinders and putting the air that you take in right where you want it instead of testing it simply filter through in the easiest possible manner, if it is guided to where it should be that would undoubtedly cool better than even letting the engine stick out in the open, but that does add such things as maintenance troubles which of course are already objectionable.

CHAIRMAN MEAD: Do you think, Monty, we could do you any good in Hartford with the 40-B? Of course we are plumbers when it comes to cowling, but we can patch up something,—it has a geared engine in it,—to try and fix up some installation which will cool there,—help anticipating trouble with the 200, or do you think the ships are so different that we would not get you very much of any valuable information?

MR. MONTEITH: Well, the ship is a little bit slower than we suspect the 200 to be so that will be an additional trouble to overcome, but what I am wondering is whether or not the additional speed we hope to get out of the cowling will overbalance the additional weight and additional cost?

CHAIRMAN MEAD: I wonder what Weick thinks about that? That is a pretty big body. Do you have any information which would check that?

MR. MONTEITH: The body is the same diameter as the engine.

CHAIRMAN MEAD: Well, it is bigger than anything we have had as a single engine ship.

MR. MONTEITH: Well, the over all diameter of the J-5 is approximately the same diameter as the engine, it does not look like it, but it is.

MR. WEICK: The over all diameter of the open-cock fuselage in the 20-foot wind tunnel test was also approximately the same as the J-5 engine used in those tests.

MR. McCARTHY: On the F2U, Monty, the body was almost exactly the diameter of the engine with the NACA cowling they put on it, the top came right directly back and the sides were probed in only a very small amount. We copied Breen's stuff almost exactly; had these deflectors behind the cylinders, and we were right at the danger point as far as cylinder temperature is concerned.

MR. MONTEITH: On the 80-A we tried the NACA deflectors on the cylinders but it made no difference.

CHAIRMAN MEAD: Mac, have you had one of your jobs with NACA on it?

MR. McCARTHY: Yes, the XA2-1.

CHAIRMAN MEAD: I don't see quite why Monty feels he is not going to get gain in speed based on these other tests.

MR. MONTEITH: I don't say we won't gain it or won't get the gain. We have managed to put about 11 miles on the mail plane by both fairing out the body behind the engine and putting the NACA cowling on, and it did run hotter, which from Mr. Colyer's point of view, is not so good. We got an appreciable increase on the XP12-A, and on the Transports, we failed to, except the nosed engines.

MR. MONTEITH: I would like to go ahead with the 200 with the NACA cowling on because we have started out that way, and if we get into trouble we can change to something else.

CHAIRMAN MEAD: We seem to be a long ways from this story, but what I was driving at here is fuselage complete with engine runs 20.18 and 23 which shows apparently that air-cooled and water-cooled do not vary a whole lot. Here is an air-cooled at 23, and water-cooled at 20, and if we should take 20 as an average, which might be safe, how much reduction would we get on cowling in its best possible form? Say I can put 20 in this column of averages, what would we have over there (indicating)?

MR. WEICK: According to our tests you get a noticeable decrease.

MR. McCARTHY: It shows six or seven percent decrease in the fuselage drag,—the total drag.

MR. WEICK: The total drag, which would be, say somewhere between ten and fifteen for that drag.

MR. MONTEITH: Here are three figures which may help you, George:

The equivalent flat plat area of model 83 which was the forerunner of the B-12 figured out to be 6.58 square feet. The B-12 figures 8.34, and XP12-A figures 5.89.

CHAIRMAN MEAD: Just a moment, will you give me those figures again, Monty?

MR. MONTEITH: The 83, 6.58; the P12, 8.34; the XP12-A, 5.89. That is calculated from flight tests from the high speed. Take the wing characteristics which were alike in each case and simply figure out your wing horsepower and parasite horsepower and back.

Now the difference between the 83 and the P12, of course, lies in the wheels, landing gear and streamlines behind the cowling, and little Military stuff like sights and things of that kind.

The difference between the P12 and XP12-A is due to the NACA cowling alone. There is some question about that last figure because I had to estimate what the power of that engine was at 2,170 r.p.m.

CHAIRMAN MEAD: It would be better to run this down then more than we have if you said from ten to fifteen, this rather proves it would be better to be down here at ten, wouldn't it?

MR. HAMILTON: You are dealing now with only top speed, how about climb—rate of climb?

MR. MONTEITH: We did not have it on the XP12-A so we don't know.

MR. WEICK: What you expect, of course, is if you get proper cooling and proper power out of your engine, and if you get the decrease in drag which you expect from your cowling, you would expect it to take slightly less power to drive it through the air at climbing speed and therefore get a little better climb; but it certainly would be a small amount.

CHAIRMAN MEAD: I would like to swing this around on one other track, and that is do we know what that fuselage proof would be with a wing radiator?

MR. MONTEITH: That would depend on what kind of ring radiator you used.

CHAIRMAN MEAD: Air-cooled engines do not look so bad on this method of scoring, but the question is how accurate it is.

MR. WEICK: The three fastest ships at Cleveland apparently had radial air-cooled engines.

CHAIRMAN MEAD: Of course, we get to this top but it certainly looks from Chatfield's figures, which I think Monty has probably checked up by now to a certain extent, that the little engine which can be completely cowled is going to give the big engine something to think about.

MR. SIKORSKY: Yes, I believe so, because you see actually the engine interference enters not to so say one time, but enters in the cube at least because the engine interference and size, which in our case, is to put the wing much higher than we want and therefore have longer struts and so on, and therefore this thing may be of very considerable value because the actual gain would be much more.

CHAIRMAN MEAD: On this item of landing gear, is it going to be correct to assume that is entirely washed out if we can have a retractable gear, or is it going to appear somewhere else in this picture?

MR. MONTEITH: It depends entirely on how you retract it. On the 200, we don't retract it at all, as you saw from the lockup last night.

MR. WEICK: Then, there is another thing, a good many ships you would have to build some special,—you would have to make some special changes to get room to put the landing gear when you retract it.

CHAIRMAN MEAD: That is what I say, if we put it down completely at zero it is going to be there in form.

MR. MONTEITH: If you build a ship with retractable landing gear on it you have to start with the gear and build the ship around it.

CHAIRMAN MEAD: From the looks of those figures, it looks as though 11 would be a fairly decent average for landing gear, and I was wondering if we dared to put down something like 4 or 5 on retractable landing gear on a basis that although that item might be wiped out it might be somewhere else equivalent to that?

MR. MONTEITH: You can wipe out about eight out of eleven,—make it three.

CHAIRMAN MEAD: The next question is,—of course, the big item seems to be this old offender, the propeller.

MR. SIKORSKY: This item is a very interesting item, and I would like to find out actually how many give us the proper full amount of revolutions of 80 percent for, if so, it is too bad because there is no room for improvement.

MR. MONTEITH: If everything was as efficient as the propeller, we would be in fine shape. I think we better leave the propeller manufacturers alone for a while.

MR. WEICK: According to our wind propeller test you can run a propeller up to 950 or 1050 for the tip speed without any depreciable loss, and from then on, the loss is very peculiar, it is almost lineal.

On ordinary installations with direct drive air-cooled engines, you can get efficiencies, that is, propulsive efficiencies, not allowing for propulsion efficiencies, including the effect of the propeller on the body; you can get efficiencies as high as 84 or 85 percent with the highest pitch propeller, and 80 to 85 for any propeller which does not run at too high tip speed and which is on a ship going say 16 miles an hour faster.

CHAIRMAN MEAD: (Interrupting) If you give the propeller the best break and put it in the proper relation to the wing (I don't know what that is myself) and run it at the correct speed what is the very best efficiency anybody dares to figure for propeller?

MR. HAMILTON: Eighty percent.

MR. WEICK: It depends on your figuring, of course—

MR. CAIDWELL: Well, we have made tests of ninety percent.

MR. WEICK: The very best we have ever gotten in the full-scale funnel, that is with a geared propeller which ran at quite low tip speed and which was large with respect to the body, a geared propeller on a J-5 engine; a very high pitched geared propeller, we actually got one case of eighty-nine percent.

CHAIRMAN MEAD: I would hate like the deuce to give you 525 horsepower and have you throw away a hundred just like it did not cost anything.

MR. WEICK: The thing is if you don't want that done, if you don't want us to throw away fifty, then you have to design the engine and airplane to get the most efficient propeller.

CHAIRMAN MEAD: That is what I am talking about. This is our "hope to be" column over here (indicating), and that is what I was wondering, whether you felt that you could do materially better than this; whether it is safe to put down any such figures, say, as ninety percent, which actually is correct, I think for certain high speed motor boats working in the water.

MR. WEICK: Of course, there is one thing, you are talking here about the high speed condition of the airplane which is not necessarily the most important condition at all. The propeller efficiency at take-off is very much lower and the propeller efficiency at climb is also much lower than the propeller efficiency at high speed.

CHAIRMAN MEAD: Aren't we interested from Colyer's standpoint of the propeller efficiency at part throttle?

MR. WEICK: It is going to be almost exactly the same unless there is tip speed loss to start with.

MR. SIKORSKY: What is the practical efficiency for, say around a cruising speed of 110 or 115 miles per hour and a top speed of 135 miles an hour?

MR. WEICK: Well, I will take from those wind tunnel tests,—I will make a guess at somewhere around 77 or 78 percent.

MR. SIKORSKY: 77 or 78? As high as that?

MR. WEICK: Yes. That is cruising at 110 which is just about exactly the same efficiency you would get at full speed if your tip speed was not above a thousand, or to be perfectly safe, 950; that is perfectly safe.

CHAIRMAN MEAD: Well, Northrop has another scheme, or course we would like to hear about from the propeller folks, whether you really think this big gain in putting the propeller behind the wing as he has,—he has buried the engine and has nothing in front of it as he gets the wing thicker he hopes—

MR. NORTHROP: (Interrupting) Our scheme was utilized really to get more efficiency out of the wing rather than the propeller and we plan to try it both ways for that reason. We did not know for sure which way would be the best. We felt the speed of the propeller was such it would be less effective by comparatively smooth flow of the wing than the wing itself would be affected by the turbulence caused by the propeller so close in front.

CHAIRMAN MEAD: Of course, these figures we are putting down here are going to affect something until we get 100 percent again. Incidentally, did they tell you that the Travel Air folks told Short they gained 60 miles an hour with the cowlings?

MR. MONTEITH: I don't believe it.

MR. SHORT: Do you expect any propeller design change due to the cowling on the front?

MR. MONTEITH: No, I do not expect that would make any change [to the] propeller.

MR. SHORT: Don't we in effect have a wall there equal to the outside band to the box the propeller is working in?

MR. WEICK: Yes, but actually it is in a portion which is very close to the hub as far as the whole propeller is concerned and which is comparatively unimportant as far as the whole propeller is concerned, and the average propeller of the present day is designed so that the pitch should reduce in towards the hub anyway because of the short air velocity with any body; that is, an ordinary air-cooled engine without cowling has a great effect in reducing the velocity of the air through the central portion of the propeller, and so had the NACA cowling type, and there is not a very great difference there between one and the other as far as the propeller is concerned.

Incidentally, we have made surveys in the plane of the propeller with both types and the air is actually slowed up more in a sort of uniform manner with the NACA cowling than it is without. Without it, it is slowed up as much but in a different manner. It does not go out to that ring, it goes off in between, but I do not think that just that would bare any appreciable effect on propellers any way. The only effect it could have that I could see would be on the pitch distribution, that is the distribution of blade angles and that is not critical in the first place, and it is not greatly different than what it would be without the cowling in the second place.

CHAIRMAN MEAD: Should we have zero for interplane bracing and interference if we had a monoplane without supports?

MR. MONTEITH: No.

MR. WEICK: The trouble is you would have more profile drag.

MR. MONTEITH: You get a certain amount of interference between the body and the wing. It is a minimum in that case, but it is not wiped out.

CHAIRMAN MEAD: Certainly these figures are much better than those which we had to begin with when we started our investigation, and Mr. Chatfield is going to keep right at that because as fast as we get information we would like to add it to this information and make it as useful as possible.

MR. SIKORSKY: I would like to ask the following question; In a single or multi-engine ship around 500 horsepower, the motor and installation in weather will give you approximately 200 to approximately 250 pounds more weight per each point unit.

Now in the multi-engine, or say any heavy Transport ship we are concerned with the drag from the very beginning, for instance, in the flying boat we are concerned with the drag of the propeller at 25 miles per hour, then with the drag of the propeller to the pull all the way further to the take off speed, then to a certain slow speed like say 75 or 80 miles per hour when we carry very much and about the full efficiency of the propeller as high as possible in order to keep a good flight, and at the present time the operating company always requires us to give a high ceiling with one power unit out of commission and therefore, we have another point where the propeller efficiency is a very high volume. Then finally comes the reducing speed, then finally, most important of all comes the top speed.

Now taking as a reasonable average of all these conditions, what do you gentlemen think would be our best guess for the future? I am speaking now of, say one or two years from now, should we ask Mr. Mead for a direct drive engine of 500 horsepower or 575 horsepower, or should we ask for a geared engine, and if geared, what ratio? Let's see what would be the answer to that?

MR. CALDWELL: I do not think there would be any difficulty to us. Probably that sort of step would be a very big increase in efficiency. At the top speed I believe we figured direct drive Hornet would be about 1100 feet a second which would be beyond the point of good condition for tip speed.

Of course, with a metal propeller we are able to go a little further in tip speed on account of thin sections that it is possible with the wood propeller.
The tunnel tests with rather thick sections of used propellers have an extreme loss of efficiency from 18 percent down to about 42 percent, going up to 1500 feet a second,—but with metal propellers, we can't go very much higher.

MR. WEICK: NACA have just completed several tests they started [when I] was there, and you get a very noticeable increase; it starts at tip speeds of around 1000 on the average, and of now as, in one case 950, and most of them are between 1000 and 1050, then it goes down fairly rapidly.

CHAIRMAN MEAD: I was getting very discouraged over this gear situation. We finally got some gears which would stand up and you gentlemen over here did not seem to get any gain except in the seaplane, that is, the high speed situation did not improve very much and so on. Lately though, I seem to be considerably cheered up by this noise situation which will probably require propellers whether we get any gain or not.

I do not know how others feel about that, but riding up on the ship the other day, which as I said before was the best transport we ever have been in, its principle deficiency is noise, and certainly we can quiet the exhaust down some but it seems as though a good deal of the noise over turning fifteen or sixteen hundred was from the propeller. I don't know how you feel about it but it seemed to me that one situation was the slowing down of the propeller.

MR. EGTVEDT: I would like to ask one question too: What would be the effect of this problem of Mr. Sikorsky's here with the variable pitch propeller? Now right on that particular problem of gearing as against direct drive with a propeller of that kind—

MR. CALDWELL: (Interrupting) Well, I think the variable pitch will give a decided improvement all right on the static thrust and the efficiency, or rather the power at the take-off. I think it would be intermediate though between the gear and the direct drive—the fixed pitch would probably be lighter; a great deal lighter.

MR. EGTVEDT: It might be possible, for instance with a land plane where the conditions are not quite as severe as has been pointed out for the sea plane, or flying boat, that that would more than off-set our increased weights and other difficulties. On the other hand with a flying boat it is very doubtful whether it would take care of the difference.

CHAIRMAN MEAD: I think we ought to consider this propeller in the most advantageous position as well as the right speed, and we have only tried one experiment in that regard which probably was not conducted very accurately, but there did not seem to be much gain in putting the propeller six inches further ahead, and possibly we should have gone further yet but I think it is worth several pounds in the engine if the propeller efficiency can be stepped up somewhere. We found according to those tests it is one mile an hour, and it cost us nearly twenty pounds in the engine to get it and that did not seem to be very worth while.

MR. HAMILTON: Mr. Mead, it seems to me there is so much involved in this matter of proportion and its effect on future design that I would like to get these propeller experts organized tonight to give you some idea of what we have in mind for next year, tomorrow, and then you ask us a lot of questions and we will try to give you some where near an answer or they will, rather, not me, but the thing always comes back to the propeller I find after the plane is all finished and the design is completed you take it out and expect the propeller men to put on the miles per hour that were laid down on the paper, and I feel that we might be able to contribute, but when they talk about ninety percent efficiency and all that sort of thing I just wonder if these fellows realize that later you are going to be asking for some of those things, therefore I would like to give you an outline tomorrow of what we have in mind.

I was not here yesterday, but we would like to ask some questions about what you want and why you want it, and then we will try to give you some idea as to what can be done with the propeller proposition as far as gearing, lighter weight and all that sort of thing.

I think unless you go at this propeller proposition as one problem, itself, and just have it tied up to this matter of the over all efficiency of the plane, why, we are not going to get much out of it from a propeller point of view at least, I have not gathered much from listening to all this so far; we are just sparring back and forth and I do not see that we are getting much information out of it. We are asking questions now that cannot be answered off-hand intelligently.

CHAIRMAN MEAD: We are just figuring an average of figures, and you all seem to agree that between 80 and 85 percent is where you will land, and that is a good place to begin because we have talked sometimes and thought "if we only had 18 percent". Well, you seem to all agree you will do better than that, and what we intended to do tomorrow was talk transport ships and Military ships and consequent improvements in performance, and from that, try to lay down the program for United as far as research went on all these various items, and I was simply trying to get down here some average which looked reasonable and everyone agreed to, and then of course looking off into the future if anybody was willing to commit themselves as to what possible improvement they might expect.

You see, we have been wondering if you really felt you could get a gain of ten percent, that is, run up from 80 to 90 percent efficiency, why, that was going to help a lot if you honestly feel, as you apparently do, that we can only take up 5 percent, that is something to figure on, and to have to look somewhere else to get a big gain.

MR. HAMILTON: This matter of propulsion is so closely tied up to what is immediately behind the propeller, and according to Mr. Weick's experience in NACA, and also on account of the fact that people cut out a dishpan and put it in front of their engine and think they have an NACA cowling, we have some ideas as far as United is concerned and as far as the industry is concerned.

Now, we have some ideas here, and we wonder whether we can utilize Mr. Weick's experience to the best advantage by undertaking some of this cowling work?

CHAIRMAN MEAD: I think you can't interest the rest of us any more than by taking that job over.

MR. HAMILTON: We are not the least bit unmindful of what is involved in such a nasty mess as to try to assist in cowling.

CHAIRMAN MEAD: What I gave up on, when you hear from around the table none of these ground rules apparently laid down work in each case and you have to change it around, but it is more of a problem than any of us thought; you just can't take some rule of thumb and do it, which seems to indicate to me that we just don't know as much as we ought to about the job.

MR. HAMILTON: I am fairly convinced of that.

MR. WEICK: Incidentally, in connection with that, I got a chance to talk with Mr. North quite a while one evening—he is the party in England who has been doing most of the work along with [Townend] down there, and he said very decidedly—and I was glad to hear him say it—that you could not expect to get any gain out of a Townend ring unless you did a lot of preliminary work. That is, the way he worked it was to put a test—put a model in a wind tunnel and then start putting Townend rings on it and small scale stuff. It was not what might be the best actually in full scale, but it was the best he could do, and he figured that if he could get it in about five or six tries of different Townend rings and different modifications then he could put the results of his work into a full scale job and hope to get some increase, but just taking a clear shot out of the sky you could not tell what you could expect to get, and the same is true of the NACA cowling.

Document 3-10(c), "The Curtiss Anti-Drag Ring," **Curtiss-Wright Review** *1 (December 1930)*

CURTISS-WRIGHT REVIEW
December, 1930

The Curtiss Anti Drag Ring

Greater efficiency—more speed with the same horsepower—is a problem on which the airplane industry is concentrating today.

The most effective way to achieve greater efficiency is by the reduction of parasite, or unnecessary air resistance.

Other things being equal, the air cooled radial type of engine, with its short crankshaft and small crankcase, is lighter than any other type. But when the parts of an airplane which offer an undue resistance to passage through the air are under consideration, the popular radial engine must be considered one of the worst offenders. Hence, a great deal of attention is being given to the problem of cowling this type of engine, so as to make it more efficient from the standpoint of drag. As a practical solution of this problem, the Curtiss Anti Drag Ring has been developed in the Garden City Laboratories of the Curtiss Aeroplane & Motor Company. This device will be manufactured in the Garden City Factory and will soon be offered to the industry for use on all makes of radial engines.

The appearance of the Curtiss Anti Drag Ring is best made clear by the accompanying photographs. It is a short barrel-shaped form of sheet duralumin surrounding the engine cylinders. It has the same contour its the upper surface of the Curtiss C-72 airfoil (wing section).

Experiments first made in England by Mr. Townend had shown the possibility of reducing drag with a ring of this type. The problem then, was to investigate the theory of this type of cowling—how and why it works—and thereafter to find out how to design and install the ring for the best results, both aerodynamically and from the standpoint of maintenance. This development from the Townend ring has resulted in the Curtiss Anti Drag Ring.

To obtain the necessary information, a one-quarter scale wood model of the fuselage of a new Curtiss airplane was built. Models of Wright radial engine cylinders were attached to the fuselage in their proper positions. A number of rings of different shapes were constructed, some spun out of brass and some built up from wood segments. The fuselage model was then set up in the Curtiss seven-foot wind tunnel, and drag tests were made. The wind resistance was measured without any ring and with each ring in place, trying different fore-and-aft locations for each one. In this way the best shape and location for the ring was determined. A large number of other tests were made to find the effects of various changes and to clear up questions of theory.

Air is violently deflected outward away from the fuselage when it strikes the projecting cylinders of a radial engine. This breaks down the smooth flow of air over the entire fuselage and leaves a turbulent wake much larger in diameter than the engine The anti drag ring prevents the turbulence caused by the cylinders from spreading out, and forces the air behind the cylinders in toward the fuselage, ironing out irregularities in the flow. The Curtiss wind tunnel tests showed that a drag reduction of forty-six percent was realized from these effects. It is an interesting fact that there is actually a forward-acting, or anti-drag, force acting on the ring, a force which may amount to as much as two hundred pounds on a fast airplane. This force is due to the angle at which the ring is set with relation to the direction of airflow over the nose of the fuselage.

As finally developed, the Curtiss Anti Drag Ring is simple and light in construction, does not require any special supports, and is easily removed for maintenance work on the engine. It has been used with great success on several Curtiss Airplanes. Speed increases of nineteen miles per hour have been obtained, and it has been found that the cooling of the engine is improved.

Document 3-11(a-b)

(a) United Aircraft and Transport Corporation Technical Advisory Committee Minutes held at the Pratt & Whitney plant, Hartford, CT, 19-23 May 1930, pp. 230-238, in Archive and Historical Resource Center, United Technologies Corporation, East Hartford, CT. [Courtesy F. Robert van der Linden.]

(b) "The Reminiscences of Harold Hicks," August 1951, transcript, pp. 61-69. Henry Ford Museum and Greenfield Village Research Center, Dearborn, Mich.

This pair of documents reflects the U.S. aircraft industry's growing interest for data about the aerodynamic value of engine nacelle placement around 1930. The first comprises an excerpt from the May 1930 meeting of the United Aircraft and Transport Corporation Technical Advisory Committee, held at the Pratt & Whitney plant in Hartford, Connecticut. At the time, no aeronautical organization was larger than the United Aircraft and Transport Organization. Just newly formed in 1930, the organization consisted of four major airplane manufacturers: the Boeing Aircraft Company of Seattle, Washington, which recently had acquired the Stearman Company of Wichita, Kansas, and the Hamilton Metalplane Company in Milwaukee; the Vought Aircraft Company of Long Island, New York; the Sikorsky Aircraft Company of Bridgeport, Connecticut; and the Northrop Aircraft Company of Burbank, California. It also consisted of a leading engine manufacturer, Pratt & Whitney, which had just come out with two excellent new radial air-cooled engines (the Wasp and the Hornet); the largest American propeller manufacturer, Hamilton Standard, which resulted from the merger of Hamilton's propeller company with the Standard Steel Propeller Company of Pittsburgh, Pennsylvania; two airlines, United, which ran from New York through Chicago to San Francisco, following the old air mail routes, and Pacific, which ran from San Francisco up to Seattle. Two other combinations of companies had been formed at about the same time, but United was the largest and most profitable.

The excerpt reproduced below from the minutes of the May 1930 meeting of United's technical advisory committee is extremely interesting from many vantage points. It captures an exchange between United executive George J. Mead; Boeing's Charles N. Monteith, formerly an officer in the engineering division of the air corps at McCook Field in Dayton (he had written a book in 1924 called *Simple Aerody-*

namics and the Airplane); and Fred Weick, who had just moved from the NACA to Hamilton. The conversation definitely highlighted that the industry did not fully recognize the importance of NACA's research to transport design—something that would quickly change over the course of the early 1930s. In fact, operating problems caused by placing engine nacelles under the wing of a multiengine transport, the Ford trimotor, were specifically mentioned during this part of the meeting.

The second document, from the reminiscences of engineer Harold Hicks, the chief designer of the Ford Trimotor airplane, also testifies to the industry's growing concern around 1930 not just for help with proper engine nacelle positioning, but with proper aerodynamic design data overall. The design of the Trimotor has always been portrayed as an empirical and unsophisticated process and a stunning example of the old methods of designing aircraft during the 1920s. Hicks's recollections illustrate that Ford aeronautical engineers were aware of the disadvantages of placing the nacelle below the wing, and they were considering the use of NACA developments to alleviate the aerodynamic disadvantages. The fact that they were willing to incorporate new innovations into a less-than-state-of-the-art design highlights the indeterminacy in design during the period. The interview with Hicks was conducted in August 1951 by Owen Bombard of the Oral History Section of the Ford Motor Company Archives in Dearborn, Michigan, and brings the reader a valuable insight into the design logic behind one of the most well-known aircraft of the period.

Document 3-11(a), United Aircraft and Transport Corporation Technical Advisory Committee Minutes held at the Pratt & Whitney plant, Hartford, CT, 19-23 May 1930.

CHAIRMAN MEAD: Monty, have you any idea from flying a tri-motored plane with the various combinations what the ultimate situation might be? As far as I understood it [N.A.C.A cowling] the outboard motors does not give any gain, is that correct?

MR. MONTEITH: That is correct, the center motor gave a gain of 5.1 [mph], but the three sets of NACA cowling added about 120 pounds to the ship, that is almost one passenger.

CHAIRMAN MEAD: I wonder if Weick knows from Langley Field whether they have arrived at any conclusions about the relation of cowling and motors to the wings? Isn't that the reason it don't work on Monty's ship?

MR. WEICK: That is the reason that I ascribe to it, that is you have an interference effect there between the nacelles and the wings, and so far in just about every case I know of, which is three actual full scale tests, also some wind tunnel tests, the interference between a cowled nacelle and a wing nearby has been very great, in fact, great enough to usually nullify any possible gain due to the use of the cowling on the engines.

The first experience we had at the NACA was that we cowled the three engines of a Fokker tri-motor with Whirlwind engines and we got a gain of about 3 ½ to 4 miles an hour from the nose cowling, and again got no gain whatever from the wing nacelles, and we did not know, of course, exactly what to ascribe it to but immediately in the twenty foot tunnel had shown just the opposite; it had shown that you get more gain from cowling an engine in a small nacelle than from cowling an engine in front of a large fuselage, and we took two methods then of going on to the problem, one was to put strings—small strings a couple of inches long all over the surface on the actual Fokker and on all the surface of the nacelle and the bottom of the wing, and we found that just behind the maximum diameter of the nacelle where the nacelle started to grow small again there was great turbulence in the first place. The nacelle was only about two or three inches from the surface of the wing and the air apparently would not follow through that and down around the nacelle, and most of the strings in that portion actually pointed forward; in other words here we are going along with strings pointing forward, going along over 100 miles an hour. It seemed very queer but it showed the condition of the flow.

Then, in the wind tunnel tests we took, in order to get as near as possible to the full schedule figures, we took a section of the Fokker wing and we built up a nacelle just like the Fokker nacelle as near as we could do it, and with a small imitation of a motor. We got about the same drag coefficient for that nacelle as we got for the complete nacelle, so we knew that part was just about right, and we found with the cowled nacelle we had just as much drag when it was next to the wing as we did with the uncowled nacelle, and then we did two things; first we started fairing that into the wing to see whether if by proper fairing we could reduce that interference, and we found in that particular case we could reduce it very materially, and by putting on a large fairing we increased the actual front projected area appreciably and we could reduce the drag very much so that we got almost the entire gain that we had anticipated we might get neglecting interference entirely.

CHAIRMAN MEAD: That is simply from the cowling into the wing?

MR. WEICK: Simply fairing from the nacelle which was already covered with NACA cowling clear to the wing without changing the location of it, and, incidentally, that type of fairing was put on the ship and a total of increase of twelve miles an hour was finally obtained. Also, the nacelle was moved in the model but it was too much of a job to do that full scale, but on the model, the nacelle was moved to several positions above and below the wing (the only one which was moved was the nacelle with the complete NACA cowling) and it was found that by moving it far enough down you could get away from fairing it, but it had to be an appreciable distance below the wing before the interference was not noticeable.

The best results were obtained by putting the nacelle exactly in front of the wing (indicating on the blackboard) with the top of the nacelle parallel to the top of the wing.

I will just take a small space here (illustrating). Supposing that is the wing (indicating), and the normal nacelle came in about like this, and when it was in that position we got no gain whatever, because the air would not flow through here, and having large fillets all around, we got an appreciable gain, but by moving this up so that the nacelle came about in this position (indicating) we got less drag than the sum of the wing drag and the nacelle drag when each was isolated, that is, the total drag of this was less than the sum of the drag of the wing and the nacelle when they were not together, so that it showed that we could get an increase in performance by changing the location of the nacelle and outing down the projected area of the entire works if you had a smooth enough cowling. Of course, if you had an exposed engine in that position you would spoil the lift over a good portion of your wing, and the Ford people with their first tri-motors ran into that problem, they had their J-5 engines right in front of the wing, and they had such high landing speed they had to remove them. They took it at the time, it was due to the fact that the propeller was in front of the wing.

MR. McCARTHY: (Interrupting) You think if they cowled those in they would be all right?

MR.WEICK: Yes, because we did not get any decrease in lift by doing that, none whatever. We only did it at low angles of attack. I do not know what would happen at high angles of attack.

MR. HAMILTON: High angles would be interesting as far as your landing speed is concerned, and that is where Ford had trouble.

MR. WEICK: There are no tests giving reliable data in this Country. There have been a few small tests made in Germany, but they also did not give just exactly what you want.

Right at the present time, in fact the last thing I did when I left NACA was to outline a series of tests they are now working on and not in the tunnel yet, which consists of tests on a combination of a wing and a propeller in a nacelle, and the nacelle has forty different positions—approximately forty different positions with respect to the wing using the propeller as a tractor, and then there will also be a series of tests made using the propeller as a pusher that includes tests with the propeller entirely above the wing and entirely below the wing and three different positions before and aft., so that it is right as near the wing as it could get clearance, and then some positions for and aft., and in some cases like that where it is built right into the wing and other cases where the nacelle is suspended above or below the wings.

CHAIRMAN MEAD: Monty, are you going to have NACA cowling on the 200?

MR. MONTEITH: That is what we had planned originally, yes.

CHAIRMAN MEAD: Based on what we know now, the trouble with NACA cowling is heating, isn't it?

MR. MONTEITH: I am still worried about that geared engine though.

CHAIRMAN MEAD: You mean in the 200 you are going to use geared engine?

MR. WEICK: With the cowled nacelle, which is the only thing we had there as

I remember it, if you got the nacelle away from the wing with a gap down close to one engine diameter you could get away from the interference, but that is more gap than you can ordinarily get in an airplane.

MR. MONTEITH: You mean the space to be once the diameter?

MR. WEICK: Supposing the diameter of that engine would be say 45 inches as it was on the J-5; if you put the top of the engine 45 inches below the wing, that is what I mean.

MR. MONTEITH: On that question of Mr. Egtvedt's here a moment ago, Doctor Zahm made some tests four years ago on propellers in front of large wings. The wings were much larger than the ones we are using however.

MR. WEICK: Yes. Well, his tests, I think, were very good and they show one thing took, they show in a case like that, or in any case where the propeller is in front of the wing, you have to put that appreciably in front of the wing or you get a very noticeable loss, and from his tests I got an idea that on an ordinary arrangement, we will say a nine-foot propeller, which was what we had on the Fokker that we were working with at the time, if we had put the nacelles up into the wing such as that and then put the engines far enough ahead so that the propeller was approximately 18 inches ahead of the leading edge of the wing instead of just, I think, about five or six inches as they were in the actual installation, there would have been an appreciable gain all the way through, and it looks to me off-hand as if that is the best condition considering high speed. Now, what that particular condition would do to that the lift at high angles or landing speed I do not know.

MR. HAMILTON: There is a Russian reproduction of the Ford tri-motor which I saw in London, where they have the engines out from the wing about better than two feet and it is not cowled. It has a Lynx engine and they claim they have tried the position of the Ford and also this position and the landing speed is no greater in their position than with the Ford, than with the low wing position. However, their top speed is considerably improved. It seems to bear out that information due to the fact they are I should say almost 30 inches ahead of the leading edge.

CHAIRMAN MEAD: Well, it strikes me that one thing we would like to know as soon as possible is the results of these tests in NACA laboratories, with the propeller turning.

MR. WEICK: It is true, but you won't get those for some time.

CHAIRMAN MEAD: I wondered if there is any chance of you or Chatfield; somebody who knows them well, to get in there ahead of the rest of the gang?

MR. WEICK: Well, there is that possibility—

MR. HAMILTON: (Interrupting) They tell me, just with that idea in mind—

MR. WEICK: (Resuming) Actually, I just did go down and got all the dope they have done on high tip speed so far. Those tests were started by me and they just finished them up after I left and I had no difficulty whatever in getting that stuff because I know the fellows down there and because also I was in on the start of the thing and they realized that I had something to do with them down there.

Now it so happens that I was in on the start of this particular thing and will have no difficulty whatever getting that information by going there, but I can't get it by getting them to write it to me.

CHAIRMAN MEAD: Then if we carry out our present scheme of getting such information to Chatfield he can broadcast it through the news letter and everyone will get it about as quickly as possible.

MR. WEICK: Only, any such information which comes out before it is published by NACA should be kept within the family because otherwise they, of course, would not let us have anything further if they discovered that we were broadcasting it. It would get them in bad.

CHAIRMAN MEAD: You would not get any more?

MR. WEICK: Yes, that is it, sure.

CHAIRMAN MEAD: Then, as far as I can see at the moment we can't help Monty very much except as to additional cooling in the engine.

Document 3-11(b), "The Reminiscences of Harold Hicks," August 1951, transcript, pp. 61-69. Henry Ford Museum and Greenfield Village Research Center, Dearborn, Mich.

Our position did not at first change when the Ford Motor Company took over Stout. At first, we ran along and did this overhaul work together with cleaning up other ends of the design on other small projects that I was doing at the time, including helping Horton on some stress analysis work.

Along about the first of September, the "Shenandoah" dirigible cracked up in Ohio. Mr. Ford wanted somebody to go there and give him a first-hand report. George Pruden, who was the Stout chief engineer and the man who is really responsible for the designing of the single engine plane, went down on the trip.

Being unfamiliar with the Ford Company way of doing things, he permitted a picture of himself to be taken which was put on the front page of the Detroit Free Press with a caption showing Pruden writing his personal report of the crash to Henry Ford. That apparently didn't suit Henry Ford very well so Pruden was fired.

Then Mr. Ford came to me and asked me to take over the engineering of the airplane. The only instructions that he made to me were very peculiar at that time. He said that I was to get in there, and run it, and to keep Stout out of the design room. He said that for the first time in his life, he had bought a lemon and he didn't want the world to know it. All the work that I did, together with any of the men, the credit would have to be given to Stout.

He was very proud to think that he was a good businessman. Apparently Mayo had led him into something, not fully realizing that Stout was a promoter and not a practical engineer.

At the time when Pruden was released from the Company, they had under construction a three engine plane which was known as the 3 AT job. That had air cooled engines, the Wright J 4. Three of those were used, one in the nose of the fuselage, the other two in the leading edge of the wing. This was the first development. It had the engine, not in the nacelles beneath the wing, but right in the leading edge.

Pruden had designed the first trimotor, using such people as Tom Towle, John Lee, Otto Koppen and McDonnell who were in the original Engineering Department with Stout. Koppen was in the Stout Engineering Department. He was not an independent outsider brought in to design the flivver plane alone. He was a part of the original group, and I believe there were possibly ten to twelve fellows in the group. When they came over, it was reduced to possibly seven because some of the men were not suited to aeronautical design.

After I took over the Engineering Department, it had to be reorganized. I had as my assistant, Tom Towle. He was an airplane designer and a very good one too. He is now the president of the Church Company that makes baking soda. He married Church's daughter, I believe.

When I took over engineering, Mayo, of course, was the general manager. Stan Knauss had charge of the shop. Hoppin was the treasurer. Shortly after that, they hired a man by the name of Rudolph Schroeder. He took over the flying end and the airport operations.

Harry Russell, I believe, worked for Stout originally as a mechanic. Manning was hired possibly a year or two after that. He was a young fellow that was a pilot and was hired into the Company after it had been fully taken over by the Ford Motor Company.

When I took over, they were producing the 2 AT planes. The 3 AT trimotored plane was being built and assembled. This first trimotor was powered by J 4 engines. It was flown by Schroeder in possibly the late fall, maybe November of 1925. I can remember that there are photographs of that ship. Mr. Ford and Edsel were there.

I can remember when Schroeder brought it in. It landed awfully fast. It landed easily at eighty miles an hour. The reason for that was aerodynamic. We didn't know how to cure it then because NACA had not developed their ring cowls to cover the cylinders and prevent the disturbance from blanketing out the wing area. There was an inadequate wing area on the plane because the wing engines destroyed the air flow behind them; that's why it landed so fast. It was considered unsatisfactory.

The wing section was one of the NACA sections, but it did not have a very effective lift as it was convex on the bottom surface. That was the same type of wing that was used on the single engines.

In his autobiography, Stout says that his original trimotor designed with the engine in the leading edge was an advanced design over that with the pods underneath the wings. He says that the only reason he changed the design was because of the fact that the officials of the Ford Motor Company thought that the plane landed too fast. In lowering the engine and putting it in a pod beneath the wing, they were actually causing a decrease in the performance of the plane.

That is probably so, but at the same time the plane with the engines in the leading edge of the wing without any NACA cowl rings would have been entirely unsatisfactory. It landed too fast, and the top speed of the plane wasn't much either. As I remember, the top speed was only about 110 miles an hour.

So with a top speed of 110 and landing speed of 80, you didn't have any range at all. The take off speed was around 80 miles an hour too.

Schroeder, according to Aviation News, is in Chicago. This was about six months ago. He is writing his autobiography.

However, the very fact that the plant burned down, in my opinion, proved a benefit to the Ford Motor Company and to aviation in general. Plans were immediately made to start a much better factory, one which was built quite quickly. It is now the body building and garage. At least the west half of it was erected very quickly after the fire.

After the fire Mr. Ford wanted to go on. The fire had occurred so early in the Ford experience of producing airplanes that that is probably the one reason why he built it up rather than to cross it off the books entirely.

Document 3-12(a-h)

(a) Harlan D. Fowler to Rueben H. Fleet, 8 January 1929, Folder "Fowler Flap," copy in San Diego Aerospace Museum, CA.

(b) Consolidated Aircraft Corporation to Harlan D. Fowler, 21 January 1929, Folder "Fowler Flap," copy in San Diego Aerospace Museum, CA.

(c) Harlan D. Fowler, "Variable Lift," ***Western Flying 10*** **(November 1931): 31-33.**

(d) Harlan D. Fowler to William B. Mayo, 15 August 1932, copy in Accession 18, Box 96, Folder "Fowler Variable Wing," Henry Ford Museum and Greenfield Village Research Center, Dearborn, Mich.

(e) H.A. Hicks to Harlan D. Fowler, 10 September 1932, copy in Accession 18, Box 96, Folder "Fowler Variable Wing," Henry Ford Museum and Greenfield Village Research Center, Dearborn, Mich.

(f) T.P. Wright, "The Application of Slots and Flaps to Airplane Wings in America," ***U.S. Air Services*** **(September 1933): 29-31.**

(g) Robert C. Platt, "Aerodynamic Characteristics of a Wing with Fowler Flaps including Flap Loads, Downwash, and Calculated Effect on Take-off," NACA ***Technical Report*** **534 (Washington, 1935).**

(h) Harlan D. Fowler, "Aerodynamic Characteristics of the Fowler Wing," *Aero Digest* 29 (September 1936): 46-50.

Harlan D. Fowler's design of a wing flap in 1924 is one of the least known yet classic stories of invention in the history of American aeronautics. An employee of the army's Engineering Division facility at McCook Field in Ohio, Fowler became interested in the concept of a variable-area wing late in World War I. For the next six years, he developed several unsuccessful prototypes until he came up with the idea of combining a slotted flap similar to the Junkers auxiliary wing with an increase in wing area and camber. In 1929, this notion led him to design what came to be known as the "Fowler flap," a small airfoil housed in the trailing edge of the wing. When needed, the airfoil could be extended backward until there was a slot between it and the main wing, and then hinged downward. Possessing a much heavier mechanism than a slotted or split flap, the Fowler flap made up for the weight with the increase in wing area and greater lift, which improved both takeoff and landing performance. Thus, the basic innovation was to increase lift by combining two effects: increasing the camber or curvature of a wing by deflecting the flap downward, and increasing the wing's surface area by mechanically extending it out downstream of the wing. Used together in such a way, the combination would greatly shorten takeoff runs, lower landing speeds, and increase rates of climb.

The army did not support Fowler's work, which was okay with Fowler as he had always intended his flap as a private venture. He began turning his idea into a practical mechanism in 1926. Working in his spare time and with his own funds, he constructed wings of his own design for various biplanes through the late 1920s. Flight tests revealed that landing speed decreased by two miles per hour and overall top speed increased by ten miles per hour. During this period, Fowler claimed that his design might deliver a whopping 90 percent boost in lift, which few people were prepared to believe.

No one paid attention to the "Fowler Variable Wing" until the NACA started taking a look at it in 1932. Wind tunnel tests at Langley during that year proved that his design did in fact contribute to the overall performance of an airplane. The NACA engineer most involved in testing the Fowler flap was Fred E. Weick, spearhead of the NACA's cowling and engine nacelle placement programs. Weick felt that Fowler's design merited at least some limited wind tunnel testing. In the first report on the Fowler flap published by the NACA in May 1932, Weick reported on "wind tunnel tests of the Fowler variable-area wing" (Technical Note 419). A year later, another report appeared, "Wind Tunnel Tests on model wing with Fowler flap and specially developed leading-edge slot" (Technical Note 459, May 1933), co-authored by Weick and Robert C. Platt.

The NACA's interest and its promising technical evaluations gave Fowler's invention its first true stamp of credibility—one that he really needed as he was no longer working as an engineer but out of the aeronautics world as a salesman in California. In 1933, he traveled to Baltimore in hope of convincing Glenn L. Martin of the value of his flap. The fact that the NACA had been experimenting with the device was a decisive factor in how Martin responded. He gave Martin a job and put him to work devising flaps for several new Martin aircraft. One of these became the Martin 146 bomber of 1935. Although it was never produced, the airplane did employ Fowler flaps.

By the mid-1930s, the U.S. aeronautics community as a whole had grown extremely interested in the promise of high-lift devices. Airplanes were flying so fast and growing so large that great improvements in lifting capacity and much higher wing loadings were becoming an absolute necessity. Lockheed incorporated Fowler flaps into its new twin-engine airliner of 1937, the Lockheed 14. It was the first production-line airplane to use Fowler flaps. (A few German aircraft with Fowler-type flaps had appeared earlier.) After that, a growing number of planes started to use slotted flaps of various kinds. Fowler became one of the most recognized names in all aeronautical engineering. Even today, his name is still associated with a generic brand of slotted flaps used on many aircraft.

In the string of documents below, one returns to the early 1930s to see how the American aeronautics community first became seriously interested in the Fowler slotted flap.

Document 3-12(a), Harlan D. Fowler to Rueben H. Fleet, 8 January 1929, Folder "Fowler Flap," copy in San Diego Aerospace Museum, CA.

FOWLER AIRPLANE WINGS, INC.
Manufacturer of Variable Area Wings
NEW BRUNSWICK, N. J.
July 8, 1929.
Mr. R. H. Fleet, President,
Fleet Aircraft, Inc.
2050 Elmwood Ave.,
Buffalo, N. Y.

Dear Sir:

The development of the Fowler Variable Area Wing is approaching the stage of practical utility after being successfully used on several types of planes.

A standard type of construction and wing area is now being prepared with the purpose of applying it to the larger majority of types of airplanes now in use giving them very substantial improvements in all around performance, that is, lower landing speed, higher top speed and good climb.

We are approaching you as a manufacturer of high grade airplanes with a proposal involving the use of our wing in place of your standard product for a cost within the range of your own factory cost wings.

If this proposal meets with your approval and you will assure us of your intention to contract for a limited number of wings, then we will install our wing on your plane and obtain complete comparative performance to be used by your company for sales purposes, and cooperate with you in giving demonstrations.

This arrangement will give your products a distinct step in advance of the conventional plane and consequently greatly influence your sales prospects.

We would appreciate learning how this procedure appeals to you and we will be pleased to submit such information as you may desire.

Respectfully yours,

President, Fowler Airplane Wings, Inc.

FOWLER VARIABLE AREA WING

The demand of aircraft operators for improved performance and load carrying capacity to meet the requirements of the public is bringing about a condition which will compel consideration of advanced wing design. Primarily, an aircraft must have sufficient wing lift to provide safe landings and take-offs. This restriction has a direct influence on high speed and load capacity and is an economic loss.

The Fowler Variable Area Wing provides for this condition. The powerful combination of the advantages of change in camber, area, angle of incidence and aspect ratio, simultaneously, has resulted in a wing design of tremendous value to aircraft operators.

As the result of further research and development of the Fowler Wing substantial improvements in performance are obtained. The following comparative data relative to the original PA-3, OXX-6 100 H.P. is interesting:

	Original	Fowler	Wing
Performance	PA-3	No. 1	No. 2
Low speed, normal, M.P. H.	46.2	58.2	57.0
Low speed extended M. P. H.	--	44.5	41.0
High speed, M. P. H.	91.0	101.0	104.0
Cruising	77.0	84.0	87.0
Rare of climb, normal, ft. / min.	515	430	570
Total weight of plane	1936	1990	2000
Wing area, normal, sq. ft.	338	136	145
Wing area, extended, sq. ft.	--	166	186
Wt./ sq. ft. normal	5.72	14.65	13.80
Wt./ sq. ft. extended	-	12.00	10.75
Speed range	1.97	2.16	2.42

The above performances are based on the best propeller design suitable for that type of plane. Performances for the PA-3 and Fouler Wing No. 1 are based on actual flight tests over a one mile course and are the average of several runs. Knowing the high speed of Fowler Wing No. l the parasite resistance was obtained after subtracting the wing resistance. With this parasite resistance still used and with the new Fowler Wing No. 2 substituted the above performance was obtained.

Reference to the above performance reveals a number of possible applications of the Fowler Wing No. 2 as applied to this particular low powered type of plane:

(a) As between the biplane the low speed is reduced almost 5 miles per hour. Between the usual monoplane (normal area) it is 16 miles per hour less in low speed when the extension area is used. Note the difference in wing area.

(b) The high speed of the normal. wing is about 13 miles per hour faster than for the biplane.

(c) The load carrying capacity of the extended wing per square foot area is over 100% greater than for the biplane, and with less landing speed.

(d) Speed range for the Fowler Wing is increased almost 23 per cent over the biplane.

(e) Rate of climb for the Fowler Wing is of a very high order considering its other remarkable features.

(f) The Fowler Wing is only 25 pounds heavier than the biplane with a strength factor 54 per cent greater than the original PA-3.

The evidence presented by these tests indicates possibilities such as:

(a) Pay load increased 100% for equal performances and power.

(b) Same pay load and performance for about once-third less power.

(c) A reduction in wing area 50%. For large planes this is of extreme importance.

Letter Patents have been granted for the design of this device.

Fowler Wing No. 2 is of such area that it can be adopted to 75 percent of the commercial types of planes now in use, giving them an all around improved performance. This will be possible on planes using power up to 225 H. P. and a total plane weight of 3000 lbs. Fowler Wing No. 2 is to be put in production and made available to the industry.

The Fowler Wing is also available to solve any special problems which it may be desired to accomplish.

Fowler Airplane Wings, Inc.
New Brunswick, New Jersey.

Document 3-12(b), Consolidated Aircraft Corporation to Harlan D. Fowler, 21 January 1929, Folder "Fowler Flap," copy in San Diego Aerospace Museum, CA.

January 21, 1929

Mr. Harland D. Fowler
The Miller Corporation
New Brunswick Airport
New Brunswick, N.J.

Dear Sir:

Could you send us information on the typical detail design of your variable area wing? Also, tell us what the span of the PA-3 biplane which you used was, both as a biplane and as a monoplane with normal and standard wings.

Very truly yours,

CONSOLIDATED AIRCRAFT CORPORATION

Asst. to General Manager

Document 3-12(c), Harlan D. Fowler, "Variable Lift,"* Western Flying *10 (November 1931): 31-33.

November, 1931

VARIABLE LIFT
BY: HARLAN D. FOWLER

In spite of the achievements of the Guggenheim Safe Aircraft Competition, little progress seems to have been made toward the solution of the problem of increasing the lift of wings without sacrificing top speed. The following article is presented with the hope that it will stimulate more thought on the subject. -The Editors.

Time consumed in taking off and landing is a matter of a few minutes, whereas straightaway flying takes hours. Simply because sufficient lift must be available for this landing and take-off, we afflict ourselves with the use of oversized wings. When once in the air, the smallest of wing, consistent with controllability, can be used safely.

Why, then, do we continue to be so far removed from the most important and far reaching development of variable lift wings? As far back as 1910, the subject of utilizing variable lift wings was frequently discussed by engineers. The methods suggested were:

(a) variable camber,
(b) variable area,
(c) variable angle of attack, and later
(d) slotted wings.

Many laboratory experiments have been made on airfoils with the camber adjustable from a streamline section to that of a heavily convexed section, with promises of a wide variation in aerodynamic characteristics. Variable area has presented a mechanical problem of the first order. The variable angle of attack offers the least, if any, improvement, although several airplanes have been tried out using this idea. Perhaps the most successful devices have been the various forms of slotted wings. All four methods have been considered extensively in published articles. However, it is unfortunately true that the benefits obtained from each type alone have not justified its continued development.

THE FOWLER WING PRINCIPLE

It was long recognized by the writer that if the best features of each one could be incorporated into a simultaneous and coordinated alteration of the wing, the fullest advantage would be utilized. This was easier "thought of" than practically solved. Fundamentally and foremost, the basic structure of the wing must not be interfered with.

Hence, the conception of the Fowler Variable Area Wing, which incorporates the following points: (1) A structurally sound basic wing; (2) An increase in area; (3) Variable angle of attack; (4) A variable camber, and (5) The slotted wing.

The lift of the normal or closed wing is increased about 100 per cent by the combined use of these relations. Each feature contributes to the increased lift in the following proportions: Variable camber, 43 per cent; variable area, 28 per cent; recessed camber, 7 per cent; and slots, 22 per cent. And it is all accomplished by the one simple expedient of extending the auxiliary area wing downwardly and rearwardly, which rapidly brings about the simultaneous characteristics of each desirable relationship.

WIND TUNNEL TESTS

Figures 1 and 2 give the characteristics for the normal and extended area. The tests were conducted at the New York University. The maximum Ky normal is .0036 and extended .00556. The normal area is 63 sq. in. and extended 80 sq. in. Viewed on the assumption that the normal wing characteristics are influenced by these alterations by virtue of being self-contained, it will have the equivalent maximum Ky of

$$.00556 \times 80/63 = .00705$$

or

$$.00705/.0036 - 1 = 0.96 \text{ increase}$$

Full flight speed trials at near stall gave a maximum Ky of .0073 as referred to the normal area—the highest lift obtained from any airfoil.

This increase in lift was obtained without impairment of aileron effectiveness, only 70 per cent of the wing span being utilized for the auxiliary wing.

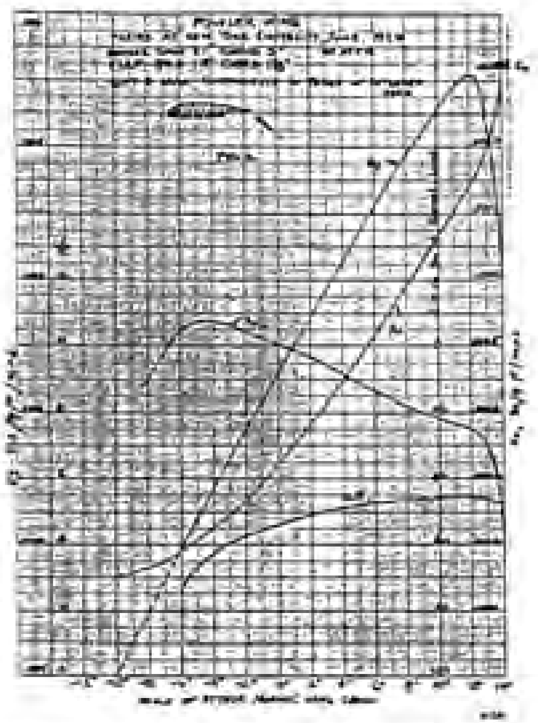
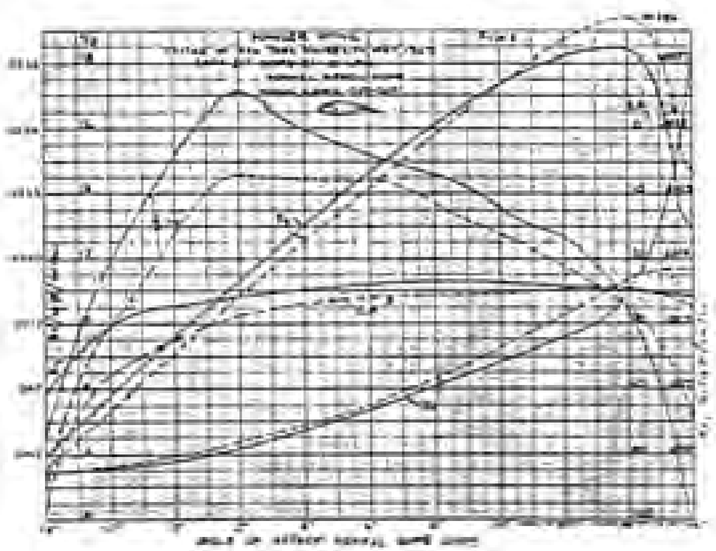


FIG. 1 (below) shows results of wind tunnel test of normal wing with and without recess.

FIG. 2 (right) shows test with wing extended. Max. Ky of 00356 occurs at only 12 deg. angle of attack



FULL FLIGHT TESTS

Performance tests were made with a Pitcairn PA-3 which was originally a biplane of 338 sq. ft. An OXX-6 engine of 100 h.p. and the same wood propeller was used throughout the trials without alterations or overhauling. The following data were obtained:

	Original PA-3	Fowler No. 1	Wing No. 2
Low speed, normal, M. P. H.	46.2	58.2	57.0
Low speed, extended, M. P. H.	---	44.5	41.0
High speed, M. P. H.	91.0	101.0	104.0
Cruising speed, M. P. H.	77.0	84.0	87.0
Rate of climb, ft./min.	515	430	570
Take-off time, seconds	10	11	10
Landing time, seconds	9	11	9
Total weight of plane, lbs.	1936	1990	2000
Wing area, normal, sq. ft.	338	136	143
Wing area, extended, sq. ft.	---	166	181
Wt. per sq. ft., normal	5.72	14.65	13.8
Wt. per sq. ft., extended	---	12.00	10.75
Speed range	1.97	2.16	2 42
Design load factor of wing	5.50	8.00	10.00
Weight of wings, complete, lbs.	377	395	409

Performances for the PA-3 and Fowler Wing No. 1 are based on flight tests over a one-mile course and are the averages of several runs.

Wing No. 2 is an entire new design with a special normal airfoil and aspect ratio of 9. It is designed for a gross-weight of 2850 pounds with a load factor of 7, which for-the 2000-pound plane is equivalent to a load factor of 10.

It will be noted that in spite of using an extended area of less than half that of the biplane, the stall speed was nearly 2 m.p.h. lower.

The stall speed for the normal wing monoplane is 58.2 m.p.h. and represents the typical monoplane. By extending the area, this speed was reduced to 44.5 m.p.h. Thus, the average monoplane can have its stall speed reduced not less than 14 m.p.h.

The high speed was increased from 91 m.p.h. for the biplane to 101 m.p.h. for the normal area Wing No. 1. For the original biplane to have attained 101 m.p.h., at least 136 horsepower would be necessary, resulting in a heavier engine and increased fuel capacity, or an increase in weight of about 185 pounds, using a water-cooled engine. This in turn would raise the stall speed from 46.2 to 48.7 m.p.h. The most important fact is this—the additional cost of a new engine at about $2,000 far exceeds the cost of a set of variable wings.

WEIGHT COMPARISON

Wing No. 2 has been designed according to the strength requirements of the Department of Commerce and approved for installation on planes with gross weight up to 2850 pounds and up to 225 h.p. The actual weight of the wing is itemized as follows:

Main wing, with fittings	242.501bs.
Ailerons	17.001bs.
Five gallons gas tank	8.50 lbs.
Main wing, complete	268.
Extension surface	36.00
Extension surface rails, etc.	24.00
Extension surface controls	6.00
Extension assembly, complete	66.
Fowler Wing No. 2, complete	334.

Struts and wires for installation to plane weigh about 75 pounds, giving a total weight of 409 pounds.

This wing can be designed for a closed cabin without special difficulties.

For comparison with the conventional wings, the following typical data are given:

	Rated H. P.	Gross Weight	Area sq. ft.	Weight Wings	Stall Speed	Load Factor
Fowler Wing	225	2850	181	409	53.0	7.00
"A" biplane	180	2877	351	425	49.0	7.00
"B" biplane	165	2702	289	450	54.0	7.00

Note—Even though Wing No. 2 is lighter than the conventional wings, no attempt was made to resort to expensive light-weight detail construction. With such an excess of lift available, this is unnecessary. However, with design refinement, at least 10 per cent further saving in weight is possible. A and B are well-known airplanes.

APPLICATION TO CANTILEVER WINGS

As shown before, for the same stall speed, the area may be reduced one-half by using the variable area wing. Since the extension feature can be easily incorporated in a cantilever wing, the span can be reduced 25 per cent for the same aspect ratio. Since the gross weight is about the same, the cantilever bending moment is correspondingly reduced. A stiffer and cheaper structure would result from this reduction in size.

From the standpoint of strength, the determination of structural sizes is simple and straightforward, being governed entirely by conventional design practice. No unusual or tricky supporting members nor articular parts are used in the wing.

For extremely high speed purposes, it is necessary to go into high landing speeds. The Schneider Cup Race planes land at speeds varying from 100 to 125 m.p.h.

Let us assume that with a high speed airfoil, the wing loading is 30 pounds per sq. ft. Substituting the variable area wing, the extended wing loading would be 60 pounds per sq. ft. at 100 m.p.h. stall speed and normal area is 76 pounds per sq. ft. at 145 m.p.h. stall speed. This is a reduction of 45 m.p.h. It is acknowledged that the wing loadings are very high, but the wing could be of a cantilever type and very small in size. The high speed would be greatly improved.

When landing with the wing extended, the glide is steep and the run on the ground short.

WING REDUCES NEED FOR INCREASED POWER

If increased high speed is desired, it is usually the practice to put in a higher power engine. This will not be necessary if the Fowler wing is used.

Let us consider a pure cantilever monoplane with a 420-h.p. engine, wing area 265 sq. ft. and a gross weight of 4500 pounds. The pay load is 600 pounds without including 150 gallons of fuel. Stall speed is 66 m.p.h.

Installation of a 525-horsepower engine would only increase the high speed of this plane from 200 to 215 m.p.h. It is an unfortunate fact that in changing to higher power, the expected increase in speed is rarely obtained. The average loss is about one-third of the theoretical increase. This is largely attributed to the increase in cylinder number and diameter, overall engine diameter, and loss of propulsive efficiency. Therefore, it is very probable that the high speed of 215 m.p.h. for the 525-h.p. engine may be more like 210 m.p.h. or even less, if available data are trustworthy.

The added cost of the larger engine is about $600. The larger gas tank to receive 30 gallons more fuel, propeller, etc., would add at least $100 more, or a total of $700.

A variable wing would cost about 15 per cent or $375 less than the original wing. The total net initial saving would be $1,075. The operating savings on gas consumption, depreciation of engine, and maintenance would also be considerable.

CLIMB AND CEILING

Numerous tests with the extension in various locations between all-in and all-out shore a very definite gain in the rate-of-climb when set about halfway out. This is, then, the best setting for take-off and climb.

With the fully extended wing, the rate of climb is lower, as is to be expected, in view of the lower L/D. Incidentally, for the same reason, the high speed was found to be about 79 m.p.h. This condition is in reality a decided safety measure in that it would be impossible to maneuver sharply. It thus prevents undue strain on the extended surface and supports.

Western Flying

FIG. 3 — Installation of Fowler wing with extension full out. Nose slot

The planes on which the Fowler wing was used had fixed stabilizers, and these were not readjusted at any time.

The extension can be operated by the pilot while flying and set at any position desired. The position of the control stick is changed forward when extending the wing, but little change of balance is noted for the two conditions of all-in or all-out.

The plane had adequate control at and beyond stalled position.

DESIGN FEATURES

The normal wing, which is of two halves joined at the center, is of conventional construction, although employing materials in a different manner. The spars are of solid spruce. The ribs are of 3/8 and 7/16-inch spruce. The nose is of solid balsa wood. (Fig. 4.) No internal drag bracing is used because the covering is of plywood which is glued and nailed directly to the spars and ribs.

The auxiliary wing has a single spar to which are secured the solid ribs and balsa wood nose piece. The whole is plywood-covered. This small wing fits snugly under the trailing portion of the main wing. (Fig.4.)

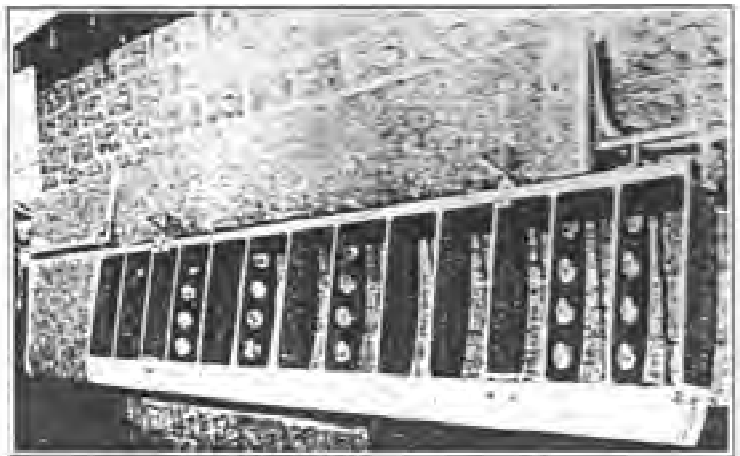


FIG. 4—Showing construction of the Fowler wing in the closed position

Figure 5 is a drawing of the complete combination, showing the supporting rail, trolley and relation of the large and small airfoils. The four supporting rails, which are 3/16 inch thick by 3 1/8 inches deep, are of steel, but may be of duralumin.

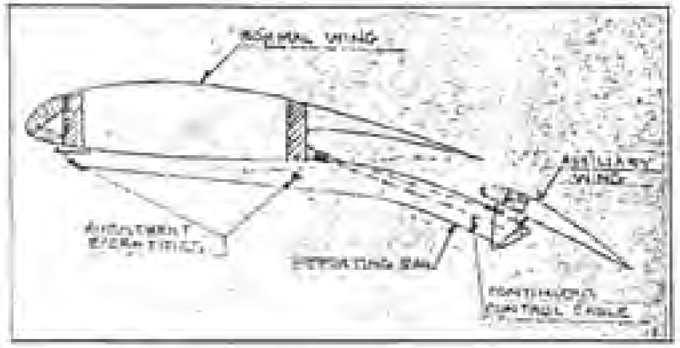


FIG. 5—Cross-section of Fowler wing. Note recess under trailing surface to receive extension

These rails are attachable in up or down movements by eccentrics located at the main fitting attachments so as to provide clearance between the auxiliary wing and the under surface of the main wing when it is closed.

Operation of the extension is by means of a continuous cable wrapping around a drum. To this drum is secured the control shaft and handle reaching down to the pilot who can operate it at his convenience. All control cables, including those for the ailerons, are led along the open face of the rear spar. Nothing is concealed.

COMMERCIAL APPLICATIONS

Due to the high lift of the Fowler wing, it can be applied as follows: (1) Light planes, stall speed 25 m.p.h.; (2) cargo planes, actual payload, 7 pounds per h.p.; (3) optional equipment, substitute for conventional wings, and (4) reduced initial cost of planes and better performance.

Document 3-12(d), Harlan D. Fowler to William B. Mayo, 15 August 1932, copy in Accession 18, Box 96, Folder "Fowler Variable Wing," Henry Ford Museum and Greenfield Village Research Center, Dearborn, Mich.

615 Fifteenth Avenue,
San Francisco, Cal.
July 19, 1932

Mr. Wm. B. Mayo,
Ford Motor Company,
Airplane Division,
Dearborn, Mich.

Dear Mr. Mayo:

With reference to yours of June 28th, I am taking the liberty of sending a copy of the N.A.C.A. report No. 419 (on a loan) so that a study can be made of the Fowler Variable Area Wing.

We have found Position No. 6 to be an all around ideal arrangement and to which the enclosed blueprint of the characteristic curves applies.

It may be to your interest to learn that several other large companies are considering the use of the Fowler Wing. If your research is convincing I shall be pleased to consider a mutual arrangement.

Very truly yours,
Fowler Airplane Wings, Inc.

Document 3-12(e), H.A. Hicks to Harlan D. Fowler, 10 September 1932, copy in Accession 18, Box 96, Folder "Fowler Variable Wing," Henry Ford Museum and Greenfield Village Research Center, Dearborn, Mich.

Sept. 10, 1932.
Mr. Harlan D. Fowler,
120 Webster Street,
San Francisco,
Cal.

Dear Sir:

In reply to your letter of August 15th concerning your variable area wing, we inform you that this company would not be interested in purchasing manufacturing rights at this time.

Yours very truly,

FORD MOTOR COMPANY
(Airplane Division)

H.A. Hicks,
Aircraft Engineering.

Document 3-12(f), T.P. Wright, "The Application of Slots and Flaps to Airplane Wings in America," U.S. Air Services *(September 1933): 29-31.*

September, 1933

The Application of Slots and Flaps to Airplane Wings in America
T. P. WRIGHT
Vice President and Director of Engineering
Curtiss Aeroplane and Motor Company, Inc.

It is the object of this article, within the space limitations set, to present in general terms the objective sought in incorporating slots and flaps on airplane wings and to indicate briefly the application which has been made in this country.

AIRPLANE CHARACTERISTICS

For the purpose of this article it appears desirable to segregate the qualities of the airplane which are affected by these devices into two main groups. These are qualities affecting safety and qualities affecting performance.

Under the heading SAFETY the functions which are affected by changes in wing characteristics are landing speed, susceptibility to spinning, and controllability, particularly beyond the stall. Other things being equal, it appears obvious that the airplane with the lower landing speed, the airplane having freedom from incipient spinning tendencies, and the airplane which maintains lateral control beyond stalling speed will be the safer craft.

Items of performance which are affected by changes in wing characteristics are high speed, rate of climb and ceiling, climbing angle, and angle of descent. Higher and higher speeds are desired each year as aviation progresses so that the advantages of any device tending to increase speed are obvious. Rate of climb and ceiling are only of relative importance, the degree of importance depending, of course, upon the particular type of airplane under consideration. The last two functions, angle of climb and angle of descent, are becoming recognized as being items of major importance more and more as the external cleanness of airplanes increases.

DESCRIPTION OF THE SLOT AND FLAP

Let us now briefly define the slot and flap. The slotted wing is one which has an auxiliary airfoil at its leading edge, either fixed or movable, so that in either case, in its extended position, a slot is formed which induces a smooth flow over the wing at higher angles of attack than are obtainable with the omission of the device. It is important to note that the slotted wing retains the same slope of lift curve as maintains with the unslotted wing. The function of the slot is to increase the angle at which burbling occurs. Therefore, if other airplane characteristics are arranged to conform with the use of a higher lift angle, then the maximum lift of the airplane is substantially increased by the use of the slot.

The flap is a device which has for its essential object the increasing of the camber of the basic wing section to which it is applied. In its simplest form the flap consists of a hinged portion of the trailing edge of the wing, which, when deflected downward, obviously increases the wing's camber. One variation of this simplest form of flap is accomplished by changing the form of the leading edge of the flap, to create a slot at that point, which, as in the case of the slot at the leading edge of the main airfoil above described, gives the beneficial effect of smoothing out the flow over the flap when deflected.

Another form of flap is described as the split trailing edge type whereby only the lower portion of the trailing edge is deflected downward in operation, rather than the whole trailing edge, as in the case of the simple-hinged flap. In a further variation of this type, this lower portion is moved aft at the same time that it is deflected downward, thus to a certain extent increasing the wing area in addition to increasing its camber. A still further variation in flap design is incorporated in the Fowler type in which the lower split flap section is made up into the form of an airfoil and is moved aft through its chord length in addition to its downward angular deflection and is so shaped that when in this final operating position a slot is formed at its leading edge.

The increase in maximum lift due to the type of flaps above described varies but in all cases is substantial. An extensive series of tests has been conducted by the National Advisory Committee for Aeronautics, and the results tabulated, to show the effect on airfoil characteristics of all of these types of flap and of the flap in combination with leading edge slots. It is shown in these tests that, based on basic wing area, lift increases of more than 100% are readily obtainable with accompanying speed range increases of more than 75%. The possibilities of such tremendous improvements in airfoil characteristics are obvious and when airplanes designed with them have other characteristics properly correlated, it is believed that without doubt they will find a substantial application in the majority of types of airplanes.

IMPROVEMENT IN AIRPLANE CHARACTERISTICS OBTAINABLE BY USE OF SLOTS AND FLAPS

Let us now note the possibilities which these devices hold for improving our fundamental airplane qualities, namely, safety and performance. Although the slot can contribute slightly to the lowering of the landing speeds, its chief utility lies in its effect in preventing spins and in permitting lateral control at high angles. It has one other possibility when interconnected with the flap, namely, the supplying of the necessary force for operating the flap, thus relieving the pilot of this operation.

A properly designed auxiliary airfoil when in the unslotted position will increase the wing drag inappreciably and therefore not affect the high speed of the airplane adversely. The high speed of the airplane can, however, be increased for a given landing speed by the use of the flap with its increased lift coefficient and resulting decrease in wing area, although the degree to which this application can be used is contingent upon the allowable resultant characteristics which are injured by the application, namely, rate of climb, ceiling, and angle of descent (perhaps better defined as sinking speed). However, increase in sinking speed or angle of descent to a limited extent is a decided advantage and one which is becoming more and more essential in preventing floating during landing in modern airplanes having ever-increasing fineness. The improved angle of climb obtainable by the use of flaps is also a quality of importance in airplanes of modern design.

In summary, therefore, we find that, with the slot, improved safety is the chief contribution with some possibility of increasing lift and therefore speed range, provided the airplane is otherwise designed to permit of the realization of the high angle of attack necessary. With the flap a very definite increase in speed range is obtainable together with improved takeoff and landing characteristics.

We thus see that it is possible to improve certain performance characteristics of the airplane by the use of slots and flaps. If the basic considerations determining the design of a particular type coincide with the characteristics which are improved by these devices, it then appears that their use is justified. There are certain other types of airplanes wherein the basic characteristics governing the design are injured by the installation of slots and flaps, thereby making their adoption of no value or of questionable value.

TYPES OF AIRPLANES

There are two major classifications into which airplanes may be placed, namely, MIILITARY and COMMERCIAL. Within each group there are subdivisions such as training, fighting, attack, observation, and load carriers, including bombers and transports, in the former; and training, utility, private owner and transport, including mail and passenger, in the latter. Each type demands a special combination of characteristics to permit of the best fulfillment of its functions. It will probably be agreed that the function of safety will be of less importance in military airplanes than commercial, and within the list of commercial types will be of less importance in the mail plane and utility plane than in the case of the private owner plane and passenger transport. This statement therefore in itself simplifies the study necessary in deciding upon the use of slots alone on a given design. The decision on the application of flaps, however, is more difficult, as there are few classes of plane in either broad subdivision which are not susceptible of some improvement by the use of flaps, or flaps in combination with slots.

SPECIFIC APPLICATIONS

In the case of military planes it appears that the attack type, requiring high speed with a lesser degree of climb and ceiling than other types, represents one possible application. This applies particularly to Army planes. An additional problem is involved in the case of the Navy where low landing speed for carrier operation is of far greater importance than in the case of the Army. Therefore in the Navy it appears logical that a wider application may be found desirable.

It is believed that in the Commercial field an even wider application is probable both because of the increased importance of safety and because of the lesser importance, in most cases at least, of the characteristics which are admittedly injured by the use of flaps or flaps in conjunction with slots.

DEVELOPMENT IN THIS COUNTRY

With the above discussion, covering a description of the devices we are considering and of the airplane characteristics which they alter, thereby indicating their possible usefulness in the various types of airplane being developed, before us, let us now see what application has been made in this country. The basic scientific development of the slotted wing was initiated in England and Germany. Apparently the basic idea of improving airflow by means of the Venturi action of a slot was coincidentally and independently worked out by Messrs. Lachmann and Handley Page. Credit for the major engineering development with reference to specific and practical devices is due to Handley Page in England. Important note of this development was first taken in this country in 1927 when the Navy Bureau of Aeronautics sent representatives to England to study the progress made and later negotiated certain license contracts with Handley Page. Shortly thereafter several applications of the slot were made on Naval Aircraft, including Curtiss F7C-1 single seater fighter, a Consolidated training plane, and a Vought observation plane. This application consisted of the installation of slots at the outer portion of the wing, forward of the aileron, only.

The first commercial application seen in this country was on the *Moth* airplane, constructed and sold here in considerable numbers under British license in 1928 and 1929. In this case also the slot at the wing tip only was used.

The first combination of slots along the whole span of the wing, with flaps also on the whole span (floating wing tip ailerons being used in this case for lateral control) appeared on the Curtiss *Tanager*, winner of the Guggenheim Safe Aircraft Competition in 1930. It should be noted that the plane most nearly approximating the *Tanager* performance in the competition also employed slots and hinged flaps, although in this case conventional ailerons were used. This was the Handley Page *Gugnunc*. It is believed significant that the winner and runner-up in this competition were both equipped with slots and flaps, especially as it is considered that the tests set up in the competition correctly represented airplane characteristics essential to safety. Subsequent developed improvements in general airplane design could equally well be applied to the *Tanager*. The *Tanager* development was the result of intensive technical research in the Garden City laboratory of the Curtiss Aeroplane and Motor Company during the period 1927 to 1929. The airplane was delivered to the Guggenheim Committee for tests in October, 1929. It is believed that this theoretical wind tunnel and full flight research was the most extensive carried out to that date in this country. The data determined from this study have been substantially augmented by the results of the very fine program of testing subsequently undertaken and accomplished by the National Advisory Committee for Aeronautics.

Although some work was carried out in 1930 on the application of slots and flaps to commercial, private owner class airplanes, the development was curtailed and then abandoned because of the need for economy at the start of the depression. Undoubtedly during the last three years a great number of successful applications would have been made but for the above economic conditions.

The next application appearing in this country, and the first one applying to a military airplane, was the use of both slots and flaps on the Curtiss XA-8 developed as an attack airplane for the Army Air Corps. The ship is the low, thin-wing, externally braced monoplane type, carrying a useful load about 50% greater than maintains for other classifications of two-seater military plane. The requirement of high sea level speed is of major importance in this class. This airplane flew in the summer of 1931, exceeding requirements substantially. The excelling performance of this plane represented the first large step in high speed increase of Army Air Corps planes since the War as it roughly stepped up speeds from the 150 m.p.h. to 200 m.p.h. bracket. This ship, at the time of its appearance, led the Military Field, when judging excellence on the criterion of load carrying times speed per given horsepower. Subsequently 13 similar ships were constructed and service tested, followed by procurement in a production order of 46 now underway.

Other applications to military developments, both in the Army and Navy, have been made and are being made, the results of which can only be described and their value judged after service testing is completed and official release of information is available.

Commercially, where from basic considerations application should be the most general, slots and flaps have been decidedly neglected at least until very recently, where several applications of the flap have appeared. This has undoubtedly been due to the economic conditions of the country and to the absence of properly charted information on characteristics of airfoils with slots and flaps, now fortunately available in published National Advisory Committee reports.

At the recent conference at the Langley Field Memorial Laboratory of the National Advisory Committee for Aeronautics considerable stress was laid on the possibilities in the use of various types of flap and of these types in combination with slots. The interest in this country, however, in slots continues meager in spite of the quite general adoption of these devices in various forms on a number of planes in several European countries. One other consideration which has undoubtedly retarded general interest in these devices is the fear of additional complexity and weight increase due to their use. The importance of these latter features is fully realized, but nevertheless it is believed that if, as is maintained, the aerodynamic improvements obtainable by their adoption are substantial, and assuming intelligent application in each specific case, the obstacles of complication and weight will be surmounted and a wider use of slots and flaps made in the future.

Document 3-12(g), Robert C. Platt, "Aerodynamic Characteristics of a Wing with Fowler Flaps including Flap Loads, Downwash, and Calculated Effect on Take-off," NACA* Technical Report *534 (Washington, 1935).

REPORT No. 534

AERODYNAMIC CHARACTERISTICS OF A WING WITH FOWLER FLAPS INCLUDING FLAP LOADS, DOWNWASH, AND CALCULATED EFFECT ON TAKE-OFF

By ROBERT C. PLATT

SUMMARY

This report presents the results of an investigation in the N.A.C.A. 7- by 10-foot wind tunnel of a wing in combination with each of three sizes of Fowler flap. The purpose of the investigation was to determine the aerodynamic characteristics as affected by flap chord and position, the air loads on the flaps, and the effect of the flaps on the downwash. The flap position for maximum lift; polars for arrangements considered favorable for take-off; and complete lift, drag, and pitching-moment characteristics for selected optimum arrangements were determined. A Clark Y wing model was tested with 20 percent *c*, 30 percent *c*, and 40 percent *c* Fowler flaps of Clark Y section. Certain additional data from earlier tests on a similar model equipped with the 40 percent *c* Clark Y flap are included for comparison. Results of calculations made to find the effect of the Fowler flap on take-off, based on data from these tests, are included in an appendix.

AIRFOIL ORDINATES

CLARK Y

(All values in percent airfoil chord)

Station	Ordinate upper	Ordinate lower	Station	Ordinate upper	Ordinate lower
0	3.50	3.50	40	11.40	0
1.25	5.45	1.93	50	10.52	0
2.50	6.50	1.47	60	9.15	0
5	7.90	.93	70	7.35	0
7.50	8.85	.63	80	5.22	0
10	9.60	.42	90	2.80	0
15	10.68	.15	95	1.49	0
20	11.36	.03	100	.12	0
30	11.70	0			

Leading edge radius = 1.50

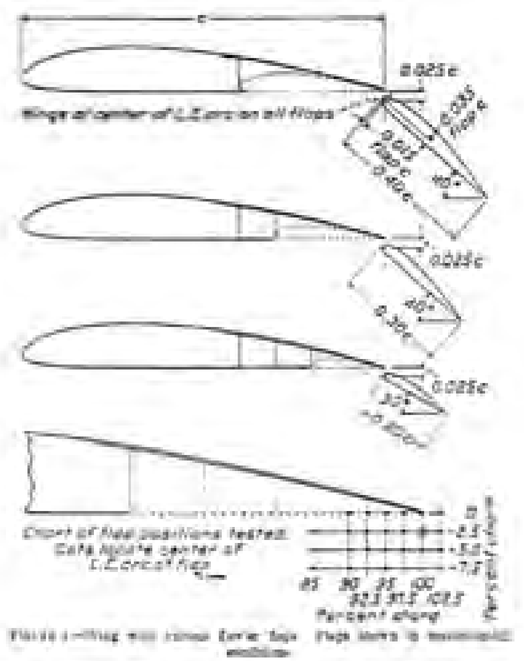
The maximum lift coefficient obtainable, based on original wing area, had a nearly linear increase with flap chord up to 40 percent, but the maximum lift force per unit of total area increased very little beyond the value obtained with the 30 percent *c* flap. The maximum load on the flap occurred very nearly at the maximum lift of the wing-flap combination and was nearly 1½ times the load that would result from uniform distribution of the total load over the total area. In general, the flap appeared to carry a large proportion of the additional lift caused by its presence and to have its center of pressure much nearer the leading edge than it would normally be in free air. The addition of the Fowler flap to a wing appeared to have no appreciable effect on the relation between lift coefficient and angle of downwash. The calculations in the appendix show that, by proper use of the Fowler flap, the take-off of an airplane having wing and power loadings in the range normally encountered in transport airplanes should be considerably improved.

INTRODUCTION

During the past few years the use of flaps on high-performance airplanes as a device for reducing space required in landing has become common. Thus far split flaps have been most generally used, probably because of their simplicity of application and their superiority in giving steep gliding approaches and short landing runs: the features of flaps with which designers have been most concerned. In order to retain satisfactory operation from normal flying fields with fast airplanes, however, the use of high-lift devices that improve take-off as well as landing is desirable. Since drag is unfavorable to take-off, the comparatively large drag of split flaps places them among the least promising of high-lift devices in this respect. The Fowler flap appears to offer a better compromise between these conflicting requirements. For equal sizes it will give higher maximum lift with no higher profile drag than most other flap arrangements and its comparatively low drag at high lifts is favorable to take-off and steep climb. This effect would normally entail some sacrifice of steep gliding ability.

Although sufficient data to form some estimate of the performance to be expected from an airplane equipped with Fowler flaps are available (references 1 and 2), they are inadequate for normal design purposes. The purpose of the tests reported herein is to provide data to form a rational basis for the design of airplanes equipped with Fowler flaps. It appears that for the present the purpose will be attained by making available the following information: effect of flap size on aerodynamic characteristics attainable, aerodynamic loads applied to the flap in various conditions, and effect of the flap on downwash. In addition, a convenient method of estimating the effect of high-lift devices on airplane take-off should prove of assistance in cases where this performance feature merits special attention.

The tests were made in the 7- by 10-foot wind tunnel of the N.A.C.A. (reference 3) at Langley Field, Va., during the summer and fall of 1934.



MODEL

The basic wing was built of laminated mahogany to the Clark Y profile (table I) and had a span of 60 inches and a chord of 10 inches. The trailing edge was cut away and the upper surface replaced by a thin curved metal plate. The lower surface was left open at the rear to serve as a retracting well for the flaps. Blocks were inserted to maintain the correct size of well for each size of flap tested. Figure 1 shows the profile of the wing with the various flaps in place.

The two smaller flaps were made of duralumin to the Clark Y profile and had spans of 60 inches and chords of 2 and 3 inches. The largest flap, which is the one described in reference 2, was made of mahogany and had a 4-inch chord. The flaps were supported on the wing by fittings attached to ribs located in the retracting well. Several sets of attachment holes in the ribs, combined with several sets of fittings, gave the range of flap positions shown in figure 1. The flaps were supported on the fittings by hinges located at the center of the leading-edge arc of the flaps, angular adjustment being obtained by set screws attached to the flap moving in quadrantal slots in the fittings. In general, where the term "flap position" is used, the position of the flap hinge axis is indicated, irrespective of angle, and flap angle is measured between the chord lines of the wing and the flap.

TESTS

Five groups of tests were made in obtaining the data presented in this report. These five groups dealt with maximum lift, optimum flap arrangement for take-off, standard force tests of optimum arrangements, flap loads, and downwash behind the wing with various flap arrangements.

Maximum lift.—The maximum lift coefficients obtainable with the 0.20 *c* and 0.30 *c* flaps at various positions shown in figure 1 were found by tests in which the flap angle was increased from 20° in 10° steps until the peak of the variation of C_{Lmax} with flap angle was defined for each position. The range of positions in both cases was sufficient to surround the point at which the highest lift coefficient was found, thus isolating an optimum position in each case. Similar surveys had previously been made with the 0.40 *c* flap (reference 2) and were not repeated at this time.

Optimum take-off arrangement.—Lift and drag data were taken at a range of flap angles between 0° and that giving maximum lift for a series of flap positions somewhat more restricted than the range used in the maximum-lift tests. Care was exercised in these tests also to surround what was judged to be the optimum setting, both as regards position and angle.

Standard force tests of optimum arrangements.—A series of final force tests, consisting of lift, drag, and pitching-moment measurements, was made at the flap positions considered to be of special interest. These included tests of the maximum-lift arrangement of each flap, of the optimum take-off arrangement of each flap, and of an arbitrarily selected arrangement representing partial retraction of each flap.

All tests in these first three groups were conducted in accordance with standard force-test procedure as described in reference 3.

Flap loads.—Air loads acting on the flaps were found by supporting the flaps independently of the wing, at the same position and angle as used in the final force tests of the wing-flap combinations, and by measuring the forces on the wing alone in the presence of the flap. The flap loads could then be readily computed. In order to find the center of pressure of the load on the flap, the flap hinge moment was measured by observing the angular deflection of a long slender torque rod required to balance the flap at the angle in question. Similar measurements of loads and center-of-pressure locations on split flaps are more completely described in reference 4.

Downwash.—Measurements were made with "pitot-yaw" tubes attached to the wing by a rigid support. The reference position thus moved in the air stream as the angle of attack was changed but remained the same with respect to the wing, as does the tail of an airplane. The angles of downwash, however, were referred to the initial direction of the free air stream. The apparatus could be adjusted to various horizontal distances behind the wing. The pitot-yaw tubes were ordinary round-nosed pitot tubes with two additional nose holes at 45° above and below the tube axis. Alcohol manometers were used to read the pressures, and the tubes were calibrated in test position in the clear-tunnel air stream.

The wind tunnel is of the open jet, closed return type, with a rectangular jet 7 by 10 feet in size. A complete description of the tunnel, balance, and standard force-test procedure appears in reference 3.

Tests were run at a dynamic pressure of 16.37 pounds per square foot, corresponding to an air speed of 80 miles per hour at standard sea-level conditions. The Reynolds number {AQ3} of the tests, based on the 10-inch chord of the wing without flaps, was approximately 609,000.

PRECISION

The accidental errors in the results presented in this report are believed to lie within the limits indicated in the following table:

Wing data:		Flap load data:		Downwash data:	
α	±0.10°	C_{Nf}	±0.10	ε	±0.5°
C_{Lmax}	±.05	C_{Xf}	±.06		
$C_{me/\lambda}$	±.008	C_{Hf}	±.004		
$C_D(C_L=0)$	±.001	Flap angle	±.25°		
$C_D(C_L=1)$	±.004	Flap position	±.003		
$C_D(C_L=2)$	±.008				
Flap angle	±.25°				
Flap position	±.0015*c*				

Consistent differences between results obtained in the 7- by 10-foot wind tunnel and in free air may be ascribed to effects of the following factors: Jet boundaries, static-pressure gradient, turbulence, and scale. In order that the present results be consistent with published results of tests of other high-lift devices in the 7-by 10-

foot tunnel, no corrections for these effects have been made. Corrections of several sets of airfoil results have indicated that the values of the jet-boundary correction factors, $\delta_\alpha = -0.165$, and $\delta_D = -0.165$, used in the standard equations (cf. reference 5) are satisfactory for a 10-inch by 60-inch wing. The static pressure in the jet decreases downstream, producing an increment in C_D of 0.0015 on normal 12 percent *c* thick rectangular airfoils. Evidence at present available indicates that the effect of the turbulence in this tunnel is small as compared with the other consistent errors.

RESULTS AND DISCUSSION

All test results are given in standard non-dimensional coefficient form. In the case of a wing with a retractable surface, the convention of basing coefficients on the area that would be exposed in normal flight, that is, the minimum area, has been adopted. The coefficients used are then defined as follows:

subscript *w* refers to the basic wing
subscript *f* refers to the flap
$q = 1/2\ \rho V^2$
$C_L = \text{lift}/S_w q$
$C_D = \text{drag}/S_w q$
$C_m = \text{pitching moment}/c_w S_w q$
C_{Nf} = normal force on flap (perpendicular to flap chord)/$S_f q$
C_{Xf} = longitudinal force on flap (along flap chord)/$S_f q$
C_{Hf} = flap hinge moment/$S_f c_f q$
$\in$, angle of downwash, degrees.

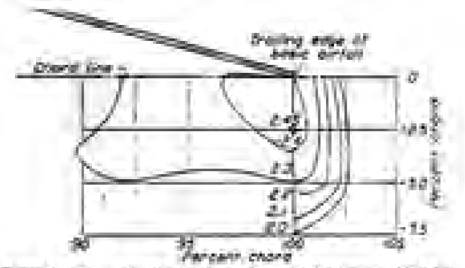


FIGURE 2.—Contours showing variation of $C_{L_{max}}$ with flap position. [illegible]

FIGURE 3.—Contours showing variation of $C_{L_{max}}$ with flap position. [illegible]

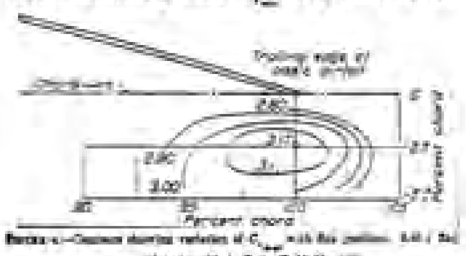


FIGURE 4.—Contours showing variation of $C_{L_{max}}$ with flap position. [illegible] (data from N. A. C. A. T. N. No. [illegible])

Maximum-lift condition.—The results of the maximum-lift tests are presented as contours showing variations of C_{Lmax} with flap hinge position, irrespective of flap angle. Figures 2, 3, and 4 show contours for the 20 percent chord, 30 percent chord, and 40 percent chord flaps, respectively. Data on the 40 percent chord flap are taken from reference 2, no further tests having been considered necessary on that size of flap after an analysis was made of the data for the two smaller flaps. The optimum position is the same for all three flaps: 2.5 percent of the main wing chord directly below the trailing edge. The optimum angle

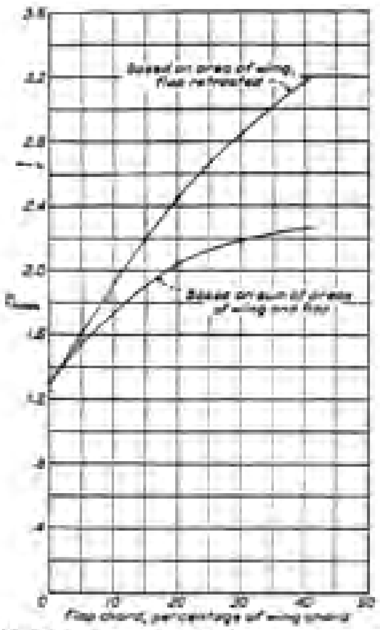


FIGURE 5.—Variation of $C_{L_{max}}$ with flap size. Flap at maximum-lift position and angle.

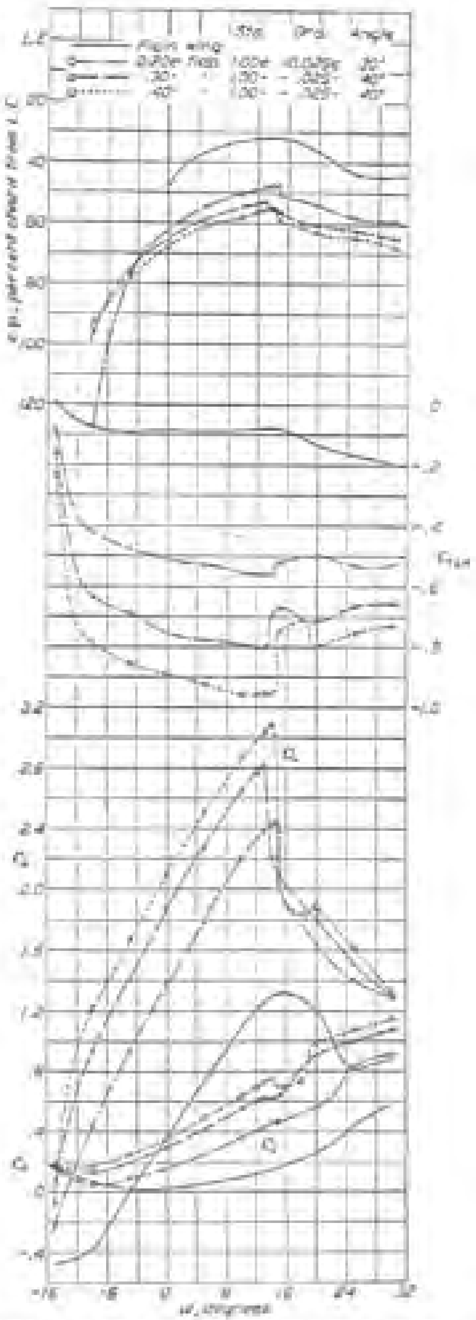


FIGURE 6.—Lift, drag, and pitching-moment coefficients for the maximum-lift condition.

TABLE II

PLAIN WING

(Flap retracted)

α	[illegible]	[illegible]	[illegible]
Degrees [illegible]	[illegible]	[illegible]	[illegible]

TABLE III

DATA FOR THE MAXIMUM-LIFT CONDITION

[illegible]

α	C_L	C_D	$C_{m_{c/4}}$	[illegible]	[illegible]	[illegible]
Degrees [illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

TABLE IV

DATA FOR THE MAXIMUM-LIFT CONDITION

[illegible]

α	C_L	C_D	$C_{m_{c/4}}$	[illegible]	[illegible]	[illegible]
Degrees [illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

TABLE V

DATA FOR THE MAXIMUM-LIFT CONDITION

[illegible]

α	C_L	C_D	$C_{m_{c/4}}$	[illegible]	[illegible]	[illegible]
Degrees [illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

was 30° for the 20 percent *c* flap and 40° for the two larger flaps.

Variation of C_{Lmax} with flap size is shown in figure 5. The maximum lift coefficient increases approximately in proportion to flap size if the area of only the original wing is considered. This is a reasonably satisfactory basis for comparison of the landing speeds of an airplane with various sizes of flap if a constant maximum speed is maintained. If the maximum lift force that a wing will give at a certain air speed per unit of structural weight is taken as a criterion, it is reasonable to compare the various sizes of flap on the basis of total (wing-and-flap) area. On this basis there is clearly little to be gained by using flaps larger than 30 percent *c*.

Lift, drag, and pitching-moment data for the wing with each of the three flap sizes, with the flap at the setting for maximum lift, are given in figure 6 and in tables III, IV, and V. Coefficients are based on the area and/or chord of the wing alone. The data for the plain wing were obtained with the 20 percent chord flap fully retracted into its well. (See table II.) It is evident that an airplane having a flap of this type would have a much larger range of center-of-pressure travel between various flying conditions than would one with a plain wing. It appears, then, that in a normal type of 2-spar wing the effect of adding a Fowler flap would be to leave the front-spar design load the same as for the wing without a flap but to increase considerably the design loads on the rear spar. If the speed at which the airplane may be flown with flap extended is limited to a value reasonably in excess of its landing speed, it appears likely that the loads with flap extended would be reduced to the same magnitude as the largest loads with flap retracted, with flap sizes not in excess of 30 percent *c*. On this basis it appears that a wing with a Fowler flap as wide as 30 percent *c* could be constructed in which there would be no increase in the weight of the wing structure proper, the only additional weight being due to the flap and its support from the spars.

Take-off condition.—Investigation of wing-flap combinations to determine the flap arrangement most favorable for take-off must involve consideration of performance parameters of the airplane in question as well as of the aerodynamic effects of the lifting surfaces. Concurrently with the tests, a series of take-off computations was made with the purpose of developing a "take-off criterion" for wings based on aerodynamic characteristics and depending on airplane design factors to the minimum extent possible. The application of such a criterion to the data would then serve to isolate the optimum flap arrangement for take-off. The development of the criterion, and associated data, are presented in an appendix to this report.

As the tests and computations progressed, it was found that some general considerations would serve to isolate the optimum arrangement, without recourse to a rigorous criterion. The computations indicated that normal transport airplanes should take off at a lift coefficient greater than 70 percent of the maximum available to achieve the shortest run to clear an obstacle. They also indicated that the principal aerodynamic characteristics affecting take-off, high lift available, and high L/D at the high lift are of nearly equal importance.

The wind-tunnel data, plotted as polar curves, are presented in figures 7 to 10 for the 0.30 *c* flap and in figures 11 to 15 for the 0.20 *c* flap. Comparison of these curves on the basis of the considerations previously stated indicated the flap position 0.025 *c* directly below the trailing edge of the wing, with an angle of 30°, to be the optimum take-off arrangement for both flaps. At this setting each flap has as high ratios of L/D throughout the high-lift region as any other setting tested, within the limits of accuracy of the tests, and has a higher maximum lift coefficient than any other setting having as high ratios of L/D. The 40° setting of the 0.30 *c* flap, at this same position, gives a higher maximum lift and lower ratio of L/D than the 30° angle, the percentage difference in L/D being greater than that in maximum lift. Computations (see appendix) verify the conclusion based on the general considerations, that the 30° angle is better with this flap.

Lift, drag, and pitching-moment data for the wing with each of three sizes of flap, with the flap at the optimum setting for take-off, are given in figure 16 and in tables III, VI, and VII. The choice of the position 0.025 *c* below the wing trailing edge, with a 25° angle, as optimum for the 40 percent *c* flap is based on the relation between optimum take-off setting and that for maximum lift of the 20 percent *c* and 30 percent *c* flaps. Although data for the 40 percent *c* flap are not sufficient for a rigorous selection, comparisons of data that are available (reference 2) indicate the choice to be sufficiently near the optimum for practical purposes.

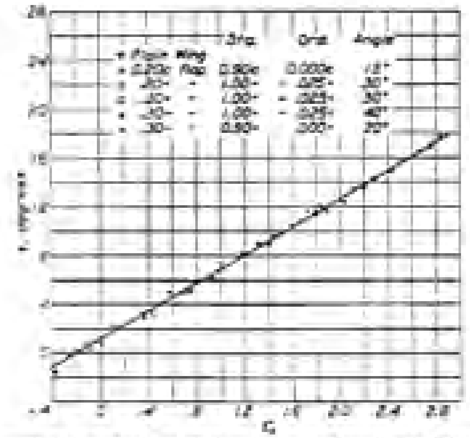
Partial retraction of flap.—Lift, drag, and pitching-moment data for the wing with the 20 percent *c*, 30 percent *c*, and 40 percent *c* flaps in a partially retracted position are shown in figure 17 and in tables VIII to XI. The settings were chosen by assuming the flaps to move along an arc from the setting for maximum lift or optimum take-off to the fully retracted position. The flap hinges crossed the wing chord line at the 90 percent *c* station, and the angles at this position were 15° for the 20 percent *c* flap, 20° for the 30 percent *c* flap, and 20° and 30° for the 40 percent *c* flap. Comparison of the characteristics at this setting with those at the maximum-lift setting shows that the change of characteristics is in the same direction and of the same order of magnitude as the change of flap setting.

Flap loads.—Curves of normal and longitudinal force coefficients, hinge moments, and center-of-pressure locations of the 20 percent *c*, 30 percent *c*, and 40 percent *c* flaps in the maximum lift, optimum takeoff, and partly retracted settings are shown in figures 18 to 23. The corresponding data appear in tables III to XI. From the magnitude of the load carried by the flap at high lift coefficients of the combination, it is evident that the flap carries nearly 1½ times its proportionate share of the total load. It appears that this type of flap may be regarded as a separate wing, operating in an air stream whose combined velocity and curvature increase considerably the load it carries as compared with the load it would experience in the free air stream. Comparison of load data for a split flap (reference 4) and a Fowler flap clearly shows the fundamental difference in the action of the two flaps. At high lifts, the split flap carries almost no lift and offers large drag; whereas the Fowler carries a large proportion of the total lift, but with less drag.

Although this condition is favorable to airplane performance, it implies a large range of center-of-pressure positions for the complete flight range, with consequent disadvantages in longitudinal-stability characteristics and possibly also in structure. In connection with structural considerations it is interesting to note that a progressive reduction in flap loads occurs with increasing flap size if the maximum angle is kept below 30°.

At flap settings giving high maximum lift coefficients, the center of pressure of the flap itself has little travel throughout most of the angle-of-attack range and is generally nearer the leading edge than it would be on an airfoil in a free air stream. As the flap angle is reduced below 30°, however, the center of pressure moves rapidly backward.

Downwash.—Some representative data from the downwash measurements are shown in figures 24 and 25. Angle of downwash as a function of lift coefficient is shown for two positions behind the wing, with data for the plain wing and for the same flap settings as were used in the flap load tests plotted on each curve. Only small consistent deviations from the mean curve, within the limits of test accuracy, were found for the variety of settings tested. It appears, then, that the addition of a



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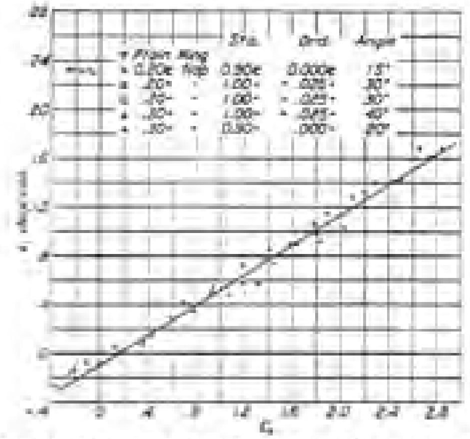


[illegible]

Fowler flap has no appreciable effect on the basic relation between lift, span, and downwash at reasonable distances behind the wing.

The foregoing conclusion is subject to some question owing to the doubtful nature of the jet-boundary effect on downwash in the 7- by 10-foot tunnel. The corrections in this particular case differ considerably from the theoretical corrections, probably on account of the combined effect of static-pressure gradient in the jet and spillage of air over the unflared lip of the exit cone. Different corrections for different positions of the reference point in the air stream might produce greater consistent differences in downwash between the plain wing and flap extended conditions than are indicated by these tests, though this effect would be small unless the variation of the corrections with position is greater than seems likely.

Although the extensive investigation required to establish the corrections might produce results of academic interest, certain effects of combining a variable-lift wing with an airplane fuselage would render the results of small technical value. Since a large difference in angle of attack occurs at the same value of C_L with different settings of the Fowler flap, a large variation of fuselage attitude and lift at a given wing lift coefficient results from changing flap settings. Thus, at a given over-all lift coefficient of the airplane, the lift coefficient and downwash of the wing may be expected to change with flap setting. The use of partial-span flaps produces an effective reduction of span as the flap is extended, causing an additional change of downwash at constant lift coefficient with changing flap setting. It appears that problems involving downwash of variable-lift wings are more susceptible of solution by measurement on the actual design in question, rather than by a fundamental wind-tunnel investigation.

CONCLUSIONS

1. The maximum lift coefficients, based on area of wing alone, found for the three sizes of flap tested were: For the 20 percent *c* flap, 2.45; for the 30 percent *c* flap, 2.85; and for the 40 percent *c* flap, 3.17. The maximum lift coefficient for the wing with flap retracted was 1.31.

2. The location of the flap leading edge for maximum lift was found to be the same in all cases, the center of the leading-edge arc being 2.5 percent *c* directly below the trailing edge of the main wing. The flap angles for maximum lift were 30°, 40°, and 40° for the 20 percent *c*, 30 percent *c*, and 40 percent *c* flaps, respectively.

3. The 20 percent *c* and 30 percent *c* flaps were found to give the characteristics most favorable to take-off with the same leading-edge location as for maximum lift. The optimum angle was 30° in both cases.

4. The maximum normal-force and longitudinal-force coefficients of the 40 percent *c* flap, based on flap area, were 2.89 and −1.25; those for the 30 percent *c* flap were 3.06 and −1.54; and those for the 20 percent *c* flap were 2.80 and −1.20. Center-of-pressure locations corresponding to these coefficients were in each case approximately at the 20 percent *c* flap chord points.

5. At positions normally occupied by the tail surfaces the relation between lift coefficient and downwash angle appears from the present tests to be the same for a wing with or without a full-span Fowler flap.

LANGLEY MEMORIAL AERONAUTICAL LABORATORY,
NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS,
LANGLEY FIELD, VA., April, 26, 1935.

Document 3-12(h), Harlan D. Fowler, "Aerodynamic Characteristics of the Fowler Wing,"* Aero Digest *29 (September 1936): 46-50.

AERODYNAMIC CHARACTERISTICS THE FOWLER WING
HARLAN D. FOWLER

Airfoil design has practically reached its maximum, comprehensive and systematic research by the NACA resulting in such several new highly efficient sections that possibility of further improvement is remote and of comparatively small value, although costly to discover.

Attempts have been made to so design an airfoil that it would develop especially high L/D ratio, better minimum drag at cruising speed, etc. Earliest efforts were devoted to increase maximum lift (to provide safer landing speed), and to increase drag so as to decrease length of the run when landing.

Research conducted by NACA for the Bureau of Aeronautics indicated that an airplane equipped with a Fowler variable area wing could reduce landing speed, decrease landing and take-off runs, and that there would also be indications of improved climb with the Fowler partially extended. This device was first tried in 1927 on a Canuck JN; in 1929 on a Pitcairn; in 1934 at the International Competition at Warsaw, Poland, where a similar device was used by Fieseler and Messerschmitt; in 1935 on a Fairchild F-22 by the U. S. Navy and NACA; and more recently on the Martin Bomber.

Combined use of variable area, camber, angle of attack and of the boundary layer, requires only an auxiliary airfoil that is extended or retracted as the occasion demands. The principle is termed a *fowler* to distinguish it from other flap devices. Individual aerodynamic properties of each of these principles is not sufficiently great to justify its use alone, as for example, the percent distribution of the maximum lift coefficient as contributed by the several features given in Table 1.

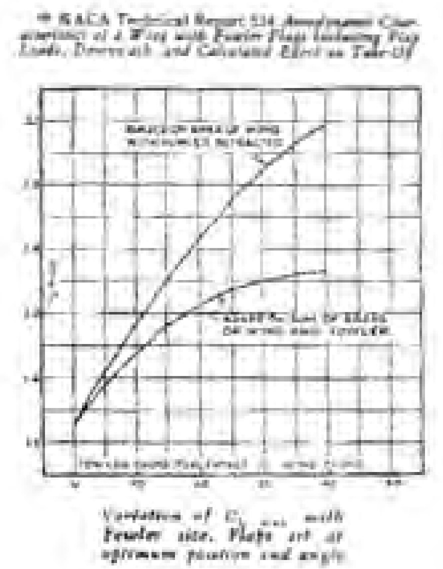
Based on area alone C_{Lmax} should vary directly with the fowler chord with it fully extended. However, data available show that with a 20% chord fowler set at -5, fully extended, the increase is slightly higher or about 25% more than for the normal wing. By virtue of the recess cutout under the trailing edge of the normal wing, lift is again increased slightly varying with fowler chord.

The major individual increase in maximum lift results from the gap, which appears to increase nearly directly with the chord of the fowler. Variable area is the next most effective factor, with camber a close third. However (by the principle involved), the combination of area and camber actually contributes a greater increase than the gap, and it is apparent that all three factors have a powerful influence in contributing to the maximum lift, and increase with enlargement of the flap chord.

PARTIAL SPAN AND LATERAL CONTROL

Distribution of maximum lift along the wingspan has been investigated and checked with actual performance data in cases where the fowler covered a portion of the wing. The loss of lift is greatest near the fuselage and least at the wing tip. On a tapered wing it is hardly worthwhile to extend the fowler beyond 80% of the span as the remaining 20% at the tip represents a loss of about 4% from the full span arrangement. However, if the inner end of the auxiliary wing is not extended over the center 20% span, there will be a loss of about 14% from the maximum.

Considering the small possible gain by extending it to the tip, the question of providing for lateral control means becomes involved. The author maintains that due to the powerful lifting capacity of the device, there is no need to sacrifice good lateral control to attain the last ounce of lift. In this respect, conventional ailerons have been used and the lateral control obtained has been satisfactory, even with the fowler fully extended. Experiments have revealed that rolling moments are substantially higher with it extended than when closed, the ideal condition. In these tests, ailerons occupied from 30 to 40% of the semi-span, the smaller span being adequately effective.



Comprehensive research on fowlers of 20, 30 and 40% chord shows that if the maximum angle is kept at or below 30 there is a progressive reduction in flap loads with increasing chord, provided advantage is taken of the corresponding decrease in stall speed. Furthermore, if the high speed with the fully extended wing is kept below twice stalling speed, it is possible to add fowlers having a chord up to 30% to a wing whose spars have already been designed for its normal critical loads (except for wings using an airfoil of constant center of pressure movement) that will be sufficiently strong to take the load from the auxiliary wing without appreciably increasing the main wing weight, particularly the rear spar. Because of the necessity of a recess under the trailing edge of the main wing, ribs are consequently shallower and must be strengthened, although not necessarily with considerable increase in weight, although there will be a weight increase by adding the fowlers and controls.

When the fowler is in its fully extended position, longitudinal force assumes a negative value—that is the force tends to close it. At some intermediate point this force becomes zero then tends towards the positive. The control system is therefore designed not to extend the fowlers but to prevent them from closing. With this arrangement and using an automatic hydraulic pressure relief valve system, should the load become too high, the fowlers will tend to move towards the closed position, thus preventing undue strain on the main wing structure. However, the hydraulic device will hold it at any desired position.

Throughout the flying angles of attack, the center of pressure on the fowler wing is uniformly forward when in the extended position, at about 30% of its chord. Since the main spar of the flap is located at about 20% chord there is no tendency for flutter since the pressure is constantly aft of the point of support. In the case of a twin-engine installation with the propellers directly in front of it, it was found that the extended fowler minimized turbulence, due perhaps to the gap action on the boundary layer over the wing.

CONTROL MECHANISM

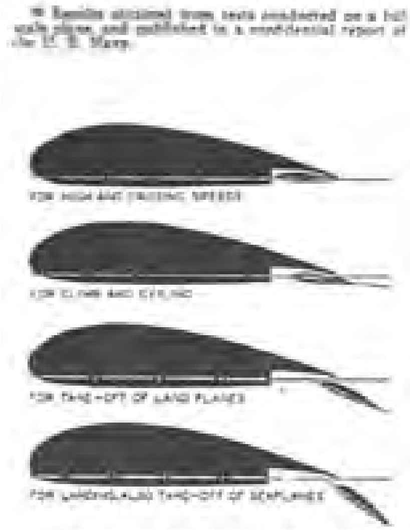
There are two methods of operating the fowlers. The original method consisted of fixed tracks, properly curved, attached to the rear spar, with a trolley which is secured to the auxiliary wing, running along the track. Back and forth motion was accomplished either by a continuous cable or by push-pull rods. The second method is the rotating or swinging arm which is pivoted to the rear spar with the fowler suspended at its outer end. The arm swings back towards the trailing edge, thus extending the fowler and automatically increasing its angle, the latter being accomplished by a universal joint. Both systems have been tested and each has merits which the other does not possess. The swinging arm presents a perfectly smooth wing when the fowler is closed. The fixed track has a small portion of its rear end exposed at all times (without causing any appreciable increase in drag) and is less costly to build and lighter in weight than the other method.

Speed of extension or retraction is obtained in several ways. Using manual crank-operated control, movement is fro m 2.5 to 3 inches to one complete turn of the crank handle. By hydraulic control in one case, movement was accomplished in 15 seconds. There is something to be said about the speed at which fowlers should be operated. They should not be operated too quickly since it gives neither the plane nor the pilot a chance to adjust for the varying conditions. If there is too long an interval, there will be a decided negative influence on landing and take-off, so that a happy medium on speed of operation is important.

Hydraulic and electrical systems are preferable for large planes, the first lend-ing itself to automatic retraction, if the plane's speed becomes excessive. Both systems are dependable and flexible.

PARTIALLY EXTENDED CHARACTERISTICS

Inherent with the extendable feature of the device is the transition or partially extended position. The fowler can be stopped at any intermediate position and several such positions contribute improvement to performance. The accompanying diagram illustrates several positions at which it may be set to give particularly desirable performances, the positions shown being spaced in four locations, the exact positions for the given type of performance varying with airfoil and airplane design.



Performance characteristics with various positions of the Fowler wing

LONGITUDINAL STABILITY AND DOWNWASH

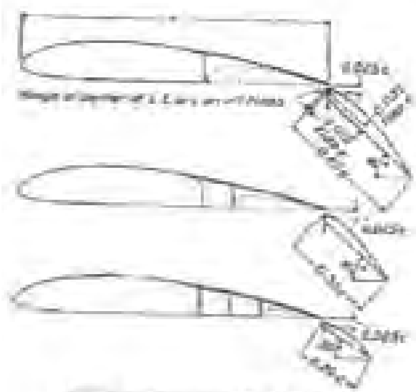
In the normal wing condition, longitudinal stability is as usually obtained on well-designed planes.

With the fowler device, as it moves outward and to about 2/3 extended position. the change in balance is barely perceptible. In the full out position there is need of more upward elevator movement for landing. Also, due to the increase in length of the combined wing and fowler chord, tail surfaces should be at a slightly greater distance from the center of gravity, or the horizontal tail surfaces made larger than normal, the area of the elevator being at least equal to one-half the total horizontal tail area. This increase is to be expected because at the very high lift coefficients obtainable with the fully extended fowler, the downwash angle is considerably higher. Combined with the low maximum lift coefficient of tail surfaces, this causes the tail to stall at certain speeds. For this condition large elevator movements are preferable to stabilizer adjustment, especially on large planes. Also, there is no tendency to nose over at the take-off or on landing.

Although not completely affirmed, extensive investigations on downwash angles behind a wing using the auxiliary wing indicate that its addition has no appreciable effect on the basic relation between lift, span and downwash at reasonable distances behind the wing. The downwash angle is a direct function of the lift coefficient, but it appears that some correction must be made for the shifting of the angle at zero lift as between the normal wing and the extended fowler. In other words, the angle of attack for all fowler positions and chord sizes is referred to the basic chord line of the normal wing. As it is extended or increased in chord size, the angle at zero lift becomes more and more negative. Since the setting of the tail surfaces is referred to the zero lift chord of the main wing, this variation must be considered.

LANDING DISTANCES AND RATE OF DESCENT

The maximum lift coefficient obtained by a fowler wing is 3.17, referred to the normal wing area, by using a 40% chord over the entire span. By means of an auxiliary nose airfoil this can be increased to 3.62. At present, for most designs in which the fowlers are used, the maximum lift coefficient is from 2.00 to 2.50, the lower value being attained by using a 20% chord and the latter by using a 30% chord both over 60 to 70% of the wingspan. A 40% chord has been used in one wing.



Wing with various Fowler flaps (From NACA report No. 534)

The accompanying chart (from NACA Technical Report 534) shows the variation of the maximum lift coefficient with fowlers of various percent chords in terms of the basic wing chord. As a high lift device is generally used to increase the maximum speed of a plane keeping a given stall speed, it is proper to consider lift in terms of normal wing area, with fowler retracted.

As an illustration of what has been done in full flight tests, some results obtained at Langley Field by NACA are given in Table 2, which shows minimum rate of descent with the fowler fully extended was about 8.7 ft./ compared to 9 ft./sec.

with the wing closed. Data were obtained on the F-22, results being average of two landings. F-22 has a Fairchild fuselage mounting a specially designed variable area wing. The plane was powered by a 95 hp engine. Gross weight was about 1600 lbs., wingspan 31 ft., normal chord 4 ft. 4 in., normal area 132 ft', fowler area 27.3 ft', total area 159.3 ft.' A 30% chord fowler over 71% of the span was used with conventional ailerons whose area was about 7.2% of the normal wing area. Wing load based on normal area was 12.1 lbs./ft ^{2}, and power loading 16.8 lbs./hp.

Running along the ground for take-off or landing, the proximity of the earth causes a reduction in the angle of attack due to induced lift. Also, the higher the lift, the greater the induced angle of attack of the wing (as this varies directly with the lift), thus allowing a shorter landing gear. The ground angle being smaller, it is possible to use a true stream-line fuselage, with a resultant saving in parasite drag.

TAKE-OFF OF LANDPLANES

Take-off distances for F-22 are in Table 3 which shows a reduction in total distance of 22%.

Theoretical study of the effect of high lift on take-off, with varying power and loading per square foot, showed the fowler wing should be able to reduce total take-off distance by from 30 to 45%, including the clearance of a 50-ft. obstacle.

It should be explained that the fowler is able to contribute to high reduction in take-off run due to the combination of high lift with low drag. The first contributes to low take-off speed, thus favoring propeller thrust. With low drag, the excess thrust is considerably improved.

While the fowler has lower drag and thus less braking effect on landing, the high lift reduces the landing speed accomplishing results similar to the split flap. However, it should be pointed out that due to the high LID ratio with the fowler wing it is necessary for the pilot to use only ordinary flight technique. This is particularly desirable in forced landings.

CLIMB AND CEILING

Wind tunnel research on a complete model indicates that with the fowler extended about 25%, L/D ratio is higher and occurs at a larger lift coefficient. When applied to a given design, service ceiling can be improved. With planes supercharged to high altitudes, the lift coefficient is larger, and with smaller drag the ceiling is bettered. This characteristic will not be detectable near sea level, as compared with the normal wing, but it should appear at altitudes.

Flight tests on F-22 at 2000 ft. indicated that, with the fowler extended 1/3, rate of climb was as good as with it closed, but the angle of climb was higher.

On another plane it was found that at 9000 ft. rate of climb was appreciably higher than for the normal wing, and at low speed the gain was about 10%.

Also, test results seem to indicate the possibility that when one of two engines fails that, by partially extending the fowler, both rate of climb and ceiling can be improved.

SEAPLANE TAKE-OFF

A valuable field of application of the fowler wing is for flying boats, especially those having long range and large payloads. While the controllable-pitch propeller has materially reduced take-off problems for such ships, its combined use with the flap offers further improvements. The problem of a large flying boat is to carry a large fuel load plus passengers, mail and equipment, but this usually results in increasing take-off speed. With the higher take-off speed, water resistance of the hull may become large enough to actually prevent a take-off. With the high lift of the fowler wing, take-off speed is reduced and propeller thrust increased, providing greater excess thrust for acceleration. Even at the full out position, L/D of the wing is high enough to cause a substantial reduction in the total water resistance plus air drag.

The most effective method of using the fowler is to start extending it about 10% below take-off speed, so that it becomes fully extended only at the instant of take-off. Thus the plane starts its run with the normal wing, gets over the hump and speeds on until at some predetermined speed the fowler begins to extend.

Table 4 calculates values for a seaplane of 15,000 lbs. (gross), 1000 ft. normal area and 1000 hp using a 20% chord fowler occupying 62% of the span. In this case the time interval between commencing to extend the fowler to the fully extended position, was about 6 seconds, illustrating the importance of not permitting too long a time for the device to completely open, although a little longer time could be permitted without materially affecting take-off characteristics.

It should also be possible to set the wing in relation to the hull at a lower angle, thus reducing hull drag at cruising speed. Another advantage is that the trim angle of the hull can be maintained nearly constant at get-away because the fowler will pull it off.

The fowler wing can also be utilized to increase fuel load with the same wing area, or alternatively, the same fuel capacity can be kept and the wing area reduced.

Document 3-13

Standard Steel Propeller Company to Ryan Airlines, undated (ca. April 1927), in Charles A. Lindbergh, *The Spirit of St. Louis* (New York: Scribner, 1953; reprint, St. Paul: Minnesota Historical Society Press, 1993), p. 116.

As with all other aspects of the design of the *Spirit of St. Louis*, Charles A. Lindbergh carefully calculated the role of his airplane's propeller before setting off on his historic transatlantic flight from New York to Paris. In the spring of 1927, the Standard Steel Propeller Company of Pittsburgh produced four nine-foot-diameter dural (aluminum alloy) ground-adjustable propellers for Lindbergh's Ryan-built airplane; the company built them according to "specification No. 1519" and sent them on to Ryan Airlines in San Diego. Inspected by Standard Steel's assembly foreman and inspector, Alexander F. Manella, the propellers consisted of twelve pieces designed for use with the Wright Whirlwind J-5C radial engine. Lindbergh and Donald A. Hall, the chief engineer for Ryan Airlines, asked Standard Steel to recommend a specific blade pitch that would help them achieve the range needed to reach Paris. The word received back came in the telegram reproduced below.

What Standard Steel did was simply calculate which propeller blade angle was best, for the propeller itself was essentially "off the shelf." Acting on Standard Steel's recommendation and influenced by urgent time constraints, Lindbergh accepted the compromise setting that favored a setting optimum for cruising conditions. This was a big gamble because it was a pitch setting for the propeller that provided minimum efficiency for takeoff. Still, the metal ground-adjustable propeller offered him a flexibility that wooden and metal fixed-pitch propellers could not. With its propeller blade set at 16.25°, the Wright J-5C engine produced 190 hp at 1,545 revolutions per minute; this generated a static thrust of 700 lbs. The maximum airscrew efficiency of the *Spirit's* propeller rated out at 74 percent.

Roosevelt Field on Long Island, from which Lindbergh lifted off for Paris, was over a mile long and the only suitable place for heavy takeoffs in the New York area. But obstacles surrounded the field, telephone wires and low tree-covered hills, dangers of which Lindbergh was well aware. As he expected, the *Spirit of St. Louis* required almost every inch of the field to lift its 5,000-lb load (mostly fuel) into the air. Years later, Lindbergh recalled his approach to the problem: "I decided to 'feel' the plane off the ground just as I had often done in underpowered planes while barnstorming. If I felt I was not going to get off the ground, my plan was to simply cut the throttle" (quoted in Cassagneres, *Spirit of Ryan*, p. 54). Perhaps more than any other element of risk in an otherwise carefully planned flight, Lindbergh's takeoff threatened failure, even disaster. Fortunately, as it happened, the straining *Spirit* lifted off the ground about 1,000 yards in front of the telephone wires at the end of the field—although the angle of the newsreel pictures made it look a lot closer. Thus, one might call Lindbergh's takeoff the most dramatic and the most decisive event of his entire flight, for as he wrote in his book *The Spirit of St. Louis* (Scribner's Sons, 1953): "My propeller is set for cruising, not for takeoff" (p. 183). This passage perhaps illustrates other innovations as well, especially flaps.

Document 3-13, Standard Steel Propeller Company to Ryan Airlines, undated (ca. April 1927), in Charles A. Lindbergh,* The Spirit of St. Louis *(New York: Scribner, 1953; reprint, St. Paul: Minnesota Historical Society Press, 1993), p. 116.

WESTERN UNION

HOLMSTEAD [SIC]
PENN.

RYAN AIRLINES
SAN DIEGO, CALIF.

15.5 DEGREES SETTING PROBABLY NECESSARY ON YOUR MONOPLANE TO GET TAKEOFF WITH HEAVY LOAD FUEL ECONOMY WILL BE IMPROVED ON HIGHER PITCH SETTING STOP IF TAKEOFF IS SATISFACTORY WITH 15.5 SETTING SUGGEST TRY 16.5 AS THIS WILL IMPROVE FUEL ECONOMY.

STANDARD STEEL PROP. CO.

Document 3-14

C. B. Allen, "Hamilton Standard Wins Collier Trophy for Controllable Pitch Propeller," *The Bee-Hive* 8 (June 1934): 1-2.

In 1933, the National Aeronautic Association awarded its Collier for the year's greatest achievement in American aviation to Hamilton Standard Propeller Company, with particular credit to Frank W. Caldwell, chief engineer, for development of a controllable-pitch propeller that was then entering general use. Caldwell's design controlled its blades via bevel gears that were turned by a drumcam in the propeller hub. Hamilton Standard later went on to refine the design by adding a "constant speed unit" or CSU. This gadget controlled the propeller automatically to permit the desired engine speed. The aircraft industry continued to make improvements. By late in World War II, the controllable-pitch propeller added reversible-pitch settings that further slowed an aircraft for landing. Even as the turbojet revolution gained momentum, aeronautical engineers continued to perfect propeller applications for reciprocating engines. Airplanes such as the Boeing B-29, Boeing B-50, Lockheed Constellation, and Lockheed C-130 Hercules all employed large and sophisticated controllable-pitch propellers of three and four blade sections. Without advanced forms of this type of propeller, piston-engine airliners such as the Douglas DC-6 and DC-7 of the late 1950s and early 1960s could not have achieved effective performance. Without question, the variable-pitch and controllable-pitch propellers made essential contributions to the reinvention of the airplane inherent to the airplane design revolution of the interwar period.

Document 3-14, C. B. Allen, "Hamilton Standard Wins Collier Trophy for Controllable Pitch Propeller,"* The Bee-Hive *8 (June 1934): 1-2.

PRESENTATION last month by President Franklin D. Roosevelt to the Hamilton Standard Propeller Company, of East Hartford. Conn., through its chief engineer, Frank Walker Caldwell, of American aviation's most coveted annual award-the Collier Trophy for 1933-has met with wide approval from the industry because of the practical benefits it has enjoyed from the first practical "gearshift of the air."

More than 500 controllable pitch propellers of the type for whose development the award was made are now in everyday service, and their users will have no complaint with the National Aeronautic Association for selecting this device as "the greatest achievement in aviation in America, the value of which has been demonstrated by actual use during the preceding year."

ELIMINATES FAULTS OF OLD MODELS

Essentially the controllable pitch propeller supplies an airplane with a "low gear" for takeoff and climb purposes and a "high gear" for economical cruising. It eliminates the chief faults of the fixed pitch propeller--the fact that its blades "bite" so much air as the plane moves slowly over the ground the engine cannot "turn up" its full rated horsepower or "race" impotently once the ship is in level flight, if the blades are set at a flat angle designed to give a reasonably quick take-off, because they do not get a sufficient grip on the medium through which the whole ship is moving at so swift a pace.

Obviously, such a development always has been highly desirable in aviation, but without it the high-speed transport planes that have made their appearance on America's air lines during the last year would not have been possible. No fixed pitch propeller could be expected to function with any sort of efficiency over the wide range that lies between the take-off and top speeds of these craft, and such devices as the wing-flap "air brake" which enable 200-mile-an-hour transports to land and take off at fifty miles an hour would be pretty much in vain save for the controllable pitch propeller.

"GEARSHIFT OF THE AIR"

Also, the safety features of the multi-motored airplane, especially the two-engine variety, would be greatly impaired without the "gearshift of the air" for the reason that, when one power plant fails, the plane inevitably slows down in the air and only by putting the remaining sound engine in "low gear," where it can "turn up" its full rated horsepower (and in some cases deliver for short periods more power than its normal rating), is it possible to continue flight and maintain altitude--particularly when the failure occurs just after leaving the ground or during flight over high-altitude terrain.

United Air Lines alone has more than 100 Hamilton Standard controllable pitch, propellers in service on the Boeing monoplanes it is operating over its transcontinental and other routes. Some of these have flown 1,500 hours and more--the equivalent of 225,000 miles flying-without failure of any sort.

The device is simple and rugged, a piston, operated by oil pressure from the engine, twisting the blades into "low gear" when the pilot pulls a lever in the cockpit. Once he is in the air and has gained the level at which he wishes to fly, he releases the lever and the blades are automatically pulled into "high" by two counter-weights, which the powerful, outward-pushing oil piston heretofore has thrust and held back.

YEARS OF DEVELOPMENT WORK

Mr. Caldwell designed and patented the controllable pitch propeller in 1928, but years of development work and $200,000 in cash were required before all of its mechanical difficulties had been conquered and its creator was satisfied to put it into production a year ago last January.

There have been three eras in the development of the airplane propellers—the age of wood, which lasted through the World War and seven years beyond, the metal propeller (both with fixed blades and blades that could be adjusted to any desired angle before flight, but not altered in the air), which held sway some nine years and won the Collier Trophy for Dr. S. Albert Reed in 1925, and now the "gearshift of the air" that may be changed at the pilot's will in mid-air.

The 1933 committee of award, appointed by President Hiram Bingham of the National Aeronautic Association, consisted of Rear Admiral Emory S. Land, U.S. Navy: Colonel Edgar S. Gorrell, Air Corps Reserve, and Chief of Staff of the Air Corps in France during the war: Colonel Clarence M. Young, former Assistant Secretary of Commerce for Aeronautics: General Frank Hitchcock, who, as Postmaster General, arranged for the first flight of air mail in America in 1909, and Earl N. Finley, Editor of U.S. Air Services.

NOTABLES AT PRESENTATION

Among the guests who witnessed the presentation were two previous holders: Harold Piteairn of Philadelphia and Glenn L. Martin of Baltimore. Others present included President Hiram Bingham of the National Aeronautic Association; Eugene L. Vidal, Director of Aeronautics of the Department of Commerce; J. Carroll Cone, Assistant Director of Aeronautics; Dr. George W. Lewis, Director of Research of the National Advisory Committee for Aeronautics; Don L. Brown, President of the Pratt & Whitney Aircraft Co.; Ray Cooper, General Manager of the National Aeronautic Association; William R. Enyart, Secretary of the Contest Committee of the N. A. A.; Joseph Edgerton and Bob Ball, newspaper men.

Document 3-15(a-e)

(a) Smith J. DeFrance, "The Aerodynamic Effect of a Retractable Landing Gear," NACA *Technical Note* 456 (Washington, 1933).

(b) Hal L. Hibbard, "Problems in Fast Air-Transport Design," *Mechanical Engineering* 55 (October 1933): 611-617.

(c) "Preliminary Study of Retractable Landing Gears for High and Low Wing Monoplanes," *Air Corps Information Circular* 7 (18 February 1933): 1-9.

(d) Richard M. Mock, "Retractable Landing Gears," *Aviation* 32 (February 1933): 33-37.

(e) Roy G. Miller, letter to editor, *Aviation*, "Retractable Landing Gears," *Aviation* 32 (April 1933), with response from Richard M. Mock: 130-131.

One of the most significant "shelf items" required to reinvent the airplane was retractable landing gear; as essential as it was, it also proved one of the most evasive. The earliest known concept for a retracting mechanism dates from November 1911 and to an obscure aeronautical inventor by the name of F. McCarroll who received a U.S. patent for his invention in November 1915. The first airplane to appear with such an undercarriage was the Dayton-Wright RB-1 of 1920, which also possessed the world's first variable-camber wing. In 1922, the idea of retracting gear got a major shot in the arm as part of Louis Breguet's agenda for aircraft streamlining (see Document 3-4). As discussed in the text, the Verville-Sperry R-3 racer of 1923 brought a lot of attention to retractable landing gear, which it incorporated, when it flew in the Pulitzer Trophy Race of 1923 in St. Louis, winning the race in Dayton the following year.

Systematic testing of the concept did not really begin until 1927, when research engineers at the NACA's laboratory at Langley Field began looking into the relationship between fixed landing gear and aerodynamic drag. The NACA tests, conducted in Langley's PRT on an army-owned Sperry Messenger airplane with its outer wing panels removed, also represented the world's first wind tunnel tests of a full-scale aircraft, not just a scale model or individual components. These tests in the PRT revealed that the resistance of the protruding fixed landing gear amounted to an astounding 40 percent of the airplane's total drag. The tests defined the "real magnitude of the landing gear drag penalty" and provided precisely the kind of empirical data needed to encourage the American aviation industry to devote more resources to the design of landing gear. This proved to be a critical step in the process of reinventing the airplane, with the NACA becoming more and more involved in evaluating the value of retractable landing gear for industry.

The following string of five documents demonstrates the growing commitment to retractable gear. The first concerns NACA tests of the Lockheed Altair monoplane in Langley's Full-Scale Tunnel for the purpose of learning the advantages and disadvantages of the new type of gear and specifically whether or not the gear contributed to the aircraft's excessive takeoff run. NACA researcher Smith J. DeFrance (future director of the NACA's Ames Aeronautical Laboratory in California, which came to life in 1941) found that the retractable landing gear improved performance generally in flight but had negligible effects on takeoff and landing. DeFrance and other NACA engineers understood that, before the industry could wholeheartedly accept retractable landing gear, precise knowledge of the flow of air caused by landing gear when it was extended had to be in hand. When the gear was retracted, the aerodynamics were better, but when the gear was extended, the aerodynamics then became problematic. Until the gear performed more effectively overall, it was difficult to overcome the inertia favoring the certainties of traditional fixed gear, such as that still being put even on innovative aircraft like the Northrop Alpha.

The second document in the string reflects the type of thinking that opposed the use of retractable landing gear. Similar to the objections to using the NACA cowling, many individuals in industry did not feel that retractable landing gear was the most practical solution. One industry representative who thought this way was Hal L. Hibbard, assistant chief engineer of the Lockheed Aircraft Corporation. In 1933, Hibbard presented the following article at the joint meeting of the Society of Automotive Engineers and the American Society of Mechanical Engineers held in conjunction with the 1933 National Air Races in Los Angeles in July. Examining all the components that made up a complete airplane, Hibbard addressed the aerodynamic differences between fixed and retractable landing gear, differences that he considered to be "greatly exaggerated." The major source of drag in landing gear, according to Hibbard, was the aerodynamic interference caused by parts of the landing gear and its relationship with the wing and fuselage. If designers would only carefully streamline their fixed landing gear, then the difference between fixed and retractable gears would be negligible; in fact, the overall advantage could even go to fixed gear. At one point in his presentation, Hibbard referred to "careful wind tunnel

tests," which backed up his conclusion. What he was no doubt referring to was a test program conducted at the GALCIT wind tunnel at Caltech on a Northrop Alpha airplane employing streamline coverings known as "spats" or "pants." These were teardrop-shaped fairings around the wheels of a fixed landing gear and enclosing the landing strut and wheels, which helped reduce drag. Spats had become popular with the Lockheed Vega and most fixed landing gear following the Vega used them.

Despite the dramatic benefits that retractable landing gear offered, widespread development and adoption was slow. As late as 1933, as we see in the third document in this string, the U.S. Army Air Corps had no clear idea which forms of retractable landing gear were feasible. The authors of the Air Corps information circular continued to stress the advantages of fixed gear with streamlined wheel fairings even for high-performance aircraft over those possible with retractable gear. The fourth article below is a February 1933 trade journal article that discussed the myriad of variables related to retractable landing gear design. Its author reported on industry's attitudes toward the mechanical difficulties it was facing when trying to integrate retractable gear into aircraft design. The string closes with a letter to the editor endorsing the tangible benefits that retractable landing gear would provide to aircraft operators.

The superiority of retractable landing gear grew more and more obvious as synergistic advances related to power plants, aerodynamics, structures, and stability and control came together through the 1930s, boosting the speed and performance of aircraft to unprecedented levels. By the start of World War II, the advantages of retractable landing gear were clear, and the U.S. aircraft industry had turned retractable landing gear into its standard.

Document 3-15(a), Smith J. DeFrance, "The Aerodynamic Effect of a Retractable Landing Gear," NACA **Technical Note** *456 (Washington, 1933).*

THE AERODYNAMIC EFFECT OF A RETRACTABLE LANDING GEAR
By Smith J. DeFrance

SUMMARY

Tests were conducted in the N.A.C.A. full-scale wind tunnel at the request of the Army Air Corps to determine the effect of retractable landing-gear openings in the bottom surface of a wing upon the take-off characteristics of a Lockheed Altair airplane. The tests were extended to include the determination of the lift and drag characteristics throughout the angle-of-attack range with the landing gear both retracted and extended.

Covering the wheel openings in the wing with sheet metal when the wheels were extended reduced the drag only 2 percent at a lift coefficient of 1.0, which was assumed for the take-off condition. Therefore, the wheel openings in the bottom

side of the wing have a negligible effect upon the take-off of the airplane. Retracting the landing gear reduced the minimum drag of the complete airplane 50 percent.

INTRODUCTION

A somewhat excessive length of run had been required by a certain low-wing monoplane, a Lockheed Altair, during take-off. This airplane has a landing gear that is completely housed in the wing when retracted; and when it is extended, openings having an area equal to the side area of the struts and wheels are exposed on the lower surface of the wing. It was desired to know whether these openings were causing the detrimental effect upon the take-off characteristics. Consequently, at the request of the Army Air Corps tests were conducted upon this airplane in the N.A.C.A. full-scale wind tunnel to determine the effect of the wing openings upon the lift and drag characteristics.

TESTS AND APPARATUS

The airplane was mounted in the wind tunnel as shown in Figure 1. The lift and drag forces were measured with the wheels retracted, with the wheels extended and wheel wells open, and with the wheels extended and the wells covered with sheet-metal. The tests were conducted at an air speed of approximately 60 miles per hour. Figure 2 shows the landing gear extended and the wheel wells open. A description of the tunnel and balance equipment is given in reference 1.

DISCUSSION OF RESULTS

The primary purpose of the investigation was to determine the effect of the landing-gear openings in the wing upon the lift and drag characteristics during take-off. Assuming an angle of attack for take-off that would give a lift coefficient of 1.0, and comparing the curves on Figure 3 for conditions with landing gear extended and wheel wells both open and closed, it can be seen that closing the wells reduced the drag coefficient only 2 percent. Therefore it can be said that the openings in the wing for the retractable landing gear of the Lockheed Altair have a negligible effect upon the take-off of the airplane.

Because of its retractable feature, the landing gear on the Lockheed Altair is not aerodynamically clean. It is interesting to note, however, that retracting the gear reduces the minimum drag of the complete airplane 50 percent. It is therefore apparent that, if the landing gear proper is streamlined to reduce the drag, the take-off characteristics will be improved; but such a change, if carried to the extreme, would have a detrimental effect upon the landing characteristics and might necessitate the installation of some device to reduce the landing distance required.

CONCLUSIONS

A retractable landing gear of the type on the Lockheed Altair may, when extended, account for 50 percent of the minimum drag of an airplane, but the openings that house the gear in the bottom surface of the wing have a negligible

effect upon the lift and drag characteristics when the gear is extended.

Langley Memorial Aeronautical Laboratory, National Advisory Committee for Aeronautics, Langley Field, Va., March 16, 1933.

REFERENCE

1. DeFrance, Smith J.: The N.A.C.A. Full-Scale Wind Tunnel. T.R. No. 459, N.A.C.A., 1933.

Document 3-15(b), Hal L. Hibbard, "Problems in Fast Air-Transport Design," **Mechanical Engineering 55** *(October 1933): 611-617.*

Problems in
FAST AIR-TRANSPORT DESIGN
by Hall L. Hibbard

In 1910 the average speed of the fastest railroad trains in this country was 34.6 miles per hour. In 1920 this average dropped to 34 and in 1930 it was 40.9. Thus, over a period of twenty years, the average speed of the nation's thirty fastest trains increased 6.3 miles per hour. This rather unimpressive change in speed has been due in a large measure to the continuously increasing traffic congestion faced by all surface transportation. Increased speed has been fraught with danger. And here is one of the many advantages that air transport has over all others. Increased speeds, almost without limit, are possible with no attendant dangers and even added safety. Air transport, even during its short existence, has shown increased speeds of 30 to 50 miles per hour, and it has barely started.

It would be impossible in the brief space allotted to this paper to discuss all points of transport design. Therefore, a few, dealing particularly with increasing the speed of transport aircraft, have been chosen.

In connection with the discussion that follows, it will be of interest to have in mind a hypothetical airplane, and to note what effect certain changes have on it. Let us assume, therefore, that this hypothetical airplane has a gross weight of 7500 lb, a horsepower of 700, a wing area of 400 sq ft, a span of 50 ft, giving an aspect ratio of 6.25, a wing loading of 18.75 lb per sq ft, and a power loading of 10.70 lb per hp. Let us further assume that this is a low wing cantilever monoplane with retractable gear, controllable pith propeller, but with no wing flaps.

THE PROPELLER

Before high-speed airplanes became so numerous and slow-speed planes were in vogue, aircraft propellers were fairly satisfactory.The speeds were not so high but that added safety. Air transport, even during its short existence, has shown increased speeds of 30 to 50 miles per hour, and it has barely started.

It would be impossible in the brief space allotted to this paper to discuss all points of transport design. Therefore, a few, dealing particularly with increasing the speed of transport aircraft, have been chosen.

In connection with the discussion that follows, it will be of interest to have in mind a hypothetical airplane, and to note what effect certain changes have on it. Let us assume, therefore, that this hypothetical airplane has a propeller set at a pitch to give best performance at high speed also had satisfactory take-off and climb characteristics. On high-speed planes, however, a propeller with a fixed pitch will not have satisfactory characteristics for both take-off and high speed. One or the other must be sacrificed. The problem has become so acute that in some cases where speed could not be sacrificed the Department of Commerce requirements for take-off have limited the gross weight of the airplane. Such was the case two years ago with the first Lockheed Origin which was equipped with an engine without a supercharger. The take-off was so poor that even though the airplane was capable structurally of carrying a higher gross weight, this was not allowed. Of course, the propeller pitch could have been changed to secure a satisfactory take-off, but tests showed that the cruising speed was so reduced of the planes entered in the present air races. The engine diameter was greater than that of the fuselage, and the cowl was carried straight back. Another cowl was built which was drawn in at the rear. Speed tests were run with both types of cowl and the plane was approximately 15 miles per hour faster, when fitted with the cowl drawn in at the rear. In the case of a plane with a larger fuselage, cowls extending straight back were found to be best.

The first tests completed by the N.A.C.A. indicated that a double nose section in the cowl was beneficial. Later tests, however, showed that increased speeds could be obtained by omitting this inner section of the nose cowl and changing the shape slightly. British tests and some private tests in this country have verified this conclusion, but in spite of this fact, some manufacturers still continue to use the double section which not only reduces the speed but is also more expensive.

For speed and cooling, it is not the quantity of air passing through the cowl that is important but the velocity and distribution. It is a common error to assume that blocking off the air at the nose decreases the drag. Such is not the case. Air must pass through the cowl in order to give the least drag. Tests show that the drag is higher when the nose shutters arc closed than when they arc open. Much is to be learned about the proper distribution of air under N.A.C.A. cowls, and further tests should be conducted along this line.

POWER PLANTS

From the point of view of the airplane designer, the engine diameter should be as small as possible. In this connection it is gratifying to note the progress that is being made along this line with all types of air-cooled engines, including the two-row radial type. Aircraft manufacturers have had the experience of substituting engines of greater horsepower and larger diameter, only to find in almost every case that the increase in speed was not what it should have been, because of the increase in frontal area.

In single-motor planes, there is but one location for the power plant, namely, in the nose of the fuselage. In multi-motor designs, if the engines are to be placed in the wings, the motors should be placed so that the propeller is located directly ahead of the leading edge of the wing by an amount equal to 2.5 percent of the chord. In this connection it is of interest to note the tests which the N.A.C.A. is at present performing in connection with air-cooled versus liquid-cooled engines with radiators in nacelles. The results indicate a drag of 74 lb with the air-cooled arrangement and drags of 72, and 76 lb, respectively, with two liquid-cooled installations. This translated into cruising speed of one of the later type bimotor transports would mean the same cruising speed for either installation. This was also verified by the Army some months ago in the case of two identical bimotor low-wing bombers, one with air-cooled and the other with liquid-cooled engines. The speeds of the two were almost identical.

Power plants in air transports should be supercharged. Transport airplanes rarely cruise at sea level. Supercharging will give the rated power at altitude and high output in emergencies at sea level for short periods of time.

LANDING GEAR

The gain in the speed of an airplane with a retractable landing gear over one equipped with a well-streamlined gear of the non-retractable type has been greatly exaggerated. A great increase in speed because of retraction of the gear usually means that the original landing gear design was faulty. Furthermore, after retractable gears are installed, there is always the tendency to compare speeds with the gear up and down, which is obviously unfair, as retracted landing gears are never faired or streamlined and arc usually quite unsightly.

Take, for example, the Lockheed Orion. This is the original Sirius model equipped with a retracting gear. The major cause of drag in landing gears is interference. In the Sirius landing gear, all struts arc nicely faired, fairings are placed over the wheels, and all joints to fuselage and wheels are well filleted. But the parts are too close together, angles are too acute, and the gear is close to the underside of the wing.The result is excessively high interference. Thus, with the retractable gear, the increase in speed was more than 2.5 miles per hour.

Careful wind-tunnel tests show conclusively that gears in which the interference has been removed add but little to the drag of the airplane. This type of gear is a pure cantilever type without external bracing of any kind, similar to that pioneered by Northrup. Care must be taken in the design of this gear to use a good streamline shape and it is important that the gear be carefully faired at the top into the wing or fuselage. A cantilever gear of this type will reduce the high speed of the airplane less than 6 miles per hour, which means a reduction in cruising speed of 3 to 4 miles per hour.

Thus it can be seen that there is some question as to the advisability of the retractable gear from the standpoint of speed increase. However, the retractable gear is still recommended as manufacturers will not install the non-interference type gear. They will read this article, or others, be much pleased and relieved to know that gear retraction gives such small increase in speed, and then install on the very next design the old type of landing gear with all its speed-reducing interferences.

When retractable gears first came on the market, rumors were spread about the terrible results that would surely ensue if an airplane was forced to land with the gear retracted. Up to the present, six cases arc known where Lockheed plants have been forced to land with the gear retracted. In no case have there been any injuries to occupants, and in no case have more than minor damages resulted from the landing. The airplane usually slides 150 to 250 ft, depending on the condition of the ground. Because of the very high cushioning effect when the wing is only a foot or so from the ground, the landing is not rough or severe.In other words, instead of being dangerous, retractable gears have become a safety feature and transport operators arc demanding them from that standpoint. It is easy to see that safer ice landings can be made in snow, in marsh lands, and on water; and in mountainous country landings can be made on very small patches of ground.

FUSELAGE AND ACCESSORIES

Fuselages for fast airplanes should be as small as possible, consistent with comfort. There is need for wind tunnel data on fuselage sizes as there is little information on this subject. The British in the Schneider Trophy racers found that decrease in frontal area did more to increase the speed than any other item. On the Super marine racer, the cross-section area was reduced from 8.46 to 8.I8 sq ft, with a resulting increase in speed of 11 miles per hour. This was in spite of the fact that the resistance per square foot of cross-sectional area was higher for the plane with the smaller cross-sectional area. In a comparison of two of last year's racers, it was shown that the drag at 100 miles per hour for the one with the larger fuselage was 89.7 lb, while that for the other was 80.4 lb, notwithstanding the fact that the latter had 30 square feet additional wing area.The straight away speeds of the two airplanes with the same horsepower were 2.47.3 and 2.77 miles per hour, respectively. It would appear then that there is a limit to fuselage size, and from most indications, the smaller the faster.

Another factor which should be taken into account in fuselage design and which has not been given enough consideration is the shape of the fuselage across the wing. Any element, even a flat sheet, placed vertically and parallel to the airstream will interfere with the airflow on the upper surface of a wing. There is a definite flow of air over the upper surface of cantilever wings, such as are being used on present-day transports, which should not be disturbed. It is evident that if a flat sheet affects the airflow, the modern fuselage must make a great disturbance. The fuselage should conform in some degree to the direction of airflow set up by the wing.

Document 3-15(c), "Preliminary Study of Retractable Landing Gears for High and Low Wing Monoplanes,"* Air Corps Information Circular *7 (18 February 1933): 1-9.

PRELIMINARY STUDY OF RETRACTABLE LANDING GEARS FOR HIGH AND LOW WING MONOPLANES

(Prepared by E. H. Schwartz, Materiel Division, Air Corps, Wright Field, Dayton, Ohio, October 29, 1932)

SUMMARY

This report sets forth the results of an investigation made to determine whether or not the use of the retractable landing gear is advantageous in view of the large number of difficulties encountered in its design, operation, and maintenance. The investigation also included cantilever landing gears with streamlined wheel fairing and fixed streamline landing gears. The results of this study indicate that the retractable landing gear can be used advantageously on the multi-engined monoplane with the landing gear retracting into the engine nacelle or wing, and that the cantilever landing gear with wheel fairing is best suited for pursuit, attack, 2-place observation, or single-engine cargo airplanes. These landing gears permit use of monocoque fuselage structure.

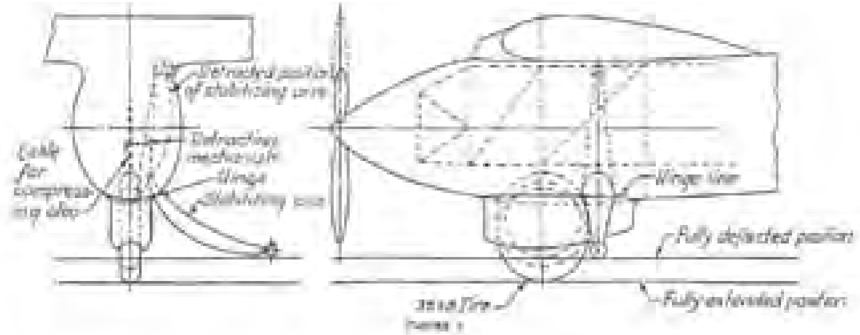
A number of retractable, cantilever, and fixed streamline landing gears were examined. The indicated increase in high speed due to retraction of retractable landing gears is misleading, as in all cases it was found that the retractable landing gear was not streamlined, and large wing or fuselage openings prevailed with the landing gear in the down position, giving a low high speed with wheels down. This accounts for the comparatively smaller speed increase obtained from wind tunnel tests on removal of faired landing gears. A study was also made of the weights of the various types of landing gears on present United States Army airplanes for comparison.

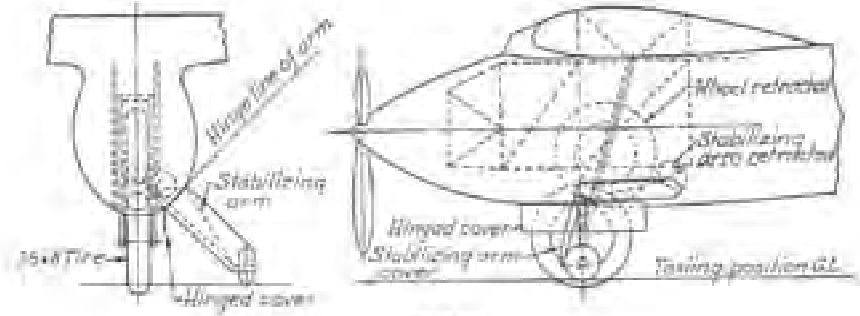
The number of retractable landing gears of the fuselage and wing type procured by the Air Corps is small. For this reason, little work has been done up to the present on improving the design of retractable landing gears. Therefore, a study was made of possible retractable landing gears, and, as a result, the landing gears shown in Figures 1 to 10, inclusive, were conceived, and it is believed that they have good possibilities for future designs.

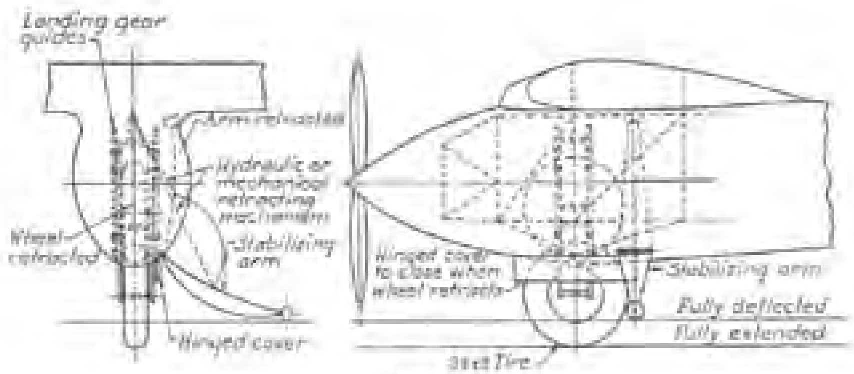
This study also showed that a good installation of a retractable landing gear to an existing airplane cannot be made efficiently or economically without a complete redesign of fuselage or wing.

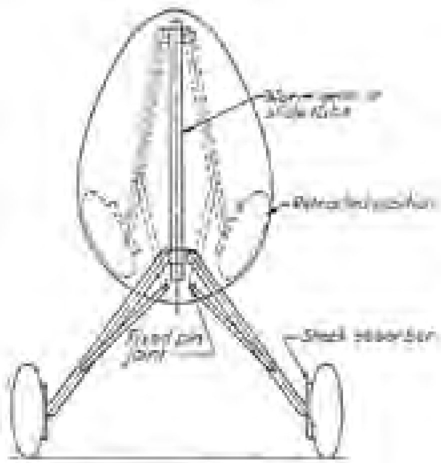
All weight data on landing gears was obtained from the actual weight statements AN-9102 and AN-9103 furnished the division by the contractors. All airplane weights were taken from the 689 inspection reports of the various airplanes.

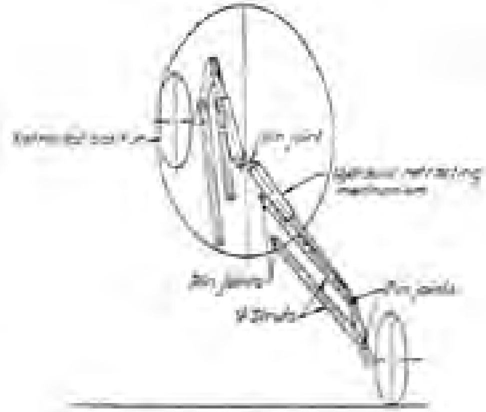
All data on the high speed of airplanes and models was obtained from official flight and wind-tunnel tests with the exception of the YlC-17 airplane.

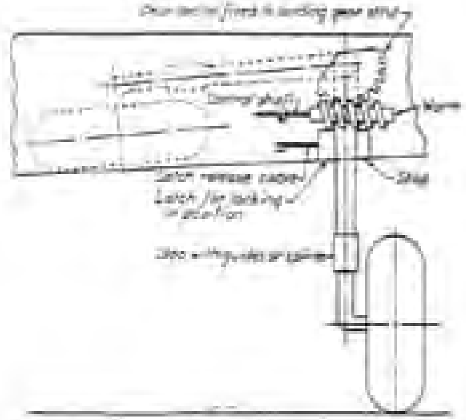

Retracting mechanism
Hinge line
Fully deflected position
Fully extended position

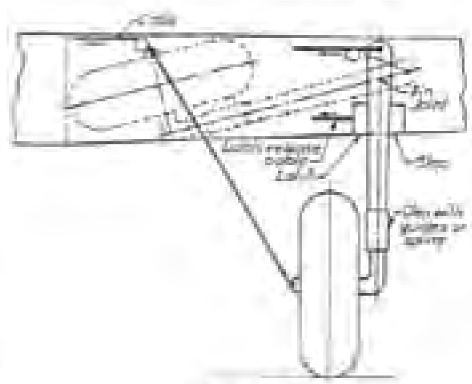

Hinge line of arm
Stabilizing arm
Hinged cover
Wheel retracted
Stabilizing arm retracted
Hinged cover
Stabilizing arm cover
Taxiing position G.L.

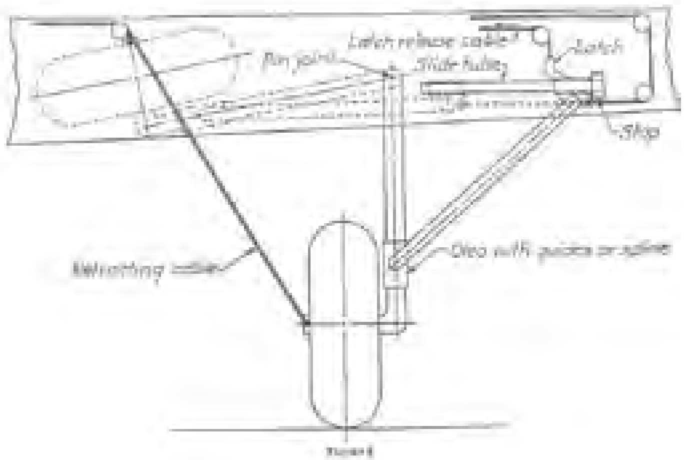

Landing gear guides
Arm retracted
Hydraulic or mechanical retracting mechanism
Wheel retracted
Stabilizing arm
Hinged cover
Hinged cover to close when wheel retracts
Stabilizing arm
Fully deflected
Fully extended

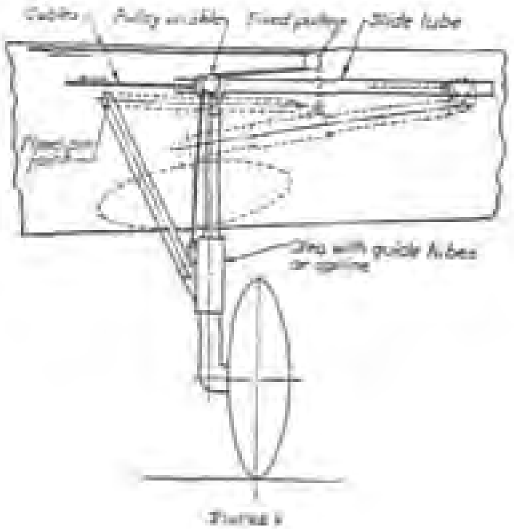

Retracted position
Fixed pin joint
Shock absorber

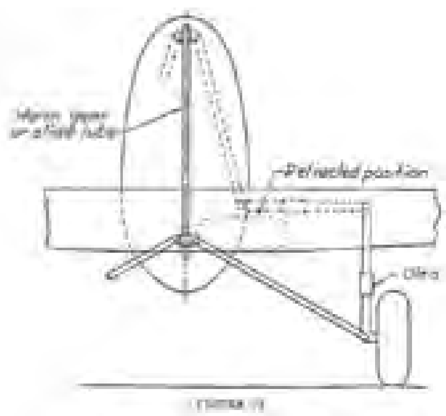

Pin joints
V Struts


Control shafts
Worm
Latch release cables
Latch for locking in position
Stop


Latch







OBJECT

The object of this report was to determine whether or not the use of the retractable landing gear is advantageous in view of the large number of difficulties involved in its design, operation, and maintenance, and to determine the type of landing gear which will give the best installation and performance on various types of monoplanes in use by the Air Corps.

DISCUSSION

In this investigation the YlC-12 airplane has been considered as a typical high-wing monoplane for the possible installation of a retractable landing gear. Similarly the YlC-23 or XP-900 may be considered in the same class for low-wing monoplanes. (Gross weight 4,000-5,000 pounds.) XO-35 or XB-9 considered typical of engine nacelle type landing gear.

It must be realized that the application of a retractable landing gear to an existing airplane will involve considerations and result in decisions that would differ somewhat from those considerations involved at the time of layout of a new design, for in the latter case fuselage shape, bulkhead locations, spar locations, wing thickness, etc., can be readily laid out or altered to best suit a retracting mechanism.

At the start of the investigation certain governing factors must be recognized, namely

(1) That the landing angle of the airplane be as large as possible, of a magnitude to enable the airplane to develop not less than 90 percent of its maximum lift coefficient in landing. A decrease in landing angle will result in an increased landing speed.

(2) That the angle in side view between a line passing through the center of gravity of the airplane and the center of the wheels, with the airplane in the level landing attitude, and the vertical must not be less than 12° for airplanes without brakes and must be 20° for airplanes with brakes.

(3) That the angle in the front view between the vertical and a line joining the center of gravity and the point of contact of either wheel and the ground shall not be less than 25°.

(4) That the propeller clear the ground by at least nine (9) inches with the airplane in either a flying or landing position.

The foregoing requirements are taken from the Handbook of Instructions for Airplane Designers, Volume I.

The purpose for which the airplane is to be employed may establish other requirements, such as:

(1) That the landing gear in its retracted position offer no interference with the pilot's location.

(2) That the cargo or bomb space remain intact in cargo or bomber airplanes.

(3) That certain space in the vicinity of the center of gravity, or elsewhere, be reserved for military equipment, etc.

(4) That the fuselage section be as small as possible to give the best vision possible.

There are other general requirements to be met by retractable landing gears, again possibly subject to alteration in the case of special-purpose airplanes:

(1) That the handling of the airplane on the ground be as simple as possible (compared with a 2-wheel landing gear with swiveled tail wheel).

(2) That the retracting gear and movement of parts be such as to permit a structure capable of carrying the fuselage design loads, and that this structure preferably be determinate and simple.

(3) That the retracting mechanism be reliable and incapable of jamming in any intermediate position.

(4) That the landing gear itself be capable of absorbing the prescribed landing loads without damage to the fuselage or airplane.

(5) That maintenance of the landing gear be readily accomplished.

(6) That the landing gear be lowered or dropped with reasonable speed.

(7) That the wing or fuselage openings be closed by means of doors, fairing, etc., on retraction of the landing gear.

Retractable landing gears may be classed as follows:

(1) Fuselage type, with landing gear retracting into the fuselage and the mechanism housed within the fuselage.

(2) Wing type, with landing gear folding up into the wing and the mechanism located in the wing.

(3) Engine nacelle type, with landing gear retracting back into the wing, engine nacelles, or both, of multiengine monoplanes with the engines located outboard of the fuselage.

The fuselage type of landing gear may be either a single-wheel or a 2-wheel landing gear. The single-wheel landing gear does not require as wide a fuselage as the 2-wheel on retraction; but requires for stabilization the use of two small auxiliary wheels, which must also be retracted into the fuselage. This requires the use of two retracting mechanisms-one for the main wheel and one for the two auxiliary wheels. This means excess weight and more fuselage cut-outs.

The fuselage type is best suited for a high-wing monoplane. The distance from the wing to ground on a high-wing monoplane prohibits the use of the wing type, since wing cut-outs and length of retracting mechanists depend on the landing gear length. This would give a very poor and inefficient installation. The fuselage-type landing gear occupies useful and valuable space at an important section of the fuselage near the center of gravity of the airplane. In cargo or bomber airplanes cargo or bomb space is valuable near the center of gravity. In other types of airplanes this place is important for the fuel tank and military equipment location. A large fuselage section is necessary in order to house the landing gear and mechanism. The size of the fuselage section depends on size of wheels, landing gear length, and length of retracting mechanism. The length of retracting mechanism is dependent on the length or travel of the landing gear. The landing gear length must be such so as to meet the requirements of the Handbook as stated on page 1. This type landing gear is unsuitable for pursuit, attack, or 2-place observation airplanes, since fuselage space is very limited and since a small fuselage section is most important for vision and performance. Due to the large cut-outs on the sides and bottom of a critical section of the fuselage, an inefficient structure must be used to carry the required loads. An inefficient structure means an excess structure weight. The large cut-outs necessary for retraction of this landing gear prohibits the use of a monocoque structure.

The wing type of retractable landing gear is best suited for a low-wing monoplane, which requires only a short landing gear, since the length of landing gear determines the size of cut-outs and space needed for retracting mechanism and retracted landing gear. In order to house the retracted landing gear and retracting mechanism, a thick wing section is necessary. For this reason, on a tapered wing monoplane the landing gear is retracted in close to the fuselage to make use of the maximum wing thickness available. This eliminates the use of outboard wing fuel tanks. On this type of airplane the landing gear cannot be retracted into the center section of the wing, due to the large cut-out necessary in the wing at the wing attachment fittings, which are at a critical section of the wing. With this type of retractable landing gear it is most important to close tip the wing openings on retraction by means of doors, plates, or fairing. An excess weight is added due to addition of the doors, plates, or fairing for closing the wing openings, due to an inefficient wing structure necessary to house the retracted landing gear and due to the addition of the retracting mechanism. Due to the wheels being forward of the front spar, the axis of rotation or hinge line must be at an angle to the chord line in order to fold the landing gear up between the spars. Where a slide tube or screw is used, unless the spars are far enough apart, ball and socket joints must be used on the struts to prevent interference with the rear spar, due to the landing gear swinging at an angle to the chord, since, with pin joints, the retracting mechanism must be in a plane perpendicular to the axis of rotation or hinge line. This type of retractable landing gear permits the use of monocoque fuselage.

The engine nacelle type of retractable landing gear is restricted to monoplanes with multi-engines located outboard of the fuselage. The engine nacelles are used advantageously to wholly or partially house or fair-in the retracted landing gear. This landing gear on a low-wing monoplane swings about the fitting on the front spar back into the wing in front of the rear spar. The wheels of this type landing gear cannot be fully retracted into the wing due to the size or diameter of the wheels and the depth of the wing section in front of the rear spar, for example, the YB-9, XB-907, and YO-27. In a high-wing monoplane with the engines located below the wing, as on the XO-35, the landing gear swings back behind the engine. The protruding wheels on the high and low wing monoplane installation are used to advantage for protection of the airplane against possible failure of retracting mechanism and possible failure of pilot to drop the landing gear. The design is such that

TABLE I.—*Landing-gear weights*

FUSELAGE TYPE, RETRACTABLE

Airplane	Total weight: Landing gear (pounds)	Total weight: Landing gear (per cent of gross weight)	Weight of landing gear chassis (pounds)	Weight of wheels, brakes, brake mechanism (pounds)	Weight of wheel fairing (pounds)	Weight of retracting mechanism (pounds)	Gross weight of airplane (pounds)	Landing load factor	Remarks
[illegible]	[illegible]	[illegible]	120	[illegible]	....	[illegible]	[illegible]	[illegible]	Retracting mechanism, [illegible]
WING TYPE, RETRACTABLE									
XP-900	[illegible]	[illegible]	[illegible]	[illegible]	—	[illegible]	[illegible]	7	Cable operated mechanism, second-class.
ENGINE, NACELLE TYPE, RETRACTABLE									
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	....	[illegible]	[illegible]	[illegible]	Cable operated mechanism, second-class.
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	....	[illegible]	[illegible]	[illegible]	Hydraulic mechanism.
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	....	[illegible]	[illegible]	[illegible]	Positive-acting mechanism, screw type.
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	....	[illegible]	[illegible]	[illegible]	Cable operated mechanism, second-class.
CANTILEVER									
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	....	[illegible]	7	Cantilever with streamlined wheel fairing.
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	....	[illegible]	7	Do.
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	....	[illegible]	[illegible]	Do.
NONRETRACTABLE									
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]		....	[illegible]	[illegible]	[illegible] type.
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	....	[illegible]	[illegible]	Do.
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]		....	[illegible]	[illegible]	Do.
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	....	[illegible]	[illegible]	[illegible] type with wheel fairing.
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]		....	[illegible]	[illegible]	[illegible] type.
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	....	[illegible]	[illegible]	[illegible] landing gear with streamlined fairing.
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]		....	[illegible]	[illegible]	[illegible] type.
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	....	[illegible]	[illegible]	Wire braced landing gear.

the exposed part of the wheel strikes the ground first, thus protecting the airplane against serious damage. For this reason the retracted wheels are not fully faired, thus saving the added weight of streamlined fairing. This landing gear is best suited for heavy cargo, bombing, or 3-place observation monoplanes and does not take up any valuable fuselage or wing space. Only small wing cutouts are necessary. Due to its short length, this landing gear is not heavy, and its weight compares very favorably with the non-retracting landing gear. It permits use of monocoque fuselage. For comparison of the retractable landing gear with the non-retractable landing gear the following table is given:

Retracting mechanisms may be classed as follows:

(1) Positive acting, as the screw or gear type.

(2) Cable operated for retraction and lowering.

(3) Cable operated on retraction only, but dropping by gravity.

(4) Hydraulic or oleo action.

The first group is positive in action and is very suitable for heavy airplanes. The screw has the advantage of self-locking the landing gear in any position. It can be readily adapted for any speed, but means must be provided for lowering the landing gear fast and retracing the landing gear with a reasonable speed. The screw type requires a long screw, which in turn requires much fuselage or wing space. The gear-

TABLE II.—*High speed of airplanes with various types of landing gears*

Airplane	High speed: Wheels up or with fairing (miles per hour)	High speed: Wheels down or without fairing (miles per hour)	Increase in high speed (miles per hour)	Increase in per cent of high speed	Engine: Type	Engine: Horsepower	Remarks
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	Wheel fairing flight tests, speed increase due to retraction of landing gear.
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	Same as above.
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	Do.
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	Do.
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	do.	do.	Do.
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	do.	do.	[illegible]
[illegible]	[illegible]	No data available without fairing			[illegible]	[illegible]	[illegible]
Curtiss	P-6E, [illegible] at [illegible] feet.	P-6A, [illegible] at [illegible] feet.	[illegible]	[illegible]	do.	do.	[illegible]
[illegible]	[illegible] with fairing.	[illegible] without wheel fairing.	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible] with landing gear off.	[illegible] with landing gear on.	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible] with wheel fairing.	[illegible] without wheel fairing.	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible] with wheel fairing.	[illegible] without wheel fairing.	[illegible]	[illegible]	[illegible]	[illegible]	Do.
[illegible]	[illegible] with landing gear off.	[illegible] with landing gear on.	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

sector type does not require much space; but means must be provided for locking in position and stops must also be provided.

The second group makes use of cables and pulleys for retraction and lowering of landing gear. Cables are easily adapted to an airplane, but the system of pulleys and cables must be complicated due to the necessity of cables for releasing lock on landing gear for retraction as well as the double system for lowering and retracting landing gear. Tension must be maintained on the free cable on lowering to prevent over-running of cable on drum and snagging of cable on parts of the airplane. Stops and catches must be provided to lock the landing gear in down position and to take the required loads on landing. Care must be used in this installation to prevent chafing of the cables. The double system of cables insures more positive locking of landing gear in lowering against any air resistance, as it does not rely on gravity. Frequent inspection of the cables is necessary.

The third class depends on cables for retraction and on gravity for lowering. It is essential to have tension in the cable at all times on dropping of landing gear to prevent the slack cable from snagging or overrunning the drum. This type gives a lighter installation than the second type. The cable system must also include means of releasing the lock or catch on retraction. Air loads on some landing gears may be such as to resist the dropping of the landing gear and prevent the locking of it in place. For this reason the single cable system is not recommended.

The fourth class, that is, hydraulic or oleo type, gives a good installation due to the absence of moving parts. The tubes from the cockpit to the cylinder can be easily located to suit the airplane. A long cylinder is required to obtain the desired travel of landing gear. For this reason the use of this type is limited to space available for the installation. With the tubes properly fastened and supported and with frequent inspection of tubes and connections, there is little danger of tube breakage which would render the retracting system useless. This type is easily adapted for any weight of landing gear and is positive in action in either lowering or retraction of landing gear. An excess weight is added due to the addition of pump, valves, hydraulic unit, and tubing.

Table 1 shows the actual weights of various types of landing gears as used on present Army airplanes. The retractable landing gear weights are grouped in their proper class. The retractable landing gear weights of the fuselage or wing type do not offer a fair comparison with the other types of landing gears, as the retractable landing gear weights of these two classes do not include the excess weight necessary in the inefficient structure of the wing or fuselage due to cut-outs, nor the excess weight due to a larger fuselage or thicker wing section necessary to house the retracted landing gear and retracting mechanism. The retractable landing gear weights of the engine nacelle type with landing gears attaching to engine mount do not include the heavier engine mount or the weight of added cowling which is charged to the engine. In using the figures of Table 1 the design landing load factors must be taken into consideration.

Of the six Army airplanes using retractable landing gears, four employ the use of cables for retraction and lowering of the landing gear. The XO-35 uses the hydraulic system for retraction, and at first glance seems rather light in weight; but, due to the engines being located below the wing, the landing gear is very short. However, the landing gear is very simple, and the only added weight is the oleo unit, tubes, valve, and pump. It is very accessible and provides for easy maintenance.

The YB-9 uses the screw type mechanism for retraction. It is heavier than the XO-35 landing gear, due partly to the wing location with respect to the ground, requiring a longer landing gear. However, it is lighter than the landing gear of the XB-907, using cables for retraction.

The cantilever landing gears from Table 1 seem to be heavier than the retractable landing gears at first glance, but the retractable landing gear weights do not include excess fuselage or wing weight. The cantilever landing gear weights are true, so they may be easily compared with split-axle type landing gear weights. Both monoplanes and biplanes using cantilever or non-retractable landing gears are listed in Table 1, but they may be compared directly, as these landing gears attach directly to the fuselage and are independent of wing location or the type of wing. The P-6 and XP-6E are good examples for comparison as they are almost identical, with the exception of the landing gear, and have same load factor. The weight of the P-6 split-axle landing gear is 5.8 percent of the gross weight, and the weight of the XP-6E cantilever landing gear is 6.9 percent of the gross weight, or an increase in weight of 1.1 percent due to the installation of the cantilever landing gear with streamlined wheel fairing. The 0-25C and the YO-31 also form a good comparison, as their gross weights are practically equal and their design landing load factors are the same. The 0-25C split-axle landing gear weighs 4.8 percent of the gross weight, and the YO-31 cantilever landing gear weighs 5.8 percent of the gross weight, which is 1.0 percent heavier due to the use of wheel fairing and cantilever landing gear.

The YO-1G uses a split-axle type of landing gear and streamlined wheel fairing. The weight of this type of landing gear is 5.6 percent of the gross weight compared to 5.2 percent for the same airplane without the wheel fairing, or an increase of 0.4 percent due to the addition of wheel fairing.

The weight of the XA-8 fixed streamline landing gear, which is attached to the wing, is very heavy, that is, 8 percent of the gross weight. This is due to the weight of the fairing (50 pounds) and the provisions for synchronized guns, etc.

Table 2 shows the high speed of various airplanes and models with retractable landing gears, with landing gears removed, with cantilever landing gears, with low-wing monoplane landing gears, and with split-axle type of landing gears. The figures show a very high increase in high speed due to the retraction of the landing gears. In the cases of the XP-900, XO-40, XB-907, and YlC-23, the retraction of the landing gear increases the high speed about 30 percent. These retractable landing gears have an exceedingly high drag when lowered, due to the large fuselage or wing openings, due to no streamline fairing of the struts or wheels, and due to exposed fairing or doors for closing the openings being open.

Flight tests of XB-907 were made for high speed with and without landing gear shielding with landing gear down and with landing gear retracted. The high speed with wheels retracted was the same (195.8 miles per hour) without landing gear shielding as with landing gear shielding, but with wheels down the high speed increased 23 miles per hour by removing the shielding. On this airplane the shielding is a detrimental clue to the excess weight and increased drag with wheels down requiring a longer distance for take-off.

These speed increases are not true indications of the actual gain in speed of a retractable landing gear over a well streamlined non-retractable landing gear. Wind tunnel tests on the XA-8 and XP-16 models show an increase of only 9.8 and 7.7 percent, respectively, for the removal of these streamlined landing gears. This is the ideal condition of retraction, but it is never realized in practice since perfect closing up of openings is not possible and since changes must be made in the fuselage or wing to accommodate the retractable landing gear and its mechanism. It is safe to assume that the actual speed increase of a retractable landing gear is not greater than 6 percent over the split-axle type or low-wing monoplane landing gears.

A good comparison of high speed can be made between the cantilever and split-axle type of landing gears by taking the P-6A and P-6E airplanes which have same engine. The P-6E with a cantilever landing gear weighing 6.9 percent of the gross weight has a high speed of 194.5 miles per hour at 5,000 feet, and the P-6A with split-axle type of landing gear weighing 5.8 percent of the gross weight has a high speed of 176.8 miles per hour at 5,000 feet, or a gain of 10 percent in high speed due to the use of cantilever landing gear which weighs 1.1 percent of the gross weight more than the split-axle type.

From actual flight tests on the YO-31 with cantilever landing gear a high speed increase of 2.2 percent was obtained by use of wheel fairing with an increase of 0.8 percent in gross weight. From flight tests on the O-1G (split-axle type landing gear) a high speed increase of 2.0 percent was noted due to the use of streamlined wheel fairing, which increased the gross weight by 0.5 percent. The high speed increase of the XP-20 (split-axle type of landing gear) obtained from the flight tests was 1.0 percent by the use of wheel fairing. This data indicates that streamlined wheel fairing is essential for high performance of airplanes with non-retractable landing gears, as a large increase in high speed results from a small increase in landing gear weight due to wheel fairing.

Table 2 shows that the YlC-17 with wire-braced landing gear and wheel fairing is 6.3 percent faster than the YlC-12 with split-axle type of landing gear with an increase in landing gear weight of 1.1 percent of the gross weight. Since the high speed on the YlC-17 is not official and since the engine of the YIC-17 is 50 horsepower greater than that of the YlC-12, no information can be gotten from these figures.

Figures 1 to 3, inclusive, show several possible methods of retracting a single wheel landing gear. These landing gears use two small auxiliary wheels for stabilization on the ground. The landing gear of Figure 1 does not fully retract, only by the amount of oleo travel. It employs the use of a good streamlined fairing for the wheel. The stabilizing arms are retracted vertically into the fuselage and are so shaped to close up the fuselage openings on retraction. Figures 2 and 3 show two different methods of retracting the auxiliary wheels in addition to showing the retraction of the large wheel. In Figure 2 the large wheel retracts up and back in the fuselage to permit more room in the fuselage for the retracting mechanism. The auxiliary wheels swing back into the fuselage and fairing is so shaped to close up the fuselage openings. Figure 3 shows the large wheel retracts vertically into the fuselage as well as the stabilizing arms, which also close up the openings in the fuselage. The landing gears of Figures 2 and 3 use a hinged cover to close up the opening in fuselage when the main wheel is retracted. Any one of the retracting mechanisms may be used, depending on the fuselage space available. These types of landing gears do not require as wide a fuselage as a 2-wheel landing gear. The use of a 3-blade propeller can be used advantageously to lower the fuselage, thereby shortening the landing gear length.

The landing gear shown in Figure 4 is very suitable for a high-wing monoplane, and does not require a complicated mechanism, but has the disadvantages of Class I landing gears as stated on page 2. It is positive in action and has good mechanical advantage throughout its entire travel. The use of cables and slide tube or screw mechanism is recommended for this landing gear.

The landing gear of Figure 5 is also of the first class. The struts form a parallelogram and the wheels are kept in a vertical plane in any position. A hydraulic mechanism is used for retraction. This landing gear requires a wide fuselage and large fuselage cut-outs. However, it does not require a deep fuselage.

Figures 6 to 9, inclusive, are landing gears of the second-class retraction into the wing. Figure 6 shows one of these landing gears using a worm gear and gear sector for retraction. A stop and catch or locking device must be used to prevent landing loads being taken out by the mechanism. Means for releasing the locking device must also be provided for retraction. This mechanism does not require a thick wing section. Figure 7 is similar to Figure 6, except cables are used entirely for retraction and lowering of the landing gear. Figure 8 shows a landing gear which uses a slide tube and cables for retraction. A latch, or locking device, and a stop is also necessary to take the landing loads. This mechanism does not require a thick wing section. It also has a good mechanical advantage in any position of the landing gear. The landing gear shown in Figure 9 has good possibilities. The pull of the cable on retraction is changed from the slide to the landing gear proper to the best advantage as the slide pulley passes the fixed pulley. Thus the first pull of the retracting cable breaks the slide loose and as the slide passes the fixed pulley the cable leaves the slide pulley and the landing gear is lifted directly, thereby giving a straight and direct pull on the landing gear. This landing gear requires a thick wing section. All of these landing gears must swing about an axis or hinge line which is at an angle to the chord line in order for the landing gear to retract between the spars. For this reason, where the landing gears are not far enough apart to permit the landing gear to swing in the same plane as the mechanism, ball and socket joints must be used.

The landing gear shown in Figure 10 comes in neither the first or second class landing gear, but is a cross between the two. The landing gear folds up into the wing and the retracting mechanism is housed in the fuselage. It has a good mechanical advantage throughout its travel. It has the disadvantage of cut-outs at a critical section of the wing; that is, the point of attachment of the wing to the fuselage, and also eliminates the use of fuel tanks in the center section of the wing. This landing gear does not require a wide fuselage section nor does it require a thick wing section, as it retracts into the thickest section of the wing.

CONCLUSIONS

As the result of this study, it was found that a good installation of a retractable landing gear to an existing airplane cannot be made efficiently or economically without a complete redesign of fuselage or wing.

The wing and fuselage types of retractable landing gears in present designs are complicated, heavy, and unsatisfactory in operation. For these reasons the retractable landing gears shown in Figures 1 to 10, inclusive, are proposed and show good possibilities for future designs. The fuselage type of retractable landing gear does not readily permit the use of monocoque fuselage structure.

As a result of the study of the nacelle type of retractable landing gears, it is believed that this landing gear will give far better high-speed performance than a cantilever or split-axle type of landing gears for multiengine monoplanes such as bombers, cargo, or 3-place observation airplanes. This type of retractable landing gear does not interfere with use of monocoque fuselage structure. The use of a cantilever landing gear on these airplanes would require a heavy landing gear and would increase the drag of the airplane. The use of a screw or hydraulic retracting mechanism gives a reliable and good installation for the engine nacelle type of retractable landing gear.

It was also found that fairing or shielding on the nacelle type of retractable landing gear may not increase the high speed with wheels retracted and that it may decrease the high speed materially with wheels down. Thus the excess weight of fairing or shielding is of no advantage, but a detriment due to increased drag with wheels down, requiring a longer distance for take-off.

The results of this investigation also show that the cantilever landing gear with streamlined wheel fairing gives the best installation for monoplanes or biplanes in the single-engine class, such as pursuit, 2-place observation, attack, or light cargo airplanes. This installation permits the use of a small fuselage, which is of vital importance in a pursuit, attack, or observation airplane for vision and high-speed performance; and it requires no fuselage or wing space, which is valuable in these airplanes for fuel tank or military equipment locations. This landing gear can be easily adapted to existing airplanes without any major changes or alterations, including monocoque fuselage structures.

It is believed that the cantilever landing gear with wheel fairing will give a better high-speed performance than the low wing monoplane landing gear and the wire braced landing gear with a lighter installation.

This study proves conclusively that the use of streamlined wheel fairing is important for high-speed performance for non-retractable landing gears with only a small increase in gross weight.

Document 3-15(d), Richard M. Mock, "Retractable Landing Gears," **Aviation** *32 (February 1933): 33-37.*

Though retractable landing gears for landplanes have only come into popular use in the last few years, the idea is by no means new. In 1876 Penaud and Gauchot patented a design of an airplane which had a front landing chasses of the wheel, all retracting into the fuselage to reduce the air resistance. However, as far as can be learned the idea never took a practical form until after the War, when Dayton-Wright built a high wing cantilever monoplane for the 1920 Gordon Bennett race with the wheels retracting flush in the sides of the fuselage. The year before Lawrence Sperry built an amphibian flying boat which is believed to have had the first practical retractable gear.

However, these were special purpose planes. On normal civilian and military planes of that time the drag of the landing gear compared with the total drag of the airplane was not sufficient for a retractable gear to give enough increase in speed to warrant the additional complication. For the next ten years the problem received little attention. Today most high speed planes are "cleaner," so that the landing gear drag is a greater proportion of the total. Thus, the gain by the elimination of landing gear drag is likely to be greater, though the value of tail wheel retraction is still doubtful.

As a rule, where the genera arrangement allows a reasonably efficient retractable landing gear it has been found that, compared with a similar design with fixed wheel type undercarriage, the speed is increased only 3-4 per cent at 130 m.p.h. Because faster planes are generally cleaner the increase grows to 6-7 per cent at 150 m.p.h. and at 165-170 m.p.h. to about 10 per cent. One well known low wing design had a high speed of about 175 m.p.h. with a streamlined external gear, while with the landing gear fully retracted the speed was increased some 25 m.p.h., or 14 per cent. It is possible that as much as 20 per cent increase in speed might result at 185-195 m.p.h. from retracting the landing gear on a very clean design. On designs where the wing bracing is combined with a streamlined landing gear, as on may low wing racers or the Bellanca Airbus, the gain due to retraction would of course be debatable.

At the speeds achieved today the exact relative merits of the streamlined external gear (whether or not it is part of the wing bracing) and the retractable type depend much upon the ingenuity of the designer and the purpose, type and size of the airplane.

SPACE

On some type insufficient space is available in the wing or fuselage to retract the landing gear. Fundamentally, for high speed flight it is believed better to increase the frontal area slightly to provide the needed space. Thus all parts are concentrated into one mass that can be given the best possible shape, rather than have a number of separate units with the accompanying interference between them and the possible disturbances where they meet. This problem is closely tied in with problems of where to house the wheels and will be discussed later.

WEIGHT

An increase in speed with the same power means less flying time, hence less fuel and oil required for a given flight distance. The saving in weight of fuel and oil goes to offset the additional weight of the retracting mechanism, linkages and accompanying increase in structural weight.

An example might help to illustrate this point. Assuming an airplane having 8,000 lb. gross weight, 3,000 lb. useful load, and cruising at 150 m.p.h., the addition of a retractable mechanism might increase the weight empty 60 lb., and reduce the useful load by the same amount. The fuel consumption might be assumed to be 35 gal. and oil consumption 3 gal. per hour, or a total of 230 lb. per hour. A 10 per cent increase in speed would reduce the time for a given flight 9.1 per cent, saving about 21 lb. of fuel and oil per hour (neglecting that slightly smaller tanks could be used.). Therefore for a 450 mile flight, if this increase in speed is achieved, the additional weight of the retracting gear would have no net effect on the pay load. This result is typical, whatever the size of the plane.

In general a fixed landing gear weighs 6-9 per cent of the empty weight of the airplane or 10-14 per cent of the useful load, though this may vary as much as 7-17 per cent. A retractable landing gear increases the landing gear weight 10-20 per cent, reduces the useful load 1-3 per cent. Airplanes of about 10 lb. per hp. Loading use 7-11 per cent of their useful load per hour in fuel and oil. Thus, with a three hour range the additional weight would be offset by the fuel and oil saving if the speed were increased 5-7 per cent. For a two hour flight, a 10-13 per cent increase would be sufficient.

COST

Outside of the greater value of a faster service, an increase in speed often means that fewer planes are needed to keep a given frequency of schedule. In addition, the flying time to cover a given distance is less, meaning a reduction of all costs proportional to flying time, such as fuel and oil, flying crews pay and maintenance and overhaul of airplane and engine.

Consider a conservative approximation of the additional expense due to a retractable gear. Assuming that the initial cost of the retractable landing gear is $500-$1,500, depending on the size, type and purpose of the plane, there will be an increase in cost of depreciation of 10 cents to 30 cents per hour based on 5,000 hours for the life of the airplane. Though some operators claim a retractable gear is cheaper to maintain than a fixed gear, with streamlining, assume an additional maintenance cost of 3 cents to 10 cents per flying hour making a total additional expense of 13 cents to 40 cents per flying hour.

Though in most instances the insurance costs would not be increased, it seems reasonable to assume a possible increase of 1 to 2 per cent in both crash and passenger liability insurance. From the Post office Department figures, insurance represents 6.5 per cent of the total operating expense. Crash and passenger liability are assumed to constitute about 40 per cent of the total insurance or 2.6 per cent of the total flying expense. If this were increased 1 to 2 per cent due to the retractable gear the total increase in operation expenses would be only 0.026 to 0.052 per cent.

Using the most probably figures within the above range in each case, and adding an imaginary plane of strictly up to the minute design and correspondingly lowered operating costs, the following table shows the increase in cost due to the addition of a retractable gear. (Only a portion of the "direct operating expenses" are directly proportional to flying time so only 30 per is used.)

An increase of 5 per cent in cruising speed, however, would mean that a flight of a given distance could be made in 4.8 per cent less flying time and hence at a saving of 4.8 per cent minus the added costs just tabulated. A 15 per cent increase would cut the time 13 per cent. It is thus apparent that in a typical case the provision of a retractable gear ought to reduce the operating expenses by from 3 to 12 per cent of the direct flying costs, or from 1.0 to 3.6 per cent of the total cost, depending upon the gain in speed. For a transport carrying from 8 to 12 passenger and crusing at 160 m.p.h., a retractable gear ought to cut total costs about $1-$2.50 per hour or 0.625—1.56 cents per mile.

DANGER OF FAILURE

Another factor to be considered is the danger of the landing gear not being in the proper position at the time of landing. This may happen from any one of four reasons: (1) A structural failure of a landing gear part or the actuating mechanism, due to faulty design, materials or workmanship; (2) fouling of the wheel bracing linkage, or actuating mechanism by ice or mud; (3) pilot's neglect to lower the wheels, which an be eliminated by proper warning devices; or (4) a mechanism that is too slow to allow the wheels to be lowered in a forced landing

As far as can be determined there has been no instance of a personal injury due to a retractable landing gear functioning improperly. In a forced landing in a small or bad field, especially with passengers a "wheel up" landing is definitely an advantage as the speed is slightly lower, the run considerably shorter and the possibility of nosing over practically eliminated. One of the leading operating companies has found it desirable to make forced landings with wheels retracted, as the expense of subsequent repairs is usually less than if the wheels were completely or partly down. In the past two years a great number of wheel up landings have been made with various privately and commercially operated planes, and invariably they resulted in less than $1,000 damage and more often $300 or less. The greatest damage usually experienced is when the wheels are in a partly retracted position. The extent of the damage due to a wheel up landing is usually limited to the propeller and the bottom covering of the fuselage.

To insure proper functioning of a retractable landing gear the mechanism should be designed to be simple, easy to service and maintain, and well protected against being fouled by foreign material thrown up by the wheel during take off. With the wheel opening in the bottom of the wing or fuselage the mechanism should not be exposed to mud, which might freeze solid after the gear has been retracted. In planning the disposition of the landing gear members, care should be taken to prevent struts from being forced into the cabin if a landing is made with partially retracted wheels. The actuating mechanism, especially cables, should not take any landing loads and the linkage must be so laid out that it never approaches dead center closely enough to require great operating force. The mechanism should not be reversible as the "down position" is approached (unless sole dependence is to be placed on a lock to hold the wheels in landing position).

WHEEL LOCATION

The best method and location for housing the landing gear varies with the type of plane. The gear can be retracted into one of five places :into the wing, the fuselage, and engine nacelle of a muti-engine design, an external fairing, or an exposed position where the frontal area or interference drag is reduced. In deciding which one to use, the aerodynamic effect of the wheel opening and the partly retracted gear during take off and landing immediately invites consideration. A disturbance of the flow by the opening might affect the stability, while a wing opening might affect the airfoil characteristics. These aerodynamic effects seem to be negligible however in most designs. Recent full scale wind tunnel tests by the N.A.C.A. on a Lockheed monoplane show that a wheel opening in the lower surface of the wing does not affect the flying characteristics when the wheels are extended. (Reference 2)

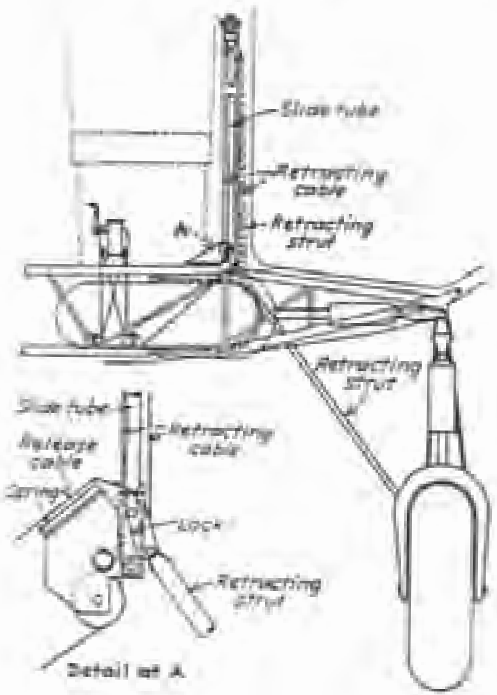
Until a year or two ago, practically all American air planes were biplanes or high wing monoplanes. As these types have inherently less drag than the low wing monoplane their general arrangement should be more desirable. (References 3, 4 and 5) The ease of retracting the gear in the low wing monoplane is apparently one of the main reasons for its increasing popularity. It is interesting that with the same power and loading a low wing monoplane, which gained 14 per cent in speed by fully retracting the landing gear, was only 3-4 per cent faster than a high wing version of the same plane with a fixed streamline, single strut, tie rod braced gear indicating that the high wing model would have been considerably faster with a retractable gear. However, as it is exceptionally difficult to retract the landing gear into the wing of a high wing type the fuselage is a more promising location for single engine designs.

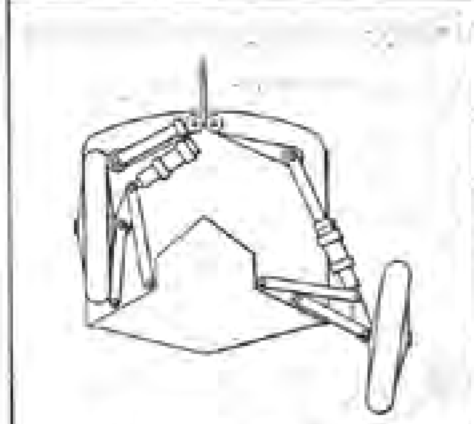
There have been many successful designs with the wheels retracting either flush into the sides of the fuselage or into the bottom. An interesting mechanism adaptable for retracting the wheels into the sides of the fuselage is shown in the illustration taken from a patent held by L. R. Grumman. A deviation from a true parallelogram is made to provide "toe in" for landing and taxing. Another type of parallelogram mechanism was used on the Great Lakes amphibian of 1929. The member parallel to the wheel was an oleo carrying bending from the axle. A diagonal strut hinged at the top of the oleo went downward and inward to a point in the hull. By raising the inner end of this member the wheel was raised and swung inward into a recess in the side of the hull.

To have sufficient tread and ground clearance on a high wing monoplane with the wheels retracting into the bottom of the fuselage it is usually necessary to either build out small stubs as on the Stinson R-3 or to have a wide fuselage as on the Burnelli. The mechanism of the Stinson is illustrated herewith. Burnelli uses a sort of curved track to give a specific path to the upper end of one member of a tripod strut arrangement. There are an infinite number of other variations with shortening or retracting struts, folding struts, etc.

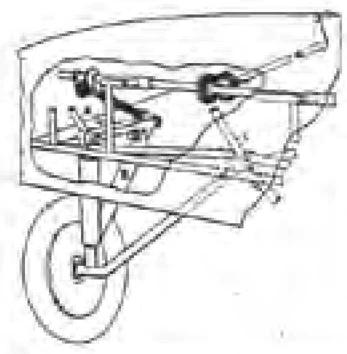
To retract the wheel into a thin wing as on a biplane or an externally braced monoplane would usually necessitate and increase in wing thickness to house the

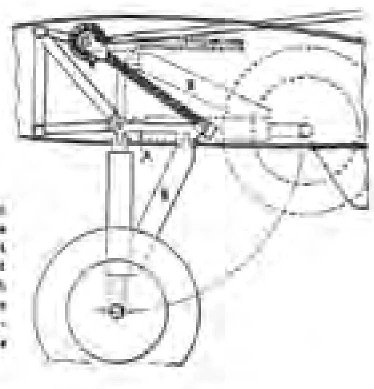
Retracting mechanism on Stinson Model R-3. There are two parallel beams in stub wing. Side brace member lies in place of rear beam and has its upper end raised vertically by a cable actuated by a winch and worm gear. The worm gear is self locking preventing gear from falling if cranking is stopped. Gear may be "cranked" down or dropped by releasing a clutch, and locks in down position automatically. Operation of hoisting crank releases lock mechanism before actually beginning lateral movement of strut. Hydraulic arresters eliminate shock in working parts. It is stated that the gear is raised in 20 seconds and lowered in two.





Drawing from Grumman's retractable landing gear patent. The shock absorber strut has the hinge effect, so that under compression the eccentricity tends to keep strut from buckling. In retracting, cables turn the sheave or wheel attached to the upper end of the shock absorber member. The upper end turns inward, "breaking" the strut.





Machinery for bringing up the wheels on the Boeing Monomail. Member "A" is a trussed structure to restrain "B" against side motion. The screw is protected against dirt and grit by a split dural cover which opens as the gear is retracted and closes as it is lowered. When electrically operated it is lowered in slightly over 30 seconds and raised in 43 seconds.

wheel or to carry the landing loads. The Bellanca "Roma," a sesquiplane built in 1928 had one portion of the wing thickened slightly to house the wheel. The landing loads were carried as in the "Airbus" but with the wheels fully retracting into the bottom of the inner bay of the lower wing. In a biplane the wheels might fold inward into the bottom of the fuselage. It is understood that the new Curtiss YO-40 Army observation plane is of this general type, with thick filleted roots on the lower wing.

INTO THE NACELLE

The problem becomes easiest when there are outboard engine nacelles, for the nacelle usually provides ample space for housing the gear and gives the proper tread with the wheels located under the concenetrated loads. On a high wing monoplane design with engines in the leading edge, the landing gear must be very high. Because of the great height it is almost impossible to take the side loads as a cantilever, and therefore the design must be complicated by side struts to the fuselage as on the General YO-27 (see illustration) where they remain out in the air-stream when the wheel is retracted. In the new Dutch built Fokker F-XX a compromise apparently has been reached by retaining the high wing and lowering the engine nacelles below the wing so that a completely retractable gear can be obtained with no side struts to the fuselage, the nacelles carrying all the load. Fokker seems to have decided that the slightly increased drag of the lowered nacelles did not offset the advantages of the high wing, with its inherently less drag and greater lift as well as its absence of buffeting and better passenger vision. (see page 59 and references 3, 4 and 5). However, some designers have reached a compromise in another fashion, locating the nacelle in the best position with reference to the landing gear and then lowering the wing to have the engine in the leading edge. Examples of this are the new Martin bomber, the Boeing bomber and the new Boeing transport, and the new Douglas monoplane transport.

The problem of retraction in these designs is quite similar to that in folding the wheel backward into the low wing of a single-engined plane. The simplest method seems to be to have the upper end of the rear brace strut move, while the forward struts pivot about their upper ends. On the Martin bomber, this is accomplished by having the rear member slide backward on a track when the wheel is hoisted by a cable attached to the axle. The gear is lowered by a cable pulling the rear strut forward to the landing position, where it is locked. In the Boeing Monomail a rotating screw and nut draw the upper end of the rear member forward and upward from the landing position. A number of equivalent systems might have been substituted such as a slide tube, rack and gear, piston and cylinder, etc.

HINGED STRUTS

A very similar plan is to provide the rear strut with a hinge near its center and "break" the strut. The Eaglerock Bullet built in 1929, the Fokker 1930 XO-27 (forerunner of the General YO-27), the Keith-Ryder San Franciscan racer of 1931 and 1932, and the new Clark-General single engined transport furnish examples. On the General transport an hydraulic piston and cylinder bearing against the landing gear strut at its "breaking" point supply the actuating force. The hydraulic system automatically locks so that if a landing is made before the wheels are fully lowered the gear will remain rigid. It is stated that the gear can be raised in eighteen seconds and lowered in eight.

The most familiar design in which the wheels swing up into the wing transversely, is the Lockheed which folds them inward. As in the Verville 1922 Pulitzer

Racer, and the 1928 Bellanca "Roma," the shock absorber is hinged to the front spar. A rear brace member is hinged to the rear spar, but further outboard than the shock absorber hinge so that when the gear is retracted about an axis passing through these hinges, the wheels move backward until they are between the spars. A similar system, but with the wheels moving outward is used on the French Bleriot transport monoplane.

As on the Lockheed the wheel on the Ballanca was raised by a cable, but instead of going to a piston and cylinder it went directly to a winch. The Bellanca had no side bracing and side loads were taken by bending in the front member, which was attached to the top and bottom of the front spar. The inner lower brace to the spar was drawn back out of the way to allow the wheel to be raised. The wheel was lowered by allowing it first to fall of its own weight and then drawing it home by a cable on the outboard side. The wheels could be manually raised in 17 to 21 seconds and lowered in seven to nine seconds.

Perhaps the best example of reducing the drag by leaving the wheels largely in the air stream and merely changing their position is the new Martin bomber. The wheels are retracted behind the outboard nacelles, but not enclosed. With the gear extended, there is a loss of approximately 12 per cent of the top speed with the wheels retracted. It was found that fairing over the wheels did not help the top speed, but with the wheels extended the fairing caused considerable disturbance.

In discussing the problem of retraction, some landing gear linkages and actuating mechanisms have been described. Let us consider the principal advantages and disadvantages of some of the various types. The most desirable seems to be the purely mechanical system, having only levers and hinges. It is usually simple, positive, quick operating, and cheap to build and maintain. A mechanical cable system usually is light, simple, and cheap, but has the problem of fouling of cables and pulleys, frayed cables, failure of synchronization due to the stretching of cables, etc. A slide tube or a track is usually light, simple, cheap, and quick operating, but it may jam. Screw and hydraulic mechanisms, though they are positive in action and can produce great forces, are usually slow unless power actuated. They have a tendency to be heavy, and though the hydraulic systems have the advantage of being very flexible they are confronted with the problems of pumps, leaks, and congealing oil in cold weather.

Though it is not difficult to make a landing gear that may be retracted or lowered manually without great exertion, often considerable time is necessary. The Department of Commerce requires that a gear be lowered in 60 seconds or less. This can usually be accomplished by a clutch which disengages the slow acting retracting mechanism of high mechanical advantage allowing the gear to drop of its own weight. Auxiliary means, such as rubber shock absorber chord, are sometimes used to hasten the lowering of the wheels.

On commercial transports and military planes it is felt that unless the landing gear naturally comes down very quickly the addition of power operated mechanism to an emergency hand actuated mechanism is desirable. The pilot's duties are ever increasing, and especially during landing in bad weather his attention should not be distracted by lowering the landing gear. On military types the power operated type is desirable to avoid distracting the pilot during formation take offs and landings. Power actuated gears are usually electrically driven, though it should be feasible to utilize the air pressure to raise or lower the gear by a flap.

Mention was made of warning devices to insure that the pilot has not forgotten to lower the wheels before landing. Most of these are electrical and are actuated if the throttle is retarded when the wheels are not locked in the down position. Originally lights were used but they proved inadequate and were replaced by a horn near the pilot's ear. The latest device, on the Clark General transport, is an electric motor with an eccentric weight which causes the control stick to vibrate (see page 59, this issue). In addition most planes have indicators, usually colored lights, showing when each wheel is locked in the up or down position. The Department of Commerce requires and indicator "to show the position of the wheels at all times."

Closing, I wish to acknowledge my appreciation to the Army Air Corps, and to those manufacturers and engineers who have so kindly supplied me with data and illustrations. Unfortunately, space did not permit credit being given in each instance nor the publication of all of the material desired.

Document 3-15(e), Roy G. Miller, letter to editor, **Aviation**, *"Retractable Landing Gears,"* **Aviation** *32 (April 1933), with response from Richard M. Mock: 130-131.*

MR. MOCK is to be congratulated on his very interesting article on Retractable Landing Gears, which was published in the February issue of AVIATION. He has invited our attention to practically every phase of this question. Some of his thoughts are, I am sure, new to many of us. There are several points, however, on which he seems to be rather too optimistic as regards the virtues of landing gear retraction.

It is stated that an airplane having a top speed of 175 m.p.h. with a faired gear of fixed design showed an improvement of 25 m.p.h. when fitted with a retractable gear. This amounts to an increase of 14 percent in speed and corresponds to a reduction in drag of about 33 percent. The drag of the landing gear, according to this, would be 50 percent, as much as the drag of the airplane with landing gear retracted.

The frontal area of a faired landing gear usually amounts to less than 10 percent of the total frontal area of the airplane, including landing gear, fuselage, pilot's compartment, wing, tail, etc. It is hard to believe that 10 percent of the frontal area would have half as much resistance as the other 90 percent. This would mean that the resistance coefficient for the landing gear was 4.4 times as high as the average for the balance of the airplane, as based on frontal area. It is true that landing gear fairing cannot be complete or perfect, but neither can the shape of other parts be such as to produce a minimum drag for a given frontal area. The fuselage, in order to offer sufficient length for tail surfaces, must usually have a fineness ratio of at least 2.5 times the ideal fineness ratio. Landing gear fairing suffers from no such limitation. Furthermore, the fuselage lines must enclose the engine, cool it and dispose of the exhaust gases. The pilot's enclosure and a number of minor items (such as door hinges, door handles, cowl fasteners and the like) add to the fuselage drag. The wings and the tail surfaces have about 2.5 times the fineness ratio of the ideal streamline strut section, and their shape is chosen for considerations other than minimum drag, as based on frontal area. When we consider that an increase of 2.5 times the ideal fineness ratio corresponds to an increase of from 50 to 100 percent in drag, it appears that the airplane we are discussing would have been about eight times as good a job of streamlining as its landing gear. The resistance coefficient of a well-faired landing gear may be equal to the average for other parts of the airplane, but it certainly is not 4.4 times as great.

In the more or less haphazard performance testing employed by the average commercial airplane organization weather conditions and other important variables may be neglected. Change of propeller setting, changing propellers, admitting more air to a starved carburetor, changing to high compression pistons or charging supercharged gears are tricks of the trade which may be employed by an enthusiast to prove his point where he is out from under the watchful eye of an Army or Navy trial board. He is apt to minimize the importance of such changes even in his own mind and consider that they are not worth mentioning.

The estimate of 60 lb. for the difference in weight between a fixed and a retractable gear for a transport of 8-12 seats seems to be rather optimistic even 60 lb. may eliminate one passenger where the weight allowance is on the ragged edge.

One feature of the retractable landing gear which has received very little attention is the effect which its drag has on stability and control. In preparing to land, a pilot ordinarily needs all of his faculties to judge the approach and to handle the controls. If the flying qualities of the airplane and the "fell" of the controls suddenly change when the landing gear is extended the result may be a bad landing or, worse, a stall while turning into the field. Roy G. Miller, Hartford, Conn.

Mr. Miller's interesting comment is very much appreciated. In the general statement I made regarding the possible improvement in an airplane with an efficient retractable gear compared with a fixed external gear, I indicated the increase in speed rather than the reduction in total drag, because I felt that a definite figure for gain in speed was more comprehensible than an abstract drag figure. I made the assumption that faster airplanes would have somewhat less drag and not achieve the improved speed through excess of power alone. Therefore, I assumed that, because of the fixed size of struts, wheels, and tires with materials and loadings now used, as the general cleanness of the airplane was improved, it was not possible to reduce the drag of the landing gear in direct proportion and that the ratio of landing gear drag to total drag would increase. This has been observed on a number of actual designs which were tested with streamlined fixed gears as well as retractable gears. As I mentioned in the article, the possible gain depends much upon the ingenuity of the designer and the purpose, type and size of the airplane.

In the general statement I mentioned that for a conventional airplane with 165-170 m.p.h. with a fixed external undercarriage a gain of 10 percent was possible, corresponding to a reduction in drag of about 24 percent, allowing 1 percent for improved propeller efficiency due to change in V/ND.

The airplane which I used as an example, as having its speed actually increased 14 percent by fully retracting the landing gear, was the low wing Lockheed Sirius. This model with external fixed gear had an exceptionally clean fuselage with filleted cantilever wing and tail. Recent tests show that the cockpit opening did not greatly affect the drag. However, the landing gear drag was far from being consistent with the low drag of the rest of the airplane. The wheel was braced to the wing by a vertical shock strut and a "Vee" of two struts on the inboard side, one to the front spar and the other to the rear spar. The space between the front and rear members of the Vee was completely filled by a flat surface meeting the wing at a fairly acute angle. I believe that this surface and the interference between the surface and the wing accounted for a large part of the drag. The intersection between the shock strut and the wheel fairing might have been improved slightly, while the wheel fairing in top view was considerably longer than the most efficient shape. In addition, the opening in the bottom and the projecting tire also contributed to the drag.

In the caption of the picture illustrating the Lockheed gear I mentioned a flap that was added to the later models completely covering the wheel when retracted. Because of the higher speed this made a difference of 6 m.p.h., according to Mr. Lloyd Stearman, president of the Lockheed Aircraft Corporation. This 6 m.p.h. corresponds to a reduction in drag of 8 percent neglecting any change in propeller efficiency, which would give some 18 percent rather than 14 percent.

I believe the figures for the increase in speed on this plane to be accurate, because a number of planes of this type, both with retractable and fixed gears, were built and all seemed to indicate about the same increase in speed.

Substantiating the above, Mr. George W. DeBell of the General Aviation Manufacturing Corporation states that their Clark GA43, low wing transport, increased its speed over 15 percent by retracting the landing gear. The plane was tested, both with a single strut cantilever gear with struts and wheel streamlined, and with the retractable gear now used. As both the speed of this plane and the Lockheed referred to are believed to be in the same range these values of 14 percent and 15 percent seem to be in agreement.

Mr. T.P. Wright, vice president of the Curtiss Aeroplane & Motor Company, states that in checking over results obtained with several of their models in which retractable landing gears have been installed and on which other data pertaining to no retractable landing gear is available he finds that the speed increase over the tripod type fixed gear is 10 percent. He draws attention to the fact that it is incorrect to compare these speeds with such gear retracted and extended, as in the extended position they have more drag than a fixed external gear initially designed for the plane. His estimate of 10 percent is on the basis of the true comparison, though he does not mention at what speed this is attained.

From the figures I gave in my article, I believe it is clear that if 10 percent, or even somewhat less were gained, it is advantageous to retract the gear.

Regarding the weight of 60 lb. for the addition of a retractable gear to an 8,000 lb. transport, I draw attention to the fact that the complete retraction mechanism on the new Martin bomber weighs only 79 lb. and this is a much larger airplane. Mr. DeBell states that the increase in weight of their plane, which is about this size, was 100 to 110 lb., considering each gear complete. He says this is relatively high, as the majority of the parts are not affected by the weight of the plane, such as the pump, and the valve mechanism, the oil lines, and the cowling to cover the landing gear after retraction. In this plane, this portion of the weight, unaffected by the size of the plane, is 38 lb. I believe that a hydraulic system is inherently heavy.

Mr. Wright agrees that if compared with a tripod gear this weight of 60 lb. appears to be correct. He mentions that on several of their recent planes they have used a single strut cantilever gear which is considerably heavier than the tripod type. With this type of gear he states that their experience is that the speed would only be increased 5 m.p.h. when going to the fully retractable gear, but that the weight would actually be lessened by some 50 or 60 lb.

In considering the additional weight of the retractable landing gear, one should not neglect the saving in fuel due to the improved speed and the weight of the streamlining on the fixed external gear. On the Lockheed Sirius the weight of the wheel fairing is 50 lb.

Regarding the fourth paragraph of Mr. Miller's letter I can only say that I attempted to be conservative in any of the general statements I made and that most were based upon data known to be accurate. I doubt if the gains indicated to me by some of the commercial manufacturers were obtained by supercharging or the use of higher compression ratios. However, I call attention to the fact that the higher speeds, due to the retractable gear, may slightly increase the dynamic pressure on the carburetor intake, slightly supercharging the engine and increasing its power. This I believe is another point supporting the retractable gear, as is the fact that at the higher speeds the opening for taking in air to cool the engine can be reduced, reducing the drag.

I do not believe that very many detailed data exist on the effect of the drag of the landing gear, when down, on the stability or flying characteristics of the airplane.

Richard M. Mock

Document 3-16(a-h)

(a) Fred E. Weick of Hampton Virginia, U.S. Patent 2,110,516. Filed 18 Jan. 1934; issued 8 March 1938. Fred E. Weick Collection, LHA.

(b) Fred E. Weick & Associates, Inc., to Waco Aircraft Co., Troy, OH, 8 Apr. 1938, Weick Collection, LHA.

(c) Weick & Associates, Inc., to Waco Aircraft Co., Troy, OH, 2 Feb. 1939, Weick Collection, LHA.

(d) C. J. Brukner, President, The Waco Aircraft Company, Troy, OH, to Fred E. Weick and Associates, Inc., Evelyn Place, College Park, MD, 16 Feb. 1939, Weick Collection, LHA.

(e) Weick & Associates, Inc., to The Waco Aircraft Company, Troy, OH, Attn: Mr. C. J. Brukner, President, 22 March 1939, Weick Collection, LHA.

(f) Weick & Associates, to Timm Aircraft Corporation, Grand Central Air Terminal, 1020 Airway, Glendale, CA, Attn: Mr. R. A. Powell, Vice-President, 22 March 1939, Weick Collection, LHA.

(g) Alpheus Barnes, The Wright Company, 11 Pine Street, New York, to Mr. Grover C. Loening, c/o U.S.S. *Mississippi,* Pensacola, FL, 16 Jan. 1915, reprinted in Grover Loening, *Our Wings Grow Faster* (Garden City, NY: Doubleday, Doran & Co., Inc., 1935), p. 45.

(h) Grover Loening, "What Might Have Been," in *Our Wings Grow Faster* (1935), pp. 43-46.

Fred E. Weick received two different patents covering different aspects of the steerable landing gear incorporated into the W-1 and W-1A. (His patent attorney was Allen E. Peck of Washington, DC, who had been a pilot during World War I and thus was familiar with aeronautics.) The first was U.S. Patent 1,848,037, filed on 16 February 1931 and issued on 1 March 1932; it mostly concerned his airplane's limited elevator control arrangement. The second (presented below) was U.S. Patent 2,110,516, filed on 18 January 1934 and was not issued until 8 March 1938; it covered some of the airplane's other features, including combinations of the tricycle gear with two-control operations. In addition, this second patent made claims involving the use of a flap to aid takeoff when a tricycle gear was used. Weick's patent application stayed in limbo for over four years because its claims interfered with an application by Joseph M. Gwinn, Jr., of Buffalo, New York. Gwinn had been working on a little airplane called the Gwinn Aircar, which had aims somewhat similar to those of the W-1 and W-1A. The patent court eventually decided that the date of Weick's conception of the W-1 was about a year earlier than Gwinn's design, and Weick received the patent.

The first six documents in the string to follow concern Weick's second patent and his efforts to protect it against infringements. As readers will see, companies building aircraft with some version of steerable tricycle landing gear in the late 1930s, such as the Waco Aircraft Company of Troy, Ohio, argued that their particular design differed from Weick's and therefore did not infringe his patent. Once the basic advantages of this sort of gear were well known within the aeronautical community, it was inevitable that other designers would apply it for their own purposes. This obviously did not work to Weick's financial benefit, but the widespread adoption of the new landing gear arrangement served the needs of American aviation well. Weick consoled himself with the fact that what he had contributed was not a product for sale but a gift to the aviation world.

To the end of his life, Weick felt that he had never seen a tricycle gear on a production airplane that was as effective in making a wide variety of landings, both easily and safely, as that of the W-1 and W-1A. His little plane's nose wheel was placed far out ahead; the gear had 12 inches of shock absorber and was designed much stronger than it needed to be. He and his associates could afford to go overboard in this direction because with the W-1's pusher arrangement, the plane's balance was helped by a little extra weight in the nose. And with the 18 inches of shock-absorber travel for the rear wheels, safe landings on smooth surfaces could be made, as long as the vertical velocity did not exceed about 20 feet per second, almost regardless of the manner in which the airplane was brought to the ground. With the wing-tip clearance made possible by the airplane's high-wing arrangement and five degrees of dihedral, pilots made satisfactory landings while still turning with a 30° bank. The airplane virtually straightened out by itself. With a tractor arrangement, however, it was almost impossible to get the full potential out of the gear. To achieve the effective wheelbase, one would have had to put the nose wheel out in front of the propeller, and this could not be done structurally.

Readers should find the last two documents in this string curious. Both relate to the U.S. Circuit Court Court of Appeals ruling in January 1914 that Glenn Curtiss and his associates had in fact infringed the Wright brothers' patent. The next-to-last document communicates a buoyant reaction of the Wright Company to that ruling. The last provides an excerpt from the autobiography of Grover Loening in which the pioneering aircraft builder gave voice to the potential that seemed to exist in 1914 for an out-and-out aircraft monopoly dominated by Wright.

The whole matter of patents in U.S. aeronautical history deserves a much closer look than historians have given it. Patent policy certainly factored into the reinvention of the airplane, but whether it enhanced or stifled innovation is not altogether clear. Scholars have paid considerable attention to the famous patent suit involving the Wrights and Glenn Curtiss, most recently in Tom D. Crouch, "Blaming Wilbur and Orville: The Wright Patent Suit and the Growth of American Aeronautics," in *Atmospheric Flight in the Twentieth Century*, eds. Peter Galison and Alex Roland (Dordrecht: Kluwer Academic Publishers, 2000), pp. 287-300. In this essay, Crouch argues that the Wright patent did not retard U.S. aviation in the way many observers over the years have claimed. What choked it far more was the failure of the U.S. government to recognize the crucial importance of the new technology.

In the same volume that includes Crouch's essay, there is another essay on patents that offers an even more significant set of important new insights about U.S. aeronautical patent policy history. In "Pools of Invention: The Role of Patents in the Development of American Aircraft, 1917-1997" (pp. 323-345), Alex Roland analyzes the infamous cross-licensing agreement of 1917 that set up a patent "pool" by which U.S. aircraft manufacturers could get around the problems of the existing Wright and Curtiss patents plus avoid the future bottlenecks and high cost that otherwise would have been sure to come with patent infringement suits. Roland's argument is too detailed to do more with here than briefly summarize. Contrary to what one might think, the patent pool was not a factor stifling innovation in the U.S. aircraft industry, because the industry chose not to protect itself in that way. "An invention published in a patent could often be worked around faster by the competition than if it were kept secret until incorporated in an actual aircraft," Roland explains, "whence the competition would have to reverse-engineer it, master its production, and redesign an existing plane to install it as a modification" (p. 339). Keeping things proprietary inside a company for as long as possible was much more vital to the industry than patenting.

Roland shows how this distinguished aircraft manufacturing from many other types of U.S. industries, in which patents played a greater role. The U.S. aircraft industry came to be dominated by very large corporations in which teams produced most innovations rather than lone inventors. Improved design and greater efficiency in manufacture was much more important to the aircraft industry than outright invention. Design process and manufacturing practice was the key to a competitive advantage, not patentable ideas per se. Roland also underscores how the federal government would finance 85 percent of U.S. aerospace research over the years. In this case, patents meant little. Industry did not have much incentive to patent inventions when the government contracts that sustained them (primarily contracts with the military) typically required free licensing. With the government sharing this technology freely with its other contractors, the value of patents diminished even further. This may be the reason why aviation historians have paid so little attention to patents, Roland speculates. Patents simply have not meant much to the U.S. aircraft industry.

This provocative conclusion suggests that the aircraft manufacturers' patent pool—which was later ruled a violation of the Sherman Anti-Trust Act—was not a factor stifling innovation in U.S. aircraft manufacturing. But at least one legal scholar and antitrust expert seems to disagree. In a May 1997 address in San Antonio before the American Intellectual Property Law Association, Joel I. Klein, Assistant U.S. Attorney General for the Clinton Administration's Antitrust Division, Department of Justice, and the main driving force behind the government's antitrust suit against Microsoft, used the early history of aeronautical patents in the United States as a case study of how cross-licensing and patent pools can in fact result in a blunting of competition.

Klein reviewed the history in the following way. On the eve of the country's formal entry into World War I in April 1917, U.S. aircraft manufacturing was in quite a mess. The government needed hundreds of new aircraft, but patents held by the Wright-Martin Aircraft Corp. and by the Curtiss Aeroplane & Motor Corp.—and those patents contested between the two firms—blocked production. Both companies demanded high royalties, which for government procurement meant excessive airplane costs. In response to numerous complaints from the army and navy, in January 1917, the National Advisory Committee for Aeronautics recommended creating an aircraft manufacturers association, one whose first task was to affect the cross-licensing of aeronautical patents. Seven months later, the Manufacturers Aircraft Association (MAA) came to life, with an endorsement from a special advisory committee convened by Assistant Secretary of the Navy, Franklin D. Roosevelt. The MAA quickly reached an agreement, which eventually turned into the Amended Patent Cross-License Agreement of 31 December 1928. From the time of the original agreement in 1917 on (until a consent decree of November 1975 brought on by the Justice Department finally abolished the MAA and terminated the cross-licensing agreement as a Sherman Anti-Trust violation), member companies could enjoy full and inexpensive access to all patents held by the other members. This agreement signaled one of the most crucial milestones in American aviation development, for it prevented a virtual deadlock in aircraft construction that could have come, with countless patent infringement suits.

The patent pool encompassed practically all U.S. airplane manufacturers; the charter members of the MAA involved several of the top companies then in airplane manufacture, including Aeromarine Plane and Motor Company, Burgess Company, Curtiss Aeroplane and Motor Corporation, L.W.F. Engineering, Standard Aero Corporation, Sturtevant Aeroplane Company, Thomas-Morse Aircraft, and Wright-Martin Aircraft Corporation. Many others soon signed on, to everyone's advantage. As Klein explained in his 1997 address, the MAA pool "resolved all pending infringement claims and bound the members to give each other nonexclusive licenses to 'all airplane patents of the United States now or hereafter owned or controlled by them.' " (Klein, "Cross Licensing and Antitrust Law," Address before the American Intellectual Property Association, San Antonio, TX, 2 May 1997, copy available at www.apeccp.org.tw/doc/USA/Policy/speech/1123.htm, p. 5). So tidy was the arrangement that MAA members agreed not to put any relevant patent out of the pool's reach. The cost to members was a flat $200-per-aircraft royalty. One hundred thirty-five dollars of that amount (67.5 percent) went to Wright-Martin and $40 (20 percent) to Curtiss until their patents ran out or until each of the two companies received a maximum of $2 million each, whichever came first; the other $25 per-aircraft-royalty (12.5 percent) stayed in the pool for MAA administrative costs. An MAA arbitration panel decided which patents merited royalties and how much those royalties were to be in dollars. In sum, the MAA worked to manage the entire field of inventions for airplanes, including patent research of all types, the prosecution of patent applications in the U.S. Patent Office, the granting of patent licenses, as well as the arbitration of disputes and royalty awards. This tightly knit arrangement laid the basis for U.S. aeronautical patent policy for the next 58 years, until the consent decree ended it in 1975.

It should be apparent from this summary why Attorney General Klein, one of the moving forces in the U.S. government's antitrust case against Microsoft, would be interested in the history of the MAA and how its management of a patent pool affected subsequent aeronautical innovation in the United States. The formation of the pool in the summer of 1917 provoked a mild uproar, with charges of an "aircraft trust." Under duress, Secretary of War Newton D. Baker, personally delighted with what the pool meant for aircraft procurement, asked U.S. Attorney General Thomas W. Gregory for an antitrust advisory opinion. It took Gregory only a few weeks to answer that the anticompetitive effects of the pool came nowhere close to outweighing "the very real procompetitive benefits resulting from assembling those patents into an affordable package available to all comers" (Klein, p. 6). The MAA pool solved what attorneys today would call the "bilateral monopoly problem;" in 1917, that sort of monopoly was on the verge of being exercised by Wright-Martin and Curtiss. These two companies had the entire embryonic U.S. aircraft industry over a barrel—not to mention each other as well. In September 1917, Attorney

General Gregory essentially concluded that combining the Wright and Curtiss patents into a wider pool would not only be procompetitive (and work out for both companies), but prove also to be the best solution for everyone involved, including the U.S. military then engaged in world war.

Much more so than historian Roland, attorney Klein questioned the effects of the patent pool on long-term innovation. Without actually evaluating American progress in aircraft manufacturing from 1917 on, he emphasized the possible anticompetitive effects of such arrangements when patent pooling deters or discourages participants from engaging in ambitious R & D, thereby retarding innovation. Granting licenses to one another for current and future technology at minimal cost encourages "free-riding" and reduces incentives to compete in R&D efforts, Klein emphasized. In 1917, the urgencies of World War I undermined this concern. The U.S. government needed airplanes, a lot of them, and fast. It was not a good time for fretting about stifling some innovation that might or might not bear fruit 20 or 30 years down the road. In this context, the ruling in favor of the patent pool made pragmatic sense.

Still, in Klein's view, the establishment of the MAA's historic cross-licensing agreement raised significant questions about just how competitive—or noncompetitive—the system of aeronautical patent management in the United States actually proved to be. How innovative could the system be if the patent pool turned out to be the only route by which new technology could travel to the marketplace and into the design of airplanes? How much incentive could there have been for new invention if the pool agreement provided for only a "reasonable" royalty, an amount typically set much lower by the industry's own arbitration jury than it would otherwise often have been?

For American aeronautical development, the matter of patent policy seems to be fundamentally important. Historians need to address the competing conclusions of Klein and Roland. Such research may prove especially important to an understanding of the airplane design revolution of the interwar period, for the young American aircraft manufacturing industry simply could not have afforded to "reinvent" the airplane if every single airplane part, including major systems such as wing design, airfoils, engine arrangement, and landing gear, were all patent-protected. What it needed was an approach that severely limited what a patent protected and that made available at low cost the "generic knowledge" of how to design effective aircraft. This seems to be what the MAA system did, however much in restraint of trade.

U.S. aeronautical patent policy in the 1920s and 1930s seems to be vitally important to the history of the NACA as well. Not only did the NACA promote the original creation of the MAA in order for it to affect the cross-licensing of aeronautical patents (NACA chairman Joseph S. Ames was one of the MAA's founding trustees), the NACA also served through this period as the country's major source of aeronautical innovation, through its R&D establishment. One might argue that by fostering the concept of the patent pool, the NACA ensured that its own organization would serve in that capacity rather than U.S. industry, thereby cementing its role for decades to come. This helped engender the very result that historian Roland cites in his study, that is, that the government would come to finance 85 percent of aerospace research in the United States. In other words, the NACA may have served as a substitute for whatever spirit of competition was lost through the pooling arrangement. Still one can wonder, as Klein does, what new inventions and innovations may have been lost from potential R&D by the U.S. aircraft manufacturing due to the coziness of the overall patent pooling arrangement.

Regarding patents in the special context of our study, it seems clear that very few advances in aerodynamics were patent-protected. With the exception of specific forms or devices such as the "Harlan flap," few aerodynamic improvements took patentable form. As readers will see in the documents below, even when an inventor received a patent, as Weick did for his landing gear, manufacturers managed to integrate the refinement without paying much, if anything, for it.

Document 3-16(a), Fred E. Weick of Hampton Virginia, U.S. Patent 2,110,516. Filed 18 Jan. 1934; issued 8 March 1938.

UNITED STATES PATENT OFFICE

AIRPLANE

Fred E. Weick, Bethesda, Md., assignor to Fred
E. Weick & Associates, Inc., Hampton, Va.,
A corporation of Virginia

Application January 18, 1938, Serial No. 185,634

This invention relates to certain new and useful improvements in airplanes; and the nature and objects of the invention will be apparent to and readily understood by those skilled in the art in the light of the following explanation and detailed description of the accompanying drawings illustrating what I believe to be the preferred embodiments or aerodynamical and mechanical expressions of my invention, from among various other forms, embodiments, designs, combinations and constructions of which the invention is capable within the spirit and the scope thereof.

This application is a substitute for and continuation in part of the pending application filed by me July 5, 1934, Serial No. 733,893, for improvements in airplanes.

Fundamentally it is a general aim and a primary object of my present invention to reduce or substantially eliminate the basic hazards and dangers that result from certain features and characteristics inherent in the prevailing designs and types of conventional heavier-than-air craft, or "airplanes" as such craft are generically termed herein. It is generally recognized and established that the conventional airplane, because of such inherent dangers and hazards, can only be practically piloted with any degree of safety under the varying conditions encountered in flight and in taking off and landing, by highly trained and skillful pilots; and that the general use of the conventional airplane is therefore restricted to those having the time and finances for the training and the ability to successfully acquire from such training the necessary and essential piloting technique and skill for safety in flight.

One of the major hazards of the conventional airplane is the landing operation which requires delicate and skillful handling of the airplane and excellent vision and a high degree of depth perception from the pilot to successfully carry perception from the pilot to successfully carry out. This hazard is a direct result of the inability of the conventional airplane to land steeply because of a flat glide and of high landing speed with a long landing run, and further due to the difficulty of contacting the ground accurately at a desired point because of the limited range of gliding angles available to the pilot, even by side slipping the airplane.

A feature of an airplane of my present invention resides in a design providing an airplane that cannot be stalled and has a wide range of gliding angles available to the pilot, and also has a low landing speed and a short landing run on the ground, which together with internal control and stability at all speeds and angles, results in the substantial elimination of the landing hazards because practically no skill is required to accurately maneuver and land the airplane even on small landing areas and over the usual landing area border obstructions.

Another feature of the invention which contributes to the safety of landing and to the ease of pilot handling on the ground, is presented by my design and arrangement of the landing gear in which provision is made for absorbing the maximum vertical landing velocities and for preventing nosing over or "ground looping" under any conditions of landing that may be encountered; and further in which the landing surface engaging elements are arranged so that the landing gear is directionally stable and upon landing surface contact always causes the airplane to tend to follow the direction of landing instead of a path defined by the fore and aft axis or direction in which the airplane is headed, unless the direction of landing is along such axis.

A further feature of the invention is presented by the relative arrangement of the directionally stable landing gear, the body of the airplane and the lifting surface therefore by which the airplane in normal position supported on the ground by the landing gear is in substantially normal cruising flight attitude with the body in substantial horizontal position and the lifting surface is at a relatively small angle of incidence, that is, substantially the angle of incidence for cruising flight.

Another feature resides in the combination with the above arrangement of landing gear, body and lifting surface, of means for increasing the lift coefficient of the lifting surface for take off of the airplane without substantially changing the normal cruising flight attitude of the airplane with the body maintained in substantially horizontal position and the lifting surface at the relatively small angle of incidence, during take off of the airplane.

The conventional airplane essentially has three controls, that is, directional (rudder), lateral (ailerons), and longitudinal (elevator). Such three controls are required in order that the conventional airplane may operably meet all of the conditions encountered in flight operations, as for example, in landing in a crosswind, or in making a landing under conditions that necessitate side slipping the airplane. Generally, the directional or rudder control is operated by the pilot's feet, while the lateral and longitudinal controls are operated manually by the pilot, so that, the pilot must acquire the essential skill and ability to coordinate and synchronize the operation of these controls through his feet and hands. Experience has established the fact that such coordination is difficult to acquire and that crossing or improper coordination of the controls, particularly when flying close to the ground, is a frequent cause of serious accidents with the conventional airplane.

An important feature and a characteristic of this investigation is the provision of a basic design of airplane in which but two controls are required for flight operations, both in the air and in landing and taking off, under all conditions encountered, with the resulting elimination of the difficulties of coordinating and synchronizing three controls as in the conventional airplane, and if desired the elimination of a foot operated control to thereby avoid the necessity of coordinating not only a plurality of controls but also in coordinating and synchronizing the feet and hands in operating such controls.

An airplane design of my invention is further featured by lateral stability and lateral control for and throughout the entire range of speed and angles of attack, to thereby eliminate the dangerous characteristics encountered in the conventional airplane due to lateral instability at low speeds and angles of attack at and approaching the stall, and the insufficient lateral control for the conventional airplane under such flight conditions.

Another feature embodied in a design of airplane of this invention in combination and aerodynamic cooperation with the other features thereof, that contributes to the safety and reduces the piloting skill required, resides in eliminating that characteristic generally found in the conventional airplane of balance at higher angles of attack with power-on than with power-off to other by avoiding the possibility of losing altitude when it is desired to climb, while always insuring climb when full power is applied with the airplane below the maximum level flight speed.

A further general object and a feature of my invention is the provision for maximum range of vision for the pilot, especially in a forward and downward direction, in a design of airplane having the foregoing characteristics for safety by reducing the piloting skill required; and further in the provisions for comfort and reduction of fire hazard, and all of the foregoing in a design that is adapted for relatively low cost production and that is capable of minimum upkeep and operating costs in use.

With the above general features, characteristics and objects in view, as well as certain other features and characteristics that will appear and be readily recognized in the following description, by invention consists in certain novel features in design and in combination and arrangements of aerodynamic and structural elements and parts, all as will be more fully and particularly referred to and specified hereinafter.

As an example of one aerodynamic and structural design expression of the principles, features and characteristics of my invention, I have illustrated in the accompanying drawings, a small, light-weight and relatively low cost airplane of the two-place, high monoplane wing type, that is particularly adapted for general private owner and novice pilot use because of the inherent safety characteristics and low degree of piloting skill required to operate it with safety as compared to the conventional airplane. I have selected the illustrated design of airplane embodying the invention, primarily because an airplane of this general design and type embodying certain of the basic features hereof has been constructed and flight tested and the safety characteristics and the low degree of piloting skill required satisfactorily indicated.

However, there is no desire or intention to limit the invention in all and its various features and characteristics to embodiment in a design and type of airplane of the example hereof. It is recognized and intended that the invention can be embodied in and expressed by various other designs and types of airplanes, as will be apparent to those skilled in the art, and my present invention includes all of such embodiments and adaptations within the broad spirit and scope of the invention.

The design of the illustrated embodiment of my invention is of the high monoplane wing type having an outrigged tail and a pusher propeller and includes the body B, high monoplane wing W and the outrigged tail or empennage E carried from the wing W. The body B is of the closed cabin or nacelle type in which the occupants are enclosed in an upwardly extended, light-weight cabin portion C with a forward windshield and side window arrangement that affords particularly good vision out of the forward side windows for the pilot. The body B and its cabin portion C enclose two seats, in this instance, the forward one S of which is shown as providing the preferable pilot's seat from which the airplane is flown.

The wing W is mounted as and provides a high monoplane wing that extends across the upper or top side of the cabin structure C and above the main body B as defined by the portion of such body that extends forwardly of cabin structure C. The wing W terminates forwardly with it leading edge structure or portion spaced rearwardly from the forward end of body B, and also preferably as here shown, terminating at or short of the forward side of cabin structure C. The trailing edge portion of wing W is disposed preferably to the rear of or approximately at and above the rear or tail end of body B although the design is not essentially limited to such relative positions of body and wing. Preferably, the wing W has a decided dihedral angle for a purpose to be hereinafter explained.

Outrigger girders or spars 10 are mounted and supported from the wing structure W spaced from opposite sides, respectively, of body B, and extend rearwardly from the wing in substantially parallel relation. The rear ends of the outrigger girders 10 mount and carry thereon an empennage E, that consists in the present example of the spaced vertical fins or stabilizers 11 mounted on the spars 10 and extending thereabove and therebelow, the horizontal stabilizer 12 extending across and between spars 10, and the vertically swingable elevator 14 pivotally mounted along the trailing edge of stabilizer 12 and between girders 10 and 11.

A motor M is mounted on the rear portion of body B, preferably the upper portion thereof and along the fore and aft axis of the body, and drives a pusher propeller P which is disposed and positioned between the outrigger girders or spars 10, the central section of the wing W between the spars 10. The design thus presents an initial safety feature in locating the propeller P as surrounded and guarded by the wing and body at the forward side, the girders 10 at the opposite lateral sides and the empennage E at the rear side, so that injury from inadvertent contact with the propeller when the airplane is on the ground is practically eliminated.

The landing gear for the airplane, which in this example happens to be of the land type, consists of the spaced rear wheels 15 having a very wide tread and disposed at opposite sides of the rear portion of body B beneath wing W and aft of the center of gravity of the airplane, and the forward wheel 16 mounted at the forward end of body B along the longitudinal axis of the airplane and forward of its center of gravity. These wheels are each of the so-called "air wheel" type familiar in the art and capable of withstanding considerable side loads without failure. The arrangement of the wheels 15 and 16 of the landing gear relative to the body B and the lifting surface W is such that in normal position supported on the ground by the landing gear, the airplane is in substantially normal cruising flight attitude with the body B in substantially horizontal position with its longitudinal axis approximately parallel to the ground and with the lifting surface W at the relatively small angle of incidence for cruising flight. The mounting and relative arrangement and operation of this landing gear and the wheels 15 and 16 thereof form important features of my invention and will be referred to and explained in detail hereinafter.

Basically, according to an airplane design of my present invention, the landing and take off difficulties and hazards of the conventional airplane, are materially reduced by providing for a wide range of glide angles and a steep angle of climb. I attain the desired result by designing the wing W of a so-called "high lift" type having a high drag and by which the airplane can attain a steep angle of glide with a relatively low rate of vertical decent. The type of high lift wing W here selected embodies an auxiliary airfoil A fixed in a certain spaced relation forwardly of and along the leading edge of wing W, and by which as familiar to those skilled in the art, a high drag and lift can be obtained. Attention is specifically directed at this point to the fact that my invention is not limited or restricted to any particular form and type of "high-lift" wing, or wing to give the required increase in drag and lift, as wings of the "flap" or "slotted" or other suitable types may be employed if desired or found expedient.

For example, I have in the present instance, included trailing edge flaps F on the wing W of the manually operated and controlled type, and such flaps can be efficiently used in conjunction with the type of high lift wing W, or a wing of the automatically operating flap type can be substituted for wing W. In connection with the flaps F, I have purely diagrammatically and without regard to efficient location, illustrated a manual control for operating flaps F, which flaps are suitable connected together for simultaneous raising and lowering. The manual control may include the operating hand lever 17 in the body B accessible to the pilot, connected with and operating bell crank 18 by the push-pull tube 19, which bell crank is connected to the flaps F by the tubes 20 operatively connected to and coupled by a suitable bell crank in the wing. Primarily, in the design of this example, the manually controlled flaps F are provided for selective use by a pilot if he finds it is difficult to accustom himself to the normal action for the design of this example with its wing W, of pulling the control stick back to increase the angle of glide in the landing maneuver. By using the flaps F, the angle of glide can be satisfactorily adjusted in landing, through operation of the control level 17, as will be readily understood by one skilled in this art.

As lateral instability at high angles and low speeds is a primary danger in conventional airplanes, I have by my present invention insured lateral stability and control throughout the entire range of speeds and angles of attack which can be maintained in flight. I have accomplished this by providing sufficient longitudinal stability in the airplane, and by limiting the upward travel of the longitudinal control of elevator 14 to a point where the airplane not only cannot be maintained in a stall but also cannot be forced into a spin. Such longitudinal control limitation is fully discussed in United States Letters Patent No. 1,848,037 issued to me March 1, 1932, and it will suffice to here state that the upward travel of elevator 14 in the present design is limited in any suitable manner, such for example as disclosed in my aforesaid patent, to a point where the airplane cannot be maintained in a stall or forced into a spin. In the present design with the high lift wing W, the stall occurs at a high angle of attack, approximately 25°, and even though the upward travel of elevator 14 is insufficient to enable flying at the stall, ample elevator control is found present and available throughout the flight range.

The problem of lateral control at and beyond the stall, inherent in the conventional airplane, is eliminated from an airplane designed in accordance with this invention, because of the longitudinal control and stability relationship and the inability of the airplane to be flown or maintained at the stall. Therefore, conventional or other lateral control means may be employed for the design and satisfactory lateral control is insured therefrom at all speeds and angles of attack at which the airplane may be flown.

By the design and arrangement, in cooperation, of the high lift wing W and the limitation of upward travel of elevator 14, together with lateral stability and control throughout the range of speed and angles of attack, the airplane requires no particular skill to land it, other than maneuvering to contract the ground at the desired point. The wind range of gliding angles including a steep angle of glide make the landing approach an easy maneuver calling for no particular or special degree of skill. The airplane will itself practically take care of contact with the ground without particular attention on the part of the pilot, and if landed with its wings approximately level laterally at any speed within approximately 30 miles per hour of the minimum speed, the landing will be safe whether the airplane is leveled off before ground contact or continued in the glide straight to the ground, with the landing gear of the present example, that embodies certain important features of the present invention.

The landing gear as hereinbefore referred to includes the rear wheels 15 behind, and the forward wheel 16 ahead, of the center of gravity of the airplane and so arranged and mounted as to make it practically impossible for the airplane to nose over. The rear wheels 15 are preferably provided with the usual or any suitable brakes (not shown) but even with a full application of the brakes the arrangement of the landing gear is such as to prevent the airplane nosing over. The rear wheels 15 are disposed spaced a wide distance apart at opposite sides of and spaced from the body B toward the rear thereof and behind the center of gravity of the airplane. Each rear wheel 15 is mounted on a truss 21 extended laterally from the adjacent side of body B and mounted for swinging thereon to permit vertical movement of the wheel, and a long travel shock absorbing strut 22 extending between truss 21 and the wing W there above.

In accordance with the invention, the forward landing wheel 16 of the landing gear is mounted and arranged so as to be normally freely laterally swingable or castering for cooperation with the directionally fixed rear landing wheels 15 to provide the directionally stable landing gear for the airplane. I have disclosed herein one possible form of mounting and arrangement to attain the lateral swinging or castering operation of the forward landing wheel 16, in which example, the forward wheel 16 is carried by a long travel shock absorbing strut that includes the upper section 23 mounted in the nose or forward end of the structural frame of body B and the lower section 24 rotatable and also vertically movable in the upper section. The lower end of the strut section 24 is provided with the fork 24a in which the forward landing wheel 16 is mounted and by which the wheel is vertically movable and laterally swingable or rotatable to carry out its castering function. The shock absorbing strut 23—24 that mounts and carries forward wheel 16 is mounted on the body B in fixed position with its vertical axis inclined rearwardly, for example, a rearward inclination of approximately twenty degrees (20°) may be used. The forward wheel 16 is thus mounted and arranged so that the area of landing surface contact of the wheel is to the rear of the point at which the projection of the rearwardly inclined vertically disposed axis about which the wheel rotates meets the landing surface, and, as a result, this forward landing wheel will function to caster or rotate into the direction of travel. The normally freely castering forward landing wheel 16 cooperates with the directionally fixed real wheels 15 that are located aft of the center of gravity of the airplane, in such a manner that upon ground contact of the landing gear the castering front wheel 16 will caster or rotate into the direction of travel of the airplane. For example, in landing the airplane with such directionally stable landing gear, if the airplane is landing with side drift, then immediately upon ground contact of the gear, the front wheel 16 will caster or rotate into the direction of travel of the airplane and in cooperation with the directionally fixed rear wheels, automatically turn or head the side drifting airplane into the direction of travel.

Such a directionally stable landing gear also enables accurate handling of the airplane in taxiing on the ground and substantially eliminates any tendency of the airplane to ground loop.

The shock absorbing mountings for the front and rear wheels of the landing gear, consisting of the shock absorbing struts 22 and 23—24, have a long vertical travel to sustain and absorb the landing loads at the maximum vertical velocities of landing. As the rear wheels 15 sustain the largest load, their shock absorbing struts have a greater vertical travel than the strut 23—24 for the directionally stable front wheel 16. The maximum vertical positions of the landing wheels with the shock absorbing struts collapsed are shown in dotted lines while the lowered positions of the wheels with the struts extended are shown in full lines. In connection with the travel of the shock absorbing means and landing wheels, I have found that with an airplane of the invention weighing approximately 1150 pounds, the landing gear should be capable of withstanding a vertical velocity of about 25 feet per second, and an 18 inch vertical travel for rear wheel shock absorbers 22 with a 12 inch travel for front wheel shock absorber 23—24, should be satisfactory.

While in the specific example hereof, I have shown a three wheel landing gear with a single forward directionally stable wheel 16, it is to be clearly understood that my invention includes a plurality of forward directionally stable landing wheels, spaced as may be desirable, and with or without the disclosed arrangement of directionally fixed rear wheels. Also, attention is called to the fact that other landing surface engaging elements than wheels may be employed including skis, water landing members such as floats, pontoons and the like, as the invention is in no sense limited to ground engaging landing wheels.

As a further feature of the invention, the forward landing wheel 16 is made steerable for ground handling and taxiing of the airplane, and in the instant example, steering of the directionally stable front wheel 16 is carried out by means of a forwardly extended horizontally disposed arm 25 that is mounted for lateral swinging on a vertical shaft 26 on which it is mounted at its rear end. A brace or truss 25a is preferably mounted extended between the forward end of arm 25 and the lower end of shaft 26, the shaft 26 being of course suitable mounted for rotation around a vertical axis in fixed position in the body B. A rod or link 27 having a bifurcated forward end pivotally connected to the wheel fork 24a, extends rearwardly upwardly and freely slidably through a vertically disposed guide 28 mounted on the forward end laterally swingable arm 25. The guide 28 is pivotally mounted on the arm 25 for free rotation about a vertical axis.

By swinging arm 25 to the right or left the rod 27 is swung to rotate or turn the front, directionally stable landing wheel 16 to the right or left to steer the airplane when on the ground. The pivotal mounting of rod 27 to wheel fork 24a permits of free vertical travel of the shock absorber section 24, and the pivotal mounting of guide 28 permits free lateral swinging of arm 25 and 37 while operatively coupled. Steering operation or movement of arm 25 is carried out in the present example by and from the pilot's control system for the airplane, as will be described and explained hereafter.

An important feature of the invention made possible by the basic design and directionally stable landing gear, as hereinbefore described, is the use of but two controls by the pilot for complete flight, landing and take-off operation under all the varying conditions encountered in such operations. By this feature, either the rudder or directional control, or the aileron or lateral control of the conventional three-control system may be eliminated. In the preferred design and control arrangement hereo the airplane is provided with only a longitudinal or elevator control and a single control for changing the direction of flight, the conventional rudder control being eliminated. Such two-controls do away with the possibility of crossing controls and materially simplify the process of learning to fly, particularly eliminating the necessity for coordinating foot and manual control operating members.

The airplane is provided with the fixed, preferably adjustable, horizontal stabilizer 12 with the usual vertically swingable elevator 14 for longitudinal control. The usual rudder or directional control is eliminated and fixed vertical fins 11 of sufficient area for directional stability are provided carried at the tail of the airplane on the outrigger 10. Directional and lateral control, that is, control in yaw and roll, is obtained from the wing mounted opposite control surfaces 30, which in this instance are of the so-called spoiler type familiar in the art. Such spoilers 30 are mounted on opposite wings in the upper surface thereof and are differentially vertically swingable to raised position and to lowered position within the wing. The spoilers 20 give a yawing moment similar to that given by a rudder and also at the same time give a rolling moment for lateral control.

A pilot operated control system for operating the two control arrangement described includes a usual pivotally mounted control column 31, rockable or swingable fore and aft of the airplane for longitudinal control, and mounting at its upper end a control wheel 32 rotatable for directional control. The control column 31 is mounted in position for operation by the pilot from forward seat S and is pivotally mounted for fore and aft rocking about the pivot 31a. Control cables 33 and 34 are connected to column 31 and below pivot 31a, respectively, and are extended rearwardly over suitable pulleys to guide them along outrigger girders 10 to the upper and lower ends of the elevator horn 14a, so that, forward movement of the control column will lower the elevator and rearward movement thereof will raise the elevator.

The hand wheel 32 on the upper end of control column 31 rotates a drum 32a, to which the opposite cables 35 and 36 are connected and from which cables 35 and 36 extend and are guided over suitable pulleys to the inner or lower ends of the horns 37 of opposite spoilers 30, respectively, within wing W. The upper ends of the spoiler horns 37 above the wing are connected in the usual manner by a cable 38 guided over suitable pulleys into and through the wing. By rotating wheel 32 to the right and left spoilers 30 are differentially vertically swung to directionally control the airplane to the right or left and at the same time generate a rolling moment acting in the proper direction.

If desired, conventional ailerons can be used in place of the spoilers 30, but due to the improved yaw characteristics from the spoilers the latter are preferred.

With the directionally stable landing gear of the invention, and the other characteristics of the design giving the high drag and lift with steep angle of glide and low landing speed and short landing run together with lateral stability and control throughout the range of speed and angles of attack, the two-control system as described gives full control for flight and for landing and taking off under all conditions and even in cross winds and with side drift.

The directionally stable forward landing wheel is steerably connected into the pilot's control system and this is accomplished by connecting opposite sides of the swingable arm 25 with the opposite spoiler operating cables 35 and 36 by the cables 40 and 41, respectively. Cables 40 and 41 are connected to opposite sides of arm 25 and then extended around opposite pairs of pulleys and connected to cables 35 and 36 leading to and operated by hand wheel 32. Thus, as the hand wheel 32 is turned to the right or left with the airplane on the ground, the directionally stable landing gear 16 is turned to the right or left to steer the airplane in handling or taxiing in movement on the ground. Of course, a separate steering control can be provided for front wheel 16, or this wheel can be unconnected and self-turning or castering if desired, but preferably, the steerable and directionally stable landing gear wheel 16 is operatively coupled with the pilot's control for ground steering.

The two controls made possible by the design of airplane of the invention may, instead of the elevator and the wing mounted directional controls consist of tail mounted rudder or directional controls and the tail mounted elevator or longitudinal control. For instance I have purely diagrammatically illustrated an arrangement of a two-control system for the airplane of the present example, which provides the elevator 14 for longitudinal control and the rudders 45 swingably mounted in the usual manner along the trailing edges of the vertical fins 11 on the outrigger girders 10. The elevator 14 is operatively coupled to control column 31 by cables 33 and 34 while the rudders 45 are connected by cables 46 and 47 with the drum 32a of the control wheel 32. Cables 46 and 47 extend over and are guided by suitable pulleys, from wheel 32 to the spaced rudders 45 where they are connected to the respective rudder horns 45a. A cable or wire 48 connects the two rudders to operatively couple them for swinging by cables 46 and 47, as will be readily understood.

With the two control arrangement, sufficient control directionally and laterally is obtainable in cooperation with the other design features of the invention, if the wing W is given sufficient dihedral as here shown, for handling and maneuvering the airplane in the air, as well as in landing or taking off even in cross winds. As in the two-control system arrangement eliminates the foot operated control and the necessity of acquiring the technique and skill required for coordinating a foot and a manually operated control.

The two controls may consist of the elevator 14 providing the pitch control, and a single control for changing the direction of flight that consists of the opposite ailerons or roll control surfaces R and the vertical rudders 45 connected with the ailerons R through the ailerons operating mechanism or cables so as to be simultaneously operated with the ailerons. In the present example, the ailerons or roll control surfaces R of this single control for changing the direction of flight are of the more or less conventional trailing type mounted on the wing W at opposite sides of the longitudinal axis of the airplane and actuated by a control mechanism for differential operation in the more or less conventional manner.

Such an aileron operating mechanism may, for example, embody a horn or crank 50 fixed on and extending above and below each aileron R together with a cable 51 interconnecting the upper ends of the opposite aileron horns 50, the cable 51 being carried over and around suitable pulleys 51a at opposite sides of the wing. A cable 52 is connected to the lower end of the horn 50 of the right-hand aileron R and this cable 52 extends over suitable pulleys 52a to the drum 32a of the control wheel 32 on the upper end of the control column 31. A cable 53 is connected to the lower end of the horn 50 of the left-hand aileron R and extends around suitable pulleys 53a to the drum 32a of the control wheel 32 on the upper end of the control column 31. Thus, by turning control wheel 32 to the right or to the left, the opposite ailerons R are differentially displaced through the arrangement of control cables 51, 52, and 53, in the usual manner to obtain roll control for the airplane.

The vertical rudders 45 have the hereinbefore described operating cables 46 and 47 extended forwardly along the outrigger spars 10 and in the present form on two control systems of the invention, these rudder operating cables 46 and 47 are connected into and with the control cables 52 and 53, respectively, of the right and left wing ailerons R. For instance, rudder operating cable 47 for the right-hand rudder 45 after passing over suitable pulleys is connected to the cables 53 from the left wing aileron R, while the cable 46 from the left rudder 45, after passing over suitable pulleys is connected to the cable 52 from the right ring aileron R. Thus, the cables in the example hereof, form a means for connecting or coupling the rudders with the ailerons for simultaneous operation as a single control. If desired, any suitable detachable connecting means may be employed for coupling cables 46 and 47 to the cables 52 and 53, respectively.

The rudder operating cables 46 and 47 so connected into and with the aileron operating cables 52 and 53 thus provide for simultaneous operation of the rudders and ailerons for roll and yaw to thereby provide a wing control for changing the direction of flight of the airplane. This single control is, through the cables 52 and

53, operable from and by the steering or control wheel 32 of the pilot operated control unit which includes the control column 31. Rotation of the control wheel 32 to the right will lower the left wing aileron R and simultaneously swing or displace the vertical rudders 45 to the right, so that the direction of flight of the airplane will be changed to the right. When the control wheel 32 is rotated to the left, the reverse movements or displacements of the ailerons R and rudders 45 take place and the direction of flight of the airplane is changed towards the left.

The foregoing arrangement of two controls consisting of the control for pitch and the single control for roll and yaw comprising the combination of aileron or roll control surfaces and vertical rudders in combination with the directionally stable landing gear of the present invention as hereinbefore described and explained, enables complete control and maneuvering of the airplane for all normal conditions of take off, flight and landing, including the landing of the airplane with side drift, by the operation solely of such two controls.

In accordance with another feature of an airplane design of my invention, the airplane illustrated balances at a slightly lower angle of attack with the power on than with the power off. As the longitudinal control is limited to prevent sustained stalling with either power on or power off, the airplane will always climb if full power is applied with the airplane at speeds below the maximum level flight speed. In this manner, I have eliminated the possibility of the airplane losing altitude in straight flight at low speed with full power on. The general design of the illustrated example which provides the pusher propeller and a high line of thrust, contributes to balancing the airplane at a slightly lower angle of attack with power on than with power off, so that it becomes impossible to maintain a stalled attitude in either case.

In connection with the landing characteristics and the ground handling of the airplane, my design provides the relation between the landing gear and wing of the airplane such that the angle of incidence of the wing is approximately 0° when the airplane is at rest on the ground. In landing, therefore, as soon as the ground is contacted, the wing angle of attack is immediately reduced to 0° so that wing lift is reduced to a negligible amount to prevent any tendency of the airplane to float off the ground if landed above its minimum landing speed. This feature of the design also materially facilitates handling the airplane on the ground in high winds.

The invention provides a relation between the body B, the wing or lifting surface W and the rear wheels 15 and forward wheel 16 of the landing gear, such that with the airplane supported in normal position by said wheels on the ground, the body B is maintained in substantially horizontal attitude with its longitudinal axis approximately parallel to the ground. Thus, the airplane, when supported on the ground in normal position has the body B and the wing or lifting surface W in substantially normal cruising flight attitude with the occupants' seats in their normal attitude for natural seated position. The invention further provides for the maintenance of this normal cruising flight attitude of the airplane during take off, that is, with the body B in its substantially horizontal position and the wing or lifting surface W at its normal cruising flight angle of incidence.

The relation of the wing or lifting surface W to the landing wheels and the body B in normal ground attitude of the airplane is such that the wing or lifting surface W has an angle of incidence, which, for the particular wing used in the present example, will give substantially the lift coefficient used in normal cruising flight. For instance, with the particular wing or lifting surface W of the examples hereof, the angle of incidence of the wing when the airplane is in normal position supported on the ground by the landing wheels, that is, is in normal cruising flight attitude, is approximately zero degrees (0°). Hence, in normal cruising flight attitude of the airplane, the wing also has such approximately zero degree (0°) angle of incidence. Such an approximate zero degree (0°) angle of incidence, where, as in the example hereof, the angle of incidence of the wing with the airplane in normal ground position is the same as the angle of incidence of the wing in normal cruising flight attitude of the airplane is such an angle as will give the wing a lift coefficient of a certain percentage of the maximum lift coefficient for the wing. In the form of the wing W, having the auxiliary airfoil A, the wing, when at its approximately zero degree (0°) angle of incidence, has a lift coefficient approximately or of the order of one-fifth (1/5) of the maximum lift coefficient for such wing. On the other hand, the wing W, without the auxiliary airfoil A, when at the approximately zero degree (0°) angle of incidence, has a lift coefficient approximately or of the order of one-fourth (1/4) to one-third (1/3) of the maximum lift coefficient for such wing.

In terms of speed, such cruising angle of incidence range may be said to be such that in order for the wing to develop a sufficient lift to enable the airplane to take off, the airplane must travel at a rate of speed of at least of the order of fifty percent (50%) in excess of the minimum landing speed for the airplane.

When taking off with the airplane having the landing wheels, body and lifting surface so relatively arranged, the normal horizontal attitude of the body B and the above-referred to angles of incidence for the wing or lifting surface W may be maintained. However, due to the aforesaid relatively low angles of incidence, a relatively long take off run is necessary in order for the airplane to attain a speed necessary for the wing at such angles of incidence to develop a lift sufficient to take off the airplane. Under certain ground conditions, and particularly muddy or sandy conditions, it is difficult, or, at times, even impossible, to attain a sufficient speed for the airplane to develop a lift that will get the airplane off the ground. With the landing gear of the invention having the wheels 15 to the rear of the center of gravity, and particularly where the airplane may, for example, as in the case of an amphibian, have a high thrust line, it is practically impossible under the foregoing ground conditions to get the tail of the airplane down because in such case, the drag on the wheels gives a moment tending to press the nose of the airplane down. Thus, an increase in the angle of attack of the wing W by changing the attitude of the body B and the wing W in order to increase the lift during take off under such conditions can not be accomplished.

In accordance with a feature of my invention, means are provided in combination with the foregoing arrangement of the landing wheels, body and wing of the airplane, through the medium of which the lift of the wing W can be increased during take off without changing the normal cruising flight or ground supported attitude of the airplane and its body B and wing W. For instance, as one example of such a wing lift varying means, I have provided a lift varying wing flap F on the wing W together with the pilot controlled mechanism for operating this flap that includes the control lever 17 as hereinbefore described. Thus, during the take off of the airplane, the lift of the wing W can be arbitrarily selectively increased by the pilot to thereby enable the airplane to take off with a minimum of ground run and without changing the normal horizontal attitude of the body B or the normal cruising flight angle of incidence of the main or fixed portion of the wing or lifting surface W.

The various features of the invention are not necessarily limited in a design of airplanes to the inclusion therewith of the two-control system, as such features may be used to advantage with the conventional three-control system, and similarly the use of a conventional wing is not precluded in a design in which certain of the other features are incorporated, as such other features can still contribute advantages when embodied in a design of airplane having conventional wings. The directionally stable landing gear of my present invention is not restricted to use with an airplane embodying any of the other features and characteristics of the invention, but is of general use on airplanes of various other designs, including the conventional.

As a result of the principles, features and characteristics of my invention, an airplane designed to embody and incorporate them has a high degree of safety and requires very little skill to fly and to land and take off. Due to the wide range of gliding angles for the pilot to select from, together with the characteristics of low minimum gliding speed, lateral stability and control throughout the full speed and angle of attack range, and the inability to remain in the stall, the airplane can be easily and accurately landed on a very small space and practically by merely guiding the airplane to the field and letting it glide into contact with the ground. The combination with the foregoing characteristics of the directionally stable landing gear further simplifies and reduces the landing skill required and enables safe landings in cross winds. The necessity for but two controls made possible by the above features of the design, still further reduces the skill required to operate the airplane and makes learning to fly and operate the airplane a simple and rapid process. The operation of taking off is also rendered easy by the characteristics of the design which result in a short take off run and steep climb, thereby making it possible to easily take off from small fields and clear surrounding obstructions.

It is also evident that various other changes, modifications, variations, substitutions, eliminations and additions might be resorted to without departing from the spirit and the scope of my invention and hence I do not desire to limit my invention in all respects to the exact and specific disclosures hereof.

What I claim is:

1. In aircraft, the combination of a body, a lifting surface, directionally fixed

landing wheels to the rear of the center of gravity of the aircraft, a normally freely castering landing wheel forward of the aircraft center of gravity and adapted for cooperation with said directionally fixed landing wheels to provide a landing means for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing means when the aircraft is landing with side drift and means for controlling the aircraft in normal maneuvers in landing and flying comprising solely a pitch control and a control for changing the direction of flight.

2. In an airplane, in combination, a system of controls for the airplane consisting solely of a pitch control and a control for roll, and a landing and taxiing gear for the airplane that is normally directionally stable when supporting the airplane on a landing surface with the airplane moving forward, said landing gear embodying directionally fixed landing surface engaging means mounted on the airplane to the rear of the airplane center of gravity, and normally freely castering landing surface engaging means mounted on the airplane forward of the airplane center of gravity and adapted for cooperation with said directionally fixed landing surface engaging means to provide a landing gear for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing gear when the airplane is landing with said drift.

3. In an aircraft, the combination of a body, a lifting surface, directionally fixed landing surface engaging means to the rear of the center of gravity of the aircraft, a normally freely castering landing surface engaging means forward of the aircraft center of gravity and adapted for cooperation with said directionally fixed landing surface engaging means for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing means when the aircraft is landing with side drift, and means for controlling the aircraft in normal maneuvers in landing and flying comprising solely a pitch control and a control for changing the direction of flight that includes a swingable vertical rudder surface.

4. In an aircraft, in combination, directionally fixed landing surface engaging means to the rear of the center of gravity of the aircraft, a normally freely castering landing surface engaging means forward of the aircraft center of gravity, and adapted for cooperation with said directionally fixed landing surface engaging means for changing the aircraft heading to the direction of landing upon ground contact of said landing means when the aircraft is landing with side drift, and means for controlling the aircraft in normal maneuvers in landing and flying comprising solely a pitch control and one or more vertical rudder surfaces laterally swingable for changing the direction of flight.

5. In an aircraft, in combination, a body, a lifting surface, a directionally fixed landing surface engaging means to the rear of the center of gravity of the aircraft, a normally freely castering landing surface engaging means forward of the aircraft center of gravity and adapted for cooperation with said directionally fixed landing surface engaging means for changing the aircraft heading to the direction of landing upon ground contact of said landing means when the aircraft is landing with

side drift, and means for controlling the aircraft in normal maneuvers in landing and flying comprising a pitch control embodying a horizontally disposed elevator surface vertically swingable about a horizontal axis, and a vertical rudder surface laterally swingable for changing the direction of flight, said vertical rudder surface being located to the rear of the aircraft center of gravity.

6. In an aircraft, in combination, a directionally fixed landing surface engaging means to the rear of the center of gravity of the aircraft, a normally freely castering landing surface engaging means forward of the aircraft center of gravity and adapted for cooperation with said directionally fixed landing surface engaging means for changing the aircraft heading to the direction of landing upon ground contact of said landing means when the aircraft is landing with side drift, and means for controlling the aircraft in normal maneuvers in landing and flying comprising a pitch control consisting of a horizontally disposed elevator surface vertically movable about a horizontal axis and a control for changing the direction of flight that includes lateral control surfaces at opposite sides of the longitudinal axis of the aircraft.

7. In an aircraft, a directionally fixed landing surface engaging means to the rear of the center of gravity of the aircraft, castering landing surface engaging means forward of the aircraft center of gravity adapted for cooperation with said directionally fixed landing surface engaging means to provide a landing means for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing means when the aircraft is landing with side drift, in combination with, an air control system for the aircraft consisting solely of a pitch control and a single control only for both yawing and rolling control for the aircraft.

8. In an aircraft in combination, means for controlling the aircraft in normal maneuvers in landing and flying comprising solely a pitch control and a control for changing the direction of flight, a landing and taxiing gear for the aircraft embodying directionally fixed landing wheels to the rear of the center of gravity of the aircraft, and a castering landing wheel forward of the aircraft center of gravity adapted for cooperation with said directionally fixed landing wheels for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing gear the aircraft is landing with side drift, and pilot actuated means operatively connected with said forward castering wheel for arbitrarily swinging said wheel for steering the aircraft on a landing surface.

9. In an aircraft in combination, means for controlling the aircraft in normal maneuvers in landing and flying comprising solely a pitch control and a control for changing the direction of flight, a landing and taxiing gear for the aircraft embodying directionally fixed landing wheels to the rear of the center of gravity of the aircraft and a castering landing wheel forward of the aircraft center of gravity adapted for cooperation with said directionally fixed landing wheels for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing gear when the aircraft is landing with side drift, mechanism for actuating said direction of flight control, and said castering landing wheel being operatively connected with said control actuating mechanism whereby operation of said mechanism to actuate the direction of flight control swings said forward castering wheel to steer the aircraft when on a landing surface.

10. In an aircraft, in combination, means for controlling the aircraft in normal maneuvers in landing and flying comprising solely a pitch control and a control for changing the direction of flight, a single manually actuated control operating unit for operating both the pitch and the direction of flight controls, directionally fixed landing wheels to the rear of the aircraft center of gravity, a castering landing wheel forward of the aircraft center of gravity adapted for cooperation with said directionally fixed landing wheels to provide a landing means for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing wheels when the aircraft is landing with side drift, and said castering landing wheel being operatively connected to said single control operating unit so that selective operation of said unit to actuate the directional control swings said castering landing wheel for arbitrarily steering the aircraft on a landing surface.

11. In an aircraft, in combination, a system of air controls for the aircraft consisting solely of a pitch control and a single control only for both yawing and rolling control of the aircraft, a single manually operable control unit for operating both said pitch control and said single control for yaw and roll, a landing and taxiing gear for the aircraft comprising directionally fixed landing wheels to the rear of the aircraft center of gravity and a castering landing wheel forward of the aircraft center of gravity, and means operatively connecting said forward castering landing wheel with the manually operable control unit whereby selective operation of the latter to actuate said single control for yaw and roll swings said castering wheel for arbitrarily steering the aircraft on a landing surface.

In an aircraft, a body, a lifting surface, a landing gear for the aircraft, embodying directionally fixed landing wheels to the rear of the air craft center of gravity and a landing wheel forward of the air craft center of gravity, said wheels being so mounted and arranged that with the air craft supported thereby in normal position on the ground, said body is in substantially horizontally disposed attitude with its longitudinal axis approximately parallel to the ground, said lifting surface being so mounted relative to the body and to the landing wheels that with the aircraft in normal position supported on the ground the lifting surface has an angle of incidence of approximately zero degrees (0°), in combination with means for increasing the lift coefficient of said lifting surface when the aircraft is moving forwardly in normal position supported on the ground by said landing wheels without changing the normal horizontal attitude of said body or the aforesaid normal angle of incidence of said lifting surface.

13. In an aircraft, in combination, a body, a lifting surface, a landing gear for the aircraft consisting of directionally fixed wheels to the rear of the aircraft center of gravity and a castering landing wheel forward of the aircraft center of gravity, said wheels being so mounted and arranged relative to the body and lifting surface

that with the aircraft supported by the wheels in normal position on the ground, said body is in substantially horizontally disposed attitude with its longitudinal axis approximately parallel to the ground, said lifting surface being mounted relative to the body and to the landing wheels so that with the aircraft in normal position supported on the ground the lifting surface had an angle of incidence such that the lift coefficient for the lifting surface at such incidence angle does not exceed approximately one-third (1/3) of the maximum lift coefficient for such lifting surface, means for increasing the lift coefficient of said lifting surface when the aircraft is moving forwardly in normal position supported on the ground by said landing wheels without changing the normal substantially horizontal attitude of said body or the aforesaid normal angle of incidence of said wing, and mechanism under the control of the pilot for arbitrarily selectively operating said lift increasing means.

14. In an aircraft, in combination, a body, a landing gear for the aircraft embodying landing wheels respectively spaced forward of and to the rear of the aircraft center of gravity, a lifting surface for the aircraft, said landing wheels being so mounted and arranged relative to said body and to said lifting surface that when the aircraft is supported by said wheels in normal position on the landing surface, said body is in substantially horizontally disposed normal cruising flight attitude and said lifting surface is at an angle of incidence such that the lift coefficient for the lifting surface at such angle does not exceed approximately one-third (1/3) of the maximum lift coefficient for such lifting surface, a flap member mounted on said lifting surface for movement to positions increasing the lift coefficient of said surface, and pilot actuated mechanism for moving said flap member whereby the lift coefficient of said lifting surface can be increased for the take off run of said aircraft to thereby enable the aircraft to take off without changing the normal horizontal attitude of the aircraft body or the normal angle of incidence of the main portion of said lifting surface.

15. In an aircraft, in combination, a body, a landing gear for the aircraft comprising directionally fixed landing wheels to the rear of the aircraft center of gravity, a normally freely castering landing wheel forward of the aircraft center of gravity adapted for cooperation with the said directionally fixed landing wheels for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing gear when the aircraft is landing with side drift, a lifting surface for the aircraft, said directionally fixed and said castering landing wheels being so mounted and arranged relative to said body and to said lifting surface that when the aircraft is supported by said wheels in normal position on the ground, said body is in substantially horizontally disposed attitude and said lifting surface is at an angle of incidence such that it requires the aircraft to travel at a rate of speed of over approximately fifty percent (50%) in excess of its minimum landing speed in order to take off, a flap member mounted on said lifting surface for movement from normal position to positions increasing the lift of said surface, and pilot actuated mechanism for selectively arbitrarily moving said flap member to a lift increasing

position during the take off run of said aircraft without changing the normal horizontal attitude of the aircraft body or the normal angle of incidence of said lifting surface to thereby enable the aircraft to take off at a rate of travel less than that required for the lifting surface at such normal angle of incidence and with the flap in normal position.

16. In an aircraft, a body, a lifting surface, directionally fixed landing wheels to the rear of the center of gravity of the aircraft, normally freely castering landing wheels forward of the aircraft center of gravity adapted for cooperation with said directionally fixed landing wheels for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing wheels when the aircraft is landing with side drift, in combination with, air controls comprising a control for pitch embodying an elevator surface vertically displaceable about a horizontal axis, a control for roll embodying control surfaces at opposite sides of the longitudinal axis of the aircraft, and a control for yaw embodying a laterally swingable vertical rudder surface, means for connecting the vertical rudder surface and said roll control surfaces together for simultaneous operation only, as a single combined yaw and roll control, and a manually operable control unit operatively connected with said combined yaw and roll control, whereby the aircraft can be landed with side drift solely by the operation of said pitch control and the said combined yaw and roll control.

17. In an aircraft, a body, a lifting surface, directionally fixed landing wheels to the rear of the aircraft center of gravity, a normally freely castering landing wheel forward of the aircraft center of gravity adapted for cooperation with said directionally fixed landing wheels for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing wheels when the aircraft is landing with side drift, in combination with an air control system comprising solely a control for pitch and a control for changing the direction of flight, said direction of flight control embodying a laterally swingable vertical rudder surface and roll control surfaces at opposite sides of the longitudinal axis of the aircraft, means for connecting said vertical rudder surface and said roll control surfaces for simultaneous operation only, as a single control, and a single manually operated control unit connected with said pitch control and with said control for changing the direction of flight whereby the aircraft can be landed with side drift solely by operation of said pitch control and said direction of flight control.

FRED E. WEICK

Document 3-16(b), Fred E. Weick & Associates, Inc., to Waco Aircraft Co., Troy OH, 8 Apr. 1938.

April 8, 1938.

Timm Aircraft Corporation, Waco Aircraft Company
Glendale, California; Troy, Ohio

Gentlemen:

We see from the aeronautical magazines that your company is manufacturing an airplane incorporating both a tricycle landing gear and a wing flap.

U.S. Patent No. 2,110,516 has just been granted us for this combination, as well as for the combination of a tricycle gear with two air controls. Both of these combinations were developed on our W-1 airplane, described in various publications (Aviation for July, 1934, and January, 1936). A copy of the patent is enclosed for your information.

We plan to issue a license to any manufacturer who desires to use the features incorporated in this patent, and will shortly send you a license agreement for your consideration. We have worked out a royalty rate that is sufficiently low to avoid burdening the industry or retarding its development.

We shall be glad to answer any questions you may have on this subject.

Very truly yours,

FRED E. WEICK & ASSOCIATES, INC.

Document 3-16(c), Weick & Associates, Inc., to Waco Aircraft Co., Troy, OH, 2 Feb. 1939.

February 2, 1939.

The Waco Aircraft Company
Troy, Ohio
Attention: Mr. C. J. Brukner, President

Gentlemen:

Last July we forwarded you two copies of a license agreement in connection with U.S. Patent No. 2,110,516, which agreement we requested you to consider.

Since that time, our President, Mr. Weick, has spoken with you twice, at which time you indicated that you might be willing to sign this agreement, but that you did not consider your present airplane as infringing on the patent under consideration. In view of this stand, will you give us in detail the reasons why you believe your airplane does not infringe of the patent.

We would appreciate a reply at your early convenience.

Very truly yours,

FRED E. WEICK & ASSOCIATES, INC.

Document 3-16(d), C. J. Brukner, President, The Waco Aircraft Company, Troy, OH, to Fred E. Weick and Associates, Inc., Evelyn Place, College Park, MD, 16 Feb. 1939.

Feb. 16, 1939

Fred E. Weick and Associates, Inc
Evelyn Place,
College Park, Maryland

Att. R. Sanders, Exec, Vice-President

Gentlemen:

Your letter of February 2nd indicates a confused impression of the discussions the writer has had with your President, Mr. Fred Weick, on the subject of Patent No. 2,110,516. Since, naturally, there would be no occasion for our signing an agreement with you until we began, or intended to begin, to utilize the patent in our production, we analyzed our tricycle model airplane with each claim of this patent and reached the conclusion that we were not utilizing the same.

Since there are several claims that contain the term, "approximately," we recognized the possibility of our interpretation of this comparative word being different from your own, and we were glad for the opportunity afforded by Mr. Weick's visit to Dayton last summer to obtain his views as to how far away from the fractions given in the claim he considered "approximately" to reach. He expressed himself on this subject, and we gave him our data, which fell comfortably outside his estimated range, and we therefore seemed to agree that our airplane did not go far enough in the direction of your patent to fall within its scope.

We assured Mr. Weick that we felt that our trend would be in the direction of your patent rather than away from it, but it has not been so far, and, therefore, the status remains the same.

We did not supply Mr. Weick with written evidence of our design details, but volunteered to show him our pertinent data if he would visit our plant on the next day after our discussion, or to fly the airplane over to Wright Field where he was then occupied, and where his extreme familiarity with the subject would enable him to verify our representations which had formed the basis of our discussion. Mr. Weick phoned on the next day that he would be unable to give time to either of these routines.

In our recent discussion with Mr. Weick on this subject, we stated that there is a possibility that we might make a design change which would bring this airplane within the scope of your patent; and that in such case we would notify you of our desire to negotiate on the subject of a license, your proposal of which we have in our file for consideration.

For your further information—there is one of our tricycle airplanes in the possession of the C.A.A., at present situated near Washington on radio and blind landing experiments, which could, no doubt, be inspected by your personnel; and if you will advise which one of the seventeen claims of your patent, in your opinion, most nearly reads on this airplane, we will supply you with whatever data will assist in a definite determination of the question.

Yours very truly,

THE WACO AIRCRAFT COMPANY,

C. J. Brukner,
President.

Document 3-16(e), Weick & Associates, Inc., to The Waco Aircraft Company, Troy, OH, Attn: Mr. C. J. Brukner, President, 22 March 1939.

March 22, 1939.

The Waco Aircraft Company
Troy, Ohio
Attention: Mr. C. J. Brukner, President

Gentlemen:

We thank you for your letter of February 16th, giving us your impressions of the conclusions reached between yourself and Mr. Weick in your recent interview.

Mr. Weick is not in full agreement with all your conclusions. However, in view of your stand, we will let the matter await further developments as we are convinced that you will shortly come to the conclusion that you must make certain changes in your airplane which will place it definitely under the claims of our patent.

Very truly yours,

FRED E. WEICK & ASSOCIATES, INC.

Document 3-16(f), Weick & Associates, to Timm Aircraft Corporation, Grand Central Air Terminal, 1020 Airway, Glendale, CA, Attn: Mr. R. A. Powell, Vice-President, 22 March 1939.

March 22, 1939.

Timm Aircraft Corporation
Grand Central Air Terminal, 1020 Airway
Glendale, California
Attention: Mr. R. A. Powell, Vice-President

Gentlemen:

We acknowledge receipt of your letters of February 7th and 13th, advising us that you do not believe that your airplane comes under the various claims in our patent, and also advising us that you had been working on your steering mechanism prior to the date of the filling of the patent.

With regard to the patent date, the patent issued was merely a reapplication of an earlier patent filed in January, 1934, and, since we established a date much earlier than 1934 in connection with an infringement which we had to fight prior to obtaining our patent, we believe that we pre-date any work which you did in this direction.

With respect to the steering of the nose wheel, you will note that our patent claims cover the steerable nose wheel as well as the freely castering nose wheel. However, our claims as to the flap used in take-off refer to a combination of wheels which tends to reduce the angle of attack to less than its maximum when the airplane is on the ground, and do not take into account the method of locking or controlling the wheels.

Although you state that you have no difficulty in taking off without resorting to flaps, the use of flaps on take-off will greatly reduce the run of such an airplane, particularly under adverse field conditions, and, if your airplane is capable of taking advantage of this feature, it falls under the claims of our patent.

In view of the foregoing, we still contend that your airplane infringes on one or more of the claims of our patent and we, therefore, request that you enter into a license agreement, a copy of which we enclose herewith for your consideration.

Very truly yours,

FRED E. WEICK & ASSOCIATES, INC.

Document 3-16(g), Alpheus Barnes, The Wright Company, 11 Pine Street, New York, to Mr. Grover C. Loening, c/o U.S.S.* Mississippi*, Pensacola, FL, 16 Jan. 1915.

The Wright Company
11 Pine Street
New York, Jan. 16, 1914

Mr. Grover C. Loening
c/o U.S.S. Mississippi
Pensacola, Florida

My dear Loening:

No doubt you have read or heard of the decision of the Court in our favor. I enclose a copy of the decision so that you can see for yourself it is quite true.

Our company will control the manufacture and sale of machines in the United States absolutely.

Am off for Dayton Sunday and look forward to a very pleasant visit there as I feel sure things will now begin to hum.

Yours sincerely,

Alpheus Barnes

Document 3-16(h), Grover Loening, "What Might Have Been," in* Our Wings Grow Faster *(1935).

To understand what follows, a little insight into Orville Wright's personality is needed.

To begin with, let me make it clear that Orville is a scientist and a real engineer. On a flying field I once found Glenn Curtiss standing near one of his new planes. I asked him a simple question about the approximate area of a tail surface. Curtiss answered, "Oh, I don't know, but if it isn't right, the boys will fix it." And in that answer is the evidence that Curtiss was a promoter and not an engineer or even his own designer, excepting in a vague way. But if Orville was asked a similar question, he would bring out of his pocket a little memorandum book he always carried and tell you exactly, not approximately, the figures inquired about. He directed all the design work in the shop, even to small metal fittings, and many a time I had designed some detail and made a fine drawing of it, only to find that meanwhile

Orville had gone into the shop and, with one of his old trusted mechanics, such as Charley Taylor or Jim Jacobs, he would not only have designed the part, but had it made right there.

Factory organization was pretty difficult, but things got done as long as Orville was well and on the job. On the other hand, he would delay making an important decision and drive us all nuts trying not to disobey his orders on the one hand and yet not knowing what to do next.

The treasurer of the Wright company at this time was Alpheus Barnes of New York, who was at Dayton most of the time, and really also represented the New York group of directors and stockholders, Andrew Freedman, August Belmont, Cornelius Vanderbilt, Robert Collier, etc.

Barnes did all the bookkeeping work as well as advertising and negotiating of contracts and left the management of the plant pretty much up to me. He was a hearty, well-built, and genial character, smoking cigars continually and full of good stories. I quickly saw that Orville really didn't like him, but that did not bother Barnes much.

Orville was good at business, I thought, in that very few people could put anything over on him, but he certainly did not have any "big business" ideas or any great ambition to expand. He seemed to be lacking in push—which I attributed to two causes: his loss of Wilbur, and his becoming increasingly ill. In addition Orville and his sister Katherine had, preying on their minds and characters, the one great hate and obsession, the patent fight with Curtiss. It was a constant subject of conversation, and the effort of Curtiss and his group to take credit away from the Wrights was a bitter thing to stand for. There was a good deal of justification for it. Once Orville showed me the letter Curtiss wrote in 1907 or thereabout, agreeing "never to use for commercial purposes" the priceless data on wing surfaces, balance, etc., which the Wrights let him have for the use of a non-profit, scientific body known as the Aërial Experiment Association, fathered by Alexander Graham Bell—and from which Curtiss presently transformed his activities into an airplane company.

Orville sued Curtiss for revenge and prestige.

But the New York bankers interested in the Wright company sued Curtiss because they wanted to establish a monopoly. And Barnes would daydream with me of what it would mean if the Wright company would win the suit.

Suddenly one day the decision was announced. The Wright patent on the fundamentals of the airplane was valid, and the Herring-Curtiss Company were found guilt of gross infringement.

There was great excitement and anticipation, because the Wright company could legally have entered upon a totally different policy and could then and there have begun a monopoly of the field that would have been no less all-embracing than that of the Telephone Company. There were untold millions of dollars ready in New

York to be invested in such a trust. In no time Curtiss and what other companies there were could have been closed down or bought up, and we would have seen a totally different development of flying starting here and spreading to Europe exactly as the telephone monopoly did. When we look back on it, it might have been a better thing for aviation. Many destructive rivalries would have been stopped, and with the World War just getting ready to start, one hesitates to think what different a great rich legal trust might have made.

At any rate it did not happen because of one man—Orville Wright. With the winning of the suit, his revenge on Curtiss seemed satisfied, and all he wanted was tribute—royalties from everyone. He did not want to expand. He fought the New York interests. The antagonism between him and Barnes became acute. And the main thing was that nothing much was done, because Orville would not make decisions. He was a sick man, and few of us knew it.

Document 3-17(a-b)

(a) A. L. Klein, "Effect of Fillets on Wing-Fuselage Interference," *Transactions of the American Society of Mechanical Engineers* 56 (1934): 1-10.

(b) James A. White and Manly J. Hood, "Wing-Fuselage Interference, Tail Buffeting, and Air Flow About the Tail of a Low-Wing Monoplane," NACA *Technical Report* 482 (Washington, 1934).

The first major explanation of the value of wing fillets came in June 1932 at a meeting of the Aeronautics Division of the American Society of Mechanical Engineers at Berkeley, California. Arthur L. Klein, a professor of aeronautics at Caltech, explained how the addition of fillets removed virtually all of the aerodynamic disadvantages of low-wing monoplanes, making them just as efficient as high-wing designs. Klein's address, reproduced here in its entirety complete with the discussion that followed, marked the appearance of yet another "shelf item" in the reinvention of the airplane. His findings, based on GALCIT tests of the Northrop Alpha, made it clear that a smooth "fairing" of the joint formed where the wing met the fuselage would greatly improve not just the efficiency, but also the handling characteristics, of low-wing monoplanes.

As shown in the second document below, NACA research supported Caltech's findings. Not only would fillets help in the case of low-wing monoplanes, they would also aid more generally with buffeting and interference problems experienced by aircraft. The NACA tests being discussed in this document involved a series of tests in Langley's Full-Scale Tunnel on a small monoplane known as the "Doodlebug." The designer of the Doodlebug, James S. McDonnell, built the diminutive monoplane in hopes of winning the $100,000 prize in the Daniel Guggenheim International Safe Airplane Competition of 1929, which was based on the notion of producing a "Model T of the Air." The Doodlebug actually never entered the contest as it was damaged in a forced landing on the way to the contest. Anyway, the Great Depression effectively nixed the market for popular light planes. McDonnell approached the NACA with the idea of its buying the aircraft, saying that it would make an interesting subject for a number of difficult experiments. The NACA did purchase it, for $5000 (McDonnell's total venture had cost about $30,000, mostly financed by Philip Wrigley of chewing gum fame), and it made many tests with it both in the FST and in flight. The NACA engineers found that with wing fillets and the addition of an NACA cowling (specifically designed by Fred Weick), the little airplane experienced greatly reduced buffeting and aerodynamic interference.

The Doodlebug was a very effective and progressive design. It had Handley Page automatic leading edge slots, slotted flaps, and long-travel shock absorbers. As intended, it could fly in and out of very small places. At the 1930 National Air Races in Chicago, McDonnell put on a remarkable exhibition with the airplane. Right in front of the grandstand he staked out a circle 150 feet in diameter with little flags. He would take off from within the circle, fly around, and land with his entire landing run within the circle also.

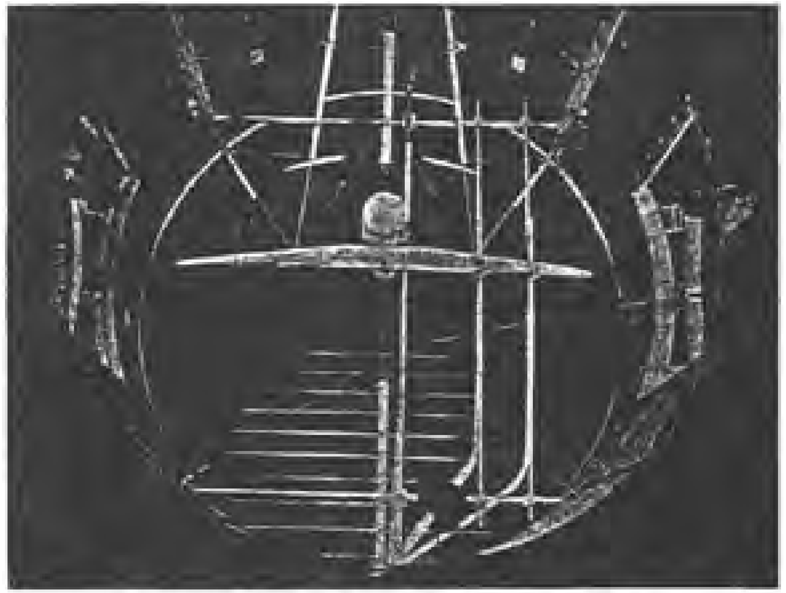


Fig. 1 View of Model as a High-Wing Monoplane in the Tunnel
(The three pitot-tube combs can be seen in the background.)

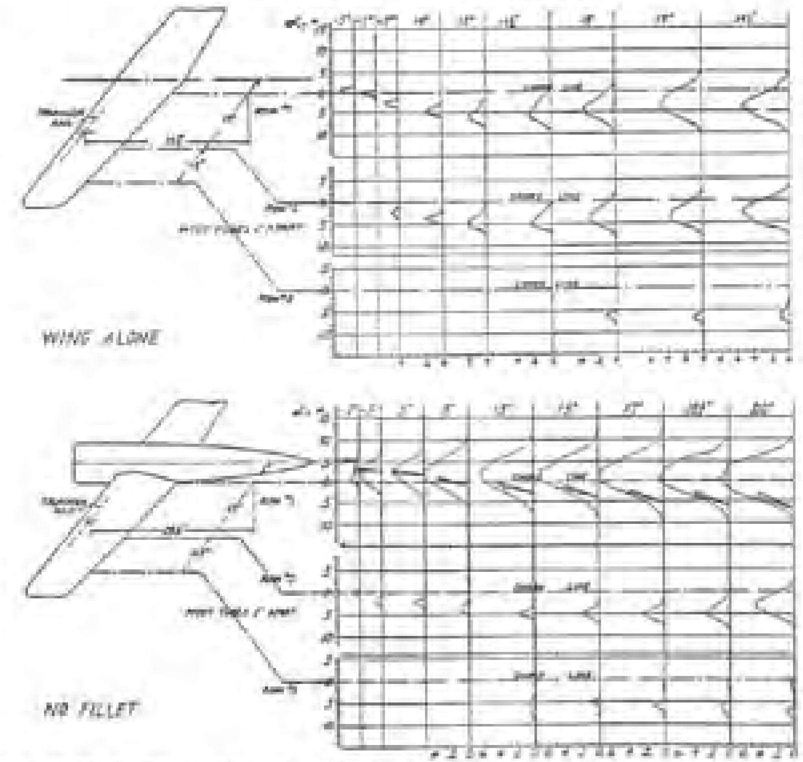


Fig. 2 Wake Distribution Behind the Wing Alone and the Wing and Fuselage With No Fillet
(The area between the vertical lines represents the energy loss.)

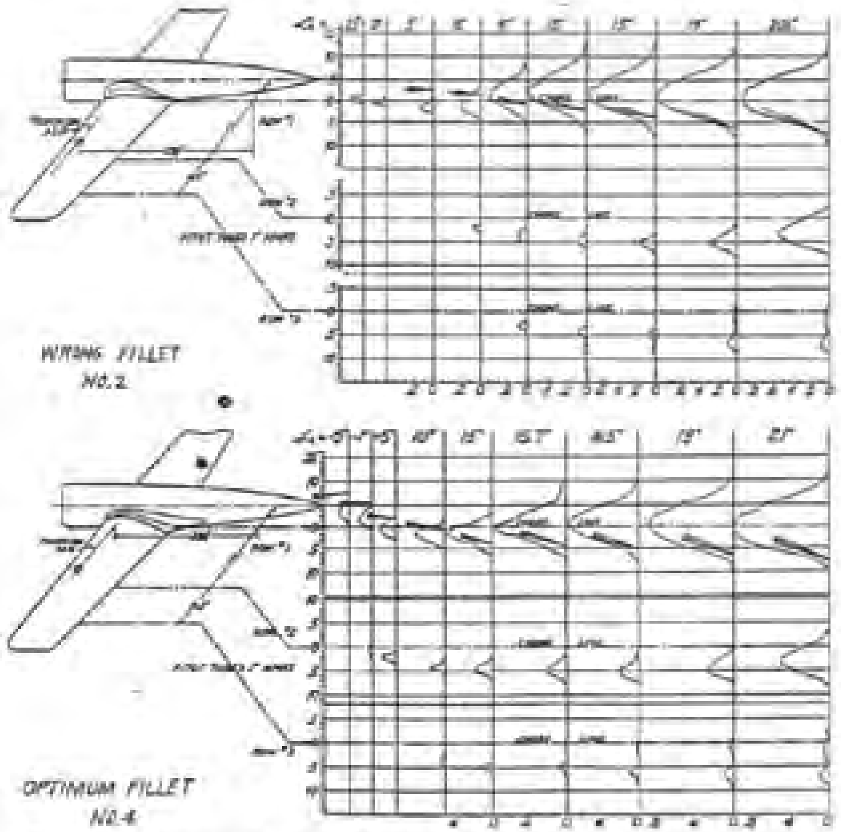


Fig. 3 Wake Distribution Behind Fillet No. 2 (Wrong Fillet) and Optimum Fillet No. 4

Fig. 4 The Wrong Fillet No. 2
(Note that the model is inverted and the pitot tubes can be seen in the background.)

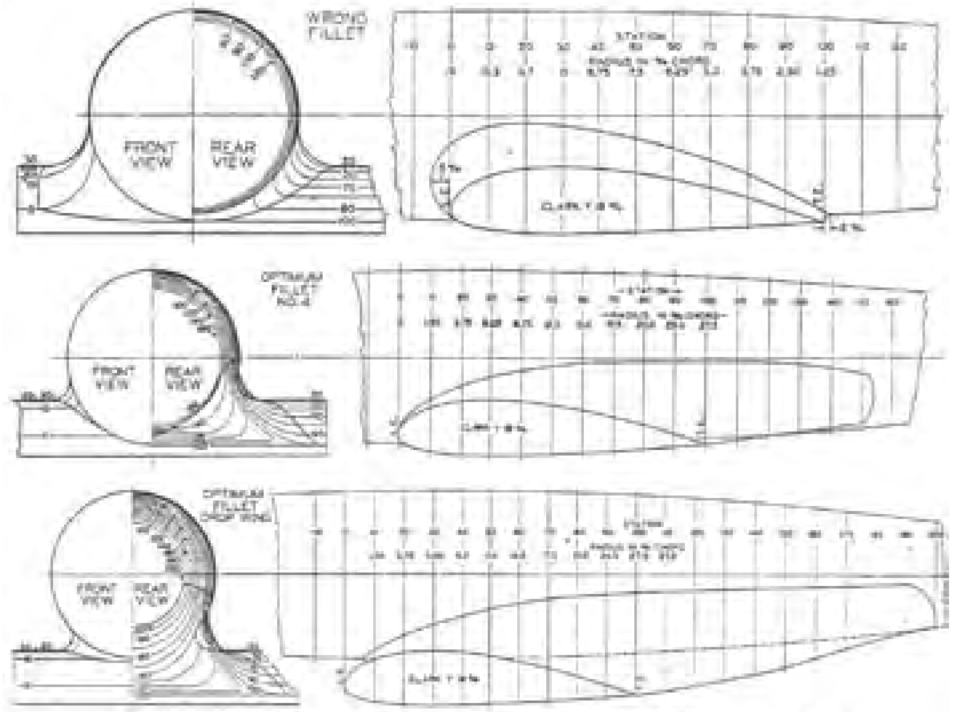


Fig. 5 Lines of the Wrong Fillet No. 2, Optimum Fillet No. 4, and Fillet D on the Dropped Wing

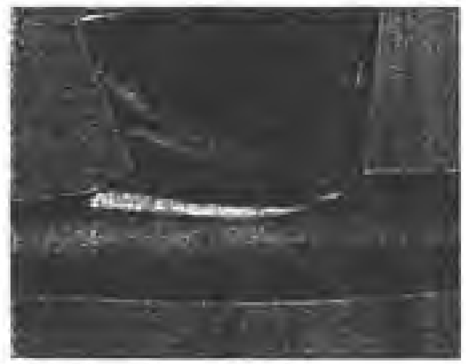


Fig. 6 Fillet No. 3

Fig. 7 Wrong Fillet No. 4

AERONAUTICAL ENGINEERING AER-56-1 5

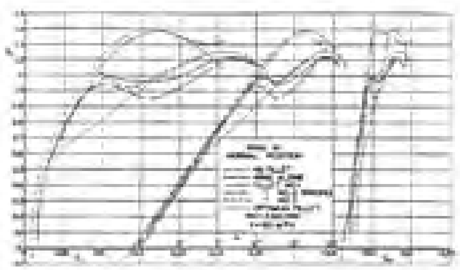


Fig. 8 Normal Low-Wing Configuration

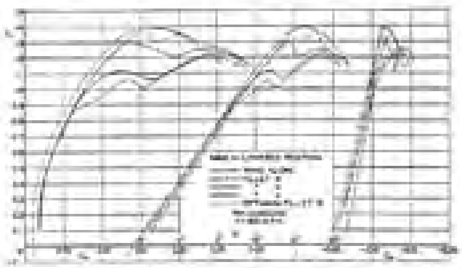


Fig. 9 Dropped-Wing Configuration

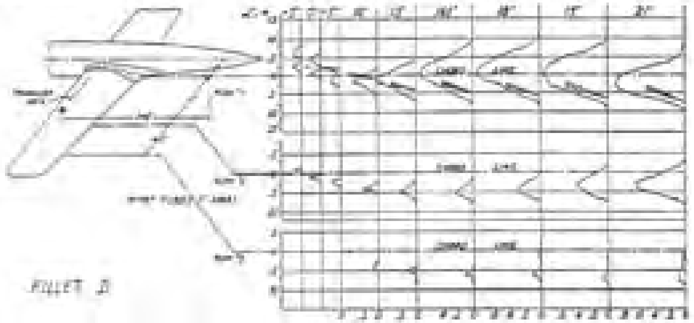


Fig. 10 Wake Diagram for Optimum Fillet D

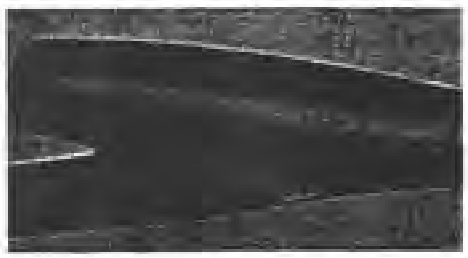


Fig. 11 Optimum Fillet D (Wing Lowered)

Fig. 12 Optimum Fillet D (Wing Lowered)

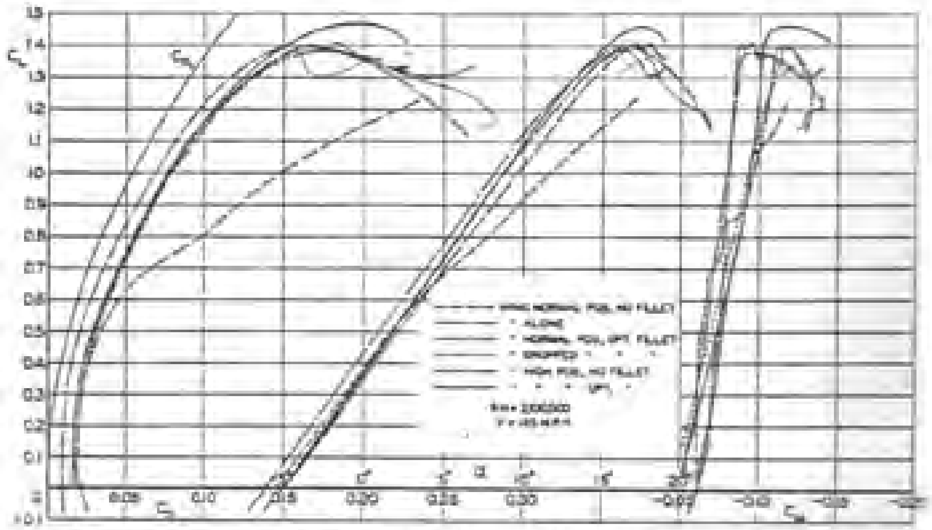


Fig. 13 Curves for Various Configurations

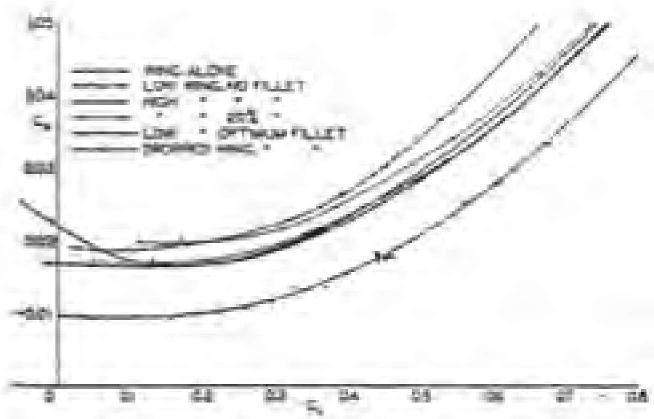


Fig. 14 Comparison of Various Configurations in the High-Speed Region Showing the Experimental Scatter

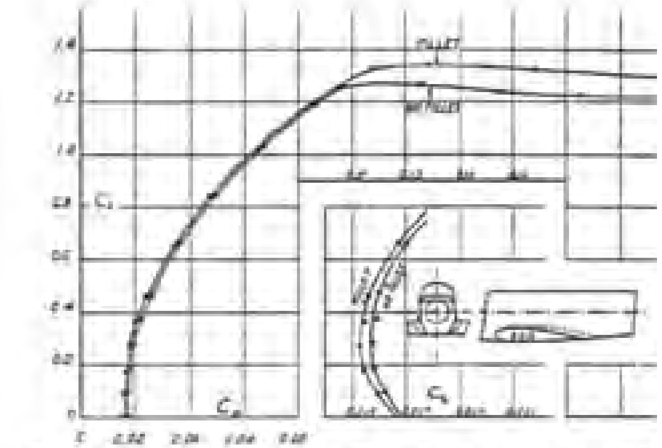


FIG. 15 LIFT AND DRAG POLAR CURVES FOR THE CASE OF A F[...]
SELAGE WING COMBINATION IN WHICH THE FUSELAGE IS STRAIG[...]

Document 3-17(a), A. L. Klein, "Effect of Fillets on Wing-Fuselage Interference," Transactions of the American Society of Mechanical Engineers _56 (1934): 1-10._

EFFECT OF FILLETS ON WING-FUSELAGE INTERFERENCE
By A. L. Klein, Pasadena, Calif.

Mutual interference of wing and fuselage has been the subject of many previous investigations, but these have been on a very small scale. A comprehensive investigation into the case of a low-wing monoplane was thought to have important possibilities, especially as this type of airplane was known to have some aerodynamical peculiarities. In a preceding investigation it had been found that the addition of large fillets to the intersection of wing and fuselage would cause a great improvement and it was decided to extend these tests and to include the case of a high-wing monoplane for comparison.

Numerous investigations have been made on the mutual interference of wings and fuselages, but much of the preceding work has been done on a very small scale. It was thought that a comprehensive investigation into the case of the low-wing monoplane might be of importance, especially as this type of airplane was known to have some aerodynamical peculiarities. In a preceding investigation it had been found that the addition of large fillets to the wing-fuselage intersection would cause a great improvement; it was therefore decided to extend these tests and to include the case of the high-wing monoplane for comparison.

The model used for the investigation was a 1/6-scale model of the Northrop "Alpha," a Wasp-engined transport plane of approximately 5000 lb gross weight. The model had a span of 7 ft and a length of 52 in. The airfoil section was 19 percent Clark-Y at the root and 12 percent at the tip. The wing area of the model was 8.33 sq ft, and the root chord of the wing was 16.67 in. The wing was mounted with reference to the fuselage as shown in Fig. 5. The model as used consisted of a wing and fuselage only. This model was presented to the laboratory by the Northrop Aircraft Corporation, a unit of United Aircraft and Transport Company.

EXPERIMENTAL METHODS

The investigation was carried on by two distinct methods. The set-up for force measurements was identical with that described in ref. 2, while in addition three combs of pitot tubes were mounted behind the model for exploration of the wake behind the wing. These combs can be seen in Fig. 1, and also some of the pitot tubes can be seen in Figs. 6 and 7. The total pressure orifices were made of 1/8-in. brass tubing mounted on a steel tube. For static-pressure measurements, jackets containing side orifices were slipped over the total-pressure tubes. These tubes were connected to the multiple manometer, and observations were made of the wake distribution behind the set-up for various angles of attack. Records of the wake observations were made by taking shadowgraphs of the manometer on ozalid paper. These records were then reduced by plotting curves showing the loss of total head as percentage of the total head in the free stream. Figs. 2, 3, and 10 show curves of this type, and also show the relative position of the model and the combs. The position of the stabilizer and elevator for each angle of attack is indicated on the curves. The fillets used in the investigation were built up from physicists' soft wax, a compound of beeswax, Venice turpentine, and rosin, and modeled to templets. It was found that an almost glasslike surface could be given this wax by rubbing it with sandpaper dipped in kerosene.

WAKE OBSERVATIONS

Fig. 2 shows the wake losses behind the wing tested alone. Noting first the wake losses in row 1, it is seen that as the angle of attack increases, the center of the wake moves downward, as one would expect. When the wing reaches an angle of attack of 15 deg, the center of the wake has reached its lowest point, near which it remains until the angle increases to 16.5 deg, after which the wake moves noticeably upward. If one now looks at the force-measured curves for the wing alone (Fig. 8), one sees that the angle for maximum lift is 16.5 deg. In row 2 the behavior is similar to row 1; the downwash causes the wake to go downward until the angle of maximum lift is reached, after which the wake moves upward. In the case of row 3 an anomaly occurs; the downwash does not reach the maximum values at 16.5 deg, but continues to increase, and apparently stalling does not occur at this section of the wing until an angle of attack of approximately 20.5 deg is reached. The wake curves for small angles in rows 2 and 3 are omitted because the wake was smaller than the distance between the pitot tubes, so that no deductions could be drawn as to its magnitude and shape.

Below on the same figure is a similar set of observations for the model assembles as a low-wing monoplane with no fillet at the intersection between the wing and the fuselage. The maximum diameter of the fuselage was 9.5 in. Pitot comb 1 was mounted approximately halfway between comb 2 and the wing tip. The positions of the stabilizer and elevator are drawn in on the successive curves. It will be noticed that the wake curves for row 1 differ completely from those for the wing alone, while the wake curves in rows 2 and 3 are hardly distinguishable from those for the wing alone. The wake curves in row 1 are not only larger in area than those for the wing alone, but also the position of their center line indicates a much smaller downwash. It will be noticed that the center line of the wake never goes below the chord line, while for the case of the wing alone the center of the wake moves down below the chord line at small angles and remains there. These two conditions represent the limiting cases of the present investigation, the wing alone giving the condition for zero interference and the wing and fuselage with no fillet the case of maximum interference.

Now consider the case of a fillet that partly remedies the interference effect. Fig. 3 shows the wake distribution for what will be called the "wrong fillet." Fig. 4 is a photograph of this fillet, and its lines are shown in Fig. 5. In the case of the "wrong fillet," the wake curves in row 1 are similar to those for the wing alone up to an angle of 6 deg, but at and above an angle of 9 deg the wakes in this row are similar to those for the wing and fuselage with no fillet. The fact that this behavior is not observed at rows 2 and 3 indicates once more that the effect is largely localized in the region of the intersection of wing and fuselage. The force-measurement curves (Fig. 8) discussed later show that at the angle of 7 deg something happens to the flow. The change in the flow pattern from the wing-alone type to the no-fillet type is completely discontinuous. An observer watching the multiple manometer when the model is set at this critical angle sees either one pattern or the other, and the change from one to the other is sudden and complete. At this angle the manometer is in a very unsteady condition. The smaller flow pattern for the downwash and the larger for the trailing vortex replace each other rapidly on the tubes. These are no intermediate stable states; either the manometer is showing one pattern or the other, or it is changing rapidly. The method of watching the multiple manometer for the change in pattern was found to be extremely sensitive, as improvements could be made and the critical angle checked visually with an accuracy of ¼ deg. This made an exceedingly rapid method of observing the effect of modifications, as no computations were necessary, and all that was needed was to start the tunnel and to run the model through a range of angles of attack. The critical angles obtained in this manner checked very accurately with the breaks in the polar, lift, and moment curves. Fig. 3 shows the wake curves for a fillet of the optimum type. The critical angle in this case was 15.7 deg. The lines of this fillet are shown in Fig. 5, and it is illustrated in Fig. 7.

FORCE MEASUREMENTS

The force measurements were made with the normal three-component set-up. The tests were run at a water head of 39.24 cm, and as the root chord of the model was 42.35 cm, the Reynolds number for the tests was approximately 2,100,000. The accuracy of the tests can be judged from the points shown on the enlarged section of the polar (Fig. 14). It is not believed that variations in the maximum-lift coefficient of the order of 2 or 3 percent are of any significance, as it was found that almost imperceptible changes in the surface condition of the model would cause this much variation. This variation is in line with that found by the N.A.C.A. in large-scale tests (ref. 3).

Before beginning the investigation on the effect of fillets, lift, drag, and pitching moment, tests were run of the wing alone. These results furnish a convenient basis for the subsequent discussion.

NORMAL LOW-WING CONFIGURATION

Results of three component tests on the normal low-wing arrangement are shown in Fig. 8. Considering the curves in the case of no fillet, it is observed that at a C_L of 0.6 the polar starts breaking over sharply, the C_L against α-curve takes up a new slope, and the moment curve becomes irregular. Now referring to the wake-loss diagrams, it is seen that for an angle of 5 deg the downwash is much smaller for the wing and fuselage than for the wing alone. The wing-alone polar parallels the induced-drag parabola (aspect ratio 5.97). The polar curve of the wing-fuselage arrangement parallels the wing-only polar up to a C_L of 0.45, and then starts deviating from it. If one attempts to fit an induced-drag polar to this curve in the region from a lift coefficient of 1.0 to 1.2, one needs an aspect ratio of approximately 2.5, or somewhat less than one-half the aspect ratio of the complete wing. This leads one to suspect that the wing is acting as two monoplanes separated by the fuselage and that there is a trailing vortex on each side of the fuselage. The wake diagrams lead one to the same conclusion, as from them one sees that that downwash is practically the same at rows 2 and 3 as for the wing alone, while in row 1 the downwash has practically disappeared for all positive angles. It will then be assumed as a working hypothesis that the foregoing is correct and that the interference corresponds to the breaking up of the horseshoe lifting vortex into two side-by-side horseshoe vortices.

This aerodynamic picture enables one to account for the peculiar shapes of the various curves, and also shows why the minimum drag is not subject to much improvement.

Three fillets were then tested to determine the best type. These fillets all had a radius of 15 percent of the root chord. Fillet 1 was a uniform fillet of this radius throughout. Fillet 2 (wrong) had a 15 percent radius at the nose and tapered aft; its lines are shown in Fig. 5. Fillet 3 had a small nose and a radius of 15 percent at the trailing edge, and was of the same type as the optimum fillet shown in Fig. 5. A photograph of fillet 3 is shown in Fig. 6. All three of these fillets were a marked improvement over the unfilleted condition, fillet 3 having the greatest maximum lift and the greatest slope of its lift curve. As all of these fillets were a great improvement over no fillet, an elaborate program was undertaken to develop fillets of various types. It soon appeared that fillets 1 and 2 could not be much improved, while fillet 3 could be developed with considerable success. Numerous variations of this type were constructed, and the following rules were deduced from the tests:

1. Increasing the trailing-edge radius improves the maximum lift and prevents induced-drag losses at medium-lift coefficients; excessive trailing-edge radii increase the drag.
2. Increasing the radius at the nose increases the drag and decreases the maximum lift.
3. The fillet should taper as uniformly as possible from the nose to the trailing edge, and its maximum size should be as close to the trailing edge as possible.
4. The fillet should be washed out smoothly to the fuselage.

Fillet 4 is an optimum design of this type. Maximum lift has been sacrificed for low drag, and it will be noticed that there is only a trace of the former ill effects of the fuselage. The wake diagrams for this fillet are shown in Fig. 3, and it will be noticed that the break in the downwash does not occur until over 15 deg, while in fillet 3 the break occurred at 10 deg. It will be noticed also that with fillet 4 the slope of the lift curve is steeper than the corresponding curves for the wing alone. This effect can be easily explained, as in this case the fuselage contributes some lift, and the coefficients are calculated neglecting this effect.

LOW-WING CONFIGURATION, WING LOWERED

It was suggested by an aircraft constructor that an investigation of the case of a low-wing monoplane in which the wing passes entirely below the cabin floor would be of great interest. Accordingly, a block 2 in. thick was made and placed between the wing and the fuselage. This corresponded to lowering the wing 1 ft at full scale. It was impossible to take any observations of the unfilleted condition for this arrangement. The first fillet put on in this configuration had a 20 percent chord trailing-edge radius and had the nose of the wing faired forward and up to the fuselage. The results are plotted as fillet α in Fig. 9. Fillet b in the same figure corresponds to a similar fillet with the leading edge cut back. The fillet was then enlarged to 27.5 percent trailing-edge radius and the nose was undercut still more. The lines of this fillet D are given in Fig. 5. Fillet D was found to be the optimum for this configuration. Fillet e was an endeavor to secure still greater improvement. The fillet was hollowed out by decreasing the radii in front of the trailing edge of the wing. This hoped-for improvement was not realized, but the curve is included in the figure. The curves for case D and for the wing alone illustrate how perfectly the pernicious effect of the wing-fuselage intersection can be eliminated; it will be noticed on the polar that the profile drag is practically independent of the lift coefficient. It was found that, as in the case of the normal low-wing configuration, the radius at the nose should be a minimum.

HIGH-WING CONFIGURATION

The model was next arranged as a high-wing airplane, the wing being mounted so that it had the same angle of incidence as in the case of the low wing and so that its trailing edge was just touching the fuselage. The curves for the high wing with no fillet and high wing with fillets are shown in Fig. 13. The fillet had quite a large trailing-edge radius (20 percent) and was of the tapered type. The lift obtained with this fillet was the highest found, but was associated with a slight increase in drag. Fillets of intermediate radius and of various types, large at the front, small at the rear, constant radii, etc., caused very little change. The curves for intermediate radii, and in fact for all of the other variations, lie between the two curves mentioned. The most remarkable point in this group is the behavior of the polar of the no-fillet case in the vicinity of zero lift. It will be noticed that this curve departs markedly from the wing-only polar in the same manner, for negative angles, as does the low-wing no-filet polar at small positive angles. This illustrates the extreme sensitivity of the suction side of the wing.

GENERAL DISCUSSION

It has been seen from the foregoing that the aerodynamic disadvantages of the low-wing monoplane can be almost completely eliminated by proper design of the wing-fuselage intersection. The maximum-lift coefficient of the wing alone can be attained with either a high-wing or a low-wing configuration. The differences in drag between the high wing and the normal low wing are inappreciable.

In the case of the dropped low wing there is found to be some drag increase, but it is thought that this difference could be reduced by further investigation. The size of the optimum fillet in the case of a low-wing design can be decreased by the following means:

1. Making the fuselage small.
2. Keeping the distance of the trailing edge of the wing below the bottom of the fuselage small.
3. Using an airfoil of small top camber.

The minimum drag of the fuselage-wing combination can be best decreased by making the fillet at the leading edge of the wing as small as possible.

Buffeting. For the purposes of discussion, buffeting will be defined as follows: Buffeting is a violent shaking of the airplane and tail surfaces by aerodynamic forces at angles below the stall. Buffeting in this sense has been observed in the case of almost all low-wing monoplanes. If one considers the wake measurements and notices the critical angles, there is no difficulty in seeking what occurs.

The buffeting seems to be due to these three causes: first, the lift over the center section of the wing disappears, causing a decrease in the total lift available; second, the wake jumps from one side of the stabilizer to the other; and third, the stabilizer is then in the trailing vortex formed at the side of the fuselage. The position of the stabilizer is indicated on each of the wake drawings.

Mr. H. J. Steiger (ref. 4) has suggested that buffeting could be eliminated by cutting away the wing root and reducing the angle of attack at the fuselage. It is dubious if any such attempt would be satisfactory, as it would inevitably result in trailing vortices forming near the fuselage, and these vortices might cause a dangerous or uncomfortable tail vibration, in addition to their pernicious effect on the induced drag. It is easily seen that for a properly designed junction between the wing and the fuselage the lift must carry entirely across the span, since if there were no lift over that portion of the wing covered by the fuselage, the polar curve for the wing and fuselage would of necessity differ more from that of the wing alone than it actually does.

ACKNOWLEDGMENTS

The author wishes to acknowledge his gratitude to the entire staff of the laboratory for their assistance in performing the tests and in preparing this report. He is especially indebted to Dr. C. B. Millikan, Mr. W. H. Bowen, Mr. W. B. Oswald, and Mr. N. B. Moore for their efforts.

DISCUSSION

E. Ower. The author's results entirely confirm the ideas formed from some similar, although not so comprehensive, work for which the writer was responsible some time ago. An account of this work was given in a lecture read to the Royal Aeronautical Society in January, 1932, and the writer thinks that the explanation he then put forward of the type of interference for which fillets are found to be beneficial is worth repeating. He suggested that this interference occurs when the airstream has to expand at more than a certain rate if it is to remain in contact with the body and wing surfaces. A certain rate of expansion can be tolerated, but if the surfaces diverge from one another too rapidly, the flow detaches itself from them and a region of turbulence is set up which leads to a loss of lift and an increase of drag. A well-known analogous case is that of the outlet cone of a venturi tube—if the angle of this cone is too great, it does not "run full;" that is, the flow breaks away from the walls, with the resulting loss of efficiency of reconversion of the kinetic energy into static pressure.

This hypothesis was confirmed by the various tests made to investigate its truth, and it explains the author's results with fillets. For in his case the geometry was such, as indeed it is in most practical body-wing combinations, that the rate of expansion increased progressively from the maximum camber of the wing toward the trailing edge. Hence, as the author found, the best fillet increases in radius toward the rear of the wing. The writer's experiments were made with fillets of constant radius, but he did in fact predict that fillets of increasing radius toward the trailing edge would be preferable. The same line of reasoning indicates why fillets on the under surface of a high-wing combination are found to have very little effect; the divergence between the surfaces of the body and the wing is much less in such a combination than it is in low-wing positions. Moreover, in the high-wing position the pressure gradient along the lower wing surface is such as slightly to assist the flow to adhere to the surfaces, whereas in the low-wing position the pressure gradient on the upper surface tends powerfully in the opposite direction.

The author mentions the importance of preserving as far as possible the normal lift distribution along the span of the wing. This again agrees with views that the writer has expressed. This principle, together with that of avoiding regions of divergent flow, will be found to be of the utmost importance to the designer in his efforts to build high-performance aircraft. The designer is always more interested in direct proof than in speculation, and the author has provided such proof, whereas the writer, mainly through lack of time, was content to put forward ideas which needed corroboration by facts before they could be accepted with complete confidence.

Richard M. Mock. This is believed to be the most interesting piece of aerodynamic research in America published during 1932, outside of that of the N.A.C.A. It is unfortunate that the tests were not made with a running model propeller, as it is possible and likely that the slipstream affects the flow around the fuselage and especially over the wing-fuselage intersection. Therefore it is believed that the comparative drag figures are somewhat questionable. During take-off and climb with high angles of attack, the same would be true as affecting the lift coefficients, while for landing the propeller effect is negligible. The comparative maximum lift coefficients, without propeller, are very valuable, as this is applicable to the landing condition.

The writer differs from the author regarding his comparison with the high-wing monoplane. The high-wing monoplane which he used as a comparison showed almost 6 percent more maximum lift and about 3 to 4 percent less drag than the best low-wing arrangement. However, the writer is under the impression that a high-wing design, rather than have the wing resting on top of the fuselage with its trailing edge just touching the fuselage, will have less drag if the wing is sunk into the fuselage, so that the top of the fuselage meets the top of the wing about one-third or one-half chord back from the wing leading edge and the combination is carefully filleted. The top of the fuselage might be lowered in front of the wing to allow a clean leading edge, and the portion of the fuselage above the rear portion of the wing might have fillets of very large radius. This should reduce the frontal area and the drag still below that of the combination used, and if the leading edge is carefully faired, should not affect the lift other than increase it by directing the flow from the fuselage over the upper surface.

Regarding the best position for the low wing, it would be interesting to raise the wing, as on the Gee Bee racer, rather than lower it as was done. As the fuselage decreases in width near the bottom and the wing is cambered on top, a pocket or cavity is formed between upper surface of the wing, just in front of the trailing edge, and the lower surface of the fuselage. The air passing over the wing and over the fuselage must fill this pocket causing eddies and consequently drag. Therefore it is logical that by fairing over this cavity, as the author has done, the drag of the combination will be reduced. This could also be done by raising the wing slightly to where the fuselage is wider and then using a fillet, and also perhaps by changing the fuselage cross-section slightly so that an excessive fillet will not be necessary. Another means would be to have the wing, in front view, curve upward at the root, meeting the fuselage side at a right angle.

Of course, if the landing gear is attached to the wing, the wheel supports will be longer if the wing is raised. The increased length means slightly greater weight, and if the undercarriage is not retractable, the frontal area and consequently the drag will be increased. Lowering the wing from the optimum (filleted) low-wing position to the optimum (also filleted) dropped-wing position (Fig. 14) means an increase in drag of approximately 17 percent. With a fixed external landing gear, the two shorter wheel supports with the dropped-wing position might partially offset the increased wing-fuselage drag. With a fully retractable landing gear, the landing-gear resistance could be neglected, and only the best wing-fuselage arrangement considered, with the tail location and fillet varied to eliminate buffeting.

The effect of lift-increasing trailing-edge flaps on buffeting would be interesting.

The writer would appreciate having the author's opinion of the meaning of the double curve near the maximum-lift coefficient of the dropped-wing combination with optimum fillet D. He also would appreciate an opinion of the double wake behind the fuselage at −5 deg, zero, and +5 deg for the same wing-fuselage combination. Another pilot combination between 1 and 2 and still within the stabilizer span would have been interesting. The writer would like to know the drag coefficient of the fuselage at the various angles of attack so that it can be added to the drag of the wing alone and compared with the drag of the combination.

G. J. Klein.The early investigations into body-wing interference were, it is true, conducted at rather low values of Reynolds number. However, they disclosed very interesting results, particularly in the case of the low-wing monoplane, and certainly showed the need for further research at higher Reynolds number. In this connection, the present paper, together with an extensive series of experiments at the N.P.L. (R. & M. 1480 and R. & M. 1300), form a valuable extension and show that the general conclusions reached in the earlier work still hold at much higher values of Reynolds numbers. When we consider all this research together, we get a fairly accurate picture of the subject.

It is now definitely established that detrimental interference is due to burbling caused by an attempt to expand the airstream too rapidly in the angle between the side of the body and the surface of the wing. It is true that the presence of the body does change the lift grading of the wing and thus increases the induced drag, but unless this burbling occurs, this effect is very small. Actually the wings so modify the flow about the body that the body contributes appreciably to the total lift of the combination, and the resulting lift grading is not very different from the lift grading of the wing alone.

For the foregoing reasons it is sufficient to confine this discussion to the case where burbling occurs and to consider the factors involved. Obviously, the worst case would be a low-wing monoplane without fillets, having a highly cambered wing root and a body of small fineness ratio and of such a cross-sectional shape that the angle between the side of the body and the upper surface of the wing is small compared with 90 deg. This combination forms a pocket between the body and the wing near the trailing edge of the latter, into which the airstream cannot expand, even at the zero lift angle of attack of the combination. Increasing the angle of attack increases the difficulties in expanding the airstream, thus causing increased burbling. There are several methods of suppressing this burbling:

(1) The wing can be raised to a higher position on the body to eliminate the pocket effect.

(2) The body can be given flat sides, making an angle of 90 deg with the wing surface, as was done in the Schneider Trophy Racer S5.

(3) A fillet of increasing radius toward the trailing edge can be employed, as was done in the present paper.

In short, anything that eliminates this pocket effect over the useful range of angle of attack will result in a combination of body and wing that will be free of any undesirable interference effect.

An interesting result brought out by the paper is that the "double stalls" found in the earlier work at low values of Reynolds number are still present at much higher values. Undoubtedly the recovery from the first stall is due to a decrease in the extent of the burbling caused by the downwash from the nose of the body.

Probably the most important point brought out by the paper is that a properly designed low-wing monoplane can be just as efficient as a high-wing monoplane.

AUTHOR'S CLOSURE

Replying to Mr. Ower, the low-wing fillet described was first developed in our wind tunnel in May, 1931, and test-flown in June of the same year. The second Northrop Beta and all succeeding Northrop ships have carried this device. The author agrees entirely with Mr. Ower's statements as to the necessity of eliminating most of the expansion of the wing fuselage intersection, as was first pointed out by Muttray (loc. cit.).

Replying to Mr. Mock, as he says, it is unfortunate that the tests were not made with a running propeller. The laboratory has under development a fuselage, in which is included an electric motor with complete dynamometer, for repeating the preceding investigation with a slipstream. The model when finished will have the proper scale horsepower and the proper ratio of propeller diameter to span. However, the author does not believe that the drag differences will be large in the case with slipstream and without, as the laboratory has been rather successful in predicting the performance and especially the high speed of the airplanes which it has tested. In a paper by Drs. Theodore von Kármán and Clark B. Millikan there are described the methods used by this laboratory in estimating the performance of the actual airplane. Due to the successful checks which have been secured, it is not very probable that the discrepancies in drag between power off and power on will be large.

Since the investigation of the fillet was undertaken as an engineering study rather than as a scientific investigation into the configuration for minimum drag of the wing and fuselage alone, we did not consider several of the cases mentioned by Mr. Mock, as the lowering of the wing in the case of a high-wing monoplane or the raising of it in the case of a low-wing monoplane would require a larger fuselage in order to maintain the headroom in the cabin.

The reduction in frontal area in the cases mentioned by Mr. Mock would be more apparent than real, as it would necessitate, if his suggestions were carried out, the enlarging of the fuselage in order to accommodate the passengers or other loads. The raising of the wing any great distance was impossible in this particular case as it would bring the floor level too high. The consolidation of a stressed-skin wing into beams in order to enable the passengers to place their feet below the top surface of the wing is exceedingly extravagant of weight, and it also causes the cabin to be encumbered with structural members which interfere with the free circulation of passengers or the stowing of freight.

The author does not believe that the position of the wing with reference to the fuselage is a vital matter, as what is gained in one place is lost in another. In the multimotor transport field there is a very interesting case. There are two modern low-wing bimotor transports of the same power loading and span loading, one with the wing passing completely below the cabin floor and the other with the wing beams passing through the cabin. The airplane with the wing completely below the cabin floor is much faster, in the neighborhood of 25 mph, than its competitor. The difficulties of designing a retractable landing gear increase with at least the square of the length of the members as does also its weight. The raising of a wing 6 in. or 8 in. farther above the ground will often make a satisfactory retractable landing gear almost impossible. The addition of lower surface flaps has been found in this laboratory to have no effect on the buffeting of a well-filleted airplane. The author believes that the double peak of the curve for the optimum fillet D is probably due either to asymmetries of the model or to asymmetries of the air stream. The model was known to have developed some aerodynamic twist. This twist raised the angle of attack of one side of the model above that of the other, and consequently one side stalled in advance of the other. This effect has also been found in the laboratory a number of times in rolling-moment tests. The author believes that the two peaks in the wake diagrams shown in Fig. 10, at −5 deg. and +5 deg, can be explained as follows: The upper one is probably the wake of the fuselage and the lower one that of the wing. When the downwash becomes large, the fuselage wake becomes merged into that of the wing. The drag coefficient of the fuselage has never been measured separately, so that Mr. Mock's question on this point cannot be answered.

Replying to Mr. G. J. Klein, he is completely correct in his statement as to the three methods of eliminating the burbling in the wing-fuselage intersection. However, in the case of a straight-sided fuselage, the burble is not completely absent. The author would like to cite Fig. 15 as evidence in this case. The results in this figure were obtained with the Douglas transport fuselage, which is straight-sided in the region in contact with the wing and for a considerable distance aft of the trailing edge. The wing section used in this transport is the N.A.C.A. 2215. This section is of only 15 percent thickness, has a very small camber (2 percent), and furthermore the point of maximum camber is only 20 percent aft from the leading edge. This wing has therefore probably the smallest top camber of any section in common use.

Nevertheless, in the test shown, the difference between fillet and no fillet is quite perceptible. The model was tested with tail surfaces but less nacelles. The separation between the two curves shown on the larger plot in the figure has been exaggerated in the region of low lift coefficient in order to enable the two curves to appear separately on the reproduction. The smaller plot shows the experimental points in their true separation. In the low drag region the addition of the fillet reduced the minimum drag for the model approximately 1.8 percent and raised the maximum lift coefficient as shown from 1.27 to 1.34. For every high angle not shown in the plot, the curves cross each other, an effect the author is unable to explain, but at all usable angles of attack the filleted case is superior to that of the unfilleted.

The discussion of why some of the lift curves have double peaks has been included in the reply to Mr. Mock. However, the double stalls in the other sense, i.e., of curves that come to a maximum and then go on to a further maximum, are apparently due to the local burbling of the wing in the center, while outboard portions of the wing do not stall as soon, but continue to work at larger and larger angles of attack. These restricted portions of the span can then go to larger lift coefficients, as it is well known that wings of small aspect ratio can reach larger lift coefficients than normal wings. It has been found in our laboratory that the characteristic sharp break in the lift curve can be explained as an effect of this kind. We always find the sharp break in untapered wings. Tapered wings usually show the rounding top which we think means that the center of the wing stalls before the outboard portions. Evidence for this can be seen in Fig. 2 on the wake diagrams for the wing alone. In one very interesting case the laboratory found that this sharp drop-off of the lift curve was obtained with a tapered wing in which an auxiliary airfoil was used only over a part of the center section. In this case the sharp drop in the lift curve was obtained, not only in the laboratory but also in flight, while the airplane without the auxiliary airfoil stalled in the more usual manner. We therefore think that in this case the auxiliary airfoil held the flow on the center section to a higher lift coefficient than that which it would normally reach, and when it stalled, the entire wing stalled at once.

Since the foregoing paper was written, two confirmations have been published of the results, and numerous airplanes have been designed and flown with these devices.

Document 3-17(b), James A. White and Manly J. Hood, "Wing-Fuselage Interference, Tail Buffeting, and Air Flow About the Tail of a Low-Wing Monoplane," NACA **Technical Report** ***482 (Washington, 1934).***

REPORT NO. 482

WING-FUSELAGE INTERFERENCE, TAIL BUFFETING, AND AIR FLOW ABOUT THE TAIL OF A LOW WING MONOPLANE

BY JAMES A. WHITE AND MANLEY J. HOOD

SUMMARY

This report presents the results of an investigation of the wing-fuselage interference of a low-wing monoplane conducted in the N.A.C.A. full-scale wind tunnel on the "*McDonnell*" airplane. The tests included a study of tail buffeting and the airflow in the region of the tail. The airplane was tested with and without the propeller slipstream, both in the original condition and with several devices designed to reduce or eliminate tail buffeting. The devices used were wing-fuselage fillets, an N.A.C.A. cowling, reflexed trailing edge of the wing, and stub auxiliary airfoils.

The use of proper fillets practically eliminated the wing-fuselage interference and greatly reduced the tail vibrations due to buffeting. An N.A.C.A. cowling reduced the buffeting and interference effects to unobjectionable magnitudes at angles of attack up to within about 3° of the stall. A large fillet alone gave the greatest reduction in buffeting effect, reducing the tail vibrations to one seventh their original amplitude, but the combination of the large fillet and N.A.C.A. cowling gave the best all-round results. This combination reduced the tail oscillations due to buffeting to one fourth their original amplitude, increased the maximum lift 11 percent, decreased the minimum drag 9 percent, increased the maximum lift/drag ratio of the whole airplane 19 percent, and increased the effectiveness of the elevator about 40 percent at angles of attack in the landing range. The reflexed trailing edge had a minor effect and the auxiliary airfoils in the best position tested were considerably inferior to the fillets. With the propeller operating, the interference effects were practically eliminated, even with the airplane in the original condition.

The elimination of the wing-fuselage interference slightly decreased the longitudinal stability of the airplane.

Records of the fluctuations in the dynamic pressure of the air stream at the tail show a prominent wake-fluctuation frequency of the order of magnitude of the natural frequency of the tail vibrations.

FIGURE 1.—The *McDonnell* airplane with large fillet in full-scale wind tunnel.

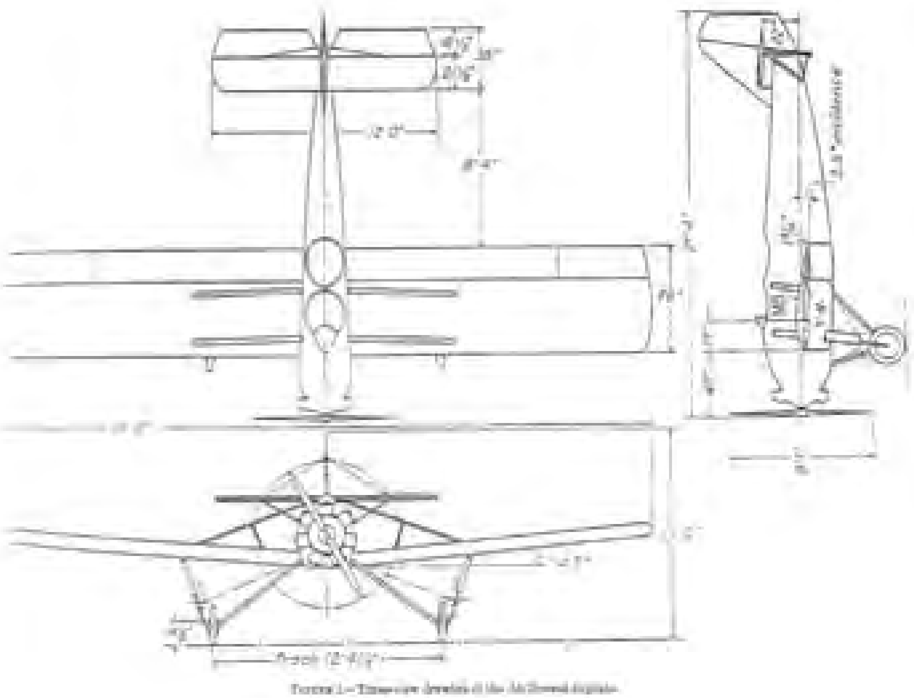


FIGURE 2.—Three-view drawing of the *McDonnell* airplane.

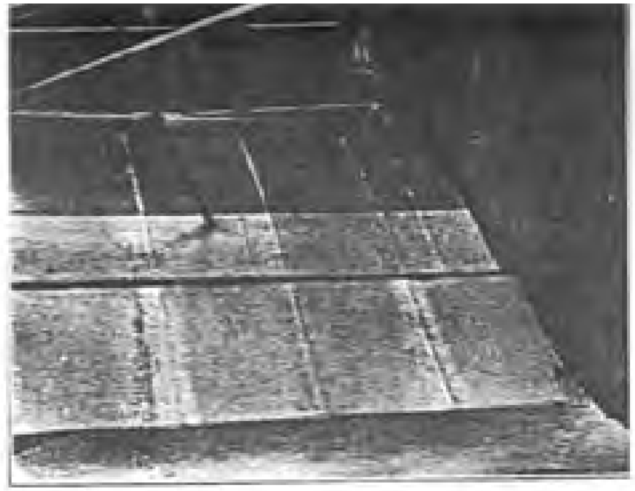


FIGURE 3.—Wing-fuselage intersection of *McDonnell* airplane.

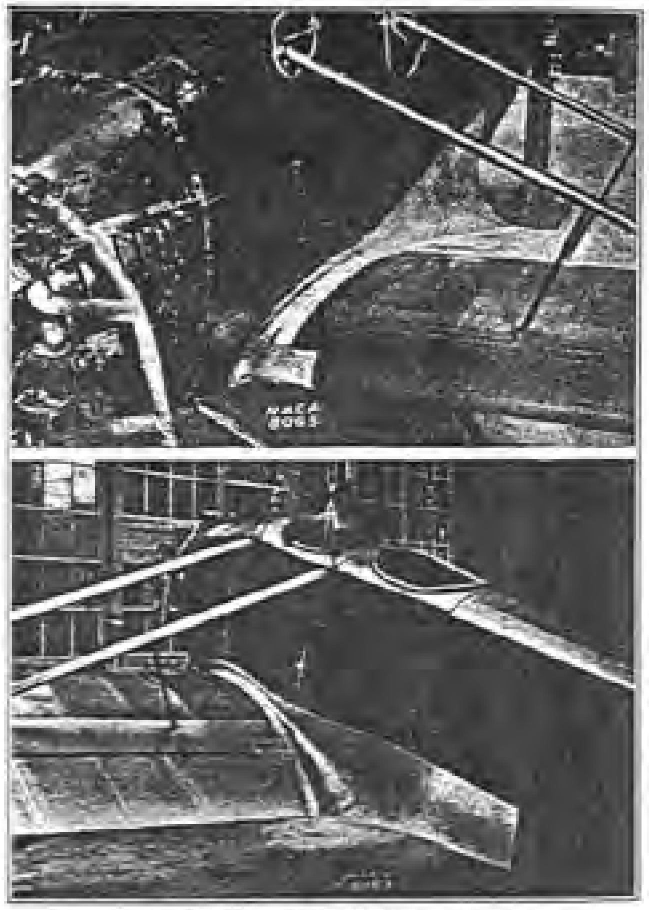


FIGURE 4.—Small fillet on *McDonnell* airplane.

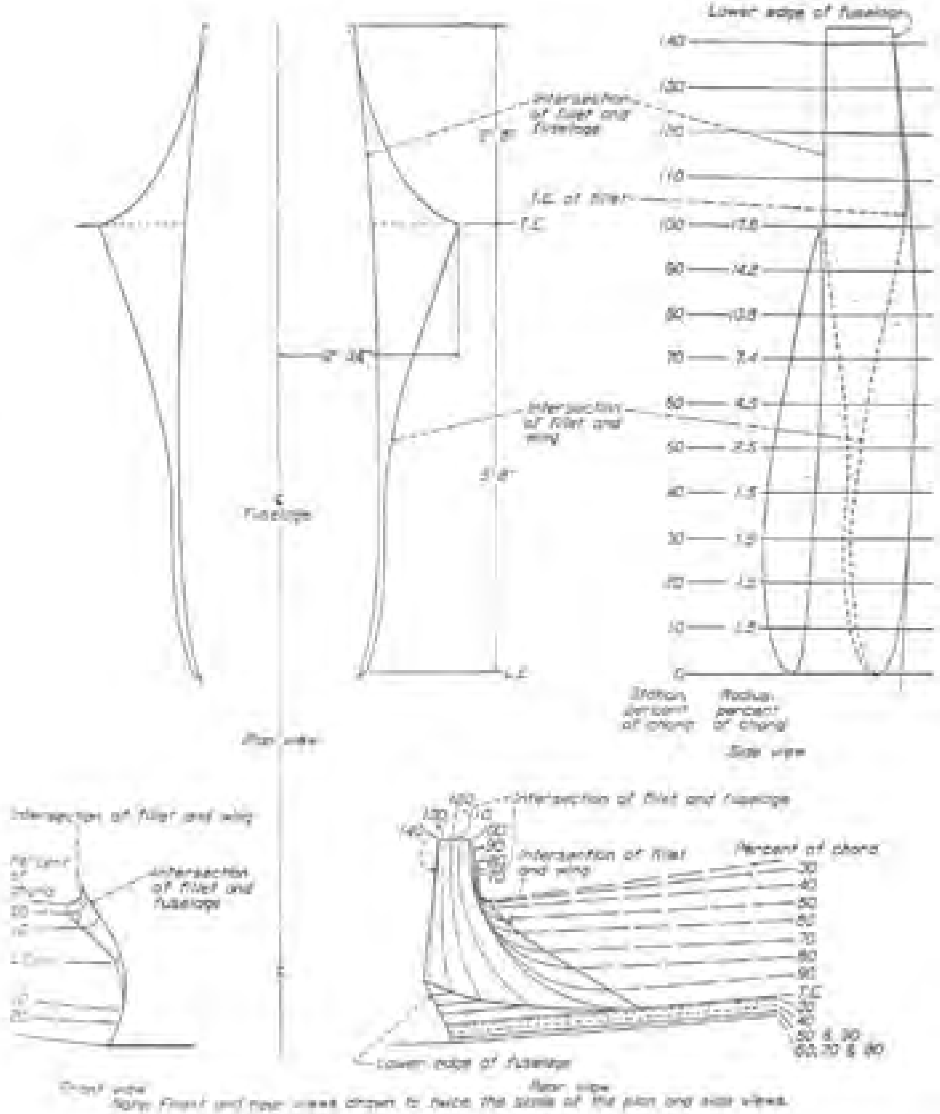


FIGURE 3.—Drawing of the small fillet.

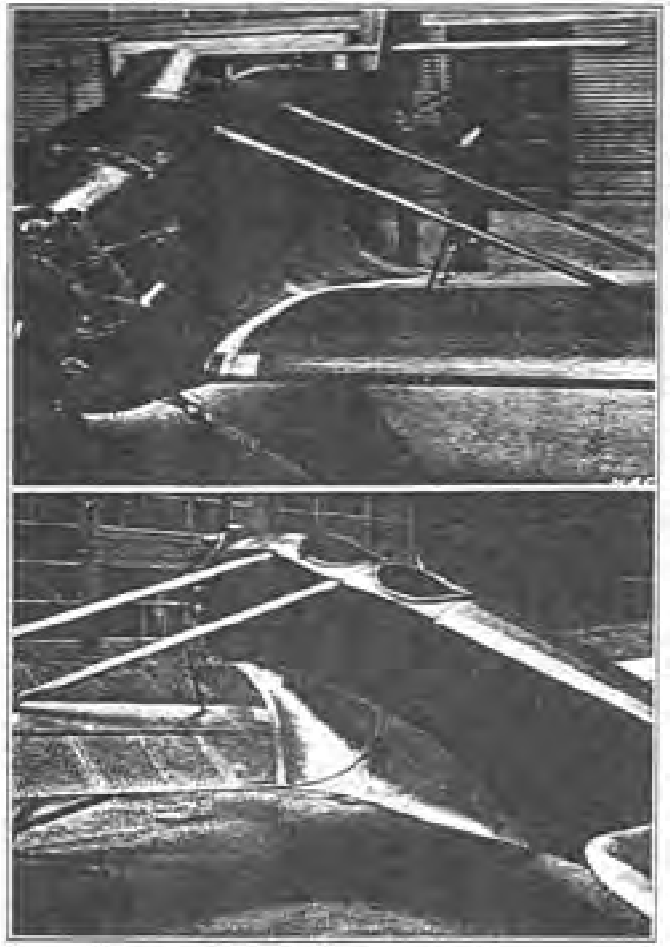


FIGURE 4.—Large fillet on McDonnell airplane.

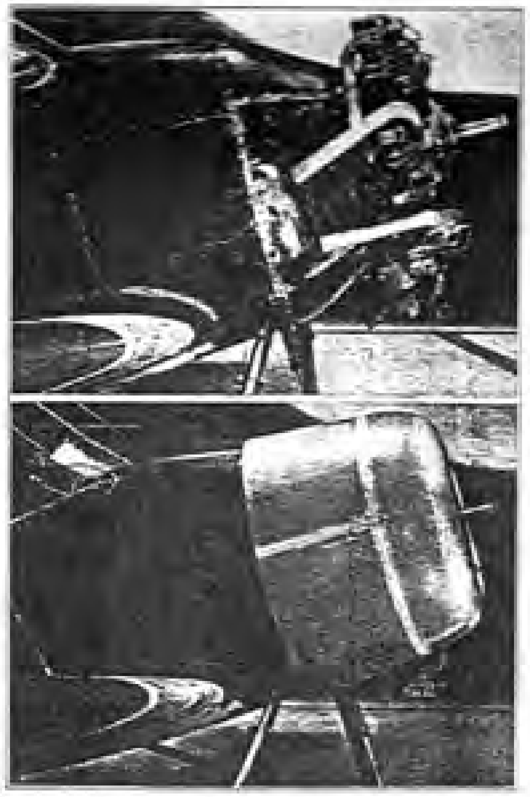


FIGURE 8.—Nose of McDonnell airplane in original condition and with N.A.C.A. cowling.

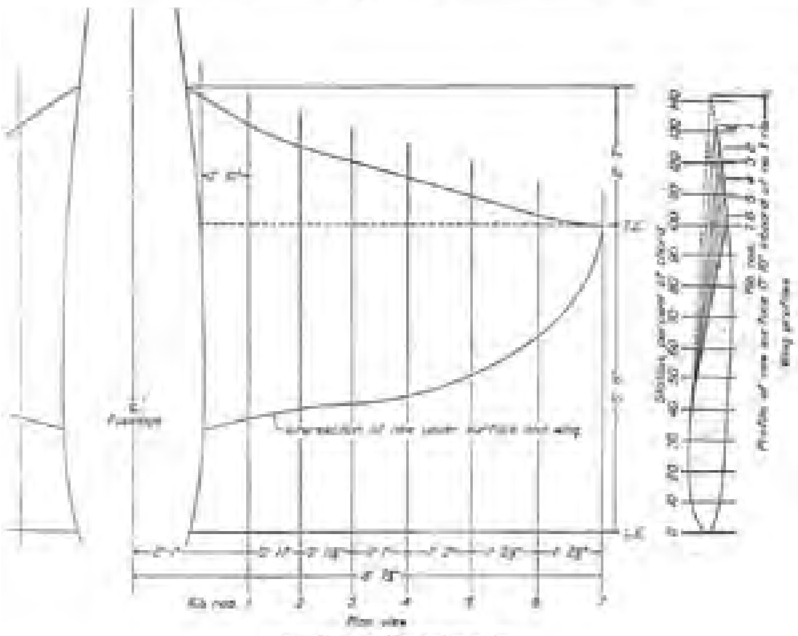


FIGURE 9.—Drawing of the reflexed trailing edge.

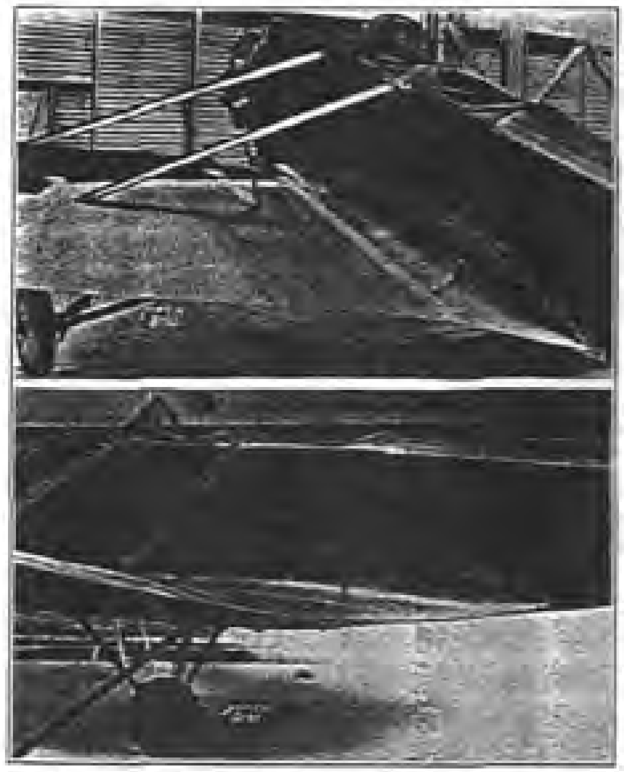


FIGURE 10.—Reflexed trailing edge with fillet on McDonnell airplane.

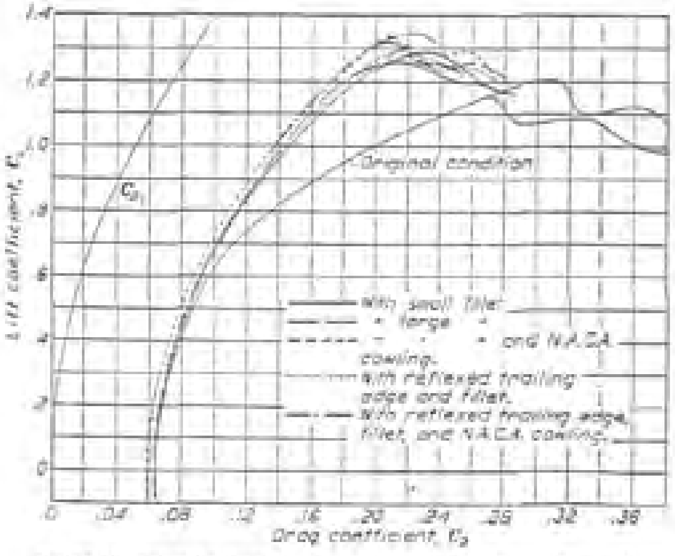


FIGURE 11.—Polars for McDonnell airplane with various fillets. Corrected for tunnel effects. Power off.

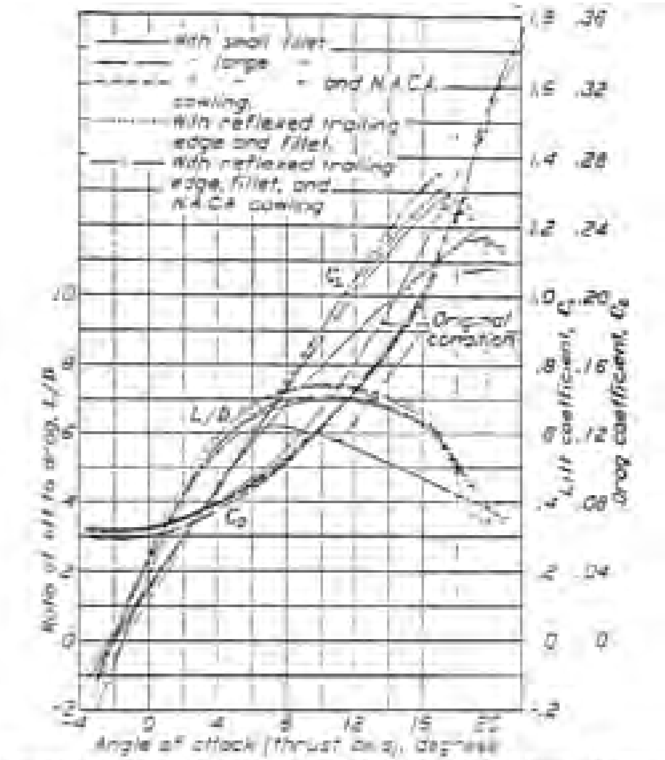
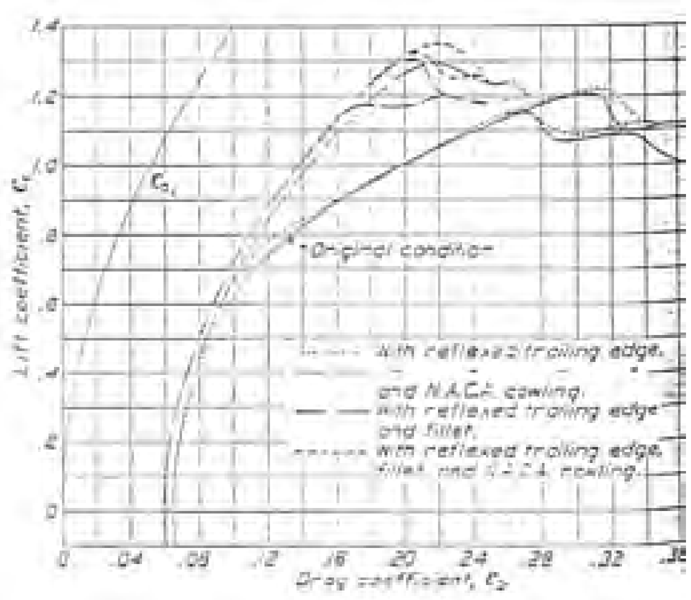


FIGURE 12.—Lift and drag of McDonnell airplane with various fillets. Corrected for tunnel effects. Power off.



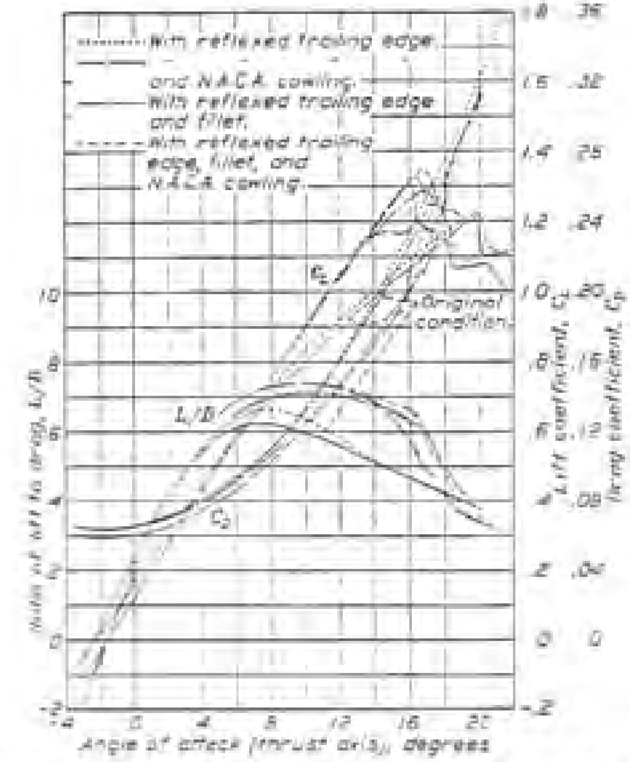


FIGURE 14.—Lift and drag of McDonnell airplane with reflexed trailing edge. Corrected for tunnel effects. Power off.

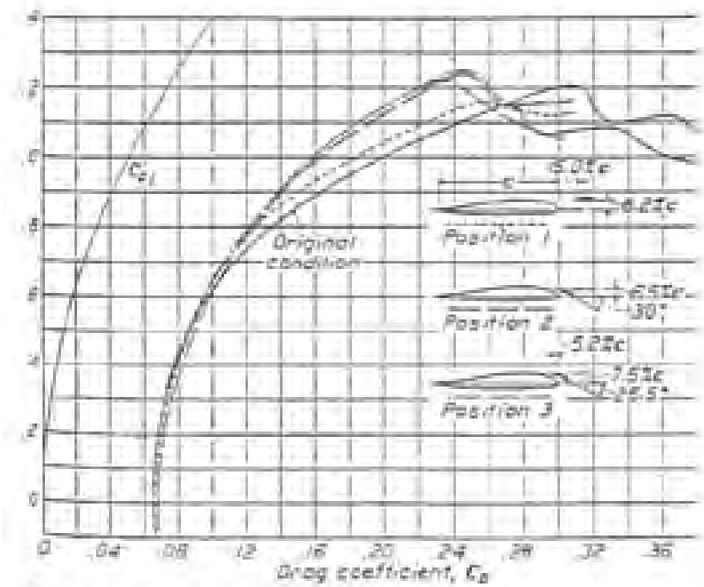


FIGURE 15.—Polars for McDonnell airplane with auxiliary airfoils. Corrected for tunnel effects. Power off.

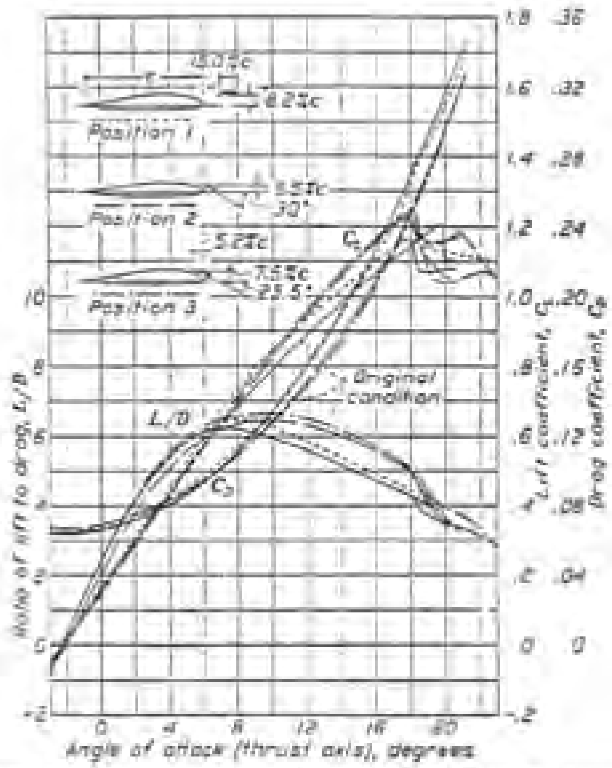


FIGURE 16.—Lift and drag of McDonnell airplane with auxiliary airfoils. Corrected for tunnel effects. Power off.

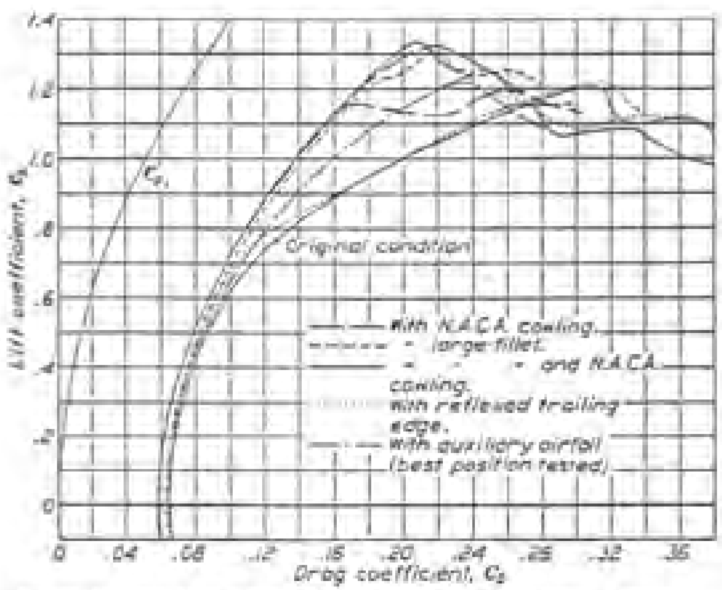


FIGURE 17.—Polars for McDonnell airplane comparing various devices. Corrected for tunnel effects. Power off.

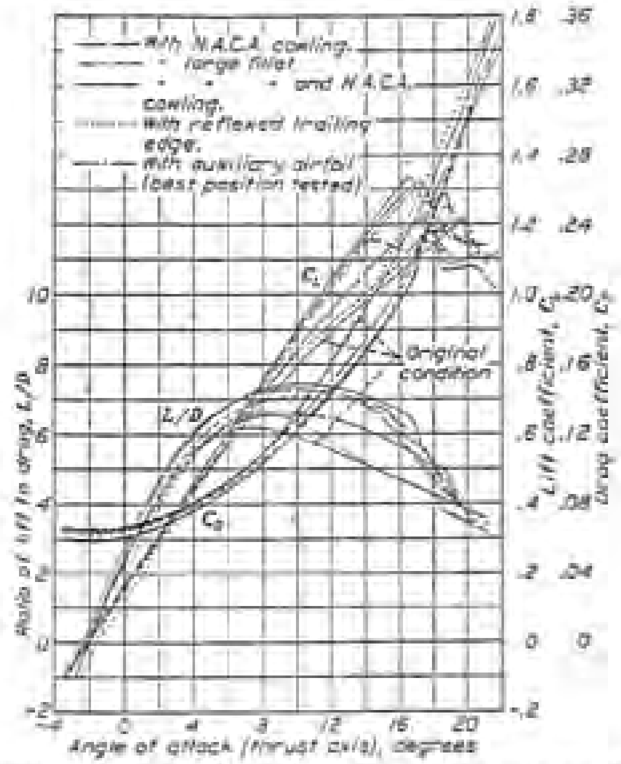


FIGURE 18.—Lift and drag of McDonnell airplane comparing various devices. Corrected for tunnel effects. Power off.

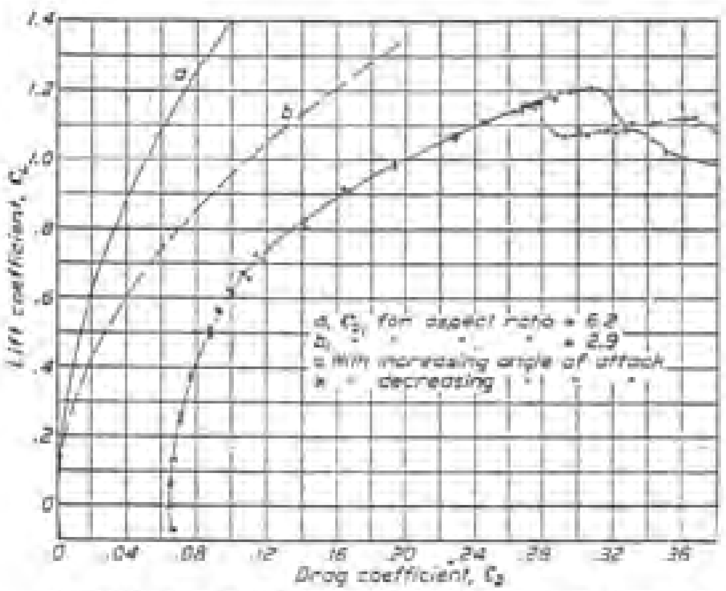


FIGURE 19.—Polar for McDonnell airplane in original condition. Corrected for tunnel effects. Power off.

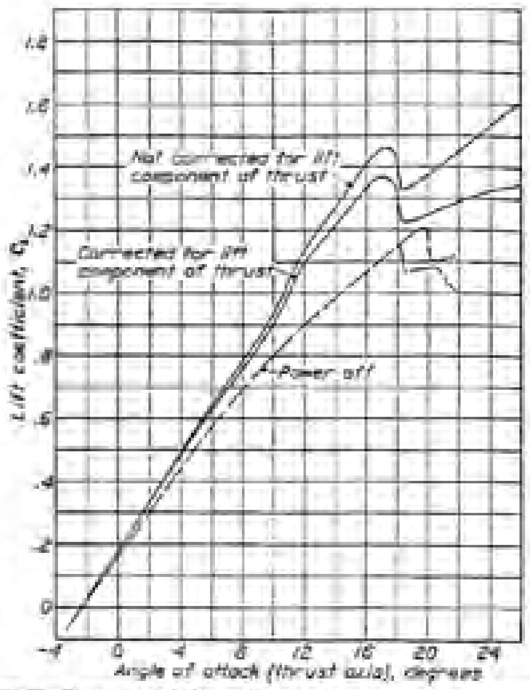


FIGURE [illegible].—Power-on lift of McDonnell airplane in original condition. Corrected for tunnel effects.

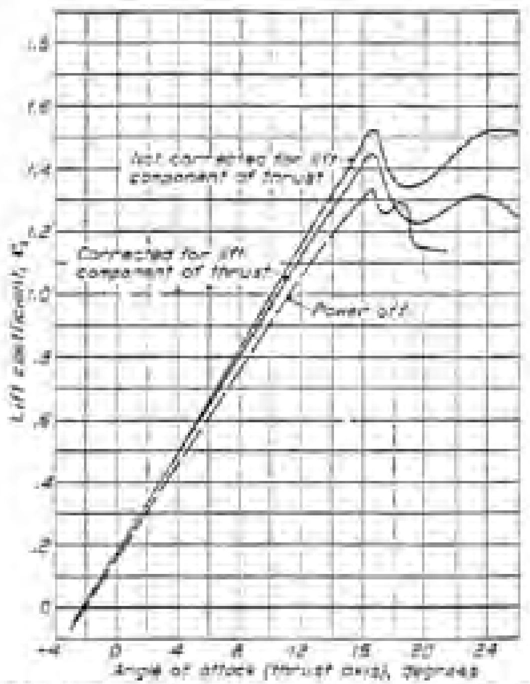


FIGURE [illegible].—Power-on lift of McDonnell airplane with large fillet and N.A.C.A. cowling. Corrected for tunnel effects.

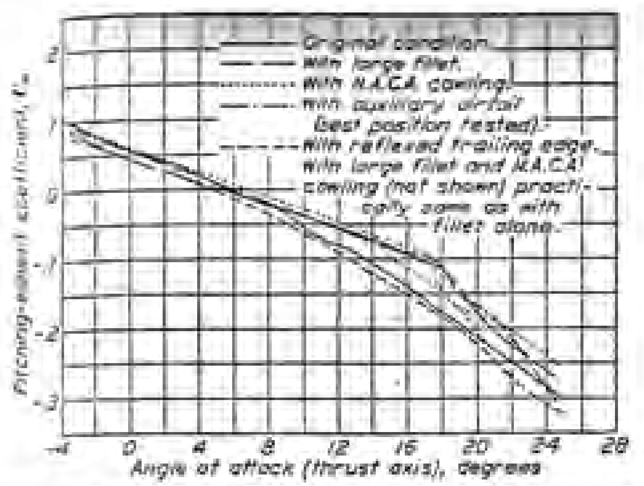


FIGURE [illegible].—Pitching moments of McDonnell airplane with various devices. Corrected for tunnel effects. Power off. Stabilizer 0.6° to thrust axis. Elevator 0° to stabilizer.

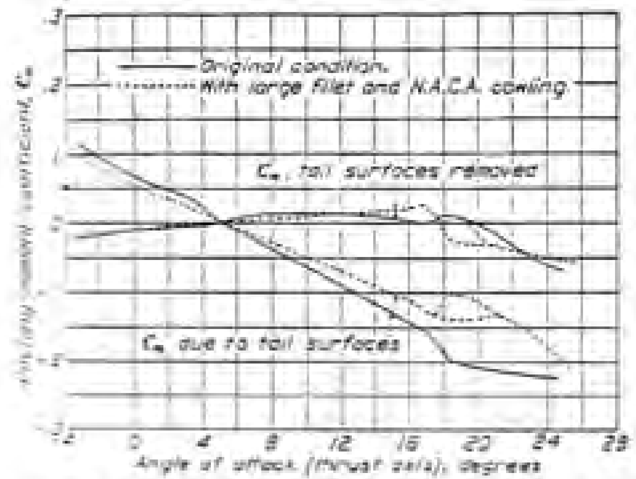


FIGURE [illegible].—Pitching moments due to tail of McDonnell airplane. Corrected for tunnel effects. Power off. Stabilizer 0.6° to thrust axis. Elevator 0° to stabilizer.

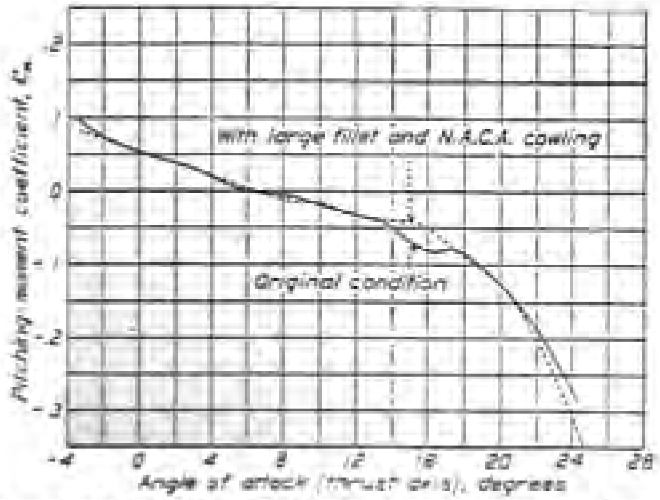


FIGURE [illegible].—Pitching moments of McDonnell airplane with power on. Corrected for tunnel effects. Stabilizer [illegible]° to thrust axis. Elevator 0° to stabilizer.

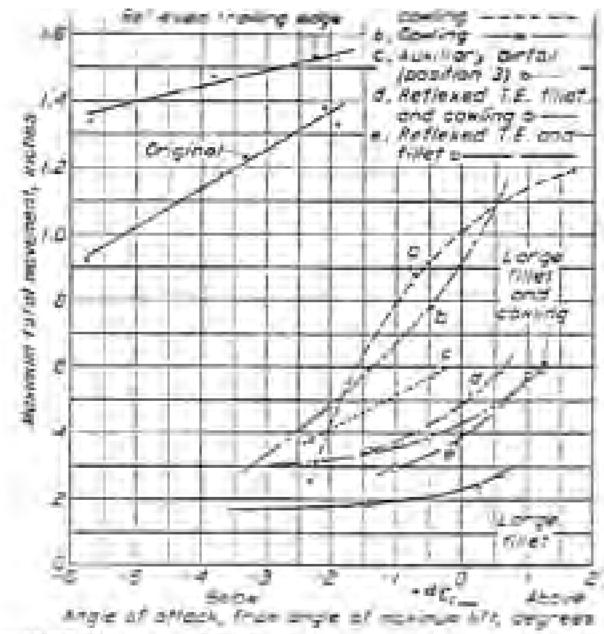


FIGURE [illegible].—Amplitude of stabilizer-tip movements under various conditions. Angle of attack corrected for tunnel effects. Power off. Air speed approximately [illegible] m.p.h.

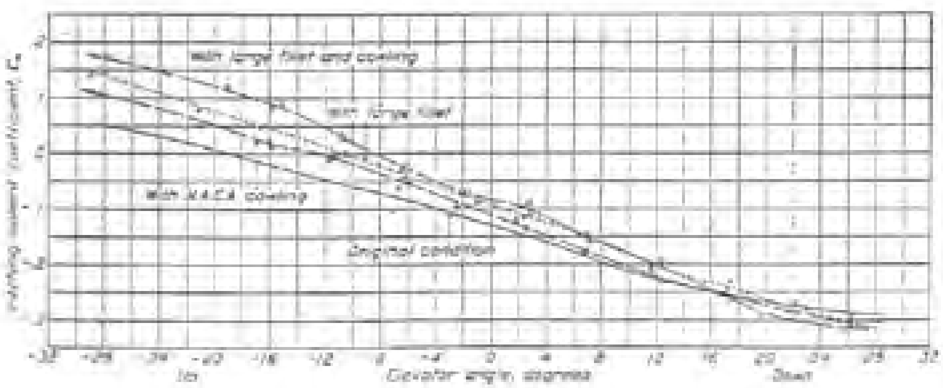


FIGURE [illegible].—Elevator effectiveness of McDonnell airplane with various devices. Corrected for tunnel effects. Power off. Angle of attack (thrust axis) = [illegible]

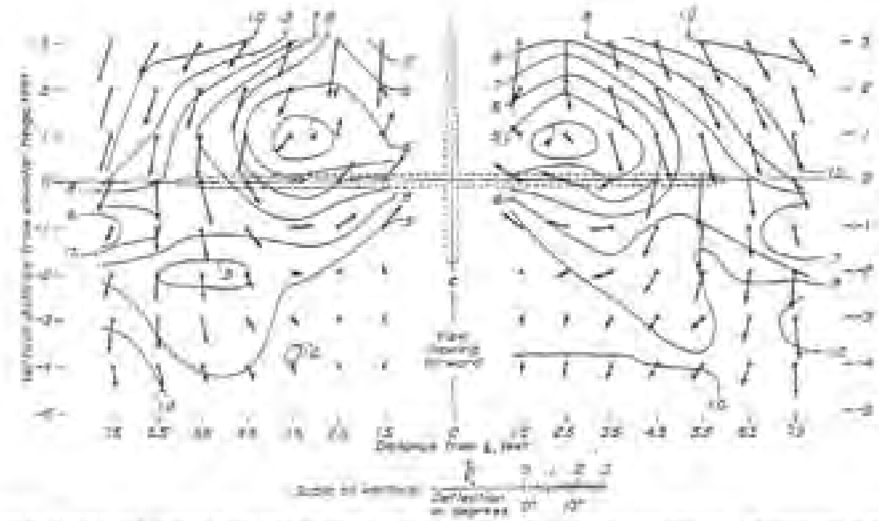


FIGURE [illegible].—[illegible]

INTRODUCTION

The increasing use of low-wing monoplanes has emphasized the susceptibility of this type of airplane to detrimental interference at the intersection of the wing and fuselage. In addition to decreasing the aerodynamic efficiency, this interference often causes a loss of longitudinal control and a violent shaking, or buffeting, of the tail of the airplane by the eddying wake from the wing roots. Tail buffeting may become so severe in some cases as to endanger the tail structure. In at least one instance it was considered as a possible cause of the failure of a low-wing monoplane that broke to pieces in the air (references 1 to 4, inclusive).

Methods have been suggested for reducing or eliminating wing-fuselage interference and buffeting, and some tests have been conducted on small-scale models and in flight (references 2 and 5 to 9, inclusive). This report covers the results of tests conducted in the N.A.C.A. full-scale wind tunnel on a low-wing monoplane that was subject to tail buffeting. The tests included an investigation of the wing-fuselage interference and buffeting with the airplane in its original condition and with various devices installed to eliminate or reduce the detrimental effects. As the detrimental effects appear to be directly due to a premature breakdown of the flow at the wing-fuselage intersection, the devices were designed with a view to their ability to postpone this breakdown of the flow to the angle of attack at which the entire wing stalls. The devices tested were two different wing-fuselage fillets, an N.A.C.A. cowling, a reflexed trailing edge next to the fuselage, auxiliary airfoils of short span in three different positions, and various combinations of the above.

The value of the various devices was determined by visual observation of the air flow at the wing-fuselage intersection by means of strings; measurements of the lift, drag, and pitching moments of the airplane; records of the vibrations of the tail; and surveys of the direction and speed of the air flow at the tail of the airplane, including records of the fluctuations of the air speed. Observations were made both with and without the slipstream from the airplane propeller.

Part of the results given here have been previously published as a technical note (reference 10).

APPARATUS

Wind tunnel.—The tests discussed in this report were conducted in the N.A.C.A. full-scale wind tunnel. The wind tunnel, the balance for measuring the forces and moments, and the apparatus used for determining the air speed and direction at any point in the jet are described in reference 11.

Airplane.—The *McDonnell* airplane, a low-wing monoplane originally built for entry in the Daniel Guggenheim Safe Aircraft Competition in 1929, was chosen for these tests because it was reported by pilots to be subject to tail buffeting. Flight tests of the *McDonnell* airplane are described in reference 12. Figure 1 is a photograph of the airplane mounted in the wind tunnel; figure 2 is a 3-view drawing showing its principal dimensions; and figure 3 is a view of the intersection of the wing and fuselage. The airplane is equipped with a Warner Scarab engine having a rating of

110 horsepower at 1,850 r.p.m. The airplane is provided with movable leading-edge slots and trailing-edge, (FIGURE 1.—The *McDonnell* airplane with large fillet in full-scale wind tunnel.), flaps, but for these tests the slots were covered with doped fabric and the flaps locked in the neutral position. After preliminary tests had been made, a walkway that extended from the fuselage to 10 inches outboard and raised the top surface of the right wing five eighths of an inch above the normal profile from 15 to 69 percent of the chord was removed, and the gaps between the wings and fuselage, which were as much as 3 inches wide on the under side, were covered. The stabilizer was set at an incidence of 0.6° with respect to the thrust axis for all the tests and, except when elevator effectiveness was being measured, the elevator was locked in the neutral position.

Fillets.—The wing-fuselage fillets were designed to reduce the rate at which it was necessary for the air in this region to diverge in order to follow the surfaces. The radius was small at the leading edge and a short distance back started increasing smoothly to a maximum at the trailing edge, behind which the fillet was faired into the fuselage. The principal difference between the two fillets was in size, hence they will be referred to as the “small fillet” (figs. 4 and 5) and the “large fillet” (figs. 6 and 7). Another difference was that the small fillet had a constant radius from the leading edge back to 41 percent of the chord, whereas the radius of the large fillet began to increase at 6.6 percent of the chord back of the leading edge.

N.A.C.A. cowling.—The N.A.C.A. cowling (fig. 8) consisted of a hood that was placed over the engine and nose of the airplane without alteration being made in the original fuselage lines. The hood was designed in accordance with the information in reference 13, except that its cross section did not resemble an airfoil profile because it consisted of only one thickness of metal.

Reflexed trailing edge.—The modification of the wing root, herein called a “reflexed trailing edge” (fig. 9), was designed to decrease the incidence at the wing root. The lower surface of the wing, which had an upward curvature (N.A.C.A.—M6 section), was extended to the rear and a new upper surface formed of straight-line elements from the new trailing edge to the points of tangency with the upper surface of the original wing. The fillet tested in combination with this reflexed trailing edge (fig. 10) was similar to the large one previously described.

Auxiliary airfoils.—The auxiliary airfoils used in these tests were of the N.A.C.A. 22 section, had a 10-inch chord (14.7 percent of the main wing chord), and extended 30 inches from the fuselage on each side. They were tested in three positions near the leading edge of the wing (see fig. 15), the first position being similar to that found to be the optimum in the investigation reported in reference 14.

METHODS

Air flow at wing roots.—The air flow at the wing roots was studied by noting the behavior of a lightweight string on the end of a slender stick held by an observer in the cockpit.

Force and moment measurements.—The power-off lift, drag, and pitching moments were all measured with the propeller removed. The power-on measurements were made with the propeller turning at such speed that its thrust just balanced the drag of the airplane (due allowance being made for jet-boundary effect), thus simulating steady level-flight conditions. As the jet-boundary corrections could be only estimated beforehand, it was not feasible to adjust the engine speed so as to give exactly zero net drag. Therefore, three readings were taken at each angle of attack at three propeller speeds near the proper value and the value of lift for zero net drag was found from a plot of these points against net drag. All tests were made at an air speed of 55 to 60 miles per hour except in the case of the power-on tests, where at high angles of attack it was necessary to reduce the speed to keep the drag within the range of the available thrust.

Records of tail buffeting.—The vertical movements of the tip of the stabilizer were recorded on a moving film by means of an N.A.C.A. control-position recorder. From these records the amplitude and frequency of the motions of the tail surfaces were determined. The instrument was mounted on a solid base and connected to the stabilizer by an 0.008-inch diameter piano wire shielded from the wind by a steel tube. (FIGURE 2.—Three view drawing of the *McDonnell* airplane). The natural frequency of the piano wire and instrument was about 34 cycles per second, which is almost four times the highest frequencies recorded. Play and friction in the instrument caused errors in indicated amplitudes of the vibrations probably not exceeding one eighth inch.

During most of these tests the tail of the airplane was supported by a rigid A-frame fastened to the tail-post. In order to determine the effect of this rigid support, records were made of the movements of the stabilizer tip and the rear end of the fuselage while the tail of the airplane was free from external support, the airplane being prevented from turning about the main supports at the landing-wheel axles only by cables secured to the forward part of the fuselage.

Most of the records were taken at an air speed of approximately 58 miles per hour, but a few were taken at speeds between 35 and 60 miles per hour to determine the effect of change in speed.

Air flow at tail.—The direction and speed of the air flow at the tail in a vertical plane through the elevator hinge line were measured with all the tail surfaces removed, using the combined pitot-static, yaw, and pitch tube and auxiliary apparatus described in reference 11. In addition to the measurements of average speed and direction, several records were made with a recording manometer connected to the pitot tube to determine the frequency of the air-speed fluctuations in the wake from the wing roots and the relative magnitudes of the fluctuations at different positions near the tail. These records were not entirely satisfactory because of the large amount of damping in the long rubber tubes required to reach from above the air stream down to the pitot tube near the tail of the airplane. Consequently the true magnitude of the fluctuations cannot be determined from these records; however, some idea of the frequencies involved can be obtained.

RESULTS

Air flow at wing root.—The action of the string held in the region of the wing-fuselage intersection indicated that, except when the airplane was equipped with some of the most effective devices, the air flow over the upper surface of the wing began to break down near the intersection of the wing and fuselage and that the turbulent region spread laterally as the angle of attack was increased. With the airplane in the original condition the turbulent flow extended approximately 3 feet outboard from the fuselage at 14° angle of attack. The approximate angles of attack at which the air flow over the root of the wing first burbled, (FIGURE 3.—Wing-fuselage intersection of *McDonnell* airplane), when the airplane was equipped with the various devices with power off were as follows:

Original condition ---------------------------------5°.
Small fillet--12°.
Large fillet ---15°.
N.A.C.A.cowling----------------------------------14°.
Small fillet and N.A.C.A. cowling---------------17° (at stall).
Large fillet and N.A.C.A. cowling---------------17° (at stall).
Reflexed trailing edge -----------------------------7°.
Reflexed trailing edge and N.A.C.A. cowling-16° (at stall).
Reflexed trailing edge and fillet-------------------Above stall.
Reflexed trailing edge, fillet, and N.A.C.A.
cowling --Above stall.
Auxiliary airfoil in position 1----------------------7°.
Auxiliary airfoil in position 2--------------------- 7°.
Auxiliary airfoil in position 3--------------------- 10°.

When the auxiliary airfoils were used, vortices trailing from their tips were evident. When the N.A.C.A. cowling was used, particularly in combination with any of the fillets and both with and without the slipstream, the action of the string indicated the presence of trailing vortices approximately concentric with the fillets.

The direction of rotation of these vortices was the reverse of what it would be for vortices corresponding to a loss of lift at the center section.

Lift and drag characteristics.—The power-off lift and drag data are presented in four groups of polar and lift and drag curves. The first group (figs. 11 and 12) compares the various fillets and fillet combinations; the second (figs. 13 and 14) shows the effects of the reflexed trailing edge alone and with the cowling and fillet; the third (figs. 15 and 16) shows the effects of the auxiliary airfoil in three positions; and the fourth (figs. 17 and 18) shows the effects of the cowling, (FIGURE 4—Small fillet on *McDonnell* airplane), alone and summarizes the other groups. In addition, a representative polar is shown with the experimental points (fig. 19). The theoretical induced-drag curve based on the geometrical aspect ratio of the wing (6.2) is included with each group of polars.

Power-on lift curves, corresponding to level flight, are presented for the original condition and for the condition with the large fillet and cowling (figs. 20 and 21). All the other conditions tested gave results practically the same as those for the large fillet and cowling. No means were available for determining the thrust of the propeller, so it was not possible to determine exactly either the effect of the slipstream on the drag characteristics of the airplane or what part of the total lift was due to the vertical component of the propeller thrust. An approximate correction for this vertical component of thrust was applied, however, in order to, (FIGURE 5—Drawing of the small fillet), make the difference between the power-off and power-on lift curves more nearly represent the effect of the slipstream; the lift curves are shown both with and without this correction. These approximate corrections were arrived at by computing, for each angle of attack, the vertical component of a thrust large enough to overcome the drag of the airplane without the slipstream.

All coefficients are based on the original wing area of 196.5 square feet. The added area due to the addition of the large fillet and the reflexed trailing edge amounted to about 2.5 percent and 7 percent, respectively.

Pitching moments.—Curves of pitching moments about the center of gravity plotted against angle of attack are shown for the power-off condition in figure 22. Curves of pitching moments with the tail surfaces removed and pitching moments due to the tail alone are shown in figure 23. Figure 24 shows the pitching-moment curves for two power-on conditions. The power-on pitching moments were found to be practically the same for all conditions. The influence of several of the devices on elevator effectiveness is shown by curves of pitching moment plotted against elevator angle for an angle of attack just below the stall (fig. 25). The pitching-moment coefficients are, (FIGURE 6.—Large fillet on *McDonnell* airplane), based on the original wing area (196.5 square feet) and the original mean chord of 5.62 feet.

Tail buffeting.—Typical records of the motion of the stabilizer tip are shown in figure 26. Curves of the maximum amplitudes of tail vibrations for various conditions of the airplane are shown in figure 27. Amplitude is here considered as the deflection between adjacent extremes of the up-and-down motion and is given in inches of motion normal to the plane of the stabilizer. The amplitude of stabilizer-tip movements with the propeller operating is not included in figure 27 because it did not vary consistently enough to permit the drawing of curves. Nearly all the maximum deflections measured with power on fell between 0.1 and 0.4 inch for angles of attack below the stall. The values in figure 27 were all obtained with the rear end of the fuselage rigidly supported. When it was free from external support the amplitude of stabilizer-tip movement was nearly doubled and the vertical movement of the rear end of the fuselage itself was only about one fifth as great as that of the stabilizer tip. Figure 28 shows the variation in amplitude with changes of air speed between 35 and 60 miles per hour. The natural frequencies of the stabilizer were as follows:

Vibrations per second

With rear end of fuselage rigidly supported---7.3

With rear end of fuselage unsupported---8.5

For each method of support the predominant frequency of the tail vibrations caused by buffeting was approximately the same as the corresponding natural frequency.

The stiffness of the stabilizer and fuselage was such that, when the rear end of the fuselage was externally supported, the stabilizer tip was deflected 1 inch by a force of 60 pounds concentrated at the tip.

Air flow at tail.—The surveys of the air flow at the tail are shown by dynamic-pressure contours and direction vectors (figs. 29 to 33, inclusive). The contours show lines of equal dynamic head expressed as the ratio of measured dynamic head to the dynamic head at the same point in the air stream with the airplane removed. The vectors show the component of the velocity in the plane of the survey, that is, normal to the tunnel axis. The length of the vector shows the magnitude of the component velocity v relative to the total velocity V_o in the direction of the flow at the point considered and therefore is also a measure of the angular deflection of the air flow from its initial direction parallel to the tunnel axis. When, as in this case, the angular deflections are relatively small, the scale of vector lengths can be divided so as to give directly the deflection in degrees in any direction from the tunnel axis by scaling the proper component of the vector. Thus, the angles of downwash and yaw of the air flow can be determined directly by scaling the vertical and horizontal components of the vectors. The surveys are presented with the vector scale graduated in terms of both v/V_o and the angular deflection from the tunnel axis. A specimen record of the fluctuation in dynamic pressure at the tail is shown in figure 34.

Wind-tunnel corrections.—All results except the velocity-component vectors shown on the surveys of air flow at the tail are corrected for tunnel effects.

DISCUSSION

Air flow at wing roots.—The visual observations of the air flow at the wing roots showed that the interference caused a premature stalling of the wing at that point. Several factors tend to cause this section to stall prematurely: The presence of the fuselage, which tapers to the rear and toward the bottom, increases the volume into which the air coming over the wing in that region must diverge; the side of the fuselage offers additional frictional resistance increasing the adverse pressure gradient; and the large drag of the engine absorbs much kinetic energy from the air and makes it less able to overcome the adverse pressure gradient. The observations showed that the disturbance started in this region at an angle of attack as low as 5° for the airplane in the original condition. The use of devices which either decreased the rate at which the air flow, (FIGURE 8.—Nose of *McDonnell* airplane in original condition and with N.A.C.A. cowling.), had to diverge or increased the kinetic energy of the air next to the fuselage postponed the breakdown of flow to much higher angles of attack.

Lift and drag characteristics.—A comparison of the polar for the airplane in the original condition (fig. 19) with the theoretical induced-drag polar for the whole wing (aspect ratio = 6.2) and for the portion at one side of the fuselage (aspect ratio =2.9) agrees with the observations of the air flow at the wing roots in indicating that even at relatively low angles of attack the smooth flow over the wing broke down next to the fuselage so that the part of the wing on each side of the fuselage tended to act independently as a wing of low aspect ratio.

The effectiveness of the fillets and N.A.C.A. cowling in preventing the premature break down of flow at the wing-fuselage intersection is attested by the straightness of the lift curves and the parallelism of the polars to the induced-drag polar as seen in figures 17 and 18. Both the large fillet and the N.A.C.A cowling postponed the breakdown of the flow to within 3° of the angle of maximum lift, although the double curve near maximum lift when the cowling was used alone indicates an unstable state or flow at high angles of attack. Figures 17 and 18 also show that the reflexed trailing edge increased the angle of attack at which the flow started to break down by about the same amount that the incidence of the wing at the root was changed (2° or 3°), but once the flow started to break down the reflexed trailing edge had little effect. The improvement due to the auxiliary airfoils in the best position tested was only about half as much as that due to the fillets or the N.A.C.A cowling. It is possible, however, that this is not the optimum position for the airfoils, as only three positions were tested.

When used alone the large fillet was found to give slightly better lift and drag characteristics than the small one, as shown by comparison of the two polars (fig. 11); but when used with the N.A.C. A. cowling the results were practically identical.

In addition to its effect on the wing-fuselage interference the N.A.C.A. cowling gave a large reduction in parasite drag. The minimum drag coefficient was reduced from 0.0637 to 0.0590 by the cowling; to 0.0625 by the large fillet; and to 0.0580 by the combination of large fillet and cowling.

The best lift and drag characteristics were obtained when the large fillet and N.A.C.A cowling were used together. The use of this combination eliminated most of the wing-fuselage interference, increased the maximum lift 11 percent above its original value, decreased the minimum drag 9 percent, and increased the maximum lift/drag ratio 19 percent.

The slipstream prevented a premature break down of the flow near the wing-fuselage intersection in all except the original condition and even in this condition the improvement was very great (figs. 20 and 21). In the original condition the lift curve begins to break over at almost the same angle of attack (about 6°) as without the slipstream; but as the angle of attack was increased, corresponding to a lower flying speed in level flight, the slipstream velocity became much greater relative to the air speed until it was sufficient to smooth out the flow, and at 12° the lift was almost as high as when the large fillet and cowling were used. Beyond 12° the flow apparently started to the slipstream for maintaining the smooth flow, especially during landing.

Preliminary tests showed that the presence of the raised walkway next to the fuselage had no appreciable effect on the characteristics of the airplane equipped with the small fillet and that removing the walkway and covering the gaps between the wing and fuselage when the airplane was not equipped with any of the special devices had a negligible effect.

The maximum lift coefficient of the airplane in its original condition, as determined by these tests, was considerably higher than the highest value measured in flight with slots closed and flaps neutral (reference 12). This difference was due to the fact that in flight the pilot was not able to maintain steady conditions long enough to take satisfactory records at angles of attach above 16°.

Pitching moments.—Improving the air flow at the wing roots resulted in a slight decrease in longitudinal stability (fig. 22), due mainly to the increased downwash at the tail (fig. 23). The curves of pitching moments with elevator neutral and with power on presented in figure 24 show that there is very little difference in pitching moments between the various conditions with power on.

The effectiveness of the elevator (fig. 25) was increased by the devices that reduced the wing-fuselage interference, probably because of the higher velocity of flow over the tail (figs. 29 to 33, inclusive). Additional data taken at other angles of attack showed that the improvement extended over about the same angle-of-attack range as the corresponding improvement in lift and drag characteristics (from about 8° to beyond the stall). (FIGURE 9.—Drawing of the reflexed trailing edge).

Tail buffeting.—The effectiveness of the various devices in reducing tail buffeting is clearly shown in figure 27. The oscillations due to buffeting were reduced to amplitudes small enough to be considered unobjectionable throughout the range of normal flight attitudes by the use of the fillets, either alone or in combination with the N. A.C.A. cowling or the reflexed trailing edge. The use of the large fillet alone gave the least buffeting, reducing the oscillations to one seventh their original amplitude. The use of this fillet with the cowling, the combination giving the best lift and drag characteristics, reduced the vibrations to one fourth their original amplitude. The slipstream was practically as effective as the fillets.

In general, the various devices decreased the buffeting in about the same proportion that they improved (FIGURE 10.—Reflexed trailing edge with fillet on *McDonnell* airplane) the lift and drag characteristics. The N.A.C.A. cowling was an exception to this rule because whenever it (FIGURE 11—Polars for McDonnell airplane with various fillets. Corrected for tunnel effects), was used the buffeting was greater than would have been expected from the improvement in the polar. This excessive buffeting was probably due to the vortices mentioned in connection with the observations of the air flow at the wing root and seen on the survey of the air flow at the tail (fig. 32). (FIGURE 12.—Lift and drag of McDonnell airplane with various fillets. Corrected for tunnel effects).

The records of the stabilizer-tip movements (fig. 26) show the nature of the vibrations. It will be noted that the vibrations had a quite definite frequency, (FIGURE 13—Polars for McDonnell airplane with reflexed trailing edge. Corrected for tunnel effects), which was practically the same as the free-vibration frequency of the stabilizer. The amplitude, however, was so irregular that to an observer the motion looked like a haphazard shaking of the tail. There appeared to be very little deflection of the stabilizer and elevator as a beam, most of the deflection being due to twisting of the fuselage.

The vibrations of the stabilizer obtained under the conditions of these tests afford good comparisons between, (FIGURE 14), the degrees of buffeting under the various conditions tested, although the results of the special tests made with the rear end of the fuselage unsupported, (FIGURE 15—Polars for *McDonnell* airplanes with auxiliary airfoils. Corrected for tunnel effects. Power off), indicate that in actual flight the magnitude of the oscillations would be about twice as great as the values given in figure 27. The frequency is apparently dependent upon the natural frequency of the tail structure, which is slightly higher with the tail unsupported.

The severity of buffeting was shown to increase rapidly with increase in air speed between 35 and 60 miles per hour (fig. 28). (FIGURE 16—Lift and drag of *McDonnell* airplane with auxiliary airfoils. Corrected for tunnel effects. Power off.) It cannot be assumed, however, that this rate of increase would continue at velocities, (FIGURE 17—Polars for *McDonnell* airplanes comparing various devices. Corrected for tunnel effects. Power off.), above those investigated, as the relations may be affected by resonance between the natural frequency of the tail and the frequency of the buffeting eddies.

Air flow at tail.—The surveys of the air flow at the tail of the airplane substantiate the observations from the other data in regard to the effects of the wing-fuselage, (FIGURE 18—Lift and drag of *McDonnell* airplane comparing various devices. Corrected for tunnel effects. Power off.), interference on the air flow and lift distribution and indicate in more detail how elimination of the interference reduced tail buffeting. The lift distribution, (FIGURE 19—Polar for *McDonnell* airplane in original condition. Corrected for tunnel effects. Power off.), near the fuselage for the various conditions is indicated by the downwash vectors and also by the vertical position of the wake from the wing roots. For the original condition, prominent vortices next to the fuselage (figs. 29 and 30) show that the part of the wing on each side of the fuselage tended to act as a separate wing, (FIGURE 20—Power-on lift of *McDonnell* airplane in original conditions. Corrected for tunnel effects. And FIGURE 21—Power-on lift of *McDonnell* airplane with large fillet and N.A.C.A. cowling. Corrected for tunnel effects.), with its pair of tip vortices. These vortices produced an upflow of the air near the fuselage which probably increased the tail vibrations by causing part of the horizontal surfaces to be stalled. In the improved conditions, such as that with the large fillet (fig. 31) and with the N.A.C.A. cowling (fig. 32), the turbulent wake from the wing roots was greatly reduced. (FIGURE

22—Pitching moments of McDonnell airplane with various devices. Corrected for tunnel effects. Power off. Stabilizer 0.6° to thrust axis. Elevator 0° to stabilizer.) (FIGURE 23—Pitching moments due to tail of McDonnell airplane. Corrected for tunnel effects. Power off. Stabilizer 0.6° to thrust axis. Elevator 0° to stabilizer.) The power-on survey (fig. 33) shows that even in the original condition the slipstream practically eliminated the vortices due to the wing-fuselage intersection.

Judging from the air-flow surveys for the original condition, it would not be possible to reduce the tail buffeting materially by moving the horizontal tail surfaces upward for any reasonable distance (figs. 29 and 30). Lowering the tail surfaces about 2 feet would cause quite an improvement, but would bring the stabilizer down near the bottom of the fuselage—an impracticable location. In any case, since the interference that causes tail buffeting also causes a loss in aerodynamic efficiency, it appears best to cure the trouble at its source by methods such as those used in this investigation. (FIGURE 24—Pitching moments of *McDonnell* airplane with power on. Corrected for tunnel effects. Stabilizer 0.6° to thrust axis. Elevator 0° to stabilizer.)

The specimen record of the fluctuations in dynamic pressure at the tail (fig. 34) shows that although the fluctuations were very irregular they had some semblance of a definite frequency. It was very difficult to determine definitely either this frequency or the true magnitude of the fluctuations from the records taken, owing to the irregularity of the changes and to the large amount of damping introduced by the connecting tubes. In spite of these difficulties, however, after a careful study of the records, the following conclusions in, (FIGURE 25—Elevator effectiveness of *McDonnell* airplane with various devices. Corrected for tunnel effects. Power off. Angle of attack (thrust axis) = 15.8°), regard to air-flow conditions at the tail seem to be justified, although they cannot be considered as definitely proved: (FIGURE 27.—Amplitude of stabilizer-tip movements under various conditions. Angle of attack corrected for tunnel effects. Power off. Air speed approximately 58 m.p.h.)

1. The principal frequency of fluctuations in the wake from the wing-fuselage intersection for the airplane in the original condition was close enough to the natural frequency of the tail to indicate the possibility of resonance (fig. 34). These high-frequency fluctuations (7 or 8 per second) are of much greater magnitude, (FIGURE 28.—Variation in stabilizer-tip movements with changes in air speed. Power off. *McDonnell* airplane in original condition. Angle of attack (corrected for tunnel effect), 3.7° below α_{CLmax}), relative to the lower frequency changes than the record indicates because high-frequency fluctuations are damped much more than slow ones. (FIGURE 29.—Air flow at tail of *McDonnell* airplane in original condition, power off. Survey in vertical plane through elevator hinge line. The contours show the ratio of the dynamic pressure behind the airplane to the dynamic pressure in the free air stream. The vectors show the components of the velocity in the plane of the survey. Angle of attack (thrust axis) =14.2° (corrected for tunnel effects). Lift coefficient = 0.984.)

2. Improving the flow at the wing root increased the frequency of the eddies in the wake to approximately 50 percent greater than that for the original condition.

3. Over the range tested, from 37 to 58 miles per flour, the frequency of the fluctuations appeared to vary proportionally with the velocity.

4. In addition to the fluctuations with a fairly definite frequency there were also irregular and sudden "bumps."

It is difficult to say just how much of the buffeting motion was due to trains of oscillations set up by the bumps mentioned in item 4 and how much was due to the more regular air fluctuations of about the same frequency as the natural frequency of the tail. Undoubtedly, some of the reduction in buffeting for improved conditions of the airplane was due to the frequency of the eddies having been increased to a value well above the natural frequency of the tail.

CONCLUSIONS

The following conclusions are drawn from the tests on the *McDonnell* airplane. Differences in engine, fuselage shape, and wing section and location might modify the results for other low-wing monoplanes.

1. In addition to the presence of sudden changes or bumps, the eddying wake from the wing roots had a predominant frequency of fluctuation of the order of the natural-vibration frequency of the tail, although the fluctuations were very irregular, which suggests that the magnitude of the tail vibrations was very probably influenced to some extent by resonance effects. (FIGURE 34—Fluctuations in dynamic pressure at tail of *McDonnell* airplane in original condition, power off. Angle of attack (thrust axis) =14.2° (corrected for tunnel effects).)

2. Fillets without cowling reduced the wing-fuselage interference and tail buffeting to unobjectionable magnitudes throughout the range of normal-flight attitudes.

3. The N.A.C.A. cowling without fillets reduced the wing-fuselage interference and tail buffeting to unobjectionable magnitudes at angles of attack up to within 3° of the stall.

4. The reflexed trailing edge had a minor effect, slightly increasing the amplitude of tail oscillations due to buffeting.

5. The auxiliary airfoils, in the positions tested, gave some improvement but were considerably inferior to the fillets.

6. Buffeting was least when the large fillet was used alone. This fillet reduced the amplitude of stabilizer tip oscillations from the 1.37 inches obtained with the airplane in the original condition to 0.18 inch at an angle of attack 2° below the stall.

7. The combination of the large fillet and the N.A C.A. cowling gave the best all-round results. This combination reduced the total amplitude of stabilizer tip oscillations at an angle of attack 2° below maximum lift from the original 1.37 inches to 0.32 inch, increased the maximum lift 11 percent, decreased the minimum drag 9 percent, increased the maximum lift/drag ratio of the airplane 19 percent, and increased the effectiveness of the elevator about 40 percent at angles of attack in the landing range.

8. The slipstream was practically as effective as the fillets in reducing tail buffeting.

9. The use of fillets or other devices for eliminating wing-fuselage interference slightly decreased the longitudinal stability of the airplane.

LANGLEY MEMORIAL AERONAUTICAL LABORATORY,
NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS,
LANGLEY FIELD, VA., December 13, 1933.

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{AQ1: Should figure legends be placed at the end of the text ?}

Document 3-18(a-c)

(a) Shatswell Ober, "Report on Wind Tunnel Model Tests, Ford Tri-Motor Plane—4-AT," 17 February 1927, Accession 18, Box 80, Folder "Wind Tunnel Report," Henry Ford Museum and Greenfield Village Research Center, Dearborn, Mich.

(b) Aircraft Engineering Department, Ford Motor Company, "Recent Design Changes on the 5-AT and their Effect on High Speed," undated (ca. 1929), Accession 18, Box 70, Folder "#6, Report on Fairing on 5-AT," Henry Ford Museum and Greenfield Village Research Center, Dearborn, Mich.

(c) "The Reminiscences of Harold Hicks," Aug. 1951, transcript, pp. 81-82, 106-108. Henry Ford Museum and Greenfield Village Research Center, Dearborn, Mich.

This trio of documents concerns the design of the Ford Trimotor—in common parlance, the "Tin Goose." This aircraft was a high-wing monoplane featuring an internally braced cantilever wing, fixed landing gear, and, as its name implies, three engines. In terms of basic configuration, the airplane resembled the Fokker trimotor, but the Ford transport was made entirely of metal, whereas the Fokker consisted of a mixture of woods, metal, and fabric. The airplane flew first in 1926 and stayed in production into 1933. It came in two versions, Model 4-AT and Model 5-AT, the latter carrying 13 to 15 passengers. A grand total of 200 of these aircraft were produced. They serviced airlines from coast to coast, at a normal cruising speed of a little less than 100 miles per hour. Aerodynamically, the transport was nothing special, with a relatively high drag coefficient ($C_{D,O}$) of 0.0471.

Nonetheless, the design reflected the growing ambition for aerodynamic streamlining. Both the 4-AT and 5-AT models were refined in wind tunnels. During January and February 1927, as we see in the first document below, engineer Shatswell Ober of the Aerodynamical Laboratory at MIT tested a model of the 4-AT transport for the Ford Motor Company. Although Ober's verbiage is sparse, his intention of aerodynamically refining the aircraft is clear.

The anonymously written "Recent Design Changes on the 5-AT and Their

Effect on High Speed," the second document below, related the intention of the Ford Aircraft Engineering Department to increase the high speed of their Trimotor design through aerodynamic means. The engineering staff weighed the advantages and disadvantages of design practices and innovations in terms of weight penalties, drag, and increase in speed, problems closely associated with the reinvention of the airplane discussed throughout this essay. It is clear from this document that Ford engineers were well aware of developments taking place elsewhere in the aeronautical community, even if they did not always use the same precise terminology.

The last selection involves the reminiscences of Ford chief engineer Harold A. Hicks, the man who oversaw the development of both Trimotor versions. This testimony from 1951 provides a wonderful insight into the design of the Ford Trimotor as well as the organizational environment in which it evolved. Hicks's comments are especially revealing because they shed light on the introduction of some of the key innovations of the airplane design revolution discussed in this chapter. NACA research on cowlings, airfoil sections, and engine placement all played a role in the design innovations incorporated by Ford.

Document 3-18(a), Shatswell Ober, "Report on Wind Tunnel Model Tests, Ford Tri-Motor Plane—4-AT," 17 February 1927.

FORD MOTOR COMPANY

AIRCRAFT ENGINEERING DEPT

DEARBORN MICH.

February, 1927

REPORT ON

WIND TUNNEL MODEL TESTS
FORD TRI-MOTOR PLANES 4-AT

To provide engineering data for the existing airplane and give information in regard to the aerodynamic effect of modifications, an extensive series of tests has been made on a model of the Ford 3-engine commercial monoplane type 4-AT. These tests were all made in the four foot wind tunnel at the Massachusetts Institute of Technology during January and February, 1927.

MODEL

The models were supplied by the Motor Company. They were well made with wings, tail, fuselages, nacelles and wheels of maple, and chassis, fin, and rudder of duralumin. The models were 1/40 full size, making a span of 20.6 inches with a net wing area of 67.8 square inches. Certain of the nacelle struts were omitted entirely in an effort to avoid excessive scale effect correction on these small parts. The original model 4-AT-3 has the Ford #2 wing, an open cockpit for the pilots, square bottom fuselage, and rounded top fuselage. Modifications consisted of two different wings, N.A.C.A. 81J and Ford #3, two different noses with closed cockpits designated as closed nose and long closed nose, a square top cowl and a small round cowl for the top of the square fuselage and a second fuselage rounded both top and bottom back of the wing. With the Ford #3 wing, additional bracing would be needed.

DIMENSIONS

It is convenient to have certain full scale dimensions. These have been enlarged from the model drawings.

Span overall (Ford #2 wing)	68.7 ft.
Net wing area (Ford #2 wing)	689 sq. ft.
Net wing area (N.A.C.A. 81J and Ford #3 wing)	677 sq. ft.
Mean wing chord (Ford #2 wing)	10.7 ft.
Aerodynamic aspect ratio	6.3
Horizontal tail area total	88.9 sq. ft.
Stabilizer area	49.8 sq. ft.
Elevator area	39.1 sq. ft.
Tail span	17.6 ft.
Aspect ratio	3.3
Distance C.G. to center of pressure of horizontal tail	31.1 ft.
Percent tail area of wing area	12.9%
Percent elevator area of tail	44%
Center of gravity located aft of leading edge at center	4.0 ft.
Below wing chord	1.33 ft.
Center of gravity percent of wing chord	32%
Power 3 Wright Whirlwind	600 HP
Wing loading (Ford #2 wing)	12.8 lb/sq/ft
Power loading	14.6 lb/H.P.
Gross flying weight	8800 lbs.

It is assumed that variations in weight and C. G. due to modifications are absorbed by variations in payload.

PROCEDURE

All tests were made in the four-foot wind tunnel at a wind speed of 40 miles per hour. The model was supported by a wing tip spindle tapering from 5/16" diameter to ¼" diameter. Conventional corrections for spindle drag and interference have been applied. The angle of attack is referred to the thrust line or the wing chord.

Tests were made as follows:

1. With original model 4-AT-3.
    a. Lift, drag and pitching moments with elevator neutral and three different stabilizer settings.
    b. Lift, drag, and pitching moments with elevator 25° up and two stabilizer settings.
    c. Lift, drag, and pitching moments with elevator 25° down and 12 ½ up and down at one stabilizer setting.
    d. Pitching moments without horizontal tail.
    e. Lift, drag, without tail and nacelles.
    f. Lift, drag, and moment, without tail, nacelles and chassis.
2. With modifications as noted, model complete with square fuselage:
    a. Lift, drag, and pitching moments with neutral elevator with:
        1. Closed nose round back cowl.
        2. Closed nose square back cowl.
        3. Closed nose small round back cowl.
        4. Closed nose without back cowl.

3. With Ford #2 wing and fuselage only:

    a. Lift, drag, with round fuselage and
        1. Open nose.
        2. Closed nose.
        3. Long closed nose.

        With square fuselage, long nose and
        1. Round back.
        2. Square back.

4. With model complete with Ford #3 wing, square body, small round back and long nose modified to fit wing:
    a. Lift, drag, and pitching moments with elevator neutral and one stabilizer setting.
    b. With these parts alone: Ford #2 wing
    N.A.C.A. 81 J wing
    Ford #3 wing
    Horizontal tail.

In addition some investigations of airflow around wing, fuselage, nacelles, and tail were made. Smoke was allowed to blow from a pipe into the airstream at different points, and the path of the axis of the jet of smoke was traced on a side view of the model. The wind speed was slow, either 15 or 20 miles per hour. These tests were made at several angles of attack with open and long closed nose. (The original airflow sketches were given directly to the Ford Motor Company's representative.)

COMPUTATIONS

To facilitate comparison with model tests of other airplanes, results have been expressed as coefficients. For tests with a fuselage in place, force coefficients have been found by dividing the force by the wing area, excluding the part above the body and the square of the wind velocity in miles per hour. The units are then lbs. per sq. ft. per (M.P.H.)^2. Pitching moments were referred to an axis through the center of gravity. The pitching moment coefficient K_M was found by dividing the moment by the net wing area, the mean wing chord, and the square of the air speed. These units are lbs. ft. per sq. ft. per (M.P.H.)^2 per ft. chord.

For tests of wings alone, the gross area was used. In addition, the center of pressure, defined as the point of intersection of the resultant force with the wing chord, was found. This is expressed as a fraction of the mean chord back of the leading edge of the wing at the center section.

Downwash was found by the method of moment differences only. The effective moment of the tail about the C.G. is found by the difference in pitching moments with and without tail plane. From the test of the tail alone, the moment of the tail without downwash is calculated. These curves of effective M_{CG} and calculated M_{CG} are plotted against angle of attack. Then at any angle on the effective M_{CG} curve, the downwash is the angular change necessary to give the same moment on the calculated M_{CG} curve.

RESULTS

The results are best considered by means of the plotted curves. Some tables of the more important data are also included. Table I gives full data for the original model, stabilizer 3 deg. elevator 0 deg., Table II complete model with Ford #3 wing, Table III lift and drag data for various fuselage and nose combinations, Table IV drag data of separate parts of original model. In general K_X and K_M are plotted against K_Y. Unless noted on the curves the Ford #2 wing was used.

DISCUSSION. LIFTS AND DRAGS

With neutral elevators the maximum lift coefficient on the original model is .00420, the minimum K_X .000198, for modified model with Ford #3 wing the maximum K_Y is .00364, the minimum K_X .000175, maximum L/Ds for these models are 9.5 and 10.2 respectively, and the ratios of maximum lift to minimum drag are 21.2 and 20.8. Some allowance must be made in the values for the plane with the Ford #3 wing for additional bracing. This might be about .000005 in K_X

at minimum drag, the maximum L/D would be reduced to about 9.9, and the ratio of maximum lift to minimum drag reduced to 20.8. Compared with tests on other models the lift with Ford #2 wing is high, but the drag is also high, the maximum L/D is good, and ratio of maximum lift to minimum drag fair.

With elevators turned up to trim at maximum lift, the maximum lift coefficient with #2 wing is .00400. The minimum flying speed corresponding is 57 M.P.H. Allowing a similar reduction, the minimum speed with #3 wing would be 62 miles per hour. Maximum speed estimated from minimum drag, 600 H.P., 80% efficiency for the two models should be 109 and 114 miles without allowance for scale effect. Speeds of about 115 and 119 miles per hour might be expected. Thus, the gain in high speed is nearly equal to the increase in landing speed.

CHANGES IN DRAGS

None of the various modifications of nose, fuselage, or back cowl have any serious effects on drag. The closed nose gives slight improvement, the long closed nose very slightly more, but from the open cockpit the improvement in K_X is only .000006, only 3%. The change in speed corresponding would be only 1 or 2 miles per hour. The form of back cowl has an almost negligible effect on the drag, except when the long closed nose is fitted, then the square cowl is not as good.

DRAGS OF PARTS

The total parasite drag of the model is affected by the wing section, as the total depth of fuselage varies with wing thickness. So with the Ford #2 wing, the total parasite at minimum drag is equivalent to 26.7 sq. ft. of flat plate, while with the #3 wing, it is 22.2 sq. ft. This difference is partly due to reduction in area of fuselage, partly to use of the better nose, and the rest to lessened interference. Against this reduction must be balanced about 1.0 sq. ft. of flat plate increase for extra bracing. The body has an effective drag of 8.7 sq. ft. with open cockpits, reduced perhaps to 7.5 sq. ft. with long closed nose. These flat plate areas are about 1/5 of the cross-sectional area. The chassis drag is some 9.8 sq. ft. of flat plate, naturally very high on account of the long struts. The nacelles together are equivalent to 6.4 sq. ft., about 1/3 of the total area to the outer ends of cylinders.

LONGITUDINAL STABILITY AND BALANCE

The curves of pitching moment about the center of gravity indicate ample longitudinal stability at all flying speeds, increasing as the speed decreases. The stability is not much affected by variations in back cowl, nose, or wing section.

With the Ford #2 wing, the open cockpit nose, and round back, a stabilizer setting of plus 3.0 deg. to the thrust line gives a trimming speed of 93 MPH. This occurs at an angle of attack of –1 deg. With closed nose, otherwise unchanged, the trimming speed is 94 MPH, practically the same. The form of back does affect the balance, however, the model with square back trims at 122 MPH, with the small round back at 98 MPH, with no cowl at 84 MPH. Except for the last, the stabilizer adjustment is ample to control the trimming speed, if such cowls should be fitted on the ship.

With the Ford #3 wing, the long nose modified to fit that wing, the square fuselage, and round back, the trimming speed with stabilizer plus .2° is 95 MPH. The engine of attack is plus 1 ½ deg.

ELEVATOR CONTROL

From these tests the elevator control appears ample. Even with the stabilizer set to trim at cruising speed, only about 17 deg. up elevator is needed to give equilibrium at maximum lift. With the stabilizer set full tail heavy –5.6 deg., the ship trims at only 58 MPH with neutral elevator.

DOWNWASH

The effective downwash may be found for moderate lifts by the equation 2° plus 1775 K_Y. The 2° is due to flow along the fuselage. The slope coefficient 1775 is about .8 of the theoretical mean downwash for this aspect ratio. It is to be expected that the downwash will be modified by the body. There is no indication of a reduction in downwash near the stalling point.

TESTS OF WINGS ALONE

The wings when tested alone give some 12% less total lift than the complete model. Most of the difference is made up by the fuselage, as with wing and body alone the lifts are nearly equal to those for the complete model. The maximum lift coefficients for wings alone are Ford #2, .00373, Ford #3 .00336, N.A.C.A. 81J, .00312; the minimum drag coefficients .000064, .000054, and .000056. The maximum L/Ds are 16.3, 18.0, and 15.9. As the N.A.C.A.J. appeared inferior, it was not tested with a fuselage or complete model.

AIRFLOW TESTS

The airflow tests did not lead to any very tangible results. Beyond the burble point, there is a distinct flow forward just above the trailing edge, but from all that could be observed, this did not extend back far enough to "blanket" the tail seriously. If there were any serious "blanketing" it would appear as a marked reduction in effective tail moment.

CONCLUSION

The question of using the thinner Ford #3 with external bracing should be given most careful consideration, both in regard to estimates of aerodynamic advantage, and changes in weight and type of structure. As to the former, it appears that the Ford #3 wing is somewhat better.

The question of open or closed nose is largely one of comfort and convenience of pilot. Aerodynamically the long, well faired, closed nose is somewhat superior.

The type of back cowl must be selected from structural or other consideration beside aerodynamic.

In regard to the wing angle, it appears that with the Ford #2 the incidence of the wing to the thrust line could well be reduced from zero to –1 deg. or –2 deg., as the plane flies nose down, even when cruising. With the Ford #3 wing the present angle of zero degrees might be increased to plus 1°.

Aerodynamic Laboratory Staff,
By—Shatswell Ober
Mass. Inst. Of Technology

TABLE I

Wind Speed: 40 M.P.H.
Model Size: 1/40 Scale
February, 1927

α	Lift	Drag	L/D	X	Z	M_{CG}	K_Y	K_X	K_M
–8	–.024	.1553	–.15	.157	–.046	.185	–.00004	.000225	.000084
–6	.278	.1403	1.98	.169	.262	.145	.00041	.000204	.000066
–4	.566	.1360	4.16	.175	.555	.091	.00082	.000198	.000041
–2	.890	.1405	6.33	.172	.884	.040	.00129	.000204	.000018
0	1.190	.1470	8.09	.147	1.190	–.042	.00173	.000213	–.000019
2	1.493	.1652	9.04	.113	1.498	–.198	.00217	.000240	–.000090
4	1.782	.1887	9.44	.064	1.791	–.359	.00259	.000274	–.000163
6	2.054	.2144	9.57	–.001	2.065	–.494	.00298	.000311	–.000224
8	2.312	.2497	9.25	–.075	2.327	–.621	.00335	.000363	–.000281
10	2.524	.2863	8.82	–.157	2.536	–.751	.00367	.000416	–.000337
12	2.735	.3322	8.22	–.245	2.742	–.853	.00397	.000482	–.000385
14	2.831	.3788	7.47	–.317	2.839	–1.048	.00411	.000550	–.000474
16	2.904	.4466	6.50	–.371	2.91	–1.263	.00421	.000649	–.000572
18	2.736	.6010	4.55	–.274	2.78	–1.462	.00397	.000872	–.000661

OC = angle referred to thrust line
3°.0 = angle of tail plane to thrust line
Elevators neutral
Tested in Aeronautical Laboratory 4 Ft. Wind Tunnel at M.I.T.

Document 3-18(b), Aircraft Engineering Department, Ford Motor Company, "Recent Design Changes on the 5-AT and their Effect on High Speed," undated (ca. 1929).

SUMMARY

Results of tests for High Speed with various units recently designed for the 5-AT have been compiled for reference. A statement is made of the most important improvements and the high speed which might be attained if all these units were incorporated on one airplane. That there are possibilities for still further improvement is indicated by a statement of several lines of development, some of which have already been demonstrated as practical and others which have yet to be proven.

The following conclusions are drawn from study of speeds obtained from the 5-AT as listed in the table:

The job with wide fuselage has the same top speed as with the standard fuselage.

Raising the wing has little if any effect on high speed. (0.6 MPH increase recorded but may be due to different prop. combination.)

Long outboard mounts. (PX-17862) with more clearance between wing and nacelle shows about 5 ½ MPH increase.

Propeller combination of 3 blade outboard and 2 blade center shows a slight advantage over 2 blades for all props. Less than 1 mile per hour is consistently recorded.

About ½ MPH increase is recorded for improved streamlines on landing gear struts SK-275-277.

Tail surface fairing SKD-297 shows a decrease of about ¾ MPH.

Nacelle to wing fairings (PX17799 revised) shows an improvement in high speed of about 3 ½ MPH. Note that this is on short mount.

Russel's outboard cowling shows about 3 ½ MPH increase.

Landing gear wheel fairings show about 1 ½ MPH increase.

Possible Improvement in High Speed with all Changes Incorporated on one Airplane.

	Increase in speed	Increase in weight
Russell's outboard cowling and Mayo exhaust	3.45	50 lbs.
Landing Gear Strut Fairing SK-275 and 277	.33	10
Nacelle to wing fairing similar to PX-17799	3.80	20
Long engine mounts PX-17862	5.50	12
3 blade outboard and 2 blade center props	.60	64
Wheel Fairing	1.60	50
Wing to Fuselage Fairing	.40	30
5-AT standard airplane	133.30 M.P.H.	
Total	148.98 M.P.H.	236 lbs.

RESULTS OF TESTS FOR HIGH SPEED

	Timed Speed (MPH)	Date Reported	Fuselage	Propellers	Nacelle C.G.	Cowl Cooling C.G.	Wing	Landing Gear	Tail Surfaces	Exhaust Pipes
[illegible]	[illegible]	11/11/39	Tide	3 Blade	Std Short	Std Nose to Wing	Std Low	Std	Std	
[illegible]	[illegible]	[illegible]	Tide	3 BL-[illegible] 3 BL-[illegible]	Std Short	Std Nose to Wing	High	Std	Std	
[illegible]	[illegible]	[illegible]	Tide	3 BL-[illegible] [illegible]	Long FX-17962	Nose to Wing Std	High	Std	Std	
[illegible]	[illegible]	[illegible]	Tide	3 BL - 421	Long FX-17962	Nose to Wing Std	High	Std	Std	
[illegible]	[illegible]	[illegible]	Tide	3 BL-421 [illegible] [illegible]	Long FX-17962	Nose to Wing Std	High	Std	Std	
[illegible]	[illegible]	[illegible]	Tide	3 BL-[illegible] 3 BL-C	Long FX-17962	Nose to Wing Std	High	Std	Std	
[illegible]	137.7	11/12/39	Std	3 BL - Std	Short Std	Nacel C.G.	Std	Flex Strut Fairing [illegible]	Std	

Page [illegible]

[illegible]

	Timed Speed (MPH)	Date Reported	Fuselage	Propellers	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible] Std	[illegible]	[illegible]		
[illegible]	[illegible]	[illegible]	Std	3 BL - Std	Short Std	Nacel C.G.	Std	[illegible] Fairing	Std
[illegible]	[illegible]	[illegible]	Std	3 BL - Std	Short Std	Nacel C.G.	Std	Flex Strut [illegible]	Std
[illegible]	[illegible]	[illegible]	Std	3 BL - Std	Short Std	Nacel C.G.	Std	Std	Std
[illegible]	[illegible]	[illegible]	Std	Std	Std	Nacelle C.G.	Std	Std	Streamline [illegible]
[illegible]	[illegible]	[illegible]	Std	Std	Std	Nacel C.G.	Std	Std	Std
[illegible]	[illegible]	[illegible]	Std	Std	Std	Nacelle to Wing FX-17999 (Original)	Std	Std	Std
[illegible]	[illegible]	[illegible]	Std	Std	Std	Nacelle to Wing FX-17999 (Improved)	Std	Std	Std
[illegible]	[illegible]	[illegible]	Std	Std	Std	Std	Std	Wheel Fairing	Std

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RESULTS OF TESTS FOR HIGH SPEED (Cont'd)

	Timed Speed (MPH)	Date Reported	Fuselage	Propellers	Nacelle C.G.	Cowl Cooling C.G.	Wing	Landing Gear	Tail Surfaces	Exhaust Pipes
[illegible]	[illegible]	9/24/39	Std	Std	Std	Std	Std	Std	Std	
[illegible]	[illegible]	10/ 3/39	Std	Std	Std	Std	Wing to Fuselage	Std	Std	
[illegible]	[illegible]	10/5/39	Std	Std	Std	Std	Std	Std	Std	

<u>Possibilities for Further Streamlining</u>

1. The use of fairing between the wing and nacelle gave a marked improvement on the short mount installation. The lengthening of the mounts giving more gap between nacelle and wing also improved high speed. The fairing of the long mount between nacelle and wing has not been tried as yet and seems to have possibilities for an added improvement. Would suggest that the radius where this fairing joins the wing be made as large as possible to properly join the airflow around the wing and the streamline. (Ref. PX-17799)

2. Mr. Manning's report of observations with strips of tape shows a very imperfect air flow around the nacelle. This condition is probably due to rotating airflow in the slipstream and the streamlining at somewhat of an angle with resultant direction of flow. Possibly we could gain an advantage by placing the streamlines on the struts at an angle to favor this resultant and thus change the direction of flow less abruptly.

3. The use of nacelles of greater length—diameter ratio is a possibility, although it has probably been previously investigated for our conditions. The length now given is of the Navy strut form, but recent propeller slipstream investigations show a lower pressure area than the remainder of the propeller disc, 15 ft. aft of the propeller.

4. The working of the exhaust manifold into an annular ring of streamline cross section with enough cowling effect to help to pay for at least its own resistance might help. Another idea along the same line is to have the exhaust outlet at the trailing edge of such an annular streamline manifold, thus throwing the exhaust gases in a direction to straighten out the flow around the nacelle. These gases, warmer than surrounding air, might set up a flow around the nacelle with beneficial lubricating effect on the much discussed boundary layer. In fact, it would be possible to approximate or better the effect produced by blowing air from slots on surfaces of bodies to decrease their resistance (German experiments about 1926).

All this has double possibility since it uses to advantage something which was formerly carried at a loss.

5. Further research on the correct way to fair the wing into the fuselage on top surface might have possibilities. Suppose experiments with 10-A models have already given considerable information on this subject.

6. Less abrupt projections around the cockpit and experiments with various cockpit enclosures seem a possibility for decreasing resistance. Decreasing the included angle of the windshield to less than 100 degrees will decrease fuselage drag—as per recent article in "Aviation".

7. Tail surfaces with more rounded tips and tapered and a possible increase in aspect ratio of stabilizer which would allow a decrease in area.

8. A considerable number of control horns and cables are exposed at present. Elimination of these projections would be an advantage.

9. A fuselage of elliptical cross section faired into the tail surfaces with large fillets would cut resistance considerably.

10. Fillets between wing and fuselage have of course been previously considered.

11. A method of changing the lift on a control surface (other than by breaking it on a hinge) such that it could be carefully faired into the fuselage is a possible future development. Slots, warping, external spoilers, and auxiliary airfoils are possibilities.

12. It is possible that a method of decreasing induced drag of a wing other than by increasing the aspect ratio can be developed, since it is a question of reducing the vortex at the tip.

Document 3-18(c), "The Reminiscences of Harold Hicks," Aug. 1951.

The 4-AT plane, although it was first powered with the J-4 engines, used the J-5 engines when Wright built them. Then George Mead, a man for whom I used to work down at McCode, and Fred Rentschler started the Pratt & Whitney Aircraft Company. They were previously with the Wright Company, and Wright didn't appreciate the fact, so they told me, that they saved Wright about a million dollars in one year. They didn't want to expand or something of that kind, so they pulled out.

They got Gordon Rentschler, Fred Rentschler's brother who was president of the National City Bank, to start them in business over in Hartford in the Pratt & Whitney Company's plant. They brought out the Wasp engine. Andy Willgoose really designed it. They came here about 1926, to get us to put the Wasp in our airplane.

We put the Wasp on the tri-motor plane, and of course that increased the performance. It upped the performance from about 114 miles an hour to about 128 miles an hour. That was before we had the N.A.C.A. ring cowls [SIC] on the job. That plane was known as the 5-AT.

Pressurized cooling was done on the big 10-AT, I think. It was a big ship we never flew, but this plane was equipped with liquid-cooled engines.

On the first Pratt & Whitney engines that we had, we designed cooling baffles that went directly across the cylinder head. In other words, we were testing sort of a cowl that was something like the final N.A.C.A. cowl before they started their cowling work. That was done largely to increase the speed of the ship. That it did, because the speed of the ship jumped up to 152 miles an hour.

Manning who had come with us at that time was our test pilot. He established the world's record for carrying a two-thousand kilogram load under a closed course. The wind shifted on the tri-angular course when he was flying the ship, enabling him to hang up a speed of about 160 miles an hour. The Ford tri-motored plane held that record for some time.

That special cowl was first used on the 5-AT. That was the one with the Pratt & Whitney engine.

We started putting the N.A.C.A. rings on all of our engines after the T.A.T. had bought a large number of the planes. The planes were brought in here for reconditioning. It was possibility in 1928 or 1929 that we started to put cowlings on all the engines. The top speed of the plane was increased from 128 to 152 or 154 miles per hour by these rings. No increase in fuel consumption ensued….

I have always felt that Tom Towle, more than anyone else, probably had more to do with the Ford tri-motor design initially than any one particular person, although it was a composite design by eight people. I myself worked up the landing gear in addition to supervising the design.

The Venturi type of exhaust was an idea that I had based upon air flow about a central exhaust tube. The idea was in connection with this low wing monoplane that we had worked upon, a fourteen place job that never materialized. It was because of Mayo's failure to give us the okay to proceed, he being interested in the 14-AT which was a large ship.

We were actually considering having externally braced wings, although it was a low wing job. We had two designs worked out. One was a fully cantilevered wing, and the other had two struts on each side as external bracing. We had figured that these tubes might easily be used as exhausts.

Al Esper, who was in charge of our wind tunnel activity in the old hangar building, had developed a hydraulic flow channel which was a return type using ink as the liquid in the channel and having aluminum powder which floated in a flecked condition on the surface.

By placing our models in this channel and the liquid being pumped around, we could see exactly then what the air flow would be like. There we worked out an exhaust system of fairing around a round tube that was truly remarkable. It was based entirely on Burnelli's theorem of flow where the energy is constant everywhere in the universe; that is, potential energy is converted into kinetic energy.

Nothing was ever done about that because that plane, of course, was never developed. It was just an interesting thing because it recalls the fact that we experimented in those early days with so many ingenious things that we could use and apply if we had to. We were always leading the aircraft industry at the time. The ideas would never get into practical operation because perhaps the immediate application for it never presented itself.

There was a freedom of thought and a freedom of opportunity to try things here in the early Ford Motor Company such as I have never seen before in any organization or since. It was truly a designer's paradise. Of course, there was always the treatment that you would get from Henry Ford that was to be feared perhaps. You knew that everything that you did was subject to immediate explanation and criticism.

For a person with any degree of courage, you never hesitated to go and try things out and do things because of the liberal attitude the Fords had on the working out of new ideas and new thoughts. They would take a personal interest in these thoughts.

You would always want to explain to them what you were doing and never attempt to conceal anything from them. You had to be very aboveboard with everything.

They would delve down into the detail. They would want to know everything about it, for example, the Venturi exhaust. As a matter of fact, you had to have everything on hand to explain to them; otherwise, they were so busy that they had little patience. If you didn't have things immediately available to show them, well, they had business elsewhere.

The wind tunnel was not particularly unique at the time the Ford Motor Company put its own tunnel in; there were wind tunnels then at that time. Because of the cramped space, we had to use a terrific amount of power and exhaust the air free into the atmosphere. That is, we couldn't build up the inertia of a continuously moving column of air.

We had a 250 horsepower electric motor up in the second story of the old hangar building that we connected with a large sirocco fan. We blew that air directly through the throat of our Venturi and in there used models about one-tenth size, as I remember. We used this for all of our aerodynamic work.

Al Esper was the technician who did the work at the wind tunnel. He was the one who was responsible for working out flow patterns, and he did an excellent job of it. The model making was done by James Lynch who is presently employed here in the Styling Department.

Document 3-19

"Lockheed 'Sirius' Monoplane," *Aero Digest* 16 (January 1930): 128.

The aviation trade journals played a vital role in announcing the introduction of aircraft that embodied new design principles—even those like the Lockheed Sirius that still retained elements of the old. In January 1930, *Aero Digest* magazine profiled the Sirius, a single-engine monoplane with fixed landing gear that was essentially a low-wing version of the Vega. But the plane incorporated Jack Northrop's innovative construction techniques, and even with its fixed gear and open cockpits, it reflected a streamline design. The Sirius was, in fact, quite similar to another sleek low-wing monoplane of the day, the Northrop Alpha, except that the Sirius was made entirely out of wood, whereas the Alpha was all metal. In 1929, Charles A. Lindbergh purchased a Sirius for his attempt to make a record nonstop flight from the West Coast to the East Coast. In 1931, he and his wife Anne Morrow flew the plane on their survey flight to Japan, via Alaska and Siberia.

Document 3-19, "Lockheed 'Sirius' Monoplane,"* Aero Digest *16 (January 1930).

LOCKHEED
"SIRIUS" MONOPLANE

The first of a new series of Lockheed planes, known as the Lockheed "Sirius," and named after one of the brightest stars in the firmament, is to go to Colonel Charles A. Lindbergh. Edward S. Evans, president of Detroit Aircraft Corporation, has announced that the Lockheed division will add the Lockheed Sirius type of plane to its regular line of models. Inasmuch as the various Lockheed models are named after stars, the name Sirius was selected as appropriate for the new type of plane.

First test flights of the Lockheed Sirius were made at the United Aircraft field near the Lockheed factory. Although no tests were run over a measured course, the Sirius is reported to have proved exceptionally fast, was highly maneuverable, showed good stability and landed at low speeds.

The Department of Commerce has paid tribute to Colonel Lindbergh's famed ship, the Spirit of St. Louis, in granting of the license of the new plane. It will be remembered that "WE" bore the identification NX-211 and the new ship will be known on Government records as NR-211. The change in the prefix on the number from X to R is due to new government regulations and will permit operation of the plane in countries outside of the United States. In its coloring Colonel Lindbergh's new ship is quite distinctive. The entire fuselage is black and has a wide gold stripe on the mid-section of each side and extending the entire length of the body. The wings, extending from the lower side of the fuselage, are painted orange-red, as are the control surfaces on the tail.

The streamline of the new ship has been especially well worked out. The engine is completely enclosed in a cowling of the type originated by the National Advisory Committee for Aeronautics. It is estimated that this cowling increases the speed of the plane fifteen miles per hour. Although the cowling fits closely over the engine and the nose of the plane, ample provision has been made for cooling. The Pratt & Whitney Wasp engine on the Lockheed Sirius develops 425 horsepower at 2,000 revolutions per minute.

The streamline effect is carried on throughout the fuselage from the engine back to the control surfaces. The ship embodies the Lockheed type of monocoque construction, resulting in freedom from external bracing to the wings, which would greatly hamper the streamline effect. Special low type wind-shields also have been fitted.

The two cockpits are located well in the rear. Both are equipped with complete controls, and the forward place has a number of special instruments. The two places are spacious and comfortable.

In the landing gear is found further evince of streamline effect. New Lockheed hydraulic shock-absorber struts are used in this construction. The wheels are almost entirely encased in a special streamline housing known as "Pants."

The wings of the plane are of low-wing cantilever construction. The mean chord is eighty-two inches and the wing area is 265 square feet.

Tanks for 320 gallons of gasoline are built in the forward part of the fuselage and an additional 120 gallons are carried in the wings. The oil tanks have a capacity of 28 gallons.

The Lockheed Sirius was primarily designed as a two-place sport plane of high performance. For average service it would carry two persons and their baggage with 150 gallons of gasoline over a range of approximately 1,100 miles. For Colonel Lindbergh's use, however, the extra gas tanks were installed, bringing the total capacity up to 440 gallons of fuel and increasing the range to 3,300 miles. This arrangement was specified by Colonel Lindbergh so that he may make extended flights without the long delays caused by landings for gasoline.

With the 3,300-mile range as desired by Colonel Lindbergh, there is a considerable increase in the gross weight, a condition which the Sirius was designed to meet. With tanks for 440 gallons of gasoline, the plane weighs 2,950 pounds. Gasoline weighs 2,640 pounds, and the twenty-eight gallons of oil, 210 pounds. Each person and equipment is allotted 200 pounds, and another 100 pounds is estimated for baggage. This results in a gross weight of 6,200 pounds, with the plane fully loaded for a flight of 3,300 miles.

Weights given below are for a standard production Sirius with a fuel capacity of 150 gallons and an oil capacity of twelve gallons.

SPECIFICATIONS

Span	42 feet 10 inches
Chord (mean)	82 inches
Wing area	265 square feet
Length overall	27 feet 6 inches
Normal fuel capacity	150 gallons
Normal oil capacity	12 gallons
Normal cruising radius	1,100 miles

Weights

Weight empty	2,950 pounds
Payload	500 pounds
Disposable load	1,450 pounds
Gross weight loaded	4,400 pounds

Document 3-20(a-c)

(a) "The New Martin Bomber: Model XB-907," Glenn L. Martin Company Engineering Report No. 240, 5 August 1932, Accession A1, Folder "B-10/char," United States Air Force Museum Research Division, Dayton, Ohio.

(b) Glenn L. Martin to Walter G. Kilner, undated (ca. 1933), Accession A1, Folder "B-10/(series)/his," United States Air Force Museum Research Division, Dayton, Ohio.

(c) Glenn L. Martin Company, "History and Development of the Martin Bomber," 20 January 1938, Accession A1, Folder "B-10/(series)/his," United States Air Force Museum Research Division, Dayton, Ohio.

This trio of documents concerns the development of the Martin B-10 bomber, one of the first aircraft in which all the various constituents of the airplane design revolution of the interwar period started to come together. A sleek, stress-skinned, all-metal monoplane, the B-10, which first flew in February 1932, delighted the U.S. Army Air Corps by reaching speeds that made all existing fighter planes obsolete. While the best fighters of the day could fly no faster than about 190 miles per hour, a B-10 could rush in to a target with a ton of bombs at 185 mph and speed away at 210. As historian Richard K. Smith has noted, as late as 1935, "there was not a fighter plane in the world that could catch a Martin B-10" ("Better: The Quest for Excellence," in *Milestones of Aviation* [New York: Hugh Lauter Levin, 1989], p. 255.). This came as a terrible shock to military establishments around the world as, up until this time, bombers could fly no faster than about 120 mph and cruise at only 95. None of these bombers possessed an all-metal structure; they were covered in fabric. None of them carried their bomb loads internally behind streamlined doors, as the B-10 did, and few of them had enclosed cockpits. And none of them had NACA cowlings and retractable landing gear like the B-10 did. All in all, Martin's integrated design of the B-10 led to a "combat airplane revolution" involving not just bomber design but also a quest for superior fighters.

Although one of the keys to the B-10's success was its two 775-hp Wright R-1820 engines, aerodynamic streamlining played a critical role. Besides utilizing an enclosed cockpit and retractable landing gear, the Martin Company also made good use of up-to-date NACA data on engine cowlings and the optimum placement of engine nacelles. The NACA's director of research, George Lewis, personally advised Glenn L. Martin sometime around 1930 to 1931 that Martin's new bomber design would not only fly significantly faster than its present 195 miles per hour, but would also land slower and more safely, if the engine's Townend ring were replaced by the NACA No. 10 cowl. Pratt & Whitney, the builder of the engine for the B-10, was contractually committed to using the ring. Martin eventually adopted the NACA cowling, increasing the airplane's maximum speed by 30 mph to 225 and also reducing its landing speed significantly. In 1932, the struggling company won the Collier Trophy for its B-10 design, and in the next two years, the army purchased more than 100 B-10s, rescuing Martin from the worst of the Depression. In fact, what the cowling did for the B-10's performance may well have been why Martin won the production contract from the army and why Boeing's B-9, which used the Townend ring, lost.

Perhaps the most illuminating document in this trio is the second one, a letter solicited by Major Walter G. Kilner of the Office of the Chief of the Air Corps around 1933, in which Glenn L. Martin explained the importance of his bomber design.

Document 3-20(a), "The New Martin Bomber: Model XB-907," Glenn L. Martin Company Engineering Report No. 240, 5 August 1932.

THE NEW MARTIN BOMBER

MODEL XB-907

CHARACTERISTICS AND FEATURES

The Glenn L. Martin Company
Baltimore, Maryland

August 5, 1932

GENERAL SPECIFICATIONS

AREAS

Wing, total, incl. ailerons	505 sq. ft.
Ailerons, rear of hinge	45.2 sq. ft.
Stabilizer	42.2 sq. ft.
Elevator (incl. 8.7 sq. ft. balance)	41.8 sq. ft.
Fin	20.1 sq. ft.
Rudder (incl. 5 sq. ft. balance)	25.9 sq. ft.

WEIGHTS

Gross Weight		10717 lbs.
Weight Empty		6967 lbs.
Structure and Fixed Equipment	4145	
Power Plant	2822	
Useful Lead—normal		3750 lbs.
Crew	600	
Fuel	1200	
Oil	105	

Equipment	102
Guns	276
Bombs (normal load)	1250
Bomb racks and controls	178
Pyrotechnics	39

PERFORMANCE—NORMAL FULL LOAD-1820-E-1 ENGINE

Maximum speed level—sea level—front cockpit covered	190 M.P.H.
Maximum speed level—8500 feet—front cockpit covered	201 M.P.H.
Maximum speed level—sea level—front cockpit open	186.4 M.P.H.
Maximum speed level—6000 feet—front cockpit open	197 M.P.H.
Maximum speed level—10,000 feet—front cockpit open	196.5 M.P.H.
Maximum speed level—15,000 feet—front cockpit open	190 M.P.H.
Take off speed—full load	76 M.P.H.
Take off speed—less bombs and ½ fuel	67.5 M.P.H.
Landing speed—full load	91.5 M.P.H.
Landing speed—less bombs and ½ fuel	83.0 M.P.H.
Rate of Climb—sea level to 6000 feet	1600 ft./min.
Time to 10,000 feet	7.8 min.
Climb in 10 minutes	12000 feet
Service Ceiling	20000 feet
Range—cruising speed (110 MPH) (200 gal.)	650 miles
Range—cruising speed (110 MPH) (300 gal.)	980 miles
Range—full speed (200 gal.)	370 miles
Range—full speed (300 gal.)	550 miles
Duration—cruising speed—200 gal.	5.9 hours
Duration—cruising speed—300 gal.	8.9 hours

GENERAL

Power Plant—2 Wright Cyclone R-1820-E-1 geared 1.58:1

Total power	1300 H.P. at 1950 R.P.M.
Airfoil	Tapered Gottingen 398
Aspect Ratio (S^2/A)	7.7

	Normal	Light	Overload	
Wing loading	21.2	17.36	22.4	lbs./sq.ft.
Power loading	8.24	6.74	8.70	lbs./H.P.
Useful Load	3750	1802	4403	lbs.

BOMB LOAD = 2500 LBS.

PERFORMANCE DATA

Engine—Cyclone R-1820-F 700 H.P. at 1900 R.P.M.

Wing Area increased to 600 square feet.

Gross Weight, lbs. (with 2500 lbs. of bombs)	12230
Wing area, sq. ft.	600
Span, ft.	70
Total Power H.P.	1400
Engine R.P.M.	1900
Propeller R.P.M.	1190
High speed, sea level M.P.H.	192
High speed, 8000 ft. (front cockpit enclosed by turret)	206
Stalling speed (less bombs and ½ fuel) M.P.H.	68
Rate of Climb (0 to 6000 ft.) ft./min.	1315
Time to 10,000 feet—min.	9.0
Time to 15,000 feet—min.	17.0
Service Ceiling—feet	21500
Wing leading—lbs/sq.ft.	20.3
Power loading—lbs./H.P.	8.7

BOMB LOAD = 1250 LBS.

COMPARATIVE PERFORMANCE

with

VARIOUS ENGINES AND INCREASED SPAN

Engines	Cyclone R-1820-E	9 cylinder R-1820-F	14 cylinder R-1535
Gross Weight	10717	10950	10950
Wing Area	505	600	600
Span—feet	62	70	70
Total power	1300	1400	1400

Engine R.P.M.	1950	1900	2500
Propeller R.P.M.	1235	1190	1670
High Speed—sea level	190	195	198
High Speed—8000 feet (front cockpit enclosed)	201	208	211
High Speed—8000 feet (front cockpit open)	197	—	—
Stalling Speed (less bombs and ½ fuel)	84	67.5	67.5
Rate of Climb— 0 to 6000 ft.—ft./min.	1600	1700	1750
Time to 10,000 feet—min.	7.8	7.0	6.7
Time to 15,000 feet—min.	15.5	13.5	13.0
Service Ceiling—feet	20000	23500	24000
Wing loading	21.2	18.25	18.25
Power loading	8.24	7.83	7.83

BASIC ARRANGEMENTS

The effect of summarizing all of the aerodynamic and structural improvements that have been developed during the last ten years may be somewhat surprising.

When Directive X-1670 for Bombardment Airplanes was issued to us for study a preliminary composition of the fundamental elements in modernized form was made, and the result indicated clearly that there would be no difficulty in exceeding the specified requirements.

The paramount importance of speed pointed out in the referenced Directive is readily appreciated, and was made a factor of first importance in the design program.

It is significant of new thought in bombing that the new Martin Bomber surpasses present standards of speed for pursuit, carrying three times pursuit useful load, but with only twice the power.

The limit of speed for bombing has by no means been reached, as will be apparent by comparison of performance data given above. The reduction of frontal area possible with 14 cylinder radial engines yields an increase of 12 miles per hour with only a minor increase of power and fuel consumption.

Still further increases of speed become immediately available with the mature

development of more powerful motors now in their experimental stages.

This unusual capacity for speed is the result of a complete analysis of the components of drag involved in bombing airplanes, and the results may be appreciated, without going into the mass of technical detail, by a simple comparison between the average drag characteristics of the series of bombers current during the past decade, and the drag characteristics of the new Martin Bomber. (Drag is expressed in the usual units of "flat plate" area.)

	Average of 6 Bombing Types	New Martin Bomber	Percent Improvement
Drag of wings	11.2	4.4	60.7%
Drag of Structure	35.6	13.6	61.8%
Total Parasite Drag	46.8	18.0	61.5%

The criterion of speed and cruising efficiency is the ratio of lift to drag, or briefly, the L/D. The new Martin Bomber has a maximum L/D = 11.4.

If the air resistance of the engines is deducted the L/D = 13.6, which compares with some of the best motor-less soaring planes that have achieved such excellent results in Germany. This fact is more remarkable in view of the relative strength factors; soaring wings being commonly designed to support about 20 lbs./sq.ft. of surface, while the Bomber wings have been static tested to 143 lbs./sq.ft.

As a result of its unusual aerodynamic finesses, the XB-907 is capable of extremely high speeds with the present available power. Power charts based on the present performance, with 650 H.P. engines show that the speed of 220 M.P.H. at 8000 feet can be readily obtained with 800 H.P. engines, and when 1000 H.P. engines become available a speed of 230 M.P.H. or more will be attainable.

A feature of XB-907 is its ready adaptability to a wide variety of engines. In its present condition it can mount either the Hornet 1860, the Cyclone 1820-E, the Cyclone 1820-F, or the twin Wasp 1535. For experimental testing the Cyclone 1820-E was installed, pending complete development of the 1820-F. The 1820-F engines will be installed ready for flight in the early Fall. Provision for other engines is made in the structure.

With the vastly increased speed there is a necessity to provide strength in proportion. The current specification of design load factors for bombing airplanes is 4.5 (Ref. page 171—Handbook of Instructions for Airplane Designers). With high speed and maneuverability in mind the XB-907 was built to factors of 8.5, (the same as observation planes) or nearly twice the strength of any other existing American Bombers.

Diving speeds of the modern high speed bomber are in excess of 250 M.P.H. and for pulling out from this speed the factor of 4.5 is inadequate and dangerous.

A twenty percent increase in the wing area of the new Martin Bomber is now being made to accommodate the 2500 pound load of bombs. This increased load can be carried with a load factor of more than 6.0 without change of the existing structure, due to the high strength provided in the basic structure in the original design.

The added wing area not only accommodates greater load but also provides a lower landing speed and greater climbing ability.

The climb of the XB-907 is excellent, and during flight trials at Wright Field it was officially found to be 1600 feet per minute with the original 1820-E engines. A substantial increase in climb is expected with the 1820-F engines.

Due to the elimination of external bracing and efficient planform of wing the additional area can be added with no material reduction of top speed. The drag of the added area at 200 M.P.H. is 96 pounds, and is equivalent to 51 H.P. The permanent engine installation (1820-F or R-1535) will provide a total of 100 H.P. more than the temporary 1820-E engines, which will amply provide for the slight increase in wing drag.

In addition to adaptability for various power plants and a complete range of load carrying capacity the new Martin Bomber has several predominating features which influence modern bombing.

Its relatively small size and high degree of maneuverability permit close squadron formation and concentration of protective fire. The size of squadron depends upon the mobility of the units. The lateral control of the new Bomber is very responsive to the touch and the ship can be wheeled through unusually small turning circles with a high degree of accuracy. All Air Corps pilots who have maneuvered the plane have remarked upon the accuracy of its turns, attesting the pronounced influences of scientific aileron proportions and proper balancing devices. In addition to aerodynamic balance of ailerons, all lateral moments are reduced to zero by special trailing edge vanes.

Particular attention is being paid to mass balance and a series of flight tests has been completed for the purpose of calibrating the dynamic action of the entire mass of the airplane in flight, permitting exact balance of lift and weight to reduce all pitching moments to zero.

Another basic feature commented upon by pilots is the steadiness of the plane upon its course in flight. This feature has been accomplished by extreme care in proportioning the dimensions of the fuselage and tail surfaces, and the components of directional stability have been made very positive for maintaining the desired course for sighting and releasing bombs with a minimum of effort.

The possibility which the new Martin Bomber offers for higher operating altitude will profoundly effect bombing practice. In conjunction with the C-4 sight, just recently perfected, bombing from 20,000 feet with heavy bombs is possible. It will be noticed from performance data (page 3) that the XB-907 is capable of a service ceiling of 23,000 feet with present engines and moderate supercharging, and there appears to be excellent prospect of realizing 25000 feet, which will put heavy

bombing well beyond the reach of anti-aircraft fire and add materially to chances of arriving over large objectives undetected.

After discharging the bombs the XB-907 has all the speed and maneuverability of the finest two seat fighters. These qualities can be used to the utmost because of the high strength factors built into the airplane, adding qualities to its power of defense and ability to escape which are not present in the large, heavy bombing types.

The internal bomb racks are an important feature of the XB-907 for three reasons. The internal racks permit carrying the widest assortment of bombs from the smallest to the 1100 pound size, all inside the body. It eliminates the reduction of speed which is involved in any attempt to carry the same variety of bombs outside. It preserves the more delicate parts of the timing mechanism from mud and ice and particularly ice which forms in the cold higher altitudes. Other advantages of the internal bomb rack, such as facility for clearing jams in the releasing mechanism, or making readjustments during flight, are readily appreciated.

Possibilities for extending cruising radius of action are important. This basic feature has been incorporated in the new Martin Bomber by a unique arrangement made possible by the internal arrangement of bomb racks. The plane as now built has racks for 1250 pounds of bombs inside, and in addition racks for two heavy bombs (600 or 1100 lbs.) under the wings. By utilizing the upper portion of the internal rack to support a 250 gallon gas tank the fuel capacity can be increased to 550 gallons with 1200 to 1500 lbs. of bombs, and a cruising range of 1500 miles.

From the pilot's cockpit vision is afforded in all desired directions. It is impossible to approach the plane without being visible to the pilot. For formation flying his vision is entirely unobstructed.

Details of the bomber's cockpit have been worked out with extreme care. The bomber's operating position is very comfortable and he has easy access to all bombing controls. Each window in the front and sides of the bomber's compartment has been carefully determined to meet requirements, both before and during the sighting operation.

At speeds of 200 M.P.H. and over it is impossible to operate a gun from the conventional Scarf mount. Anticipating this difficulty the original nose design of the new Martin Bomber was so shaped and cowled that the front gunner received protection from the blast. However, the development of the larger type bomb sights (L and C type) required a greater depth of cockpit and the desired lines of the nose could be only partially maintained. The shelter afforded the front gunner was insufficient and research was begun upon the completely enclosed type of turret. This type has now been developed and will shortly be mounted on the XB-907 nose.

The turret consists of a "bee-hive" shaped enclosure of transparent, fire-resisting material upon an aluminum alloy skeleton frame. It is mounted on ball bearings and is easily rotated by hand, or locked in any position as desired. The gun carriage moves vertically in an open slot four inches wide, upon an accurately machined track built in the surface of the turret enclosure. The gun may be trained on any point in the entire forward hemisphere, and also about two thirds of the rear hemisphere. Practically the only limitation of fire is the physical parts of the airplane and propellers. The predominating merit of this turret is that it makes the gun actually usable in a convenient and practical manner when flying at high speed. It also provides complete protection for the bomber when he is operating his sight. The interior arrangement of the turret has been studied in a full size mockup, and so laid out that the bomber may transfer immediately from bombsight to gun, or the opposite without a second's loss of time. When seated at the bombsight he has only to raise his eyes to obtain full vision forward, up, to the side or to the rear.

The self-landing characteristics of the XB-907 have been generally remarked. From the pilot's viewpoint the approach is steady and the landing very easy and regular. All pilots who have landed the ship are unanimous in their description of the semi-automatic landing qualities of the plane. From the engineer's viewpoint these qualities have been obtained by careful analysis of the airplane's stability curves and arrangement of the wing planform to prevent premature stalling of either wing tip. Lateral control is present in positive quantity throughout the landing.

INTERIOR ARRANGEMENT

The design and arrangement of the body is an example of combined aerodynamic and structural efficiency.

The body structure is entirely new and it surpasses previous standards of weight/strength ratio by a considerable margin. During static tests it supported the greatest bomb load and loads equal to the breaking strength of the tail surfaces and the tail wheel, with scarcely measurable deflection. Its efficiency factor of weight to size is 1.11 compared to 1.33 for the best previous types. This remarkable gain is due to a novel type of construction known as the "restrained shell" monocoque. This type of body has no separate longitudinals. The full strength of the skin is derived by the use of a series of closely spaced tubular rings, and the backbone strength is contained in corrugated arches which run along the top and bottom of the body.

The XB-907 body lines were laid out after the manner of flying boat hull lines. Fluid air pressures at 220 M.P.H. approach the fluid pressures of water, corresponding in dynamic pressures, to water at a speed of about eight miles per hour. Advantage of this fact was taken in perfecting the airplane's stability characteristics and maneuvering qualities, and the body lines were proportioned accordingly.

In addition to the advantage of variety and protection of bombs on the internal racks, the body shape is ideal for comfortable and commodious radio and navigation installations. The radio compartment is a small enclosed room with ample space for all equipment to be arranged conveniently.

Excellent protection is provided for the upper rear gunner by the natural shape of the body and he is entirely out of the blast when training his gun through the entire upper hemisphere.

The lower hemisphere is covered by a really efficient floor gun mounting, and the upper slope of the underbody makes the floor gun particularly effective.

The shape of the body nose is ideal as a base for a gun mount, and it accommodates the turret enclosure without increase of drag, a feature which is impossible without suitable body lines.

Passage from both ends of the body to the bomb bay is permitted by the deep streamline shape of the body, and as a result access may readily be had in flight to all bomb gear, fuel valves, pumps, strainers, etc. The distribution of stresses in the body was examined in flight by structural engineers, and in carrying out this check-up various persons passed without difficulty from the rear cockpit, back to the stabilizer inside the body on several occasions.

In spite of its capacity for 2500 lbs. of bombs the XB-907 is small in overall dimensions. Although it is a monoplane the span, with increased wing area, is only 70 feet, five feet smaller than present service biplane bombers.

During flight tests the airplane was flown on one engine with full load. The particularly interesting part of this test was the powerful action of the rudder trailing edge vane. This vane is controlled from the cockpit at will. By its use the offset thrust of one-engine flight is completely balanced. In fact, there is no difficulty in making turns contrary to the live engine and this was done repeatedly during tests. This vane has a very fine adjustment in the cockpit and it can be used to compensate for drift in a cross-wind to a very exact degree.

The landing wheel retracting gear of the new Martin Bomber has been entirely successful from the beginning. A ratchet arrangement is provided on the operating handle to increase the pilot's mechanical advantage. The gear operates so easily that the ratchet is seldom used. The punishing tests given the landing gear during trials showed no effect whatever upon the landing gear. The operating arrangement in the cockpit has no complications of any kind as all the parts operate automatically and the pilot has only to set the operating handle for "up" or for "down," as desired. No wrong motions of the operator can jam or damage the gear in any way, and it has proven to be a highly successful device.

The wing construction is extremely strong. It consists of a backbone of corrugated dural of an average thickness of 1/16 inch, with substantial wing beams in the usual location. The form of the wing is maintained by substantial formers spaced rather widely. The entire wing assembly is simple and of few parts. The conventional multiplicity of small pieces is avoided. The bottom of the wing is screwed on and can be removed as desired, exposing the entire interior for inspection or repair.

The elimination of fuel tanks from the body avoids the unpleasant fumes of gasoline in the body and materially reduces the possibility of fire.

UTILITY OF THE BOMBER

The Martin ship is unique in its variety of possible uses. It was originally designed to be a fast day bomber. It not only exceeds the requirements for fast day bombing, but it is also able to meet all the requirements for heavy bombardment except the 60 M.P.H. landing speed, and in this characteristic the landing qualities of the ship are so far in advance of normal that the pilots' impression of the landing is very satisfactory. It will carry the normal heavy bombardment load of bombs (2000 lbs.) and with the 20% addition to wing area now being made it will carry the 2500 lb. bomb load and take off at less than 70 M.P.H. without difficulty. The Martin ship exceeds the high speed required for heavy bombardment by some 50 miles per hour, and the ceiling requirements by several thousand feet.

Furthermore, without bombs it has a maneuverability comparable with the two seat fighters.

The XB-907 has performance as an observation type far superior to requirements or existing types. Its large and roomy body affords excellent facilities for observation equipment and the complete vision which all members of the crew have is a decided asset.

Although no attempt was made in the design to develop characteristics suitable for a ground attack plane, it happens that the Martin ship fulfills the ideal attack requirements to a high degree. Its speed and maneuverability would be invaluable for this class of airplane. Its present guns can be trained to front or rear towards the ground, particularly the floor gun, and the mounting of additional guns can be readily accomplished.

The XB-907 with its internal bomb rack can carry all the smaller sizes of bombs. The use of this airplane as a raiding type has been suggested, carrying ten to twenty of the smaller bombs, up to the 100 pound size, making it a formidable weapon against troop trains, small bridges and similar objectives. Its speed and mobility will make it highly valuable for this work.

The excellent flying qualities, high performance and simplified construction of this airplane have resulted in a fundamentally new type of airplane which greatly surpasses existing airplanes and specification requirements in several classes. Its adaptability to existing and future engines speaks well for its further development. It brings into actuality performance and abilities for military use which have never existed before.

Photograph No. 7489 herewith shows the excellent accommodations afforded for radio with proper and adequate spacing of each item of equipment. A full size folding navigator's table is provided and the radio room is lighted for night navigation operations.

Pursuit escort will not be necessary for the XB-907 because the forward gun is made completely effective by the turret enclosure described on page 7. In combination with the upper rear gun and the lower rear gun this provides complete protection, even if the bomber becomes separated from its squadron. This eliminates the necessity of limiting the radius of action of long distance bombers because of the short range of the pursuit escort.

Document 3-20(b), Glenn L. Martin to Walter G. Kilner, undated (ca. 1933).

Maj. Walter G. Kilner,
Office of Chief of Air Corps,
Washington, D.C.

Dear Maj. Kilner:

Complying with your telephone request we are pleased to summarize the outstanding features of the New Martin Bomber recently submitted to and successfully tested by the Air Corps, and to review briefly this accomplishment as an event of historical significance.

On August 22, 1930 this design amongst others was presented to the Material Division at Wright Field and we were advised that if we felt confident of results the Engineering Division would give endorsement to the development of the airplane.

When the fully developed airplane was delivered to Dayton on October 7, 1932 its performance so far exceeded expectations that current directives, of not only bombing airplanes, but all other combat types as well, were forthwith discarded as obsolete.

For the first time in the history of military aeronautics the bombing airplane became independent of pursuit protection or attack.

The new Martin Bomber is the first twin engined airplane and the first bomber of the world to surpass the 200 M.P.H. speed mark. It is the fastest airplane of its size in existence.

This airplane places at the disposal of the United States Government a weapon of military strength unequalled anywhere else.

The brilliant demonstration of the new bomber in its acceptance trials opened a whole new picture of the tremendous potentialities of aircraft in military operations, and thereby brought aeronautics a long step nearer to its ultimate place in world affairs.

Not only in military operations, but in commercial transportation as well, the superlative benefits of this design are appreciated and in the space of a few months, already imitated.

The new Martin Bomber has broken away from conventional practice in almost every particular whether it be structures, aerodynamics, control, propulsive efficiency execution of detail.

Its wing construction for the first time rationalizes variations of applied air pressures with supporting strength, and for the first time a wing construction of this type has been proven in flight.

The construction of the body is one of those inventions that inspires the old comment—"Why wasn't it thought of before?"

The retractable lauding gear of the new Bomber functions in its perfected simplicity with the monotonous regularity of any common event.

The shake-proof cantilever tail surfaces are more readily recognized as an innovation. There is also something new in a twin engined airplane which flies "hands and feet off" with only one engine, as the new Bomber does for extended periods. This all important increase in safety is a vital contribution to military and commercial aeronautics alike.

The propulsive efficiency of the nacelle-propeller-wing combination may be appreciated in full contrast from the simple fact of propeller efficiency obtained at 88%, a degree of efficiency hitherto reserved for Schneider Cup Racers.

Air Corps officers have stated that the New Martin Bomber is the best designed and built airplane which ever came to Wright Field.

This Company has been and is continually engaged in the research for better aeronautics. During the years since the Old Martin Bomber we have steadily accumulated improved aerodynamic and structural elements.

The creation of such new elements opens the way to progress, but it is also conceivable and frequently demonstrated that a few such elements do not insure successful aircraft, nor does the unskilled aggregation of miscellaneous innovations produce lasting benefits.

The New Martin Bomber is proportionally advanced in all its elements. Its parts have such mutuality of improvement that the complete composition is more than a list of detail inventions. It has the power and perfection that comes from the blending of related parts, each perfect in itself—in a word, unity.

Document 3-20(c), Glenn L. Martin Company, "History and Development of the Martin Bomber," 20 January 1938.

HISTORY AND DEVELOPMENT OF THE MARTIN BOMBER
January 20, 1938
The Glenn L. Martin Company
Baltimore, Maryland
U.S.A.

Up to the year 1932 the Army's bombardment equipment consisted of large twin engined biplanes having a high speed of 114 m.p.h.

In 1932 the Martin Company re-entered the bomber field and submitted an airplane to the Army which so far surpassed the Army's equipment at that time, that the Martin Company was given the 1932 award of the Collier Trophy in recognition of this outstanding contribution to aeronautical science. This airplane (See Figures 1 and 2), which was originally known as the Model XB-907, incorporated new military design features which up to that time had never been used. Many basic aerodynamic and structural features were also incorporated in this model which made it the forerunner of the modern high speed commercial transports.

This model so far surpassed any previous airplanes of its type that it was found necessary by military authorities to change entirely military tactics as far as aerial warfare was concerned. Fundamentally, it was so far in advance of its day that since then major basic changes have not been required, however, the Martin Company has continuously incorporated improvements in structure, aerodynamics, power plant and equipment. Because of this policy, the airplane today is still outstanding in performance, striking power, and general utility. Added to this, the incorporation of modifications as a result of operating experience has made it one of the simplest airplanes to maintain in service. The general improvement may be noted by comparing Figure 1 to Figure 10.

The following outstanding flights have been made with this airplane besides the constant, day in and day out, routine flying.

(a) On August 24, 1935, Major General Frank M. Andrews, commanding officer of the G.H.Q. Air Force, accompanied by two assistants, flew his Hornet powered Martin B-12 bomber with Edo floats on a 2000 kilometer non-stop flight at 266 km. per hour. The course was laid from Hampton Roads, Va. to New York and return by way of Washington, D.C. Flying at an altitude of 3040 meters or more, General Andrews completed the second lap of 1000 kilometers in three hours 45 minutes and 13 seconds, and carried a payload of 1000 kg. made up of two big aerial bombs. At the speed of 266 km./hr. General Andrews broke three world seaplane speed records, for 1000 kilometers, one without payload, the second with payload of 500 kilograms and the third with payload of 1000 kilograms. (See Figure 4).

(b) A squadron of ten airplanes flew from Washington, D.C. to Alaska and returned. This flight covered a total of 12,000 km. While making this mass maneuver, the only incident of the flight was a forced landing caused by a piloting error. It was necessary to land in the water due to the excessively rough terrain in that section of the country. A water landing was made without any difficulty. The airplane was beached, repaired, and in two days continued the flight with the rest of the squadron. This particular incident indicates the ability of this airplane to meet all conditions. A notable record established as part of this flight was the photographing of 20,000 square miles of unexplored territory in three days.

(c) One of the later models was flown from the manufacturer's plant in Baltimore, Md., down through Central America, the west coast of South America, and across the Andes to Buenos Aires, a total distance of 13,000 km. without difficulty. The airplane on this flight carried a crew of four men, their personal equipment, and a full load of miscellaneous equipment.

(d) Recently a squadron of nine ships of this model was flown a total distance of over 13,000 km. on a tour over the Dutch East Indies. This flight, which included several long over water flights, was completed without any maintenance of any sort except refueling and the addition of engine oil.

The following is a chronological list of the outstanding features incorporated in the original Model XB-907 and added in succeeding models. The tabulated list enclosed gives more detailed information.

(1) Due to the high speed of this model it was found impossible to operate a flexible gun without wind protection for the gun and gunner. This was the first airplane in the United States to incorporate a gun turret which has since been incorporated on most military aircraft of its type. (See Figure 3).

(2) In order to keep the airplane aerodynamically clean and to protect the vital bomb rack mechanism from dirt, this airplane was the first one to incorporate bomb bay doors which completely enclose the bomb bay.

(3) In order to further increase the aerodynamic efficiency and to give added protection and comfort to the crew, enclosures were incorporated for each cockpit opening. These enclosures were designed to give easy access, to the cockpit and from the cockpit, and also give the rear gunner ample wind protection when operating the gun. (See Figure 6). The enclosure on the most recent model has been redesigned as one continuous structure covering both front and rear cockpits, with special arrangement so that the radio compass loop may be installed within it.

(4) The wings and horizontal tail surfaces were designed with water tight compartments to enable the airplane to remain afloat in case of an emergency water landing. This type of flotation requires no servicing and does not deteriorate or depend upon mechanisms for its operation, the floats being ready at all times without any action on the part of the pilot.

(5) Full cantilever wings and tail surfaces were incorporated which, while being aerodynamically efficient, contribute to simple maintenance due to the elimination of constant rigging and alignment which was necessary with previous airplanes. The wing construction is of a type developed through the cooperation of the Martin Company and Army engineers, which structurally has proven very sound from a strength, weight, stiffness, and manufacturing point of view. This type of wing construction has since been incorporated on Martin airplanes up to 62,000 lbs. gross weight with the same efficiency, thus demonstrating its basic soundness.

(6) The landing gear, which at the time of its inception, was an outstanding development for airplanes of its size, is cantilevered for side loads which tends to simplicity and cleanliness. The original main strut of the landing gear was of a twin oleo type later changed to a half fork to facilitate the changing of tires. It is of interest to note that the landing gear on this airplane, including the retracting mechanism, was of equal weight to the landing gear on previous bombers of the same gross weight which were not retractable nor cantilever.

(7) The development of the present power plant installation from the original has been the result of continuous testing and research. One of the most striking examples of this development is the engine cowling as may be

seen by comparing Figures 1 and 10. The latest design has been commented on by the engine manufacturer as being the most efficient type for properly cooling and regulating the temperatures of the most modern type of high horsepower engine. This cowling has also permitted the realization of a substantial increase in high speed.

(8) On the B-10B and succeeding models, wing flaps were incorporated to

allow for increased gross weight without increasing the landing speed of the airplane. (See Figure 5). The prototype model was originally designed for 10,580 lbs. gross weight with an area of 550 sq. ft. which was changed to 682 sq. ft. on the succeeding airplanes. Since then the airplane's normal gross weight has been increased to 15,400 lbs. gross weight without affecting its flying characteristics. The increase in gross weight has been due to increase in fuel capacity and additional equipment which has recently been developed. To provide for this increase in gross weight the Martin Company has successively reanalyzed the structure of the airplane and incorporated changes in it to maintain full strength requirements. The most recent structural analysis has included the latest specification of the U.S. Army Air Corps on gust conditions combined with a diving speed of 300 m.p.h.

(9) On Model 139-WH2 and succeeding models, the rear section of the wings, that is from the rear spar to the trailing edge, which was fabric covered on preceding models is metal covered. This was done in the interest of maintenance as it eliminated the necessity of putting the complete airplane in the shop for replacing the fabric. The only fabric on the airplane at the present time is on the movable control surfaces which may be easily replaced with the spare surfaces when in need of recovering. This allows the airplane to be kept in service at all times.

(10) Many items of equipment have been incorporated in the later models, namely:

Gyropilots.
Constant speed propellers.
Deicers.
Abrasion shoes for horizontal stabilizer.
Supercharger regulators.
Latest types of radio and radio compasses.
Aerial camera.
Oxygen equipment.
Engine fire extinguisher.

Document 3-21

Jack Frye to Douglas Aircraft Corporation, "Attention: Mr. Donald Douglas," 2 August 1932. A facsimile of this letter appeared in *American Heritage of Invention and Technology*, Fall 1988.

One important link that cannot be overlooked within the context of the airplane design revolution of the interwar period—or airplane design at any time, for that matter—is the link between technical innovation and economic incentive. In the austere financial conditions brought on by the Great Depression, competition between the airlines, and between the aircraft manufacturers, grew extremely keen. This tight situation can be seen clearly in the document below; in it, Jack Frye, vice-president for operations for Transcontinental and Western Airlines (TWA), acknowledged in the summer of 1932 that, for his airline to survive, it needed to adopt a popular new transport airplane. TWA's aging fleet of Fokker trimotors were becoming more and more suspect in the eyes of the public and government regulators ever since the fatal 1931 crash of a TWA airliner carrying famous Notre Dame football coach Knute Rockne. Frye desperately wanted a new design like the Boeing 247, but the corporate connection between Boeing and United Air Lines blocked TWA from buying any of the new aircraft.

Document 3-21, Jack Frye to Douglas Aircraft Corporation, "Attention: Mr. Donald Douglas," 2 August 1932. A facsimile of this letter appeared in* American Heritage of Invention and Technology*, Fall 1988.

TRANSCONTINENTAL & WESTERN AIR INC.
10 RICHARDS ROAD
MUNICIPAL AIRPORT
KANSAS CITY, MISSOURI

August 2nd, 1932

Douglas Aircraft Corporation,
Clover Field
Santa Monica, California.

Attention: Mr. Donald Douglas

Dear Mr. Douglas:

Transcontinental & Western Air is interested in purchasing ten or more trimotored planes. I am attaching our general performance specifications, covering this equipment and would appreciate your advising whether your Company is interested in this manufacturing job.

If so, approximately how long would it take to turn out the first plane for service tests?

Very truly yours,

Jack Frye
Vice President
In Charge of Operations

JF/GS
Encl.

N.B. Please consider this information confidential and return specifications if you are not interested.

TRANSCONTINENTAL & WESTERN AIR, INC.

General Performance Specifications Transport Plane

1. Type: All metal trimotored monoplane preferred but combination structure or biplane would be considered. Main internal structure must be metal.

2. Power: Three engines of 500 to 550 h.p. (Wasps with 10-1 supercharger; 6-1 compression O.K.).

3. Weight: Gross (maximum) 14,200 lbs.

4. Weight allowance for radio and wing mail bins 350 lbs.

5. Weight allowance must also be made for complete instruments, night flying equipment, fuel capacity for cruising range of 1060 miles at 150 m.p.h., crew of two, at least 12 passengers with comfortable seats and ample room, and the usual miscellaneous equipment carried on a passenger plane of this type. Payload should be at least 2,300 lbs. with full equipment and fuel for maximum range.

6. Performance

Top speed sea level (minimum)	185 m.p.h.
Cruising speed sea level—79% top speed	146 m.p.h. plus
Landing speed not more than	65 m.p.h.
Rate of climb sea level (minimum)	1200 ft. p.m.
Service ceiling (minimum)	21000 ft.
Service ceiling any two engines	10000 ft.

This plane, fully loaded, must make satisfactory take-offs under good control at any TWA airport on any combination of two engines.

Kansas City, Missouri.
August 2nd, 1932

Document 3-22(a-b)

(a) Clark B. Millikan and Arthur L. Klein, "Report on Wind Tunnel Test on a Model of the Douglas Transport DC-1 with Various Modifications," GALCIT Report 119, 7 June 1933, GALCIT Wind Tunnel Test Reports, Boeing Company Historical Archives, Long Beach, California.

(b) Engineering Department, Douglas Aircraft Company, "Development of the Douglas Transport," Technical Data Report SW-157A, undated (ca. 1933-34), Folder AD-761184-05, Aircraft Technical Files, National Air and Space Museum, Washington, D.C.

This duo of documents reveals the unprecedented extent to which the results of fundamental wind tunnel data went into the design of the DC-1, information that soon after would be expanded into its later incarnations, the DC-2 and DC-3. Bailey Oswald, the Douglas Company's first aerodynamicist, arranged for wind tunnel tests of the DC series at the Guggenheim Aeronautical Laboratory at the California Institute of Technology, his alma mater. GALCIT's new 200-mph tunnel enabled a more direct approach to design, and Douglas's direct use of the tunnel helped bring the discipline of aerodynamics out into the open for manufacturers to exploit by showing how successful such an experimental approach could be. Douglas's wind tunnel tests of the DC-1 model resulted in the refinement of several important design elements. The inclusion of an aft sloping windscreen, an NACA cowl, retractable landing gear, cantilever monoplane wing, tail surfaces, and streamline fuselage all proved to be aerodynamically sound. In key respects, Oswald's test program set an important precedent; it was soon almost commonplace to be using wind tunnel testing in aircraft design. No longer would there be such dependence on intuition or on theoretical calculations alone. Systematic physical tests would be required for nearly all proposed designs. This proved to be one of the most critical, and lasting, contributions of the process of reinventing the airplane.

Document 3-22(a), Clark B. Millikan and Arthur L. Klein, "Report on Wind Tunnel Test on a Model of the Douglas Transport DC-1 with Various Modifications,"

GUGGENHEIM AERONAUTICS LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA
REPORT ON

WIND TUNNEL TESTS ON
A MODEL OF THE DOUGLAS TRANSPORT DC-1 WITH
VARIOUS MODIFICATIONS

PREPARED BY:

C. B. Millikan

A. L. Klein

Date: June 7, 1933

I. Introduction, General Description of Model and Tests.

This report describes the results of wind tunnel tests on a 1/11th scale model of the twin-engined Douglas Transport airplane DC-1, and also the results of several auxiliary wind tunnel tests on models of component parts of the airplane. The experiments were made in the closed working section of the 10 degree wind tunnel of the GALCIT (Guggenheim Aeronautics Laboratory at the California Institute of Technology)*. Practically all of the tests were made at a wind speed of about 185 m.p.h., corresponding to a Reynolds' Number based on mean wing chord of approximately 1,700,000. A few measurements were made at lower speeds to investigate scale effect, and certain pressure distributions were also run at lower speeds. The critical Reynolds' Number at which a 27 cm. sphere has a drag coefficient of 0.3 is about 330,000 indicating a wind stream with very low turbulence. The entire model was lacquered and rubbed down to a high polish.

During the course of the investigation many modifications were made. Table 1 gives the notation employed throughout the report in designating the various elements and configurations.

*cf. Clark B. Millikan and A. L. Klein: "Description and Calibration of 10-foot Wind Tunnel at California Institute of Technology". Trans. of A.S.M.E., Aeronautical Engineering, 1932-33.

TABLE 1

Notation Used to Describe Configurations Tested

Symbols		Configurations
W0	=	Original wing
Wt	=	Original wing with large tips
W1	=	Intermediate wing
W2	=	Final wing with straight trailing edge
F	=	Fuselage
N, N1,…N8, etc.	=	Various nacelles with wing-nacelle fillets
C	=	Chassis retracted
Cd	=	Chassis down in position for landing
T	=	Tail wheel
H1	=	Original horizontal tail surface (small area)
H2	=	Final horizontal tail surfaces (large area)
V1	=	Original vertical tail surfaces
V2	=	Final vertical tail surfaces
w1	=	Original auxiliary airfoil (large chord)
w2	=	Final auxiliary airfoil (small chord)
A	=	Free air ailerons
S1	=	FV1N8CTH1 = Original Standard configuration less wing
S2	=	FV2N8CTH2 = Final standard configuration less wing

S1, S2 are always associated with one of the wings, e.g. S1Wt, S2W2, etc. and unless otherwise specified correspond to stabilizer, elevator, and flettner all clamped at 0° setting.

S2W2 = S2W2w2 = S2W2 with auxiliary airfoil w2 set at –5°.

TABLE 2

Data on Various Surfaces

	W(sub 0)	W(sub t)	W(sub 1)	W(sub 2)	H(sub 1)	H(sub 2)	V(sub 1)	V(sub 2)	w(sub 1)	w(sub 2)
Area, S(ft.^2)	910	944	910	939	120	145.6	53.2	71.4	17.01	11.01
Span, b (ft.)	79	85	85	85	23.9	25.83	10.3	11.17	4.131	4.131
Aspect Ratio, AR	6.85	7.66	7.94	7.7	4.75	4.58	1.28	1.75	-	-
M.A.C., t(sub ft.)	12.22	12.06	11.71	11.86	-	-	-	-	-	-
Root Chord (ft.)	14.86	14.26	14.26	14.26	-	-	-	-	-	-
Rudder Elevator Area (ft.^2) (Total)	-	-	-	-	46	48	31.7	42.5	-	-
Flettner Area (ft.^2) (Total)	-	-	-	-	7.5	7.72	4.1	6.58	-	-

All dimensions correspond to full scale. The model was 1/11th full scale.

The wings were all similar in tapering from a 2215 root section to a 2209 tip section. They differed chiefly in area and plan form. The basic data for the four wings, the two horizontal surfaces, the two auxiliary wings, and the vertical surfaces, are given in Table 2 below. Dimensions are for the full scale airplane. Wing areas are gross areas with no allowance for the portion of the wing covered by the fuselage, while tail surface areas do not include any area in the fuselage. Mean aerodynamics chords were calculated by the Douglas Company. *SEE TABLE 2

A three view of the airplane in its original forms (S1W0, S1Wt, S1W1) is given in Fig. 1, and one in its final form (S2W2) is given in Fig. 2. Photographs of the models and of various details are given in Photos 1-8. The design data furnished by the Douglas Company and used in the performance and stability estimates are given in Table 3.

TABLE 3

W = Gross Weight = 17,000 lbs.

Po = Design Maximum Brake Horsepower = 2 X 712 = 1424 H.P.

No = Propeller R.P.M. corresponding to Po = 1340 R.P.M.

Assumed C.G. positions to which moments are referred are given in Figs. 1,2. If a is the distance of the C.G. behind the leading edge of the M.A.C.

a/t = 24.135 % for wing Wt

a/t = 20.0 % for wing W2

Note: The C.G. position was assumed to be the same for wings Wo, W1, and Wt.

1 X Tail Length = distance from elevator hinge to C.G. = 40.62° for S1

36.62° for S2

For details of the model and its various parts reference must be had to the following Douglas Co. drawings: 130150, 131198, 131255, 132309, 230678, 233258, 430679, 529827, 529828, 529973, 529974, 530113, 530148, 530234, 530325, 530447, 530448, 531062, 531371, 531404, 531949, 532086, 533596.

In particular drawings #430679 and 533596 give dimensioned three views of S1Wo and S2W2w2 with references to detailed drawings.

The tests were divided into the following broad groups:

1) Three component measurements on wings alone, including scale effect.

2) Effects of fuselage, nacelles, chassis, vertical and horizontal tail surfaces, and windshields; chiefly on lift and drag.

3) Effects of auxiliary airfoils.

4) Longitudinal stability and control.

5) Lateral (including directional) control.

6) Effects of bottom surface flaps.

7) Pressure distribution on an N.A.C.A. cowl.

Since nearly two hundred experimental Runs were made, no detailed list of the Runs will be included, but the results will be discussed in terms of the above grouping in section III.

II. Method of Making Tests and Calculations, and of Presenting Results: Notation.

The normal experimental setup is indicated schematically in Fig. 3. Unfortunately, at the time the tests were made, only five balances were available. Hence for lift, drag, and pitching moment investigations the two yaw balances were not used. For rolling and yawing moment tests the drag balance and one lift balances were moved into the positions of the two yaw balances. For this reason lift and drag could not be measured simultaneously with rolling and yawing moments. (This point it further discussed in III, 5 below).

The tare drag of the pyramid wire systems running to the wing trunnions (cf. Photo 2) was known from previous GALCIT investigations. The tare drag and moment of the sting, tail wire, and counterweight wire were determined by a) testing a wing alone with an auxiliary sting attached, and b) completely enclosing the tail and counterweight wires in streamlined windshields. The tare drag at the high speed attitude of the airplane was about 65% of the total parasite drag at this attitude. The tare moment was extremely small at all angles of attack.

All drags, angles of attack, and pitching moments were corrected by the Prandtl theory of tunnel wall interference to give free air conditions. Rolling and yawing moments and side force were uncorrected for the effect of wall interference.

All observations were reduced to the standard American system of absolute units (the notation for rolling moment, yawing moment, and side force is different from that recommended by the N.A.C.A.)

CL = Lift CD = Drag CM = Stalling

Moment
ρ/2(V2s) ρ/2(V2s) ρ/2(V2s) t

Cs = Side Force CR = Rolling Moment CY = Yawing Moment

ρ/2(V2s) ρ/2(V2s) b ρ/2(V2s) b

where
ρ = mass density of air (note: a correction was applied to the experimental observations so that in this formula ρ is to be taken as the free air density uncorrected for compressibility effects, at least up to 200 m.p.h.)

V = velocity
S = total wing area (See Table 2)
t = mean aerodynamic chord (See Table 2)

The conventions as signs are the same as those used by the N.A.C.A. and are as follows: Taking directions as the pilot sees them CM is positive when it tends to raise the nose, CS is positive when it corresponds to a force to the right, Cr is positive when it tends to lower the right wing, Cy is positive when it tends to move the right wing back. All control surface angles are positive when they tend to increase the lift (or side force) on the surface in question.

In certain cases the parasite drag coefficient CDp was determined (for the wing alone CDp = CDo = profile drag coefficient). For the wing alone the angle of attack for infinite aspect ratio was used, αo. The formulae employed in obtaining these quantities were:

$$\text{CDp (or CDo)} = \text{CD} - \frac{\text{CL2}}{\pi \text{AR}}$$

$$\alpha\text{o} = \alpha - \frac{\text{CL}}{\pi \text{AR}} \bullet 57.3 \text{ (degrees)}$$

where AR = aspect ratio = (span)2/area. It will be noticed that the lift distribution was assumed to be elliptical.

Unless otherwise specified pitching moments are referred to the appropriate C.G. position, cf. Figs. 1, 2. (Original Figures and Drawings not included herein) For wings Wo,W1,Wt the C.G. was assumed to be that of fuselage S1 shown in Fig. 1. For wing W2 fuselage S2 (Fig. 2) was used. In certain pitching moments are given with respect to a point 25% or 22% to the rear of the leading edge relative to the M.A.C., in which cases the figures are labeled so as to cause no confusion. The angle of attack is referred to the thrust axis throughout, even for the wing alone observations.

In making performance estimates and in comparing the effects of various modifications the equivalent parasite area has been used, where

f = equivalent parasite area = parasite drag = CDp S
ρ/2(V2)

Certain additional symbols are used in Section III which are there defined.

It should be mentioned that the plotted experimental points represent direct observations with no fairing, except that the tare drag results were faired before being subtracted from the observed total drags to give the final values.

III. Experimental Results and Discussion.

In view of the tremendous number of individual observations all results are given in the form of plotted experimental points and faired curves, practically no tabular data being given in the Report. Such detailed tabular data are available in the files of the GALCIT. The results are discussed below in accordance with the grouping previously mentioned.

1) Three Component Measurements on Wings Alone (Figs. 4, 5)

The results for the four wings reduced to infinite aspect ratio conditions are plotted in Fig. 4. CLmax for wing W1 is 4 or 5 percent lower than that for the other three which attain the quite normal values (for the GALCIT tunnel) of between 1.30 and 1.32. The rather low value of CDo for wing Wt is not especially surprising when it is considered that Wt was obtained from Wo by adding on large tips of comparatively thin section. The rather curious shape of the CDo curve between CL ~ 0.3 and 1.0 for wing W2 seemed a little suspicious when the results were finally plotted. Hence it was decided to make the series of Runs including 192 as a check on the earlier series including 152. Run 192 was taken ten weeks after Run 152, during which time the balance system had been completely shifted twice and the model completely disassembled and reassembled. The agreement of the two runs as indicated by the two sets of experimental points in Fig. 4 given an idea of the accuracy and reproducibility of the experiments. The value of CDomin = 0.0105 compares very satisfactorily with that obtained at the GALCIT and elsewhere with similar tapered wings. The pitching moment curves are unusual in two respects. First, all four curves indicate that CM is approximately constant in the normal flying range about a point between 1½ and 3 percent forward of the 25% point on the calculated M.A.C. Second, both of the more highly tapered wings W1 and W2 give a considerable and unusual instability just before the stall. With wing W2 this is especially pronounced. The reason for this anomalous behavior is not yet clear. It might be mentioned that a subsequent test at the GALCIT on a tapered wing with considerably more sweepback than W2 gave the same type of result but in a still more exaggerated degree. Whether the effect is due primarily to the large taper or to the sweepback cannot yet be stated.

As indicated by the Run numbers only wings Wo, Wt, and W1 were at first tested. From the results with them it was decided by the Douglas Co. to use wing Wt for the airplane, so that the subsequent Runs up to 150 were made with this wing. On the basis of the stability results discussed in section 4) it was later decided to change to wing W2 with which the subsequent tests were made.

In Fig. 5 infinite aspect ratio characteristics of wing W1 are plotted for three Reynolds' Numbers, in order to give an idea of the scale effect. The results at the lowest Reynolds' Number are considerably less exact than those at the higher velocities as is evidenced by the different amounts of scatter in the corresponding experimental points. Probably the most interesting features, the variation of CLmax and CDomin, are discussed in connection with other results of the same nature in section 8) below. In view of the similarity of the other wings tested to W1 it is probable that the scale effect for them all would be very similar to that shown for W1 in Fig. 5.

2) Effects of Various Modifications; Chiefly on Parasite Drag and CLmax

The first characteristic to be investigated in this category was the fuselage-wing interference. From Fig. 6 it appears that the addition of the fuselage and vertical tail surfaces to wing Wt causes an increase in CDp which is practically uniform over the entire flying range. Also CLmax for the combination is almost identical with that for the wing alone. Hence it appeared that the original fuselage-wing fillet was satisfactory in that there were no unfavorable interference effects. This conclusion was later verified in connection with wing W2 as is indicated in Fig. 9. Unfortunately in the latter case the horizontal tail surfaces could not be removed from the fuselage so that the additional drag and lift shown in Fig. 9 are due to the combination of fuselage interference and horizontal tail surfaces. Since the fuselage-wing fillet appeared to be so satisfactory no further modifications of it were investigated. A series of tests on wing Wt with fuselage and vertical surfaces at various Reynolds' Numbers gave results which are reproduced in Fig. 7. The scale effect is very similar to that given in Fig. 5 for wing W1 alone.

The wing engine nacelles were then mounted on the wing and the additional drag and interference effects noted. The original nacelles (N) gave rise to a considerable decrease in CLmax, to a fairly high ΔCDpmin and to a rather large decrease in effective aspect ratio (e). The first difficulty was ameliorated by the addition of fillets, but the other two did not respond to such simple treatment. A rather elaborate series of modifications was therefore investigated, including changes in cowling, nacelle shape and size, fillets, etc. The most important results are given in Fig. 8. The large variations caused by the various modifications are quite striking, as is the curve for NS corresponding to a case in which the engine and cowling were removed and the nacelle completed with a streamlined nose. It is, of course, possible that with power on the normal nacelle configurations might give curves approximating more closely to that for the streamlined nacelle. As a result of this series of tests it was decided that N4 was the best nacelle investigated and a new wooden nacelle, N7, was copied from it. The final nacelle, N8, which was used for all subsequent tests was identical with N7 except for a small fairing at the wing intersection. The effect of the addition of the final nacelle N8 to wings Wt and W2 is almost identical as may be seen by comparing the curves of Figs. 6 and 9.

The parasite drag coefficients for FV1Wt and FV2W2 with various modifications are plotted in Fig. 10. In Fig. 11 are given the results obtained by replacing the normal wind-shield by a Fokker type (as used on the Boeing 247) and then by eliminating the wind-shield entirely and giving the fuselage a smooth, completely streamlined nose. From these two Figures the increase in parasite drag coefficient caused by adding various parts was determined and plotted in Fig. 12. The extremely small drag added by the horizontal tail surfaces H1, the tail wheel T, the chassis C, and the two types of windshield are noteworthy, as is the tremendous drag of the chassis in the unretracted position Cd.

The final parasite drags are discussed further in section 8 below.

3) Effect of Auxiliary Airfoils

In view of the beneficial influence which the addition of small auxiliary airfoils between the nacelles of the Douglas Dolphin had previously been found to exert, similar investigations were conducted on the present model. Two sets of auxiliary wings running between nacelles and fuselage were tested. Of these w1 had a large chord and w2 a smaller one. Both airfoils were tested at a series of angles of incidence to the fuselage axis.

The results for S1Wt with airfoil w1 (cf. Photos 4 and 7) are given in Figs. 13, 14. Referring to Fig. 13 it appears that for S1Wt the flow near CLmax is unstable so that the wing stalls prematurely and the value of CLmax is rather low. The addition of the auxiliary airfoil apparently stabilizes this flow condition and leads to considerably higher values of CLmax with only a very slight increase in CDp. It appears that a setting of about -5° is approximately the optimum. The influence on pitching moment is noticeable as is indicated in Fig. 14. The auxiliary airfoil decreases the static longitudinal stability perceptibly but not violently.

The model S2W2 was investigated with both auxiliary airfoils, the results being indicated in Fig. 15. The increase in CLmax for practicable settings of the auxiliary airfoil is not nearly so striking as was the case with S1Wt. This is largely because of the fact that at the stall the flow with model S2W2 is more stable than with S1Wt so that CLmax without any auxiliary airfoil is considerably higher in the former case. It appears that the smaller airfoil w2 is perceptibly better than the larger w1 so far as both CLmax and CDpmin are concerned. Again a setting of -5° seems to be about the best possible. From these tests it was decided to build the airplane using airfoil w2 set at approximately -5°. All subsequent tests demoted by S2W2' were made with this configuration.

4) Longitudinal Stability and Control

In view of the fact that the observed longitudinal stability of S1Wt (to be discussed below) was less than the designer had calculated, the moment coefficient for wing Wt alone was calculated about three axes. The results are given in Fig. 16 and indicate that CM is approximately constant about the 22% point of the M.A.C. instead of the 25% point as would be expected. This means that the conventional method of determining M.A.C. is not satisfactory for this wing and explains much of the discrepancy between calculated and observed stabilities.

One unusual feature of the model S1Wt was that the elevators of tail surface H1 were supported from the stabilizer on ball bearing hinges and were statically balanced by a counterweight forward of the hinge axis and inside the fuselage. Clamping arrangements were also provided so that tests could be made with the elevator free or with it clamped at any of a series of angles to the stabilizer. The pitching moment about the C.G. for S1Wt, with elevator free, as well as for Wt, is plotted in Fig. 6. It appears that the stability of S1Wt is small but positive for CL less than 0.8. Above this lift coefficient the model is first neutrally stable and then unstable until the stall. Because of this unsatisfactory behavior another wing, W2, was designed so as to have the same root chord as Wt and also to have the same fuselage connection as Wt, but so as to have its M.A.C. further to the rear and hence give greater stability. At the same time new horizontal and vertical tail surfaces (H2, V2) were constructed. Unfortunately it was not possible to wait for the completion of these surfaces in carrying on the experiments, so that most of the control surface investigations were made with S1Wt(i.e. with Wt and H1).

First a series of measurements was made with elevator free and various stabilizer and flettner settings. The results are given in Fig. 17. The first important conclusion to be drawn from this figure is that, with power off the airplane should trim hands off at approximately the high speed attitude for a stabilizer setting of 0° and a flettner setting of 0°. The next conclusion is that with fixed stabilizer (at 0°) the flettners have much more than enough effectiveness to enable the plane to be trimmed hands off at any attitude in the flying range. Fig. 18 gives the results of corresponding tests made with stabilizer, elevator, and flettner clamped rigidly at various angles. The results are self-explanatory and require no further discussion. In Fig. 19 are plotted the observed elevator angles as function of the angle of attack for various flettner angles with elevator free. Up to 10° throw the elevator angles have approximately the same numerical values as the flettner angles. The flettner effectiveness decreases at flettner angles above about 10° and apparently becomes very small at angles greater than about 20°.

The results of Figs. 17 and 18 have been replotted in Fig. 20, which gives at a glance the stability, trim, and control characteristics of S1Wt with elevator fixed and free. The auxiliary scales to the right, giving approximate origins for the CM scale for various stabilizer, elevator, and flettner settings, furnish a very simple means of seeing quickly what is the effectiveness of the various controls. It appears that the stability is too small to be satisfactory but all controls are extremely powerful.

The differences in CM, from Figs. 17 and 18, for the model with and without horizontal tail surfaces have been plotted in Fig. 21 as CMt pitching moment furnished by the tail. This figure enables a determination of the "tail efficiency factor"

to be made. This factor occurs in a formula for the pitching moment coefficient due to a tail surface which is used by C.B. Millikan in his courses on "Aerodynamics of the Airplane":

$$CMt = CMo' - \eta t \frac{1}{t}\frac{St}{S}\frac{1 - R/\pi(AR)}{1 + R/\pi(ARt)} = CL$$

where

CMt = pitching moment coefficient about the C.G. due to the horizontal tail surfaces.

CMo' = a constant depending on stabilizer and elevator settings.

1 = distance of C.P. of tail behind C.G.

t = mean aerodynamic chord of the wing.

St = total area of horizontal tail surfaces.

S = wing area

R = slope of curve of CL vs. ∝ (radius) for infinite aspect ratio.

~ 5.5 for all normal airfoil sections.

AR = effective aspect ratio of wing cellule.

ARt = aspect ratio of horizontal tail surfaces

ηt = tail efficiency factor.

The above formula is deduced theoretically for elevator fixed to take into account all downwash effects. The factor ηt is an empirical correction factor introduced to take account of the interference effect of a fuselage on the tail. The numerical value of ηt will naturally depend on the particular wing-fuselage tail combination under consideration, but an average value of between 0.75 and 0.80 has been recommended in general.

For the Douglas Transport S1Wt we have (from Table 2 and from Douglas Co. drawings scaled up to full size).

1 = 40.6 ft.	St = 120 ft. 2	S = 944 ft. 2
AR = 7.66	ARt = 4.75	t = 12.06 ft.
		R = 5.5

Hence from the formula

$$\frac{dCMt}{dCL} = -\eta t \frac{40.6}{12.06} \cdot \frac{120}{944} \cdot \frac{0.771}{1.386} = -0.241\,\eta t$$

$$\therefore (dCMt/dCL)\text{theor.} = -0.241\ \eta t$$

From Fig. 21 we have approximately for elevator fixed:

$$(dCMt/dCL)\text{exp.} = -0.15$$

Comparing these two results we find that for the S1Wt model as indicated by the Present tests:

$$\eta t = 0.62$$

This is a somewhat lower value than has been observed before in the GALCIT tunnel, and an explanation of this fact is not as yet apparent. One very interesting characteristic brought out by Fig. 21 is the remarkably slight decrease in tail effectiveness (i.e. in |dCMt/dCL|) which is introduced by changing from elevator fixed to elevator free.

In view of the unsatisfactory stability characteristics of S1Wt, a second wing W2, and a new set of horizontal tail surfaces H2, were constructed, as has already been mentioned. Unfortunately the stabilizer of H2 was built rigidly into the fuselage at a setting of 0°, so that experiments with wing W2 and horizontal ail surfaces removed could not be made. Also, because of the elaborate set of tests with S1Wt, no complete investigation of elevator and flettner effectiveness was undertaken with S2W2. In addition, the elevators of H2 were not statically balanced so that no elevator free measurements could be made. The pitching moments for wing W2 alone and for S2W2 (i.e. stabilizer 0°, elevator 0°, flettner 0°) are plotted in Fig. 9. It is seen that the stability of S2W2 is very good over the normal flying range (dCM/dCL = - 0.15) while even at the stall there is no actual instability.

The effectiveness of elevator and flettner in permitting trim at the stall is discussed later in connection with the measurements on bottom surface flaps (section 6). CM vs. CL curves are given in Fig. 34 in this connection for S2W2 and for S2W2w2 = S2W2'. Comparing the two curves one sees that the addition of w2 decreases the stability very slightly and makes the plane approximately neutrally stable for quite a range of angles of attack at the stall. The curves of Fig. 33 for elevator up indicate ample control to enable the tail to be gotten down for landing. The effect of flaps on stability and trip is discussed in section 6 below.

5) Lateral (including Directional) Control

As mentioned in section II, the number of wind tunnel balances available at the time of the tests was insufficient to permit six components to be measured simultaneously. With the normal "lift, drag, pitching moment" setup the rolling moment about an axis through the diamond shaped suspension frame (cf. Fig. 3) could be measured. It was at first assumed that this was identical with the rolling moment about the fuselage axis and a series of rolling moments with various aileron displacements was made with this normal setup. Later the drag balance and one lift balance were moved to the yaw balance positions. With the latter setup side force, yawing moment, and rolling moment could be determined. It was discovered that the side force was quite appreciable and this introduced rather considerable corrections in transferring rolling moments from the suspension frame axis to the fuselage axis. In addition a slight lack of symmetry in the model or in the airstream was found to cause measurable rolling and yawing moments with no aileron deflection. The final results given here contain correction factors introduced to take all of these elements into consideration and represent averages of Runs made with both balance arrangements.

The first set of observations dealt with the effectiveness of the conventional Frieze type ailerons whose outline is indicated in Fig. 1. Expressed in percent of the gross wing area these ailerons had an area of 11.2% behind the hinge axis or 14.1% overall. The results are given in Figs. 22 or 23. In plotting such aileron curves the convention has been adopted that solid lines correspond to moments of the desirable sign, while dashed lines correspond to reversal of control or to moments of undesirable sign. Fig 22 shows that the aileron effectiveness is good as would be expected from their rather large size. However, the effectiveness near the stall is not as large as Weick recommends for satisfactory control (Cr ~ 0.075 for CL = 1.0). The aileron effectiveness holds up quite well at the stall, no reversal appearing until $\alpha \sim 18°$. The yawing moments (Fig. 23) are favorable for "up aileron" at small angles of attack or large aileron angles. However, for "down aileron" the yawing moments have an unfavorable sign for all conditions tested. For a 2 to 1 differential, the normal arrangement of – 20°, + 10° has unfavorable yawing moments for angles of attack above about 7°.

Since it was planned to investigate the effects of bottom surface flaps a few experiments were next performed on free air ailerons. (A). The ailerons had a symmetrical profile, and an area of 5.85% of the wing area. Their location and shape is indicated in Fig. 24 where the chord line used as a reference axis for angles also appears. All aileron angles refer to the angle which this reference axis makes with the thrust line of the model. It was first determined that the minimum drag occurred with ailerons set at about +3° and this was accordingly chosen as the neutral setting. Since it appeared that plus angles (trailing edge down) gave reversed rolling moments, it was assumed that the aileron system had complete differential, i.e. only negative angles were considered. The results are given in Figs. 25 and 26. The rolling moments are quite satisfactory, comparing very favorably with those for the normal ailerons. The favorable yawing moments are of considerable magnitude. Hence the present wind tunnel tests show very promising characteristics for the free air ailerons. However, flight test results would be required before it could be ascertained whether or not the actual behavior in flight would be satisfactory.

A series of tests on rudder control was next undertaken. In Fig. 27 are plotted Cr and Cy as functions of rudder angle for flettner at 0°. These measurements refer to S1Wt, i.e. to the first vertical surfaces V1. The rolling moments are very small as was to be expected and the yawing moments are quite satisfactory, having nearly twice the maximum value which Warner states is desirable (E.P. Warner, Airplane Design, p. 444). The rudder effectiveness drops off very rapidly at rudder angles above 30°, so that there would be little purpose in designing for throws larger than ± 30°. The flettner effectiveness, shown in Fig. 28, is not quite so satisfactory. The largest rudder angle which can be obtained with rudder free is about 13° and the corresponding yawing moment is less than one half of that corresponding to a rudder angle of 30°. Fig. 29 gives the results of measurements with ailerons fairly well displaced and rudder set to counteract the yawing moment so produced. It appears that the rudder is able to neutralize the aileron yaw but without much favorable yawing moment to spare. For this reason, when the model was rebuilt with wing W2, a new vertical surface V2 was also installed. The results of tests on the complete model SW2' with full rudder angle and with flettner angles of 0° and 15° are given in Fig. 30. The rolling moments remain negligible, but the yawing moments are very large for both configurations and at all angles of attack.

6) Bottom Surface Flaps.

A series of six flaps (denoted by A, 100%, B, C, D, E) which were investigated is shown diagrammatically in Fig. 31. All were of the bottom surface, split trailing edge type, and all were tested at an angle of 45° to the chord line. Three configurations of the horizontal tail surfaces were tested at various times: elevator 0° flettner 0°, elevator -30° flettner 0°, and elevator -30° flettner +150°. The last two were investigated because it was desired to ascertain whether or not the plane could be trimmed at the stall with flaps down. The values of CLmax for the various configurations tested are given tabularly in Fig. 31. Curves of CL vs. α for some of the configurations are given in Fig. 32 and curves for CM vs. CL in Fig. 33. Flaps A and B give very similar results and comparing them with C it is evident that the extension of the flaps under the wing root gives a marked increase in CLmax and a considerable decrease in the undesirable diving moment. On the other hand the further extension under the fuselage as in D gives no appreciable increase in CLmax and only a slight further improvement in the diving moment. A comparison of flaps 100% and C shows that the Vee notch caused by the dihedral at the junction between center section and wing flaps, has no noticeable deleterious effect. The elevators have ample control to permit the plane to be trimmed at the stall with any of the flaps extending completely across the center section, but not with any of the other flaps. All of the pitching moment curves with flaps indicate a region of longitudinal instability from the stalling angle of attack (about 12°) up to about 15°. Hence if the airplane reached an angle of attack of about 12° with flaps down it would rapidly increase its angle of attack to about 15° and would be stable from this angle up. A very interesting and satisfactory result obtained with flaps 100%, C, D, and E is that the CM vs. CL curve with flaps down is very nearly an extension of the corresponding curve without flaps. This means that lowering the flaps in flight would be accompanied by only very small moments altering the trim. A complete set of curves of CD, α. CM vs. CL for the W2 wing only, SW2, SW2', and SW2' with flaps B, C, and E is given for comparative purposes in Fig. 34.

7) Pressure Distribution on N.A.C.A. Cowl

An investigation of the pressure distribution at the surface of the N.A.C.A. cowl on one of the engine nacelles was made using a multiple tube manometer. The setup is shown in Photo 8. A special N.A.C.A. cowling was built up having five pressure orifices on the inside of the cowl spaced along a fore and aft plane as shown in the pressure plots. Small copper tubes ran from these orifices to the rear outside of the cowl and were connected to the manometer by long rubber tubes. 4/9ths of the circumference from these orifices (so as not to interfere with engine cylinders) was

a similar row of pressure orifices on the outer surface of the cowl. The tubing for these orifices ran to the rear inside the cowling. In order to investigate a possible anti-symmetry in the flow the entire cowl was removed after one set of observations, was then rotated through 4/9ths of a revolution, and the observations repeated. The circumferential location of the orifices is indicated on Fig. 35.

The individual pressures were measured relative to an arbitrary origin which was approximately the static pressure in the wind tunnel working section. All pressures were then divided by the dynamic pressure q and plotted as vectors giving the ratio of p to q. Pressures measured at the outer surface are plotted as solid vectors in Figs. 35 and 36, inner surface pressures as dotted vectors. The sign of the pressures, i.e. whether suction or positive pressure is indicated by the arrows on the vectors. In view of the rather arbitrary origin for the measurement of the individual pressures, the algebraic sum of the two pertinent vectors should be used to give the net force on the cowl at any point. In other words the external and internal pressures should be considered together and not separately. From the two figures it is clear that there are considerably bursting forces acting on the cowl as well as a resultant force tending to pull the cowl forward into the propeller.

8) Performance Estimation

In this section estimates are made of the full scale values of the aerodynamic parameters entering into a normal performance calculations. These are extrapolated from the wind tunnel results given in the present report. Actual performance computations are not attempted in view of uncertainties in the final non-aerodynamic design parameters. The three aerodynamic parameters necessary for a performance calculation are CLmax, the airplane efficiency factor e, and the equivalent parasite area f. We discuss these individually below.

CLmax and Stalling Speed

In Fig. 37 are plotted values of CLmax vs. R for wing N.A.C.A. 2412 and model FV1Wt from results obtained at the GALCIT. The values of CLmax interpolated from the results of the present tests for S2W2' and S2W2' with flap E are also plotted. The two dotted curves represent the author's judgment as to a reasonable extrapolation of these results to full scale at the stalling attitudes. From the design data and these curves it appears that the full scale stalling speeds, Reynolds' Numbers, and CLmax's for the two cases are:

	Uncorrected Model Results			Model Results Extrapolated to full scale from Fig. 37		
	CLmax	Vs(mph)	R model	CLmax	Vs(mph)	R full scale
S2W2'	1.30	73	1.6 x 106	1.40	71	7.9 x 106
S2W2' with flap E	1.90	61	1.6 x 106	2.00	59	6.6 x 106

These are obtained from Oswald's Fig. 40, (W. Bailey Oswald "General Formulas and Charts for the Calculation of Airplane Performance", Technical Report No. 408, 1932.), assuming a wing loading of 17000/939 = 18.1 lb/ft2.

Efficiency factor, e.

The curve for S2W2' of Fig. 34 is used to obtain an estimate of e. This is done by constructing the induced drag parabola which gives, as nearly as possible, constant CDp over the flying range, and finding the effective aspect ratio ARe corresponding to this parabola, where CD1 = CL2/πARe

Then e is determined from e = ARe/AR = ARe/7.70

If one determines the parabolas which give CDp = CDpmin for a series of values of CL, and calculates e for each parabola one gets approximately the following

CL =	0.4	0.5	0.6	0.7	0.8	0.9
e =	0.91	0.88	0.85	0.82	0.78	0.74

Hence e is not constant but decreases continually as CL increases. This is apparently due to the unfavorable interference effects between wing and nacelles, since the variation of e is much less when the nacelles are removed. For power-on flight it is probably that these unfavorable effects are much reduced by the action of the slipstream. "e" is of principal importance for performance estimation in the calculation of maximum rate of climb at sea-level and absolute ceiling. For the present airplane a rough calculation shows that the first occurs at CL ~ 0.5, while the second occurs at CL ~ 0.8. Hence for power-on performance calculations a value of e of 0.85 should be conservative. We therefore take

e = 0.85

For single engined ceiling estimates it may be necessary to decrease this value somewhat.

Equivalent parasite area, f.

From Fig. 15 (Run 166) we get for the model S2W2'. CDpmin = 0.0208. We denote the value of f corresponding to this unextrapolated result by fm and obtain

fm = 0.0208 x 939 = 19.5 ft.2 ~ R = 1.6 x 106

The first estimate which is required deals with the effect of the increase to full scale Reynolds' Numbers on the equivalent parasite area. Karman has recently given a theory of turbulence skin friction for smooth surfaces which apparently gives a good approximation to experiment even for very large Reynolds' Numbers. This theory is presented in "Proceedings of the Third International Congress for Applied Mechanics", Stockholm, 1930, Vol. I, page 85. If we write

F = Cf (ρ/2) U2bl

Where

F = skin friction on a smooth plane surface of breadth b and length l (in the direction of flow)

U = undisturbed velocity far from the plate

ρ = fluid density

Cf = skin friction coefficient

R = Reynolds' Number = U1/V

Then Cf is given as a function of R in Fig. 38 which is reproduced from Fig. 6 of "Quelques Problemes Actuels de l'Aerodynamique" by Th. von Karman (Paris 1932).

We can use this theory in getting an estimate of the effect of Reynolds' Number on the drag of the model discussed in this report. We consider two limiting cases:

Case 1: The wing profile drag is assumed to arise entirely from turbulent skin friction for which Karman's theory is applicable. The remaining parasite drag is assumed to arise from eddy resistance and to have a drag coefficient independent of R. The characteristic length 1, is assumed to be the M.A.C.

Case 2: The entire parasite drag is assumed to be caused by turbulent skin friction. The characteristic length 1, is again taken as the M.A.C. since it is assumed that the large length of the fuselage and the small lengths of nacelles, tail surfaces, etc. will lead to an average effective length of about the wing chord magnitude.

For both cases we have:

CDpfull scale = Cf (Rfull scale)
CDpmodel = Cf (Rmodel)

where CDp is the portion of the parasite drag coefficient which is assumed to vary with R. We use subscripts 1, 2 to denote the two cases.

For Case 1 we take the wing profile drag

CDpomodel = 0.0105 Rmodel = 1.6 x 106

For Case 2

CDp2model = 0.0208 Rmodel = 1.6 x 106

From preliminary estimates we get Vmax ~ 200 m.p.h. The corresponding full scale Reynolds' Number is

Rfull scale ~ 22 x 106

From Fig. 38 this gives

Cfmodel = 0.0042 Cffull scale = 0.0027

Hence for both Case 1 and Case 2

Cf(Rfull scale) = 0.0027 = 0.64
Cf(Rmodel) 0.0042

In Case 1 we have now

CDpofull scale = 0.0105 x 0.64 = 0.0067

To this must be added the constant portion of CDp = 0.0208 – 0.0105 = 0.0103.

Hence finally

CDp1 = 0.0172 f1 = 16.1 ft.2

In Case 2

CDp2full scale = 0.0208 x 0.64 = 0.0133

∴ CDp2 = 0.0133 f2 = 12.5 ft.2

We must now estimate the effect of the lap joints and rivets of the actual airplane which were not present on the model. From N.A.C.A. Technical Note #457 one can estimate that the lap joints may increase the wing profile drag 3 or 4%, corresponding to a Δ CDp = 0.0003. From Technical Note #461, Fig. 6 it appears that the addition of a normal arrangement of rivets adds a drag to a smooth wing given by Δ CDp = 0.0018. For the fuselage and tail surfaces the presence of rivets, window depressions, etc., may be estimated to furnish an additional Δ CDp ~ 0.0009. Hence one may estimate the addition of rivets, roughness, etc., to furnish

Δ CDp = 0.0030 Δf = 2.8 ft.2

Collection the results we have

	CDp	f
Model results scaled up without modification	0.0208	19.5 ft.2
Model results scaled up and roughness included Case 1	0.0202	19.0 ft.2
Model results scaled up and roughness included Case 2	0.0183	15.3 ft.2

Case 1 is probably too pessimistic and Case 2 too optimistic, so that the value to be actually expected should lie between the two. It should be explicitly noted that no radio mast or antenna has been considered, nor has any other departure from the model configuration except rivets and surface roughness.

CONCLUSION

The above experiments were carried out from December 12, 1932 to May 25, 1933 under the direction of Drs. A. L. Klein and C. E. Millikan who were largely assisted by Messrs. N. B. Moore, Rosooe Mills, W. Bowen, A. Reed, and other graduate students at the GALCIT.

Guggenheim Aeronautics Laboratory
California Institute of Technology
Pasadena, California
June 7, 1933

Document 3-22(b), Engineering Department, Douglas Aircraft Company, "Development of the Douglas Transport," Technical Data Report SW-157A, undated (ca. 1933-34).

DEVELOPMENT OF THE DOUGLAS TRANSPORT

INTRODUCTION

After fourteen years of continuous, successful experience in building airplanes of all types for the United States Army, Navy, Post Office Department and Coast Guard and for private persons and foreign governments, the Douglas Aircraft Company started plans for the design and development of a high performance passenger airplane for airline use. Profiting by extensive experience in the design and production of aircraft, the Company decided to make an extremely thorough investigation of all factors, however minor, that might affect performance and passenger comfort. Before construction was started, hundreds of wind tunnel and structural tests were made in addition to an intensive mock-up investigation and studies and tests of special items, such as fuel systems, control mechanisms, heating, lighting and ventilating systems and sound control. When the various parts of the airplane were ready, they were each tested to show their static strength and freedom from vibration or flutter.

The finished airplane was, in all probability, subjected to more thorough flight tests than any other known type of passenger transport or even military airplane. Over two hundred flying hours and fifteen thousand gallons of fuel were used in making these flight tests. Not only were the usual tests for speed, stability and general performance made but also tests subjecting the airplane to dynamic loads in flight to prove its structural strength, to determine the best soundproofing practical, to eliminate vibration and to determine the effect of certain variables, such as different engine cowls, fairings, oil temperature regulators, propellers, wing and control surface flaps, engine cooling and power. In conjunction with these tests, several entirely new conceptions in flight testing were put into practice and a new technique for airline cruising operation was developed.

The development cost of the first airplane, including all research directly connected with the project, was approximately $325,000. In addition, the airplane incorporates a great amount of the experience obtained during airline operation of the highly successful single-engined Northrop transports, which represent an engineering and development cost of approximately $290,000.

It is desired to outline briefly in the following pages some of the work done in the development of the Douglas DC-1 and its successor, the DC-2. Tests are still being carried on daily, both on the ground and in flight, to improve and refine this airplane and make it a still more superior product, both from a manufacturing and an operating viewpoint.

AERODYNAMIC DEVELOPMENT

The aerodynamic design of the Douglas Transport was the subject of exhaustive study for a period of more than eighteen months. This study included aerodynamic calculations and wind tunnel and flight tests, which were carried out in a scientific and comprehensive manner. Through the correlation of these calculations and data, it was possible to predict and analyze the actual aerodynamic characteristics which were later obtained in service. The high degree of performance and safety offered by the Transport is the realization of features that have been thoroughly studied and tested in the wind tunnel and in flight.

The aerodynamic calculations were particularly concerned with performance and control at all attitudes of flight, both in normal and single-engine operating conditions. Special design of the controls, wing and fairing makes possible continued single-engine operation at high altitudes with sufficient controllability to ensure safety for meeting emergency conditions. The performance studies for obtaining the desired velocity, range and climb led to the choice of the bi-motor type with controllable-pitch propellers and high-lift wing flaps being adopted as best meeting the requirements of the high-performance airliner. The flaps give a gain in lift of 35% and a drag increase of 300%.

An extensive series of wind tunnel tests, including approximately 200 test runs, were carried out on a one-eleventh scale model of the Transport in the 200 mile-an-hour wind tunnel at the California Institute of Technology.

The large scale of the model and high speed of the tests were particularly valuable for this work. All items of the airplane affecting aerodynamic operation were tested with the view not only of obtaining the desired performance, stability and controllability, but also of perfecting each item to the greatest practical degree. Briefly, the investigation included tests on three complete wings with various modifications, various wings to fuselage fillets, tail surfaces, landing gears end tail wheels, several sets of ailerons, of normal and special types, six arrangements of high-lift wing flap devices, and other special arrangements. Tests on controllability and stability were made with controls both fixed and free. The lift and drag of the final model were tested at various Reynolds numbers in order to indicate the trend in passing to full-scale. The wind tunnel tests resulted in the final aerodynamic design providing an increased degree of performance with satisfactory stability and ample controllability for all normal and emergency conditions of flight.

It is interesting to note that some of the early models tested in the wind tunnel showed instability and that the tests revealed that it was necessary for satisfactory stability to have a hitherto untried arrangement of center of gravity, wing sweepback and general configuration. The actual airplane was built in accordance with this new plan of arrangement and the stability in flight proved to be exactly as predicted. If the wind tunnel tests had not been made, it is very possible that the airplane would have been unstable because ordinary investigation had indicated that the original arrangement was satisfactory.

The actual measured flight test results showed an excellent agreement with predicted performance in all phases and fully justified the extensive aerodynamic study and wind tunnel investigation. These flight data have further been used to modify aerodynamic features that indicated possible improvement, so that the final aerodynamic characteristics of the Douglas Transport are extremely satisfactory and very advanced for a transport airplane. In fact, the total resistance of the complete airplane is less than twice the resistance of the wing alone.

STRUCTURAL DEVELOPMENT

The studies of aerodynamics and general arrangement showed the desirability of having the engine nacelles well ahead of the wing leading edge. It was also found desirable to house the retractable landing gear within the nacelles. Sweeping back the outer part of the wing offered the advantages of getting the landing gear well forward of the center of gravity and having the center of gravity come well forward on the wing for stability. With these points in mind and recognizing the fact that the size and performance desired for this machine presented an entirely new problem, an exhaustive study of the various possible types of construction was made.

In developing a structure having the maximum strength and rigidity with a minimum of weight, it is preferable to design a wing with the material so distributed that there is no great variation in the stresses in the various parts. Such variation is apt to be caused by rigidly attaching very thin members, such as the skin, to very heavy members, such as spars or beams with heavy stresses, if very thorough and careful investigation of the distribution of loads, deflections, local stresses, etc., is not made. At the same time, the wing must have little or no torsional deflection, a minimum of vertical deflection, and no excessively large unsupported flat metal surfaces.

A first investigation showed that most metal wings were merely an adaptation of wooden designs in other material. However, the characteristics of wood and metal are quite different and, therefore, the design principles of one do not apply to the other. In a metal wing, having a thin skin rigidly attached to a heavy spar, sudden changes in cross section are apt to cause very objectionable stress concentrations. If precisely the proper proportions of material are not made, or if the designs of the various attachments are not exactly correct, there are apt to be cracks in the skin and popping of rivet heads due to the deflecting spars pulling against the skin.

In the Douglas and Northrop types of multi-cellular wing construction, there are a multiplicity of full length span-wise stiffeners, and the fact that they have no abrupt changes or "breaks" results in no concentration of stresses. With the centroids of the stiffeners located at the maximum distances from the neutral axis of the section, a most efficient structure for absorbing the bending load is obtained.

In a highly stressed airplane, torsional rigidity of the wing is of paramount importance in the prevention of wing flutter at high speeds and torsional deflection of the structure must therefore be kept to an absolute minimum. When under load, there will always be some vertical deflection but this must not be excessive since a wing with large vertical deflections might cause jamming of aileron controls and by no means inspires confidence in the passengers or pilots.

If unsupported flat metal surfaces are even moderately large, there is always a tendency for the middle of the surface to vibrate in flight even when there is no stress. This is termed "oil canning" and will, in time, cause fatigue in the sheet metal and in the rivets and cause rivet heads to work and to pop off. These unsupported flat surfaces continually drum and cause a noise that cannot be completely eliminated in a cabin because part is carried as vibration through the structure. Even when on the ground with the engines running, this "oil can" action and drumming is apparent. "Oil can" action should be differentiated from wrinkling in the skin. Wrinkling of the skin will be present in every metal wing with a flat metal covering taking stress. These wrinkles are deflections of the skin under load and ordinarily do not have any tendency to vibrate.

In determining the wing construction of the Douglas Transport, single, two, three and multi spar designs were considered as well as shell type and multi-cellular designs. .

After a thorough investigation of all types, the Northrop multi-cellular wing construction was finally decided upon. This type of structure consists of a flat skin reinforced by numerous longitudinals and ribs. The bending is taken by the combination of flat skin and full length stringers. Three main flat sheets or webs carry the shear loads and torsion and indirect stress are carried by the skin with frequent ribs preserving the contour and dividing the structure up into a number of small rigid boxes or cells. Since the major loads are carried in the outer surface of the wing as well as in the internal structure, an inspection of the exterior gives a ready indication of the structural condition. The unit stresses in the material are low and therefore the deflections are at a minimum giving a maximum in rigidity. This construction has proven to be a happy medium of those considered since it combines practically all of the advantages of each; namely, very small unsupported areas, extreme lightness for its strength and rigidity, also ease of construction, inspection, maintenance and repair. The Northrop wing being comparatively small, it is economical to have many of the stringers run from the top to the bottom of the wing as shear webs or spars. However, when the principle is carried out on a larger scale, as in the Douglas Transport with its deeper wing, it is more efficient to have only three shear webs or spars. Thus it was not necessary to evolve a new type of structure but merely to adapt a time proven type to the dimensions of the Douglas Transport.

In the fuselage, the structural problem was basically the same. However, the Douglas Company had had extensive experience in building metal monocoque fuselages. This experience, combined with that of the Northrop Company, resulted in the present fuselage construction. This construction consists of a smooth, stressed skin in contact with closely spaced over-strength bulkheads and numerous longitudinal stringers (either flanged members or extruded angles) as a rigid part of the skin passing through the bulkheads, thus all parts are securely attached together and the skin has very small unsupported areas.

The coast to coast airline, Transcontinental and Western Air, Inc., which has been using a fleet of Northrop mail planes in daily service with notable satisfaction, advised on the design of the Douglas Transport from an operator's viewpoint. The airline encouraged this type of wing, fuselage and tail construction principally because their actual experience of many thousands of flying hours in hard service with the Northrop mail planes showed that the maintenance costs of this type of construction are negligible.

CONCLUSION

It is gratifying to note that the time and expense of all the preliminary aerodynamic, wind tunnel, mock-up and design studies were more than justified by the results obtained. The performance, stability characteristics and wing deflections, as determined in flight, conformed almost exactly with the predicted results. In fact, no major changes were necessary in the arrangement of the various parts of the airplane. Similarly, other wind tunnel predictions were proven in flight to be accurate.

The superior strength and rigidity of the Douglas multi-cellular wing and all-metal fuselage construction has been proven in both static and dynamic tests as well as in service to more than justify the time and expense of the thorough investigation made of the structure. There is no doubt left regarding the strength and reliability of any part.

From the viewpoint of passenger and pilot comfort, the mock-up, soundproofing, heating and ventilating investigations have more than proven their value as shown in the quietness and comfort of this multi-engined transport.

To add further to the completeness and excellence of this airplane, carefully worked out maintenance aids have been so constructed and mounted as to provide for servicing and replacement with a minimum of time and expense. In fact, the complete power plant section, including the engine, propeller, oil tank and all cowling, may be completely removed in seventeen minutes.

In general, this airplane is the product of a painstaking study of all the problems concerned and a thorough and methodical investigation of every possible solution, combined with the extensive experience of the Douglas Company in producing a great quantity of experimental and production airplanes. The Douglas Transport takes the air fourfold in supremacy - in comfort, performance, safety and service - the luxury liner of the airways.

Document 3-23

"Douglas Airliner for Transcontinental Service," *Aviation* 32 (October 1933): 331-332.

The Douglas DC-series aircraft were marvels of their age. Major innovations in aerodynamics, structures, propulsion, and flight instrumentation came together in an elegant way to produce the world's most advanced long-range airplane. In this article from October 1933, *Aviation* magazine introduced the DC-1 to the aeronautical community. The article provides an excellent summary of the innovations employed in the DC-1 design.

Document 3-23, "Douglas Airliner for Transcontinental Service,"* Aviation *32 (October 1933).

Except for a few amphibians of the "Dolphin" class delivered for private use or for limited transport services, the output of the Douglas plant at Santa Monica, Cal., has to date been definitely militaristic. To meet Transcontinental & Western Air's requirements for new equipment, however, Douglas has entered the commercial field on a large scale. The delivery of the first of an order of twenty DC-1 transport airplanes marks the beginning of a new high-speed shuttle service between New York and Los Angeles over the TWA system.

Safety, speed, comfort and economy have been the keynotes of 1933 transport airplane design and an analysis of the specifications indicates that the new Douglas machine yields an exceptionally high rating on all four points. Under the head of safety may be listed such features as all-metal construction; a comparatively great amount of shock-absorbing structure ahead of and below the passenger compartment; high performance with one engine cut (an altitude of 9,000 ft. has been reported on one engine with full load); low landing speed (60 miles per hour at sea level), and steep gliding angles through the use of trailing edge flaps; the ability to land without damage (except to propellers) and with full braking facilities with wheels fully retracted; a wide range of vision from the pilot's cockpit (including a full view of the retracting undercarriage); quick-acting dump valve on fuel tanks; full fire-extinguishing equipment, and a full complement of the latest navigational and communication instruments.

High speed was forecast through exhaustive wind tunnel tests on interference and careful attention to aerodynamic cleanness. Engine nacelle positions follow N.A.C.A. recommendations for low drag and high propulsive efficiency. The ship shows a high speed of 188 m.p.h. at sea level, 210 m.p.h. at 8,000 ft. It cruises at 184 m.p.h. at sea level and at 190 m.p.h. at 8,000 ft., all figures being in excess of any so far reported for ships of its size and capacity. With the landing speed indicated above, the speed range of this airplane is worthy of note.

On the score of passenger comfort, nothing has been overlooked. The cabin, 6 ft. 3 in. high throughout and 5 ft. 6 in. wide, normally accommodates 14 passengers in 7 rows of two each, spaced 40 in. from seat back to seat back. For short hauls where extreme roominess is not essential (and where gasoline capacity can be reduced) accommodations can be installed for 21 passengers. The seats are of special Douglas design, fully adjustable, and are mounted individually on rubber to minimize direct vibration. Seat backs are reversible to permit passengers to sit face to face if desired. Since the cabin floor passes over the top of the wing structure, there is no obstruction of any sort in the cabin. Entrance is through a door on the left side of the fuselage in the rear. Aft of the door is a complete buffet, and beyond, a fully equipped lavatory.

Stephen J. Zand, acoustical engineer of the Sperry-Gyroscope Company, whose work with the new CurtissWright Condor has been previously reported (AVIATION, July, 1933) handled the sound-proofing of the new Douglas. By careful attention to rubber-insulating the engine mounts; by eliminating all direct contact between the structure and the cabin lining; by extraordinary care to eliminate all direct leaks (even to the extent of providing rubber gaskets for the cabin doors and designing special door locks without keyholes); and by a liberal use of Seapack (kapok processed into sheet form) and other sound-deadening materials in the 3 in. space between outer and inner shell, the average sound intensity in the cabin at cruising speed was reduced below 70 decibels—an outstanding achievement.

In connection with the sound-proofing development a complete ventilating and steam heating system has been worked out. Controlled ventilation is effected by admitting air through a vent in the nose of the fuselage and distributing it throughout the ship by ducts. A thermostatic control ensures that the temperature in the cabin will remain constant at 70 deg. With outside air temperatures going as low as –20 deg. F.

Economy results from good aerodynamic design (maximum performance from minimum power) and practical common-sense arrangement of component parts (for minimum maintenance and servicing expense). Exhaustive wind tunnel tests and careful calculations ensured acceptability on the first count, and an extensive specification of features required for minimum maintenance based on TWA's long operating experience guaranteed the second.

Getting down to structural details the entire machine is built of Aluminum Company of America's 24 ST and 24 SRT Alclad. The wing is of a cellular multi-web construction (similar to that used in the Northrop Gamma and Delta) and is tapered in plan and in thickness. N.A.C.A. Airfoil Section No. 2215 was used at the root, and No. 2209 at the tip. The central portion of the wing is built integral with the fuselage and serves as a mounting for the engine nacelles and the retractable landing gear. Outer wing panels are demountable by means of bolted joints. Two

main fuel tanks of 180 gal. each and two auxiliary tanks of 70 gal. each are mounted in the center section on each side of the fuselage.

The fuselage itself is a full monocoque with both the vertical fin and the horizontal stabilizer built integral. Longitudinal and directional trim are obtained by tabs in the rudder and elevators. The filleting of the wing and tail surface intersections has been carefully studied for minimum interference effect. All controls, including those for the trimming tabs, are internal.

Landing wheels retract upward and forward into the engine nacelles by a simple hydraulic mechanism. Counterbalancing the landing gear has made possible the use of hand operation only. Retraction is accomplished in 25 seconds and lowering in 20 seconds by means of a pump within easy reach of either pilot or co-pilot. When in the retracted position the axles rest in sockets attached to the main nacelle bulkhead. Hydraulic brakes with a differential control operating through the rudder pedals are provided. Shock absorbers are of the Douglas hydraulic type. A 42x15.00-16 tire is used.

The engine nacelles are monocoque except for the steel tube supporting structure forward of the firewall. The entire mount, including engine and all accessories, is quickly detachable and interchangeable right and left. Removal is facilitated by grouping all connections at the firewall and by using quickly detachable plugs for all electrical connections. Carburetor air intake is of sufficient capacity to prevent icing. Direct-cranking electric starters with shielded booster coils are used. Hamilton Standard controllable pitch three-bladed propellers are standard equipment.

Every possible feature for the comfort, convenience and safety of the pilots has been included. The control columns are located between the pilots and the outside walls so that there is no obstruction between the seats. Full instrument equipment has been installed on rubber-mounted vibration-proof boards. Under TWA specifications the latest type of Western Electric two-way radio has been installed including directional beacon receiver. All wiring for the radio or for other electric circuits is carried in aluminum conduit.

The ship is designed to accommodate either two Wright Cyclone F-3 engines (geared 11:16), each developing 710 hp. at 8,000 ft., or two Pratt & Whitney Hornet D geared engines with an output of 700 hp. at 6,500 ft. The general specifications as given by the manufacturer are: Length over-all, 60 ft.; span, 85 ft.; wing area, 948.6 sq. ft.; fin area, 28.9 sq. ft.; rudder area, 42.5 sq. ft.; stabilizer area, 97.6 sq. ft.; elevator area, 48.0 sq. ft.; aileron area (total), 86.8 sq. ft.; weight empty, 11,780 lb.; useful load (including radio and all equipment), 5,720 lb.; gross weight, 17,500 lb.; power loading, 12.3 lb. per hp.; wing loading, 18.5 lb. per sq. ft.

Document 3-24

Donald W. Douglas, Sr., "The Development and Reliability of the Modern Multi-Engine Air Liner," *The Journal of the Royal Aeronautical Society* (November 1935): 1010-1046.

In November 1935, Donald W. Douglas, then the president of the Institute of Aeronautical Sciences (since the 1960s, the American Institute of Aeronautics and Astronautics), addressed the Royal Aeronautical Society in London. What follows below is the section of his address concerning the major developments in aerodynamic design that had been helping to reinvent the airplane in the 1930s. In his speech, Douglas stressed the importance of single-engine operation within the demands of long-range operation, an issue that set the American flight environment apart from Europe and that served as a major stimulating factor in U.S. technological development.

Document 3-24, Donald W. Douglas, Sr., "The Development and Reliability of the Modern Multi-Engine Air Liner,"* The Journal of the Royal Aeronautical Society *(November 1935).

THE DEVELOPMENTS AND RELIABILITY OF THE MODERN

MULTI-ENGINE AIR LINER

(with special reference to Multi-engine Airplanes after Engine Failure)

By
DONALD W. DOUGLAS, Esq.
(President of the Institute of Aeronautical Sciences)

INTRODUCTION

Four essential features are generally required of any form of transportation: Speed, safety, comfort and economy. The airplane must compete with other forms of transportation and with other airplanes. The greater speed of aircraft travel justifies a certain increase in cost. The newer transport airplanes are comparable with, if not superior to, other means of transportation. Safety is of special importance, and improvement in this direction demands the airplane designer's best efforts.

SAFETY AND RELIABILITY

Statistics show that the foremost cause of accident is still the forced landing. The multi-engine airplane, capable of flying with one or more engines not operating, is the direct answer to the dangers of an engine failure. It is quite apparent, however, that for an airplane that is not capable of flying with one engine dead the risk increases with the number of engines installed. Hence, from the standpoint of forced landings, it is not desirable that an airplane be multi-engine unless it can maintain altitude over any portion of the air line with at least one engine dead. Furthermore, the risk increases with the number of remaining engines needed to maintain the required altitude. In general, therefore, the greatest safety is obtained from—

1. The largest number of engines that can be cut out without the ceiling of the airplane falling below a required value;
2. The smallest number of engines on which the airplane can maintain this given altitude.

For airplanes equipped with from one to four engines, it follows that the order of safety is according to the list below.

a. Four-engine airplane requiring 1 engine to maintain given altitude.
b. Three-engine airplane requiring 1 engine to maintain given altitude.
c. Four-engine airplane requiring 2 engines to maintain given altitude.
d. Two-engine airplane requiring 1 engine to maintain given altitude.
e. Three-engine airplane requiring 2 engines to maintain given altitude.
f. Four-engine airplane requiring 3 engines to maintain given altitude.
g. One-engine airplane requiring 1 engine to maintain given altitude.
h. Two-engine airplane requiring 2 engines to maintain given altitude.
i. Three-engine airplane requiring 3 engines to maintain given altitude.
j. Four-engine airplane requiring 4 engines to maintain given altitude.

Reliability and safety, however, depend upon other factors. These can generally be classified as the airline over which the airplane operates, and the aerodynamic design of the airplane. The airplane designer has control of reliability in so far as he is able to modify the multi-engine design to enable it to complete its flight over the necessary terrain after engine failure.

…performance has already been calculated or determined from flight test, Δ and l_t can be obtained from the normal full-power curves in Figs. 17 and 18.

2. The percentage drag increases resulting from the idling propeller and additional drag of the airplane are obtained from Figs 5 to 10, hence z is determined. Different values of z should be used for determining the various characteristics.

3. The principal performance characteristics are found from the normal values of the parameters l_p, l_s, l_t and Δ, and z by Figs. 16, 17 and 18. Alternately, the new values of the parameters can be calculated, or obtained through Figs. 13, 14 and 15, and z. Then the charts of Reference I are used to obtain the performance after engine failures.

AERODYNAMIC DESIGN

Achieving reliability in multi-engine airplanes is largely dependent upon the proper aerodynamic design of the airplane. It is necessary that special attention be given to the design of the airplane for flight with one or more engines out of operation. The multi-engine airplane should be definitely designed for this condition if the reliability consistent with the additional complication of many engines is to be realized. As had been previously pointed out, safety and reliability of the multi-engine airplane are not necessarily greater than that of a single-engine airplane, unless after engine failure it is capable of completing its mission in an efficient manner with adequate control.

Certain aerodynamic features are available which enable the engineer to design a multi-engine airplane that has good performance after engine failure. The principal considerations for obtaining most favorable operation are outlined below:

1. The rudder should produce sufficient yawing moment at the zero yaw attitude of flight to counterbalance the moments resulting from offset thrust, idling propeller drag, and any adverse yaw due to aileron action. The magnitude of this moment should be ample to provide for the lowest velocity intended for operation, usually near the stall.

2. The vertical surface should be designed as an efficient lifting surface based essentially on the criteria that would be used for judging a good wing design. It should be of high aspect ratio and so disposed that the efficiency of the surface is not impaired by adverse interference effects. The rudder angle necessary for producing equilibrium should not be too large because of the high parasite drag that might result.

3. A rudder flap is needed of sufficient size to hold the rudder in the required position with zero control force. Likewise, the ailerons and elevators require trimming flaps sufficient to enable the airplane to be flown "hands off."

4. Propellers should be of the feathering type in order to reduce their

drag when an engine fails. When turning against the torque of the engine, the negative drag of a propeller in high pitch is slightly less than when in low pitch. For higher pitch settings a free-wheeling propeller has considerably more drag than the feathered type, but less than when it is turning against engine torque. Reducing the propeller drag decreases the rudder size required and the drag of the rudder.

5. Span loading of the airplane should be as small as feasible (that is, large span and/or low weight) since span and weight have the greatest effect on ceiling of any of the airplane parameters. The problem of flight after engine failure is largely one of designing an airplane that has more than the usual capabilities in absolute ceiling.

6. The general "cleanness" of design of the airplane and efficiency of all component parts must be kept at their highest possible value, since all favorable and unfavorable effects are magnified when operating after engine failure.

7. The rudder moment required for balance should be investigated at the ceiling; especially for airplanes with highly supercharged engines, because aerodynamic force would there be reduced while thrust still remained large. In fact, bi-motored airplanes with highly supercharged engines will be found to require for single-engine operation a ruddersomewhat out of proportion to that which would normally be satisfactory.

All of these factors should be given due consideration if it is desired to achieve the reliability and safety factor possible when flying with one or more engines out of operation.

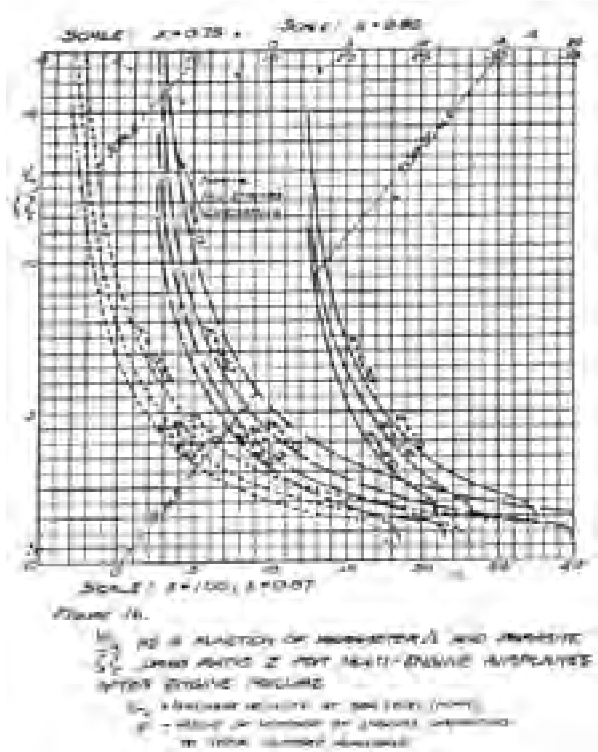
EXAMPLE OF USE OF METHOD

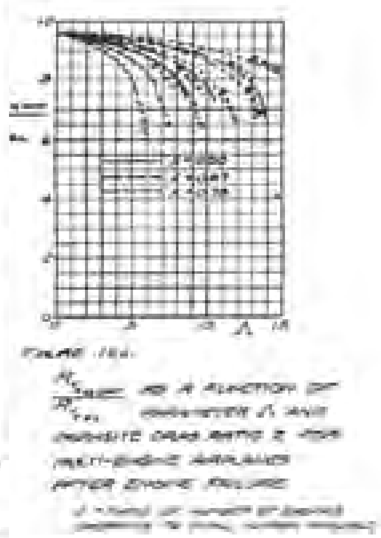
In order to illustrate the application of this method and the agreement between flight test results and calculations, the Douglas DC-2 Bi-motor Transport is used as an example since complete data on it are available.

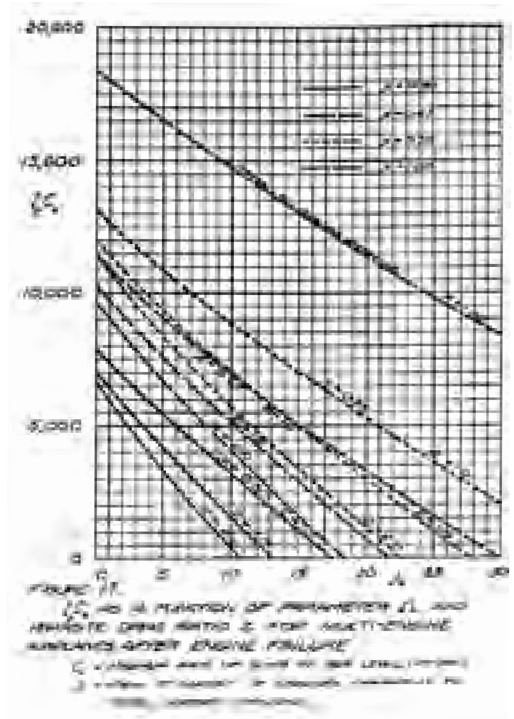
Flight test results for full gross load reduced to standard performance conditions are as follows:

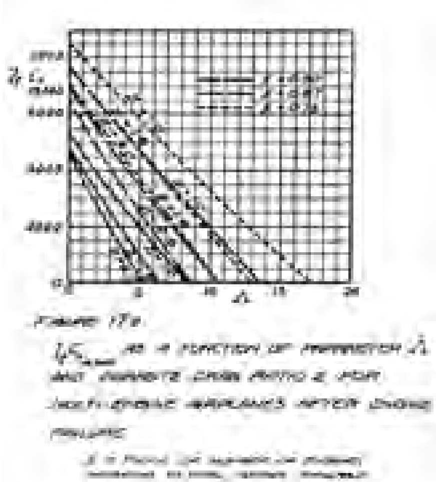
ON FULL POWER

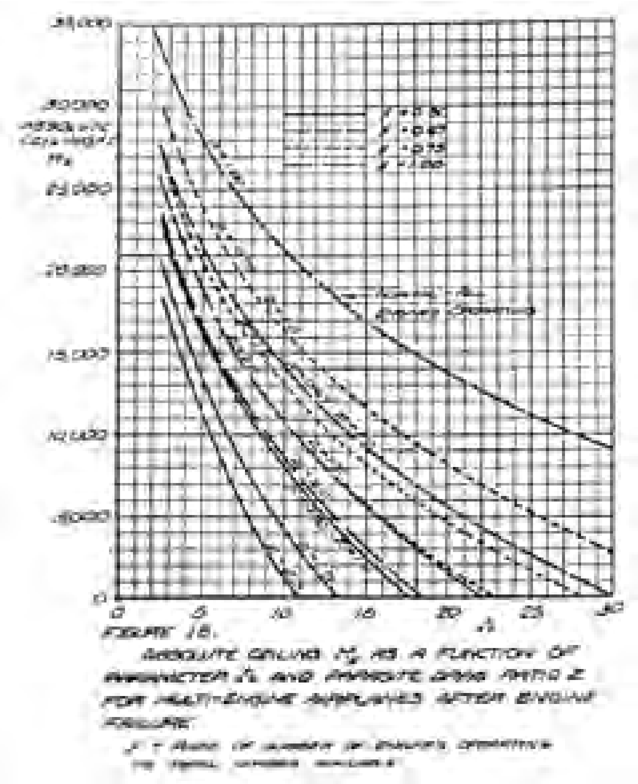
High speed at 8,000 ft.	213 m.p.h.
Cruising speed at 75 percent. power at 14,000 ft.	200 m.p.h.
Maximum rate of climb at 8,000 ft.	1,000 ft./min.
Absolute ceiling	25,400











ON SINGLE ENGINE:

	Full Load:	Less 1,000 lb. Fuel. (Half Normal Fuel Load).
Absolute ceiling	9,500 ft.	10,800 ft.
Maximum rate of climb at 6,000 ft.	70 ft./min.	110 ft./min.

The performance on full power by reverse solution of the charts of Reference I yields the following performance parameters:

Δ_o = 6.55 (see Fig. 18, normal operation).
l_{to} = 10.4 (see Fig. 17, normal operation).
f = 18.0.

PROPELLER DRAG

For the 1,820 cubic inch engine, geared 16:11 at 9,500 ft.,

Q_t/r.p.m.$_p$ = 0.000057 X 1,820 (1 + 0.00002 X 9,500) 1.2/(11/16) = 0.216 (equation 15).

With the three-bladed 11-foot propeller at 113 miles per hour set 23° at 0.75 R. in low pitch and 32° in high pitch,

Q_n = (17,200 X 0.216)/(0.75 X 113 X 11.0^4) = 0.0300 (equation 20).

Then

T_c = –0.0136 (Fig. 8 for two-bladed propellers) for 23° blade angle.
= –0.0076 (Fig. 8 for two-bladed propellers) for 32° blade angle.

And

f_p = 3 X 11.0^2 X 0.0136 = 4.9 for 23° blade angle.
= 3 X 11.0^2 X 0.0076 = 2.8 for 32° blade angle.

Likewise, the drag of the propellers locked and feathered are found from Fig. 9 to be:

	Locked.		Free-Wheeling.		Feathered.
Angle at 0.75 R.	23°	32°	23°	32°	87°
T_c	–0.00225	–0.0020	–0.0072	–0.0035	–0.0013
f_p	8.2	7.3	2.6	1.3	0.5

Additional Drag

For the offset thrust of 9.0 ft. and tail length of 33 ft.,

F = (9.0/33) {2 +(1+9.0/9.0) (4.9/18.0)} = 0.69 at absolute ceiling, and

Δf / f = 0.28 (for efficient design, Fig. 10).

Total Parasite Drag Increase Ratio

z = 1 + (4.9/18) + 0.28 = 1.55 23° at 0.75 R.
= 1 + (2.8/18) + 0.26 = 1.42 32° at 0.75 R.

Similarly for other cases of propeller settings,

	Locked		Free-Wheeling		Feathered
Angle at 0.75 R.	23°	32°	23°	32°	87°
F	0.79	0.76	0.63	0.58	0.56
Δf / f	0.32	0.31	0.26	0.24	0.23
f_p / f	0.46	0.41	0.14	0.07	0.03
z	1.78	1.72	1.40	1.31	1.26

Absolute Ceiling

The absolute ceiling according to Fig. 18, for Δ_o = 6.55 and z = 1.55 is

H_1 = 9,500 ft. with inoperative propeller in low pitch.
H_1 = 10,500 ft. with inoperative propeller in high pitch.

Approximately 1,000 ft. increase in ceiling is obtained by setting the idling propeller in high pitch.

For the other propeller settings, the comparative single-engine ceilings are

	Normal Engine Torque		Locked		Free-Wheeling		Feathered
Angle at 0.75 R.	23°	32°	23°	32°	23°	32°	87°
Single-engine ceiling	9,500	10,500	8,400	8,700	10,500	11,100	11,500
Change in single-engine ceiling	—	+1,000	–1,100	–800	1,000	+1,600	+2,000

By dumping 1,000 lbs. of fuel, which is about half the normal fuel load, Δ_o is increased to

$$\Delta_o = 6.55 \text{ X } [17{,}000/18{,}000]^2 = 5.84 \text{ (–1,000 lb.)}$$

and the normal single-engine ceiling increases 1,400 ft. from 9,500 ft. to

$$H_1 = 10{,}900 \text{ ft. (less 1,000 lb. fuel with propeller at 23°).}$$

OPERATION IN FLIGHT

Control of this airplane when operating on one engine was carefully studied with the result that it can be flown in this condition "hands off."

In one test a take-off was affected during which one engine was cut out after having traversed half of the take-off runway at 4,200 ft. above sea level. The airplane was climbed, flown at 8,300 ft. altitude, 1,000 ft. above the highest point of the transcontinental air line which is 7,300 ft. above sea level, and landed at the next regular airport, 5,100 ft. above sea level. The entire flight was made on the remaining engine without causing any unusual strain on the pilot or airplane. It is notable that on this entire single-engine flight an average terminal-to-terminal speed of approximately 124 miles per hour was made, which practically equals that formerly obtained with tri-motored equipment.

This illustration serves to demonstrate that a high degree of safety in operation of an airplane can be achieved by thorough study and application of the various principles and problems to flight after engine failure.

CONCLUSION

Reliability and safety in the operation of an air line are best served by multi-engine airplanes capable of completing under good control all required flights after engine failure.

In order that the airplane be capable of developing its full potentialities in flight after engine failure it is necessary that comfortable and sufficient rudder control be provided to maintain flight at approximately zero angle of yaw. The control must be sufficient to handle emergencies such as failure just after take-off, etc.

The performance of the airplane after engine failure can be calculated by a rapid parameter and chart method. The principal parameters required are the usual parasite, span and thrust horse-power loadings, l_p, l_s and l_t respectively, and parameter $\Delta = l_s l_t^{4/3} / l_p^{1/3}$, and parasite drag ratio z. Charts for finding z and the performance after engine failure are included.

The influence of the various factors affecting the performance after engine failure is readily seen. It is important that span be increased to a maximum, and that weight and propeller drag be reduced to a minimum. It is further desirable that the propeller offset from the airplane's plane of symmetry be as small as possible. The airplane should fly nearly at the attitude of zero yaw in order that drag remain low. The vertical surface should be of efficient aerodynamic design and have low drag with the rudder deflected.

For construction of the charts, all increase in drag of the airplane has been assumed to be of the parasitic type. This assumption has been found to be consistent with available flight test data. Should it be found desirable when sufficient data are available, all variations could be determined from the general relations, and charts developed therefrom.

Careful study of the various factors involved in flight after engine failure, particularly regarding control and performance, should enable the engineer to design an airplane that has control and performance satisfactory for any reasonable requirement. For multi-engine airplanes, the performance after engine failure should be regarded as a definite problem of design.

Outside of the purely mechanical reason that very large engines are not yet available, the principal justification for the multi-engine airplane is its possibility of continuing safe and satisfactory flight after partial engine failure. The importance of the problems of performance and control involved is, therefore, directly comparable to the basic problems of normal flight.

REFERENCES

1. Oswald, W. Bailey: General Formulas and Charts for the Calculation of Airplane Performance. N.A.C.A. Technical Report No. 408, 1932.

2. Oswald, W. Bailey: Methods of Performance Calculation for Airplanes with Supercharged Engines. Prepared by R. B. Ashley. Air Corps Information Circular No. 679, 1933.

3. Hartman, Edwin P. : Negative Thrust and Torque Characteristics of an Adjustable-Pitch Metal Propeller. N.A.C.A. Technical Report No. 464, 1933.

Mr. Douglas showed a film and added the following commentary:

I consider it a great honor to be here before you tonight on such an outstanding occasion to be able to join with your Society in again honoring the memory of Wilbur Wright. I bring to the members of the Royal Aeronautical Society the hearty and cordial greetings of the Institute of Aeronautical Sciences, our younger but similar association in America. Tonight I am not going to talk directly on the technical paper which I have submitted to you, but rather on the general subject of the developments in the air transport field in America. I not only feel that this might be less trying to many of you, but I am hopeful that the moving picture film I am about to have shown to you will so engross your attention that any defects in my talk will be unnoticed. As might be said in Hollywood, film by Fox, Warner and others—sound effects by Douglas!

The film to be shown is somewhat historical in that we will see at the start the first really successful air liners, namely, the early Fokker and Ford tri-engined planes. To touch but lightly on American air transport history, let me recall to you that it really started in 1926, when our Post Office Department transferred the operation of mail planes to private carriers. The revenues resulting to private concerns from this gave the needed subsidy to permit several capable concerns to embark seriously in the business of carrying passengers on regular schedules. Fokker and Ford made the first real contributions to the passenger equipment of these lines, and the records which these early air liners have to their credit is one to be proud of.

The stock market must be given some credit for the rapid increase in air travel in the early phase, since the money secured by the industry from Wall Street made possible a great expansion of facilities. In this period the development was most rapid in the ground organization of our large air carriers and in the navigational facilities supplied, in the main, by our constructive Department of Commerce. Thus, by, we will say, 1930, and in spite of an already hesitant tone in general business in America, the stage was set for the really rapid technological progress of the past five years.

It seems to me it has been the golden age of aeronautics for us—although not an age for gold in America!—an alert and ambitious military technical personnel played its part in accelerating the use of brains by our designers. A growing competition among our air lines spurred the development of faster and safer air liners. A need on the part of the manufacturers for a broader market for their products and an ability, represented in favorable balance sheets, to seek this with newer and better types, both military and civilian, added to the activity of this field. Early attempts at really fast airplanes, such as the original and successful Lockheed, pointed the way to others as to what might be done. But even beyond all this, and most certainly marking the time as one of real accomplishment, has been the fact that all agencies concerned in and contributing to aviation have been most alert, co-operative and constructive. Our engines have been developing at a pace to permit the airplane designer to raise his sights from time to time. Our instrument and radio people have aided tremendously by furnishing us with the means to fly in bad weather. Propeller makers have been most helpful and, in fact, I can say that the development of the variable pitch propeller to the practical point it has reached today, is probably the most fundamentally important development of this period. Without it many of our present air liners would be impracticable.

As the first modern multi-engine air liner development of this golden age we see on the screen the Boeing. This excellent ship with its high speed, twin instead of three engines, with its ability to continue flight on only one engine, was the first really modern transport. United Air Lines made use of these airplanes and was immediately successful in stimulating air travel. The public was offered flight schedules across our continent that attracted business on the basis of comfort, safety and speed. Other air lines were forced to seek new equipment and to try to still further increase speed and comfort. T. & W.A. embarked on an ambitious development program, enlisting the Douglas Company in the project. The result was our DC-2 type, which most of you have undoubtedly seen flying in and out of Croydon and which you will also see over some of our western mountainous country in the film. We of the Douglas Company are proud of this airplane, but we concede that we were but one agency in its development. From its inception we had the most excellent and wholehearted aid from the air lines. Pilots and maintenance engineers gave of their time and knowledge. The thought was developed that speed in the air was but one function to be solved in the solution of the equation of a successful ship. Speed of inspection and maintenance during fuelling stops, and studies to cut the time required to make replacements, were given great consideration. Safety was a factor which, of course, had to be developed to the utmost possible. Here we were furnished aid by the engine maker, the instrument people, the propeller designers and the wind tunnel and research agencies of our government and of our technical institutions. Flying, as our major air lines do, over great distances across high country in many places and through or over weather of the worst at times and continuation of flight at any point with one of our two engines stopped, was the prime requisite we had always before us in our design. As developed, the DC-2 actually took off after cutting one engine at a point 40 percent of the take-off run from a field about 4,500 ft. high, climbed on up to about 8,500 ft. and completed its flight across the highest part of the air line, some 300 miles, and all with full load.

Comfort was studied with care, and sound engineers developed efficient and practical methods of eliminating the formerly disagreeable and tiring noise of air transport. Heating and ventilation comparable to that found in modern buildings was affected after the aid of related industries was obtained.

The film also gives you a view of other interesting and remarkable achievements in American civil aviation. The Lockheed Electra has been built for and is being used very successfully by some of the air lines needing a smaller capacity airplane than the DC-2. The Lockheed has speed, comfort, and safety that are now demanded by our air travelers. An equally interesting and worthwhile air liner is the Curtiss Sleeper.

You will see on the screen a rather interesting view of the interior showing the comfortable quarters and a fair passenger turning in for a good sleep while en route over the American desert. I have had the surprising experience of seeing a mother settle her small family of two children, aged 2 and 4, for a quiet sleep at 10,000 ft. with the greatest of nonchalance. I felt at that moment that air transportation was really here and was spurred to plan more comfortable and roomy night quarters and facilities for our next developments.

I have spoken only about land transportation. The film will show you flying views of the latest and finest of our flying boats—the Sikorsky and Martin. These have been brought out by those two great concerns working closely with Pan-American Airways. They, in the Ford-Fokker era of land plane transportation, started their ambitious and highly successful foreign air lines with flying boats largely developed from naval boats. These latest pieces of equipment represent the last word as we know it in America for the safe and successful operation of overseas transport. With their four great engines it is difficult to imagine a flight failure due to engine trouble. Seaworthy hulls of large size seem adequate to permit of safe landings under all conditions likely to be met. Comfort is even better served in these commodious hulls than in the smaller bodies of our land planes.

The development of our modern air liner technically from 1926 to 1935 is quite apparent to the eye of the engineer. The development of the traffic in America that has spurred this on is revealed by a few statistics. In 1926 the total of commercial flying was about 4,000,000 plane-miles, a plane-mile being the equivalent of one plane flying one mile, whilst in 1934 this amounted to 41,000,000 plane-miles, a tenfold increase. The number of passengers carried increased from less than 6,000 in 1926 to 460,000 in 1934. Air mail volume increased from slightly over 200,000 lbs. in 1926 to an annual rate of 10,000,000 lbs. in the last quarter of 1934. Air express, an as yet practically undeveloped source of revenue, increased from 3,500 lbs. to 2,000,000 lbs. in this period. The time of flight from coast to coast of America has shown a great betterment in the past nine years, being now 16 hours against about 48 hours in 1926.

That we have achieved a marked gain in safety is proved by the records which show that whilst there was one fatality per 897,000 miles of air line operation of 1926, there was only one death for 6,817,000 miles in 1934. This improvement is most significant when consideration is given to the fact that we are now flying a great deal at night, which was not the case nine years ago and, furthermore, we are now operating in weather which even two years ago we would have regarded as impossible.

The story of the development of modern air liners would not be complete if we did not give credit to the effect of greater technical thought as applied to the flight of the air liner, rather than just to the design of the machines. Formerly the aeronautical engineer had completed his work when the final flight tests of his airplane were completed. From then on it was up to the pilot who, whilst vitally concerned with the proper speed and altitude at which to fly, was not informed completely as to just what factors determined his best course. With the idea of realizing the ultimate in air line operations from the standpoint of speed and economy, many engineers and engineering pilots have necessarily collaborated. Results of their work in theory and flight tests have been made available to all and in such form that they are of great practical value and aid. Dr. Osborn, of our staff, and Pilot Eddie Allen have blazed the path along these lines, and further valuable information is being gathered by others inspired by this work. As can be seen from what I have said, we are only just learning to utilize the machine that we have developed. This brings me to the thought of the future development and I believe we should think for a moment at least of this. What lines do we wish to develop along speed, safety, comfort, size, efficiency? Speed to its ultimate, most certainly. Speed is the first thing that the air line ticket man has for a selling point. There may be many who fly on an air liner for the thrill or the change as compared with surface travel, but these will be fewer each year. There are undoubtedly many, and always will be, who fly because they prefer air travel on the grounds of cleanliness and lack of vibration, but speed must always be the greatest reason for traveling by air. Future development, therefore, must include the development of all the speed possibilities in air travel consistent with other necessary considerations. What the limit is, I cannot say. The aerodynamic engineer will, with some hesitancy, place a limit, but no one can be sure of it. I believe we should be cruising at 300 miles per hour in another ten years. I cannot say how; if I could we would be doing it now. Stratosphere or sub-stratosphere flight, while not to be developed necessarily primarily for speed, may open up avenues of development now not clear to us. Better utilization of slow landing devices will undoubtedly give us our next measurable increase in practical cruising speeds. There is undoubtedly some room for increases in overall aerodynamic efficiency. Possibly some expectations of somewhat better propulsive efficiencies is sound.

What of the development towards more safety? We must admit that whilst great strides have been made in this direction, more can be and must be done. We can reasonably expect to continue to improve the reliability of our engines, our instruments, our radio, our plumbing, and other vital gadgets, but at the moment we can only visualize a general betterment in the airplane itself by such changes as four-engined transport capable of flying on any two engines, rather than twin-engine airplanes. It seems at the moment that the most important advance in safety that we should be concerned in is in better protection against icing up, against electrical storms and in better development of blind flying and blind landing facilities. The icing problem is partially but inadequately solved at the moment. Its better solution may be mechanical and may also be along the line of greater altitude capabilities of the air liner. Hand in hand with this solution for icing goes the possibility that protection against dangerous electrical conditions may lie mainly in high altitude flight. Comfort development certainly is an open field and takes in the problem not only of comfort but sustentation of life at the high altitude that we may soon wish to fly at. Pressure cabins and/or free oxygen in the cabins are both being experimented with today. Our air lines are working diligently with our designers on these points and we look with confidence to some solution soon.

Our future developments from the standpoint of size seem, except in the case of trans-oceanic flying boats, to be controlled by traffic and comfort. Intrinsically, one of the advantages of air transport is its flexibility, so it would seem that size, beyond an efficient point from a structural standpoint and an economic one, is not to be expected to show any great change. Efficiency we must certainly hope to continue to improve, for air transport to be able to show an operating profit without depending on any subsidy. This will be improved by better pay-load percentages, lower first costs and lower maintenance charges. Already in America we find the operating costs of the modern ships quite markedly lower than the older and slower planes. Therefore, we need not fear that even with our speed development we shall cease decreasing costs.

Might I say that it seems to me that aviation is so tied with ground facilities of such a world-wide and international scope that the most important developments will come when we find means to more completely co-operate as nations. Also it appears that so much time and money is wasted because of the duplication of the same experiments in different countries, that something constructive should be considered at the present time. From this point of view military aviation seems to be a deterrent factor on more rapid and less costly progress. Why, however, can we not hope to take certain general aviation problems that do not involve anything intrinsically military, and allot them among the major nations: certain problems to one nation with more experience on a certain phase and certain problems to another? A complete interchange of the knowledge resulting would then follow. Surely this is possible, and if it is, the aviation industry, instead of being a potential breeder of war with its increasingly dangerous weapon, might become an agent in spreading understanding. I offer these last thoughts humbly and in the belief that engineers and scientists such as form the membership of the Royal Aeronautical Society, are able most fully to grasp the possibilities. A glorious future lies before aviation. Its development has only just commenced. May I hope that those interested in it here will believe that we in America extend to you our fellowship and promise of full co-operation for the common good.

The PRESIDENT: In asking Mr. Fairey to propose a vote of thanks to Mr. Douglas, he said that Mr. Douglas was indeed a remarkable man because he had given them three kinds of entertainment. First of all there was the scientific paper which they were all going to read in bed; then there was the film which they really ought to see again; and finally, there were the comments which Mr. Douglas had made in giving that film. In his quiet way Mr. Douglas had said many things in his talk which must not be lost, and therefore he was going to ask him to allow the Society to print his talk in the Journal separately from the lecture, which had already been printed, because personally he felt that was very necessary.

Mr. C. R. FAIREY (Past-President): They had listened to a very vivid, practical and interesting talk, but they all had in their hands the print of an equally interesting and even more important lecture, since in it Mr. Douglas proved that high speed in aviation could be obtained without any loss of economy, of safety, or of comfort, as the aircraft on the world's airways had already proved. It was to be hoped that in the hands of our designers this lecture would have some effects on the future of British air transport. If it led to the fact that we never heard again of that strange doctrine under the headline of "the fallacy of speed," it would have done something, although he was never quite clear himself as to whether the protagonists of that amazing doctrine were out to prove that aeroplanes ought not to go fast or to apologize because they did not do so. After all the basic ratio for aeronautical engineers, L/D, had a commercial as well as a technical significance. L was the income and D represented the outgoings, and anything we could do to improve L/D was all to the good of commercial aviation, and Mr. Douglas had certainly done that.

Everybody was grateful to Mr. Douglas for having traveled the long distance he had to deliver this lecture and also for the trouble he had taken in preparing it, and further, for giving them the opportunity of meeting him personally.

The vote of thanks was carried with hearty enthusiasm.

Mr. DOUGLAS, in acknowledging the vote of thanks, said the pleasure he had in coming to England to give this lecture had been enhanced by the great kindness shown to him and the indication that had been given that those who had listened to him had, at any rate, enjoyed the lecture to some extent.

Document 3-25(a-e)

(a) George W. Lewis, Director of Aeronautical Research, NACA, to Mr. R. D. Kelly, Research Supervisor, United Air Lines Transport Corporation, 5936 South Cicero Avenue, Chicago, IL, 3 Sept. 1937, copy in Research Authorization File No. 565, NACA Langley Historical Archives, Hampton, VA.

(b) John W. Crowley, Jr., Senior Aeronautical Engineer, to Engineer-in-Charge [Henry J. E. Reid], Langley Memorial Aeronautical Laboratory, Langley Field, VA, 5 Oct. 1937, copy in RA file 565, LHA.

(c) R. D. Kelly, Research Supervisor, United Air Lines Transport Corporation, Field Headquarters, 5936 South Cicero Ave., Chicago, IL, 6 Oct. 1937, to Dr. George W. Lewis, Director of Aeronautical Research, National Advisory Committee for Aeronautics, Navy Building, Washington, DC, 6 Oct. 1937, copy in RA file 565, LHA.

(d) Melvin N. Gough, Senior Test Pilot, Langley Memorial Aeronautical Laboratory, Langley Field, VA, to Engineer-in-Charge, "Suggestion that conference on 'Stalling' be held with interested personnel outside the Laboratory," 2 March 1938, copy in RA file 565, LHA.

(e) Edward P. Warner, New York City, to Dr. George W. Lewis, NACA, 6 Apr. 1938, copy in RA file 565, LHA.

In September 1937, NACA Langley performed stalling and icing studies with a DC-3 Mainliner passenger transport belonging to United Airlines. In order to warn the pilot of an approaching stall, the NACA engineers installed sharp leading edges on the section of the wing between the engine and fuselage. These sharp edges disturbed the airflow enough to cause a tail buffeting that could be felt by the pilot in his control column. When the pilot felt this buffeting, he knew that his airplane was approaching a stall and needed pilot correction.

Document 3-25(a), George W. Lewis, Director of Aeronautical Research, NACA, to Mr. R. D. Kelly, Research Supervisor, United Air Lines Transport Corporation, 5936 South Cicero Avenue, Chicago, IL, 3 Sept. 1937.

September 3, 1937.

Mr. R.D. Kelly,
Research Supervisor,
United Air Lines Transport Corporation,
5936 South Cicero Avenue,
Chicago, Illinois.

Dear Mr. Kelly:

Thank you for your letter of August 30. I am pleased that the DC-3 airplane will be available for the investigation of stalling characteristics sometimes near the end of September. The small instrument which the Committee has developed for indicating the stall is quite simple, and there will be no difficulty about making another for permanent installation on the DC-3 if you so desire.

In discussing the stalling characteristics with Mr. West, I mentioned the possibility of modifying the contour of the leading edge of the wing near the tip to obtain better stalling characteristics of the wing in front of the aileron. He forwarded to me a blueprint showing the planform and section of the tip portion of the wing of the DC-3. It was thought that we would suggest a section and size of spoiler for the leading edge for installation by you for the investigation. On further discussion of the question with members of our staff, it appears that it would be more desirable for the Committee to make such an installation, as in their opinion one or two positions should be tried for the modification of the wing section.

We have conducted an investigation for the Army Air Corps on a Consolidated two-seated fighter for the purpose of modifying the wing so as to indicate to the pilot the approach of the stall condition. In this particular case it was desirable to place a small sharp leading edge section on the wing near the center of the wing.

Sincerely yours,

G. W. Lewis
Director of Aeronautical Research.

Document 3-25(b), John W. Crowley, Jr., Senior Aeronautical Engineer, to Engineer-in-Charge [Henry J. E. Reid], Langley Memorial Aeronautical Laboratory, Langley Field, VA, 5 Oct. 1937.

Langley Field, Va.
October 5, 1937

MEMORANDUM For Engineer-in-Charge.
Subject: Cooperative tests with the United Airlines Company on the Douglas DC-3 airplane.

1. The tests that were conducted in cooperation with the United Airlines Company on their 21-passenger DC-3 airplane during the week of September 27 have been completed. The tests were primarily for the purpose of investigating the stalling characteristics of the machine and determining methods for improving the same so as to increase the safety of low-speed flight such as is encountered in take-off, landing, and approaches to landing. In addition, tests were made in which ice formations were simulated to determine the effect of ice formation such as is encountered in service on the ability of the airplane to fly and handle safely. Furthermore, tests were made of certain of the general flying and handling and stability characteristics of the airplane and also of the landing and take-off characteristics for the purpose of increasing our fund of information on these characteristics that we are currently accumulating for all modern airplanes for use in establishing what are desirable flying and handling qualities.

2. We found that the stalling characteristics, particularly in the condition in which the airplane is generally flown, i.e., with power on, was definitely undesirable and likely to be dangerous. The airplane stalled violently and with no warning to the pilot of the approach of the stall. The tip portion of the wing at the position of the ailerons stalled first so that lateral control was entirely lost at the final stall. Since in a machine of this type there is no necessity for stalling, it was decided that the most suitable solution would be to provide the pilot with a definite warning that he was approaching the stall so that he could avoid same. To accomplish this we installed sharp leading edges on the section of the wing between the engines and the fuselage. Those were so adjusted that at a few miles an hour before the stall was reached the flow on this part of the wing became disturbed and in passing over the tail surfaces caused a buffeting of the tail that could be felt by the pilot in the control column. These were very successful in definitely warning the pilot of the approach of the stall and were enthusiastically received by the United Airlines personnel. Tests showed that the sharp leading edges had no deleterious effect on the performance of the airplane in level or cruising flight nor did they appreciably increase the stalling speed flight. During the course of the other tests of this machine the development

of a stall warning indicator that mounted on the wing and indicated the approach of the stall to the pilot by means of light on the dash was carried out. While not perfected for this machine the development was carried sufficiently far to show that the principle involved was entirely practicable and that such a device would definitely increase the safety of flight on this machine. The pilots appreciated, in particular, the knowledge obtained that the stalling speed in a turn was higher than in steady flight. The United Airlines personnel, aside from the use of the sharp leading edges, expressed the opinion that they had learned more of the stalling characteristics of the DC-3 from these few cooperative experiments than in all their other flying of this machine.

3. In the tests to establish the effects of ice formation, pieces of sponge rubber were cemented to the forward part of the wing in positions that the United Airlines personnel, from their experience, were able to say was representative of that obtained in actual operation. The airplane was then flown with these in place and the performance of the airplane in steady flight and climb and the stalling speed were measured. In general, it was found that the speed in level flight and the stalling speed were not affected appreciably, but that the climbing performance was noticeably reduced. This was the first direct information obtained upon the effect of ice formation on the performance of the DC-3 and the United Airlines personnel felt that the information was of great value to them.

4. Our measurements of the general flying and handling and stability characteristics in conjunction with the observations of the United Airlines pilots we believe to be very valuable in that we now have a better appreciation of what the transport operators expect and are willing to accept in these respects. It was found definitely that the longitudinal stability was poor.

5. The landing and take-off characteristics measured constitute a valuable addition to our knowledge of these characteristics for large machines.

6. In summary, I believe the information that we received as a result of our tests of the aerodynamic characteristics of this the latest and largest land transport machine now in use, together with the direct knowledge that we received through discussion with the pilots of the problems of transport operation has been invaluable. On the other hand, according to the statements made by the United Airlines personnel, they were highly gratified with the information they have received in regard to a method of providing stall warning, the increased knowledge of the character of the stall, and the information of the effect of ice information on the performance of the airplane.

John W. Crowley Jr.,
Senior Aeronautical Engineer.

Document 3-25(c), R. D. Kelly, Research Supervisor, United Air Lines Transport Corporation, Field Headquarters, 5936 South Cicero Ave., Chicago, IL, 6 Oct. 1937.

UNITED AIRLINES TRANSPORT CORPORATION

Field Headquarters
5936 South Cicero Ave.
Chicago, Ill.

October 6, 1937

Dr. George W. Lewis
Director of Aeronautical Research
National Advisory Committee for Aeronautics
Navy Building
Washington, D.C.

Dear Dr. Lewis:

As you know, we took one of our DC-3 airplanes to Langley Field September 26th to be used there in the course of special stalling tests as per previous arrangements. We remained there all of last week and accumulated considerable data for your own organization and for ourselves. From our standpoint, this trip was extremely successful and we hope that the N.A.C.A. found the investigation to be beneficial to them also.

We were surely accorded every courtesy at Langley Field and the cooperation could not have been better. We feel that the amount of work accomplished was all that anyone could have asked for, particularly since it was the first time that our groups had worked together. We hope that such cooperative investigations may be continued in the future inasmuch as the information obtained should be of benefit to the entire commercial aircraft industry as well as to ourselves.

We have not completed our report of these tests as yet, but we plan to make immediate use of the information obtained by passing on some of the highlights to our pilot personnel at once. We know this information will be very interesting to them and that it will give them a better knowledge of the characteristics of this airplane. Therefore, they will be able to take advantage of those characteristics which were found to be particularly good and to avoid those which were shown to be somewhat more critical.

As you know, ice presents one of our greatest existing problems. Therefore, near the end of the week your people installed small blocks of sponge rubber upon the leading edges of the airplane in an effort to simulate ice deposits. This installation was made at our suggestion and the blocks were located as per our recommendation. By making flights with these in place, we incurred certain losses in performance which coincided with experiences which had been reported to us by our pilot personnel when they had encountered certain types of icing conditions. Had time permitted, we might have extended these tests still further, but inasmuch as the different types of ice formation are so varied, it seemed best to not carry the simulated tests further at this time. While these simulated ice tests were all of a very preliminary nature, they gave all of us a better understanding as to the good which might be accomplished by more thorough simulated tests.

Before we left Langley we had discussed the following arrangement with your engineers, which we hope you will see fit to approve. It was agreed that we should be able to furnish the N.A.C.A. with fairly accurate information concerning the various types of ice formations which are encountered; this information to be based upon actual questionnaires which will be furnished to our pilot personnel and upon actual observations which will be made by our Engineering and Research Organization. When this ice formation information is received, the N.A.C.A. can simulate it by means of plaster paris or any other satisfactory method and apply it to airfoils which can be tested in your wind tunnels. Thus, even the worst conditions can be carried throughout the complete range of flight analysis.

The N.A.C.A. stall indicator appears to us to have very great possibilities and we would like to encourage its development. It seems to us that this unit is about the only feasible method which can be expected to provide a sure warning ahead of the actual stalling condition, regardless of how that condition may be influenced by ice, load, accelerations, etc. While we did not consider the unit to be developed sufficiently that we should retain it on the airplane for service test, we are hopeful that it can be brought up to this point very shortly. We will keep the mounting attachment on our airplane in the hope that you will wish to forward the improved unit to us for further testing. We believe that these improvements can be made through the medium of wind tunnel tests.

We were very much interested in the moving pictures which were shown us concerning the behavior of tufts on the top surface of the wing during actual flight stalls. Some of these pictures were taken from our own airplane and if it is in line with your policy, we would like to purchase a print of these pictures just as soon as possible in order that we may exhibit them to our pilots and engineering personnel for educational purposes.

Again thanking you for this opportunity of working with the Committee and assuring you of our appreciation for the courtesies extended to us while we were at Langley Field, we are

Very truly yours,

R. D. Kelly
Research Supervisor

Document 3-25(d), Melvin N. Gough, Senior Test Pilot, Langley Memorial Aeronautical Laboratory, Langley Field, VA, to Engineer-in-Charge, "Suggestion that conference on 'Stalling' be held with interested personnel outside the Laboratory," 2 March 1938.

Langley Field, Va.,
March 2, 1938.

MEMORANDUM For Engineer-in-Charge.

Subject: Suggestion that conference on "Stalling" be held with interested personnel outside the Laboratory.

1. Much has been written and said by Laboratory personnel in the past 2 years regarding stalling and stall control, and its relative importance. Little has been heard from outside. Either they are awed or disagree. In talking with service personnel and others outside, I have heard both adverse and constructive criticism regarding our relative evaluation of the problem.

2. Feeling that the N.A.C.A. must seek out and be aware of the stand of others on the problem, I suggest that our future activities on the subject might best be guided by the understanding and changed viewpoints which might be gained as a result of a "stalling conference" held with members of the N.A.C.A. staff, representatives of the Army, Navy, Department of Commerce, and others interested.

3. It is suggested that such a conference be arranged.

Melvin N. Gough,
Senior Airplane Test Pilot

Document 3-25(e), Edward P. Warner, New York City, to Dr. George W. Lewis, NACA, 6 Apr. 1938.

EDWARD P. WARNER
New York City
April 6, 1938
Dr. George W. Lewis
National Advisory Committee for Aeronautics
Navy Building
Washington, D.C.

Dear George:

I have returned the copy of the DC-3 stall report that you were good enough to let me borrow. I've dictated some notes suggested by reading it; and if they're somewhat dogmatic in tone, that's to provoke an argument. Incidentally, I may be revealing an embarrassing ignorance of recent developments in the field; but I'll risk that.

The use of a sharpened leading edge on the portion of the wing inside the nacelle is apparently capable of improving the stalling characteristics, but that remedy always seems to me a very unsatisfactory one. Its object is to knock the top off the lift curve on the inner portion of the wing, so producing an early stall in that region, and giving the pilot warning by buffeting; but in so doing it reduces the maximum total lift on the wing, and increases the minimum speed, and it really does nothing fundamental about the stalling characteristics of the tip. What we should work for is not only a delayed stall at the tips, but a more gradual stall when it appears. The use of the sharpened leading edge in flattening off the lift curve inside the nacelle actually produces the shape of curve in that region that we would like to have out near the tips. The sharpened leading edge is then less a means of really improving the stall than a mere stall warning device.

The great virtue in the change of the leading edge of course is that it can be applied on existing ships. In all future development, the stall should be thoroughly investigated during the tests of the prototype, and if it is unsatisfactory, something more fundamental than a leading-edge change on the inner part of the wing should be required. Changes in mean camber near the tips are perhaps the most promising of the more fundamental modifications.

I observed with amazement the statement on page 10 of the report that with power off "the airplane is laterally controllable up to speeds 10 miles per hour above the minimum beyond the stall". If this is literally true, it suggests an immunity of the wing-tips from sudden stalling which is quite at variance with the indications gained in flight with power on; yet the introduction of power is ordinarily expected to have little effect beyond the scouring of the wing in the neighborhood of the

nacelle, and the delay of stalling on the inner portion. I ordinarily expect, in other words, that the elimination of power will itself provide a stall-warning, rather than a stall-preventive, device. In this case it appears to be definitely stall preventive. That point would deserve further examination.

One factor in this difference of stall with the introduction of power is no doubt the change of the span-loading curve with the introduction of the slip-stream across the wing. The effect on span loading of bringing in power is similar to that of increasing the taper ratio, and that may be responsible for the sharpness of the tip stall. Even that doesn't seem an entirely satisfactory explanation; though I should be interested in getting the comments of some of your people, on the magnitude of the effect. If that's an important item, it's likely to be inherently worse on a four-engined than on a two-engined ship.

I was much interested in the longitudinal stability measurements on the ship, as recounted on pages 15 to 18 of the report, but the conclusions seem to me really to repeat the original premise, and therefore to be without relation to the special problem of the ship's stalling behavior. It is specifically stated that the tests were made "with the most rearward position of the center of gravity recommended according to the Douglas loading chart". The conclusion is then reached that the longitudinal stability is inadequate, being almost exactly neutral; but it is that very fact that determines where the most rearward position of the c.g. shall be located. It is a pity that some of these tests couldn't be repeated with a more normal center of gravity location. While I agree with all that is said about the undesirable effects of too small a measure of static longitudinal stability, I don't see any way of preventing that quantity from diminishing steadily towards zero as the c.g. is moved to the rear on any ship. We have then in any case to determine how small a degree of stability is acceptable, and that in turn determines how far back the c.g. may go. If, as your report suggests, the static stability characteristic is in fact a major difficulty here (on which point I am a little skeptical, as I understand that there's complaint of the stall even when the load distribution puts the c.g. well forward) the conclusion is not that there is an inherent fault in the airplane, but that the loading charts should be revised so that the c.g. will never be allowed to go as far back as its present rearmost position. I presume that that is what the report was really intended to imply.

In the matter of a stall-warning indicator I agree entirely with Mr. Jacobs (see page 25 of report) that the instrument as set up for these tests was probably more of an air-speed than an angle-of-attack indicator. Nevertheless the idea seems worth following up, for this general type of indicator has the advantage of seeming the only type that will give warning of the advent of a stall directly caused by local icing. If it were not for icing, I should prefer an indicator depending directly on measurement of the angle of attack, like the old Savage Bramson device, but when the ice forms on the wing it changes the conditions under which, and the angle of attack at which, the stall will appear. On the other hand the direct measurement of angle of attack has the advantage of indicating an approach to the condition of stall, whereas the boundary-layer device seems to be capable of indicating only at the time when a stall is actually developing.

Were it not for the special problem of icing, however, I should prefer a stall warning by mild buffeting to any sort of instrumental warning; and if the instrument is introduced it should be primarily to meet the special case of the modification of the stall condition by icing. Preferable to any sort of warning or indication, whether instrumental or inherent in the airplane, is a design of the wing which will not only retard the stall near the tip slightly, as compared with that at the root, but will also ensure as gradual a development of tip stall as possible, corresponding to a very flat top on the section-lift curve.

I was very much interested in the work that had been done on ice deposits as simulated by blocks attached to the wing; and especially in the conclusion that further work on this point could profitably be done in the wind tunnel. I am in full accord on that point.

In short, the stalling problem seems to present six questions.

1. Is it desirable to secure increased warning of the stall on the DC-3 by inducing buffeting with a sharpened leading edge on the wing roots?
2. Assuming that bad stall characteristics might develop on the DC-4, what line of experiment should be planned in advance for use in looking for a solution?
3. Specifically, can modifications of the section or angle of the outer wing be made which would smooth out the tip stall and render it more gradual, in case the flight test of the DC-4 should show that any improvement in characteristics is needed?
4. Is a deliberately induced buffeting a sufficient warning of approach to the stall (assuming that the entry into the stall cannot be made so gradual, and the lateral control be so well maintained, as to obviate the need for any warning), or should it be supplemented with an instrumental indication?
5. In connection with the effect of icing, is there anything to be gained by a change from the type of wing section now in general use to one which may have somewhat lower efficiency, but will be less sensitive to the effect of minor changes of form near its leading edge?
6. If buffeting gives a satisfactory warning of the approach of the stall when there is no ice, may the introduction of ice on the outer portions of the wings nevertheless change the stalling characteristic so that buffeting will become inadequate as a warning, and should be supplemented by an indicator acting directly in the region of danger near the tip?

Sincerely,
Edward P. Warner

Document 3-26(a-b)

(a) Excerpts from Lewis A. Rodert and Alun R. Jones, "Profile Drag Investigations of an Airplane Wing Equipped with Rubber Inflatable De-Icer," NACA Advance Confidential Report, Dec. 1939, copy in LaRC Technical Library.

(b) Excerpts from George W. Gray, Chapter 14, "Heat Against Ice," in *Frontiers of Flight: The Story of NACA Research* (New York: Alfred A. Knopf, 1948), pp. 307-329.

Perhaps an even more dangerous problem for airline operations was icing, a problem to which the NACA would commit considerable time and energy from the mid-1930s on. As historian Glenn Bugos has explained, icing was a critical systems-wide problem during this period of time:

> Ice caused aircraft to crash by adding weight and preventing the pilot from climbing above the icing clouds, so that the aircraft gradually lost altitude and slammed into the ground. . . . [I]ce accreted along the wing and tail leading edges disturbing lift and adding drag. Ice clogged the interstices of rudders and ailerons, preventing control and inducing buffeting. It changed the aerodynamic profile of the propeller, causing it to vibrate and exert less thrust per horsepower. It coated windshields, so the pilot flew blind. Ice made antenna wires oscillate and snap, and generated static that rendered useless most radio communication and navigation. It distorted pitot shapes, so that pilots got erroneous airspeed readings. And it clogged carburetors, suffocating the engine. Frequently, the pilot lost each of these systems—engine, wings, control surfaces, indicators, radio, sight—within minutes (Bugos, "Lewis Rodert, Epistomological Liaison and Thermal De-Icing at Ames," in *From Engineering Science to Big Science: The NACA and NASA Collier Trophy Research Project Winners*, ed. Pamela E. Mack (Washington, DC: NASA SP-4219, 1998), p. 32).

Although the problem of aircraft icing would never totally go away, the comprehensive study of the icing problem begun by the NACA, and continued by NASA, along with its development of different de-icing systems represents a major contribution of government-sponsored research to modern aviation. Its importance was recognized throughout the industry, for the first time in 1946 when Lewis Rodert of the NACA won the Collier Trophy for developing a thermal ice-prevention system.

The documents below concern the genesis of the NACA's icing research program, as well as the NACA's research into the stall problem that was experienced by the Douglas DC-3. In order to simulate the effects of ice formation on the DC-3's performance, the NACA engineers cemented pieces of sponge rubber to the forward parts of the wings, where ice was thought to form most often, and then measured the resulting changes in the plane's climb, cruise, and stalling speeds. In a December 1939 NACA Advanced Confidential Report on these tests, entitled "Profile Drag Investigations of an Airplane Wing Equipped with Rubber Inflatable De-Icer," Langley engineers Alun R. Jones and Lewis A. Rodert (the future Collier Trophy winner for his anti-icing work) emphasized just how big of an impact even a thin layer of ice could have on the lift, drag, and stalling of an aircraft.

It is not surprising to find, in the last document in this string, that aviation dynamo Edward P. Warner was very interested in both the stall and icing problems and how the two related, as he was a special consultant to Douglas Aircraft Company with particular concerns about the operation of large transports.

Although many aircraft systems were involved in the icing problem, the NACA's icing research definitely concerned aerodynamics. As ice built up on a wing, it destroyed lift and increased drag. Control and stability suffered as rudders and ailerons froze up. Propellers vibrated intensely and lost their thrusting power. Simply put, aircraft could crash—and many of them did.

The second document below is an excerpt from aviation writer George W. Gray's 1948 homage to NACA's wartime contributions, *Frontiers of Flight*. Gray examined the problem of aircraft icing and the various solutions that the NACA offered to alleviate it in the period from 1938 to 1948. The excerpt below does not include three (nonconsecutive) sections near the end of his chapter on icing. The first of these omitted sections concerned "Ice and the Engine" (pp. 320-323); the second, "Icing Problems of Jet Propulsion" (pp. 323-324); and the third, "Problems of Cooling and Ventilation" (pp. 328-29).

Figure 1.- Lockheed 12-A airplane equipped with inflatable de-icers.

Document 3-26(a), Excerpts from Lewis A. Rodert and Alun R. Jones, "Profile Drag Investigations of an Airplane Wing Equipped with Rubber Inflatable De-Icer," NACA Advance Confidential Report, Dec. 1939.

PROFILE-DRAG INVESTIGATION OF AN AIRPLANE WING

EQUIPPED WITH RUBBER INFLATABLE DE-ICER

By Lewis A. Rodert and Alun R. Jones

SUMMARY

The National Advisory Committee for Aeronautics has made profile-drag measurements in flight of a wing which was equipped with a rubber inflatable de-icer and to which various simulated ice formations were attached. Tuft observations at the stalling speed of the wing with the various drag conditions were made in order to determine the influence on the maximum lift coefficient.

The de-icer installation caused an increase of from 10 to 20 percent in the profile drag of the plain wing and reduced C_{Lmax} about 6 percent. Simulated ice, when confined to the leading-edge region of the de-icer, had no measurable influence upon the profile drag at the cruising speed. This ice condition, however, reduced the value of C_{Lmax} to about three-fourths that of the plain wing.

Simulated ice in the form of a ridge along the upper and lower de-icer cap-strips increased the profile drag by about 560 percent at cruising speed. This condition reduced the C_{Lmax} to approximately one-half that of the plain wing value.

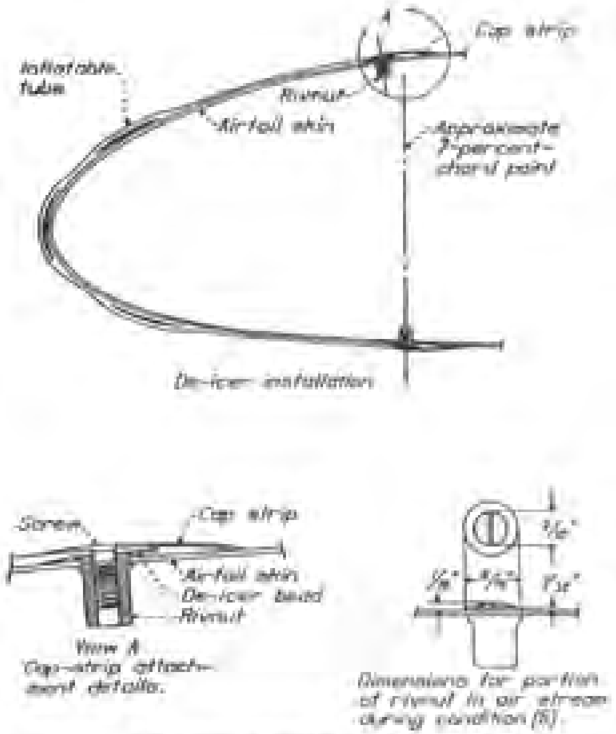


Figure 2. Rubber inflatable deicer details.

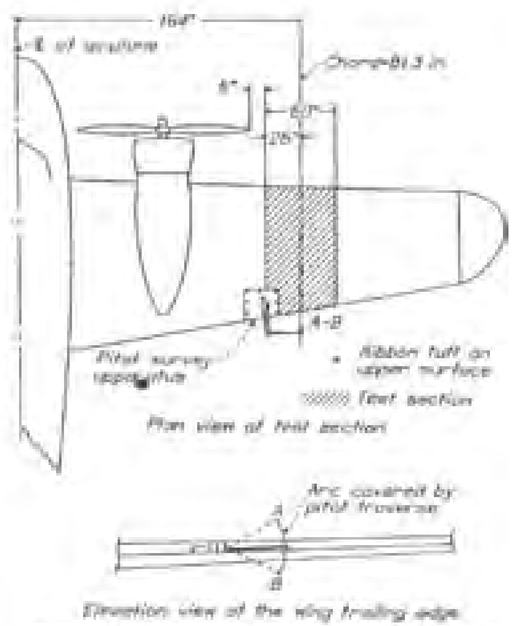


Figure 3. Location of test section and traverse apparatus.

Figure 4.- Wake-survey apparatus installed on the trailing edge of the wing.

Figure 5.- Rubber inflatable de-icer installation. Drag condition (1).

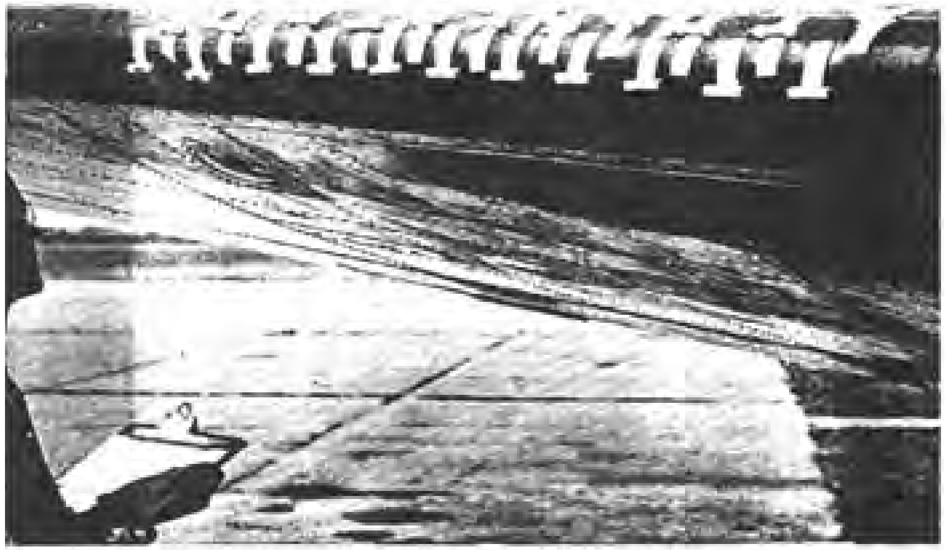


Figure 6.- Rubber de-icer with simulated ice formations on leading edge. Drag condition (2).

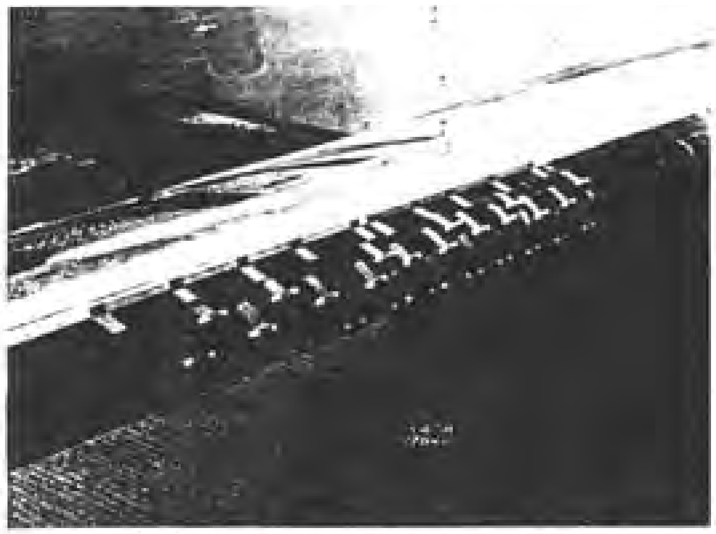
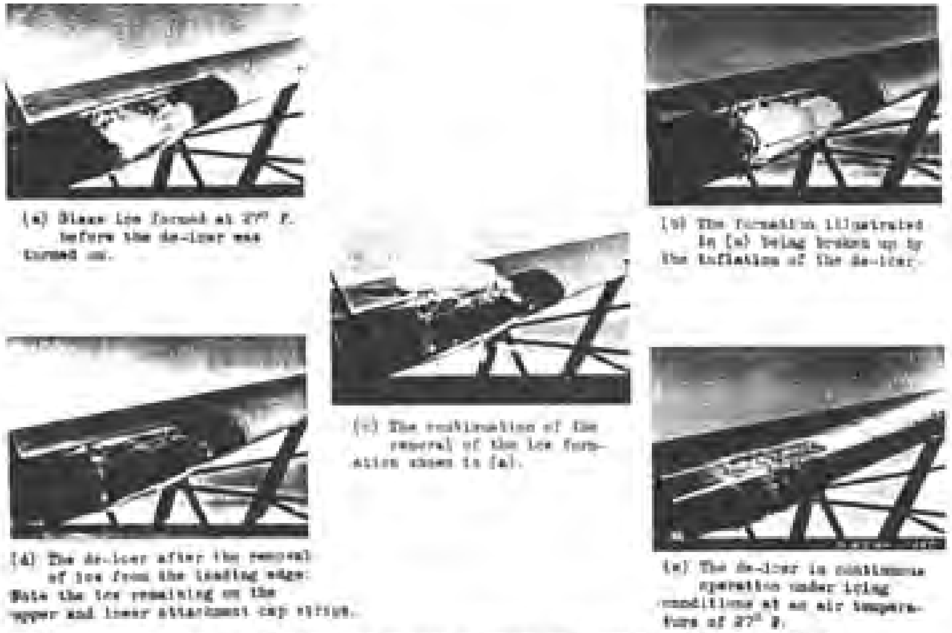


Figure 7.- Rubber de-icer with simulated ice formations on leading edge and with upper and lower cap strips. Drag condition (3).

Figure 8.- Rubber de-icer with simulated ice only on cap strips. Drag condition (4).

Figure 9.- Rubber de-icer removed and rivnuts faired. Drag condition (6). The unfaired rivnuts can be seen at the left of the test section.



(a) Glaze ice formed at 27° F. before the de-icer was turned on.

(b) The formation illustrated in (a) being broken up by the inflation of the de-icer.

(c) The continuation of the removal of the ice formation shown in (a).

(d) The de-icer after the removal of ice from the leading edge. Note the ice remaining on the upper and lower attachment cap strips.

(e) The de-icer in continuous operation under icing conditions at an air temperature of 27° F.

Figure 10.– The Goodrich de-icer in operation during a test at the N.A.C.A.

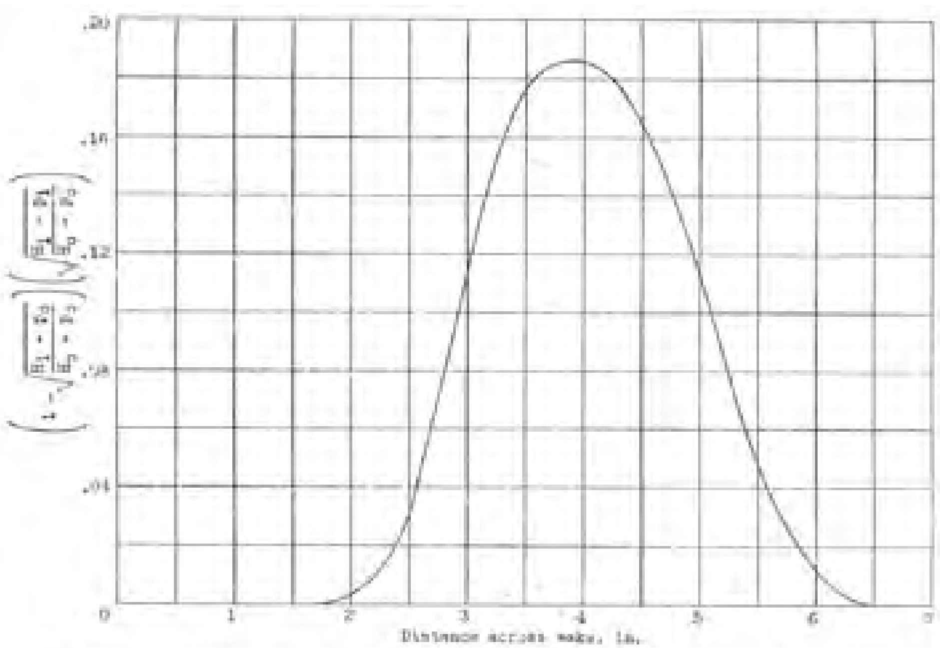


Figure 11.– Wake-survey curve for Lockheed 12-A wing with rubber de-icer and simulated ice on leading edge. Air speed, 175 miles per hour; chord at survey head, 81.3 inches.

INTRODUCTION

The large number of airplanes equipped with rubber inflatable de-icers makes this device of considerable importance in aircraft operation and performance. Inquires have frequently been made relative to the aerodynamic effect of the de-icer installation during flight in normal fair weather, and during icing conditions. A partial answer to these questions has been provided by previous investigations as reported in references 1, 2, and 3. In order to obtain more complete information, the National Advisory Committee for Aeronautics has made flight measurements of the profile drag of an all-metal wing as influenced, first, by a rubber inflatable de-icer installation and, second, by simulated ice formations on the wing loading-edge region. Observations were also made on the approximate effect that the ice simulations had upon the maximum lift coefficient.

The pitot traverse method of measuring profile drag presented a relatively simple and accurate means of obtaining the required drag information (reference 4). Information regarding the lift coefficient was obtained by making tuft observations at the stalling speed of the test section.

APPARATUS AND METHOD

The flight tests were conducted on a Lockheed 12-A airplane which was equipped with inflatable wing de-icers (fig. 1). The details of the de-icer installation are shown in figure 2. The Lockheed wing is an all-metal structure with brazier-head rivets and lap-jointed skin. The traverse apparatus was located on the airplane in the position shown in figures 3 and 4. The necessary mechanism was provided for the movement of the traverse head through the arc AB (fig. 3) in measured increments during flight. To obtain a profile-drag measurement, a survey was made of the stream pressures between the two points A and B which were, respectively, above and below the wing wake. A swiveling air-speed head, shown in figure 1, was placed at the end of a boom projecting from the front of the fuselage, the calibration of which was established by means of a suspended pitot-static head. Pressure recordings of the following were made: (1) dynamic pressure of undisturbed stream; (2) dynamic pressure in the wake at each position of the traverse head; and (3) difference between the static pressures at these two points.

Profile-drag measurements and stalling-speed observations were obtained with the test section in the following conditions: (1) with de-icer, no ice (fig. 5); (2) with de-icer and formations on the leading edge (fig. 6); (3) with de-icer and formations on the leading edge and the de-icer attachment cap-strips (fig. 7); (4) with de-icer and with formations on the cap-strips only (fig. 8); (5) plain wing without the de-icer and with the de-icer attachment rivets unfaired; and (6) with the de-icer removed and the rivets faired (fig. 9). The unfaired rivets of condition (5) can also be seen in figure 9.

Condition (1) corresponds to the state of the wing in normal operation when no ice is present. The simulations for conditions (2), (3), and (4) were obtained by fastening wooden blocks to the wing. The leading-edge blocks were 3/8 inch by 3/8 inch by 2 inches. The cap-strip pieces were triangular in section with ½ inch base and height, and ran continuously along the span of the test section. The simulated ice formations extended 50 inches along the leading edge of the wing. (See fig. 3).

Attention is invited to the fact that the de-icer apparatus on the Lockheed 12-A airplane was installed within the past year and, therefore, is similar in surface smoothness to other de-icers currently in use and installed prior to January 1939. It is understood that various modifications of the de-icer are now under consideration for use during the coming winter (1939-1940). These modifications vary in degree from rather minor changes of existing equipment to completely new equipment for installation on new airplanes and are intended to reduce or eliminate the discontinuity in the wing contour at the rear edge of the de-icer. It is also understood that an improvement in surfacing the de-icer is now possible. These modifications are intended to reduce the profile drag of the plain wing and to prevent residual ice accretions on the leading-edge and the cap-strip regions. Inasmuch as these modifications are still in the development stage, no data are available regarding their effectiveness in eliminating ice accretions.

The determination of the thickness, shape, and location of the simulated ice was based on unpublished reports of flight observations on a rubber de-icer in action and on replies to a questionnaire circulated to transport airline operators. According to these sources, two different types of failure to remove ice are common. Ice may remain in narrow ridges along the leading edge of the de-icer when in operation, or an accumulation of ice may gather on the upper and lower cap-strips, or both types of failure to remove ice may occur simultaneously. Pictures of such ice formations, which were taken by the N.A.C.A. during a recent flight investigation, are shown in figure 10. For condition (5), the de-icer was removed and the rivets (to which the cap-strips are attached) were not faired. This situation corresponds to the normal condition of the wing during the summer. The rivets were then faired (condition (6)) in order to approach as closely as possible the profile drag of the plain Lockheed wing.

Tests were made with each of the six drag conditions at speeds of 125 and 175 miles per hour. Additional tests at 140 and 150 miles per hour were made with drag condition (1). The corresponding Reynolds number range was from 6,500,000 to 9,000,000.

An approximation of the stalling speed of the test section for the various drag conditions was obtained from tuft observations. Ribbon tufts were fastened to the upper wing surface at the points indicated in figure 3. The air speed was decreased slowly until the ribbons indicated that the region of the test section had stalled. The speed at this instant was noted and was used to calculate C_{Lmax} for each condition. These tests were all made with the flaps up and the landing gear retracted.

TABLE I.- DRAG OF WING AS INFLUENCED BY RUBBER INFLATABLE DE-ICER AND SIMULATED ICE FORMATIONS

Drag condition	Description of condition	Test section profile-drag coefficient, C_{D_o}	Airplane lift coefficient, C_L	Reynolds Number	1 Profile drag in percent of profile drag of plain wing	2 Profile drag in percent of profile drag of wing with de-icer
(1)	De-icer, no ice	0.0105 .0106 .0112 .0116 .0124 .0146	0.23 .28 .31 .36 .44 .47	9.0 x 10^6 8.0 7.0 7.1 6.5 6.4	110 - - 120	100 - - 100
(2)	De-icer, and ice on leading-edge region	.0105 .0139	.23 .44	9.0 6.6	110 135	100 114
(3)	De-icer, and ice on leading edge and cap-strips	.0435	.23	9.1	458	414
(4)	De-icer, and ice on cap-strips only	.0455	.23	9.2	480	433
(5)	De-icer removed, rivets unfaired	.0099 .0111	.23 .45	9.0 6.6	104 107	94 86
(6)	De-icer removed, rivets faired	.0093 .0095 .0103 .0106	.22 .23 .45 .45	9.6 9.3 6.8 6.7	100 100	91 81

RESULTS AND DISCUSSION

The equation for the profile-drag coefficient as given in reference 4 may be written

$$C_{Do} = 2 \int_{W} (1 - \sqrt{(H_1 - p_o) / (H_o - p_o)}) (\sqrt{(H_1 - p_1) / (H_o - p_o)}) \, d\,(y/c)$$

Where:

y	is distance measured across wake.
c,	wing chord.
H_o,	total pressure of undisturbed stream.
p_o,	static pressure of undisturbed stream.
H_1 and p_1,	corresponding values in the survey plane.

Since $H_o - p_o$ equals the dynamic pressure in the undisturbed stream and $H_1 - p_1$ equals the dynamic pressure at the survey head, then $H_1 - p_o$ equals the difference in static pressures subtracted from the survey dynamic pressure. The three pressure readings taken in flight were sufficient, therefore, to evaluate the quantity

$$(1 - \sqrt{} (H_1 - p_o) / (H_o - p_o)) (\sqrt{} (H_1 - p_1) / (H_o - p_o))$$

for each value of y across the wake.

This quantity was plotted against y and, by multiplying the area under the curve so established by 2/c, the value of C_{Do} was obtained. A typical curve of the data is shown in figure 11. During the 125-mile-per-hour runs of conditions (3) and (4), the width of the wake was so great that it could not be traversed by the survey mechanism; therefore the values for C_{Do} were not obtained.

The results of the profile-drag investigation are shown in table I. The repetition of the data which will be noted for some of the drag conditions indicates that check flight tests were made for these conditions. The airplane lift coefficient during each drag test was calculated on a basis of the airplane wing loading and air speed. The profile-drag coefficient of the wing (at $C_L = 0.23$) with the rivets faired was found to be $C_{Do} = 0.0095$. This is assumed to be substantially the same as the profile-drag coefficient of the wing section without the rivets and is called the plain-wing profile drag. The profile drag of the test section at the various conditions is compared with the plain-wing profile drag in column A (table I) and with the profile drag of the wing with de-icer in column B.

Removing the de-icer had the effect of reducing the section profile drag about 5 percent, and fairing the rivets resulted in a further reduction of 5 percent. Placing simulated ice on the de-icer at the leading edge did not measurably increase the profile drag at cruising air speed, although as the speed was reduced this was not the case. At 125 miles per hour, the formations on the leading edge had caused an increase of 12 percent in the profile drag over that of the wing with a clean de-icer. Attaching a simulated ridge of ice on the upper and lower cap-strips and retaining the formation on the leading edge caused the profile drag of the wing at 175 miles per hour to become over four times its normal value. Removing the ice simulations from the leading edge but allowing the ridges on the cap-strips to remain caused a further increase in the profile drag at 175 miles per hour.

The results of the tuft observations at the speed of stall are given in table II. The maximum lift coefficient was calculated on a basis of the wing loading and the indicated air speed at the instant of stalling of the test section. The values for C_{Lmax} at the various drag conditions are compared with C_{Lmax} for the plain wing in column A (table II) and with C_{Lmax} of the wing de-icer in column B.

Obtaining the values for CL by tuft observations is admitted to be subject to error because of the effect of several factors which could not be evaluated in the present investigation. However, the lift data are thought to be of sufficient accuracy to show the approximate effect of the de-icer and the ice simulations on the lift coefficient. The de-icer installation without ice simulations reduced the value of C_{Lmax}

below that for the plain wing by about 6 percent. Ice on the cap-strips reduced the C_{Lmax} to one-half the value obtained with clean de-icers.

Although the data were not determined for the landing conditions, the assumption is made that the lift coefficient during landing is similarly affected. A tendency of the airplane to roll during take-off and landing with only the 50 inches of test span affected indicates that the calculated data for C_{Lmax} are satisfactory approximations. The simulated ice formation on the cap-strips necessitated a landing speed between 110 and 115 miles per hour, whereas the normal landing speed is in the vicinity of 70 miles per hour.

The significance of the data in table II and the observations made during the tests as applied to the Lockheed 12-A airplane may be briefly stated as follows:

The increase in the profile drag caused by the de-icer or by ice formations which are confined to the leading-edge region on the de-icer result in a reduction of less than 1 percent in the cruising speed, assuming constant power. At the speed for best rate of climb, however, the increase in profile drag due to the ice on the leading edge will result in a material reduction in the rate of climb.

Although the clean de-icer causes only a slight reduction in C_{Lmax}, the presence of ice on the leading edge reduces this factor to the extent that the minimum safe landing speed in increased by about 14 percent above that of the clean wing. The minimum safe landing speed is calculated on a basis of the stalling speed of the wing cellule outboard from the engine nacelle to the wing tips and on which the de-icers are installed. According to similar calculations, ice on the de-icer attachment cap-strips increases the minimum safe landing speed about 50 percent above that for the plain wing.

CONCLUSIONS

1. The installation of the rubber inflatable de-icers increased the profile drag of the plain Lockheed wing by 10 percent at $C_L = 0.23$ and by 20 percent at $C_L = 0.45$.
2. The attachment of simulated ice to the leading-edge region of the de-icer resulted in a wing profile drag (at $C_L = 0.23$) which was not measurably different from that obtained with the clean de-icer. At $C_L = 0.45$, however, this condition increased the profile drag 35 percent above that of the plain wing and 15 percent above that of the wing with clean de-icer.
3. A formation of simulated ice along the upper and lower de-icer cap-strips increased the profile drag of the plain wing by approximately 360 percent at $C_L = 0.23$.
4. On the basis of tuft observations at the stalling speed, ice on the leading edge of the airplane wing may reduce the value of C_{Lmax} 25 percent, while ice only on the de-icer cap-strips may reduce C_{Lmax} 59 percent below that of the plain Lockheed wing.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics
Langley Field, Va., September 14, 1939.

Document 3-26(b), Excerpts from George W. Gray, Chapter 14, "Heat Against Ice," in **Frontiers of Flight: The Story of NACA Research** *(New York: Alfred A. Knopf, 1948).*

HEAT AGAINST ICE

Ice on the wing or tail adds weight, but more serious is its effect on the drag, lift, and pitching characteristics which may become so adverse as to render the airplane unmanageable. There is also the pilot's need to know at all times where he is, and that involves keeping windshields, cockpit windows, and radio antennas free. Heavy coatings on the antenna impose extra drag that sometimes tears it off. Even if the antenna is not broken, water freezing on the wire and insulators reduces the reception and transmission of the pilot's voice, and without a properly functioning radio the cloud-bound pilot is not only blind but deaf and dumb.

During the war, more than a hundred cargo planes of the Air Transport Command, flying from bases in India over the Hump to battlefronts in China, crashed in the Himalayas. Most of them were brought down by ice. In a single day in 1944, nine of these big Army transports, loaded with sorely needed supplies for the Allies' fighting forces, were lost.

Many of the fatal crashes of commercial aviation have been traced to this same cause. For years commercial transports have been equipped with anti-icing devices, but the apparatus in common use was designed to assist in meeting an emergency when it arises, not to prepare the plane for deliberate flight into ice clouds. If dangerous icing conditions are inadvertently encountered, transport pilots are instructed to turn back or land at a safe alternate airport.

"The greatest advance that can be made in air transport from its present level," said Edward P. Warner in 1946, "is not in speed, or even in economy, important as that is, but in regularity. When cancellations on account of weather are eliminated, or even reduced to a fifth or a tenth of their present number, air transport's whole status will be changed." It is difficult to see how aviation can come into its full destiny as an every-day competitor of steamships and railroads unless airplanes are made capable of flying in any weather.

MECHANICAL, CHEMICAL, THERMAL

Various ways of preventing or removing ice have been experimented with, but basically there are just three methods. In the familiar case of ice encrusting a doorstep, for example, if you hack it with a shovel and scrape it off, that's mechanical ice removal; if you sprinkle salt over the ice, and, by lowering its freezing point, cause it to melt, that's chemical; if you pour boiling water over the frozen surface, or wait for the solar rays to do the job, that's thermal. All three methods have been applied in aeronautics.

The mechanical method is represented by the rubber de-icer which was pioneered by the B. F. Goodrich Rubber Company in the early thirties. Its characteristic feature is an inflatable rubber covering which is placed over the leading edge of wing and tail. When freezing occurs, the pilot can bring the de-icer into action by turning a switch. Compressed air then alternately inflates and deflates the rubber surface, the ice is cracked and broken by these changes in shape, and the wind blows the fragments away. The de-icer has been progressively improved since its introduction in 1930, and by 1939 had become fairly common equipment on commercial transports and long-range military airplanes. There is no doubt that many a dangerous icing of wing or tail has been alleviated by this simple device. But ice rarely cracks off completely or smoothly, and the rough line of cleavage left by the broken ice may seriously increase the drag. Also, under severe icing conditions, the freezing may extend far back over the airfoil, beyond the influence of the pulsating rubber covering of the leading edge.

Chemical methods have been intensively explored both in the United States and abroad. Two schemes have reached application. One makes use of an anti-freeze liquid which when sprayed or exuded over a surface prevents water from freezing there. The liquid most commonly used is alcohol, and for propeller protection the alcohol is released onto the blades by a slinger ring mounted on the propeller hub. The flow over the blade is assisted by a feed shoe, a rubber pad covering the leading edge, and grooves in the pad distribute the alcohol to critical areas of the blade. The alcohol system has been applied also to the protection of carburetors and windshields, and for them as well as for propellers it has proved effective against moderate icing conditions. When the conditions are severe, however, ice can get the upper hand over alcohol.

The second chemical method is the device of coating surfaces with a lacquer or paste whose nature is such as to inhibit the formation of ice. For the most part the lacquers have been applied only to propeller blades, but some tests have been made of their use on the leading edges of wings, and in particular as a coating for the rubber de-icer which covers the leading edge. None of the lacquers provides permanent protection, however, although different compounds vary in their resistance to abrasion and erosion. In most cases, under severe icing conditions, the coating is worn off in an hour or so of operation.

Heat is the most obvious enemy of ice, and the NACA was early impressed with the desirability of trying it. In 1927, a study of the requirements for thermal ice prevention was begun at the Langley Laboratory by Theodore Theodorsen and W. C. Clay, and since then the researches have been expanded and extended to the two other laboratories. There is practically no phase of the airplane icing problem that has not received attention. Tests have been made, not only of thermal systems, but also of mechanical and chemical systems. Both Army and Navy have supported these efforts, and there has also been co-operation with many other government and private agencies, including the Civil Aeronautics Authority, the National Bureau of Standards, the Weather Bureau, operators of commercial airlines, and manufacturers of airplanes, engines, propellers, heat exchangers, and other components. Certain auxiliary researches were carried on under contract by the University of California and the Massachusetts Institute of Technology. Even the enemy contributed unwittingly to the effort, for a captured German airplane which embodied thermal protection for wings, tail, and propeller was sent to the Ames Laboratory for analysis, and certain of its features were adapted, improved and applied to American use.

RESEARCH ON THE GROUND AND IN THE SKY

The early studies at Langley concentrated on the wing. How to keep a wing free of ice was the first problem tackled, and models of wings were built with interior passages for conducting the heating medium. There was then no wind tunnel equipped with refrigeration, and so, when the time came to test these heat-protected models, the men at Langley decided to use the upper air as their refrigerating device. One of the models was placed on the wing of an airplane, mounted firmly a foot or two above the top of the wing on struts, the pipes were connected so that the heat could be turned on in an instant, and then the airplane took off for a high-altitude flight. When the desired height had been reached, water was sprayed on the leading edge of the model and in the below-zero temperature it quickly coated the edge with ice. Then the heat was turned on, and a camera recorded the melting of the ice and how long it took a given amount of heat to free the surface of its coating. In this way, working out changes in the laboratory, and testing them with artificially formed ice in the sky, the practical fundamentals of thermal ice-prevention were derived.

The first requirement of such a system is a source of heat, and that is freely available in the torrent of exhaust gases discharged by the engine. The second need is the ability of the surfaces likely to collect ice to conduct heat, and that need too is already met in the metal skins of wings, tails, and bodies, the glass transparencies of windshields, and the like. It would seem a simple matter to design a system for taking heat from the exhaust and conducting it to the surface where ice is prone to form. But anti-icing, like every other problem of safety, is conditioned by the necessity of keeping the structural weight as low as possible. Moreover, since the heat-conducting pipes must pass through spars, ribs, and other critical elements of the framework, there is also the necessity of guarding against any weakening of the structure.

The problem was attacked from many angles, first using steam as the heating medium. A boiler was placed in the path of the exhaust gases, and water boiled there supplied steam for the pipes leading to the front part of the wing. This arrangement was successful so far as the transfer of heat was concerned, but it added a lot of weight to the airplane. Moreover, the steam often leaked. So steam was abandoned, the boiler discarded, and the next scheme tried was that of piping the hot gases of the exhaust directly to the wing's leading edge. This reduced the weight to a minimum, and surely no heating system could be more direct. The idea was worked out by a group under Lewis A. Rodert, who then was in charge of the icing research program, and by 1938 results were so promising that a full-scale demonstration seemed called for. It was felt that the time had come to install the ice-prevention system in an airplane, seek out ice in the clouds, and determine under the critical conditions of flight the ability of the device to protect the airplane.

A thermal ice-prevention system is not something that can be bolted on, like a windshield wiper. By its very nature it partakes of the structure, and those who proposed full-scale flight experiments realized that to adapt an airplane for such research would involve ripping open the wings, body, and tail and rebuilding them with the added system enclosed and integrated with the structure, a project almost as expensive as the first cost of the airplane. Fortunately the previous experimental results were sufficient to impress both Army and Navy, and each agreed to finance a full-scale installation.

In 1939 the Navy provided funds to remodel a PBY Catalina with the wing-and-tail heating system built into it, and the Army did the same for a Lockheed 12 airplane. The Catalina was flight-tested by the Navy, with results so successful that almost immediately it was assigned to service in tough climates, eventually to Alaska. Meanwhile, the Lockheed 12 remained with the NACA and became the flying laboratory in which Rodert and his young men first tried out full-scale thermal systems under natural icing conditions.

SEEKING ICE IN THE CLOUDS

The reconstruction of the Lockheed 12 with wing and tail protection was completed in 1940, soon after the NACA broke ground for the Ames Laboratory in California. The Committee had already decided to transfer its icing program to the new laboratory on the West Coast; and so, toward the end of that summer, the Lockheed was flown from Langley to Moffett Field; and with it went the icing research staff.

One reason for the transfer was the need for more icing weather than the Virginia skies were wont to provide. The research at Langley too often had to resort to artificial means of imposing ice on the airplane. In California, the interaction of mountain ranges and ocean air provided an infallible mechanism for creating supercooled clouds. W. H. McAvoy and Lawrence A. Clousing were the pilots assigned to fly the airplane into these clouds, and Rodert and his researchers soon found plenty of ice to combat. The flights around San Francisco and along the Sierra Nevadas demonstrated that the heated wings would keep ice off. It was also shown that, if the wings were left cold for a while, only a few seconds heat were necessary to de-ice them. "This," says Rodert, "was the beginning which gave us assurance that eventually airliners would fly from Omaha to Chicago to New York and any other place, irrespective of ice clouds."

Sometimes the ice seekers got more than they were bargaining for. One Sunday morning the Lockheed was riding a storm along the Pacific coast north of San Francisco. "At about 9,000 feet altitude we came out of a large cloud and headed for another," relates Rodert. "By experience we had learned that the severest ice was found in the center of the cloud which looked the blackest. On this day the clouds were tremendous, towering to 17,000 feet. Turbulent air rocked and tossed us. As we headed to the next cloud in the eerie mist between the denser masses, a violent explosion shook us with a blinding flash. The lightning apparently was drawn out of the clouds by the path of our airplane and the blast went right through the craft, stem to stern. Our first thought was fire. We all sniffed. No smoke. Then—did we still have radio communication? Yes—as much as usual in a storm. We went over the airplane and found nothing out of order. After a little hunting for Moffett Field with our radio compass, we landed and were able to make an inspection of the exterior. The lightning had burned metal off the propeller, wing, and tail, but other than requiring minor repair on one or two places, our craft was intact. Airplanes have been struck by lightning many times without any evidence that such events, spectacular as they may be, result in catastrophe."

It was experiences such as this, and the danger of collision in that heavily traveled coastal area, that prompted the group to look for still other skies. Rodert and Cousing were Missesotans, and they remembered that a region northeast of Minneapolis was not traversed by airlines. They knew that a winter there would provide plenty of icing without the extra hazards of mountains and lightning. So in November of 1941 the Lockheed was flown to Minneapolis, and within five days the researchers were delighted with a rich harvest of data. They stayed on, and bucked the icy skies with their hot wings all winter. In January, the Consolidated-Vultee Company asked that the system be applied to their B-24 Liberator, and the Army turned over one of its Liberators for that purpose. Then it was requested that the anti-icing system be installed in the B-17 Flying Fortress. That gave the group two heavy bombers to experiment with, and after their heating systems had been built in, these airplanes were ferried from Ames to Minneapolis and flight tested at Minnesota Ice Research Base. This base, meanwhile, had been taken under the wing of the Army Air Forces and made the cooperative center for icing research, beginning with the winter of 1942-43. That winter there were seven pilots and seventy-five mechanics on duty at the Ice Research Base, in addition to research engineers; and thirteen airplanes with newly installed ice-prevention systems were tested.

In 1943, the Army turned over a C-46 airplane to the Ames Laboratory, and all the research techniques and devices that had been developed in the 9,000 pound Lockheed 12 were used on this 45,000 pound Commando—thermo-couples, strain gauges, air pressure orifices, and the necessary equipment to power the apparatus and record the results. The cabin provided ample space for the complicated equipment with its maze of wires, instruments, and 50,000 watt generator, and there was plenty of elbow room for the crew of three research engineers, two pilots, and a mechanic to move around in. The improvement of thermal systems was rapidly advanced by the use of this large airplane with its extensive instrumentation.

THREE WAYS OF TRANSMITTING THE HEAT

In the course of the flight experiments, several methods of transmitting heat to the surfaces were investigated. The scheme of piping hot gas from the engine exhaust, which had been substituted in the Lockheed 12 airplane for the earlier experimental steam system, was effective so far as heat conduction was concerned, and it added very little weight to the airplane, but in other ways there were disadvantages. In the first place, the hot vapors from the engine carry corrosive acids and other chemicals which have a destructive effect on aluminum and steel. In military airplanes, there was the additional risk that an enemy bullet might puncture the piping and release noxious fiery gases into the wing or fuselage. So the experimenters passed on to a new arrangement employing heated air, and this too was tried first in the Lockheed 12. The idea was to place a heat exchanger in the stream of the engine exhaust, use it as a furnace to heat fresh air drawn from the atmosphere, and then pipe this heated air into the leading-edge region of the wing.

The heated-air system was applied first to wings, and its value demonstrated there before attempts were made to serve other parts. The method was extended to the tail, to protection of the windshield, and finally to heating the cabin. In the case of the windshield, the transparency is made of two sheets of laminated safety glass separated by a space of less than half an inch. It is this intermediate space that receives the heated air. For satisfactory protection, the front of the windshield must be kept at 50 degrees F. How much heat is necessary to maintain that temperature depends on the airplane's speed, among other things. At a speed of 150 m.p.h., a heat flow of 1,000 British thermal units per hour per square foot of windshield surface is necessary. For protection of wing and tail, the system must be able to heat the forward ten percent of their surfaces 100 degrees F. above the outside temperature when flying through dry air speeds up to 200 m.p.h.; surfaces back of the forward ten percent should be heated one-fourth to one-half of this amount.

It was this type of thermal protection—the heated-air system—that the Ames Laboratory applied first to the Lockheed 12, then to the B-24, the B-17, and the C-46.

Compact, highly efficient heat exchangers were developed in the course of these experiments. A design built at the Ames Laboratory weighed only 30 pounds, occupied a cylindrical space eight inches in diameter and twenty-two inches long, and was able to put into the circulating air 315,000 British thermal units per hour—an average of more than 10,000 for each pound of its weight. The exhaust gasses enter the heat exchanger with a temperature of 1,500 degrees F., and there was a problem of finding metals refractory enough to stand up. Not only were they found, but the weight was kept so low that the complete system, including heat exchangers, connecting pipes, and all auxiliary equipment, adds no more than one-and-one-half percent to the gross weight of the airplane; in some cases much less. For example, an installation designed by the Ames group for a four-engine bomber of 60,000 pounds gross weight added 300 pounds, or only half of one percent. This compares with 230 pounds for the less effective inflatable rubber de-icer which breaks off ice after it is formed, and serves only wings and tail, whereas the thermal system is preventive, protects windshield as well as tail and wings, and in addition keeps the cabin comfortably warm.

Still a third method of getting heat from the exhaust to the surfaces has been tried out in the Lockheed 12 airplane, and with very promising results. In this system, air is taken from the atmosphere and mixed with ten to fifteen percent of its volume of exhaust gas, whereupon the mixture is conducted by pipes in the usual way to wing, tail, and windshield surfaces. The exhaust gas is so hot, around 1,500 degrees F., that even the small percentage is sufficient to deliver the mixture to the wing at 300 degrees F. The use of heat exchangers is dispensed with, which simplifies the installation as well as saves weight. Mixing valves are required, but for a four-engine airplane they add only about 20 pounds, to compare with 120 pounds for four heat exchangers. Use of the gas-air mixture thus promises a saving of about 100 pounds on an installation of this size. Experiments in an industrial laboratory and at Ames indicate that the corrosive effect for so small a percentage of exhaust gas is quite minor, indeed negligible when non-corrosive coatings are applied to the inner surfaces.

A still more recent series of experiments is investigating the use of electrical heating. The idea is to build resistance wires into the surfaces of wings and tails, or build them into pads which can be bonded to these surfaces—applying the principle of the electric blanket. For the windshield, the heat is supplied by passing current through a transparent conducting surface film on the windshield glass or through a network of fine electrical resistance wires embedded in the glass. These experiments follow earlier studies in the application of electricity to the protection of the propeller.

ICING RESEARCH IN THE WIND TUNNEL

At the time that headquarters for icing study were transferred from Langley to Ames, plans were being formulated for the Flight Propulsion Research Laboratory whose construction was begun the following January. It was early decided to equip the new research center with a specialized wing tunnel for controlled studies of icing on the ground, something that was needed to check results obtained in flight and also to prospect problems in advance of or independent of the flights studies.

A prime reason for placing the new tunnel at Cleveland was the presence there of the huge refrigerating plane which serves the altitude wind tunnel. This refrigerating plant in the largest "cold factory" in the world, and the installation is such that its full capacity can be turned to cooling either tunnel. Temperatures as low as -65 degrees F., and wind velocities as high as 320 miles per hour can be generated in the icing-research tunnel, and there are three test sections. One, a high-speed section measuring six-by-nine feet across, is used for studying ice problems in an engine, on a wing, windshield, antenna, or other part. The second test section, twelve-by-fifteen feet, is primarily for research on the propeller. The third section was planned

to study icing problems of the rotation wing. Artificial blizzards and other freezing conditions can be turned on at will in the tunnel, and most of the icing situations likely to be encountered in flight can be set up in experiments and be investigated quite irrespective of the outside weather.

The tunnel was completed and came into use in 1944, with Wilson H. Hunter in charge. V-J Day thus found both Ames and Cleveland engaged in studies of ice formation and seeking methods of preventing it. The work of the two laboratories was closely correlated. At the height of the wartime effort in 1945 the group at Ames numbered 32 men, including such varied specialists as a chemist, a metallurgist, a physicist, a meteorologist, and an electrical engineer, in addition to mechanical engineers, pilots and mechanics. At the same time 49 men were engaged in icing research at Cleveland, most of them employed in the tunnel. Cleveland also makes flight studies. During the war it used a P-38 Lightning to investigate the effect of the turbo-supercharger on carburetor icing, and it is currently using two Army bombers, a B-24 and a B-25, in icing research. The B-24 has been fitted out as a flying laboratory to study turbo-jet icing; the B-25, with considerable instrumentation, is specializing on the effects on airplane performance of ice formations on propellers and other aircraft components.

PROPELLER PROTECTION

The propeller problem is different from that of wing, tail, and windshield, because it involves a rotating element. Ice on the whirling propeller, in addition to its damaging effect on thrust, can set up destructive vibrations. One requisite in designing an ice-prevention system is to guard against imposing any mass on the blades, or subtracting any, which would unbalance the propeller and be a possible source of resonance and vibration. It is of course essential that anti-icing have no adverse effects on propeller aerodynamics. Other objectives are economy of operation, safety of operation, and pilot convenience. The ideal would be a robot controlled by a thermostat that automatically turned on the system at the first encounter with ice; but this is yet to be.

During the war, the chief reliance was placed on fluid systems which flow alcohol or some other anti-freeze liquid over the blades, in the manner described earlier in this chapter. Not only military airplanes but the commercial transports made wide use of this method. A three-blade propeller, of the type commonly driven by 1,100-horsepower engine, requires about three quarts of alcohol per hour, and that involves carrying a reserve supply of the liquid. There are stories of airplanes lost when gunfire punctured and ignited their tanks of anti-freeze. Moreover, the presence of the feed shoe on the blade affects aerodynamic efficiency; the grooved surface has less thrust than the naked blade, and as a result the take-off requires a longer distance, the rate of climb is reduced, and the airplane's forward speed slowed down. With the trend to bigger broader blades of the paddle type turning fewer revolutions per minute, the adverse effect of the feed shoes on propeller performance becomes even more serious. The NACA has made many studies of fluid systems, first at Langley, and then at Ames, and its staff is convinced that for propellers, as for wings and tails, the best hope lies in the use of heat.

Several thermal systems have been projected, some using the principle of internal heating, others supplying heat only to the surface. For surface heating, two electrical systems are in the testing stage, one using a pad containing resistance wires, the other a special type of synthetic rubber which itself conducts the electric current, and in doing so generates its own heat. These thin heating pads are cemented over the leading edge of each blade, and their smoothness is such that the airflow characteristics of the propeller are not impaired by the added surface. However, the added surface does not remain smooth, for grit, sand, gravel, and other rough particles in the air abrade the pads and make frequent replacements necessary. It is this relatively short life of surface installations that has caused researchers to turn more and more to experiments with internal heating.

Internal heating requires special blades. The NACA's first experiments with this idea made use of hollow blades through which heated air was passed and released from holes in the blade tips. At first it was thought that the tip holes imposed an objectionable drag, but wind-tunnel studies have shown that the effect is inappreciable. The system requires a heat exchanger at the engine exhaust, and in the effort to eliminate this extra complication with its added weight the workers at Cleveland have been experimenting with a scheme for injecting fuel into the blade cavities, burning it there, and discharging the combustion gases through the tip holes, thus heating the blades directly by setting off a fire inside each of them.

Electricity has also been applied to internal heating. For this the blade must be built with a hollowed section under the area to be served, and into the hollow a rubber housing containing the usual electrical resistance wires in installed. Fortunately aluminum and steel, the usual structural materials for propeller blades, are good heat conductors, so any temperatures engendered within the metal quickly communicate their effects to the surface.

A principal drawback to the use of electricity is the large drain that must be made on the power system to generate the current. This is true of both internal and surface systems. Ames found that complete protection of each propeller of the Flying Fortress required about 5,000 watts. Fairly satisfactory protection for all but extreme icing conditions was had with only 2,400 watts per propeller. The former figure seems a heavy tax. It represents a total of nearly 28 horsepower for the four propellers. And, in addition to subtracting that much power from propulsion, it adds the weight of electrical generators which may total 120 pounds, for the four engines. Manufacturers are working on the problem of designing generators of lighter weight, and meanwhile the group in the icing research tunnel has investigated cyclic heating. Picture a system for a four-blade propeller in which the electrical current is on two blades for thirty seconds, then on the other two blades for the next thirty seconds, and so on alternately as long as protection is needed. Such an

arrangement would require a smaller generator than a continuous system.

In one of its research flights, the C-46 at Ames had a quick demonstration of what ice can do to propeller performance. On this occasion it became necessary to shut off the electric heating from one of the two propellers while a new combination of instruments was being set up and plugged in. Within less than a minute ice began to form on the unheated propeller, and in the three minutes that were required to change the instrumentation the blade became so encrusted that the airplane speed dropped from 144 to 128 m.p.h. Energetic action was required from the pilot to keep the big airplane under control. Finally the new connections were plugged in, the electricity turned on, and—it was happiness to the crew to see the ice melt away. One minute of heating was sufficient to clear the blades, and the speed returned to 144 m.p.h.

POSTWAR FLIGHT RESEARCH

During the war, icing research was focused on obtaining results that could be quickly applied, and various approximate solutions were worked out. Many of the solutions came to application, and by 1946 thermal ice prevention systems were being installed as part of the standard design of a number of leading transport and cargo planes as well as of long-range bombers. The NACA, however, felt the need of more basic knowledge, and after V-J Day there was a return to fundamental problems. The primary objective was the establishment of more exact design data. It was recognized that an ice-prevention system should be especially designed for the airplane it is to serve, with no excess heating and no unnecessary weight added.

The size and shape of the wing, blade, antenna, windshield, or other part are necessary criteria for designing such a system, but so too are the operating conditions under which the airplane will be flown and the performance to be required of it. For example, its speed affects an airplane's capacity for collecting ice; so does its angle of attack; so does the altitude; and a whole series of weather items enter into the picture—factors such as the liquid content of the air, the water droplet size, the size distribution of droplets in icing clouds, and the temperature of the air. The postwar program has sought precise information on all these criteria, rigorously evaluating the influence of each on ice formation, and measuring the amount of heat required to protect a surface of given shape and size under every combination of circumstances. The end result will be design curves from which may be drawn the exact specifications for protecting any particular airplane.

Flight research has been the primary tool of these studies, and both the Ames and the Cleveland laboratories have been active, employing techniques that were largely pioneered at Ames. The C-46 remains the principal flying laboratory of the Ames group, with Alun R. Jones in charge. But instead of the wartime practice of going to the Ice Research Base above Minneapolis for its skies, the C-46 has been making flights over a triangular course of 1,000 miles—between San Francisco, Seattle, and Salt Lake City—with pilots of the United Airlines at the controls. The

Cleveland laboratory, for its research, has been using the B-24 and the B-25, and these planes are flown as far east as upstate New York and as far west as Fargo, North Dakota, wherever the weather offers the icing conditions best suited for the study. The Cleveland flight research is under the direction of Mr. Rodert, who became attached to that laboratory in 1946.

Much of the B-25 research was concentrated on studies of propeller icing, but the program also included careful checking of other parts as well. In the propeller studies, the rest of the airplane is kept free while the propeller is allowed to gather ice, and then measurements are made to determine how deposits on the propeller affect the performance of the airplane. The research is also concerned with how rapidly the propeller accumulates ice under a given combination of circumstances, how much heat and electric power are required to protect it, how much time is required to de-ice it, and so on. Wing icing is studied in the same way: all other parts are kept warm, and as the cold wing panel collects ice, instruments measure the effect of this on the performance of the airplane and determine in thermal units the minimum amount of heat needed for adequate protection. And so with the nose of the fuselage, the cowl rings, and other parts—the icing tendencies and performance effects of each are appraised in turn, and the indispensables of good protection are determined.

In the same way, Cleveland's other ice-seeker, the B-24, flying over the same regular airline routes between northern New York and North Dakota, has been used for special studies of two critical components: the antenna and the windshield. In the case of the former, a number of experimental antennae of different lengths were installed on the B-24 and set at angles ranging from 0 degrees to 90 degrees with respect to the direction of flight. Strain gauges mounted on the antennae measured the stresses set up by air drag. The increases in drag caused by ice encrustations are quite substantial. In a study in the icing research tunnel at Cleveland, more than 90 horsepower was required to overcome the drag of a heavily iced antenna, whereas without ice the antenna's drag presented only 30 horsepower. Small slender shapes, such as rods and wires, collect ice more rapidly than larger structures, and efforts have been made to design antennae in streamlined form.

While the ice-prevention specialists are concerned with the antenna because of its proclivity for collecting ice, the aerodynamicists are troubled with its drag. And as airplanes go to high speeds, this drag problem becomes increasingly serious. The idea of burying the antenna beneath the surface of the airplane has been experimented with in the effort to eliminate its drag. If installed within the metal skin of the airplane itself, an antenna ceases to function normally, since the metal shields it from radio signals; but there are schemes to make grooves or depressions in the airplane body or tail as housings for the antenna, and then fair these places over with non-conducting plastic. When this idea is successfully applied it will solve both the drag problem and the icing problem.

For studying windshield icing, the nose section of the B-24's fuselage was

modified to provide for the installation of test panels of experimental windshields. These electrically heated panels were mounted at various angles of incidence and instrumented with the idea of determining the influence of shape, size, and angle. While the observer, sitting in what was the bombardier's compartment, watches and records these effects, automatic instruments measure the meteorological factors of the atmosphere through which the airplane is passing. In this way the amount of heat required to protect a windshield of given size, shape, and angle is correlated with the ice-making conditions of the air.

The postwar research contrasts with the wartime research in the greater exactitude of its measurements, and this in turn has been made possible by new and more refined instruments. Many of these new instruments were developed by Mr. Jones and his group at the Ames Laboratory, assisted by advice from the U.S. Weather Bureau. For example, it is extremely difficult to measure free-air temperature. Even in dry air such a feat is difficult, and to do it in a cloud of unknown moisture content required no end of ingenuity. "Similarly," remarked Mr. Jones, "the determination of the amount of free water in a cubic foot of cloud through which you are flying at 150 to 200 m.p.h. is a problem to be approached with respect." The co-operation of the Weather Bureau has been an important element of the icing research program, and Ames and Cleveland each has a Bureau meteorologist attached to its staff.

A primary object has been the development of a theory of thermal ice-prevention based on the observed and measured reactions of the heated surfaces to known meteorological conditions. A theory has been proposed by J. K. Hardy of the Royal Aircraft Establishment at Farnsborough, England, and during the war Mr. Hardy spent two years in research at the Ames Laboratory and demonstrated how the dissipation of heat in conditions of icing may be calculated from that in free air. It was interesting to find that his theoretical calculation of the heating requirements of the C-46 wing was in reasonable agreement with the standards adopted by the Ames researchers on experimental indications. Perhaps theory will not make too drastic changes in practice; but good design requires that an airplane be stripped to its essentials and carry no useless weight, and for such calculations a sound theory which has been thoroughly checked and confirmed by experiment is necessary.

The group at Ames has also investigated the effect of the thermal system's heating on the strength and durability of the airplane's structure. Temperature measurements taken of the inner skin, nose, ribs, baffle plate, and other "insides" of the C-46 wing show that they remained at a temperature below 300 degrees F., when the wing's ice-prevention system was operating efficiently in dry air at the conditions for which it was designed. If we assume that the wing will not get hotter than 300 degrees F., assuming also that it is made of the usual 24ST aluminum alloy, it appears that the designer must reckon on a reduction of 11 percent in the tensile strength of the structure.

Heat can also render metal more susceptible to corrosion. An extensive review has been made of the published information on this subject, and it would seem to indicate that so far as previous experience goes no such effect need be feared from 24ST sheet at temperatures below 300 degrees F., provided the sheet has been properly aged.

Heat causes metal to expand, and there is a question how the difference in temperature, for example, between the cooler outer surface and the hotter inner surface of a heater wing's skin may affect its internal stresses. Certain measurements made in the C-46 showed substantial increases in tension, compression, and shear stresses in the wing's leading edge. At some points these internal stresses amounted to 5,000 pounds to the square inch.

There are thus three metallurgical effects of heating to be considered: the effect of heating on the strength of the metal, on its corrosion-resistance properties, and on the internal stresses of skins and other parts whose outer surface is exposed to freezing temperatures. All three problems are receiving continued scrutiny at the Ames Laboratory.

Document 3-27(a-h)

(a) Richard V. Rhode, Associate Aeronautical Engineer, Langley Field, VA, to Mr. Charles Parker, Secretary, Aeronautical Chamber of Commerce, 300 Madison Ave., New York City, "Load Factors--Comments by Richard V. Rhode, NACA, for minutes of meeting held in room 240, Statler Hotel, Detroit, April 5, 1932," 14 Apr. 1932, copy in RA file 287, LHA (In pencil at top of letter, "This letter not sent.")

(b) H. J. E. Reid, Engineer In Charge, to NACA, "Program proposed by Mr. Rhode for determination of dynamic overstress of wings in gusts," 15 March 1933, RA file 287.

(c) G. W. Lewis to LMAL, "Research authorization on study of load and load distribution on commercial type airplanes," 5 Apr. 1933, RA file 287.

(d) Richard V. Rhode, Aeronautical Engineer, Langley Field, VA, to Engineer In Charge, "Proposed daily weather flights and gust tail load measurements on modern airplane," 6 June 1935, RA file 287.

(e) "Extracts from Letter Dated April 12, 1936, From Honorable Edward P. Warner to Dr. J. S. Ames, National Advisory Committee for Aeronautics, With Comments of Dr. D. M. Little, U.S. Weather Bureau," 12 May 1936, RA file 287.

(f) Edward P. Warner, 122 East 42nd St., to Dr. J. S. Ames, NACA, Navy Buillding, 18 Apr. 1936, RA file 287.

(g) Dr. G. W. Lewis, Director of Aeronautical Research, NACA, to Mr. L. V. Kerber, Chief, Aircraft Worthiness Section, Bureau of Air Commerce, Department of Commerce, Washington, DC, 26 Feb. 1938.

(h) Interview transcript, Richard V. Rhode to Michael D. Keller, Hampton, VA, 10 Jan. 1967, pp. 1-6, copy in LHA.

This string of documents on the early days of the NACA's gust loads research represents another major nexus in the airplane design revolution—one linking aerodynamics, meteorology, and structures. It also shows the importance of instrumentation—in this case the V-G (Velocity-Gravity) recorder, an instrument that recorded speed and the acceleration along one or more of the principal axes of an aircraft. This was not the first—and certainly not the only important flight instrument; by that time, there were also accelerometers, strain gauges, pitot tubes, and motion-picture cameras. But the V-G recorder certainly proved to be an essential instrument, one that provided key data guiding aircraft design. In essence, it was aviation's first "black box."

NACA Langley engineers Richard V. Rhode (pronounced "Road-ee") and Henry J. E. Reid (Langley's Engineer-In-Charge from 1926 to 1940) invented this crucial new recording instrument in 1929-30. What led to their design was a series of thoughts about the relationship between atmospheric gusts and airplane aerodynamics. Flight researchers investigating gusts noticed that when an airplane encountered a gust, the airplane behaved just as if it had changed its angle of attack. Changes in angle of attack changed lift, causing the airplane to move vertically; analysis showed that how much of this vertical "acceleration" took place was a function of the airplane's speed and the degree to which the angle of attack changed. Rhode and Reid recognized that the relationship between velocity and acceleration due to gravity could be computed—better yet, the aircraft itself could be used to measure the gusts encountered. The NACA engineers also understood that larger aircraft were generally less maneuverable than smaller ones. Both large and small aircraft would inevitably encounter gusts brought on by thunderstorms and other bad weather, but it was undoubtedly more critical to design the right amount of strength into large aircraft, transports in particular, because their greater size and versatility resulted in more strenuous gust loads. With these conclusions in mind, Rhode and Reid built their first V-G recorder. They screwed on the instrument, which was about the size of an old-style alarm clock, to a firm structural part of the airplane near its center of gravity. It then automatically recorded the greatest accelerations encountered by the airplane in relation to its speed. The last document in the string below is an excerpt from a 1967 interview with Rhode, in which he recalled the invention of the V-G recorder.

Though developed for NACA flight research, the V-G recorder became ubiquitous. A number of U.S. airliners began carrying them in the 1930s, including several trans-continental passenger planes. The famous Clipper flying boats of the era all carried V-G recorders on their flights across the Pacific to Hawaii, the Philippines, and China. In World War II, thousands of U.S. and Allied aircraft (a number of the instruments were lent to the British Air Ministry early in the war) came equipped with the instrument. The V-G's contribution to airplane design proved substantial. When the scratchings on the blackened disks of the instrument were examined, sometimes after months or even years of service, the influence of gusts and all the other vicissitudes of flight were recorded precisely in terms of speeds and inertial loads. In doing that, the V-G recorder not only told the story of what happened to an aircraft in an accident or fatal crash but also generated data that was extremely helpful in aircraft design. Before the end of the 1930s, there was enough data on gust loads for U.S. aviation regulators to establish a design criterion of 55-feet per second "effective gust velocity."

As this string of documents suggests, the NACA pioneered the field of gust loads research during this period. In 1936, it purchased two light airplanes—a Fairchild and a Stinson Reliant—and with them Langley flight researchers studied the structure of what they called the "natural gust." In 1937 Langley also started to design a ground facility in which the effects of gust gradient on aircraft response could be measured. Not completed until after World War II began, this facility consisted of a vertical jet of air for simulating a gust, a catapult device for launching dynamically-scaled models into the air jet, curtains for catching the model after it traversed the gust, and instruments for recording the model's responses. By 1940, the NACA had elicited several different means by which to improve an airplane's response to gusts, including gust "alleviators" that did such things as affect longitudinal control through variations in elevator angle.

Without question, gust loads research offered information critical to the design and operation of all American aircraft designed from the mid-1930s on.

Document 3-27(a), Richard V. Rhode, Associate Aeronautical Engineer, to Mr. Charles Parker, Secretary, Aeronautical Chamber of Commerce, "Load Factors," April 14, 1932.

April 14, 1932.

Mr. Charles Parker, Secretary,
Aeronautical Chamber of Commerce,
300 Madison Ave.,
New York City

Dear Mr. Parker:
Subject: Load factors—comments by
Richard V. Rhode, N.A.C.A., for minutes of meeting held in room 240.
Statler Hotel, Detroit, April 5, 1932.

We believe the line of attack being followed by the Department of Commerce is proper and rational in so far as the correlation of the loads with the speed and lift coefficients is concerned. The question of the value of the total loads that are likely to be applied in flight is, however, still entirely unsettled and probably will remain so for sometime to come.

It is my opinion that the stresses in gusts will ultimately be found to be the important factor for the time design of civil airplanes. I do not believe that stresses in the ordinary maneuvers exceed the maximum gust stresses. A great deal of accelerometer data show very conclusively that the "common garden variety of maneuver" does not impose accelerations greater than about 3g even for single and two-seated airplanes. On the other hand, accelerations in gusts as high as 3g have been measured. These accelerations, however, do not necessarily represent the stresses. It is a fact that in the more serious gusts—those with relatively steep velocity gradients—there is an excess stress due to the dissipation as internal work in the structure of the kinetic energy gained by the wing as it deflects with respect to the main mass of the airplane. This "dynamic overstress" may vary from 0 to 50 percent or more of the stress due to the applied acceleration considered as a static load. Thus if an acceleration of 3g is measured in a gust, the stress may be anywhere from that caused by a static load factor of 3 to that caused by a static load factor of 4-1/2 or more. The exact overstress depends on the gradient of the gust and the flexibility of the wing structure, as well as on the speed of flight and the gust intensity. These matters at present are very imperfectly understood, and no attempt to calculate them with precision is justified because of the lack of sufficient definite information on gust gradients.

The N.A.C.A. has purchased an instrument for measuring structural deflection in flight and also strain gauges suitable for use on airplanes in flight. It is hoped that by making use of such instruments in conjunction with a combined air-speed accelerometer developed by the N.A.C.A. some definite conclusions regarding gust stresses may be formulated from the statistical evidence gathered.

The most enlightening information on loads at present in the possession of the N.A.C.A. is applicable almost entirely only to military aircraft and in particular to aircraft required to undergo the most severe military tactics. This information has not been published, although the N.A.C.A. contemplates doing so in the near future. From the standpoint of the design of civil aircraft this information will be of value only as applied to a special acrobatic or unrestricted category. I think it will demonstrate rather clearly that the critical load factors are determined primarily by physiological and psychological considerations of the pilot and that a logical design will proceed from a knowledge of the applied load factor coordinated with certain characteristics of the airplane, such as speeds and wing loading. I am tempted to believe, from the limited data available thus far, that for all practical purposes the probable maximum applied load factor will be found constant or nearly so throughout a range of speeds from V_{max} to $V_{terminal}$, with the low-angle load factors somewhat higher, perhaps, than the high-angle factors. Only further statistical data, however, will definitely prove or disprove this opinion.

Yours very truly,

Richard V. Rhode,
Associate Aeronautical Engineer.

RVR.IT

EWM

Approved:

H. J. E. Reid
Engineer-in-Charge.

Document 3-27(b), H. J. E. Reid, Engineer-in-Charge, to NACA, "Program proposed by Mr. Rhode for determination of dynamic overstress of wings in gusts," March 15, 1933

Langley Field, Va.,
March 15, 1933.

From LMAL
To NACA

Subject: Program proposed by Mr. Rhode for determination of dynamic overstress of wings in gusts.

1. There is forwarded for your information and approval a memorandum prepared by Mr. Rhode suggesting a program of flight tests involving measurements of accelerations and wing stresses in gusts. Mr. Crowley is of the opinion that this work is an essential step in the interpretation of the load factors they are attempting to establish, particularly on transport airplanes. He feels that this program will enable us to maintain an advanced position in this field of work, as compared with the work of other laboratories.

2. Our investigation of load factors and gusts involving the use of V-G recorders on commercial airplanes has all been charged to R.A. 287, but it is realized that we have gone beyond the scope originally intended in this authorization since R.A. 287 provides for an investigation on our Fairchild cabin monoplane. It is believed that R.A. 287 should be replaced by a new research authorization, or that the method of procedure should be revised somewhat as follows: "Flight tests are to be made on a number of commercial airplanes in which pressure distribution, stresses, and load factors will be measured as may be desirable. These flights will be made in all weather conditions and over all types of terrain. The main source of information will be transport airplanes in regular operation on established airways."

H. J. E. REID

Engineer-in-Charge

Document 3-27(c), G.W. Lewis, Director of Aeronautical Research, to LMAL, "Research authorization on study of load and load type distribution on commercial type airplane," April 5, 1933.

Washington, D.C.
April 5, 1933.

From NACA
To LMAL

Subject: Research authorization on study of load and load type distribution on commercial type airplane.

Reference: LMAL letter, March 15, EWM.DD, paragraph 2.

1. Consideration has been given to the question of reference as to the suitability of continuing to charge to Research Authorization No. 287 the work on the study of load factors and gusts on commercial airplanes in actual transport operation by the use of the N.A.C.A. V-G recorder.

2. In view of the fact that the work the Committee is actually doing on this problem at the present time comes within the scope of the title and purpose of the investigation as stated in the research authorization, it is believed unnecessary to request a new authorization or an extension of No. 287 to cover the very different procedure being followed in conducting the investigation. Since it was not possible at Langley Field with the Fairchild cabin monoplane to obtain information regarding the load factor on commercial airplanes under the varying conditions encountered in actual operation, approval was given by this office, as within the scope of administrative discretion, to the extension of the procedure to airplanes in operation in commercial air transport and the study of accelerometer and V-G recorder records obtained in such operation.

G.W. Lewis.
Director of Aeronautical Research.

Document 3-27(d), Richard V. Rhode, Aeronautical Engineer, to Engineer-in-Charge, "Proposed daily weather flights and gust tail load measurements on modern airplane," June 6, 1935.

Langley Field, Va.,
June 6, 1935

MEMORANDUM For Engineer-in-Charge.

Subject: Proposed daily weather flights and gust tail load measurements on

modern airplane.

1. The increasing importance of the atmospheric gust as it effects many phases of aeronautics makes it desirable to prosecute research on gust phenomena more intensively than we have done in the past. There seems to be no question that transcontinental transport flights will be carried out at altitudes upwards of 10,000 feet. The question of the applicability of our transport V-G data, which have been obtained at relatively low altitudes, at these higher altitudes therefore arises. It is also of increasing importance to know the relations between gustiness and types of weather in addition to the relation with altitude.

2. While we have attempted in the past to obtain some of these relations by (a) asking transport pilots to fill out information forms to accompany the acceleration records, and by (b) cooperating with the Weather Bureau in obtaining acceleration records on their daily airplane flights at Cleveland, these attempts have proved unfruitful. In the first case, pilots have failed to cooperate, and in the second case, suitable accelerometer equipment for the conditions of the job was not available.

3. I believe, however, that we should and can obtain useful information along the lines mentioned by making more or less regular daily flights at Langley Field with certain equipment as set forth below.

4. It is proposed, therefore, to use a modern airplane carrying our optograph equipment modified to take certain pertinent measurements as follows:

a. Acceleration.
b. Air speed.
c. Temperature.
d. Pressure.
e. Humidity.

Certain other measurements may be desirable, but those above are fundamental

to the main research. The airplane would be climbed daily to 17,000 or 18,000 feet with some few short level flight runs at intermediate altitudes. It would be necessary to equip the airplane with blind flying instruments, including radio, as are now used by all military and commercial airplanes, so that cloudy or overcast conditions would not interrupt the flights. Oxygen equipment is also desirable.

5. It is proposed also to extend our gust tail load measurements to this airplane, tasking measurements along with the measurements listed above. The present gust tail load project on the O2H was undertaken primarily to test the feasibility of the method, with the understanding that the O2H was not entirely suited to the job and would be useful for only a limited period. It was anticipated that, if the method proved feasible, the tail measurements would be continued on a more suitable airplane. It is believed that the preliminary results on the O2H indicate both the feasibility of the method and the importance of gust loads on tail surfaces, and hence justify further research on a modern airplane.

6. The airplane desired is one of modern type and performance. From the standpoint of the tail load measurements, it should be similar to modern and probable future transport types; via., a low-wing tapered monoplane. This requirement is not essential from the standpoint of the acceleration and weather measurements. It is therefore proposed to start the weather and acceleration measurements on the XBM-1 airplane; this work can proceed immediately. It is also proposed that the committee take steps to procure a fast low-wing monoplane on which to continue the weather measurements and also on which to make simultaneous measurements of gust tail loads. A Vultee transport would be ideal, although one of the new Northrop attack airplanes ordered by the Army would be suitable. In this connection it seems worthwhile to point out that if a cabin job were procured, it could also be used to carry on the present functions of our Fairchild FC-2W2, which is rapidly becoming obsolete and worn out. The installation required for the researches herein mentioned would not interfere with the cabin space or use of the airplane for transport work.

Richard V. Rhode,
Aeronautical Engineer.

Document 3-27(e), Extracts from letter dated April 12, 1936 from Edward P. Warner to Dr. J. S. Ames, National Advisory Committee for Aeronautics, with comments of Dr. D.M. Little, United States Weather Bureau.

EXTRACTS FROM LETTER DATED APRIL 12, 1936, FROM HONORABLE EDWARD P. WARNER TO DR. J. S. AMES, NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS, WITH COMMENTS OF DR. D. M. LITTLE, UNITED STATES WEATHER BUREAU

"The most important of aerodynamic problems from the transport point of view seem to be those lying midway between aerodynamics and meteorology, and perhaps deserving the special notice of the Subcommittee on Meteorological Problems as well as that of the main Aerodynamics Committee. Specifically, it is of the greatest importance that all possible information be secured, and that existing knowledge be extended with all possible rapidity, on the subject of the structural effects of atmospheric disturbances. The type of indirect analysis of gust structure upon which Mr. Rhode and the flight section at Langley Field are currently engaged (seemingly in a comparatively small way) has a potential value that cannot be exaggerated. The transport operators in attendance at the recent meetings assured their fullest cooperation in the installation of instruments on their airplanes, and in the keeping of such records as would enhance the value of the instruments' indications and facilitate their interpretation. I am supplementing discussion at the meeting with a letter to each member of the Committee urging further that they make it a practice to submit to the N. A. C. A. a report of any actual experience with possible structural implications that any of their pilots may encounter, such as an unusually violent bump, or an apparent rolling, pitching, or yawing moment of extraordinary intensity due to atmospheric disturbances. If, in all such cases, proper records can be secured of the attendant conditions, it will put in hand a powerful tool for correlating structural effects with identifiable meteorological phenomena and also for determining the degree in which the worst conditions that happen to have been recorded in a somewhat limited amount of recording can, in fact be taken as representative of the worst that need ever be anticipated in design."

Perhaps the Weather Bureau could assist in this by suggesting that V-G recorders be installed on the airplanes taking upper air soundings. We would even undertake to change the glass slides once a day if Mr. Rhode would be interested in getting individual records for each sounding for use in correlating it with the meteorological data.

"There is another matter that touches both on meteorology and on aerodynamics—the question of ice formation and its effects. I am sending you a separate memorandum on the general subject of the ice problem, and the possible organization of research thereon, but there are some obvious aspects that are almost purely

aerodynamic. One useful job for the full-scale tunnel would be a study of the effect of de-icers, both in their deflated and in their operating condition, on the drag and the stalling characteristics of wings. Another would be an investigation of the temperature drop of air flowing around a wing. Where moisture in the air forms as ice on an aerodynamic surface, it is presumably the result of a change of temperature on contact with that surface. If there is to be any possibility of using heat effectively in the prevention of ice, data are needed on the exact conditions under which heat ought to be used and the exact amount that ought to be applied. Even if heat is considered out of the question, the investigation of temperature changes would still be worthwhile as providing a basis for determining the probable locations of ready ice deposit. Experience indicates that certain types of control surfaces, for example, accumulate ice, while others do not. It suggests the need for particular care in the design of slotted controls, the more especially as de-icers can hardly be used in a slot. While of course much of this work can only be done in a refrigerated tunnel where practical tests of actual ice deposit can be made, there is more that can be done in any kind of a flowing stream if ice formation can be redefined in more general terms as a thermodynamic problem."

Considerable icing data are being accumulated by the Weather Bureau in its program of upper air soundings. We have in mind tabulating and correlating the data with the meteorological records at a later date. The work can probably be started next year as soon as other pressing projects are out of the way. It will then be possible for Mr. Samuels to bring up to date his work on the meteorological aspects of the icing problems.

"Another meteorological problem on which the transport operator badly needs light is the range of variation of wind conditions in the upper air. While the Weather Bureau is badly handicapped by having to depend on visual observation of its balloons, it is my understanding that more data exist than have ever been put into a usable form, and that the funds for personnel for interpreting and qualifying them have been lacking. I hope that the Aerodynamics Committee or its meteorological Subcommittee can take an active interest in sponsoring that work, and do whatever lies within its power to encourage its active prosecution."

A detailed summary of all pilot balloon observations (Army, Navy and Weather Bureau) made in the United States previous to 1923 was published in the Monthly Weather Review Supplement No. 26. This summary was based on levels above the surface and confined to the area east of the Rocky Mountains. Another summary showing wind rises, resultant winds and average velocities for individual stations by months and for the levels 750, 1500, 3000, and 4000 meters above sea level is being prepared and the first section covering the eastern third of the United States has been published in Monthly Weather Review Supplement No. 35. The data for the remainder of the country will be ready for publication in about six months. A third summary of winds at high levels (6, 8, 10, 12, and 14 kilometers) by seasons and for individual stations is now being prepared and will probably be completed in about three months. Considerable work has also been done in tabulating the frequency of different upper air wind velocities, grouping them by directions and the velocity limits as recommended by the International Commission for Air Navigation. It will require about one year to prepare these data for publication. Upper air wind data have been tabulated for all ocean areas with a view to publishing a summary of winds over the oceans and for island and coastal stations. A great amount of work is yet to be done on this project, however, and without outside assistance, it will probably require two or three years to complete it. It is proposed also, when time permits, to compute and publish in great detail, upper air wind data for the stations having the longest records. Another study which should be undertaken is the grouping of upper air winds along with other meteorological elements according to different air masses.

Document 3-27(f), Edward P. Warner to Dr. J. S. Ames, National Advisory Committee for Aeronautics, April 18, 1936.

CONFIDENTIAL
COPY
ONE HUNDRED TWENTY-TWO EAST FORTY-SECOND STREET
NEW YORK CITY

April 18, 1936

Dr. J. S. Ames,
National Advisory Committee for Aeronautics,
Navy Building,
Washington, D. C.

Dear Dr. Ames:

Among the subjects considered at the recent meeting of the special Sub-committee on Problems of Transport Construction & Operation were one or two which I believe might be of interest to the Committee on Structures & Materials, and might provide it with some suggestions for lines of work which would have a special value for air transport. Though the special sub-committee took no formal action to the extent of requesting specific researches by resolution, I am taking the initiative as its chairman in rendering a report of the apparent sense of the meeting, and in including also some points of which lack of time forbade any discussion at the meeting, but of the significance of which I am persuaded by personal conference outside the meeting with members of the group who were in attendance and with others active in the transport field.

Distinctly the most important of questions is that of load factor due to atmospheric disturbances. It has been our practice up to the present time to assume a constant factor, based on an assumed constancy of vertical gust. With the development of very large airplanes and of airplanes having a large part of their weight distributed along the wings, the simple assumptions of the past become less tenable. The possibility of change of stress or the introduction of new types of loading through the action of gusts having linear dimensions small compared with those of the airplane becomes important. In this connection it is important also that we have some direct appreciation of the actual extent to which the effective strength of an airplane is increased by the suddenness with which the gust load is applied. I am not speaking now of the dynamic overloading factor but of its converse, the well-known ability of structures to stand a larger load very quickly applied and very promptly removed than they can sustain under continued applications. The more complex the internal organization of the structure and the larger the part played by internal friction during deformation, the more important that factor becomes. At the present time structures are conventionally designed for a factor of safety of 1.5 and for a gust velocity of 30 feet per second. We have evidence that gusts approaching 50 feet per second have actually been encountered at times, yet I am personally cognizant of no single case of apparent loading beyond the yield point in a modern monocoque structure. The reconciliation of these apparently conflicting facts ought to be of marked interest to the Structures Committee.

Of marked interest also is the question of the structure of fuselages carrying internal pressure. I do not propose that for any specific investigation under the Committee's auspices at the present time, for I understand that the Army has it in hand and is carrying forward as rapidly as possible. Nevertheless it might be a subject that the Committee would find interest in discussing, and upon which some cooperative grouping of researches by the various governmental agencies may in due course be desirable.

The matter of structural loading in gusts of course involves structures and aerodynamics jointly. I have already suggested its reference, from a slightly different point of view, to the Aerodynamics Committee.

Sincerely,
Edward P. Warner

Document 3-27(g), C. W. Lewis, Director of Aeronautical Research, to Mr. L.V. Kerber, Chief, Aircraft Airworthiness Section, Bureau of Air Commerce, February 26, 1938.

February 26, 1938

Mr. L.V. Kerber
Chief, Aircraft Airworthiness Section,
Bureau of Air Commerce,
Department of Commerce,
Washington, D.C.

Dear Mr. Kerber:

I have referred to our laboratory for the consideration of Mr. R. V. Rhode your letter of February 9, 1938, inquiring as to the extent to which the values of gust-load factors for airplanes of high wing loading recommended in his recent report have been confirmed by data obtained from N.A.C.A. V-G recorders installed in commercial airplanes.

Mr. Rhode states in reply that the factors recommended in the report cannot be said to be confirmed by evidence from the V-G records, nor can they be said to be not confirmed. Unfortunately, variations in the maximum effective gust velocities determined from the V-G records depend to so great an extent upon operating policies of the air lines routes flown, the number of hours of flying time recorded on each airplane type, and unreliability and inconstancy of the weather, that the effects of airplane variables are completely masked.

It is the Committee's viewpoint that the program of the collection of records from N.A.C.A. V-G recorders provides a general view of gust-load conditions under which the various transport airplanes operate. We believe it is important to continue to install the recorders in airplanes of new types, especially those that differ appreciably from previous or current types in important respects such as wing loading or size, but we have not yet obtained data as a result of these records that could be called comparable in respect to the external variables. It is therefore the opinion of the Committee that theoretical and gust-tunnel investigations are needed to supplement the information obtained on gust loads by analysis of V-G records, so that the effects of such variables as wing loading, gust gradient, stability, and other factors may be determined.

With such a procedure the V-G data will ultimately provide a basis for the establishment of a datum or normal level of conditions and the investigations conducted at the Committee's laboratory will provide the information required for a rational application of gust-load criteria to different airplane types. Mr. Rhode's report is therefore considered to be a preliminary step toward this objective. The material in this report may be said to be confirmed by results obtained in the Committee's gust tunnel to the extent that such results apply, but, as indicated above, it cannot be said to be confirmed by the V-G data, for the reasons stated.

Sincerely yours,

C. W. Lewis

Director of Aeronautical Research

Document 3-27(h), Interview of Richard V. Rhode by Michael D. Keller, January 10, 1967.

January 10, 1967
RVR: meh

1. Q. What, generally, was the sequence of events which led to the development of the VG and the VGH recorders?

A. The need for obtaining measurements of gust load factors on airplanes in actual service operations was recognized by NACA during the late 1920's. The earliest of measurements were made with a modified version of a commercially available instrument known as a Tetco TIM recorder. I believe the initials TIM stood for Time in Motion, because the instrument was normally used on vehicles such as trucks to record the periods during which the truck was idle or operating. A stylus, which reacted to vibrations or jarring of the vehicle, impressed a record of its motions on a circular waxed paper disc, which had a 24-hour time scale on it. The NACA converted this instrument to an accelerometer of suitable sensitivity, and, when properly installed near the center of gravity of an airplane, the instrument would register accelerations of the machine normal to the plane of the wings. It was calibrated to read acceleration in g units.

Although the converted Tetco TIM Recorder served to give some idea of the accelerations or load factors during service operations, it was not very accurate and data on acceleration alone were not enough to describe the structural loading situations of importance. Measurements of airspeed were also required. In maneuvering flight, for example, a given acceleration occurring at low speed is associated with moderate or high angle of attack of the wing, whereas the same acceleration occurring at high speed is associated with a low angle of attack of the wing. Since the pressure or load distribution on the wing is normally greatly affected by angle of attack, it was of great importance that airspeed and acceleration be simultaneously measured to permit interpretation of the records in terms of angle of attack or load distribution.

The importance of this aspect of aircraft structural loading was greatly illuminated as a result of analyses of acceleration and airspeed records taken by the NACA on an open time scale during dive tests of the U.S. Navy's first dive bomber, the XBM-1, a biplane built by the Glenn L. Martin Company in Baltimore. In analyzing the records from these tests, Mr. Eugene E. Lundquist plotted the data on a chart in which the scale of ordinates was acceleration normal to the wing chord and the scale of abscissas was indicated airspeed. Parabolic lines, each representing a condition of constant lift coefficient (or pressure distribution) could be drawn in on

this chart. With such a chart, the loading situation could be seen at a glance.

Because of these qualities, the chart became of great importance as a means of representing the critical or design loading conditions for aircraft both civil and military airworthiness requirements.

In this same time period, it became evident from studies of gust loading on aircraft that simultaneous values of normal acceleration and indicated airspeed measured in flight could be converted to "effective gust velocities" when certain simple characteristics of the airplane were known and applied to the derivation. Such effective gust velocities could, within limits, be used to determine the loads on other airplanes having different characteristics and flying at other speeds than the one on which the measurements were made. There were, however, no instruments available at that time which could be used in service operations to measure simultaneous values of acceleration and airspeed. The NACA recording accelerometers and airspeed meters, which were commonly used in flight tests at the Langley Laboratory, employed optical systems and light beams to trace their records on photographic film. They were delicate instruments and required constant maintenance and frequent re-calibration. Moreover, the operating time period for one loading of the film drums was in all cases much too short for use in service operations. All in all, it was not practicable to employ instruments of this type, even if modified to permit long-period use, in service operations. The thought of collecting and analyzing "miles of records" in order to determine the critical loading conditions that occurred only rarely in service also acted as a strong deterrent to the development of an airspeed-acceleration recorder that would register against time with an open time scale.

The unique instrumentation requirements of the gust load problem, viz., (a) the determination of simultaneous values of airspeed and acceleration in rough air for conversion to "effective gust velocities;" (b) the necessity for measurements over long periods of time on many airplanes in service operations; and (c) the requirement for simple and rugged instrument characteristics, when considered in the light of the Lundquist acceleration-airspeed chart, led me to suggest that a simple instrument, which would register acceleration against airspeed, be devised for use in determining the desired gust-load measurements.

H.J.E. Reid, Engineer-in-Charge of the Langley Laboratory, and a skilled instrument designer and technician, undertook to devise the required instrument which he designed to employ a thin, flexible steel stylus to scratch a record on a small smoked-glass plate. The light pressure of the stylus against the smoked glass resulted in sufficiently low friction to reduce error to acceptable small limits. The original instrument has been modified to improve the damping means and the balancing of the linkages, but today's version is still the same size and shape as the original and employs the same basic principle. Mr. Reid and I jointly held the patent on this instrument.

The instrument was given the name VG recorder for brevity and euphony at my suggestion. "V" is the standard symbol for airspeed and "G" is the capitalized version of the standard symbol, g, for the acceleration of gravity.

The Lundquist acceleration-airspeed chart was, for the same reasons, called the VG diagram, although it later became more properly known as the V-n diagram, because "n" is the symbol for normal load factor.

Although the VG Recorder filled the great need for a simple practical instrument to measure critical loading conditions in aircraft operations, it was not capable of providing much desired detail respecting aircraft operating and loading conditions. A practical instrument capable of measuring the three quantities acceleration, airspeed, and altitude against time on a reasonably open time scale for long periods was still much to be desired.

At a later period, Mr. Philip Donely, who was then Chief of the Gust Loads Branch of the Aircraft Loads Division, pushed for perfection of an optical system and other qualities which would make the desired instrument practical and, working with Mr. Edmond Buckley (now AA/Tracking and Data Acquisition) and others of the Langley Instrument Research Division, finally succeeded in developing the fine instrument now known as the NASA VGH Recorder.

Both the VGH Recorder and the VG Recorder continue to be used as complementary devices for the measurement of aircraft load statistics.

2. Q. How did relations of the laboratory personnel with the people of Hampton change over the years? Why?

A. The laboratory personnel at Langley have always consisted in part of local people and in part of outsiders. In general, the professional employees have come from outside the community and I assume the question relates primarily to these persons and to the people of Hampton who were not employed or who had no friends or relations employed at the Laboratory.

There is no question in my mind that, in the earlier years of the NACA, the professional employees of the Langley Laboratory were, on the whole, not warmly welcomed into the community.

Document 3-28 (a-h)

(a) Floyd L. Thompson, Associate Aeronautical Engineer, to Engineer-in-Charge [Henry J. E. Reid], NACA Langley, "Preliminary study of control requirements for large transport airplanes," 14 July 1936, with attachment "Suggested Requirements for Flying Qualities of Large Multi-Engined Airplanes," copies in RA File No. 509, Langley Historical Archives, Hampton, VA.

(b) Edward P. Warner to Mr. Leighton W. Rogers, Aeronautical Chamber of Commerce, Shoreham Building, Washington, DC, 19 Nov. 1936, copy in Research Authorization (RA) file 509, LHA.

(c) William H. Hernnstein, Jr., Assistant Aeronautical Engineer, to Engineer-in-Charge, NACA Langley, "Visit of Mr. E. P. Warner to the Laboratory, December 3, 1936," 3 Dec. 1936, copy in RA file 509, LHA.

(d) Eastman N. Jacobs, Aeronautical Engineer, to Engineer-in-Charge, NACA Langley, "Mr. E. P. Warner's visit to the Laboratory to discuss airplane tests—December 3, 1936," undated, copy in RA file 509, LHA.

(e) Hartley A. Soulé, Assistant Aeronautical Engineer, to Engineer-in-Charge, NACA Langley, "Visit of Mr. Warner to flight section on June 14, 1937," 15 June 1937, copy in RA file 509, LHA.

(f) Floyd L. Thompson, Associate Aeronautical Engineer, to Engineer-in-Charge, NACA Langley, "Tests of flying qualities

of Martin M-10B airplane," 14 July 1937, copy in RA file 509, LHA.

(g) Edward P. Warner, 11 West 42nd Street, New York City, to Dr. George W. Lewis, NACA, Navy Building, Washington, DC. 28 Dec. 1937, copy in RA file 509, LHA.

(h) Robert R. Gilruth, "Requirements for Satisfactory Flying Qualities of Airplanes," NACA Advanced Confidential Report, Apr. 1941.

For a thorough analysis of the torturous 25-year path leading to flying-quality specifications for American aircraft, one must read Walter Vincenti's chapter, "Establishment of Design Requirements: Flying Quality Specifications for American Aircraft, 1915-1943," in *What Engineers Know and How They Know It: Analytical Studies from Aeronautical History* (Baltimore and London: The Johns Hopkins University Press, 1990).

Below, readers will find several of the documents on which Vincenti based his analysis of the NACA's initiation of a systematic research program into flying qualities, which the government organization began from 1936 to 1937 in response to the request for help from Douglas consultant Edward P. Warner on the DC-4.

A few of the major figures in the history of NACA/NASA research appear as authors in this document string. Although hardly a household name, Hartley A. Soulé (born 1905) ranks as one of the great aeronautical minds within the NACA. A New York University engineering graduate who came to Langley in 1926, Soulé made significant research contributions in the field of stability and control. In the late 1940s and 1950s, he served as the leader of the NACA's Research Airplane Project, responsible for high-speed experimental airplanes like the Bell X-1 and Douglas D-558. He went on to major management responsibilities with the Mercury program.

A much better known figure is Eastman N. Jacobs (born 1902), a University of California-Berkeley graduate who came to work at NACA Langley in 1925. As readers will see in the next chapter, Jacobs played *the* central role in NACA airfoil development in the 1930s and early 1940s, until his abrupt (and rather mysterious) departure from the lab (and from the American aeronautics community almost entirely) late in World War II.

Floyd L. Thompson (born 1898), who came to work at Langley in 1926, served as director of NASA Langley Research Center from 1960 to 1968. Among the many roles he performed during the venturesome early years of the U.S. manned space program, Thompson chaired the committee that looked into the 1967 space capsule fire on the ground at Cape Canaveral, which killed three astronauts.

Although not nearly as prominent, William H. Herrnstein (born 1905) was another productive researcher with the NACA. After graduating from the University of Michigan's aeronautical engineering program in 1927, he came to work at Langley, serving in Fred Weick's PRT section. In this capacity, he became deeply involved in the PRT's low-drag cowling and engine nacelle placement tests, among other investigations. He worked at Langley into the NASA period.

A University of Minnesota aeronautical engineering graduate, Robert R. Gilruth (born 1913) took over the NACA's flying qualities research program when Soulé moved to wind tunnel duties almost immediately after arriving at Langley from Minnesota in 1937. (Vincenti's chapter discusses Gilruth's role at length.) Gilruth went on to become the head of the NACA's Pilotless Aircraft Research Division at Wallops Island, Virginia, the head of Project Mercury, and the first director of the Manned Spacecraft Center (later Johnson Space Center) in Houston, Texas.

Gilruth's April 1941 NACA report on the "Requirements for Satisfactory Flying Qualities of Airplanes," the last document in this string, is, without question, one of the most impressive papers in all NACA literature. As Vincenti noted in his analysis of the report, "To arrive at his results, Gilruth reviewed the mass of flight data and pilot opinion to see what measured characteristics proved significant; he also considered what was reasonable to require of an airplane and its designers." Although originally a classified report, in 1943, the NACA reissued it as *Technical Report No. 755*. "As the first generally accepted set of flying-quality requirements," the work quickly became an "oft-cited milestone in its field" (Vincenti, "Establishment of Design Requirements," p. 92). In an interview conducted with Gilruth 45 years later, in 1986, Gilruth looked back at what he had done: "I boiled that thing down to a set of requirements that were very straightforward and very simple to interpret. [The] requirements went right back to those things that you could design for" (Interview of Robert R. Gilruth by James R. Hansen, Kilmarnock, VA, 10 July 1986, copy of transcript in LHA). The requirements that Gilruth laid out in his report gave "comprehensive coverage" to longitudinal stability and control, lateral stability and control, and stalling characteristics. Within each, there were numerous subdivisions and each subdivision "stated a requirement, usually in a number of parts, gave a simple explanation of the reasons for it, and briefly discussed the design considerations necessary to its achievement" (Vincenti, p. 93). Aircraft designers were thus supplied with clear and precise quantitative limits for all the basic "flying qualities," ones based on extensive testing. To many in the airplane industry at the time, it seemed remarkable that "the same requirements could be applied (with minor exceptions) to all current types and sizes of airplanes," but that was what Gilruth's NACA work provided. Before long, such requirements became something "taken for granted" (Vincenti, p. 92), a normal part of the new design paradigm generated by the airplane design revolution of the interwar period.

For an even fuller treatment of the contents of Gilruth's classic report, see the entirety of the Vincenti chapter.

One cannot leave the topic of Gilruth's report without making a final observation. It is tempting to see much of the basis for Gilruth's later success as head of NASA's Mercury program already reflected in his brilliant laying out, two decades earlier, of satisfactory flying qualities for American aircraft. Historians and others have given too little credit to previous aeronautical experience, and previous NACA research experience, for the early successes of NASA, particularly in the manned space program.

Document 3-28(a), Floyd L. Thompson, Associate Aeronautical Engineer, to Engineer-in-Charge [Henry J. E. Reid], NACA Langley, "Preliminary study of control requirements for large transport airplanes," 14 July 1936.

Langley Field, Va.,
July 14, 1936.

MEMORANDUM For Engineer-in-Charge.

Subject: Preliminary study of control requirements for large transport airplanes.

Reference: (a) NACA Let. Jan. 14, 1936, CW:MW.
(b) LMAL Let. May 4, 1936, FEW.T.

1. Some time ago, the Honorable Edward P. Warner recommended that the N.A.C.A. undertake a study of the control requirements of large transport airplanes, which was authorized in letter of reference (a). In reference (b) an outline of the work to be undertaken in this research was given. The study indicated in section (a) of that letter has been competed and the results are attached hereto.
2. The first part of these results is entitled, "Suggested Requirements for Flying Qualities of Large Multi-Engine Airplanes." These requirements are set down in forms such as one would use in preparing specifications for the flying qualities. They are similar in some respects to the suggested specifications prepared by Mr. Warner and submitted to Mr. Weick for comment, as noted in a letter from Mr. Warner to Mr. Weick December 22, 1935. An attempt to write down the requirements in a detailed form such as this is beneficial, in that it helps to crystallize ideas regarding what items are important and indicates wherein data are lacking concerning quantitative values for various items.
3. The second part of the results is entitled, "General Program of Tests of Airplane Flying Qualities." This program indicates the type of tests required to obtain the various items specified in the previous section on requirements, and also indicates the instruments required.
4. The next step in the investigation will be to carry out section (b) of letter of reference (b). This work will consist of a trial of the methods outlined in the program on one or more of the airplanes available at the laboratory. It is expected that a Stinson Reliant will be available for the tests some time after the first of August. In the meantime, attention will be given to the design of any special instrument required.

Floyd L. Thompson,
Associate Aeronautical Engineer.

SUGGESTED REQUIREMENTS FOR FLYING QUALITIES OF LARGE MULTI-ENGINED AIRPLANES

Note: The chief purpose of these suggested requirements is to show what items are believed to be important. Although numerical limits have been stated, they are, for the most part, considered to be quantitatively unreliable, owing to the present lack of data concerning what constitutes satisfactory flying qualities.

I. Longitudinal stability and control characteristics:
A. Longitudinal stability:
With elevator free the airplane shall be dynamically stable throughout the speed range for all loading conditions. The period of the longitudinal oscillations should never be less than 40 seconds and the damping should be sufficient to reduce the amplitude of the oscillation to one-fifth the original amplitude in four cycles.

1. Longitudinal stability is required for the following conditions:

   a. Flap up–full power–entire range of trimming speeds.

   b. Flap up–level flight–entire range of trimming speeds.

   c. Flap up—power off—entire range of trimming speeds.

   d. Flap up–any three engines operating or the outboard engine on one side and the inboard engine on the other side cut out. Entire range of trimming speeds with aileron and rudder trimming tabs adjusted to give lateral balance.

e. The same conditions should be met with the flap down except that the high speed shall be limited to the placarded speed.

B. Elevator control:

1. Range of elevator control:

The range of elevator control should be sufficient so the following conditions may be met:

a. It shall be possible to maintain steady flight at any speed from the designed probable diving speed to the minimum speed. This condition shall be met with any loading condition and with any power condition with the flap up.

b. With the flap down it shall be possible to maintain steady flight at any speed from the placarded speed to the minimum with any loading and any power condition.

c. If the conventional landing-gear arrangement is used it shall be possible to land the airplane power off with the flaps either up or down with the loading condition that gives the most forward position of the center of gravity in a three-point attitude without aerodynamic bouncing. It shall be possible for the same loading condition to hold the tail wheel down while braking vigorously enough to give a deceleration of 0.3 g during the landing run down to a speed of 30 m.p.h. With the loading condition that gives the most rearward position of the center of gravity it shall be possible to raise the tail wheel off the ground in the take-off run by the time a speed of 30 m.p.h. is attained.

d. If a tricycle landing-gear arrangement is used it shall be possible with the loading condition that gives the most forward position of the center of gravity to raise the nose wheel off the ground in the take-off run by the time a speed of 30 m.p.h. is attained.

2. Variation of elevator angle with speed:

a. The curve of equilibrium elevator angle against speed for every setting of the trimming tab both power on and power off for all loading conditions and with the flaps up and down shall be smooth and shall everywhere have a positive slope.

3. Range of elevator control forces:

With every setting of the trimming tabs, with every loading condition either power off or power on, and with the flaps up and down, it shall be possible to fly the airplane within the ranges given for the elevator control in the previous section with a range of stick forces of no greater than 100 pounds.

4. Variation of elevator force with speed:

a. A curve of stick force required for steady flight plotted against the speed of flight for every tab setting and every loading condition with either power on or off or flaps up or down shall be smooth in form without discontinuity or sudden changes of curvature. Its slope shall be everywhere negative throughout the specified speed ranges and shall nowhere be less than ¼ pound per mile per hour.

5. Variation of elevator forces with throttle setting:

a. The force required on the stick to overcome, without change of tab setting or speed, the effect of any change in engine operating condition from full power to fully throttled shall not exceed 100 pounds.

6. Elevator trimming tabs:

a. It shall be possible to trim the airplane at a low enough speed so that no greater force than a 30-pound pull will be required in meeting the landing requirements previously given.

b. It shall be possible to trim the airplane at its maximum speed.

7. Effectiveness of elevator control:

a. As an indication of the effectiveness of the elevator for maneuvering the airplane, it shall be possible to obtain an acceleration of 0.8 of the design load factor at any speed with the elevator alone with the airplane originally trimmed for cruising speed without applying a force of more than 200 pounds and not less than 60 pounds to the control stick.

b. As low speeds down to 10 miles per hour above the minimum where the theoretical maximum acceleration approaches 1, it shall be possible to change the attitude of the airplane in space with respect to its transverse axis in either direction by 5° in 1-½ seconds through use of the elevator alone.

II. Lateral stability and control characteristics:

A. Lateral stability:

The airplane shall be laterally stable for the same conditions for which longitudinal stability is required. The period of the lateral oscillations should not be less than 20 seconds and the damping should be sufficient to reduce the amplitude of the oscillation to one-half of the original amplitude in two cycles.

Between the minimum trimming speed and the minimum speed the airplane shall show no negative dihedral effect nor any autorotative tendencies.

From the minimum speed to the limit of the elevator control there shall be no sudden development of marked autorotative tendencies nor any sudden change in the lateral stability characteristics.

1. Limits for roll due to sideslip (dihedral effect);

a. The dihedral effect should be sufficient so that when the ailerons are freed immediately after putting the machine into a 15° bank, and using the rudder to avoid a change of heading, the angle of bank shall be reduced to 2° within 15 seconds and with the loss of altitude of not over 300 feet. This condition shall be met for all conditions for which longitudinal stability is required.

b. The rolling acceleration accompanying abrupt rudder displacement shall be less than one-half of the yawing acceleration.

B. Aileron control:

1. Aileron power:

The aileron power shall be sufficient so that the following conditions can be met:

a. At a speed of 70 miles per hour with flaps down or 80 miles per hour with flaps up, it shall be possible to bank the airplane 15° in 2-½ seconds with the ailerons alone, and at 120 miles per hour or higher the same

angle of bank shall be obtained in 2 seconds.

b. At a speed 2 miles per hour above the stall with the flaps down it shall be possible to bank the airplane 10° in 2 seconds with the ailerons alone.

c. The aileron effectiveness shall be proportional to the aileron deflection.

2. Aileron forces:

a. The force required to obtain the aileron reactions given should not exceed 50 pounds applied tangentially at the rim of the wheel.

b. The aileron force shall be approximately proportional to the aileron deflection.

3. Yaw due to ailerons:

a. At speeds above 10 percent in excess of the minimum speed the ailerons should not produce a yawing acceleration greater than 0.1 the acceleration in roll. At speeds below 10 percent in excess of the minimum the acceleration in yaw should be less than one-fifth of the acceleration in roll.

4. Aileron trimming tabs:

The range of power of the aileron trimming tabs should be sufficient so that the following conditions may be met:

a. It shall be possible by the use of the tabs to balance the airplane against the dissymmetry of loading corresponding to full gas tanks on one side of the center of gravity and empty gas tanks on the other.

b. It shall be possible by the use of the tabs to compensate for any rolling tendency accompanying steady flight with asymmetrical power and loading conditions.

C. Rudder control:

1. Rudder power:
The power of the rudder control should be sufficient so the following conditions may be met:

a. It shall be possible during steady flight at 70 miles per hour with the flaps down or 80 miles per hour with the flaps up, as well as at any higher speed, to produce a change of heading of 15° in 3 seconds by use of the rudder alone. At 120 miles per hour the same change shall be possible in 2 seconds.

b. At 2 miles per hour above the stall with flaps down it shall be possible to make flat turns up to a change of heading of 10° in 2 seconds.

c. At 20 miles per hour above the minimum speed as well as at any higher speed it shall be possible to hold a straight course with the wings laterally level with both engines on either side cut out and those on the other side operating at full rated power.

d. With any three engines operating or one outboard and the opposite inboard one cut out, it shall be possible to hold a straight course on the ground down to 50 miles per hour with the flaps either up or down.

e. The rudder effectiveness shall be proportional to the rudder deflection.

2. Rudder forces:

a. It shall be possible to obtain the rudder reactions given without applying a force greater than 180 pounds on the rudder pedals.

b. The rudder force shall be proportional to the rudder deflection.

3. Rudder trimming tabs:

The range of power of the rudder trimming tabs should be sufficient so the following conditions may be met:

a. Down to 10 percent in excess of the minimum speed, it shall be possible by adjustment of the trimming tabs to fly straight with any three engines operating or one inboard and the opposite one outboard cut out with no force on the rudder pedals.

b. Down to 20 miles per hour above the minimum speed with the flaps down, it shall be possible with the trimming tabs to reduce the force on the rudder pedals required for straight flight with both engines on either side cut out and those on the opposite side operating with full rated power to 30 pounds.

D. Combined operation of rudder and ailerons:

1. It shall be possible to enter a 45° banked turn at 140 miles per hour in 5 seconds without the rudder force exceeding 100 pounds or the aileron force exceeding 75 pounds. The same limitations on forces which apply to a 30° banked turn at 200 miles per hour entered in 4 seconds. It shall be possible to make normally banked turns up to a 15° bank at speeds within 5 miles per hour of the minimum with the flaps either up or down. It shall be possible with the flaps either up or down to fly the airplane steadily for at least 10 seconds up to the limit of the elevator control if the elevator control is sufficient to stall the airplane. It shall be possible at speeds above 10 percent in excess of the minimum to maintain a steady sideslip with an angle of bank of 20°.

GENERAL PROGRAM OF TESTS OF AIRPLANE FLYING QUALITIES

Longitudinal Stability and Control

Longitudinal Stability – (Section I-A)*

Item — Period and damping of oscillations.

Procedure — Elevator free:— Trim airplane for desired speed and then push stick forward until airplane gains 5 to 10 miles per hour. Release the stick and record the ensuing oscillations as the airplane returns to a steady state at the original trimming speed.

Elevator fixed: — Repeat tests as before except that after setting up a disturbance the stick is returned to the original setting or neutral point and held during the ensuing oscillations.

Observations or records — If recording instruments are used, it is only necessary for the pilot to start the instruments at the start of the oscillations and to turn them off when steady flight is reestablished. If indicating instruments are used, it is necessary to observe and record the air speed at successive peaks of the oscillations and to determine the time between successive peaks for four oscillations.

* The reference in brackets and similar references throughout this program apply to "Suggested Specifications for Flying Qualities of Large Multi-Engined Airplanes." This referencing system is employed to show the correlation between the flight program and the items to be specified.

Document 3-28(b), Edward P. Warner to Mr. Leighton W. Rogers, Aeronautical Chamber of Commerce, Shoreham Building, Washington, DC, 19 Nov. 1936, copy in Research Authorization (RA) file 509, LHA.

November 19, 1936.

Mr. Leighton W. Rogers
Aeronautical Chamber of Commerce
Shoreham Building
Washington, D.C.

Dear Mr. Rogers:

I have just heard from George Lewis that Mr. Clayton visited Langley Field on November 13th to discuss the test work being done by the National Advisory Committee for Aeronautics that bears on airplane design requirements, and that you are planning a meeting of your own people on December 3rd to discuss some of these matters.

My special purpose in writing is to say that I expect to be in the east myself by December 1st and stay for a couple of weeks thereafter, and that I should very much like to attend the meeting of the 3rd, if an invitation for me to be present would be consonant with the meeting's purposes. I am, as you know, very deeply interested in all questions bearing on transport design and regulations; I have been particularly concerned with promoting testing of the flight qualities of commercial airplanes at Langley Field; in connection with the development of the DC-4, I have lived very close to all manner of questions of specifications and regulation; and I was designated as chairman of the special National Advisory Committee for Aeronautics subcommittee on the aerodynamic problems of transport operations (which, despite its title, has not by any means been limited to aerodynamics in its considerations).

I shall probably be staying here through next Monday or thereabouts. If you want to reach me after that, I would suggest that I be addressed at the Harvard Club, 27 West 44th Street, New York City.

Sincerely,

Edward P. Warner.

Document 3-28(c), William H. Hernnstein, Jr., Assistant Aeronautical Engineer, to Engineer-in-Charge, NACA Langley, "Visit of Mr. E. P. Warner to the Laboratory," December 3, 1936.

Langley Field, Va.
December 3, 1936.

MEMORANDUM For Engineer-in-Charge.

Subject: Visit of Mr. E. P. Warner to the Laboratory December 3, 1936.

1. Mr. Warner visited the Laboratory on the morning of December 3 primarily for the purpose of discussing the program of measuring flying qualities of airplane and the progress we have made to date on the flight tests contemplated with the Committee's Stinson in connection with this program.

2. A meeting was held in Mr. Reid's office which was attended by Mr. Warner and Messers. Reid, DeFrance, Crowley, Jacobs, Thompson, Soule, and Herrnstein of the Laboratory staff. Mr. Warner suggested that airfoils of 30 to 40 percent thickness be tested for possible future use on transport airplanes of high wing loading and high aspect ratios.

3. Mr. Crowley stated that the Stinson was being prepared for its part in the program of measuring flying qualities of airplanes to check out the method to be used when we receive various airplanes for flight tests. He stated that the instrument installation would call for some special instruments but that many of the standard instruments in the airplane would be used and the readings photographically recorded. Instruments to be added to the airplane would include control-position and control-force recorders, two turnmeters, an accelerometer, and an air-speed recorder.

4. Mr. Warner inquired as to how long it would take to make an instrument installation in any airplane furnished. Mr. Thompson thought it would take two days at the most and probably less time. He added, however, that a different control force recorder would have to be used than that which is now being used in the Stinson as it takes too long to install it.

5. Mr. Warner wanted to know how long a pilot could sustain forces on the controls. He was already familiar with our work done on the maximum force a pilot can exert on the controls. Mr. Crowley replied that such measurements had been made at the Laboratory and also added that the pilots complained of aileron forces on the stick of 10 pounds as being excessively heavy, whereas they can exert a maximum force of about 30 pounds. Mr. Warner asked how the maximum wheel forces a pilot can exert check up with his specifications on this subject. No one knew exactly but the data are all on record.

6. Mr. Crowley stated that with the artificial horizon the flight section hoped to be able to discard most of the recording instruments provided the horizon proved to give accurate enough results. He also added that the Stinson would probably be flying next week. One difficulty with using the Committee's Stinson as a guide in the measurement of flying qualities of larger airplanes to be tested later is that the machine rolls very rapidly as compared to larger airplanes.

7. Mr. Warner stated that very complete wind-tunnel tests made by the Douglas Company showed that they could meet any of his specifications easily with the exception of control on the ground in take-off with unsymmetrical power conditions. The tests showed that about two times ordinary rudder size would be needed to meet this specification. Mr. Warner also stated that he thought it a common error to overestimate the safety of twin-engine and 4-engine airplanes. In most cases the twin-engine airplane, which is the worst, needed about 90 miles per hour forward speed to enable it to be held on straight course after one engine was cut out.

8. Mr. Warner inquired if the control-position recorder to be used would be mounted in the cockpit. Mr. Crowley answered that it would.

9. In answer to a question by Mr. Warner, Mr. Soule answered that no flight routine had yet been settled for these tests. Such a routine would depend upon test results obtained with the Stinson and further work. It was agreed that this work should have a broad base at the start particularly.

10. In regard to control at the stall and warning of the stall, Mr. Warner stated that the Douglas DC-4 was specified to have the lift spoiled at the center section so as to maintain lateral control on the tapered wing, and that any buffeting resulting from this was to be felt through the controls only and not by the passengers through the structure.

11. Mr. Jacobs expressed concern about the ability of manufacturers to meet lateral control specifications when partial span flaps were to be used on tapered wings. He was afraid that the tips of these wings would always stall first due to the flaps.

12. Regarding torsional stiffness of the wings, Mr. Warner stated that the big problem now was to get a lateral control that would not subject the wing to torsion. He said that the devices such as slot-lip ailerons were out of the picture at present because of icing difficulties. Regarding the use of heat for the prevention of ice formation, Mr. Warner observed that ice would form on heated surfaces when the outside temperature is much below freezing when no ice would be present if heat were not applied.

13. Mr. Warner asked if in our studies of long-range flight, neglecting the wing factor, we had determined that it would be economical to fly at high altitudes. Mr. Jacobs replied that he was not sure but doubted if there was any advantage from the pure-range standpoint. He said that there would be an advantage in speed but it would not be as large as was commonly supposed.

14. Mr. Soule pointed out the difficulty in adapting the present control-force device to all types of wheels and sticks with which test airplanes will be supplied.

It was decided that the Laboratory would have to obtain advance information on the control columns, etc., of the airplanes to be tested. Mr. Warner said that there is nothing definite to date as to what airplane would be the first to undergo such tests.

15. Mr. Warner observed that in all the large future airplanes the control will probably be irreversible, doing away with any feel in the controls of the airplane.

16. Following this meeting Mr. Warner visited the individual activities of the Laboratory.

W. H. Herrnstein, Jr.,
Assistant Aeronautical Engineer.

Document 3-28(d), Eastman N. Jacobs, Aeronautical Engineer, to Engineer-in-Charge, NACA Langley, "Mr. E. P. Warner's visit to the Laboratory to discuss airplane tests—December 3, 1936."

Langley Field, Va.
Undated

MEMORANDUM For Engineer-in-Charge.

Subject: Mr. E. P. Warner's visit to the Laboratory to discuss airplane tests—December 3, 1936.
Reference: LMAL Let. (L.F.1) Dec. 3, 1936, IHD. EWM.

1. A conference between Mr. Warner and Messrs. Reid, Crowley, Thompson, Soule, and Jacobs was held mainly to answer questions of Mr. Warner about the proposed tests and the instruments developed for the work (covered in Mr. Herrnstein's memorandum of reference). Later, the instruments were inspected. Many related problems were discussed at length; for example, lateral control and high-lift devices, servo controls, ice formation, sub-stratosphere flying, etc. The main discussion was hampered by the fact that we had no copy of Mr. Warner's specifications for the new transport to refer to. Later, I had a chance to go over the specifications with Mr. Warner, the flight section group, and Mr. Gough, and also had an opportunity to discuss our airfoil work with Mr. Warner while I drove him down to Old Point to get the specifications. The part about stalling characteristics, the part in which I was particularly interested and the only part I had time to consider in detail, seemed to me essentially reasonable and definitely desirable.

2. As I understand it, however, the vital part of Mr. Warner's proposal as far as the Committee is concerned, is that we develop a short flight-check procedure for commercial airplanes. Mr. Warner is undoubtedly right in contending that every effort should be made to shorten the procedure to an extent that the airplane is

required for only about one week. On this basis, it would probably be possible to flight-check many new types. This object is of vital importance to the Laboratory, because a familiarity with new types will tend to get us out of the dark with regard to the practical effects of the application of new developments.

3. It is my personal opinion that the flight-check procedure might be improved and shortened by going over the subject carefully with the pilots so as to take full advantage of their qualitative observations during a few preliminary flights. I agree with the flight section personnel, however, that we will be in a better position to discuss the final procedure among ourselves and with Mr. Warner after the preliminary tests on the Stinson have been completed.

4. Mr. Warner offered suggestions that will be followed up on both our airfoil and our fuselage work. He wants us first to investigate further variations of airfoil thickness distribution, and second, the best practical maximum thickness on the simplifying assumption that a span to thickness ratio of the order of 60 must not be exceeded. He criticized the fairness of our fuselage models and suggested that we test fairer modifications. He also thought that we have not gone to sufficiently low fineness ratios on our fineness ratio series. I agreed that we should extend it if the tests show that we have not definitely gone beyond the optimum.

Eastman N. Jacobs,
Aeronautical Engineer

Document 3-28(e), Hartley A. Soulé, Assistant Aeronautical Engineer, to Engineer-in-Charge, NACA Langley, "Visit of Mr. Warner to flight section on June 14, 1937," 15 June 1937.

Langley Field, Va.,
June 15, 1937.

MEMORANDUM For Engineer-in-Charge.

Subject: Visit of Mr. Warner to flight section on June 14, 1937.

1. Mr. Warner visited the flight section on June 14 to discuss the section's work pertaining to the measurement of flying qualities. He was familiar with the status of the project and the more important results. His object on this visit was to inspect the results from the Stinson and B10-B (Martin bomber) airplanes in order to familiarize himself with the extent of the results, details of presentation, and the time required for the tests. He was also interested in the precision of the measurements and agreement of runs under supposedly identical conditions as an indication of the practical tolerances when specifying the various factors.

2. The inspection took about 3 hours. Mr. Thompson was present about half this time. During the discussion accompanying the inspection Mr. Warner made several comments and suggestions, the most important of which was that for pull-ups for large airplanes we should investigate the flight path for the representative cases to assure ourselves that the maximum normal acceleration is indicative of the attitude gained in the maneuver in a given time. He is interested in the delay between the control movement and the upward motion of the airplane. After going over all the figures Mr. Warner requested permission to look over the text of the memorandum on the Stinson tests. This was granted and he spent about a half hour reading the memorandum. He made note of further suggestions but as time was not available for more discussion, he intends to submit these in writing.

Hartley A. Soulé,
Assistant Aeronautical Engineer.

Document 3-28(f), Floyd L. Thompson, Associate Aeronautical Engineer, to Engineer-in-Charge, NACA Langley, "Tests of flying qualities of Martin M-10B airplane," 14 July 1937.

Langley Field, Va.
July 14, 1937.

MEMORANDUM For Engineer-in-Charge.

Subject: Tests of flying qualities of Martin B-10B airplane.

Reference: (a) NACA Let. Mar. 29, 1937, L.F.
(b) LMAL Let. Apr. 3, 1937, WHH. DLC.

1. The investigation of the flying qualities of the Martin B-10B airplane obtained on loan from the Army Air Corps in accordance with letter of reference (a) has been completed. The airplane made available for the tests was no. 34-34 and was one of those attached to the 20th Bombardment Squadron. It was made available to the Committee for tests on May 1 and was released to the Army on July 2. Throughout this period the airplane was housed in the Army hangar and was serviced by Army personnel, and at various times it was made available to Army personnel for flights, when such flights did not interfere with the test program. The actual test program required approximately 26 hours of flying, most of which was done during June. The installation of instruments and some preliminary flights were made early in May, but because of the Engineering Research Conference and because of the necessity of repairs and routine checking of the airplane, no tests were made during the latter part of May.

2. The B-10B airplane is the second one with which tests have been made in the investigation of flying qualities. The program followed was in essential agreement with that submitted with letter of reference (b), but with minor modifications suggested by experience gained in testing the single-engine Stinson monoplane, the first airplane used in this investigation.

3. In the original plan for this investigation it was intended to develop a method wherein the measurements could be obtained, as much as possible, by means of indicating instruments as well as by means of the recording instruments usually employed by the Committee. A certain amount of development along this line was followed in the test of the Stinson, but further development was handicapped with the B-10B airplane because of the arrangement of that machine. The use of the indicating instruments requires that an observer have access to the pilot's cockpit whereas in the B-10B airplane the observer does not have access to the pilot's cockpit and can only communicate with the pilot by means of the interphone system. It should be noted, however, that although in the tests of the Stinson the elevator and aileron forces were recorded, with the B-10B an indicating system was used, the observations being made by the pilot. Regardless of the system used for making measurements testing is greatly handicapped when the observer does not have access to the pilot's cockpit.

4. Equipment used in the tests performed satisfactorily with the exception of the control-force indicator. This instrument, as arranged at present, is suitable for the measurement of steady forces but maximum recording hands required in determining suddenly applied forces did not work satisfactorily. The instrument is now undergoing further development in order to make possible the determination of suddenly applied forces as well as steady forces.

5. With the B-10B airplane it is possible to vary the throttle settings, the propeller pitch, the flap position, and to raise or lower the landing gear. In addition, the center-of-gravity position is a variable. Thus, as for nearly all modern machines, there are a great number of possible conditions of the airplane to consider in arranging the test program, but after consideration the program was restricted to 5 conditions of the airplane representative of the following regimes of flight:

(1) High-speed flight.
(2) Climbing flight.
(3) Power-off flight.
(4) Take-off.
(5) Landing.

The complete series of tests was made with a center-of-gravity position corresponding approximately to the rearmost position recommended for the airplane. Because the airplane exhibited static longitudinal instability with this position of the center of gravity, a portion of the tests related to the longitudinal stability was repeated with the airplane loaded for the most forward center-of-gravity position.

6. The data for the tests have not been completely evaluated but some points of general interest can be stated. With the rearmost center-of-gravity position, the airplane with power on possessed longitudinal static instability but with the power off the airplane was stable. This instability was eliminated when the center of gravity was shifted to the most forward position. In the latter case, however, the trimming tab did not have sufficient power to balance the airplane within the permissible speed range with the flaps lowered. The airplane had very little effective dihedral, as was evidenced by the slow leveling off when a wing was dropped and slipping toward the lower wing developed. With one engine operating at full power and the other engine idling, the power of the rudder tabs was insufficient to balance the turning moment due to the asymmetric thrust at any speed. With the tab set to assist the pilot straight flight could be maintained down to a minimum speed of 90 miles per hour. At this speed the rudder force required was of the order of 160 pounds. A complete report on the results of these tests will be made as soon as possible.

7. As a result of the experience gained, first with the Stinson monoplane, and second with the Martin B-10B airplane, it is felt that the procedure has been fairly well perfected. Some further development of instruments and procedure will be required, but in general it is believed that from now on the major point of interest will be the actual results obtained, rather than the perfection of procedure. It is believed that in machines wherein the observer has access to the pilot's cockpit, the complete program can be carried out in approximately one month. Some advance notice, however, is required to permit the preparation of instruments; in particular, the construction of parts necessary to adapt the control-force recorder to the particular airplane involved. Drawings of the control wheel installation should be made available at least two weeks in advance of the delivery of the airplane for testing.

Floyd L. Thompson,
Associate Aeronautical Engineer.

Document 3-28(g), Edward P. Warner, 11 West 42nd Street, New York City, to Dr. George W. Lewis, NACA, Navy Building, Washington, DC, 28 Dec. 1937.

EDWARD P. WARNER
11 West 42nd Street
New York City

December 28, 1937.

Dr. George W. Lewis,
National Advisory Committee for Aeronautics,
Navy Building,
Washington, D. C.

Dear George:

As you know, the completion of the DC-4 is near at hand. Naturally it will be necessary to run a most exhaustive series of tests on a ship of such exceptional size and cost before deciding on a production policy; and as far as possible the factor of difference of personal opinion ought to be eliminated from the results of those tests.

Furthermore, from a scientific point of view it would in any case be desirable that we check with particular care on the flying qualities of this ship, in view of the attempt that it represents to put flying-quality specification on a quantitative basis.

For both of those reasons, I am very hopeful that it will be possible for the Advisory Committee to participate in the testing, and to provide for the use of your recording instruments and the techniques that you have developed in their use.

The tests will of course be made at Santa Monica; nor would it seem entirely practicable to take the machine to Langley Field for this specific purpose, at least until a number of months after its construction and after the first and most urgent need for instrumental test methods would have passed. Could it not then be arranged for one or two members of your staff, competent in the use of the flight recording instruments, to go to Santa Monica and remain there as part of the test crew for the duration of the test period?

While I do not definitely speak for any of the air lines, and have no authority to commit any of them, I should think that it would be possible to meet whatever expenses might be involved in such a transfer of personnel; but in any event I am sure that the results that you would get from such a participation would be of the greatest interest, and I have no doubt that the parties involved in the test would be glad to agree that a substantial proportion at least, if not the whole, should be considered available for early publication by the Committee.

I of course make this suggestion with less hesitation than I might otherwise feel because of the cooperative nature of the project. In participating to increase the value of the tests and to increase the accuracy of the immediate appraisal of the qualities of the ship, you would be rendering service not merely to a single air line, {AQ1}but substantially to the entire air transport industry; since the companies directly involved in the DC-4 purchase and presumably all equally anxious to get the most reliable information possible upon the ship, represent nearly 80% of the mileage flown and nearly 90% of the passenger traffic handled under the American flag.

In this same connection of flying-quality testing, which is obviously going to assume increasing importance as airplanes increase in size, it occurs to me that it ought to be possible at the same time to increase your background for the judgment of test results, and to render an immediate service to the Army and Navy, by applying the technics of measurement that you have been developing on the Stinson and the Martin to the Boeing XP-15, and to the new Sikorsky and Consolidated four-engined patrol boats.

Would it be appropriate to suggest to the Army and Navy personnel concerned the availability of your new procedure and its possible interest in connection with these large aircraft, or would it be better that I take the initiative by corresponding with the Service personnel? Personally I should think that you could better take it up directly, unless you have some fixed principle against seeming to promote the use of the Committee's services in a particular instance in advance of their being asked for.

I should think that another very useful application of the same methods would be on the flying model which Martin has been using recently to predict the behavior of the big boat for which he has just received a Navy order. Aside from being of considerable interest to Martin and to the Navy, in its greater delicacy of indication of existing flying qualities and of suggestion of the changes that perhaps should be made in the full-scale ship to ameliorate them, direct measurement of flying qualities in this instance would be of obviously immense scientific interest in establishing a specific quantitative correlation between the behavior of the flying model and that of the final machine, and in showing just how much scale effect there is in these matters in free flight.

I shall be very much interested in knowing what you think about all this. Please let me know if I can be of any assistance.

Sincerely,
Edward P. Warner

Document 3-28(h), Robert R. Gilruth, "Requirements for Satisfactory Flying Qualities of Airplanes," NACA Advanced Confidential Report, Apr. 1941.

REQUIREMENTS FOR SATISFACTORY FLYING QUALITIES OF AIRPLANES

By R. R. Gilruth
Langley Memorial Aeronautical Laboratory
April 1941

INTRODUCTION

The need for quantitative design criterions for describing those qualities of an airplane that make up satisfactory controllability, stability, and handling characteristics, has been realized for several years. Sometime ago, preliminary studies showed that adequate data for the formulation of these criterions were not available and that a large amount of preliminary work would have to be done in order to obtain the information necessary. It was apparent that flight tests of the flying qualities of numerous airplanes were required in order to provide a fund of quantitative data for correlation with pilots' opinions.

Accordingly, a program was instituted which covered the various phases of work required. The first step involved the development of a test procedure and test equipment which would measure the characteristics on which flying qualities depend. Thin phase of the work is reported in reference 1, although since that time the test procedure has been expanded and modified on the basis of additional experience and several changes have been made in the equipment used.

Another phase of the investigation has involved the measurement of the flying qualities of a number of airplanes. The procedure used has been in general in accord with that described in reference 1. At the present time, complete tests of this nature have been made of 16 airplanes of varied types. These airplanes were made available largely by the Army and more recently by private companies at the request of the Civil Aeronautics Board. In addition, flying-qualities data of more limited scope have been obtained from time to time on a number of other airplanes, the tests of which covered only particular items of stability and control but which, nevertheless, augment the fund of data now available.

A third phase of the investigation, one which also has been pursued throughout the duration of the project, has involved the analysis of available data to determine what measured characteristics were significant in defining satisfactory flying qualities, what characteristics it was reasonable to require of an airplane, and what influence the various design features had on the observed flying qualities.

In order to cover this work adequately, a number of papers dealing separately with the various items of stability and control are necessary. Several such papers have been prepared or are in preparation at the present time. Detailed studies of all items will require considerable time for completion, but it is believed that the conclusions reached to date are complete enough to warrant a revision of the tentative specifications set forth in reference 1. As opportunity for additional analysis occurs, it would be desirable to cover the individual requirements at more length than is possible at this time. As a result of further studies, it may also be desirable to revise again the flying-qualities specifications given here.

In addition to the actual specifications, the chief reasons behind the specifications are discussed. Wherever possible, interpretation of the specification is made in terms of the design features of the airplane unless the subject is covered in reports of reference.

In formulating the specifications, every attempt has been made to define the required characteristics in easily measurable, yet fundamental terms. It was necessary to consider all stability and control requirements in arriving at each individual item because of the varied functions of the individual controls and the conflicting nature of many of these functions.

The specifications require characteristics that have been demonstrated to be essential for reasonably safe and efficient operation of an airplane. They go as far toward requiring ideal characteristics as present design methods will permit. Compliance with the specifications should ensure satisfactory flying qualities on the basis of present standards, although as additional knowledge is obtained it may be possible to demand a closer approach to ideal characteristics without in any way penalizing the essential items of performance.

FLYING-QUALITY REQUIREMENTS

It has been convenient to present the flying-quality requirements under the following individual headings. They appear in the report in this order.

- I. Requirements for longitudinal stability and control:
  - A. Characteristics of uncontrolled longitudinal motion.
  - B. Characteristics of elevator control in steady flight.
  - C. Characteristics of elevator control in accelerated flight.
  - D. Characteristics of elevator control in landing.
  - E. Characteristics of elevator control in take-off.
  - F. Limits of trim change due to power and flaps.
  - G. Characteristics of longitudinal trimming device.
- II. Requirements for lateral stability and control:
  - A. Characteristics of uncontrolled lateral and directional motion.
  - B. Aileron-control characteristics.
  - C. Yaw due to ailerons.
  - D. Limits of rolling moment due to sideslip.
  - E. Rudder-control characteristics.
  - F. Yawing moment due to sideslip.
  - G. Cross-wind force characteristics.

H. Pitching moment due to sideslip.

I. Characteristics of rudder and aileron trimming devices.

III. Stalling characteristics.

These requirements pertain to all flight conditions in which the airplane may be flown in normal or emergency operation, with the center of gravity at any point within the placarded limits, some of the specifications are based on the behavior of the airplane at some specified airspeed. The airspeed in such cases shall be taken as the indicated airspeed. Where minimum airspeed is referred to, unless otherwise stated, it shall be taken as the minimum airspeed obtainable with flaps down, power off.

With the exception of part III of the requirements, which deals exclusively with characteristics at or close to the stall, the requirements pertain to behavior of the airplane in the range of normal flight speeds at angles of attack below that at which the stall would occur.

In the specifications which follow, the lower limits of the control-force gradients are specified in terms of the ability of the controls to return to trim positions upon release from deflected positions. This is a very desirable characteristic because it assures {AQ2} a control friction sufficiently low in comparison with the aerodynamic forces to allow the pilot to feel the aerodynamic forces on the controls. However, some additional interpretation of the specifications is necessary, because no control system can be made entirely free of friction and, therefore, there will always be some small deviation from return to absolute trim. At the present time, it is not possible to fix the allowable limits for these deviations. It is known, however, that controls reasonably free from friction, as measured on the ground, have satisfactory self-centering characteristics in the air as long as there is a definite force gradient. For elevators, force gradients as low as 0.05 pound per mile per hour have been satisfactory when the friction was small. For relatively small airplanes such as fighters, trainers, and light airplanes, it appears that about 2 pounds of friction in the elevator control system and 1 pound in the aileron represent an upper limit. In several cases, where push-pull rods with ball bearings were used throughout the control system, friction in both elevator and aileron systems has been found to be under 1/2 pound.

For large airplanes not intended to maneuver where visual or instrument references are always available, self-centering characteristics are not believed to be essential, although they are very desirable. In these airplanes, control friction should be kept as low as possible, although there is indication that considerable more friction can be tolerated. A representative amount of control friction for a transport or medium bomber would be about 10 pounds in the elevator system and 6 pounds in the ailerons.

Irreversible controls have somewhat similar characteristics to controls with high friction; that is, they are not self-centering and therefore tend to destroy control feel. They are not considered desirable, although on very large airplanes where the rates of deviation from steady flight are slow they have been used successfully on ailerons.

I. Requirements for longitudinal stability and control.

<u>Requirement (I-A):</u> Characteristics of uncontrolled longitudinal motion.

When elevator control is deflected and released quickly, the subsequent variation of normal acceleration and elevator angle should have completely disappeared after one cycle.

<u>Reasons for Requirement (I-A)</u>: The requirement specifies the Degree of damping required of the short-period longitudinal oscillation with controls free. A high degree of damping is required because of the short period of the motion. With airplanes having less damping than that specified, the oscillation is excited by gusts accentuating their effect and producing unsatisfactory, rough-air characteristics. The ratio of control friction to air forces is such that damping is generally reduced at high speeds. When the oscillation has appeared at high speeds as in dives and dive pullouts it was, of course, very objectionable because of the accelerations involved.

The short-period oscillations involve variations of the angle attack at essentially constant speed and should not be confused with the well-known long-period (phugoid) oscillation, which involves variation of speed at an essentially constant angle of attack. As shown by the tests of reference 2, the characteristics of the latter mode of longitudinal motion had no correlation with the ability of pilots to fly an airplane efficiently, the long period of the oscillation making the degree of damping unimportant. Subsequent tests have not altered this conclusion. The case of pure longitudinal divergence of the airplane (static instability) will be covered later under requirements of the elevator control in steady flight. No requirement for damping of the long-period phugoid motion appears justifiable at the present time.

<u>Design considerations</u>. – A theoretical analysis of this problem (unpublished) has shown that the damping of the control-free (short-period) oscillation is dependent chiefly on the magnitude of the hinge-moment coefficient of the elevators and on the mass balance and moment of inertia of the control system. The analysis shows that the damping is improved by increasing the hinge-moment coefficient, increasing the mass balance, and reducing the moment of inertia. The introduction of friction damping in the control system should, of course, also be effective although control friction is very undesirable for other reasons.

<u>Requirement (I-B)</u>: Characteristics of elevator control in steady flight.

1. The variation of elevator angle with speed should indicate positive static longitudinal stability for the following conditions of flight:

a. With engine or engines idling, flaps up or down, at all speeds above the stall.

b. With engine or engines delivering power for level flight with flaps down (as used in landing approach), landing gear down, at all speeds above the stall.

c. With engine or engines delivering full power with flaps up at all speeds above 120 percent of the minimum speed.

2. The variation of elevator control force with speed should be such that pull forces are required at all speeds below the trim speed and push forces required at all speeds above the trim speed for the conditions requiring static stability in item 1.

3. The magnitude of the elevator control force should everywhere be sufficient to return the control to its trim position.

4. It should be possible to maintain steady flight at the minimum and maximum speeds required of the airplane.

Reasons for Requirement (I-B): Items 1 and 2 require positive static stability for flight conditions in which the airplane is flown for protracted lengths of times, or where opportunity exists to establish a trim speed so that stable characteristics can be realized. Positive static stability at this time is not considered particularly helpful to a pilot at very low speeds with full power on or with flaps extended with full power on, because of the large trim changes due to power usually experienced. The conditions are classed as emergency conditions because in actual operation they are entered suddenly from approach conditions, where relatively little power is used. In these cases the elevator force and position changes, due to applied power and change of flap setting, are usually far greater than any inherent stable or unstable force or position gradients which exist due to the degree of static stability present. For these reasons, static stability in these conditions is not considered essential, at least not until trim changes due to power are reduced to much lower values than are experienced at the present time. The magnitude of allowable trim change due to power and flaps, is covered later in Requirement (I-F).

In other conditions of flight, however, static stability is regarded as an essential flight characteristic. Item 1 pertains to the elevator-fixed condition. This requirement ensures that the airplane will remain at a given angle of attack or airspeed as long as the elevator is not moved, and provided that disturbed motion of the airplane is not left uncontrolled for long periods of time. Positive stability eliminates the need for constant control manipulation in maintaining given conditions and, furthermore, simplifies the control manipulation when a speed change is desired, because the direction of control movement required to start the rotation in pitch corresponds to that required to trim at the new angle of attack. A negative slope to the elevator-angle curve is a necessary requirement for elevator control feel, and the degree of control feel increases as the variation of elevator angle with angle of attack is increased (reference 4). 1n general, it may be said that the variation of elevator angle with angle of attack should be negative and as large numerically as is consistent with other requirements of elevator control.

Item 2 requires that the elevator-free, static longitudinal stability shall always be positive. This specification ensures that the airplane will not depart from a trim speed except as a result of definite action on the part of the pilot.

Item 3 requires that the elevator control be self-centering a characteristic which is necessary for the attainment of control feel.

The reason for item 4 is obvious.

Design considerations. - A detailed analysis of the static longitudinal stability characteristics of various airplanes and the influence of various design features on the observed characteristics, is given in reference 3.

Requirement (I-C): Characteristics of the elevator control in accelerated flight.

1. By use of the elevator control alone, It should be possible to develop either the allowable load factor or the maximum lift coefficient at every speed.

2. The variation of elevator angle with normal acceleration in steady turning flight at any given speed, should be a smooth curve which everywhere has a stable slope.

3. For airplanes intended to have high maneuverability, the slope of the elevator-angle curve should be such that not less than 4 inches of rearward stick movement is required to change angle of attack from a C_L of 0.2 to C_{Lmax} in the maneuvering condition of flight.

4. As measured in steady turning flight, the change in normal acceleration should be proportional to the elevator control force applied.

5. The gradient of elevator control force in pounds per unit normal acceleration, as measured in steady turning flight, should be within the following limits:

a. For transports, heavy bombers, etc., the gradient should be less than 50 pounds per g.

b. For pursuit types, the gradient should be less than 6 pounds per g.

c. For any airplane, it should require a steady pull force of not less than 30 pounds to obtain the allowable load factor.

Reasons for Requirement (I-C): Item 1 of this specification requires that sufficient elevator control should be available to execute maneuvers of the minimum radius inherent in the aerodynamic and structural design of the airplane. Since the curvature of the flight path is directly related to the normal acceleration, it is obvious that the attainment of either the maximum lift coefficient or the allowable load factor is the limiting condition.

Item 2 is a requirement for stability in turning flight. Airplanes that do not meet this requirement tend to "dig" and overshoot desired accelerations in maneuvers, even though every use is made of visual and instrument references.

Item 3 specifies the amount of stability required of an airplane which must be maneuvered at or close to maximum lift without resort to visual or instrument references. It has been demonstrated by tests of several pursuit airplanes that longitudinal stability and control characteristics as specified are necessary for airplanes that require a high degree of control feel. The provision of such characteristics also reduces the time required to change angle of attack in entering rapid turns or zooms due to the simplified control manipulation associated with a definitely stable airplane.

The linear stick-force gradients specified in item 4 are, of course, very desirable as an aid to the pilot in obtaining the accelerations desired.

The numerical limits specified for the force gradients in item 5, are such that the minimum radius may be readily attained in any airplane. For pursuit types, gradients greater than 6 pounds per g were considered heavy by pilots. For airplanes where the load factor is lower, such as bombers, transports, etc., which are not required to maneuver continuously, a gradient of 50 pounds per g is not excessive. To ensure against inadvertent overloading of the structure, the 30-pound lower limit (item 5-c) is necessary. For pursuit airplanes with allowable load factors of 9, this lower limit would correspond to a gradient of about 4 pounds per g. For airplanes with lower load factors, such as bombers, transports, or light airplanes, the gradient in pounds per g would be proportionately higher.

Important design factors. - In turning flight, due to the curvature of the flight path, a stabilizing effect is obtained which increases the slope of the elevator-angle curve over that obtained in straight flight. The stick forces required to maintain a given lift coefficient are considerably greater than those for straight flight, however, because the elevator angles are higher and because they are obtained at greater speeds. For this reason, it is necessary to specify the upper limit of elevator-force gradients only for accelerated flight.

A linear relation between stick force and normal acceleration is always obtained provided the elevator-angle curve and hinge-moment coefficient curve have linear variations with angle of attack and deflection, respectively.

Requirement (I-D): Characteristics of the elevator control in landing.

l. (Applicable to airplanes with conventional landing gears only.) The elevator control should be sufficiently powerful to hold the airplane off the ground until three-point contact is made.

2. (Applicable to airplanes with nose-wheel type landing gears only.) The elevator control should be sufficiently powerful to hold the airplane from actual contact with the ground until the minimum speed required of the airplane is attained.

3. It should be possible to execute the landing with an elevator control force which does not exceed 50 pounds for wheel-type controls, or 35 pounds where a stick-type control is used.

Reasons for Requirement (I-D): For airplanes with conventional landing gears, the three-point attitude usually corresponds closely to that for the development of minimum speed for landing. In addition, an airplane alighting simultaneously on main wheels and tail wheel, is less likely to leave the ground again as a result of possessing vertical velocity at the time of contact.

The reason for item 2 is obvious.

The limits of allowable control force in landing were determined from considerations of the pilot's capabilities. The limit forces given are 80 percent of those which a pilot can apply with one hand to the different control arrangements with the control 12 inches from the back of the seat. (See references 4 and 5.)

Design factors. - The requirements of the elevator in producing three-point or minimum speed landings are by far the most critical from a standpoint of control power. Flight-test data show that low-wing monoplanes with flaps down require about 10° more up elevator to land than to stall in comparable conditions at altitude. Without flaps this increment due to ground effect is not so great, and with high-wing monoplanes without flaps the landing frequently requires less elevator than the power-off stall at altitude.

Requirement (I-E): Characteristics of elevator control in take-off.

During the take-off run, it should be possible to maintain the attitude of the airplane by means of the elevators at any value between the level attitude and that corresponding to maximum lift after one-half take-off speed has been reached.

Reasons for Requirement (I-E): The attitude of an airplane for optimum take-off characteristics depends upon the condition of the runway surface. On smooth, hard surfaces with low-rolling friction the shortest take-off run is obtained with a tail-high attitude. Where rolling friction is high, however, it is advantageous to maintain an attitude which gives high lift.

Design considerations. - Adequate control of the attitude angle during take-off depends more on the proper location of the landing gear with respect to the center of gravity than on the characteristics of the elevators themselves. This requirement certainly is not critical from a standpoint of elevator control. An airplane that has sufficient tail volume to be stable and sufficient elevator control to perform three-point or minimum-speed landings should meet this requirement easily, as long as the main landing-gear wheels are properly located.

Requirement (I-F): Limits of trim change due to power and flaps.

1. With the airplane trimmed for zero stick force at any given speed and using any combination of engine power and flap setting, it should be possible to maintain the given speed without exerting push or pull forces greater than those listed below when the power and flap setting are varied in any manner whatsoever.

a. Stick-type controls - 35 pounds push or pull.

b. Wheel-type controls - 50 pounds push or pull.

2. If the airplane cannot be trimmed at low speeds with full use of the trimming device, the conditions specified in item 1 should be met with the airplane trimmed full tail-heavy.

Reasons for Requirement (I-F): It is desired that emergency manipulation of flaps or throttles do not require simultaneous adjustments of the trimming device. The force limits specified are approximately 80 percent of the maximum that a pilot can apply with one hand. The one-hand limit is necessary to allow the adjustment of throttles, flaps, or trimming device while complete longitudinal control is maintained. It is, of course, desirable that the trim changes be less then the limiting values given. The ideal condition would be one where the stick forces required for trim were not influenced by the position of the flaps or throttles.

It is also desirable that the control position required to maintain a given speed or lift coefficient be independent of the power and flap position insofar as possible. It is not, however, believed reasonable or necessary to specify any definite limits at this time.

Design factors. - Because of simultaneous changes in downwash, dynamic pressure at the tail, and pitching moment of the airplane less tail, the trim change produced by variations of power and flap setting are very difficult to predict. Several of the effects, however, have opposite signs, so that with sufficient care it should be possible to restrict the trim changes to a reasonably low value. Wind-tunnel tests of a powered model of the design under consideration would be a great help if not an absolute essential in this connection.

Requirement (I-G): Characteristics of the longitudinal trimming device.

1. The trimming device should be capable of reducing the elevator control force to zero in steady flight in the following conditions:

a. Cruising conditions - at any speed between high speed and 120 percent of the minimum speed.

b. Landing condition - any speed between 120 percent and 140 percent of the minimum speed.

2. Unless changed manually, the trimming device should retain a given setting indefinitely.

Reasons for Requirement (I-G): It is, of course, desirable to be able to reduce the elevator force to zero in conditions where the airplane must be flown for protracted lengths of time. It is also desirable to be able to establish a trim condition within the allowable speed limits of the airplane so that release of the controls will not put the airplane in a dangerous position.

The reasons for item 2 are obvious.

II. Requirements for Lateral Stability and Control.

Requirement (II-A) : Characteristics of uncontrolled lateral and directional motion.

1. The control-free lateral oscillation should always damp to one-half amplitude within two cycles.

2. When the ailerons are deflected and released quickly, they should return to their trim position. Any oscillations of the ailerons themselves shall have disappeared after one cycle.

3. When the rudder is deflected and released quickly, it should return to its trim position. Any oscillation of the rudder itself shall have disappeared after one cycle.

Reasons for Requirement (II-A): Because of its relatively short period, the lateral oscillation must be heavily damped. It is not logical to specify limits for the period of the oscillation because the period is dependent on factors covered by other specifications and also because the period is dependent on the size, speed, and weight of the airplane. The amount of damping specified in item l has been obtained with all satisfactory airplanes tested.

Items 2 and 3 of the requirement (II-A) are included to ensure stability in the behavior of the lateral controls themselves.

Attention is called to the omission of a requirement for spiral stability. Tests have shown that the lack of spiral stability has not detracted from the pilot's ability to fly an airplane efficiently. In fact, it is very difficult to determine whether an airplane is inherently spirally stable or not, because divergence will occur with a spirally stable airplane if perfect lateral and directional trim do not exist or if slight asymmetry in engine poser occurs in a multiengine airplane. For these reasons a large amount of inherent spiral stability would be required to ensure against lateral divergence under control conditions.

Since it appears that the degree of spiral stability or instability is inconsequential or at least of doubtful importance under actual conditions, it is desirable to avoid any such requirement because the design conditions for spiral stability conflict with other factors known to be essential in the attainment of satisfactory flying qualities.

Design considerations. - The theory of dynamic stability has been rather extensively developed from a mathematical standpoint. The charts of reference 6 make the calculation of the dynamic characteristics a relatively simple matter, provided the stability derivatives are known. In general, however, the stability derivatives are not known and cannot be estimated to a reasonable degree of accuracy, particularly with power on.

On the basis of experience, however, it appears that the damping requirement is now a critical design condition. There is every indication that when other requirements of fin area and dihedral are met, the uncontrolled lateral motion will be satisfactory.

Items 2 and 3 of the requirement (II-A) are dependent, as was the elevator-free motion (requirement I-A), on the control-hinge moments, mass balance, and moment of inertia of the control systems.

Requirement (II-B): Aileron-control characteristics (rudder locked).

1. At any given speed, the maximum rolling velocity obtained by abrupt use of ailerons should vary smoothly with the aileron deflection and should be approximately proportional to the aileron deflection.

2. The variation of rolling acceleration with time following abrupt control deflection should always be in the correct direction and should reach a maximum value not later than 0.2 seconds after the controls have reached their given deflection.

3. The maximum rolling velocity obtained by use of ailerons alone should be such that the helix angle generated by the wing tip, pb/2V, is equal to or greater than 0.07.

where: p maximum rolling velocity, radians/second

b wing span

V true airspeed, feet per second

4. The variation of aileron control force with aileron deflection should be a smooth curve. The force should everywhere be great enough to return the control to trim position.

5. At every speed below 80 percent of maximum level-flight speed, it should be possible to obtain the specified value of pb/2V without exceeding the following

control-force limits:

a. Wheel-type controls: ±80 pounds applied at rim of wheel.

b. Stick-type controls: ±30 pounds applied at grip of stick.

Reasons for Requirement (II-B): Item 1 of this requirement states an obviously desirable condition for any control; i.e., that the response shall be proportional to deflection.

Item 2 is designed to eliminate controls that are unsatisfactory from a standpoint of lag in the development of the rolling moment, or controls in which the initial rolling action is in the wrong direction.

Item 8 was obtained by correlation of pilots' opinions and measured characteristics for some 20 different airplanes of various types and sizes (reference 7). It was found that pilots judged the adequacy of their lateral control on the basis of the helix angle generated by the wing tip of the airplane. Airplanes giving values of pb/2V less than 0.07 were always considered unsatisfactory.

Item 4 is a requirement for self-centering characteristics of the lateral control. This is a necessary condition for satisfactory control feel.

The specifications of item 5 was determined by the limitations of pilots in applying forces to the lateral controls. Lower forces are, of course, desirable.

Design considerations for Requirement (II-B): Item 1 represents a normal characteristic of flap-type ailerons, provided they are not deflected beyond the range where effectiveness is linear. Certain spoiler-type ailerons, however, have been unsatisfactory because of their failure to meet this requirement. In these cases, the variation of effectiveness with deflection was either markedly nonlinear or such that appreciable movements of the control about the neutral point were required before the ailerons became effective.

Item 2 also is met by all conventional flap-type ailerons. Again, however, certain arrangements of lateral controls that depend on spoiler action have proved unsatisfactory because of lag or incorrect initial development of the rolling moment. Detailed information on various satisfactory and unsatisfactory spoiler types may be found in reference 8 and later reports on the subject.

The specification of the helix angle, pb/2V ≥ 0.07 of item 3, corresponds approximately to requiring a rolling-moment coefficient, C_2, of 0.035 or greater. Actually, since pb/2V is equal to the ratio of the rolling-moment coefficient to the damping-moment coefficient, C/C_{2p}, a criterion in terms of C_2 alone is not strictly applicable. The damping-moment coefficient tends to decrease with increased taper of the wing and to increase with increased aspect ratio. However, for the aspect ratios and taper ratio likely to be used, the criterion considered in terms of rolling-moment coefficient alone, $O_2 \geq 0.035$ should be satisfactory. In several types tested, particularly the very large airplanes, control-cable stretch resulted in a very serious loss of aileron effectiveness. There is also indication that wing twist under the torsional loads applied by ailerons, should be considered in an interpretation of the rolling-moment coefficient required to obtain the specified value of pb/2V.

Item 4 sets the upper limit for aileron control friction since the ability of a control to center itself depends on the ratio of the inherent force gradient to the frictional force.

The control-force limits of item 5 are, of course, critical at the high speed specified. This requirement can be met by using existing design methods without servo control or mechanical booster systems except, perhaps, for the very largest airplanes that appear at this time.

Requirement (II-C): Yaw due to ailerons.

With the rudder locked at 110 percent of the minimum speed, the sideslip developed as a result of full aileron deflection should not exceed 20°.

Reasons for Requirement (II-C): Aileron yaw is responsible not only for annoying heading changes as a result of the use of ailerons, but also for a reduction of aileron effectiveness unless the rudder is carefully manipulated to eliminate the sideslip induced. This latter effect is also dependent on the rolling moment due to sideslip (dihedral effect).

The requirement for aileron yaw expressed in this manner clearly separates satisfactory characteristics from those considered unsatisfactory by pilots and, moreover, has the merit of relating the factors responsible for aileron yaw in a fundamental manner. The limiting condition of 20° sideslip seems surprisingly high, but the number of satisfactory airplanes that develop sideslip angles substantially this great, cannot be ignored. The requirement, however, is written to cover the critical low-speed conditions. At cruising speeds, comparable tests would give sideslip angles of the order of 5°.

Design considerations. - The sideslip due to ailerons is chiefly dependent on the aileron yawing moment, the yawing moment due to rolling, the dihedral effect, and the directional stability of the airplane. Compliance with the requirement depends mainly on the provision of sufficient directional stability, since the aileron yawing moment and the yawing moment due to rolling are determined by the aileron power. Of course, the designer has some control over the adverse aileron yawing moment through the use of differential in the control system and by increasing the profile drag of the up aileron. These effects, however, are generally small in comparison with inherent yawing moments due to ailerons and rolling velocity, which are always adverse in sign.

* The measured sideslip angle on which this and subsequent specifications are based, should not be confused with the angle of bank of the airplane. The angle of sideslip is simply that given by a vane free to pivot about a vertical axis and align itself with the relative wind.

The required amount of directional stability is simply that which will give an equilibrium of the yawing moments at or below the angle of sideslip specified. The adverse aileron yawing moments can, of course, be determined in the wind tunnel.

The yawing moment due to rolling for wings of various planforms is given in the charts of reference 9.

Requirement (II-D): Limits of rolling moment due to sideslip (dihedral effect).

1. The rolling moment due to sideslip as measured by the variation of aileron deflection with angle of sideslip should vary smoothly and progressively with angle of sideslip, and should everywhere be of a sign such that the aileron is always required to depress the leading wing as the sideslip is increased.

2. The variation of aileron stick force with angle of sideslip should everywhere tend to return the aileron control to its neutral or trim position when released.

3. The rolling moment due to sideslip should never be so great that a reversal of rolling velocity occurs as a result of yaw due to ailerons (rudder locked).

Reasons for Requirement (II-D): Item 1 ensures that the roll due to rudder will always be in the correct direction and that any lateral divergence will not be of a rapid type. It is also a necessary but not a sufficient condition for the ability to raise a wing by means of the rudder.

Item 2 was required to ensure that the rolling moment due to sideslip will be of the correct sign with controls free. The ability of the control to self-center here again is a requirement for control feel.

The reason for item 3 is obvious.

Design considerations. - Wind-tunnel data showing the effects of flaps, wing planform, and fuselage-wing arrangement on the rolling moment due to sideslip are given in references 10 and 11. These results are generally substantiated by flight test. With single-engine, low-wing airplanes, however, the dihedral effect in sideslips made to the left sometimes became negative at low speeds with power on, even though it was satisfactory with power off or with power on at higher speeds. Low-wing monoplanes generally required from 4° to 8° more geometric dihedral angle than high-wing monoplanes, to obtain the same effective dihedral effect. On airplanes with the trailing edges of the wing swept forward, flaps reduced the effective dihedral and where the trailing edge of the wing was a continuous straight line, flaps had little or no effect on the dihedral effect.

In order to meet item 2, the friction in the aileron control system must be low and the aileron required to overcome the rolling tendencies in the sideslip (dihedral effect) must exceed that at which the ailerons would tend to float due to the spanwise angle-of-attack variation.

The upper limit of the rolling moment due to sideslip (item II-D-3) is dependent on the yaw due to ailerons (item II-C-1) and the power of the aileron control (item II-B-3).

Requirement (II-E): Rudder-Control characteristics.

1. The rudder control should everywhere be sufficiently powerful to overcome the adverse aileron yawing moment.

2. The rudder control should be sufficiently powerful to maintain directional control during take-off and landing.

3. The rudder control should be sufficiently powerful to provide equilibrium of yawing moments at zero sideslip at all speeds above 110 percent of the minimum takeoff speed under the following conditions:

a. Airplanes with two or three engines: With any one engine inoperative (propeller in low pitch) and the other engine or engines developing full rated power.

b. Airplanes with four or more engines: With any one engine inoperative (propeller in low pitch) and the other engines developing full rated power.

4. The rudder control in conjunction with the other controls of the airplane provide the required spin-recovery characteristics.

5. Right rudder force should always be required to hold right rudder deflections, and left rudder force should always be required to hold left rudder deflections.

6. The rudder forces required to meet the above rudder-control requirements should not exceed 180 pounds (trim tabs neutral).

Reasons for Requirement (II-E): The reasons for these various items are obvious. Item 1 must, of course, be met if satisfactory turns are to be made at low speeds unless, of course, the directional stability is very great. Item 2 represents one of the most important functions for rudder control, although if a tricycle landing gear is used it becomes much lees important.

Items 3 and 6 should ensure adequate control over a symmetric thrust following engine failure subsequent to take-off. It does not seem necessary to retain directional control below the speed specified because of the probability that lateral instability due to stalling would set in first. The 180-pound force limit specified, is about 90 percent of the maximum that an average pilot can apply.

Design considerations. - The rudder power needed to meet item 1 of the above requirement, can be determined in the same manner that the directional stability required by aileron yaw was found (requirement II-C).

In at least one instance, item 2 of the above requirements was met without any rudder control. This was accomplished by using a tricycle landing gear and by eliminating the rudder-position variation with speed and power. However, due to the inherent instability of conventional landing gears, a certain amount of rudder control during take-off and landing will always be required when this arrangement is used, even though the rudder-trim change due to power or speed, were eliminated. Just how much rudder is needed here is not known. The efficiency of the brakes, type of tail wheel (lockable or free-swiveling), and the magnitude of the inherent ground-looping tendency undoubtedly enter into the problem. Also, in landing, the stalling characteristics of the airplane may have an important bearing. On the basis of data on hand, however, it appears that a rudder control that is sufficiently powerful to meet the other requirements outlined should generally be satisfactory from a standpoint of ground handling.

Items 3, 4, and 5 do not appear to require additional discussion.

Requirement (II-F): Yawing moment due to sideslip (directional stability).

1. The yawing moments due to sideslip (rudder fixed) should be sufficient to restrict the yaw due to ailerons to the limits specified in requirement (II-C-1).

2. The yawing moment due to sideslip should be such that the rudder always moves in the correct direction; i.e., right rudder should produce left sideslip and left rudder should produce right sideslip. For angles of sideslip between ±15° the angle of sideslip should be substantially proportional to the rudder deflection.

3. The yawing moment due to sideslip (rudder free) should be such that the airplane will always tend to return to zero sideslip regardless of the angle of sideslip to which it has been forced.

4. The yawing moment due to sideslip (rudder free with airplane trimmed for straight flight and symmetric power) should be such that straight flight can be maintained by sideslipping at every speed above 140 percent of the minimum speed with rudder free with extreme asymmetry of power possible by the loss of one engine.

Reasons for Requirement (II-F): The reasons for item are covered in discussion under requirement (II-C).

Item 2 of this requirement states a desirable characteristic for any control; i.e., the response should be proportional to the deflection.

Item 3 is designed to ensure satisfactory directional stability, particularly at large angles of sideslip where vertical tail stalling has frequently led to trouble. This requirement follows directly from the results of reference 12.

Item 4 is included to prevent the directional divergence following an engine failure from being excessively rapid. Although the ability to fly with rudder free on asymmetric power, is probably not in itself important, it is undoubtedly strongly related to the rate of divergence, and therefore the required quickness of action on the part of the pilots when this emergency occurs.

Design considerations. - The directional stability required to fulfill item 1 has been discussed under requirement (II-C).

General discussion of the factors that determine the fin area required to meet items 2 and 3 of this requirement, is given in reference 12. However, the interference effects of wing-fuselage position, vertical tail arrangements, etc., are so great that wind-tunnel tests would appear a necessary aid to design for these requirements. Since the directional stability at large angles of sideslip, however, is related to the manner in which the flow breaks down on the vertical surfaces and on its effect on the floating characteristics of the rudder, the scale of the test should be kept as great as possible.

Requirement (II-G): Cross-wind force characteristics.

The variation of cross-wind force with sideslip angle, as measured in steady sideslips, should everywhere be such that right bank accompanies right sideslip and left bank accompanies left sideslip.

Reasons for Requirement (II-G): Under normal conditions in a sideslip or skid, a force is produced which acts toward the backward-lying wing tip. Since this actual angle of sideslip cannot be observed by the pilot, the cross-wind force developed

allows appreciation of the fact that sideslip exists because of the lateral acceleration which occurs. In steady sideslips the cross-wind force is balanced by a component of the weight of the airplane, so that an angle of bank results. The greater the cross-wind force the greater is the angle of bank. An approximate relation between angle of bank ϕ and the crosswind force may be written as follows:

$$\phi = \sin^{-1} \frac{\text{cross-wind force}}{\text{weight of airplane}}$$

In addition to providing the pilot with "feel" of the sideslip or skid, the lateral attitude from which it is possible to recover with the rudder alone (without permitting a heading change) is directly related to the magnitude of the cross-wind force. Obviously, a positive dihedral effect is also necessary for the performance of this maneuver, but the fact remains that turning toward the low wing will always occur if the lateral attitude from which recovery is attempted, exceeds that which can be held in steady sideslip with full rudder.

For these and other reasons, large values of cross-wind force are desirable and more rigid specification than that given would lead to better flying qualities. On the other hand, it is not known whether this could be done without increasing the drag of the airplane.

None of the airplanes tested to date has failed to meet the requirement as written. It is included, however, because there is indication on the basis of wind-tunnel tests, that some future designs may actually develop cross-wind force of opposite sign to that normally experienced. Obviously, this condition could not be tolerated.

Requirement (II-H): Pitching moment due to sideslip.

As measured in steady sideslip, the pitching moment due to sideslip should be such that not more than 1° elevator movement is required to maintain longitudinal trim at 110 percent of the minimum speed when the rudder is moved 5° right or left from its position for straight flight.

Reasons for Requirement (II-H): A pitching-moment change due to sideslip is undesirable because it requires that the elevator as well as the rudder must be coordinated with the ailerons. Also, since sideslip of considerable amounts may be carried inadvertently, a marked variation of pitching moment with sideslip will tend to produce inadvertent angle-of-attack changes. The condition is critical at high lift coefficients, so compliance with the specifications given should automatically ensure satisfactory characteristics at higher speeds.

Design considerations. - It is believed that the change in pitching moment with sideslip occurs as a result of the downwash change experienced by the horizontal tail as it moves from behind the wing center. In most cases, the moment produced is a diving moment because of the relatively high concentration of downwash at the wing center due to the propeller or partial-span flaps. It has also been noted that the magnitude of the pitching moment due to sideslip, progressively decreased as the angle of attack was reduced, presumably because of the corresponding reduction of downwash angles.

Requirement (II-I): Power of rudder and aileron trimming devices.

1. Aileron and rudder trimming devices should be provided if the rudder or aileron forces required for straight flight at any speed between 120 percent of the minimum speed and the maximum speed, exceed 10 percent of the maximum values specified in requirements (II-B-5) and (II-E-6), respectively, and unless these forces at cruising speed are substantially zero.

2. Multiengine airplanes should possess rudder and aileron trimming devices sufficiently powerful in addition to trim for straight flight at speeds in excess of 140 percent of the minimum speed with maximum asymmetry of engine power.

3. Unless changed manually, the trimming device should retain a given setting indefinitely.

Reasons for Requirement (II-I): The reasons for the items listed above are obvious.

Requirement (III): Stalling characteristics.

1. The approach of the complete stall should make itself unmistakably evident through any or all of the following conditions:

a. The instability due to stalling should develop in a gradual but unmistakable manner.

b. The elevator pull force and rearward travel of the control column should markedly increase.

c. Buffeting and shaking of the airplane and controls produced either by a gradual breakdown of flow or through the action of some mechanical warning device, should provide unmistakable warning before instability develops.

2. After the complete stall has developed, it should be possible to recover promptly by normal use of controls.

3. The three-point landing attitude of the airplane should be such that rolling or yawing moments due to stalling, not easily checked by controls, should not occur in landing, either three-point or with tail-first attitude 2° greater than that for three-point contact.

Reasons for Requirement (III): The items of this requirement are in keeping with all others given; i.e., it demands all that can be obtained with existing knowledge and yet is sufficiently rigid, so that any airplane that complies with the specification will be reasonably safe in terms of our present standards. Since there is never occasion in the normal operation of an airplane for a pilot to stall intentionally, such characteristics that provide warning of the stall are given first importance. If the warning is unmistakable, the relative violence of the actual stall loses much of its significance because it would then occur only as an intentional act on the part of the pilot and at a safe altitude. Item 2 is included to ensure that recovery from an intentional stall can be promptly made.

Item 3 is an outgrowth of some experience in studying ground-handling problems. In most cases, poor stalling characteristics are troublesome in landing because of wing dropping either during the actual landing flare or after the airplane has alighted during the landing run. In other cases the ring stall has influenced the flow

at the vertical tail in such a manner that powerful yawing moments have developed. Unless the stall itself can be made to develop in a gentle manner, the cure for these characteristics can be effected by preventing the occurrence of the stall altogether in the landing maneuver.

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(a) C. H. Dearborn, "Full-Scale Wind-Tunnel Tests of Navy XF2A-1 Fighter Airplane," NACA Confidential Memorandum Report, 17 May 1938, RA file 603, LHA.

(b) H. J. E. Reid, Engineer-in-Charge, LMAL, to NACA, "Drag-reduction investigation on XP-40 airplane," 7 Apr. 1939, RA file 637, LHA. Includes attachment from Clinton H. Dearborn, Aeronautical Engineer, LMAL, "Test Program on XP-40 Airplane in Full-Scale Wind Tunnel," undated.

(c) Don R. Berlin, Chief Engineer, Curtiss Aeroplane Division, Curtiss Wright Corp., to Major Carl F. Greene, Materiel Division, Liaison Officer, NACA Laboratory, Langley Field, VA, 4 Aug. 1939 [originally classified "Confidential"], RA file 637, LHA.

(d) C. H. Dearborn, Aeronautical Engineer, LMAL, to Engineer-in-Charge, "Investigation for determining means of increasing the high speed of the XP-41 airplane," 8 June 1939, RA file 672, LHA.

As one can see in the following documents involving its drag reduction program, the NACA followed a very systematic method of experimental parameter variation. First, its engineers examined an airplane in detail, identifying those external features most suspected of causing unnecessary drag. They then made the airplane as aerodynamically clean as possible, by carefully removing protuberances like the radio antenna and using putty or tape to cover holes and leaks and to reshape irregular surfaces such as the cockpit canopy. Following this, they mounted the plane in the test chamber, and measured its drag at various wind speeds. In this faired and sealed condition, the airplane naturally proved to have less drag than the original body, but it was impossible for this pristine shape, with essential parts covered up or removed, actually to fly. The wind tunnel workers returned the plane to its service condition, item by item, and evaluated the change in drag caused by each action.

In the case of the cleanup tests of the Seversky XP-41 in late 1939, Langley studied the drag of the airplane in 18 different configurations. The data indicated that the changes in drag values corresponding to the steps of the cleanup process were generally small, amounting to only a few percent of the total drag coefficient and thus involving only small speed changes. Taken together, however, increments like these often resulted in impressive gains in total performance.

Document 3-29(a), C. H. Dearborn, "Full-Scale Wind-Tunnel Tests of Navy XF2A-1 Fighter Airplane," NACA Confidential Memorandum Report, 17 May 1938.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

CONFIDENTIAL MEMORANDUM REPORT
For the
Bureau of Aeronautics, Navy Department
FULL-SCALE WIND TUNNEL TESTS OF NAVY XF2A-1 FIGHTER AIRPLANE

By C. H. DEARBORN

INTRODUCTION

At the request of the Bureau of Aeronautics, Navy Department, an investigation has been conducted in the full-scale wind tunnel on the Navy XF2A-1 fighter airplane to determine: (1) its minimum and high-speed drag coefficients, (2) reductions in drag to be obtained by improving the fairing of various parts of the airplane, and (3) probable points at which carbon monoxide enters the cockpit.

This report presents as Part I the wind-tunnel investigation, and as Part II recommended changes in the airplane which would involve major alterations to the design.

PART I - WIND-TUNNEL INVESTIGATION TESTS

The Navy XF2A-1 airplane was mounted on the full scale wind-tunnel balance as shown in figure 1. The propeller was removed and all control surfaces locked in their neutral positions for the tests. Inasmuch as the time allotted for the performance of the tests of this airplane was very limited, it was necessary to reduce the amount of tunnel testing to a minimum.

First a force test was made on the airplane in its normal condition over a small angle-of-attack range, including the minimum and high-speed drag angles of attack at four test speeds from 60 to 100 miles an hour. From this test it was ascertained that an angle of attack of 1.05° approximates the mean angle between the reported high speeds for sea level and an altitude of 15,200 feet, and all further force tests were therefore confined to this angle of attack.

In order to study the general flow conditions, wool tufts were placed at frequent intervals over the fuselage. To determine points of possible entry of carbon monoxide into the fuselage, tufts were placed at the edge of the opening for the arrester gear hook at the end of the fuselage and at the opening around the tail-wheel assembly. An observer with a fine wool tuft on the end of a rod was stationed in the cockpit to note the entry of air around proximity of the cockpit. The observation of the tufts was carried out at a tunnel speed of 60 miles an hour, and, in addition to visual observation, moving pictures were made.

Next a series of force tests was conducted on the airplane at the high-speed attitude and for the same four tunnel speeds as noted for the standard condition but with the following modifications to the airplane:

(1) With streamline fairings over the engine exhaust stack to obtain an indication of what drag reduction might be made by altering the stacks (fig. 2).

(2) With cover plates over the gaps between the top of the wheels and the ends of the landing-gear strut fairings in addition to change (1) (fig. 2).

(3) With landing wheels removed and flush cover plates installed over wheel wells (fig. 2).

(4) Changes covered by items (1) to (3) were removed, and modifications made to the engine cowling, which included refairing the carburetor intake scoop, the oil-cooler intake scoop, the openings at the forward ends of the blast tubes, and fairing out of the indentation at the forward end of the fuselage at the point where the blast tubes leave the fuselage. Figure 3 shows the modification to the oil-cooler scoop, figure 4 shows the modifications made on top of the cowling, and figure 5 presents sections of the changes in fairing which were accomplished by the addition of plasticine. The sections in figure 5 were patterned after low-drag engine cowls developed in the high-speed tunnel and during the modifications plasticine was added until the turbulent wakes noted by the behavior of the tufts for the original condition disappeared. The openings at the forward ends of the blast tubes were sealed as the greater part of the disturbing effect at these points appeared to be due to the flow of air from within the cowl through the areas between the blast tubes and openings in the cowl. As there was no flow of air through the blast tubes, the arrangement tested appears to simulate a possible service installation.

(5) With gun sight removed to determine its drag.

(6) The break in the contour of the cockpit enclosure at the end of the movable section was faired out by the addition of a sheet-metal patch. This change was made in the presence of changes (4) and (5).

(7) The final change consisted of adding a rounded windshield, which is shown with item 6 in figures 6(a) and 6(b). (Original figures not included herein.)

At the completion of the force tests, surveys were made in the wake of the left wing by means of a comb of total-head and static tubes. The measurements were made at the five spanwise stations shown in figure 7 for a test speed of 90 miles an hour and for angles of attack corresponding to zero lift and to the high-speed condition. Surveys were repeated at stations 4 and 5 with the ailerons sealed in order that the drag due to the aileron slot might be determined.

RESULTS AND DISCUSSION

The tuft observations are summarized in figure 8. It will be noted that in general the flow is satisfactory over the fuselage with the exception of the areas in back of the air-intake scoops, small cowls at the forward ends of blast tubes, and the exhaust stacks. A reproduction of several frames of the moving-picture film in figure 9 shows that the flow at the wing-fuselage juncture is very satisfactory. The flow at the top of the engine cowling, however, was particularly turbulent. The reproduction of the moving-picture film covering this area in figure 10 unfortunately does not give as clear a picture of this condition as might be desired, and it is therefore recommended that recourse be taken to the projection of the moving-picture film. It will, however, be noted in figure 10 that there is considerable movement of the tufts from one frame to another as for example may be seen by the changed attitude of the tuft marked in the upper frame. One of the greatest disturbing factors at the forward edge of the cowling appeared to be the flow of air outward between the blast tube and the cowling.

The observer in the cockpit during the tuft observation reported air flow into the cockpit at the following points, which may be locations at which carbon monoxide enters the fuselage.

TABLE 1

Locations	Nature of inflow
Case and link ejector chutes	Slight inflow
Flap indicator control	Slight inflow
Opening in side of fuselage for aileron control tube	Strong flow
Flow from rear end of fuselage	Steady flow
Bomb sight windows.	No flow or so small as not to be discernible
Gas drain tubes.	No flow or so small as not to be discernible

The principal point of air flow out of the cockpit was at the gap at the end of the movable portion of the cockpit enclosure. The tufts placed around the edges of the opening at the end of the fuselage for the arrester hook and around the opening for the tail-wheel assembly showed a infinite inflow of air into the fuselage.

The force-test results for each individual run (figs. 11(a) to 11(h)) show that the scale effect on the lift and drag coefficients is small within the speed range of the tests. The coefficients have been corrected for all wind-tunnel effects and are based on a wing area of 209 square feet. To aid in visualizing the condition of the airplane for each run, the letter accompanying the figure number is the same as that used to designate the various conditions of the airplane, shown on the chart of figure 13. The results of the force tests are summarized in figures 12 and 13.

It will be noted that the fairing over the exhaust stacks, the cover plates over the wheel wells, and the modifications to the engine cowling accounted for appreciable reductions in the drag of the airplane, with the latter accounting for by far the largest reduction. The summation of the drag reduction, which is obtained by adding the difference in the drag coefficients for conditions A and D and A and H, is equal to the drag-coefficient increment 0.0103, which constitutes about 27 percent of the high-speed drag coefficient of the normal airplane. It is also of interest to note that the modifications to the cockpit enclosure did not account for any reduction in drag.

A recommended type of exhaust stack to reduce the drag over the existing type is suggested by an investigation of air discharge openings conducted in the atmospheric wind tunnel. The stack would consist of a turn to the rear of 90° or preferably a slightly smaller angle to keep the exhaust gases off the fuselage with one or two guide vanes to reduce the bend loss.

The profile drag coefficients obtained by the momentum method for the five stations along the semispan of the wing are shown by the experimental points in figures 14 and 15. The profile drag coefficients for the two stations in back of aileron for the conditions of the normal aileron slot and sealed slot given in the table show that the aileron slot does not increase the wing drag.

Document 3-29(b), H. J. E. Reid, Engineer-in-Charge, LMAL, to NACA, "Drag-reduction investigation on XP-40 airplane," 7 Apr. 1939.

R.A. 637

Langley Field, Va.
April 7, 1939

From LMAL
To NACA

Subject: Drag-reduction investigation on XP-40 airplane.

Reference: NACA Let. Mar. 30, 1939, CW and Enc.

1. A conference was held in the full-scale wing tunnel to discuss the test program for investigating means of increasing the high-speed of the XP-40 airplane. Those in attendance were: Mr. Hovgard, of the Curtiss Company, and Messrs. Reid, DeFrance, Silverstein, and Dearborn, of the Laboratory staff. The items of the Curtiss letter to the Material Division forwarded with reference were reviewed and a number of them have been included in the test program. An investigation of the cooling system drag is of particular interest as it is believed that a substantial reduction in drag may be made by revising the present system.

2. It now appears that the tests will be under way not later than April 12.

3. The test program as drafted at the conclusion of the conference is enclosed.

H. J. E. Reid,
Engineer-in-Charge

CHD. AMD
Enc* 3 copies program

EWM

TEST PROGRAM ON XP-40 AIRPLANE IN FULL-SCALE WIND TUNNEL

1. Smooth fuselage conditions, power-off tests, gun-blast tubes and carburetor inlets removed, exhaust pipes off, Prestone radiator retracted into fuselage and radiator cowling removed, original windshield, propeller off, holes in propeller spinner sealed, oil radiator ducts sealed.
2. Same as 1, with exhaust stacks added.
3. Same as 2, with carburetor inlets added, blast tubes omitted.
4. Same as 3, with blast tubes added.
5. Same as 4, with oil-cooler duct inlets open.
6. Power-on, with airplane in same condition as 5. This is the same as normal condition of airplane except the Prestone cooling system is retracted into fuselage, and the lower fuselage line faired into a smooth line.
7. Modified radiator installations radiator raised above present position and proper opening allowed for cooling airflow. Force and airflow measurements will be made at high speed, and climbing conditions for both power-off and power-on conditions.
8. Original airplane conditions, with radiator in normal position. Force and airflow measurements for high-speed and climbing conditions, power-off and power-on.
9. Power-on tests to determine the effect of covering the gaps in the nose of the spinner and around the propeller blades.
10. Study of methods for improvement in carburetor inlets and blast tubes, in the event that previous tests indicate it to be desirable.
11. Measurements of drag and airflow for various modifications of the original radiator installation.
12. Modifications to fairings over the retracted landing gear.
13. Measurements of wing-profile drag by means of the momentum method for high-speed conditions.
14. Measurements of the drag change due to sealing the control surface slots by momentum method at high-speed flight condition.
15. Study of leading edge fillets at wing roots. Measurements at high-speed and landing conditions, flaps up and down.
16. Measurement of the critical compressibility velocities by means of pressure measurements along fuselage and wing.
17. Measurement of boundary-layer transition location in order to determine the possibilities for drag reductions due to smoothing the wing.

It is the opinion of the staff that no large improvement in the windshield can be made, so tests for this purpose will not be included.

Clinton H. Dearborn

Aeronautical Engineer

CHD:RL
8JD
EWM

Document 3-29(c), Don R. Berlin, Chief Engineer, Curtiss Aeroplane Division, Curtiss Wright Corp., to Major Carl F. Greene, Materiel Division, Liaison Officer, NACA Laboratory, Langley Field, VA, 4 Aug. 1939.

August 4, 1939

Major Carl F. Greene
Material Division
Liaison Officer
NACA Laboratory
Langley Field, Va.

Enclosure: (A) Three View Drawing P-2251.
(B) Power Plant Drawing P-2278.
(C) Alternate Power Plant Drawing P-2414.

Dear Carl:

We have your letter dated July 31st in which you inquire concerning the status of the Curtiss CP-39-13 full scale model to be tested in the NACA full scale tunnel. In answer to your questions which you have asked the following is submitted for your information:

(a) As you know, the program which involves testing at NACA was arranged and tentatively approved by conference at Wright Field on July 19, 1939. Although no written approval has been received here at this date, we believe that you can feel assured that the program is definitely approved by Wright Field. Mr. C.H. Dearborn of the NACA staff paid us a visit this week for the purpose of arranging details concerning the design of the model and the various tests which are desired. The points which were discussed and agreements which were reached are detailed as follows:

(1) The model will be complete and all details affecting aerodynamics will be represented as nearly as possible.

(2) It will not be necessary to incorporate a motor of propeller. Mr. Dearborn feels that the effect of power can be calculated and it would simplify the model and the test procedure to omit this refinement.

(3) The landing gear will be represented and it is desired to obtain tests with the landing gear in both the extended and retracted positions.

(4) The balance supports on the wing will be spaced somewhere between 16 and 18 feet. We will advise the exact spacing as soon as this can be determined. Supports for the model will be made in accordance with NACA drawing D-6947.

(5) Control surface gaps will be represented but control surfaces will not be made movable. The design of control surfaces will be such that they may be made movable should this appear to be desirable at a later date. The flap will be represented and will be made movable to determine its effect upon the lift characteristics of the airplane.

(6) No tail wheel will be represented.

(7) Jacking the hoisting provisions will be made. Leveling points will also be included.

(8) Two types of cooling systems will be completely simulated. Actual radiators installed within the wing contour together with suitable inlet and outlet ducts will be provided. The design of the ducts will conform to the latest NACA research on this subject. Another cooling system complete with radiators and ducts will be provided in a position below the motor. It is planned to conduct tests both ways to determine which of these cooling systems or combination of them will be most desirable.

(9) External ducts, exhaust manifolds, and gun blast tubes will be represented, but made removable with flush covers for the openings for the purpose of obtaining comparative tests.

(10) No tests will be required on the wing alone.

(11) Templates for the wing and tail surface profiles will be furnished for checking purposes.

(12) The construction of the entire model will be made rugged and a structural investigation will be made to insure that the strength requirements of NACA specifications will be met.

(13) In general, it is desired to secure lift and drag information only although this may be modified later if conditions make it advisable. In particular, we wish to make a thorough study of the drag characteristics of the design with the idea of securing every practicable refinement which can be realized.

(b) It is anticipated that the model will be available about August 25th for NACA to start their tests.

(c) Regarding your proposed inspection of the model we estimate that you could obtain a very good idea of its construction and appearance if you would come here between August 15 and 20. If you can arrange this we shall be very glad to discuss in further detail the outline of the testing procedure.

Enclosures (A), (B), and (C) will give you a preliminary idea of the important characteristics of the design as they now exist. It is requested that you convey the information contained herein to the proper NACA authorities including Mr. Dearborn as this letter constitutes a confirmation of the conference with Mr. Dearborn.

Sincerely yours,

Curtiss Wright Corporation
Curtiss Aeroplane Disivion

Don R. Berlin
Chief Engineer

Document 3-29(d), C. H. Dearborn, Aeronautical Engineer, LMAL, to Engineer-in-Charge, "Investigation for determining means of increasing the high speed of the XP-41 airplane," 8 June 1939.

L. M. A. L.
Langley Field, Va.
June 8, 1939.

MEMORANDUM For Engineer-in-Charge.
Subject: Investigation for determining means of increasing the high speed of the XP-41 airplane.

1. Attached is a test program of the XP-41 airplane. Two additional copies of this program have been prepared for transmittal to the Materiel Division Liaison Office.

C. H. Dearborn
Aeronautical Engineer.

CHD.P
Enc.

SJD.
EWM.

PROGRAM OF TESTS ON THE XP-41 AIRPLANE IN THE FULL-SCALE WIND TUNNEL

Test Description
1. Drag tests of the airplane in smooth condition with:
oil-cooler scoop removed and duct openings closed
ejector chute removed
accessory exit closed
landing gear faired
intercoolers removed
canopy rail faired
carburetor scoop removed
gaps in cowling at sheet-metal joints closed
sanded walkway surface removed
cockpit ventilator opening closed and faired
blast tube fairing removed
venturi tubes through cowling removed and openings closed
aerial removed
cowling exit closed.

(a) with existing N. A. C. A. cowling.
(b) with closed streamlined spinner.

2. Drag and air-flow quantity measurements of the ship as in 1 except with cowling exit open; flaps open and closed:

(a) with existing N. A. C. A. cowling.
(b) with faired spinner and three variations of inlet nose shape.

3. Momentum measurements behind the wing to determine the profile drag for:
(a) normal wing surf ace.
(b) a section of the wing with aerodynamically smooth leading edge.
4. Boundary-layer velocity surveys to determine the transition point on the wing:
(a) normal wing surface.
(b) on the section noted in 3 (b) above.

5. Drag and air-flow quantity measurements for new oil-cooler installation under the engine.

6. Determination of the drag of the airplane in normal condition as received and the individual drag of the following component parts of the airplane:

a. oil-cooler installation
b. ejector chute
c. cooling airflow
d. landing gear fairing
e. intercooler installation
f. canopy rail
g. carburetor scoop
h. gaps in cowling at sheet-metal joints
i. sanded walkway surface
j. cockpit bentilator
k. blast tube fairings
l. cowling venturi-tubes

7. Static pressure surveys on fuselage, canopy, cowling, and wing to determine the critical compressibility velocity.

8. Measurement of the engine-cooling air quantity for the original installation.

9. Measurement of the maximum lift in the normal service landing condition and tuft surveys to observe the progression of the stall.

10. Measurement of rolling moments and stick forces over the range of aileron deflections with flaps undeflected and deflected for service landing.

Document 3-30(a-c)

(a) General Henry H. Arnold to NACA, "Full-Scale Wind-Tunnel Tests of XP-39," 9 June 1939, copy in Research Authorization (RA) file 674, LHA.

(b) Smith J. DeFrance to Chief, Aerodynamics Division, Langley Aeronautical Laboratory, "Estimated High-Speed of the XP-39 Airplane," 25 Aug. 1939, copy in RA file 674, LHA.

(c) Larry Bell, President, Bell Aircraft, to George W. Lewis, NACA, 17 Jan. 1940, copy in RA file 674, LHA.

Although the NACA's drag cleanup work usually proceeded in a typical way, each aircraft that went through the process was different, posed its own problems, and required special attention. This was certainly true in the case of the Bell XP-39 Airacobra, the eleventh in the series of military planes subjected to the NACA cleanup operation. The following string of documents takes the reader through roughly eight months of NACA investigation into the design details of the XP-39, from August 1939 to April 1940. It is important to remember that, early in this stretch of time, Nazi Germany invaded Poland, World War II began, and President Franklin D. Roosevelt called for U.S. production of an incredible 50,000 planes a year. In August 1940, the air war known as the Battle of Britain would be fought. The development of advanced combat aircraft had grown extremely urgent.

Bell Aircraft designed the XP-39 as a 400-mph fighter. In the spring of 1939, at Wright Field in Ohio, the unarmed prototype flew to a maximum speed of 390 mph at 20,000 feet. The aircraft reached this speed, however, with a gross weight of only 5,500 pounds, about a ton less than a heavily armored production P-39. That meant that the existing aircraft, when normally loaded, would have a hard time exceeding 340 mph. Still, the test performance impressed the Air Corps enough for it to issue a contract three weeks later for 13 production model YP-39s. It should be clear from the first item in the string below that Gen. Henry H. "Hap" Arnold was desperate for a new fighter and hoped that the speed of the airplane could be increased considerably by cleaning up the drag.

The second item documents some of what NACA Langley was doing with the XP-39 in the late summer of 1939. Tests in the Full-Scale Tunnel indicated that the prototype in a completely faired condition had a drag value of only 0.0150 compared to 0.0316 in the original form. If everything possible were done to reduce drag, the airplane's maximum speed might be raised by as much as 26 percent. But as NACA realized, not all of the changes were feasible for the production aircraft. Bell made what changes it could and even considered the NACA's nonaerodynamic recommendation to try a more powerful engine, one with a geared supercharger. Even without that sort of engine, the refined aircraft, designated XP-39B, managed to reach a speed of 375 mph at 15,000 feet in its first trials. (This seemingly meant a *reduction* in speed of 15 miles per hour, but the XP-39B weighed about 300 pounds more than the original, due mostly to the addition of armament.) In the third item below, Lawrence Bell, the president of Bell Aircraft, wrote to the NACA in appreciation of what he called "extraordinarily satisfactory results."

At the request of Bell and the air corps, the NACA continued to pursue drag cleanup on the airplane. Unfortunately, the XP-39B's top speed never came anywhere near 400 mph—for that matter, no version of the Airacobra ever would. In September 1940, the first YP-39, having incorporated most of the suggestions called for by the NACA, notably the installation of propeller cuffs and wheel well covers, flew to a top speed of 368 mph at 13,300 feet. Deliveries of the first production model P-39s, which were very similar to the service-test YP-39, began four months later. In 1941, the United States sent nearly 700 Airacobras to Great Britain and the Soviet Union under Lend-Lease. After the Japanese attack at Pearl Harbor, the Air Corps rushed P-39 units into action in the South Pacific.

The production P-39s could never fly more than 368 mph. This speed ceiling seems to indicate that the NACA's drag cleanup program failed, as the first P-39 prototype in test flight had reached 390 mph. But the reason for this slightly slower speed was the hundreds of pounds of weight that had to be added in the form of a bigger power plant, guns, and heavy armor plate. In other words, if the P-39 had not gone through drag testing, it would have been slower than it ultimately was, maybe as much as 25 to 30 miles per hour. Although limited in its top speed, the plane possessed reasonable stability and roll rates and maneuvered well at low altitudes. These qualities made the airplane useful in combat ground support as a strafer and as a fighter-bomber.

Document 3-30(a), General Henry H. Arnold to NACA, "Full-Scale Wind-Tunnel Tests of XP-39," 9 June 1939.

Address reply to
Chief of the Air Corps
War Department
Washington D.C.

452.1

WAR DEPARTMENT
Office of the Chief of the Air Corps (4-e)
Washington

June 9, 1939.

SUBJECT: Full Scale Wind Tunnel Tests of XP-39.

TO: National Advisory Committee for Aeronautics,
Navy Building, Washington, D.C.
(Attention: Dr. G. W. Lewis,
Directory of Aeronautical Research)

It is requested that the XP-39 be placed in the full scale wind tunnel at Langley Field, Virginia, for the purpose of determining performance characteristics of this airplane and what improvements are recommended to increase aerodynamic and performance characteristics.

H. H. Arnold
Major General, Air Corps,
Chief of the Air Corps.

Document 3-30(b), Smith J. DeFrance to Chief, Aerodynamics Division, Langley Aeronautical Laboratory, "Estimated High-Speed of the XP-39 Airplane," 25 Aug. 1939.

Langley Field, Va.,
August 25, 1939.

MEMORANDUM For Chief Aerodynamics Division.
Subject: Estimated high speed of the XP-39 airplane.

1. In accordance with a telephone conversation with Dr. Lewis on August 24, a survey of the data obtained in the full-scale tunnel on the XP-39 airplane has been made to determine the high speed with a 1,350-horsepower engine and geared supercharger. The drag coefficient for the XP-39 airplane, as tested in the tunnel at 100 miles per hour without the turbo supercharger and with the original wing cooling ducts in both wings, was found to be 0.0210. From experience on other airplanes, this value has been corrected for Reynolds number of the tests to Reynolds number of flight speed, and the value is 0.0191, which would produce a high speed of 408 miles per hour. The wing cooling duct originally installed in this airplane is not of the best design, both as to the inlet and the structure within the duct. The properties of the duct can be improved, and if the structure can be removed from the duct, the corrected high-speed drag coefficient would be reduced to 0.0180, corresponding to a high speed of 416 miles per hour. The wheels at present on this airplane are too large to be completely housed within the wing and protrude below the lower surface. By decreasing the size of the wheels so that they can be completely faired within the wing, the corrected high-speed drag coefficient would be reduced to 0.0164, corresponding to a high speed of 429 miles per hour. The speeds quoted are based on 1,350 horsepower at 16,000 feet and a propulsive efficiency of 85 percent.
2. The above figures are based on the radiator size and quantity of air required for the engine which is now in the airplane. It is not known how much additional air would be required for cooling a 1,350 horsepower engine, and if this is increased by a large amount, the high-speed values given would be reduced, but it is not believed that there would be any difficulty in obtaining at least 410 miles per hour unless a compressibility shock wave is experienced on some part of the airplane. It appears that the recommended changes to obtain the higher speed, namely, installation of a 1,350-horesepower geared supercharger engine, improved wing ducts in both wings, and smaller wheels completely faired into the wing, could easily be made on the present airplane without much delay.

Smith J. DeFrance,
Principal Aeronautical Engineer.

Document 3-30(c), Larry Bell, President, Bell Aircraft, to George W. Lewis, NACA, 17 Jan. 1940.

BELL AIRCRAFT CORP.
2050 Elmwood Avenue
Buffalo, N. Y.

January 17, 1940.

Dr. George W. Lewis
National Advisory Committee for Aeronautics
Munitions Building
Washington, D. C.

My dear Dr. Lewis:

I wanted to tell you that after we had incorporated the changes in the XP-39B, as a result of the wind tunnel tests at Langley Field, we are getting extraordinary satisfactory results. From all indications the XP-39 will do over 400 m.p.h., with 1150 H.P. All of the changes were improvements and we have eliminated a million and one problems by the removal of the turbo supercharger.

The cooling system is the most efficient thing we have seen. The inlet ducts on the Prestone Radiator are closed up to 3% and the engine is still overcooling.

I want to convey to you personally and your entire organization, both at Washington and at the Laboratory, our very deep appreciation of your assistance in obtaining these very satisfactory results.

With kindest regards,

Sincerely yours,

Larry Bell
President

Document 3-31(a-c)

(a) Excerpts from Clinton H. Dearborn, "The Effect of Rivet Heads on the Characteristics of a 6 By 36 Foot Clark Y Metal Airfoil," NACA *Technical Note* 461, May 1933.

(b) Excerpts from Manley J. Hood, "The Effects of Some Common Surface Irregularities on Wing Drag," NACA *Technical Note* 695, March 1939.

(c) Charles Peyton Autry, Boeing Aircraft Company, "Drag of Riveted Wings," in *Aviation* (May 1941): 53-54.

In the 1930s, aeronautical engineers came to understand that low drag required more than a correct wing profile or overall effective aerodynamic shape. Without extremely smooth surface conditions, an airplane's design would be seriously compromised, especially as aircraft speed increased to well over 200 miles per hour. Wrinkles in metal, dents, even scratches and the grain embedded in paint upset the airflow. These surface irregularities caused various instabilities. They generated rising pressures in places where pressure otherwise would be falling, and in the critical boundary layer, they hastened transition from laminar to turbulent flow. Means had to be found to smooth out all these bumps and nicks if an airplane were ever to perform to the maximum potential foreseen by its designer.

One of the most upsetting surface features damaging the aerodynamic performance of an airplane once more and more aircraft came to be built out of stressed-skin metal were rivet heads. As indicated in the documents below, NACA researchers began looking into this problem in the early 1930s (although the first proposal for flush riveting seems to have been in a patent proposal made by Charles Ward Hall, president of the Hall-Aluminum Aircraft Corporation in Buffalo, New York, in 1926). They found that the dome shapes of rivet heads could increase a wing's parasite drag (i.e., the drag forces that are not formed in the production of lift) by as much as 25 to 30 percent over that of a smooth wing. If happening with a big transport plane, it could mean a drain on an engine amounting to about 150 horsepower. If the rivets were countersunk, the increase in drag would be considerably less, only 5 to 6 percent.

Both the NACA and the aircraft industry experimented throughout the late 1930s with various means by which to improve the surface finish of airplanes. They tried slicing off rivet heads and eventually moved to an advanced technique of flush riveting. (For a detailed analysis of this development, see Walter G. Vincenti, "Design and Production: The Innovation of Flush Riveting in American Airplanes, 1930-1950," in Vincenti's *What Engineers Know and How They Know It*, pp. 170-199.)

The three documents below, two from the NACA and one by a Boeing engineer, report on the problem of protruding rivet heads and other surface irregularities especially as it related to wing drag. It is noteworthy that none of the revolutionary American airplanes of the early 1930s, such the Northrop Alpha, Boeing Monomail, as well as the DC-1, DC-2, and DC-3, incorporated flush rivets to any significant extent. They all still had the problem of rivets protruding into the airstream. By the early 1940s, however, virtually all U.S.-built metal airplanes benefited from flush rivets. As with the NACA's drag reduction program, the development of flush riveting represents the final stage in the airplane design revolution, during which the last minor details of reinventing the airplane took form.

Document 3-31(a), Excerpts from Clinton H. Dearborn, "The Effect of Rivet Heads on the Characteristics of a 6 By 36 Foot Clark Y Metal Airfoil," NACA **Technical Note** *461, May 1933.*

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE NO. 461

THE EFFECT OF RIVET HEADS ON THE CHARACTERISTICS OF A 6 BY 36 FOOT Y METAL AIRFOIL

By Clinton H. Dearborn

SUMMARY

An investigation was conducted in the N.A.C.A. full-scale wind tunnel to determine the effects of exposed rivet heads on the aerodynamic characteristics of a metal-covered 6 by 36 foot Clark Y airfoil. Lead punching simulating 1/8 inch rivet heads were attached in full-span rows at a pitch of 1 inch at various chord positions. Tests were made at velocities varying from 40 to 120 miles per hour to investigate the scale effect.

Rivets at the 5 percent chord position on the upper surface of the airfoil produced the greatest increase in drag for a single row. Nine rows of rivets on both surfaces, simulating rivet spacing of multispar construction, increased the drag coefficient by a constant amount at velocities between 100 and 120 miles per hour. Extrapolation of the curves indicates that the same increase would be obtained at speeds over 120 miles per hour. Accordingly, if rivets spaced the same as those on

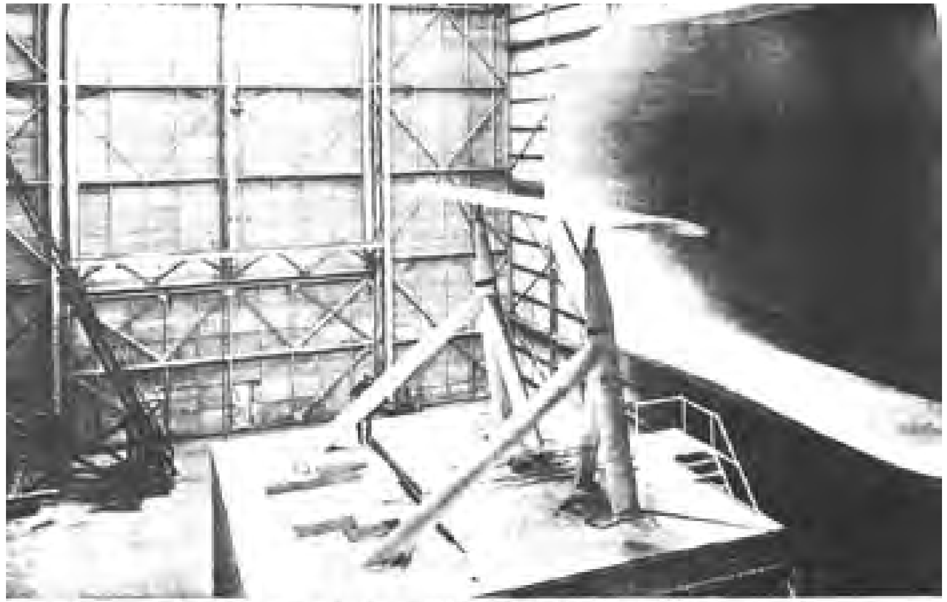


Figure 1.-The 6 by 36 foot Clark Y airfoil mounted on balance.

the test airfoil were used on a Clark Y wing of 300 square feet area and operated at 200 miles per hour the drag would be increased over that for the smooth wing by 55 pounds and the power required would be increased by 29 horsepower. The effect on the lift characteristics due to the rivets was found to be negligible.

INTRODUCTION

One of the most promising possibilities of improving the performance of airplanes lies in the reduction of drag. A recent airfoil investigation conducted in the N.A.C.A. variable-density wind tunnel on full-span protuberances (reference 1) and on short-span protuberances, including wing fittings (reference 2), showed that small protuberances have an important effect on the aerodynamic characteristics of an airfoil. This investigation was extended to include the determination of the effects caused by exposed rivet heads of a type common to metal airplane wing construction. The latter tests were conducted in the full-scale wind tunnel on a 6 by 36 foot airfoil.

Lead punchings formed to simulate rivet heads were attached to the airfoil first in single rows at various chord positions on the upper surface, then in nine rows on the upper surface, and finally in nine rows on both surfaces.

APPARATUS AND METHODS

The 6 by 36 foot Clark Y airfoil used in this investigation is shown mounted in the tunnel in Figure 1. Two structural steel H beams with steel-angle connecting members form the primary structure of the airfoil; the ribs and skin are of 1/16 inch

Figure 2.-Nine rows of rivet heads on upper surface of airfoil.

sheet aluminum. The outer surface of the skin was made as smooth as practicable by the use of butt joints and countersunk attaching screws. Rivet heads were simulated by gluing lead punchings to the surface of the airfoil as shown in Figure 2. These punchings were made from sheet lead with a die conforming in dimensions to the head of a 1/8 inch brazier head rivet. (Fig. 3.)

The airfoil was supported on the balance by two braced struts shown in Figure 1. All members were encased in fairings except the tops of the supports and the short struts for changing the angle of attack. The exposed members were made as small as practicable so that the tare drag would be a small percentage of the minimum drag of the airfoil. Tare-drag tests in which the airfoil was independently supported showed that the drag of the supports was only 4 percent of the minimum drag of the plain airfoils at 100 miles per hour. A description of the balance will be given with the description of the tunnel now being prepared as a Technical Report.

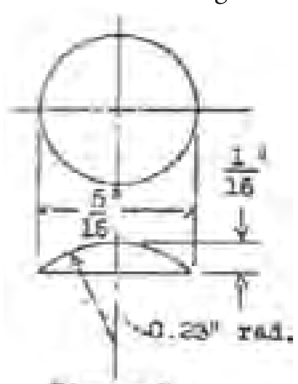


Figure 3.

TESTS

The effect on the drag of the airfoil of a single row of rivet heads at the leading edge, and at 5, 15, and 30 percent of the chord back of the leading edge on the upper surface was first investigated. The single rows, as well as the combination of rows at 10 percent chord intervals tested later, extended over the full span of the airfoil with the rivets spaced 1 inch apart.

Starting with the 5 percent chord position, nine rows were attached to the upper surface at increments of 10 percent of the chord and the drag measured. Nine

additional rows of rivet heads were later attached to the lower surface at the same chord positions as those on the upper surface and the drag again measured. The last condition of test is representative of the spacing of rivets on metal-covered wings of multispar construction. These tests were made at a dynamic pressure of 7.8 pounds per square foot, which corresponds to an indicated velocity of 55 miles per hour.

The plain airfoil and the airfoil with the nine rows of rivets on both the upper and lower surfaces were next tested at angles of attack in the region of minimum drag over a speed range from 40 to 120 miles per hour to investigate the magnitude of the scale effect. The effect of the rivets on lift was investigated by testing the airfoils from -8 degrees to 21 degrees angle of attack at a dynamic pressure of 16 pounds per square foot (79.2 miles per hour indicated velocity).

RESULTS AND DISCUSSION

Tunnel jet-boundary corrections have not been applied to the results presented in this report because the differences in lift were negligible and the differences in drag therefore would not be affected.

A comparison of the results obtained from the plain airfoil with these obtained with a single row of rivets at the various chord positions on the upper surface showed that the single row at the 5 percent chord position produced the greatest increase in minimum drag. This increase in drag amounted to 19 percent of the minimum drag of the plain airfoil. (Fig. 4.)

The nine rows of rivets on the upper surface of the airfoil at 10 percent chord intervals extending from the 5 to 85 percent chord positions caused a 21 percent

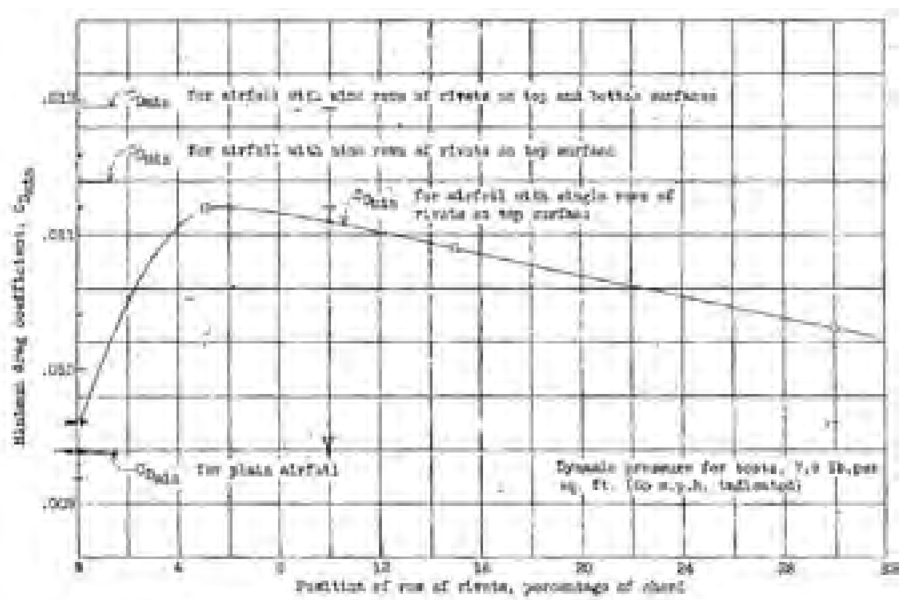


Figure 4.-Minimum drag coefficient for airfoil with rivet arrangements tested.

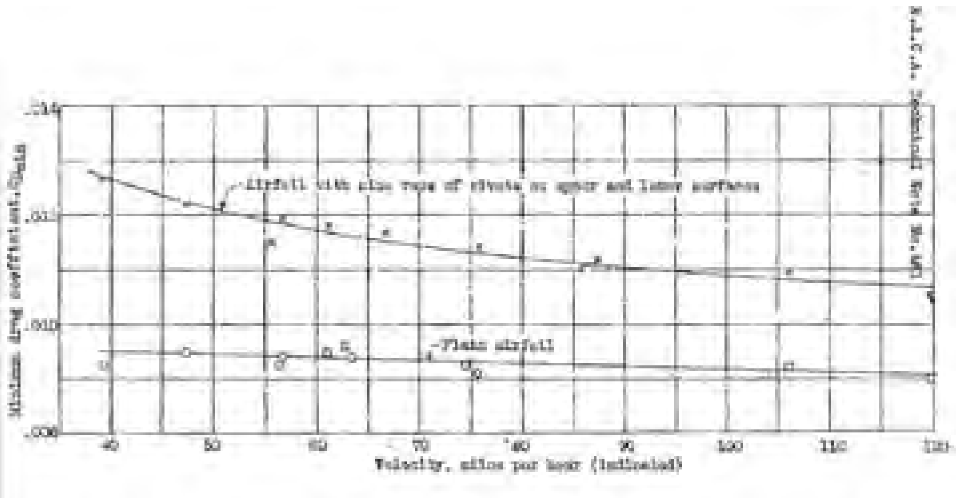


Figure 5.-Scale effect on $C_{D_{min}}$ for the plain airfoil and airfoil with 16 rows of rivets.

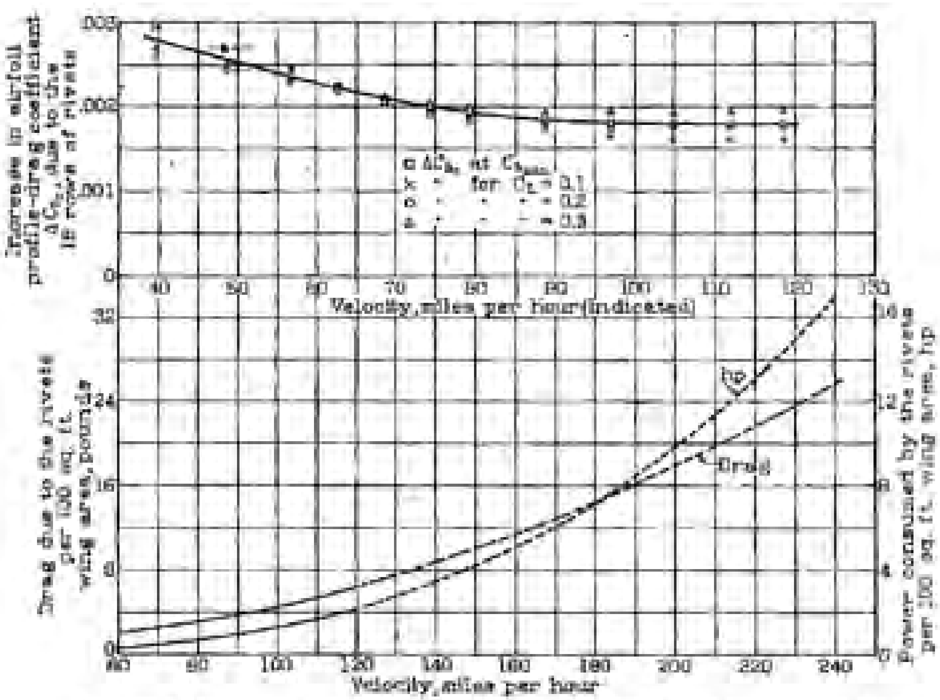


Figure 6.-Increase in profile-drag coefficient, drag, and power required due to the rivets.

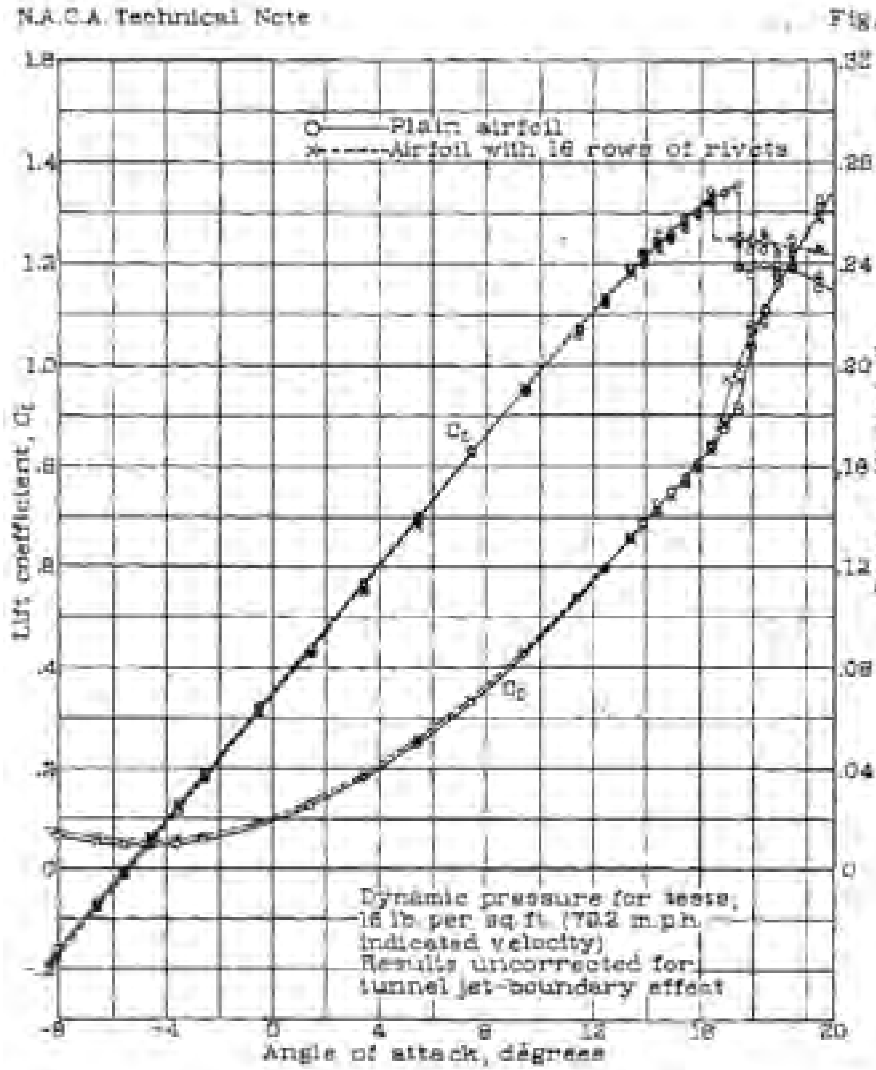


Figure 7.-Variation of lift and drag with angle of attack for plain airfoil and airfoil with rivets.

Fig.8

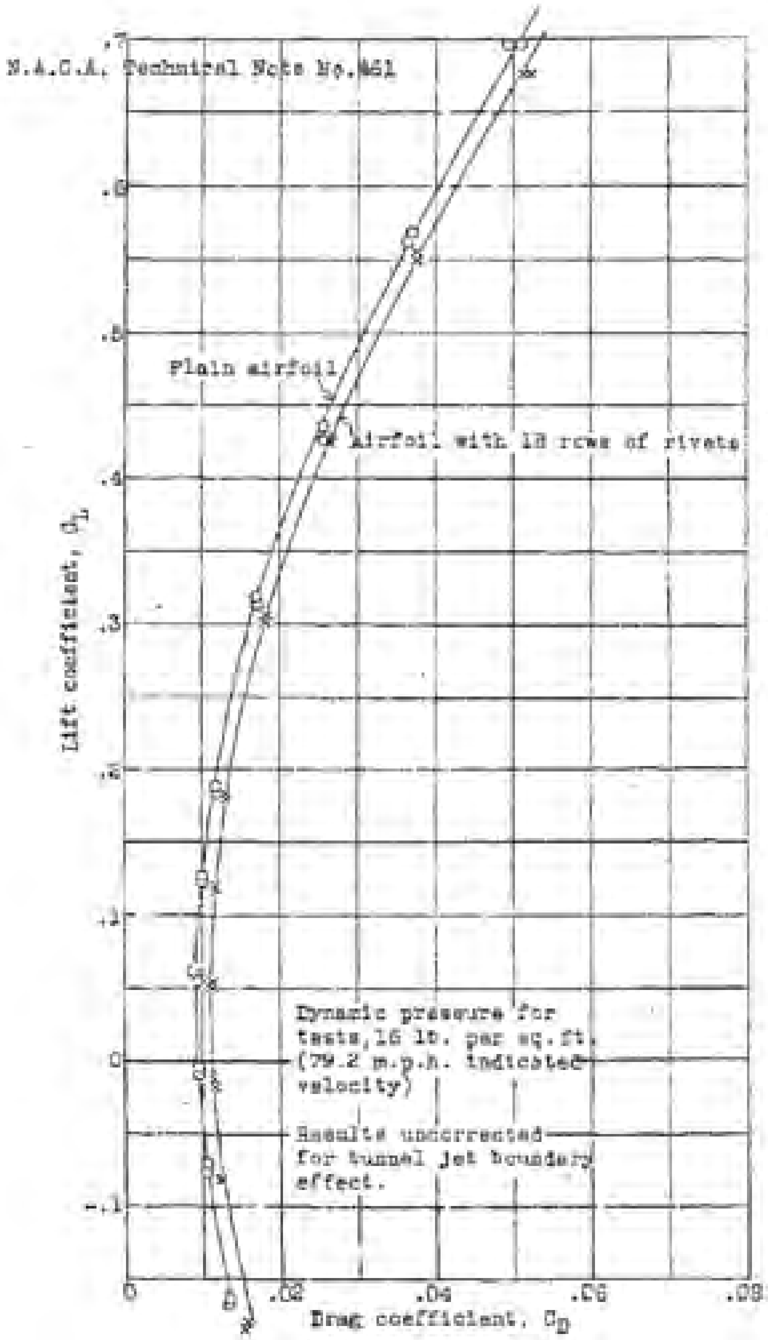


Figure 8.- Polars for plain airfoil and airfoil with rivets.

increase in minimum drag. This increase in drag is small compared with the increase of about 60 percent that would be obtained from the summation of increases in minimum drag for single rows shown in Figure 4. The fact that the increases in drag due to the single rows failed to become additive for a combination of the same rows was probably due to a serious disturbing effect in the boundary layer caused by the first row of rivets.

The nine rows of rivets on both surfaces produced an increase of 27 percent in drag. This is less than one-third more than the amount obtained with the rivets on the upper surface alone.

The preceding results were obtained from tests at 55 miles per hour. It will be noted in Figure 5 that the increase in minimum drag at 120 miles per hour for the airfoil with rivets on both surfaces is only 18 percent of the minimum drag of the plain airfoil. This difference in increase of minimum drag may be attributed to scale effect; it may be assumed that the same scale effect would be present with the single row of rivets at the 5 percent chord position and with the nine rows on the upper surface alone and that the percentage increase in minimum drag for these conditions would be proportionally reduced at the higher speed.

Figure 5 shows a greater scale effect for the riveted airfoil than for the plain airfoil at the lower test velocities. However, at the higher velocities this difference in the scale effect disappears, resulting in a constant difference in minimum drag. Differences of the minimum drag coefficients and drag coefficients corresponding to the lift coefficients of 0.1, 0.2, and 0.3 for the two airfoils throughout the speed range are plotted in Figure 6. The increase in the drag coefficient due to the rivets is, for practical purposes, due solely to an increase in the profile drag, as indicated by the parallelism of the polars in Figure 8. The difference in drag coefficients at velocities between 100 and 120 miles per hour is 0.0018. It appears reasonable to assume that this difference in drag coefficients would remain the same at velocities even higher than those employed for this investigation.

The effect of the rivets on lift is practically negligible, as shown in Figure 7. The burble angle occurs 1 degree earlier with a decrease of about 1 percent in the maximum lift coefficient.

The significance of the increase in profile drag may well be illustrated by estimating what effect it would have on the performance of an airplane. For this purpose an airplane with the following specifications was chosen and the assumption made that the wings were metal covered with exposed rivet heads on both surfaces in the same locations as those covered by the tests.

Wing area	300 sq. ft.
Wing section	Clark Y
Engine	500 b.hp
Fuel consumption	0.5 lb./b.hp-hr.
Propulsive efficiency	80 percent

High speed	200 m.p.h.
Cruising speed	170 m.p.h.

These specifications are representative of a modern high-speed transport or a military observation airplane.

The extrapolated drag curve in Figure 6 shows that the increase in drag caused by the rivets would be 40 pounds at the cruising speed of 170 miles per hour and 55 pounds at the high speed of 200 miles per hour. These drag forces, taking the propulsive efficiency into account, would consume 23 and 37 brake horsepower, respectively, at the cruising and high speeds. The increase in fuel consumption due to the rivets at the cruising speed, based on a weight of 6 pounds per gallon, would be 1.9 gallons per hour. This amount represents about 7 percent of the fuel consumption at the cruising speed. The high speed would be increased from 200 to 205 miles per hour by the elimination of the exposed rivet heads.

CONCLUSIONS

1. A single row of rivets located at the 5 percent chord position on the upper surface of the airfoil produced a greater increase in the minimum drag than any other position investigated.
2. Rivets added on the upper surface of the airfoil back of a single row at the 5 percent chord position had little effect on drag.
3. Nine rows of rivets on the lower surface increased the drag less than one-third of the amount that the same number of rows did on the upper surface.
4. The effect of rivets on maximum lift was negligible.
5. Exposed rivet heads of the type and spacing investigated would have an appreciable detrimental effect on the fuel consumption and high speed of an airplane.

Langley Memorial Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Field, Va., February 4, 1933.

Document 3-31(b), Excerpts from Manley J. Hood, "The Effects of Some Common Surface Irregularities on Wing Drag," NACA* Technical Note *695, March 1939.

TECHNICAL NOTES
NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

NO. 695

THE EFFECTS OF SOME COMMON SURFACE IRREGULARITIES ON WING DRAG

By Manley J. Hood

Langley Memorial Aeronautical Laboratory

Washington
March 1939

SUMMARY

The N.A.C.A. has conducted tests to provide more complete data than were previously available for estimating the effects of common surface irregularities on wing drag. The irregularities investigated included: brazier-head and countersunk rivets, spot welds, several types of sheet-metal joints, and surface roughness. Tests were also conducted to determine the overall effect of manufacturing irregularities incidental to riveted aluminum alloy and to spot-welded stainless-steel construction. The tests were made in the 8-foot high-speed wind tunnel at Reynolds numbers up to 18,000,000.

The results show that any of the surface irregularities investigated may increase wing drag enough to have important adverse effects on high-speed performance and economy.

A method of estimating increases in wing drag caused by brazier-head rivets and lapped joints under conditions outside the range of the tests is suggested. Estimated drag increases due to rivets and lapped joints on a wing of 20-foot chord flying at 250 miles per hour are shown.

INTRODUCTION

Improved streamlining has reduced form drag so much that on modern airplanes skin friction often constitutes more than half of the total drag. It is therefore important that skin friction be made as small as possible. It is obvious that protruding rivet heads, roughness, and other surface irregularities will increase skin friction. A knowledge of the magnitude of the drag increases is necessary before the designer can decide to what extent it is economical to eliminate these irregularities from the surfaces exposed to air flow.

Previous tests have shown that rivet heads (reference 1), certain arbitrary protuberances (references 2 and 3), and roughness (references 4 and 5) increase the drag of wings by important amounts. The N.A.C.A. has conducted additional tests to provide more complete data on the drag caused by irregularities of types commonly occurring on airplane wings so that their effects may be more accurately estimated. Some of the results of these tests are presented in this report. The irregularities for which data are presented include protruding rivet heads, spot welds, several types of lapped joints, imperfections in butted joints, surface roughness, and manufacturing irregularities. Most of the irregularities were tested in various systematic arrangements over different portions of the surface of an airfoil of N.A.C.A. 23012 section and 5-foot chord. The tests were made at lift coefficients of 0 to 0.30 and at Reynolds numbers up to 18,000,000.

A method of applying the results to predict the drag increases due to rivet heads and lapped joints on airplane wings under conditions outside of the range of these tests is presented.

APPARATUS

The tests were conducted in the N.A.C.A. 8-foot high-speed wind tunnel. The air flow in the closed circular test section of this wind tunnel is quite uniform and the turbulence of the air flow is so small that sphere tests have shown virtually the same critical Reynolds number as in free air (reference 6).

The airfoils used for the tests were of N.A.C.A. 23012 section and, except for a few supplementary tests of a 2-foot airfoil, they were of 5-foot chord. The noses of the airfoils were bare steel to reduce erosion; the rest of the surface was bare steel in some instances and painted in others. Except for the irregularities being tested, the surfaces were aerodynamically smooth; that is, further polishing would not reduce the drag.

The airfoils were mounted horizontally across the center of the test section as shown in figure 1. The tunnel-wall interference was reduced by enclosing the ends of the airfoils in shields that did not touch the airfoils or their supports but were supported independently of the balance. The span of each shield was 10 inches and the active span of the airfoils between the shields was 6 feet.

Actual rivets with their shanks pressed into holes drilled in the airfoil were used for most of the tests of protruding rivet heads but for some of the tests it was found more convenient to simulate the rivet heads with properly shaped lead punchings cemented to the surface with airplane dope. Countersunk rivets were simulated by annular cuts in the surface of the airfoil to represent the indentations around rivet heads that result from making countersinks by forming the sheet metal with punches instead of by cutting. Details of the rivets and the simulations are shown in figure 2. The rivet-head simulations tested on the 2-foot airfoil were two-fifths as large as the heads of the 3/32-inch brazier-head rivets.

Spot welds were simulated on the otherwise smooth model by depressions of the dimensions shown on figure 3. The cylindrical simulations were cut in the surface and the spherical simulations were formed by filling the cylindrical simulations with plasticine and forming the depressions with tools having spherical ends.

Most of the lapped joints were represented by cuts made in the surface of the airfoil but, in addition to this method, the plain lap facing aft was also simulated by a ridge built up on top of the normal airfoil surface with paper and lacquer-base glazing putty. Gaps such as occur between the edges of sheets in butt-jointed construction were simulated by square-edge grooves cut spanwise in the surface of the airfoil. Adjoining edges of the gaps were of equal height, representing construction in which the butted sheets are of exactly equal thickness. The dimensions of the simulations of lapped and butted joints are shown in figure 4. All the simulations represented sheets 0.018 inch (0.0003 chord) thick.

The chord positions at which sheet-metal joints and spanwise rows of rivets and spot welds were tested are shown in figure 5. The spanwise pitch of the rivets and spot welds was ¾ inch (0.0125 chord) except where otherwise noted.

Photomicrographs of samples of the different surface roughnesses tested, all to the same magnification, are shown in figure 6. The carborundum-covered surfaces were produced by spraying carborundum grains mixed with thin shellac onto the smooth airfoils. The common designations of the grain sizes are:

Average grain size k(in.)	Carborundum Company's designation
0.0037	180
.0013	FF
.0005	800-RA

From figure 6 it is apparent that the 0.0005-inch grains were piled on top of each other in such a manner that the degree of roughness was not equivalent to the grain size, as was the case with the larger grains. The photomicrographs also indicate that the shellac used to hold the grains was sufficiently thin that the effective size and shape of the grains were not appreciably changed. The sizes of the grains were determined from measurements made with a microscope and from measurements of the photomicrographs. The density (spacing) of the grains varied somewhat over the airfoils but the photomicrographs represent average conditions. The spray-painted surface was produced by spraying a lacquer-base primer surface onto the airfoil, probably a little rougher than is common practice. The sandpapering was done with No. 400 sandpaper lubricated with water. No attempt was made to limit the sandpapering strokes to any one direction but chordwise strokes predominated. The surface was polished by rubbing with a polish of the type used in polishing automobiles, waxing, and rubbing with a soft cloth.

Two airfoils, one of riveted aluminum-alloy construction and the other of spot-welded stainless steel, were tested to obtain a measure of the over-all effect of manufacturing irregularities incidental to conventional metal-wing construction. These "service wings" were both made by manufacturers accustomed to the respective types of construction involved. The manufacturers were instructed to employ conventional design, tolerances, and workmanship in order to make the models as nearly as possible representative of actual wings being produced at that time (1936). The riveted model employed the same rivet size and arrangement and the same lapped-joint positions as were tested in one instance on the more accurate wind-tunnel model but the thickness of the lapped sheets was 0.032 inch. The skin of the stainless steel model was 0.015 inch thick on the forward 45 percent and 0.0008 inch thick on the rear 55 percent. The average dimensions of the spot welds are shown in figure 3. The arrangement of spot welds and lapped joints on the stainless-steel model is shown in figure 7. Except for the discrepancy in the profile of the riveted service wing shown in figure 8, there were no departures from true profile large enough to have important effects. Figures 9 and 10 are photographs of the service wings arranged to reflect the image of a lattice so as to show the irregularities of the surfaces. The riveted service wing was furnished by the Bureau of Aeronautics, Navy Department, and the stainless-steel wing was furnished by the U. S. Army Air Corps.

METHOD

For each arrangement of surface irregularities tested, the lift, the drag, and the pitching moment were determined at left coefficients of approximately 0, 0.15, and 0.30, respectively. The tests at lift coefficients of 0.15 and 0.30 were made at speeds varying from 80 to 370 and 80 to 270 miles per hour, respectively, the upper limit in each case producing a wing loading of approximately 50 pounds per square foot. For the tests at zero lift, the speed was varied from 80 to about 430, and in some instances 500, was varied from 80 to about 430, and in some instances 500, miles per hour. At the highest speed compressibility effects were so large that the drag coefficient increased very rapidly as the speed was increased. The drag of the smooth airfoil was checked frequently during the tests.

Because of the high test speeds employed, the method used for determining dynamic pressure, air speed, and Reynolds number in the N.A.C.A. 8-foot high-speed wind tunnel must allow for compressibility effects. Bernoulli's equation for a compressible fluid, in a form given in reference 7, is

$$P_a = P_s + ½\, \rho_s V_s^2\, (1 + ¼\, M^2 + 1/40\, M^4 + 1/1600\, M^6 \ldots.)$$

where P_a is the atmospheric pressure which, in the case discussed in reference 7, was virtually equal to the total pressure in the test section.

P_s, static pressure in the test section.

ρ_s, density of air in test section.

V_s, air speed in test section.

M, Mach number (the ratio of the air speed to the speed of sound in the air).

The quantity within the parentheses is the factor by which the impact pressure (q_c) shown by a pitot-static tube can be divided to give true dynamic pressure ($q = ½\, \rho\, V^2$). This factor is called the "compressibility factor" and is often designated by $(1 + \eta)$. Accordingly,

$$(1 + \eta) = 1 + ¼\, M^2 + 1/40\, M^4 + 1/1600\, M^6 \ldots.$$

Substituting first $\sqrt{}1.4\, P_s/\rho_s$ for the speed of sound, then 2q for $\rho_s V_s^2$, and finally, $q_c/(1 + \eta)$ for q gives,

$$(1 + \eta) = 1 + 0.357\, (q_c/P_s(1 + \eta) + 0.051\, [q_c/P_s(1 + \eta)]^2 + 0.0018\, [q_c/P_s(1 + \eta)]^2$$

From this relation, the compressibility factor $(1 + \eta)$ is computed in terms of q_c/P_s and plotted for use in computing results.

During tests, measurements are made of the pressure difference $P_1 - P_s$, where P_1 is the static pressure in the low-speed part of the return passage. From this pressure difference, qc for the model position is computed in accordance with a relation previously determined by pitot-static surveys of the air flow in the test section. The absolute value of P_s is computed from the barometric pressure, a previously determined value of $P_a - P_1$, and the measured pressure difference $P_1 - P_s$. The ratio q_c/P_s is then computed and, from this ratio and the curve described in the proceeding paragraph, $(1 + \eta)$ is determined. The true dynamic pressure on which force and moment coefficients are based is then computed from the relation

$$q = q_c/(1 + \eta)$$

The air temperature in the slow-speed part of the return passage, T_1, is measured with remote indicating thermometers. From T_1, P_1, and P_s, the temperature and the density of the air in the test section, T_s and ρ_s, are computed on the assumption that the flow is adiabatic. The air speed in the test section is then easily computed from ρ_s and q. The speed of sound in the air in the test section in miles per hour is $33.5\sqrt{T_s}$ where T_s is the absolute temperature in Fahrenheit degrees. The viscosity of the air follows from T_s and, since ρ_s has already been determined, the Reynolds number can be computed.

The assumption that the flow is adiabatic between the slow-speed part of the return passage and the test section is supported by tests which have shown that, except in the boundary layer near the tunnel walls, there is no appreciable difference between the total pressures at these two sections.

When the air in the wind tunnel is cool and its relative humidity is moderately high, fog condenses in the test section when the tunnel is operated at high speeds. Such condensation has appreciable effects on the thermodynamics of the air flow. When this condition is encountered, the tunnel is operated until the air becomes warm enough to dissipate the fog before test data are taken. Aside from this precaution, no allowance is made for the effects of humidity.

Air-flow measurements ahead of the model have indicated that blocking effects are unimportant under the conditions of these tests.

PRECISION

The only known systematic errors affecting the results herein presented are due to errors in the dynamic pressure resulting from constriction by the model of the flow through the test section. No correction for constriction has been applied because its magnitude is not yet accurately known. Preliminary tests have indicated, however, that it is not more than 6 percent at speeds up to 270 miles per hour or 9 percent up to 500 miles per hour. The drag increases herein presented may, therefore, be too high by 6 percent at the lower speeds and 9 percent at the higher speeds. Since most of the increases are small relative to the smooth-wing drag, these errors are generally unimportant.

The scatter of experimental points for separate determinations of the smooth-wing drag indicates that random errors in drag coefficients generally do not exceed ±0.0001, corresponding to ±1.4 percent of the smooth-wing drag; however, at speeds below 100 and above 400 miles per hour and at all speeds at a lift coefficient of 0.30, the variations are about twice this large.

RESULTS AND DISCUSSION

Method of Presentation

All results are presented in terms of increases in drag coefficient C_D, the amount by which the drag coefficient for any condition exceeded the drag coefficient of the smooth airfoil at the same speed and angle of attack. No corrections for tunnel-wall effects have been applied because this method of presentation makes corrections unnecessary except those due to constriction effects, which have been discussed under Precision.

Because of the rapid variation of drag coefficient with Mach number at the high test speeds, it was necessary to compute the drag differences at equal values of the Mach number. Equal Mach numbers correspond approximately, but not exactly, to equal Reynolds numbers for a given size model. This variation from test to test of the Reynolds number corresponding to a given Mach number is, however, small enough to be of little consequence and the results are therefore presented in terms of Reynolds numbers representing the averages for the various tests. For the 5-foot airfoils, an average Reynolds number of 10,300,000 corresponded to a Mach number of 0.3 and an average Reynolds number of 17,600,000 to a Mach number of 0.6.

That the effect of compressibility on the drag increments herein presented may be neglected is indicated by figure 11, which shows that rivets of geometrically similar size and arrangement increased the drag coefficients of the 2-foot and the 5-foot airfoils by substantially equal amounts at equal Reynolds numbers even though the Mach numbers differed widely. The results may therefore be applied solely on the basis of Reynolds number. In accordance with Reynolds' law, rivet size and arrangement must be considered in terms of wing chord when the results are used directly to predict the drag of rivets on wings of different chord.

RIVETS

Figure 12 shows the drag increments due to the various types and sizes of rivets in 13 spanwise rows on each surface. The spanwise pitch in each row was ¾ inch (0.0125 chord) and the most forward row on each surface was 4 percent of the chord behind the leading edge. The approximate percentage increments are spotted on the curves for a few representative points to aid in visualizing the magnitude of these drag increments. It is obvious that the drag increments are large enough to have important adverse effects on performance, being as much as 27 percent of the smooth-wing drag for the 3/32-inch brazier-head rivets. Even countersunk rivets may increase wing drag by amounts too large to be neglected.

Rivet heads increase the drag of a wing in two ways: first, each rivet head, being exposed to the air flow, has some drag in itself; and, second, rivets on the forward part of a wing cause the transition point to move forward and thereby cause an increase in skin friction. That this second factor may be responsible for a large part of the drag increase is indicated by figure 13. This figure shows that, when the front row of rivets was 4 percent of the chord from the leading edge, the rivet drag was reduced only slightly by increasing the spanwise pitch from 3/4 inch to 1½ inches. When the forward rows of rivets were 28 percent of the chord from the leading edge, they were behind the transition point and, as the pitch was varied, the rivet drag varied in proportion to the number of rivets on the wing.

Figures 14 and 15 show the variation of rivet drag with position of the forward rows. The position of the forward rows was varied by adding or removing rows at the front, rows behind the most forward ones always remaining in place at the chord position shown in figure 5. These figures illustrate again the importance of the shift of the transition point caused by rivets. As rows of rivets are added, starting at the back and progressing forward, the drag increases slowly until the region where transition occurs on the smooth wing (fig. 15) is reached, after which the drag increases much more rapidly. The shaded area in figure 15 indicates the excess of the rivet drag over what it would be if the original rate of increase were maintained forward of the transition point. This excess drag is plotted in figure 16 along with the computed difference between turbulent and laminar flat-plate skin friction for the Reynolds numbers involved. The agreement of the computed and the experimental curves indicated that the rapid rise of the curves of figures 14 and 15 forward of the 25-percent-chord position is largely due to forward movement of the transition point.

From the test results shown in figures 12 to 15, it can be concluded that the following measures are most effective in keeping rivet drag small:

(a) Rivets should be as far back on the wing surface as possible. It is especially important that there be no rivets forward of the point at which transition occurs on the smooth wing. With the rivet arrangement shown in figure 5, for example, 70 percent of the rivet drag was caused by the rivets on the forward 30 percent of the airfoil.

(b) Rivet heads should be as small as possible.

(c) There should be as few exposed rivet heads as possible; reducing the number by increasing the pitch in spanwise rows forward of the transition point, however, has little effect unless the pitch is larger than 0.025 chord.

SPOT WELDS

The drag increases caused by the three sizes of spot welds are shown in figure 17. These increases are smaller than those caused by any of the protruding rivet heads but are large enough so that an effort should be made to keep the depressions at spot welds as shallow as possible. The two points shown for the 0.012-inch-deep spot welds in figure 14 indicate that spot welds, as well as rivets, should be avoided especially forward of the smooth-wing transition point.

SHEET-METAL JOINTS

Increases in drag due to the presence of sheet-metal joints are shown in figure 18 for lapped joints and in figure 19 for butted joints with gaps between the edges of the sheets. The conventional plain laps facing aft are only slightly superior to the same type facing forward. Joggled laps cause only about one-half as much drag as either type of plain lap. It is important that the laps be accurately formed to lie inside the true airfoil profile because, if they rise outside the true profile (see fig. 4), appreciably more drag may be created.

Rounding the exposed edges of the sheets to a radius equal to the sheet thickness reduced the drag of the forward-facing laps to about the same magnitude as that of the square-edge laps facing aft. Rounding the edges of the plain laps facing aft to this radius reduced the drag very slightly but fairing the edges back to a width equal to three times the sheet thickness reduced the drag caused by this type of lap to about two-thirds of its previous magnitude.

The variation of lapped-joint drag with chord position of the forward laps was similar to that shown for rivets in figure 14.

From figure 18, it is seen that the drag increments caused by rivets and laps are not additive. Presumably, this condition is due to the fact that the rivets caused transition to take place at the most forward row. Adding laps back of this point therefore had no further effect on the transition point, and the additional drag due to the presence of the laps was only the direct drag of these laps in the turbulent boundary layer.

The 0.024-inch square-edge grooves caused only small increases in drag (fig. 19), indicating that only reasonably small tolerances need be maintained on the permissible gap between the edges of butted sheets.

ROUGHNESS

From figure 20, it is evident that even a small degree of surface roughness increases wing drag sufficiently to have serious adverse effects on high-speed performance and economy. Even the roughness due to spray painting may increase wing drag 10 to 14 percent in the high-speed and cruising range. Except at the lowest speeds, the 0.0013-inch roughness increased the drag considerably more than 3/32-inch brazier-head rivets. As in the case of other surface irregularities, it is especially important to keep the forward portions of wings smooth, but roughness may cause important increases in wing drag even when entirely behind the smooth-wing transition point. (See fig. 21.)

In the range of these tests, the drag increases caused by surface roughness vary considerably with scale (fig. 20). At the lower speeds, the drag due to roughness decreases rapidly as speed is reduced and the curves indicate that, for each degree of roughness, there is a speed or Reynolds number below which that roughness has no effect on drag. Conversely, it is indicated that for every speed or Reynolds number there is a limiting "permissible roughness," which will cause no increase in drag. The existence of such a permissible roughness has been shown by other tests and by the theory that roughness wholly submerged in the laminar sublayer will not increase the drag (reference 8 and 9). Estimating permissible roughness from the results herein reported involves questionable extrapolations, but nevertheless the results do indicate the same order of magnitude of permissible roughness as is tabulated for a flat plate with a wholly turbulent boundary layer in reference 8; even though, in the case of the airfoil, part of the boundary layer is laminar. This agreement suggests the conclusion that neither the 0.0005-inch grains nor the roughness due to spray painting had any great effect on the transition point. This conclusion is supported by figure 21 because the curves for the two smaller degrees of roughness do not rise so rapidly forward of the 25-percent-chord position as they would if the transition point moved forward with the roughness. There is need for further investigation of the degree of roughness a wing surface may have before premature transition is induced. The permissible roughness in the laminar boundary layer probably varies widely with dynamic scale, airfoil pressure gradient, and initial air-stream turbulence, so the indicated conclusion should not be applied where conditions differ from those of the present tests.

It is of practical interest to note that, within the limits of accuracy of the tests, the drag of the sandpapered airfoil was the same as that of the highly polished airfoil. For airplanes to have the smallest possible skin friction the surfaces must be smooth but need not be highly polished.

Because of the large scale effect on the drag of rough surface (fig. 20), it is essential that experimental investigations of the effects of surface roughness be made at large scale. Degrees of roughness large enough to have serious adverse effects under flight conditions may have no effect whatsoever under conditions of small-scale tests.

The effects of roughness on airplane wings can be estimated only approximately from the results of these tests because the effects depend on grain shape and grain spacing as well as grain size. The variation of drag with grain spacing is especially large (reference 10).

MANUFACTURING IRREGULARITIES

Figure 22, showing the results of tests of the two service wings (figs. 9 and 10), indicates that manufacturing irregularities may cause important drag increases over and above those due to the rivets and the lapped joints. For example, the drag of the riveted service wing was 42 percent greater than that of a smooth accurate wing (at a lift coefficient of 0.15 and a Reynolds number of 10,300,000), whereas the same arrangement of rivets and lapped joints caused only 29 percent excess drag on the more accurate wind-tunnel model. The rivets on the two models were identical but the lapped sheets were 0.032 inch thick on the service wing compared with 0.018 inch on the wind-tunnel model. It is estimated, by a method hereinafter explained, that the extra sheet thickness on the service wing was responsible for a difference in drag coefficient of about 0.0004, or 5 percent of the smooth-wing drag. There remains a difference of 8 percent of the smooth-wing drag to be attributed to manufacturing irregularities such as sheet waviness, departures from true profile, and imperfections in the lapped joints. In the absence of protruding rivet heads on the forward part of the wing, equivalent manufacturing irregularities would probably have a much larger effect.

The drag of the riveted service wing at zero lift increased rapidly at Reynolds numbers above 14,000,000, corresponding to a Mach number of 0.43 or a speed of 330 miles per hour at 60° F. It is believed that this rapid rise in drag was due to a shock wave prematurely induced by a bulge in the lower portion of the nose of this model (fig. 8). The importance of making wings for high-speed airplanes accurately to true profile is evident.

The drag of the spot-welded stainless-steel service wing averaged about 20 percent greater than the drag of a smooth accurate wing. It is difficult to say how much of the excess drag of this model was due to the lapped joints because of uncertainty as to whether the spot welds and the sheet waviness would induce premature transition in the absence of the lapped joints.

APPLICATION OF RESULTS

It has been shown that the increase in wing drag caused by rivet heads can be divided into two parts: the increase in skin friction due to the fact that the rivets cause transition from laminar to turbulent boundary layer to occur abnormally far forward on the wing, and the direct drag of the rivet heads themselves. The separation of the drag increments into these two parts affords a basis for estimating the drag due to rivet heads and lapped joints under conditions of scale and of rivet and lap size and arrangement outside the range of these tests. The increase due to a forward shift of the transition point can be approximated by estimating the distance through which the transition point is moved and then applying the known difference between laminar and turbulent skin friction for the Reynolds numbers involved. The direct drag of the rivet heads can then be estimated from computed local velocities at the rivet heads and suitable rivet-drag coefficients.

In order to estimate the drag increase resulting from a forward shift of the transition point, it is necessary first to know where the transition point would be if there were no rivets on the wing. The results of recent tests by Becker, as yet unpublished, have indicated that, on smooth conventional airfoils, the transition point approaches the point of peak minimum pressure as the Reynolds number increases. When there is more than one peak of minimum pressure on a surface of the airfoil, transition will generally occur at the farthest forward peak even though it may be smaller than later peaks. Until a more complete understanding is available, transition at large Reynolds numbers may be assumed to occur at the farthest forward peak of minimum pressure on each side of the wing. It appears safe to assume that any protruding rivet heads large enough to be practicable will, if forward of the smooth-wing transition point, cause transition to occur at the rivets. Dust patterns on models in the 8-foot high-speed wind tunnel have indicated that the wake of turbulent boundary layer behind individual rivet heads in an otherwise laminar boundary layer spreads laterally with a total included angle of about 15°. This angle may be used in estimating the area over which premature transition is created by individual rivet heads.

Calculated skin friction based on free-stream dynamic pressure and flat-plate coefficients agreed with experimentally determined values on an N.A.C.A. 0012 airfoil. In the absence of further evidence, the increase in skin friction resulting from the estimated shift of the transition point can be estimated as the product of the area affected, the free-stream dynamic pressure, and the difference between the laminar and the turbulent skin-friction coefficients. The difference between local flat-plate skin-friction coefficients for turbulent and laminar boundary layers (reference 8) varies only slowly with Reynolds number. A constant value of 0.0026 is correct within 10 percent for Reynolds numbers between 1,000,000 and 10,000,000. This value of the coefficient is based on the total area involved instead of the wing area as usually defined, and the Reynolds number is based on the distance from the leading edge to the center of the area affected.

From the test results herein reported, a coefficient for the drag of brazier rivet heads in the turbulent boundary layer has been computed. This coefficient, based on the local dynamic pressure in the boundary layer at a distance from the surface equal to the height of the rivet heads and on the frontal area of the rivet heads, has the value of 0.32. A more convenient form of this coefficient for use with standard brazier-head rivets is that based on the square of the shank diameter in inches instead of on the frontal area of the head in square feet. In this form, the coefficient becomes 0.0020.

A coefficient computed in the same manner for plain lapped joints facing aft has the value 0.20 based on the frontal area in square feet but may be as high as 0.30 if the laps lie outside the true airfoil profile.

Since transition will occur at the most forward rivets, all the rest of the rivets will be in a turbulent boundary layer and may be treated as outlined in the preceding paragraph. The most forward rivets, if forward of the smooth-wing transition point, will be in a laminar boundary layer and this boundary layer will generally be thinner than the height of the rivet heads. These rivets, therefore, have a higher drag coefficient than those wholly submerged in the turbulent boundary layer. The coefficient expressing the drag of 3/32-inch brazier-head rivets in a laminar boundary layer about 0.004 inch thick was computed from the test results to be 1.3, based as before on the dynamic pressure at the top of the rivet head and the frontal area of the head. Based on the square of the shank diameter in inches, this coefficient becomes 0.0079.

In the application of these drag coefficients, the airfoil pressure distribution may be computed by Theodorsen's method (reference 11) and the boundary-layer thickness by the method of Dryden and Kuethe (reference 12). On the assumption that the one-seventh power law applies to the distribution of velocity in the boundary layer, the velocities and the dynamic pressures at the tops of the laps and the rivet heads can be computed for the different positions. The drags of the different laps and rivets can then be computed as the products of these dynamic pressures, the corresponding areas, and the applicable coefficients and be summed up along with the drag increment resulting from the shift of the transition point to obtain the total drag caused by the rivets and lapped joints on the wing.

This method of estimating rivet and lapped-joint drag has been applied to most of the arrangements of lapped joints and 1/16-inch and 3/32-inch brazier-head rivets tested and has yielded results in good agreement with the experimental values.

EXAMPLE

The magnitude of drag increments resulting from the presence of rivet heads and lapped joints on small airplane wings can be judged directly from the test results. As an example to illustrate the magnitude of the increments on large wings, the drag due to rivets and lapped joints on a wing having an average chord of 20 feet has been estimated. The chord positions of the spanwise rows of rivets and lapped joints assumed for this example are shown at the top of figure 23. It was assumed that the rivets were standard 3/32-inch brazier-head rivets, that the spanwise pitch was 1½ inches, and that the thickness of the lapped sheets was 0.040 inch. The flight speed was taken as 250 miles per hour at sea level, the corresponding Reynolds number being 45,000,000.

One set of ordinates in figure 23 shows the drag caused by the rivets and the lapped joints remaining on the wing when all the irregularities forward of the chord positions indicated by the abscissas have been eliminated. For example, it is estimated that all the rivets and the lapped joints shown on the wing would increase the drag coefficient by 0.00115 but that, if the rivets and the lapped joints on the forward 30 percent of the wing were eliminated, the excess drag would be reduced to 0.00035.

The excess power required just to overcome the drag caused by the rivets and the lapped joints is shown by the second set of ordinates in figure 23. The additional assumptions that the wing area is 3,600 square feet and that the propulsive efficiency is 85 percent, have been used in computing this power. With all the rivets and the lapped joints shown on the wing, more than 500 horsepower would be required just to overcome their drag. If the forward 30 percent of the wing were made smooth, about 160 horsepower would be required to overcome the drag caused by the remaining rivets and lapped joints.

CONCLUSIONS

The most important conclusions drawn from the tests described in this report can be summarized as follows:

1. Rivets at 3/4 –inch pitch in 13 spanwise rows on each surface of an airfoil of 5-foot chord increased the drag from 6 percent for countersunk rivets of 27 percent for 3/32-inch brazier-head rivets. About 70 percent of these drag increments were due to the rivets on the forward 30 percent of the airfoil.

2. Lapped joints, arranged six on each surface, increased the drag of the airfoil from 4 percent for joggled laps to 9 percent for conventional laps.

3. Surface roughness may cause serious increases in drag; for example, the roughness due to spray painting increased the drag 14 percent and roughness of 0.0013-inch grain size increased the drag 42 percent.

4. Manufacturing irregularities increased the drag of a typical wing 8 percent of the smooth-wing drag over and above the increments due to the rivets and lapped joints.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., March 7, 1939.

Document 3-31(c), Charles Peyton Autry, Boeing Aircraft Company, "Drag of Riveted Wings," in* Aviation *(May 1941): 53-54.

DRAG OF RIVETED WINGS

With so much emphasis placed on speed in present designs even rivet drag is an important factor. The following is the result of some NACA tests.

By Charles Peyton Autry, *Boeing Aircraft Company*

This discussion is intended to provide a practical insight of the excess power required in the drag of riveted metal wings with surface laps, being particular to small areas at high speeds. For certain types of wing surface construction and disposition the excess power required is of such an amount that this construction is highly unsatisfactory. Charts for a range of wing areas, Reynold's numbers and rivet dispositions are plotted to extend a means of estimating the drag in terms of the power increments. These power values have been plotted with the inclusion of an

.85 propulsive efficiency.

The NACA tests conducted by Hood on wing surface irregularities included the effects of rivet heads and surface laps on wing drag. These tests were principally on an NACA 23012 airfoil of 5 ft. chord. The drag coefficient increments of the tests were employed in the power formula forming the curves as shown in Figs. 2, 3 and 4.

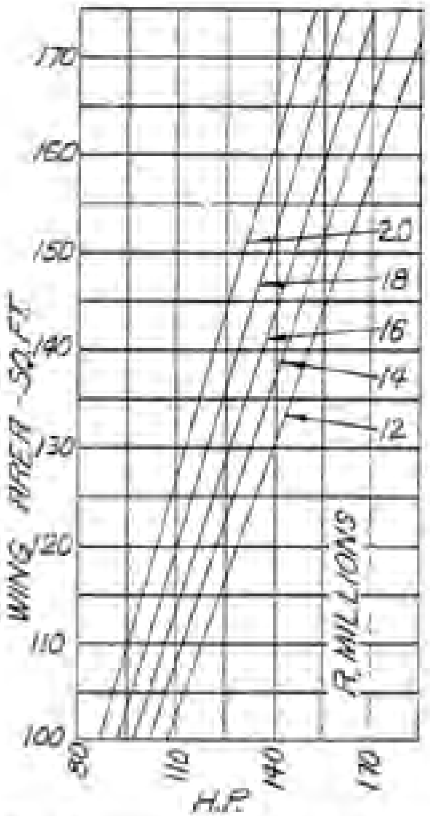
Fig. 1. Illustration of several types of rivets and joints used in fastening sheet metal for aircraft.

The rivets and laps considered herein are those shown in Fig. 1. The rivets are of 3/32 in. shank dia. The increments are based on these rivets for a 5-ft. chord. The tests also consisted of runs on a like airfoil and rivet arrangement, of which the chord was 2 ft. and rivets were two-fifths the size of the 3/32 in. rivets, drag coefficients of the two airfoils being increased by practically equal amounts at equal Reynold's numbers, although the Mach numbers were considerably apart. The tests are thereby applicable on the basis of Reynold's numbers alone.

Chordwise locations of the rivet rows and laps, in these tests, were as given further for the power charts herein. Spanwise pitch of the rivets for the following power curves is .0125 of the chord or ¾ in. for the 5-ft. chord. The tests showed that rivet drag reduction was negligible up to the point of doubling the ¾ in. pitch with the forward rivet row at .04 of the chord from the leading edge. This was probably due to disturbance of the boundary layer, the drag increments being only slightly reduced by the increase in rivet pitch. With the forward row at .28 of the chord from the leading edge the rivet drag varied proportionately to the number of rivets as the pitch varied.

The tests also included the effect of various positions of the forward rows of rivets on the drag increments. Sheet thickness of laps as dealt with herein was .018 in. Drag increments as applied were taken at values when C_L = .15.

Power Increments of Two Wing Types at 400 m.p.h.



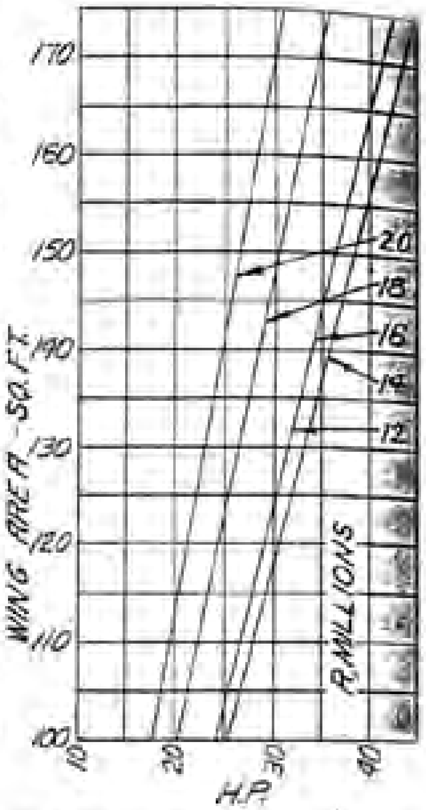


Fig. 2. Curves of horsepower plotted against wing area for 80 to 170 hp. range for various Reynold's Numbers.

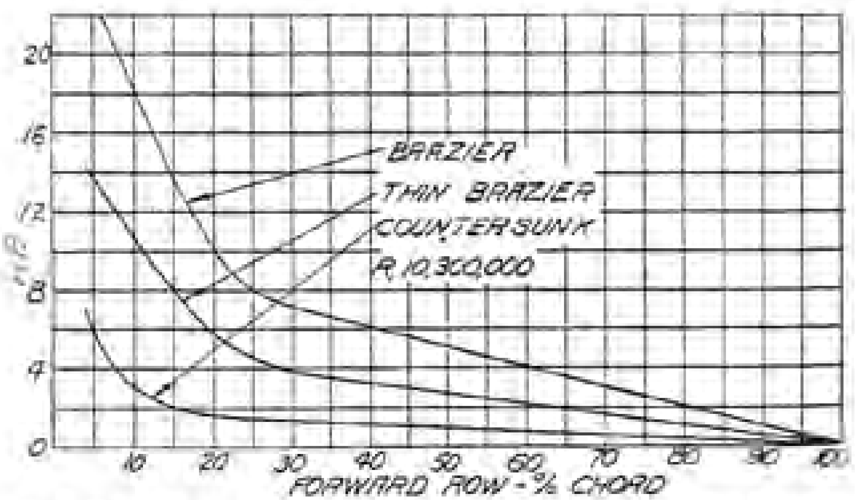


Fig. 4. The curves are for various types of rivets at a constant Reynold's Number of 10,300,000.

Fig. 2 and Fig. 3 present the values of these wings, Fig. 2 being with brazier-head rivets and plain laps. Fig. 3 gives the values for countersunk rivets and joggled laps with the bend fissure of the after sheet filled with a smooth surface filler. This lap is assumed to be representative of a lap having no drag, or that the lap drag has been successfully eliminated. The joggled laps at the plain lap positions without a filler in the after-sheet bend fissure has a zero drag increment at an R of 7,000,000 and its maximum increment at an R of 13,000,000. Such a wing of 175 sq.ft. at 400 m.p.h. would produce a power increment of 45 at an R of 13,000,000. The drag under consideration in this wing is then due to countersunk rivets, which is caused by an indention around the rivet head, in the sheet, the result of punching. Fig. 2 is not typical of a wing of excessive drag increment, probably being as low as any average wing having plain laps and brazier rivets, with the exception of any rearward difference of position of the forward rivet rows and laps or effective rivet pitch increase that might be made. Sheets conformed fairly close to profile without any excessive or needless variations.

The rivets are in 13 rows on each surface of the airfoil. The plain laps are 6 in number on each surface. The number or position of the joggled-filled laps has no bearing on the results, since they are assumed to offer no drag. The position of rivet rows on each surface of each wing are at .04, .08, .12, .20, .28, .36, .44, .52, .60, .68, .76, .84 and .92 of the chord from the leading edge. The position of the plain lap centers of each surface are .08, .20, .36, .52, .68, and .84 of the chord from the leading edge.

The charts of Fig. 1 and Fig. 2 are for various Reynold's numbers which will give usable power values for the range of wing areas given, the 400 m.p.h. speed, average chords for the areas given and an average value of ρ^s/μ.

The drag coefficient increment curves of the tests, plotted against Reynold's numbers were extended from the maximum of 18,000,000 to 20,000,000 to include the increment for the latter value.

In accordance with the power charts, Fig. 1 and Fig. 2, a wing with brazier-head rivets and plain aft-facing laps of the number and disposition given, the power increment may be reduced to about 22 percent by the use of countersunk rivets and the joggled-filled lap or its equivalent.

EFFECT OF FORWARD RIVETS ON POWER INCREMENT

Fig. 4 is illustrated of the power increments for a wing of 200 sq.ft. at 200 m.p.h., with three types of rivets. The abscissas of the chart is the most forward row of rivets. Rows aft of the first row are at the positions previously stated with the exception of the rows forward of the position in question, which are non-existent. The joggled-filled lap or its equivalent is assumed. Rivets forward of the smooth wing boundary layer transition point cause this point to move forward resulting in an increase of the turbulent boundary layer extension and a corresponding decrease in the laminar layer. Lapped joint drag varied similar to that of rivets. In Fig. 4 the rivets forward of the .30 chord point constitute about 75 percent of the total power value when the forward rivet row is at the .04 chord point.

Reduction of the power increment by increasing the spanwise pitch of the rivets forward of the smooth wing transition point may not be appreciably accomplished unless the pitch is of some marginal consequence above .025 of the chord, as previously stated from the referred tests.

CONCLUSIONS

A typical high-speed single-engine pursuit airplane incorporating the types of rivets and laps of Fig. 2 would be required to yield as much as 20 percent of its total power to the rivet and lap drag of the wing. This could be reduced to about 4 percent or less by the use of joggled-filled laps and countersunk rivets and elimination of rivets forward of the .30 chord point would further reduce the power increment to about 1 percent or less. The ultimate desired at high speeds would be complete elimination of rivet and lap protrusions or indentions. Preformed or smooth leading edges up to the .30 chord point and any practicable amount above are of much value.

Document 3-32

J. D. Akerman for the Boeing Aircraft Company, excerpts from *Manual for Design of Details Affecting Aerodynamical Characteristics of an Airplane*, Boeing Report No. D-2789, 12 Sept. 1940, Microfilm Roll No. 154a, Boeing Archives, Seattle, WA.

By the early 1940s, American aircraft manufacturers had come to understand the value of an extremely clean detail design. The problem was communicating this understanding to their engineering and production staffs. In the fall of 1940, a consultant for Boeing, aeronautical engineering professor J. D. Akerman of the University of Minnesota, prepared the following manual by which to emphasize to Boeing employees the importance of various meticulous details in the proper design and manufacture of airplanes. Akerman left nothing unexamined: rivets, seams, and joints; fasteners and screws; inspection cover plates; vents and relief drains; hinges; gaps or clearances between moving surfaces; spanwise gaps; chordwise gaps; cowlings and cowl flaps; fillets; air inlet openings; scoops; air outlet openings and leaks; streamlined covers on exposed parts; windows; doors and handles; and de-icers. All were carefully scrutinized as potential sources of disturbed airflow.

It is clear from Akerman's references at the end of his manual that NACA research played a formative role in defining the problem of the "numerous protuberances, crevices, or irregularities" affecting aerodynamic performance. Of his 14 references, 9 of them were NACA reports.

Akerman was an interesting and controversial character. For an analysis of his stormy career at the University of Minnesota, see Amy E. Foster, "Aeronautical Science 101: The Development of Engineering Science at the University of Minnesota," Master's thesis, University of Minnesota, 2000.

Document 3-32, J. D. Akerman for the Boeing Aircraft Company, excerpts from* Manual for Design of Details Affecting Aerodynamical Characteristics of an Airplane*, Boeing Report No. D-2789, 12 Sept. 1940.

BOEING AIRCRAFT COMPANY

MODEL A11
REPORT NO D-2789

MANUAL FOR DESIGN OF DETAILS AFFFECTING AERODYNAMIC CHARACTERISTICS OF AN AIRPLANE

Number of Pages: 30 Prepared by: J.D. Akerman

Date: 9-12-40 Approved by: Flight and Research Dept.

INDEX

INTRODUCTION

In order to meet the requirements for high top speed and economy of operations at cruising speeds, as specified by contractors for latest model airplanes, the aerodynamics efficiency and cleanness of design of every detail exposed to the wind has become of necessity one of the most important factors contributing towards the satisfactory performance of the airplane.

Low overall drag and aerodynamic efficiency, secured by well and tediously developed shape of major parts of an airplane, may be completely offset by numerous protuberances, crevices or irregularities appearing on the finished airplane. Very often such detrimental increases in total drag are due to a multitude of details designed, or simply installed on the airplane, without proper consideration of their individual or accumulative drags.

Extensive studies made by the N.A.C.A. and by the B.A.C. have definitely established the value and importance of clean detail design. On one actual service airplane, the combined total drag of excrescences alone constituted 45.4% of the total drag of that particular airplane, or 83% of the total drag of an equivalent ideal aerodynamically clean airplane (see B.A.C. report D-2487).

The following discussions and suggestions as to how to decrease the drag of each exposed part of an airplane by rational and careful design of the details may serve as a guide to the designers in order to eliminate most of the unnecessary drag on future airplanes. All given percentages of drag values are approximate averages for purposes of comparison.

FROM PRODUCTION AND COST STANDPOINT, IT IS OFTEN EASIER AND CHEAPER TO MAKE A SMOOTH, SIMPLE, WELL-DESIGNED PART THAN TO PRODUCE A RAGGED, HAPHAZARDLY designed part, assembly, or accessory, provided proper care is taken in design of details.

The accumulation of minute barnacles on a sea-going giant renders the ship's operation uneconomical and required periodical hull scraping. On airplanes we have reached a stage where small protuberances and irregularities are worse than barnacles and their removal is mandatory on modern high-speed aircraft.

It is essential to keep always in mind the following three basic rules:

1. AVOID MAKING ANY DETAIL WHICH WOULD CAUSE DEVIATION FROM ORIGINAL AERODYNAMIC CONTOUR – It can be done. There is always another way to do it. Somebody else probably did it on some other airplane.

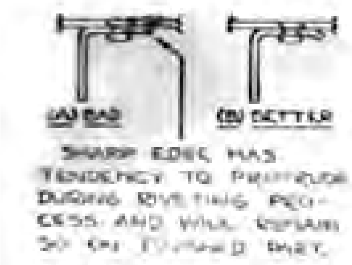
2. IF YOU CAN NOT AVOID MODIFICATION AND HAVE TO DESIGN A DETAIL PRODUCING SURFACE IRREGULARITY, MAKE THIS IRREGULARITY SMALL, SMOOTH AND WITH MINIMUM DRAG. Consult the following discussion for guidance and aerodynamic section for approval.

3. IF YOU HAVE TO MAKE "BARNACLES" MAKE THEM FEW, SMALL AND WITH MINIMUM DRAG. One "Barnacle" is negligible, but 300 designers may eliminate thousands of them and improve the performance, appearance and utility of the airplane.

4. WHEN IN DOUBT CONSULT FLIGHT AND RESEARCH DEPARTMENT CONCERNING EFFECTS OF THE PROTRUSION ON THE DRAG OF THE AIRPLANE. At high speeds even small irregularities may be critical.

RIVETS, SKAMS AND JOINTS

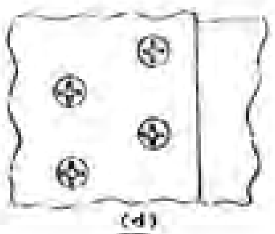
Tests conducted by NACA indicated that a smooth flush rivet construction service wing had 17% extra drag due to: "(1) sheet waviness, (2) departure from true profile, and (3) imperfect laps." All three causes may be traced back, in most cases, to the original design of the wing parts. A design should be such that fast manufacturing practices could be applied and still the product would be uniform, regular and within specified tolerances without special hand adjustment of "fussing" by the craftsman. Even if the tolerances are indicated, the product can not be expected to be uniform and smooth if the original design is not fool-proof against subsequent variations and distortions of the part during the process of manufacturing.

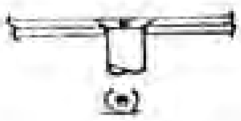


Since joints and fastenings on airplanes at the present time are made by means of rivets, and since riveting constitutes 60% of manufacturing cost of assembly of an airplane, IT IS JUSTIFIABLE FROM THE PRODUCTION AND FINAL COST STANDPOINT TO EXERCISE PARTICULAR CARE IN DESIGNING EACH JOINT. Some general rules will always help:

1. Air hammered rivets will give a worse surface than squeezed or "Kuck" pulled rivets. Therefore, make the design so that squeeze or Kuck riveting processes are applicable in subassemblies or assembly. It will be faster, cheaper and smoother.
2. Air hammer used on surfaces, and bucking bar on the inside, will produce large indentation, distortion of sheet, and rolling up of the edge.
3. Exercise care in designing joint "back up" parts. Inclined approach of riveting tool will produce bump.
   Theoretical joint as in (d) will result in distortion as shown in (e).
   A flat and sharp edge "backing part" will never produce smooth surfaces. (f)
4. In making seams the following principles should be kept in mind:

(1) Any seam perpendicular to air stream will have approximately 70% more drag than one parallel to the air stream.





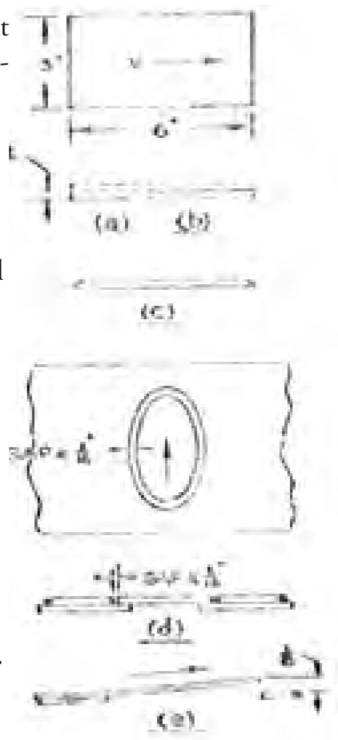
(2) Design part so that it is possible to produce a gap of not more than 1/32 of an inch.
(3) SPECIFY on drawing that gap should not be over 1/32 in.
(4) Specify [ ] 1:2 as in (b).
(5) DO NOT MAKE JOINTS ON FIRST 33% of streamlined body or curvature. It may produce compressibility wave.
(6) AVOID MAKING joints in next 30%.

FASTENERS AND SCREWS

With the introduction of smooth wing and fuselage construction, particular care should be given to provide smooth surface where a row of permanent fasteners or screws are used in place of rivets. Some places to mention: gas tank covers, fillets, door hinge line, etc.

1. A row of fasteners just behind the front spar, as shown on Fig. (a) or (b), would destroy all the benefits secured by smooth riveting behind the row.
2. A Fastener, as in (e) and screw, as in (c), are preferred to the others.
3. Recommended use of Reed and Prince screws wherever possible in countersunk holes (d).
4. Use Camlock Fastener's, precision type (e) where Reed and Prince screws cannot be used.
5. AVOID SCREWS OR FASTENERS IN FIRST 33% WING OR STREAMLINED BODY.
6. A single, double, or triple row of Reed and Prince screws or fasteners would leave a very detrimental effect on the wing or fuselage, and therefore, special attention should be made to [ ] a COVERING STRIP of thin adhesive transparent weather resistant tape with proper water proofing. This type should be required for two reasons:
   a. To provide aerodynamically smooth surface.
   b. TO ASSURE INSTANTANEOUS INSPECTION THAT ALL SCREWS AND FASTENERS ARE AS ORIGINALLY INSPECTED, SINCE THEY CARRY STRESS AND PROVIDE NECESSARY AIR, GAS OR WATER TIGHTNESS.
7. No Finger Lifts (f) on covers should be used. They provide dangerous air leaks, place to collect water and ice, increase drag, are useless when mechanics are working with gloves.
8. "Snap On" fasteners (f) should not be used. Securely closed air and water tight inspection doors and covers should be provided with inside spring to open cover automatically if locking device is not closed.

INSPECTION COVER PLATES



Do not make external cover plates. Consult Flight and Research Department if outside cover plate is contemplated.

1. Best installation:
    a. Depressed smooth plate
    b. Groove not more than 1/16
    c. Short and perpendicular to airstreams
2. Drag on same plate 6" X 3" fig. (a) and (b) will be decreased if short side is perpendicular to the airstream instead of long side
    For t = 1/16" by 33%
    t = 1/4" by 53%
3. Drag will be decreased if
    t = 1/16" is substituted for t - 1/8 by 69%
    t = 1/16" is substituted for t - 3/16 by 83%
    t = 1/16" is substituted for t - 1/4 by 88%
4. [ ] edges will decrease plate drag 20 to 30%.
5. Make covers to close flush. A gap 1/8 inch produces drag equal to drag of a 1/8 [ ] plate.
6. Covers should close tight to prevent air and water leaks.
7. No "Finger Lifts" on outside covers. (See notes on fasteners)

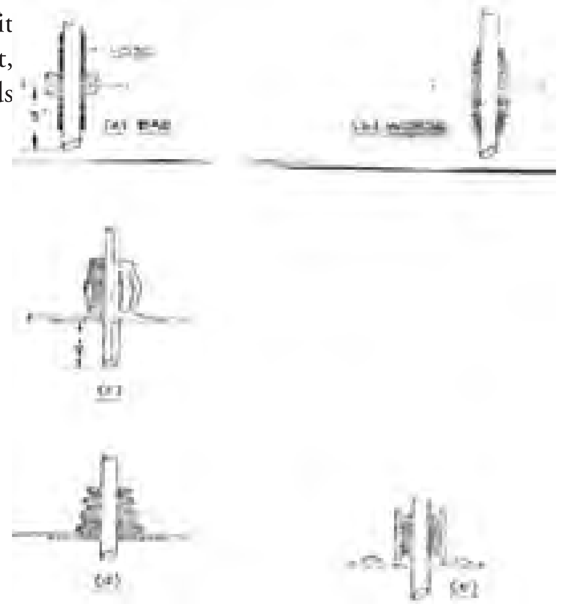
GENERAL CONSIDERATIONS

1. Do not make inspection plate line in leading edge of wing, control and tail surfaces.
2. No joint lines in forward 10% of chord; it may destroy [ ] flow at high speeds.
3. A joint line in first 10% of chord will have 50% more drag than one at 30% of chord.
4. Use flush rivets on cover plates. They are small parts and can be riveted on "Brco" rivets with flush rivets just as easy as with round head rivets.
5. The plate adds some drag; do not add extra rivet drag.

VENTS AND RELIEF DRAINS

On a recent model airplane there were counted 44 relief drains or vents. The drag of those drains constituted 1.5% of the total drag of a smooth airplane. In terms of N.P. it required 76 H.P. to drag those extensions through the air, using 50 gals. of gas during a service flight.

Present drain installation is intolerable as shown on Fig. (a) or (b).



In order to get some benefit of the 50 gals. of gas per flight, the drag of the vent or drain ends should be eliminated by:

1. COMBINING SEVERAL LINES IN ONE AND DRAINING OUT THROUGH TRAILING EDGE OF WING.
    Care should be taken in doing this that NO LOW BENDS are in the line WHERE GASOLINE MAY BE TRAPPED, and that at the trailing edge, tubing does not stick out as a sharp point WHERE STATIC ELECTRICITY DISCHARGES would be produced to IGNITE GAS VAPOR or interfere with radio. Use static proof preventive coating.
2. Designing a special [ ] and short pipe and, as suggested in Fig. (c), (d) or (e) may be allowable on cowling or other places where draining through the trailing edge is impossible. Consult B.A.C. Dug. 1-17524 and 1-17525. COMBINING SEVERAL LINES IN ONE OUTLET STILL SHOULD BE [ ].

HINGES

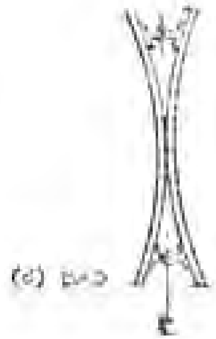
If for some reason a hinge is unavoidable on one side of plate:

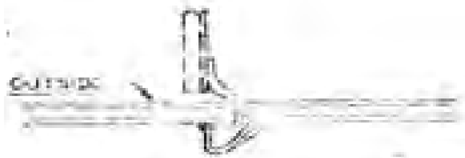
1. Make hinge line flush.
2. Use flush rivets – hinge lines are bad; do not make them worse with rivet heads.
3. Hinge line parallel to air stream will have 85% less drag per linear foot than one perpendicular to air stream.
4. No hinge line in first 30% of chord of wing, control surface, tail surfaces or any curvature less than 9° radius.
5. Faulty installation, as fig. (b), is very common. To avoid protrusion "E" installation (c) is suggested IF hinge line IS unavoidable and

(1) Perpendicular to airstream,

(2) More than 2 inches long,
(3) Located on forward portion of fuselage or cowling where local air velocities are high.

6. On many former models on latches, doors or covers, hinges are of the external type as in Fig. (d). Such external hinges are not found on latest high speed aircraft models (not even on modern automobiles) and will not be tolerated on new designs. A suggested alternate inside hinge for curved surfaces is shown in Fig. (c).





GAPS OR CLEARANCE BETWEEN MOVING SURFACES

Gaps between fixed and moving parts of surfaces fall into two types:

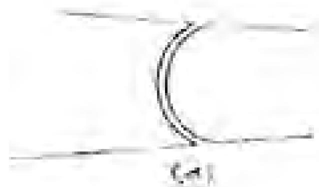
(1) Spanwise
(2) Chordwise

SPANWISE GAPS

(1) Spanwise gaps along front part (nose) of control surfaces play a very important part in the effectiveness of the control surfaces. By a slight change of shape or size of the gap, effectiveness may be decreased easily from 10 to 20%, and in some cases up to 30%. SOME SLIGHT CHANGES MAY MAKE THE GAP VERY DANGEROUS FROM THE STANDPOINT OF ICE FORMATION. Therefore, extreme care should be exercised to design the gap according to the specifications from the Flight and Research Dept. Proper dimensioning and design of details should be made so that the production departments could produce and duplicate the same gap on all airplanes of the same model and still make parts interchangeable. Very often all efforts to produce good control surfaces and aerodynamically efficient gaps are rendered ineffective by poor design of parts and surfaces inside of the gap.

To illustrate the case, Fig. (a) was specified by aerodynamics section. When completed it had some or all protuberances as shown on Fig. (b) causing:

(1) Change of effectiveness of slot and controls; extreme drag of lap joint in high velocity air

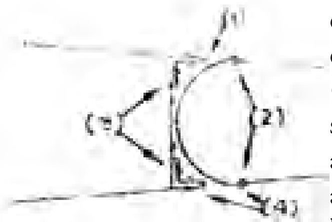


stream: change in hinge moment of surface.

(2) Extreme drag in high velocity air stream capable of producing turbulent flow and compressibility wave. Reduction of air flow through gap and reduction of effectiveness of gap.

(3) Pocket producing whirling flow reducing effectiveness of gap and causing extra large drag.

(4) Changes of flow through gap produce excessive drag. Laps and rivets produce areas for excessive ice formation.



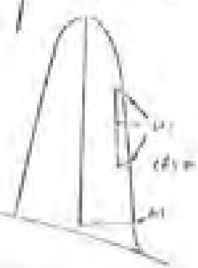
(5) Brackets, supports or partitions inside of spanwise gaps and slots should be just as smooth and free from rivets, bolt heads, and sharp corners as wing surface since the air velocity in the gap may be even higher than on the outside of the wing.

Any turbulence of air INSIDE OF GAP (or outside) does not "just happen." It is created by force and causes drag using power. The only bank from which continuous withdrawals are made to pay for those forces or drags is the power plant. Like tax taken—they amount to a large total.

EVEN IF YOU CANNOT SEE THE IRREGULARITIES FROM OUTSIDE, THEY ARE EXPOSED TO HIGH VELOCITY AIR STREAMS AND CAUSE LARGE DRAGS AND QUITE OFTEN ARE DANGEROUS FROM THE STANDPOINT OF CHARGES IN EFFECTIVENESS OF CONTROLS AND ICE FORMATIONS.

CHORDWISE GAPS

On one recently manufactured airplane gaps (1), (2), (3), and (4) fig. (a) and (b) were measured and the clearance space varied from 1/4" to 1-1/4." Such variations of gap could not all be attributed to poor manufacturing processes. Most of the variations are due to inconsistency of design and lack of specific detail dimensions to provide interchangeability of parts, and give the same clearances between parts.



There is no reason at all why any clearance gap should be more than 1/4" on a metal airplane when 1/8" would provide ample room to take care of discrepancies in overall dimensions of sub-assemblies.

On one end of a moving part a gap width measured on top of gap 1-1/4" and on T.B. 1/4", fig. (c). Such discrepancy is not permissible on future aircraft.

A very common cause for excessive gap on control surface is the bump of cloth on the trailing edge corner. Such bumps are due to the poor practice of sewing on fabric and finishing the corner. Care should be taken to specify that corners should not protrude (it is done on other airplanes) and gap should be no help to the production because it requires adjustments and fits in assembly. Design specifications and detail dimensioning will help to avoid misfit of sub-assemblies.

FABRIC COVERINGS

On fabric covered parts of control surfaces on present designs, the ribs, rib tape, and stitching protrude from 3/32 to 3/16 of an inch above the surface of the covering. Such protrusions are bad from the standpoint of drag and amount to a large drag item since they are on all horizontal and vertical control surfaces.

Such irregularities of surface are also bad from the standpoint of control surface effectiveness since their protrusion tends to increase the thickness of the turbulent air layer over the surfaces.

Suggest a rib stitching as in Fig. (f) or better.

COWLING

The general outlines of cowling shape specified by the aerodynamics division should be adhered to in every detail.

The front part of the cowling is the critical place where high velocities are encountered. It has been determined by tests and calculations that even on present models, air velocities approaching the velocities of sound are encountered in several places on cowlings. If the front part of the cowling is continuously exposed to such high air velocities, it is obvious that certain practices in use now should be discontinued.

sq. in. gauge pressure difference between inside and outside of cowl.

6. Design cowling joints parallel to the axis in places of circular joints by designing cowling which would open as orange peels. Provision should be made to have joints smooth and tight. Fig. (a)

COWL FLAPS

On many present day airplanes cowl flaps stay open when they should be closed or tend to close when they should be open, see fig. (a) and (b). In the design of cowl flap particular care should be given to the following:

1. Opening and closing mechanics should be so designed that the pilot could definitely set cowl flaps at any desired position to secure necessary cylinder temperature control.
2. Deflections in operating mechanism and in flaps should be reduced to minimum so that flap position would not vary due to outside or inside air loads. See Fig. (a) and (b)
3. When flaps are closed longitudinal and circumferential joints should be airtight to prevent air leaks, fig. (c). Such air leaks reduce the efficiency of air outlet opening and may affect cooling of the motor. Such air leaks also produce very undesirable high drag.
4. There should be enough control to pull flaps in closed position, fig. (b).
5. Inside of flaps and flap mechanism are exposed to airstream and should be made as smooth as possible.
6. All seams and edges should be smooth and flush.

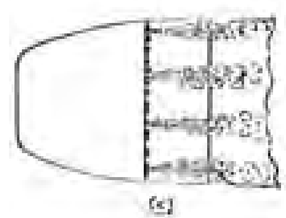
FILLETS

Filleting should be provided wherever there are two surfaces intersecting at an angle over 45%. See (a). Such filleting should be installed according to the specifications of the Flight and Research Dept. and if such specifications are missing, it should be provided in detail design, see (b). Moreover, the Flight and Research Dept. should be consulted, because there are many cases where special filleting would not be justified unless the curvature is incorporated in the parts where there are rivets, bolts or screws to cover up.

A very common mistake is shown in Fig. (c). The drag of exposed edges, rivets, or bolts may be greater than the gain provided by the fillet.

Air leaks between fillets and body are extremely undesirable. See note on air leaks.

A gap between body and fillet edge, or protruding edge is very objectionable.



AIR INLET OPENINGS

(1) FOR LOCATION OF AIR INLET HOLES, CONSULT FLIGHT AND RESEARCH DEPT.

(2) Edges and inside corners should be rounded by proper size radii (b).

(3) Edges, radii and inside surface of ducts should be just as smooth as the outside of an airplane. Flow in ducts produces drag, turbulent flow and compressibility [wave] same as outside air.

(4) Bends in ducts should be smooth and gradual in order to avoid turbulence, drag, and resistance to flow of air in duct (c).

SCOOPS

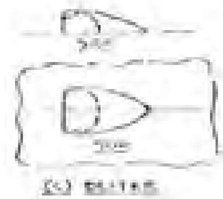
A modern airplane should not have individual scoops for each accessory hanging over all parts of the airplane. All cooling air for individual use should be taken from one or two main air inlet openings as motor cowl, intercooler air ducts or main ventilating air shaft. If, with permission of the Flight and Research Dept., a scoop has to be installed, the following principles should be observed:

(1) Do not install scoop in region of high velocity air stress. On a recent small high speed airplane, tests disclosed that a large scoop was installed in the location of the maximum air velocity on the airplane. The designer could not have selected a worse place if he had been instructed so.

Flight and Research Dept. should be consulted for locations of each scoop.

(2) On a recent airplane were counted 14 scoops, as shown on figures (a) and (b), not counting several large cooling scoops.

A streamlined shape with tail and rounded edges figure (c) and (f) would have cut the drag at least 50%.



(3) In designing a scoop, particular care should be taken in the design of the forward edge.

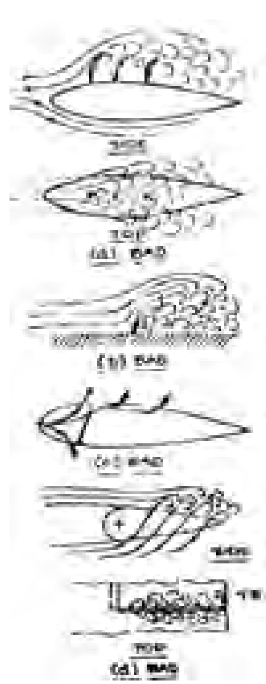
A very misleading conception is that a sharp edge, as figure (d), would have small drag.

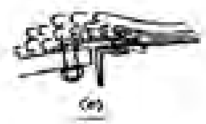
The point is that all scoops are designed to admit enough air at low speed in climb. At high speed or cruising only part of the air goes through the scoop and outside streamlines are deflected as shown.

(4) Seams and ridges as shown in figure (e) are very bad, since they produce turbulent flow around the whole body, resulting in very large drag.

Do not make scoops without consulting Flight and Research Dept. or trying to eliminate them completely.

AIR OUTLET OPENINGS AND LEAKS

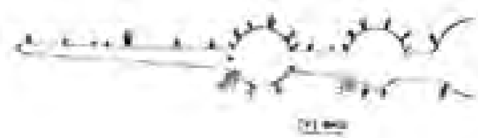




(1) The first and most important principle in designing air or gas outlets is:

(a) Air streams discharges as shown on (a) disturbs smooth laminar flow over the body behind the discharge hole and over a large area causing considerable increase in body resistance.

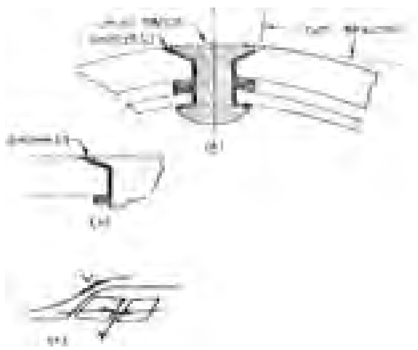
(b) The discharged air stream itself constitutes a body which has to break up and deflect the streamlines of the passing air (b). Such drag is just as bad, and combined with distortion of flow behind, even worse than the drag of a solid body.



(c) AIR DISCHARGE SHOULD BE REARWARD

(2) The real bad hidden culprits are the air discharge holes which are not supposed to be there at all, but which are there as a result of unintentional openings, holes, slits, gaps, etc., air discharges into low pressure areas breaking up smooth streamlined flow and causing accumulative drag of considerable magnitude.

It should be realized that each air jet resembles a "whisk broom" or a "floor mop" sticking out in the air stream. THE AIRPLANE WOULD LOOK PRETTY BAD with all these many "brooms" and "mops" sticking out and particularly if they were painted red, Fig. (f). The air jets are just as bad even if they are invisible and are not painted red.



The airplane of yesterday had too many air leaks on the wing, body, and nacelles but the airplanes of tomorrow should be airtight.

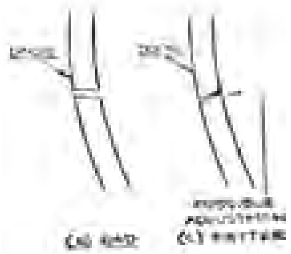
Recent studies of the effect of small air jets on the drag of a streamlined body (Ref. 6) show that a small air jet on a 10 inch dia. and 30 inch long streamlined body increases the drag of the body from 3 to 6 times depending on the location of the hole.

It is obvious that all efforts to produce streamlined low drag parts of an airplane may be useless if unnecessary air leaks are allowed.

STREAMLINED COVERS ON EXPOSED PARTS

A common malpractice will produce the cover as shown in Fig. (a).

A much better streamlined cover will be one which is even easier to manufacture as shown in Fig. (b).



Whenever there arises a question as to whether to make a short and wide cover or a long and narrow one, decrease the cross-sectional area. The forward part should not be blunt and should not have too sharp a break in its curvature.

NACA report (RIM 154) on windshield resistance gives valuable information for design of windshields, domes, covers and fairings over protruded parts. Shape 4-0-3 fig. (a) although very similar to shape 3-1-3 fig. (b) has only half of the drag and only 30% of the drag of form 1-1-3 fig. (c), and 10% of drag from 9-1-3 fig. (e).

All screws, rivets and fasteners should be the flush type and should provide an air tight joint.

When transparent fairing is installed as in Fig. (d) and (e) a joint similar to the one in (e) is often designed, destroying most of the advantages gained by the installation of the fairing.

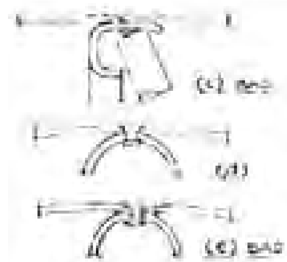
Joint as in (f) or better is suggested.

WINDOWS

All windows should be flush and conform to the curvature of the airplane, except on places where a flat surface is specifically called for by the contractor.

In case a flat surface is required, care should be exercised that this flat surface is not increased by flat frames around this flat surface. This can be avoided by providing slight curvature in frame surfaces. Fig. A.

In case a frame is on a spherical surface, bending the frame in one plane only would produce a flat ring in a plane perpendicular to the plane of bend. This flat can be eliminated by giving very slight curvature to the frame, as in figure (a). Unless the frame is curved a flat circular ring would be produced on curved surface.



Provisions should be made that the groove (1), fig. (a) and (b) is filled with suitable filler which is water, gas, and crack proof. Such fillers are available.

MAKE ALL WINDOWS FLUSH

In design of windshields, it should be remembered that at points the air velocity is very high. Tests indicated that the velocity of sound is reached there at normal flying speeds.

Therefore,

(1) Make corners round and not less than [9 M].
(2) Make smooth without ridges, grooves, or rivet heads. They will cause tremendous drag and destroy laminar flow.
(3) The shape of corners on fuselage are of such importance that they should not be designed without consulting Flight and Research Department for data secured by wind tunnel and flight tests.

DOORS AND HANDLES

One most common fault of door design is the poor fit as far as flush surface is concerned on the side of the smooth body. There is always a protrusion of the door from 1/8 to 3/8 of an inch above the contour of the body.

Such protrusions are very often the result of improper design of joints as in (b), where inside corner of door requires a large gap in order to clear the frame and does not provide the solid stop of flush installation.

Joint (b) also does not make provision to prevent air leaks.

HANDLES

Handles formerly used, as shown in (d) should be eliminated and handles such as on the Yale-Clipper lock (e) or Krug disappearing lock (f) should be used.

There is a possibility of providing a very simple flush handle as in (g). Care should be taken that the spring flap comes flush and the gap is small. The dependency of action, rugged construction should not be neglected in choice of handles and locks. It is important from the standpoint of operation.

A very poor handle is commonly used as in (h). If such a handle is mandatory it may be modified as shown in Fig. (3). A step as in (k) may also be very easily modified as shown in Fig. (1).

It seems that on each airplane in the field, an after-thought has made it necessary to provide eyes to fasten the motor covers. Such an eye is very crude and could be improved by using a flush eye belt as in (c).

DEICERS

Present deicer installation is very bad and is installed on planes as an afterthought producing six spanwise ridges along the L edge of the wing.

The new type of deicer, C-313, type 3 and 4, is smoother and better adapted for modern aircraft, since this new deicer is on continuous thickness with beveled edge. Provisions should be made in the contour of the wing to accommodate the deicer. However, the surface of the part under the deicer should be smooth and the recess made gradual in order to have a smooth wing if the deicer is taken off.

The question of designing a wing with a recess for deicer or installing a deicer above the smooth L.E. may be answered from the following considerations:

- (a) Most flying is done with the deicer on.
- (b) For real service flight, deicers will be on, and then best airplane performance (airfoil characteristics) are needed.
- (c) To meet (a) and (b) a recessed wing construction may be justified. Fig. (c) or exceptionally smooth installation should be provided by:
  - (1) [Rivets] should be flush and smooth.
  - (2) Joint must be smooth since the top joint is in a place where ridge and roughness may increase the wing drag from 20 to 30%.

Document 3-33(a-b)

(a) Laurence K. Loftin, Jr. Chap. 6, "Design Maturity, 1945-1980," in *Quest for Performance: The Evolution of Modern Aircraft* (Washington, DC: NASA SP-468, 1985), pp. 137-161.

(b) Laurence K. Loftin, Jr. Chap. 6, "Design Trends," in *Quest for Performance: The Evolution of Modern Aircraft* (Washington, DC: NASA SP-468, 1985), pp. 151-162.

The following documents from Laurence K. Loftin's classic 1985 book *Quest for Performance* serve well as an abbreviated review of the aerodynamic development of propeller-driven aircraft since 1945.

Although a historical survey, it also represents a "primary source" document in its own right, in that it reflects the experience and perspective of a veteran NACA/NASA aeronautical engineer looking back over a period of 35 years in which he was internationally active as a research aerodynamicist. Loftin's views on what has happened to propeller-driven aircraft since the end of World War II are thus both timeless and a reflection of their time.

As readers will clearly see, Loftin believed that propeller-driven aircraft reached a plateau during World War II above which it may not ever rise. Too many factors inherent to the propeller limit how fast a propeller-driven aircraft can travel.

Furthermore, basic physics interfers with improving maximum lift/drag ratio, a value that has not improved much since 1945. In Loftin's view, it may still be possible to acquire small increases in lift/drag through improved structural materials, by enabling minor reductions in drag and/or increases in wing aspect ratio. But other than that, little further improvement can be anticipated. Breakthroughs relevant to smoothing out turbulent flow and controlling laminar flow in the boundary layer — complicated matters to be discussed in the next chapter on airfoil development — represented perhaps the last best hope for another round of major aerodynamic progress. But Loftin does not mention these topics here, as he was skeptical that these developments would ever materialize.

Document 3-33(a), Laurence K. Loftin, Jr. Chap. 6, "Design Maturity, 1945-1980," in* Quest for Performance: The Evolution of Modern Aircraft *(Washington, DC: NASA SP-468, 1985), pp. 137-161.

CHAPTER 6: DESIGN MATURITY, 1945-80

BACKGROUND

In the years since the end of World War II, turbojet- and turbofan-powered aircraft have come to dominate an increasingly large segment of aeronautical activity. The propeller-driven aircraft, however, remains an important part of aviation, both in this country and in various other parts of the world. The new propeller-driven aircraft that have appeared since 1945 differ little in configuration from those seen in the years immediately before and during World War II, nor has the level of aerodynamic refinement exceeded that of the earlier aircraft. The turboprop propulsion system is probably the most significant technical advancement to be incorporated in propeller-driven aircraft. In the realm of reciprocating engines, the supercharger has come into widespread use, both in commercial transport aircraft and in contemporary general aviation aircraft. The supercharger, together with the advent of cabin pressurization, has resulted in highly efficient cruising flight at high altitudes. High-altitude operation also offers the passengers freedom from the discomfort of rough air to a degree that was not possible in unpressurized aircraft.

A few examples of propeller-driven transports of the post-World War II period are described and discussed here, as are a number of contemporary general aviation aircraft.

Two families of large, long-range, propeller-driven transports dominated U.S. airlines, as well as many foreign airlines, until the jet transport began to appear in significant numbers toward the end of the 1950's. These families of aircraft, which served on both long-range domestic and international routes, were the Douglas DC-6 and DC-7 series and the Lockheed Constellation series. Both were derived from aircraft developed during World War II; they had four supercharged engines and pressurized cabins, and both series underwent large increases in size, power, and weight during their development history.

Representative of the long-range, four-engine transport is the Lockheed L. 1049G Super Constellation illustrated in figure 6. 1. The prototype Constellation, known by its USAAF designation of C-69, first flew on January 9, 1943, and the model L. 1049G first flew on December 12, 1954. The total number of all models of the Constellation constructed was 856.

The Lockheed L.1049G was powered by four Wright turbocompound engines of about 3250 horsepower each. The Wright 3350 turbocompound engine employed a two-speed gear-driven supercharger and, in addition, was equipped with three exhaust-driven turbines. The three turbines were geared to a single shaft that in turn was hydraulically coupled to the engine crankshaft. Each turbine was driven by the exhaust of six cylinders. About 15 percent of the total power of the engine was obtained from reclamation of exhaust gas energy. The specific fuel consumption was probably the lowest ever achieved in a reciprocating aircraft engine.

The gross weight of the aircraft was 133,000 pounds, which was more than twice that of the Boeing B-17 "heavy" bomber of World War II fame. The wing loading was 80.6 pounds per square foot, and.the corresponding stalling speed was 100 miles per hour. The wings employed very powerful Fowler-type extensible slotted flaps to maintain the landing speed within acceptable limits. The landing gear was of the tricycle type that was standard on most post-World War II transports. The maximum speed of the aircraft was 352 miles per hour, and the normal cruising speed was 331 miles per hour at 23 000 feet. The pressurized cabin was capable of seating 71 first-class passengers or 91 coach passengers. Some versions of the aircraft were capable of carrying an acceptable payload nonstop from the east coast of the United States to the west coast. The zero-lift drag coefficient of 0.0211 and the maximum lift-drag ratio of 16 indicate a highly refined and efficient aerodynamic design.

Figure 6.1 - Lockheed 1049G Super Constellation 91-passenger four-engine airliner; 1954. [mfr]

Today, many Constellations and their Douglas counterparts are in operation in nonscheduled activities in different parts of the world. The use of these aircraft in long-range scheduled operations, however, terminated in this country during the 1960's following the introduction of the high-performance jet-powered transport.

The turbopropeller, or turboprop engine, is basically derived by gearing a conventional propeller to the shaft of a gas generator composed of a compressor, burner, and turbine. The turboprop engine may therefore be thought of as a turbojet engine that transmits power to the air by means of a propeller rather than through the jet exhaust. The turboprop engine is light and relatively simple as compared with the large high-power reciprocating engines. For example, a modern turboprop engine may develop between 2 and 3 horsepower per pound of weight, as compared with a maximum of about 1 horsepower per pound for a reciprocating engine, and has been made in sizes of up to 15,000 horsepower. The specific fuel consumption of the turboprop engine, however, is somewhat higher than that of the best reciprocating engines. The turboprop engine has been used in a number of highly successful transport aircraft and is still in fairly widespread use, particularly for short-haul, commuter-type transports.

Figure 6.2 - Vickers Viscount 810 40-passenger turboprop airliner,- 1948. [Peter C. Boisseau]

The first civil airliner to be equipped with turboprop engines was the Vickers Viscount depicted in figure 6.2. The aircraft employed four Rolls-Royce Dart engines of 1600 horsepower each and had a gross weight of about 60 000 pounds. Depending upon the configuration, 40 to 59 passengers could be carried in the pressurized cabin. The cruising speed of the Viscount was 334 miles per hour at 25 000 feet. The aircraft employed double-slotted flaps and was equipped with a tricycle landing gear. The Viscount made its first flight in July 1948 and subsequently was used by airlines all over the world. A total of 441 Viscounts were built and many are still in use.

Two turboprop aircraft of much larger size were constructed in the United Kingdom. These were the Vickers Vanguard, with a gross weight of 146 500 pounds, and the Bristol Britannia, with a gross weight of 185 000 pounds. Many types of turboprop transport aircraft have been designed and built in Russia, as well. The largest passenger carrying turboprop ever built was the Tupolev Tu-114. This aircraft has a gross weight of 377 000 pounds and is equipped with four 14 795 equivalent shaft horsepower turboprop engines. Each of these engines drives two counterrotating propellers. The wings are sweptback, which is unusual for propeller-driven aircraft; the amount of sweep is 34°. The aircraft carries 220 passengers and cruises at a speed of 478 miles per hour at an altitude of 29 500 feet. The Tu-114 is no longer in airline use, but a version known as the Bear is employed by the Soviet military forces as a reconnaissance aircraft. The Lockheed Electra is the only large turboprop airliner to be developed in the United States. Although the Electra was an efficient high-performance aircraft, it was never produced in large numbers because it was introduced at about the same time as the Boeing 707 jet airliner and could not compete with this aircraft. A few Electras are still in service with the scheduled airlines, and a number are employed in nonscheduled activities. The naval version of the aircraft, known as the P-3 Orion, is employed by the U.S. Navy for antisubmarine patrol work.

Figure 6.3 - Lockheed C-130 turboprop cargo transport; 1955. [mfr]

A number of highly successful turboprop aircraft have been developed for use as cargo carriers. The largest of these aircraft is the Russian Antonov AN-22, which weighs over 550 000 pounds and is equipped with four 15 000-horsepower engines. The Lockheed C-130 is perhaps the best known of the turbo prop-powered cargo aircraft and the one that has been produced in the greatest numbers. The C-130 is used by all branches of the United States military forces and by the military forces of over 20 foreign governments. A commercial cargo version of the aircraft is also available. The first production contract for the aircraft was placed in 1952; over 1500 models of the C-130 have been built, and the aircraft is still in production.

A Lockheed C-130 is shown in figure 6.3. Many variations of the C-130 have been produced, and engines of slightly different power ratings have been employed. The aircraft has an unswept wing mounted in the high position at the top of the fuselage and is equipped with four Allison T-56 turboprop engines of 4910 equivalent shaft horsepower each at takeoff. In order to minimize weight and complexity, the landing gear is retracted into blisters located on either side of the fuselage, rather than into the wing or engine nacelles. The high wing position is advantageous for a cargo aircraft because it allows trucks and other types of equipment to move beneath.the wing, and the fuselage can be brought close to the ground without causing interference with the engines and propellers. A rear loading door may be deployed from the bottom of the upswept, aft portion of the underside of the fuselage. The proximity of the forward portion of the fuselage to the ground results in an aft-loading ramp with only a small inclination to the ground so that vehicles can be readily driven or pushed into the aircraft. The Lockheed C-130 has a gross weight of 155 000 pounds and cruises at a speed of 386 miles per hour at 20 000 feet. The wing loading is 88 pounds per square foot, and the landing speed is 115 miles per hour.

A great variety of twin-engine airliners has been developed both in the United States and abroad during the postwar years. These aircraft are smaller than the large, long-range, four-engine aircraft and are employed on short-haul types of operations. Twin-engine airliners have been developed with both reciprocating and turboprop engines. The twin-engine Martin 404 and Convair 440 aircraft and earlier versions of these machines were perhaps the most-used postwar twin-engine transports powered with reciprocating engines. These aircraft are similar in configuration to the Douglas DC-3 but are larger, faster, and are equipped with pressurized cabins; in addition, they both employ the tricycle type of landing gear. The Fairchild F-27 (a Dutch Fokker design built under license by Fairchild in this country) and the Japanese YS-IIA are probably the best known turboprop twins in the United States. The British Hawker Siddeley 748 turboprop-powered twin-engine airliner is widely used in many countries of the world.

Although the long-range propeller-driven transport is essentially a thing of the past, smaller, short-range aircraft of this type are becoming more numerous. Since the advent of airline deregulation in the United States in the latter part of the 1970's, there has been a large growth in short-haul, commuter-type airline operations. Many aircraft employed in this type of service are foreign built, are of high-wing configuration, and are equipped with two turboprop engines. Passenger capacity varies between 20 and 30, and at least one four-engine aircraft of this type carries 50 passengers. Generally speaking, these aircraft have straight wings and employ no new configuration concepts. In fact, some of them have fixed landing gears and strut-braced wings. Since high speed is unimportant and low initial and maintenance costs are critical, these retrogressive technical features are justified on a cost-effectiveness basis. The final forms of the commuter-type transport, however, are yet to emerge.

GENERAL AVIATION AIRCRAFT

The term "general aviation" covers all types of flying except military and commercial airline operations. Only contemporary aircraft designed for business and pleasure are considered here. General aviation aircraft designed for business and pleasure are available in both single-engine and twin-engine models; most models are equipped with horizontally opposed reciprocating engines. However, several high-performance turboprop types are offered. Single-engine types may be had with high- or low-wing location, retractable or fixed landing gear, controllable-pitch or fixed-pitch propeller, and in sizes varying from two place to seven place. The twin-engine aircraft usually employ the low wing location and have retractable landing gear and controllable-pitch propellers. The twins may be had with or without turbosupercharging, with or without pressurized cabins, and with varying seating capacities. The modern aircraft designed for business or pleasure is almost invariably of all-metal construction, as contrasted with the metal, wood, and fabric construction typical of the pre-World War II general aviation aircraft. Reliability of the internal systems employed in the aircraft and the precision of the radio and navigational equipment have greatly improved as compared with pre-World War II standards. The general aviation aircraft of today are almost universally equipped with an electrical system to power the radios and other types of equipment installed in the aircraft and to operate the self-starter. Hand starting of production aircraft is a thing of the past. The cabins of these aircraft are generally relatively comfortable, are equipped with heaters for wintertime and high-altitude use, and are sometimes equipped with air conditioning for use on the ground and at low altitudes in the summer. The open cockpit is a thing of the past in production aircraft, except for special sport and aerobatic aircraft. Many aircraft employ complete instrumentation and communication equipment for flight under IFR conditions. Most contemporary aircraft employ a tricycle gear that greatly eases the problem of aircraft handling on the ground. The basic aerodynamic configuration of contemporary general aviation aircraft, however, differs little from those in use in 1939.

CONTEMPORARY TYPES, 1970-80

General aviation aircraft are manufactured in a number of different countries; however, the majority of these aircraft are produced in the United States. The major U.S. producers are the Cessna Aircraft Company, the Piper Aircraft Corporation, and the Beech Aircraft Corporation. Each offers a wide variety of aircraft designed for various needs and markets. Six aircraft of different levels of performance, size, and price produced by these manufacturers for different segments of the market are briefly described here.

Two single-engine aircraft representative of the lower performance and price spectrum are shown in figures 6.4 and 6.5. The Piper Cherokee 180 shown in figure 6.4 is an all-metal aircraft with an internally braced, cantilever wing mounted in the low position. The aircraft shown has four seats and is equipped with a 180-

Figure 6.4 - Piper Cherokee 180 contemporary general aviation aircraft. [NASA]

horsepower, four-cylinder Lycoming engine of the opposed type. The engine drives a fixed-pitch propeller. The landing gear on the aircraft is fixed, and although not visible in the photograph, the horizontal tail employed on the Cherokee is of the all-moving type equipped with a geared tab. The Cherokee 180 has a maximum speed of 148 miles per hour at sea level and cruises at 141 miles per hour at 7000 feet. The stalling speed with the split flaps deflected is 61 miles per hour. The gross weight of the aircraft was 2450 pounds. The Cherokee 180 is representative of one of the lower cost members of a complete family of Piper aircraft that carry the Cherokee name. Some of these aircraft have six or seven seats and more powerful engines that drive controllable-pitch propellers. Other versions of the Cherokee employ a retractable landing gear. The flight of the first production aircraft was made in February 1961, and well over 25 000 Cherokees of all types have now been produced.

The Cessna Skyhawk shown in figure 6.5 is one of the lower cost members of an entire series of Cessna aircraft of the same basic configuration. The Skyhawk, like the Cherokee 180, is equipped with a fixed tricycle landing gear and has a four-cylinder, horizontally opposed engine driving a fixed-pitch propeller. Unlike the Cherokee 180, however, the Cessna Skyhawk is a high-wing configuration with a single wing strut on either side of the fuselage to brace the wing. The Skyhawk has a maximum speed of 144 miles per hour and cruises at 138 miles per hour at 8000 feet. The stalling speed with the flaps deflected is 49 miles per hour. The gross weight of the Cessna Skyhawk is 2300 pounds, and the wing loading and power loading are 13.1 pounds per square foot and 15.3 pounds per horsepower, respectively. These values are in the same order as for the Piper Cherokee. The zero-lift drag coefficient of the Skyhawk is 0.0319 as compared with 0.0358 for the Cherokee, and the maximum lift-drag ratios for the two aircraft are 11.6 and 10.0, respectively.

Figure 6.5 - Cessna Skyhawk contemporary general aviation aircraft. [mfr]

Two representative high-performance single-engine general aviation aircraft are shown in figures 6.6 and 6.7. The Beech Bonanza V-35B shown in figure 6.6 is of all-metal construction, has an internally braced wing mounted in the low position, has single-slotted flaps, and is equipped with a fully retractable tricycle landing gear. The aircraft is equipped with a six-cylinder, horizontally opposed Continental engine of 285 horsepower that drives a controllable-pitch propeller. The aircraft can be configured for four, five, or six seats. The unique Butterfly tail combines the stability and control functions of both the conventional vertical and horizontal tails. The gross weight of the aircraft is 3400 pounds. The aircraft has a maximum speed of 210 miles per hour at sea level, cruises at 203 miles per hour at 6500 feet, and has a stalling speed of 63 miles per hour. The zero-lift drag coefficient is a very low 0.0192, and the corresponding maximum lift-drag ratio is 13.8. The prototype of the Bonanza first flew in December 1945, and the aircraft has been continuously in production since 1947. Approximately 10 000 Beech Bonanzas have been built.

Figure 6 6 - Beech Bonanza V-35B contemporary general aviation aircraft. [mfr]

The Cessna Cardinal RG II shown in figure 6.7 is a high-performance aircraft with an internally braced wing mounted in the high position. The aircraft is equipped with a fully retractable tricycle landing gear and is equipped with a four-cylinder, horizontally opposed, Lycoming engine of 200 horsepower that drives a controllable-pitch propeller. The Cardinal is of all-metal construction and is equipped with trailing-edge flaps and an all-moving horizontal tail employing a geared trim tab. The aircraft has a maximum speed of 180 miles per hour at sea level, cruises at 171 miles per hour at 7000 feet, and has a stalling speed of 57 miles per hour. The aircraft weighs 2800 pounds. The.zero-lift drag coefficient of the Cardinal is 0.0223, and the corresponding maximum lift-drag ratio is 14.2.

Figure 6 7 - Cessna Cardinal RG II contemporary general aviation aircraft. [mfr]

The first twin-engine aircraft designed specifically for business use was probably the Beech Model D-18, first produced in 1937. This aircraft was similar to the Douglas DC-3 in general appearance, although much smaller, and was in continuous production from 1937 until the early 1970's. A wide variety of twin-engine aircraft of various sizes and with different levels of performance are now offered for business use. Two contemporary twin-engine aircraft are shown in figures 6.8 and 6.9.

The Cessna 310 shown in figure 6.8 is representative of one of the smaller contemporary twin-engine aircraft offered for business use. The aircraft is a low-wing configuration with an engine mounted in each wing on either side of the fuselage. The aircraft can be had with both normally aspirated engines or with turbosuperchargers. The engines are six-cylinder, horizontally opposed, Continental engines

of 285 horsepower each that drive controllable-pitch, full-feathering propellers. The aircraft normally has a seating capacity of five but can be configured for six. Maximum speed is 238 miles per hour at sea level, and cruising speed is 223 miles per hour at 7500 feet. The wings are equipped with split flaps which with a wing loading of 30.7 pounds per square foot result in a stalling speed of 77 miles per hour. The gross weight of the aircraft is 5500 pounds. The. Cessna 310 has a zero-lift drag coefficient of 0.0267 and a maximum lift-drag ratio of 13. The Cessna 310 was first flown in January 1953 and has been in continuous production ever since. The aircraft is unpressurized and may be thought of as the smallest of a whole line of Cessna twins, both pressurized and unpressurized.

Figure 6.8 - Cessna 310 contemporary twin-engine general aviation aircraft. [mfr]

The Beech Super King Air 200 shown in figure 6.9 is an example of a new, relatively large, high-performance twin-engine business aircraft. Provision is provided for 2 pilots and 6 to 13 passengers, depending on the configuration. The cabin is pressurized to permit comfortable cruising flight at high altitudes. Power is provided by two Pratt & Whitney PT6A-41 turboprop engines of 850 shaft horsepower each. The engines drive controllable-pitch, full-feathering, reversible propellers. The low-wing configuration of the aircraft is conventional although the use of a T-tail on a straight-wing propeller-driven aircraft is somewhat unusual. The use of this tail arrangement is said to reduce both vibration resulting from the slipstream of the engines and trim changes with flap deflection. The aspect ratio of the wing is 9.8, which must be considered as relatively high for any aircraft. The King Air 200 has a maximum speed of 333 miles per hour at 15,000 feet and a maximum cruising speed of 320 miles per hour at 25,000 feet. The aircraft is equipped with single-slotted flaps that together with a wing loading of 41.3 pounds per square foot give a stalling speed of 92 miles per hour. The gross weight of the aircraft is 12,500 pounds. The Beech Super King Air 200 was certified in December 1973 and is now in series production.

Figure 6.9 - Beech Super King Air 200 contemporary twin-engine turboprop general aviation aircraft. [mfr]

OTHER TYPES OF GENERAL AVIATION AIRCRAFT

The six aircraft just described may be considered as representative of generic classes of aircraft designed for business and pleasure use. In order to gain a true appreciation of the wide variety of such aircraft offered today, the reader is referred to the current year's issue of Jane's All The World's Aircraft. Other types of aircraft of interest and not described here are specially designed agricultural aircraft intended for spraying and dusting crops. These aircraft will also be found in Jane's, as will many types of sport and aerobatic aircraft. Another segment of general aviation aircraft is made up of the so-called home builts. These aircraft, which are built by individuals or clubs at home, are gaining in popularity and are flown in relatively large numbers in this country. They are usually not certified under any of the pertinent federal air regulations but, rather, operate in an experimental category. Many of the more popular types of home-built designs are also described in Jane's All The World's Aircraft.

Document 3-33(b), Laurence K. Loftin, Jr. Chap. 6, "Design Trends," in* Quest for Performance: The Evolution of Modern Aircraft *(Washington, DC: NASA SP-468, 1985).

CHAPTER 7: DESIGN TRENDS

INTRODUCTION

This chapter briefly summarizes the progress in design of propeller-driven aircraft since the end of World War I by showing how a number of important design and performance parameters varied over the years 1920 to 1980. The following parameters are discussed:

(1) Maximum speed, V_{max}
(2) Stalling speed, V_s
(3) Wing loading, W/S
(4) Maximum lift coefficient, $C_{L,max}$
(5) Power loading, W/P
(6) Zero-lift drag coefficient, $C_{D,O}$
(7) Skin friction parameter, C_F {Ptr, in number 7, make a line over the "C" as shown.}
(8) Maximum lift-drag ratio, $(L/D)_{max}$

The values of each of these parameters are plotted against the appropriate year in figures 7.2 to 7.9. All of the parameters could not be obtained for some of the aircraft; in particular, the zero-lift drag coefficient and the maximum lift-drag ratio could not be determined for a number of the aircraft because of insufficient performance data from which to make the desired calculations. The symbols identifying each aircraft are given in figure 7.1 and have been used throughout the subsequent figures. At the left side of each figure (figs. 7.2 to 7.9), bars have been drawn to indicate the spread of each parameter during World War I. The year for which the

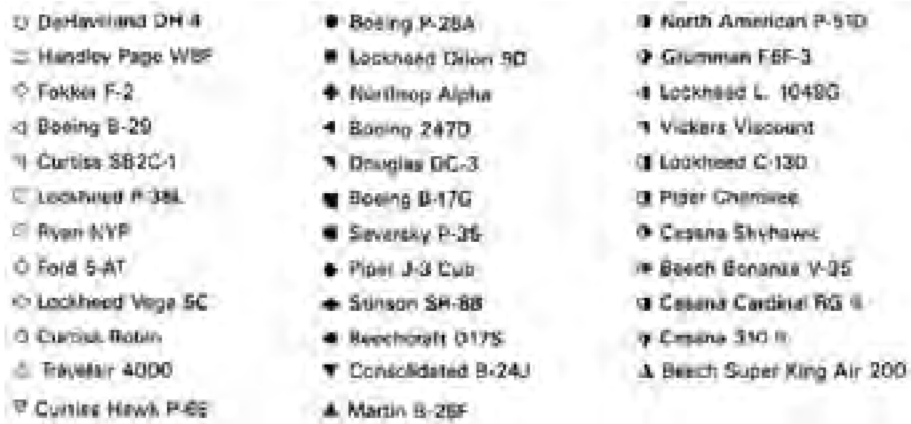


Figure 7.1 - Symbols used in figures 7.2 to 7.9.

characteristics of a given aircraft are plotted is in some degree arbitrary. For example, most of the World War II aircraft characteristics are plotted for the year 1942. In other cases, aircraft that were used for a number of years are shown at a year corresponding to the first year of production, or after the aircraft had achieved a fully developed status. The points for the different aircraft show a large spread in the different figures; hence, lines representing an upper and lower bound are shown on each figure. (The shape of these bound lines may be varied according to the manner in which the data are interpreted. The lines shown are only suggested fairings of the data points presented.) One of these bounds corresponds to aircraft developed with the highest technology available at a particular time, and the other is for aircraft of a relatively low and slow-changing level of technology. Neither of these bounds represents boundaries of maximum and minimum values but, rather, corresponds to higher and lower levels of technology for operational aircraft of a particular time period. No data for racing or special performance aircraft are given in the figures.

MAXIMUM SPEED

Trends in maximum speed of propeller-driven aircraft are shown as a function of time in figure 7.2. The maximum speeds of high-technology operational aircraft are seen to increase steadily from about 125 miles per hour in 1920 to nearly 450 miles per hour in the World War II years. The highest maximum speed shown is for the P-51D aircraft, which had a speed of 437 miles per hour at 25 000 feet. Late in the war, a Republic P-47J achieved a speed in level flight of 507 miles per hour at 34 000 feet. The upper bound through the years closely follows the advancement of fighter-type aircraft. The large increases in maximum speed that occurred between World War I and World War II resulted from increases in engine power and reductions in drag area through improved aerodynamic efficiency. For example, the 10 000-pound P-51 fighter of World War II had a drag area of only 3.8 square feet (this corresponds to a circular disc 2.20 feet in diameter) and was equipped with an engine of 1490 horsepower; by comparison, one of the highest performance fighters in use at the end of World War I, the 1807-pound SPAD XIII C.1 (chapter 2), had a drag area of 8.33 square feet (a circular disc 3.26 feet in diameter) and was powered with a 200-horsepower engine. The corresponding values of the ratio of power to drag area are 392.11 and 24.01, respectively. Also contributing significantly to the large increases in maximum speed were the development of the supercharger and controllable-pitch propeller, both of which permitted efficient high-power flight in the low-density, high-altitude environment. No increases in the maximum speed of operational propeller-driven aircraft have been achieved since the end of World War II because of the inherent limitations imposed by the effects of compressibility on the efficiency of conventional propellers.

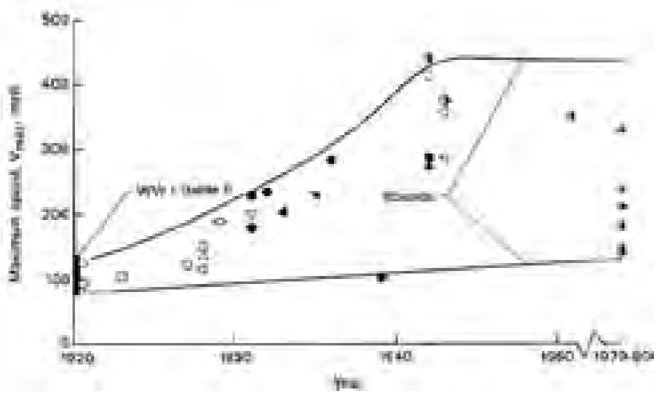


Figure 7.2 - Trends in maximum speed of propeller-driven aircraft.

The lower bound in figure 7.2 shows an increase in maximum speed from about 80 miles per hour to about 130 miles per hour. This bound indicates a continued desire for low-performance aircraft throughout the years. The general aviation aircraft of today are seen to encompass a range of maximum speed from about 130 miles per hour to almost 350 miles per hour, which indicates the wide range of technical sophistication in contemporary propeller-driven aircraft.

Although not shown in the data presented in figure 7.2, the performance of representative, specially built, propeller-driven racing aircraft through the years may be of some interest and is indicated as follows:

1. 1913, absolute speed record of 126.64 miles per hour established by French Deperdussin landplane
2. 1920, absolute speed record of 194.49 miles per hour established by French Nieuport 29V landplane
3. 1923, absolute speed record of 267.16 miles per hour established by American Curtiss R2C-1 landplane
4. 1927, absolute speed record of 297.83 miles per hour established by Italian Marcchi M-52 seaplane

5. 1931, absolute speed record of 406.94 miles per hour established by British Supermarine S-6B seaplane
6. 1934, absolute speed record of 440.60 miles per hour established by Italian Marcchi-Castoldi MC-72 seaplane (This record for propeller-driven seaplanes still stands and is unlikely to be surpassed in the near future.)
7. 1938, absolute speed record of 469.22 miles per hour established by German Messerschmitt 209VI landplane
8. 1969, absolute speed record of 483.04 miles per hour established by highly modified American Grumman F8F landplane

The world speed records cited above are officially recognized by the Federation Aeronautique Internationale and were established under sea-level flight conditions.

STALLING SPEED, WING LOADING, AND MAXIMUM LIFT COEFFICIENT

The stalling speed, wing loading, and maximum lift coefficient are shown as a function of years for various aircraft in figures 7.3, 7.4, and 7.5. The short, unpaved fields that served as airports in the early 1920's, together with the relatively poor flying characteristics of aircraft of that period, dictated the necessity for low values of the stalling speed. Values of the stalling speed of 40 to 50 miles per hour were not unusual, although precise data are not shown in figure 7.3 for the year 1920. High-lift devices were essentially unknown at that time; hence, the wing loadings needed to give the low values of the stalling speed were correspondingly low, as shown in figure 7.4. Values of the wing loading from 5 to 10 pounds per square foot were typical, and the 14-pound wing loading of the DH-4 was considered high in 1920. For a given atmospheric density, the wing loading is, of course, related to the square of the stalling speed by the value of the wing maximum

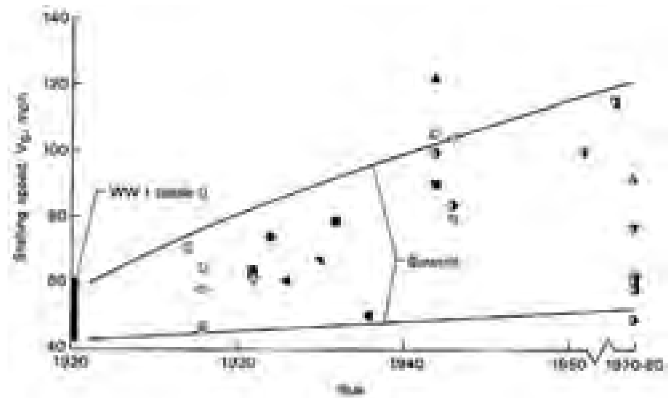


Figure 7.3 - Trends in stalling speed of propeller-driven aircraft.

lift coefficient. Values of the maximum lift coefficient slightly in excess of a value of 1 were typical of unflapped aircraft with thin airfoil sections in 1920, as shown in figure 7.5. The demands for increased high-speed performance resulted in increases in wing loading and, hence, increases in the stalling speed. By the time of World War II, the stalling speeds of high-performance military aircraft were in the range of 80 to 100 miles per hour; wing loadings were in the range of 40 to 60 pounds per square foot. The development and the associated use of powerful high-lift devices, such as described in chapter 5, resulted in aircraft maximum lift coefficients of the order of 2.0 to 2.5 for high-performance aircraft in the World War II period. These high-lift devices, and consequent high maximum lift coefficient, prevented the stalling speed from increasing to an even greater extent than that shown in figure 7.3. Since World War II, the stalling speed of high-performance aircraft has continued to increase and is seen in figure 7.3 to be 115 miles per hour for the contemporary Lockheed C-130 cargo transport. The wing loading for this aircraft is about 90 pounds per square foot, as shown in figure 7.4, and the maximum lift coefficient is about 2.75. The highest maximum lift coefficient of any of the aircraft for which data are shown in figure 7.5 is about 3.0 and was obtained by the Lockheed Model 1049G Constellation. The corresponding wing loading for this aircraft is about 80 pounds per square foot. The high maximum lift coefficient of the Constellation gave a relatively slow stalling speed of about 100 miles per hour.

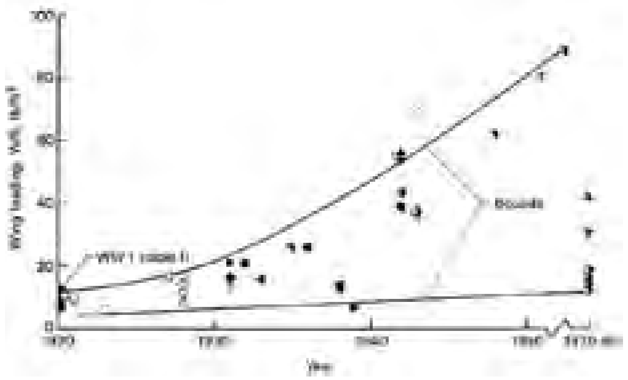


Figure 7.4 - Trends in wing loading of propeller-driven aircraft.

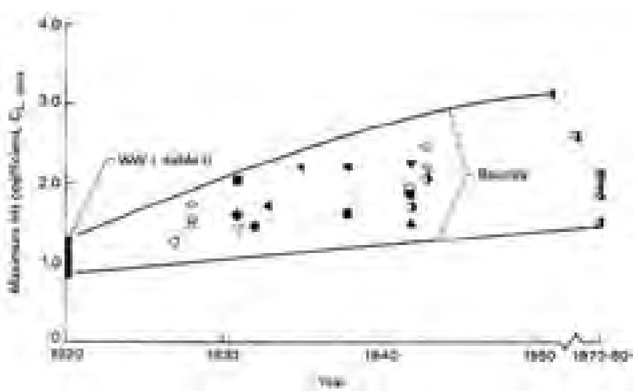


Figure 7.5 Trends in maximum lift coefficient of propeller-driven aircraft.

The lower bounds in figures 7.3, 7.4, and 7.5 show modest increases in stalling speed, wing loading, and maximum lift coefficient for aircraft of relatively low performance. The data for current general aviation aircraft show a wide spread in level of technology, insofar as maximum lift coefficients are concerned, and a wide range of values of stalling speed and wing loading. Values of maximum lift coefficient for these aircraft vary from about 1.3 to about 2.2. The higher values of maximum lift coefficient achieved by current high-technology general aviation aircraft are about the same as those of military aircraft in World War II. The wing loading and stalling speeds of the high-performance general aviation aircraft of today are also seen to be in the same order as those of World War II military aircraft.

POWER LOADING

The power loading data shown in figure 7.6 appear to have nearly constant values for the upper and lower bounds. Within these bounds, the transport and bomber-type aircraft have power loadings that vary from about 12 pounds per horsepower in 1928 to 8 to 10 pounds per horsepower by the 1950's. Low-performance aircraft have a higher upper bound value of the power loading of about 16 pounds per horsepower although the venerable Piper Cub J-3 had a power loading value of about 19 pounds per horsepower. The lower bound of the power loading is formed by fighter aircraft, which tend to have power loadings in the range from 5 to 6 pounds per horsepower. These low values of power loadings have, through the years, been dictated by the rate of climb and maneuvering performance characteristics required in fighter-type aircraft. Present-day general aviation aircraft have power loadings that vary from nearly 16 pounds per horsepower for the very low-performance type of pleasure or training aircraft to about 8 pounds per horsepower for the high-performance Beech King Air 200 (at low altitude).

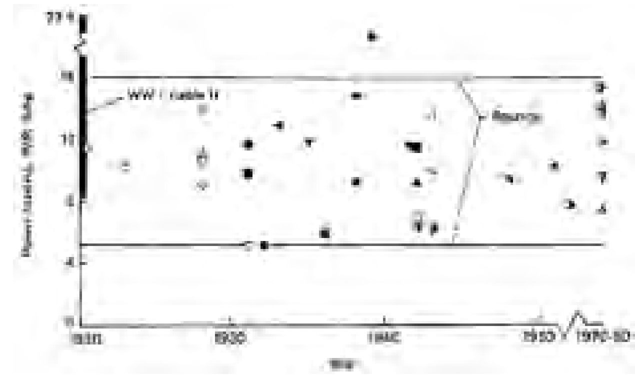


Figure 7.6 Trends in power loading of propeller-driven aircraft.

ZERO-LIFT DRAG COEFFICIENT AND SKIN FRICTION PARAMETER

The value of the zero-lift drag coefficient $C_{D,0}$ is often used as an indicator of the aerodynamic cleanness or refinement of an aircraft. Values Of $C_{D,O}$ calculated according to the methods of appendix C are shown as a function of years in figure 7.7. The lower bound of $C_{D,0}$ drops sharply from a value of about 0.040 in 1920 to a value of about 0.021 in the early 1930's. A smaller reduction in the lower bound values

Of $C_{D,0}$ took place in the years between the early 1930's and the years of World War II. The general aviation aircraft of today show a spread in the values Of $C_{D'O}$ from near the upper bound to near the lower bound. The lower bound curve shows the dramatic reduction in $C_{D,0}$ that accompanied the basic change in airplane configuration from a strut-and-wire-braced biplane with a fixed landing gear to the highly streamlined, internally braced monoplane with retractable landing gear. As indicated in chapter 4, this transformation had largely taken place for high-performance operational aircraft by the early 1930's. Detailed aerodynamic refinements such as described in chapter 5 were responsible for further improvements in aerodynamic efficiency as indicated by the lower bound curve. The zero-lift drag coefficient, although useful as a measure of comparative aerodynamic refinement, has a basic limitation because the coefficient is based on wing area, and, for a given wing area, many different fuselage and tail sizes may be employed. Thus, differences in zero-lift drag coefficients may be interpreted as a difference in aerodynamic refinement when the difference may result from a significant difference in the ratio of wetted area to wing area.

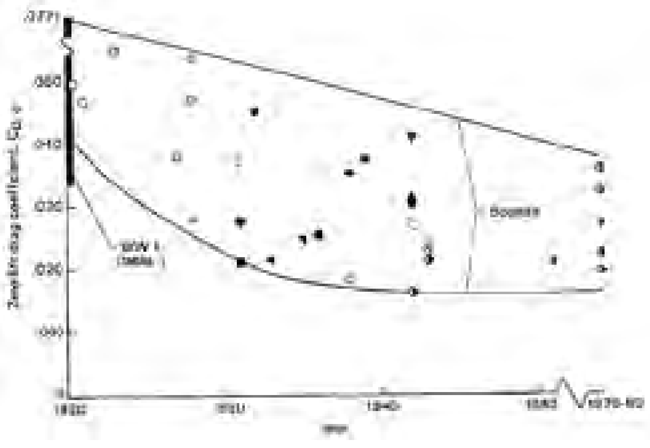


Figure 7. 7 - Trends in zero-lift drag coefficient of propeller-driven aircraft.

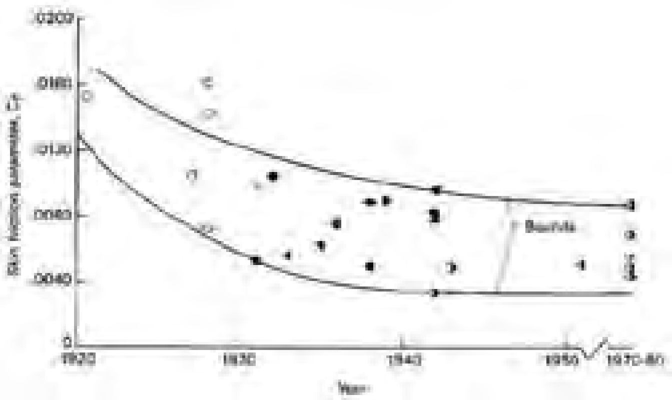


Figure 7.8 - Trends in skin friction Parameter C_F, {ptr: line over C} of propeller-driven aircraft. [ref. 90].

In order to remove the effect of variations in the ratio of wetted area to wing area, a zero-lift drag coefficient based on total wetted area rather than wing area was estimated in reference 90 for most of the aircraft for which drag data are given in figure 7.7. The reference area for

this coefficient, termed the skin friction parameter C_F {ptr: line over C} consisted of the total surface area of the fuselage, wings, and tail surfaces. The parameter C_F {ptr: line over C} was obtained from multiplication of $C_{D'O}$ {ptr: line over C} by the ratio of wing area to total wetted area. Values of C_F, {ptr: line over C} taken from reference 90 are shown as a function of years in figure 7.8. The upper and lower bounds of the data show the same trends as do those for the zero-lift drag coefficient shown in figure 7.7. The lower bounds of the skin friction parameter indicate that essentially no progress has been made in reducing C_F {ptr: line over C} since World War II, and little progress has been made since the early 1930's. The data for the current general aviation aircraft fall generally between the upper and lower bounds but do not reach as low a value as that of the lower bound curve. This suggests that these aircraft can be refined to a value at least as low as that achieved during World War II. There is little likelihood, however, that values of C_F {ptr: line over C} significantly lower than the lower bound shown in figure 7.8 can be achieved unless some breakthrough is made that permits the achievement of a significant extent of laminar flow on the aircraft. Other than reductions in the value of the skin friction parameter, future reductions in the airplane zero-lift drag coefficient $C_{D,O}$ {ptr: line over C} can perhaps be achieved through configuration design aimed at reducing the ratio of wetted area to wing area. The pure flying wing represents the ultimate improvement by this means.

MAXIMUM LIFT-DRAG RATIO

The maximum lift-drag ratio of the various aircraft was calculated according to the methods described in appendix C and is shown as a function of years in figure 7.9. The value of the maximum lift-drag ratio $(L/D)_{max.}$ is a measure of the aerodynamic cruising efficiency of the aircraft. The upper bound of $(L/D)_{max}$ varies from values of about 9 in 1920 to a value of 16.8 for the World War II Boeing B-29 and 16.0 for the Lockheed 1049G in 1952. The $(L/D)_{max}$ upper-bound curve shows a sharp rise between 1920 and the early 1930's, which corresponds to the reduction in zero-lift drag coefficient shown in figure 7.7 and to the emergence of the monoplane with its higher aspect ratio as compared with the biplane. Little change in maximum L/D has taken place since the end of World War II. Any further increases in maximum lift-drag ratio will require reductions in the value of the zero-lift drag coefficient and/or increases in wing aspect ratio that may be possible through the use of improved structural materials.

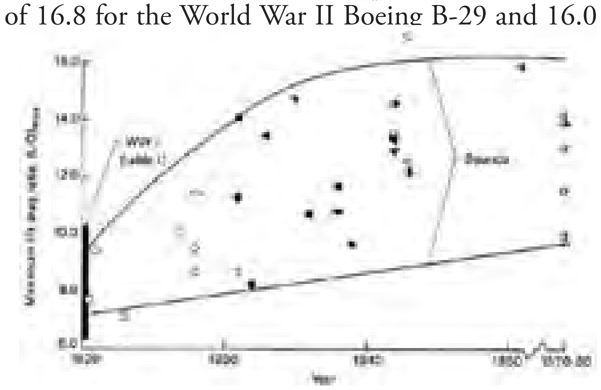


Figure 7.9 - Trends in maximum lift-drag ratio of propeller-driven aircraft.

Chapter Four

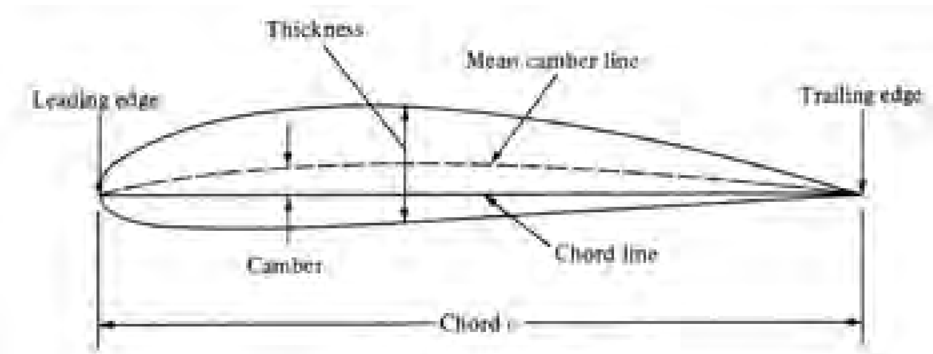
On the Wing

Destination Document:

Historical retrospective from aeronautical engineer Ed Rees of North American Aviation, Inc., in "A Tribute to Dutch Kindelberger: The Mustang—A Great War Horse," The Airpower Historian [9] (Oct. 1962): 201.

The design touchstone of the Mustang was the laminar flow wing, a high-lift, low-drag airfoil developed by the National Advisory Committee for Aeronautics (NACA) and considered by most engineers as being too revolutionary for use in a mass-produced airplane. The chiefs of the aerodynamics sections believed in it so thoroughly that they promised in case of failure to produce a conventional wing within 30 days. Since wing design is the toughest of all components and usually determines the schedule for the rest of the design job, the entire project hung tenuously on the then-unproven laminar flow concept.

"The Bird is on the Wing" is a well-known line from Edward Fitzgerald's 1859 translation of *The Rubaiyat of Omar Khayyam*, written in the late 11th century. The medieval Persian poet refers metaphorically to the fleeting "Bird of Time," which "has but a little way to fly," and certainly not to anything practical about the actual technology of flying—not even to flying carpets. Still, from an engineering perspective, nothing could be truer about the mechanics of flight. The essence of an



Basic to airfoil geometry are the coordinates of the upper and lower surface, as well as such parameters as maximum thickness, maximum camber, position of maximum thickness, position of maximum camber, and nose radius. From the beginning of airfoil research and development, aeronautical engineers and aircraft designers generated airfoil sections simply by adjusting these parameters. Over the years, these adjustments became more and more analytical and systematic. Courtesy of John D. Anderson, Jr.

airplane is unquestionably its wing; it is what the "bird" is all about. Designed to lift the machine up into the air and sustain its flight, it is the structure that performs the most basic, required functions; it also embodies the most aesthetic and ethereal aspect of the airplane's overall form and function.

Designing a highly effective wing for a particular aircraft requirement has proven over the decades to be one of the most challenging tasks facing aerodynamicists. More theoretical work and experimental energy has gone into exploring the complexities of wing design than into the delineation of any other aerodynamic component, and none has brought greater dividends. Similarly, nowhere in aeronautical technology has "engineering as artistry" been more evident than in the creative process of designing wing shapes. Higher mathematics and the most advanced wind tunnels have systematically explored the mysterious interactions between wing shapes and airflows, but aesthetic considerations have played a vital role in wing design even into the modern electronic age of the high-speed computer.

As the last line of the Destination Document for this chapter suggests, whether it is for a fabulously successful airplane like the P-51 Mustang of World War II fame or for some miserable failure of a machine (and there have been more than a few in aerospace history), the wing has proven to be not only the toughest of all aerodynamic components to design but also the one on which the ultimate fate of most aerodynamic configurations has depended. It should be obvious why, in a documentary study of aerodynamic development, the history of the evolution of the wing should be singled out. So much about the progress of aeronautics from before the Wright brothers to the present has depended fundamentally upon determining ever more effective airfoil shapes for various wing applications.

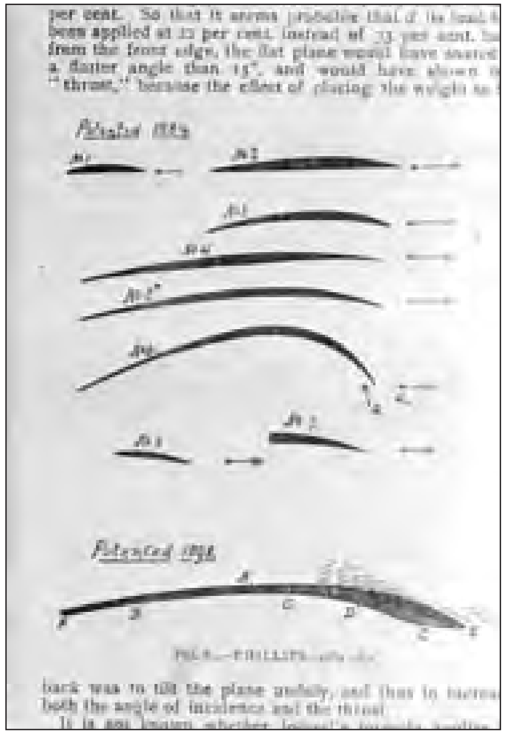
No era of aeronautical history witnessed more rapid improvement in airfoil design than the period from the mid-1920s into the early years of World War II. In a little over a decade's time, as the propeller-driven airplane reached its mature (and some might argue, ultimate) form, the wings of the world's airplanes evolved from intuitively derived, cut-and-try, and aerodynamically primitive shapes into advanced streamlined, highly reliable, and sometimes counterintuitive structures that were systematically engineered, a great number of them in the NACA laboratory. Demonstrating how that critical evolution took place, and how the NACA's research program contributed to it from the 1920s into World War II, is the main purpose of this chapter's documentary collection. More specifically, the drama of the chapter is provided by the NACA's quest for the so-called laminar flow airfoil, the aerodynamicist's dream of the ultimate low-drag wing that would enable a streamlined aircraft to fly more aerodynamically "pure" than ever before.

As the reader will see in the collection of documents to come, the path to the achievement of laminar-flow wings by the NACA, to the limited extent they *were* achieved, was anything but simple or straight. Like the search of medieval alchemists to turn base metal into gold, or anyone else looking for the pot of gold at the end of a rainbow, the quest of NACA aerodynamicists for a wing that would possess

the many advantages of laminar flow was full of adventure, promise, and expectation of final achievement. In the end, however, it fell short of its lofty goal, a critical disappointment, made worse for those most directly involved by the exaggerated claims and hype that the NACA and other American aviation publicists had made for the endeavor along the way.

To place the NACA's laminar flow airfoil development in full historical context, one must go back to the history of wing section research and aerodynamic development more generally, starting right after the Wright brothers.[1]

Even with the Wright brothers' experience with airfoils in mind, the earliest airplane designers possessed such scanty knowledge of aerodynamics that they could do little more than guess at how any sort of lifting surface they drew up would actually perform in flight. Common sense and observation of the long graceful wings of many large birds in flight suggested that efficient wings needed to be long and slender so as to use as much air as possible to support the carrying weight. Moreover, the proven successes of the wing shapes employed by Lilienthal, Langley, and especially the Wrights indicated that a certain amount of wing curvature (or camber) gave much more satisfactory results than did flat surfaces. Thus, airplane designers of the early 20th century followed the lead of the inventors of the airplane and went with the long, thin wing. This shape complicated the actual construction of the airplane, of course, because a long, slender wing arrangement could only be accomplished structurally at the time by employing a deep truss. Typically,



The first serious work on the development of airfoil sections began in the 1880s. Although earlier experimenters had shown that flat plates produced lift when set at an angle of incidence, some suspected that shapes with curvature more closely resembling bird wings would produce more lift. British experimenter Horatio F. Phillips (1845-1912) patented a series of more highly cambered airfoils after testing them in a wind tunnel he had built. He continued to patent curved airfoils into the 1890s. National Air and Space Museum, Smithsonian Institution (SI A-32247-D)

[1] Significant parts of the introductory essay to follow derive from James R. Hansen, *Engineer in Charge: A History of the Langley Aeronautical Laboratory, 1917-1958*, NASA SP-4305 (Washington, 1987), by permission of the author.

Octave Chanute understood the importance of wing shape for flight. In 1893 he wrote that "it seems very desirable that further scientific experiments be made on concavo-convex surfaces of varying shapes, for it is not impossible that the difference between success and failure of a proposed flying machine will depend upon the sustaining effect between a plane surface and one properly curved to get a maximum of 'lift'." National Air and Space Museum, Smithsonian Institution (SI 84-10696)

this type of construction meant a biplane arrangement involving an aerodynamically messy array of supporting wires and struts bracing two sets of wings. Only a very minor degree of streamlining was achieved, which was done not by delineating improved airfoil shapes but simply by enclosing the framework of a wing completely rather than by just covering it with a single stretched-tight cloth surface on one side, as had been the case with the pioneering gliders and airplanes.

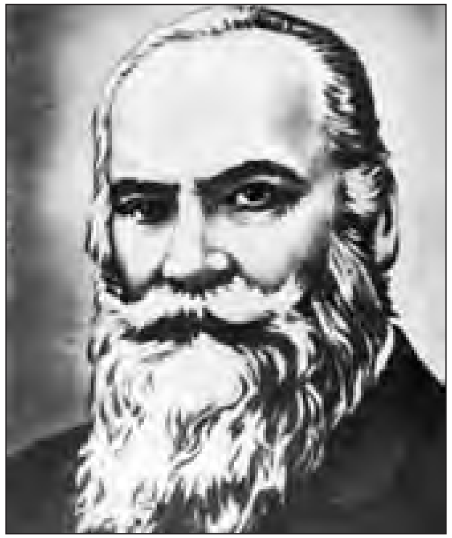
Some basic theory that would later prove extremely helpful to the understanding of lifting surfaces developed early in the century, but few airplane builders or others interested in the practical engineering of airplanes knew of it—or knew enough higher mathematics to take advantage of this information. In an article first published in Germany in 1912, but not available in the United States until after World War I, Professor Ludwig Prandtl at the University of Göttingen proposed what came to be known as the "lifting-line theory." (Prandtl was influenced in this hypothesis by the earlier work of England's Frederick W. Lanchester, who in key respects initiated the first great age of theoretical aerodynamics in the 1890s with his concepts—rejected by fellow scientists at the time—of how lift was generated in relation to a circulatory flow about an airfoil and how trailing vortices at the tips of wings caused drag, what later became known as "induced drag.") Without delving into the complexities of the mathematics involved, what the lifting-line theory did was quantify the study of a wing (or any other type of airfoil, whether it was a propeller blade, tail fin, or a rudder) by permitting its "characteristics" to be segregated into two elements that could be considered separately: those intrinsically associated with the shape of the wing (cross-) section and those associated with

Designing "by eye" and without any attempt at systematization, Louis Blériot was able to increase curvature over the forward part of the wing section of his historic number 11 airplane. The added wing camber in that location helped the airplane to make its historic crossing over the English Channel in 1909. National Air and Space Museum, Smithsonian Institution (SI 78-14792)

the wing planform (the outline of the wing when viewed from above) or twist (any varying angles of attack along its span or length). With greater knowledge of wing section characteristics in mind, the aerodynamicist could then calculate roughly the angle of zero lift, the lift curve slope, the span loading, the drag for high speed and cruise conditions, and the maximum lift coefficient and stalling characteristics. Lifting-line theory did not work for every performance parameter of a wing, and generalized solutions of the equations were far too complex for complete answers. Nonetheless, aerodynamicists following Lanchester's and Prandtl's lead were able to provide some useful approximations and began to pursue studies of their own that eventually promoted a greatly enriched mathematical understanding of how wings worked and how their shapes might be improved.

Wing sections of different outlines, thicknesses, and curvatures began to proliferate in the 1910s, and virtually all of them were designed "essentially by eye and without any attempt at systematization."[2] In this manner, Frenchman Louis Blériot was able to increase the curvature over the forward part of his wing section successfully, which helped his airplane to make its historic crossing of the English Channel in 1909. Similarly, between 1912 and 1915, British engineers at the fledgling Royal Aircraft Factory managed to design a series of practical airfoils, some of them made thicker and with a rounder leading edge and sharper trailing edge, resulting in slightly improved aerodynamic performance. In truth, though, the Royal Aircraft Factory engineers evolved their shapes principally with structural considerations in mind rather than any novel aerodynamic insight. Designers in the United States followed the British lead, either copying the dimensions of a superior R.A.F section exactly (notably R.A.F, 6 or 15) or modifying one only slightly, as in the case

[2] Clark B. Millikan, *Aerodynamics of the Airplane* (New York: John Wiley & Sons, 1941), p. 66.

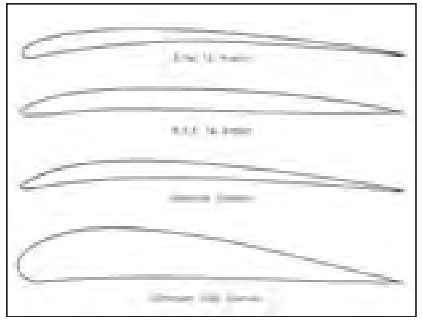


An equation derived independently by Martin E. Kutta and Nikolai Joukowsky (Zhukovsky) between 1902 and 1906 allowed the lift of an airfoil to be calculated for the first time with mathematical precision. In notes published in Russian and French journals in 1906, Joukowski (1847-1921) calculated the lift per unit span of an airfoil by using the following relationship, $L = \rho V \Gamma$, where L is lift and is Γ is the circulation of air around an airfoil. In Joukowski's derivation, the latter was a quantity equal to the line integral along a closed curve of the velocity resolved along that curve. The most important historical point to remember about this equation is that it represented a revolutionary development in theoretical aerodynamics, one that enabled the precise calculation of an airfoil's lift. National Air and Space Museum, Smithsonian Institution (SI 83-7699)

of the United States Army's most effective wing shape of the World War I era, the U.S.A. 27.

Still, bias for the thin wing persisted. Empirical test methods reinforced it. Because the small, atmospheric wind tunnels of the early years were incapable of testing at anything but low Reynolds numbers, their data showed that thin, highly cambered (or arched) wing sections had the most favorable properties. At low Reynolds numbers, airflow over thick sections "separated" early and resulted in unsatisfactory performance. Furthermore, especially in America, there was always the memory that the Wrights had achieved their historic flight in 1903 with a long, slender airfoil. Convinced that the longest span with the thinnest sections generated the greatest lift, some German propeller designers even went so far as to make their blades from mere fabric stretched by centrifugal force. Nearly all World War I aircraft, with the important exceptions of some advanced German aircraft designed by Junkers and Fokker, employed extremely thin wings requiring for external strength and rigidity a messy conglomeration of wires, struts, and cables. Junkers and Fokker tried thicker wing sections in a few of their aircraft, in part because the leading German aerodynamical laboratory at Göttinger was recommending them in confidential wartime reports.

It is hardly surprising that the world's first systematically engineered wing sections developed at Prandtl's Göttingen laboratory, the institution where advanced aerodynamic theory and premiere wind tunnel technology met really for the first time anywhere—doing so early in World War I, just in time for use by the German air force. Göttingen based its wing sections, interestingly, on a family of shapes derived mathematically by the age's foremost Russian aerodynamicist, Nikolai E. Joukowski (1847–1921), who operated a wind tunnel laboratory in Moscow. In 1910, Joukowski showed how a circle could be transformed into airfoils by a mathematical trick known as "conformal transformation." The trick allowed a bright aerodynamicist to compute the surface pressure (or pressure distribution) on an



These four airfoils—one French, one British, and two German—were typical of the types of airfoil shapes employed in wings of World War I airplanes. Laurence K. Loftin, Jr, *Quest for Performance*, Fig. 2.9, p. 24. http://www.hq.nasa.gov/office/pao/History/SP-468/ch2-2.htm

airfoil with some accuracy, from which some rather accurate lift values could be determined. In practice at Göttingen, the transformation method enabled the design of a series of "reasonable-looking airfoils" that were to be tested in wind tunnels and in flight, with researchers noting differences in performance brought on by the slightest variation of a wing parameter.[3] Iterating the Joukowski family of airfoils into effective, streamlined wing shapes proved extremely slow and laborious because of the sophisticated mathematics it involved, but the rigorous thinking and methods of Prandtl and

The most advanced feature of the Fokker D-VII airplane of World War I was its internally braced cantilever wings involving very thick airfoil sections. Many of the fine characteristics of this biplane, one of the most renowned of Germany's fighters, were due to its wing thickness, which was unusual for its day. Fokker built the D-VII wing around a Göttingen 418 airfoil. Laurence K. Loftin, Jr, *Quest for Performance*, p. 34. http://www.hq.nasa.gov/office/pao/History/SP-468/ch2-2.htm

his cohorts at Göttingen created some useful wing shapes. Generally of a thicker shape and featuring large-radius leading edges and very thin trailing edges, the very best Göttingen airfoils, notably Göttingen 387 and 398, proved very effective in flight. They were to be used on airplanes for many years to come, not only in Europe but in the U.S. as well. The famous Clark Y airfoil, designed by Colonel Virginius E.

[3] Anderson, *A History of Aerodynamics*, p. 289.

Airfoil research took many different forms. At NACA's Langley Memorial Aeronautical Laboratory in 1921, the performance of a model wing was tested by being suspended beneath a Curtiss JN-4 "Jenny" aircraft, which employed a Royal Air Force 15-airfoil section. NASA Image #L-00130 (LaRC)

Clark of the U.S. Army in 1922 and employed on a number of noteworthy American aircraft of the 1920s, including the Ryan monoplane that flew Lindbergh across the Atlantic in 1927, was in fact a design offshoot of the Göttingen family.

In the United States, one of the first things that the National Advisory Committee for Aeronautics did when it came to life in 1915 was focus American attention on wings. In its first *Annual Report* to Congress, the NACA called for "the evaluation of more efficient wings of practical form, embodying suitable dimensions for an economical structure, with moderate travel of the center of pressure and still affording a large angle of attack combined with efficient action." The Committee could not carry out this work itself, because Langley Memorial Aeronautical Laboratory was at that time still no more than a dream. The best the NACA could do toward improving wing design was to support wind tunnel tests at the Massachusetts Institute of Technology (MIT), which were under the auspices of the airplane engineering department of the Bureau of Aircraft Production. It was this experimental program that resulted, in 1918, in the introduction of the U.S.A. family, the largest single group of related airfoils developed in America up to that time.

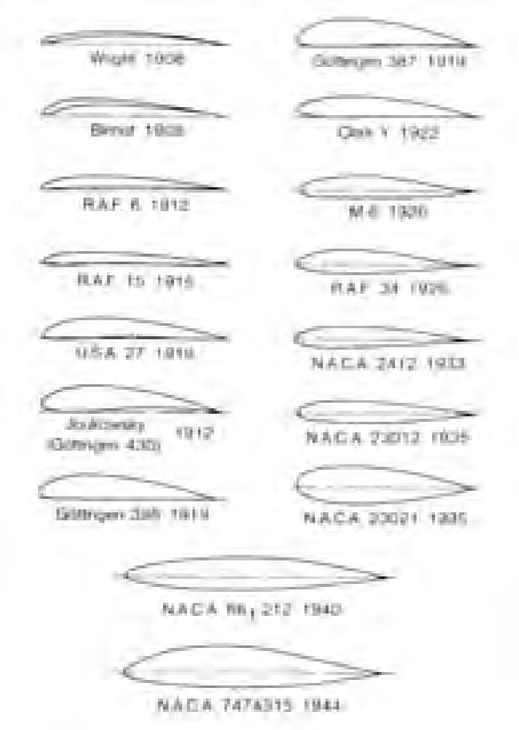
This chapter's first document is the public announcement of the initial six airfoils in the U.S.A. series. Entitled "Aerofoils and Aerofoil Structural Combinations," by Lt. Col. Edgar S. Gorrell and Maj. H.S. Martin of the U.S. Army Signals Corps, the NACA published this report as *Technical Report (TR) 18* in 1918. Also in 1918, sponsored wind tunnel experiments at MIT looked specifically into the suitability of utilizing thick U.S.A. wing sections in the design of internally braced monoplanes. Unfortunately, there was still no effective theory or sufficiently capable wind tunnels to help design them. But at least a long and dedicated American search for truly effective airfoils for different advanced airplane requirements had begun.

At the end of the war, the NACA supplemented its support of the MIT wind tunnel program with a laborious effort by its small technical staff in Washington to bring together the results of airfoil investigations at all the European laboratories but still not including Göttingen or any other lab in Germany. In June 1919, the Committee opened an intelligence office in Paris to collect, exchange, translate,

Workmen in the patternmakers' shop at NACA Langley manufacture a wing skeleton for a Thomas-Morse MB-3 airplane for pressure distribution studies in flight, June 1922. The MB-3's wing employed a Royal Air Force 15 airfoil. NASA Image #L-00184 (LaRC)

and abstract reports as well as miscellaneous technical and scientific information related to aeronautics. One of the early fruits of this labor was the NACA *TR 93*, published in 1920. Too technical in nature to include in our collection, *TR 93* provided a comprehensive and handy digest of standardized test information about all the different airfoils employed by the Allied powers. The report offered graphic illustrations of the detailed shapes and performance characteristics of more than 200 airfoils, as well as four index charts that classified the wings according to aerodynamic and structural properties. The intention was to make it easier for an American designer to pick out a wing section suited to the particular flying machine on which he was working. In retrospect, it is plain that many of the plots were totally unreasonable—no doubt because the NACA personnel who interpreted the collected data, like those who made the original tests, did not really understand how and why certain shapes influenced section characteristics as they did. Despite the flaws, however, the effort that went into the preparation of this report and others like it mobilized the NACA staff to manage a solid program of airfoil experiments once research facilities were ready at Langley laboratory.

When NACA Langley began routine operation in June 1920, the empirical approach was still the most sensible way to better wings. Wing section theory as developed before World War I by Joukowski, Prandtl, and others (including Martin W. Kutta, who, along with Joukowski, was responsible for inventing the circulation theory of lift, a "revolutionary development in theoretical aerodynamics"[4]), permitted the rough determination of lift-curve slopes and pitching moments, but little else. It was possible to transform from the pressure distribution around a circle, which was known theoretically, to the flow distribution measured around an airfoil, and thus create an approximate airfoil shape, but the mathematics required for the transformation was too abstruse for the average engineer, perhaps especially in America. Further, there was no way to measure the practical value of the mathematical formulations other than via systematic wind tunnel testing. Prandtl had refined the Kutta-Joukowski method, but his refinement still allowed only for the rough calculation of wing section characteristics.



The historical evolution of airfoil sections, 1908-1944. The last two shapes are NACA low-drag sections designed to have laminar flow over 60 to 70 percent of chord on both the upper and lower surface. NASA Image #L-1990-04334 (LaRC)

Some of the most popular airfoils of the 1920s would thus still be produced by highly intuitive methods: cut-and-try procedures based neither on theory nor on systematic experimentation. For the wing section of his successful seaplane, Grover Loening took the top curvature of the RAF 15 wing section and, for the underside, drew a streamlined curve with a reverse in the center, which enclosed the spars. The net result of this artistic rendition was so good that Loening, who did not want other people to copy his product, decided not to submit it for tests anywhere. Col. Virginius Clark, U.S.A., designed one of the 1920s' most popular airfoils for wings, the Clark Y, simply by deploying the thickness distribution of a Göttingen airfoil above a flat undersurface; he chose the feature only because it was highly desirable as a reference surface for applying the protractor in the manufacture and maintenance of propellers.[5]

[4] Anderson, p. 250.

[5] See Grover Loening, *Our Wings Grow Faster* (New York, 1935).

Aviation magazine published a brief but illuminating interview with Virginius Clark in its 17 October 1927 issue. Although it appears in our documentary collection slightly ahead of its time (that is, out of chronological order), we have placed it early in the chapter as a representation of the intuitive stage of airfoil design that predominated in the U.S. into the late 1920s.

Clark certainly enjoyed great success with his airfoils, but the days of such trust in personal intuition were numbered. The cut-and-try method, although successful in the hands of a few talented practitioners like Clark and Loening, had too spotty a success record to continue forever. More and more aeronautical engineers from the 1920s on realized that a wide range of effective airfoils would be created only by using some more systematic analytical method involving tests in a significant and reliable wind tunnel.

The scene of the most important airfoil research conducted anywhere in the world shifted in the early 1920s from Göttingen to NACA Langley. This change happened when Prandtl's brilliant young protégé Dr. Max Munk emigrated to the U.S., bringing not only his idea for a variable-density tunnel with him (see chapter 2) and his extraordinary theoretical abilities but also his experience in developing Göttingen airfoils.

Munk's variable-density tunnel (VDT), which began operation at Langley in October 1922, made possible a huge advance in the experimental method by which to understand airfoils. Airfoil expert Ira H. Abbott, an aeronautical engineer who went to work at Langley immediately after graduating from MIT in 1929 (and who remained with the government research agency until his retirement from NASA in the early 1970s), described the VDT's significance for airfoil development in a 1980 historical retrospective on the evolution of aircraft wing design:

> All previous wind tunnel research on airfoils had been severely handicapped by the small scale of the tests, measured by Reynolds number. Experience had shown that the results of tests of small models could not be applied directly to full-scale flight conditions, and neither theory nor experience provided any means for correcting the results. Small-scale models were, however, essential to research, both because the small wind tunnels then existing would not accommodate large models, and because big models for the large wind tunnels later to be built were too expensive and cumbersome for extensive research.

As Abbott emphasized, the VDT "avoided this problem" by obtaining full-scale results from tests of small models (usually 5 by 30 inches), at 20 atmospheres pressure.[6]

[6] Ira H. Abbott, "Airfoils: Significance and Early Development," in *The Evolution of Aircraft Wing Design* (American Institute of Aeronautics and Astronautics/Dayton-Cincinnati Section, 1980), p. 22. Abbott presented this paper at an AIAA meeting held at the Air Force Museum in Dayton, Ohio, in March 1980.

With Munk supervising the work from his NACA office in Washington, Langley began its first systematic investigation of a series of wing sections in the VDT early in 1923. Although the research was to be essentially empirical, the idea behind the design of the airfoils derived from a highly intuitive theory conceived by Munk as an outgrowth of his Göttingen experience and his further thinking about the design application of Joukowski's mathematical trick of conformal mapping. In his "General Theory of Thin Wing Sections," published by the NACA in 1922, Munk expressed a more helpful engineering approach to the theoretical prediction of airfoil lift and moments when the airflow in which it operated was considered, for calculation purposes, nonviscous.[7]

The actual airflows in which wings fly, of course, do experience viscosity (an internal fluid friction) that definitely affects aerodynamic performance; for theoretical purposes, however, given the tremendous complexities of dealing with viscous flows and skin friction, aerodynamicists conceived of a perfect fluid, one that has no viscosity. "In such a perfect, nonviscous fluid," as professor of aeronautical engineering Alexander Klemin of New York University's Daniel Guggenheim School of Aeronautics wrote in a textbook published in 1930, "all bodies are perfectly streamlined and experience no resistance to motion." The value of the conception of a perfect fluid for Munk and other aerodynamicists of the time was that "the flow in such a fluid is easily calculated for simple bodies, and at least approximates the flow in an imperfect fluid such as air or water."[8] Results from such theory were not perfect themselves; for one thing, there were scale effects associated with viscosity that "lent continuing uncertainty to the applicability to full-scale airplanes of model results from wind tunnel tests."[9] Still, approximate nonviscous theory such as Munk's thin-airfoil theory did offer significant help where otherwise none could have been achieved.

Convinced that contemporary aerodynamicists would fail to produce significantly improved airfoils if they continued to let the wing section be dictated by the mathematical method, Munk decided to "start with a wing section, any technically valuable wing section, and fit the mathematics to the section." Even though the method required some simplifying assumptions and did not permit the calculation of maximum-lift coefficients, Munk's idea was still a major breakthrough, if not a watershed in the history of airfoil design.[10] By replacing the airfoil section with an infinitely thin curved line, it permitted the calculation of certain airfoil characteristics (e.g., lift-curve slope, pitching moments, and chord-wise distribution) directly in terms of easily identified parameters of the shape. A complete reproduction of Munk's *TR 142* of 1922 appears in this chapter's collection of documents.

[7] Anderson, p. 290.

[8] Alexander Klemin, *Simplified Aerodynamics* (Chicago: Goodheart-Willcox Co., Inc., 1930), p.200.

[9] Walter G. Vincenti, *What Engineers Know and How They Know It: Analytical Studies from Aeronautical History* (Baltimore and London: The Johns Hopkins University Press, 1990), p. 35.

[10] For expert technical discussion of the importance of Munk's thin-airfoil theory in the history of aerodynamics, see Theodore Von Karman, *Aerodynamics: Selected Topics in the Light of the Historical Development* (Ithaca, New York, 1954), pp. 9-10, and Ira H. Abbott and Albert E. Von Doenhoff, *Wing Section Theory* (New York, 1959).

TR 142 implied, specifically, that characteristics such as the angle of zero lift and moment coefficient were determined primarily by the shape of the "mean" or "camber line" of the airfoil, the line halfway between the upper and lower surfaces. But in the report, Munk also extended his method to suggest what would happen if the camber, thickness, and thickness distribution of a wing section were varied independently, according to a method of systematic parameter variation. Munk's analysis suggested that a design having a slight upward camber near the trailing edge would result in a stable center of pressure travel. So, starting with a mean line pulled out analytically from one of the better Göttingen airfoils, the VDT research team wrapped a thickness form about the upper and lower surfaces of an airfoil. Then, by pulling the mean line (or camber) out, going to a symmetrical section, and changing all the ordinates to correspond to the correct proportion of thickness, it prescribed a family of 27 related airfoils. The NACA announced this airfoil family to the world in *TR 221*, published in 1925, also found in our documentary collection; NACA named the airfoils "M sections" in Munk's honor.

Ironically, Munk was forced to leave the NACA for good in early 1927, a result of a bitter dispute over his autocratic style of supervision that had led to the mass resignation of all the aerodynamic section heads. Fortunately, his approach to airfoil development stayed happily and securely in place. In fact, it climaxed in the 1930s in work at NACA Langley that would be directed by one of the other most brilliant and controversial characters in American aerodynamics (and certainly NACA) history, Eastman N. Jacobs. A 1924 graduate in mechanical engineering from the University of California at Berkeley, Jacobs reported to work at Langley in 1925 and remained there until his own, even more mysterious, departure (and complete disappearance from the aeronautical scene) in 1944. Put in charge of the VDT soon after Munk's resignation as chief of aerodynamics at Langley, Jacobs built upon the excellent work that Munk had started and took it to a significantly higher level of technological achievement. Under Jacobs's direction, the NACA developed and standardized a complete system of mathematically constructed airfoils, based on the results of actual wind tunnel testing and flight research. These airfoil families proved instantly and hugely successful not just in the U.S. but around the world. Instead of taking the time and bearing the great expense and time of conducting their own airfoils, most aircraft manufacturers relied entirely on the published characteristics of the NACA-derived wing sections. From NACA reports and their voluminous catalogs of shapes and pertinent data, the industry picked airfoils that became the wings for some of the best aircraft of their era, including the Douglas DC-3 transport and the B-17 Flying Fortress, as well as a number of postwar general aviation aircraft.

Some interesting insight into Jacobs's early plans for what would become the famous NACA 4-digit series of airfoils can be gleaned from a sequence of correspondence reproduced in this chapter from the year 1929. In essence, this chain of

Eastman N. Jacobs (1902-1987), head of the Variable Density Wind Tunnel (VDT) section at NACA Langley, played a critical role in the delineation of airfoil shapes in interwar America. The VDT research team at Langley in 1929. Eastman Jacobs is sitting (far left) at the control panel. NASA Image #L-3310 (LaRC)

Eastman Jacobs was adventurous in many ways besides aeronautical research. In the late 1920s, he bought a Pitcairn Mail Wing-type airplane, a small open-cockpit biplane with a 110-hp engine. In 1933, he flew back across Hampton Roads from nearby Norfolk, Virginia, without realizing he was in the eye of a hurricane. NASA Image #A76-1405 (Ames)

correspondence suggests something that has never been placed into the historical record—that a basic idea leading to the 4-digit series came from outside NACA, from a young and then little-known engineer, Ralph H. Upson, then working for the Aeromarine Klemm Corporation in Keyport, New Jersey. (Upson later became chief engineer for Henry John Heinz, grandson of the founder of the Heinz pickle and catsup company in Pittsburgh, when Heinz sought to build cargo- and troop-carrying gliders for air service in World War II. After the war, he worked at Boeing and taught aeronautical engineering at the University of Minnesota.) In no NACA history or history of aerodynamics has Upson ever been credited with any sort of role in the genesis of the 4-digit airfoils. As the documentation shows, however, he urged the NACA as early as 1928 to focus its airfoil research on the effects of varying section thickness and mean camber line, the fundamental concept behind the 4-digit series.

Whether Jacobs and his colleagues at Langley would have focused on the effects of thickness and mean camber line as the fundamental properties for further investigation without Upson's input is uncertain. What is definite is that the Jacobs team quickly determined in 1929 that their research should concentrate on the effects of varying those parameters and to move on to the selection of an initial thickness distribution and shape of camber from which to iterate a family of test airfoils. Ira Abbott, who joined Jacobs's VDT team in June 1929, many years later recalled only the technical details of how this happened. Robert Pinkerton, another VDT section member, found that the thickness distributions of efficient airfoils such as the Göttingen 398 and the Clark Y were nearly the same when the camber was removed and they were reduced to the same thickness. Jacobs accordingly selected a mathematically defined thickness distribution, which corresponded closely to that for such airfoils. The mean lines to be used were selected as those defined by two parabolic arcs tangent at the position of maximum mean line ordinate. These cambers appeared suitable for positions of the maximum ordinate varying between 20 and 70 percent of the chord.[11] Having selected these key variables, Jacobs and his fellow engineers were ready to start the research—almost ready, that was, given that their wind tunnel had gone haywire.

As soon as planning began for the test program that was to become the NACA 4-digit series, the precious piece of experimental equipment in which the tests were to take place, the VDT, started misbehaving badly. Redesigned and rebuilt with a closed-throat section, variable-speed drive system, and new balance following a catastrophic fire and explosion caused by a broken light bulb, which scorched its insides in August 1927, the new VDT simply did not work right. Its airstream, which needed to be steady, constant, and uniform to be of much help, was atrocious, and all attempts to smooth out the turbulence failed. There was no alternative but to rebuild the tunnel once again, a remedy that "horrified and exasperated" NACA

[11] Abbott, "Airfoils," p. 22

officials, as there was little if any chance of an additional appropriation in the wake of the start of the Great Depression.[12] Because no special funds from Washington were available, the engineers at Langley had no alternative but to scavenge parts and materials, scrape up what little money they could from the lab's existing budget, and rebuild the machine themselves. It took about a year, but by late 1930, the tunnel was running again—although its turbulence problems were never totally solved. As early as 1932, Jacobs was telling his superiors that the rehabilitated VDT would never be a fully acceptable facility. What was needed, he said, was a new low-turbulence tunnel.

The next few documents in the chapter feature the outstanding results of the aerodynamic research that nevertheless ensued from the existing VDT: the NACA 4-digit series of airfoils. The first is a reproduction of *TR 460*, "The Characteristics

In the course of developing its second family of airfoils in the late 1920s and early 1930s, NACA devised a numerical code—patterned after that used to identify the composition of steel alloys—by which to describe the physical shapes. Until that time, researchers in the United States and abroad all designated airfoils simply by numbering them in the sequence in which they had been tested (M-1, M-2, M-3, and so on). In the new system, however, numbers would indicate the airfoil's critical geometrical properties. This digital code did not signify much to the man on the street, but to aeronautical engineers, it suggested everything important about an airfoil. What the integers meant in the case of NACA's 4-, 5-, and 6-digit series of airfoils is expressed below:

4	4	12
Maximum Camber	Position of Maximum camber in 1/10 of chord	Maximum thickness in percentage of chord

A popular airfoil from the NACA 4-digit series: N.A.C.A. 4412.

2	30	12
Approximate maximum camber in % of chord	Position of maximum camber in 2/100 of chord	Maximum thickness in % of chord

After the 4-digit sections came the 5-digit sections. These sections had the same thickness distribution but used a camber line with more curvature near the nose. One of the most effective in the 5-digit series was N.A.C.A. 23012, the major geometric characteristics of which were built into its numerical code.

6	3	2	2	12
Six-series	Location of minimum center of pressure in 1/10 of chord	Half width of low drag bucket in 1/10 of lift coefficient	Ideal lift coefficient in tenths	Maximum thickness in % of chord

The NACA's 6-series airfoils departed from the more simply designed 5-digit family in that they were generated from a more or less prescribed pressure distribution and were meant to achieve laminar flow. Below is an expression of the digital code built into the N.A.C.A. 63,2212 airfoil. After the six-series sections, airfoil design became much more specialized for particular applications.

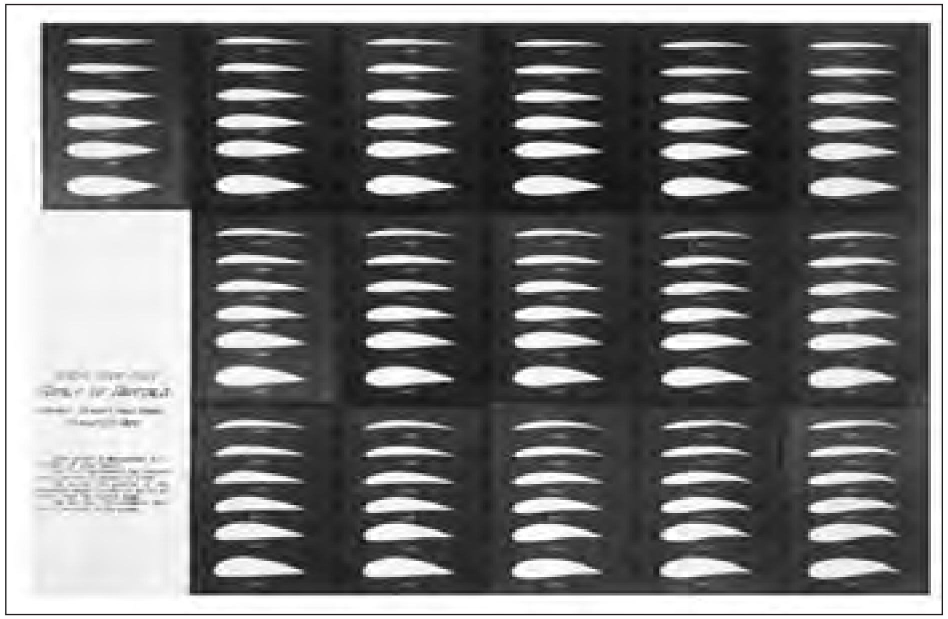
[12] Abbott, "Airfoils," p. 22.

of 78 Related Airfoil Sections from Tests in the Variable-Density Wind Tunnel," by Eastman N. Jacobs, Kenneth E. Ward, and Robert M. Pinkerton. Immediately upon publication by the NACA in 1933, the report became a classic, an airplane designer's bible.

The NACA 4-digit series introduced by *TR 460* were instantly successful not only because of the systematic presentation of data and ingeniously simply numerical code for identifying an airfoil's geometrical properties; several of the airfoils were also highly efficient wing shapes. As the reader will read in the final section of the report, test results indicated, most significantly, that "the maximum lift increases with increased camber, the increase being more rapid as the camber moves forward or back of approximately the mid-chord position."

The chapter's next five documents appear in a string and bear testimony to the nearly universal acclaim given to the 4-digit series.

The NACA did not stop at four digits. Under Jacobs's direction, the Langley engineers expanded their research in the mid-1930s to look at the effects of still other variations of wing section characteristics, particularly at what happened when the position of maximum curvature was moved forward and backward. Moving maximum curvature to the rear of the section proved ineffective due to large pitching moments, but some airfoils with their camber quite far forward performed in very promising ways. In 1935, the NACA published *TR 537*, "Tests in the Variable-Density Wind Tunnel of Related Airfoils Having the Maximum Camber Unusually



By the end of the summer of 1929, tests in NACA Langley's VDT had produced the family of airfoils N.A.C.A. 0006 through N.A.C.A. 6721, shown here in cross section. Data on all of these airfoils were presented in NACA 1933 *Technical Report No. 460*, "The Characteristics of 78 Related Airfoil Sections from Tests in the Variable-Density Wind Tunnel," by Eastman N. Jacobs, Kenneth E. Ward, and Robert M. Pinkerton. NASA Image #L-05344 (LaRC)

In 1937, NACA's Eastman Jacobs received the Sylvanus Albert Reed Award for his contributions to the aerodynamic improvement of airfoils. NASA Image #L-43999 (LaRC)

Far Forward," by Jacobs and Pinkerton. This report asserted that some of new sections, notably N.A.C.A. 23012, were "markedly superior to well-known and commonly used sections and should replace them in applications requiring a slightly cambered section of moderate thickness, having a small pitching-moment coefficient." Some aircraft designers worried about airfoils with maximum camber so far forward; their experience in full-flight suggested that airfoils of this type might be inclined to a sharp break in lift at the stall. But further NACA research alleviated this doubt, and by the late 1930s, the better sections in this new airfoil series had become extremely popular and widely used, notably a handful of sections in the 23000 series.

On 8 April 1937, shortly after seeing an updated report on the forward-camber airfoils (*TR 610)*, Charles H. Chatfield, head of the research division of United Aircraft Corporation in East Hartford, Connecticut, wrote to the NACA's George Lewis with a positive response to the new 5-digit series. Declaring that "these new airfoils are good structurally as well as aerodynamically," Chatfield went on to recommend another publication focusing just on the better of the airfoils, "so that a designer may have in one publication all the airfoils that he would be likely to consider seriously for any particular airplane." Chatfield's letter and Lewis's response to it are included in the chapter's documents.

In December 1937, the Institute of the Aeronautical Sciences selected Eastman Jacobs as the recipient of its prestigious Sylvanus Albert Reed Award for his contribution to the aerodynamic improvement of airfoils. Included in this chapter's documentary collection is George Lewis's nomination letter to the Institute of Aeronautical Sciences, along with a memorandum from Langley engineer R.C. Platt, who supplied Lewis with up-to-date information about the military and commercial aircraft employing NACA wing sections.

Ironically, not everything had actually been so rosy all the while with NACA airfoil research. Scale effects and unusually high turbulence in the VDT airstream had continued to plague its data throughout the mid-1930s, and no matter what Jacobs and his colleagues did to alleviate them, the problems would not go away. The first really shocking demonstration of how bad the VDT data were turning

out to be came as early as 1932 when Langley's new Full-Scale Tunnel (see chapter 2) first got around to making some wing tests. As Ira Abbott of the VDT section recalled, "These tests showed that the VDT results indicated a rate of increase of drag with thickness ratio much greater than the Full-Scale Tunnel. This discrepancy was important because it directly affected the choice of wing thickness for the inner sections of monoplane wings."[13] As more and more tests in the full-scale tunnel (FST) showed good agreement with results obtained in flight, some of the prouder and less circumspect proponents of the FST began to say that results from the VDT bore little relation to what really happened in flight and that correct airfoil data could only be obtained from tests on full-scale wings in the FST. VDT defenders, although fully aware by this time of their facility's inherent defects, answered the charges of their peers by asserting that their machine was still the NACA's best cheap means of obtaining a wide range of comparative data on a multitude of related airfoils. FST test specifications called for aircraft and aircraft models that were simply too cumbersome and expensive, they argued, to permit the kind of systematic, scale-model programs accomplished in the VDT.

Other criticism came from outside of the NACA, as can be seen in Langley's response on 8 February 1933 to an article by Theodore von Kármán, entitled "A Few Present Problems in Aerodynamics," in which the renowned Caltech aerodynamicist questioned the validity of VDT airfoil data. Although the memorandum enclosed was signed by Henry J.E. Reid, Langley's engineer in charge, the substance of the memo repeated Jacobs's very strong, and negative, reaction to von Kármán's criticisms.

Jacobs answered the more general challenge to the value of VDT tests by introducing the concept of "effective Reynolds number." This was essentially a stopgap method of predicting the aerodynamic effect that could have been obtained if the VDT had zero turbulence. Jacobs mentioned the concept several times in *TR 610* but actually defined it first in *TR 530*, "Characteristics of the N.A.C.A. 23012 Airfoil from Tests in the Full-Scale and Variable-Density Tunnel," published in 1935. "In a wind tunnel having turbulence," Jacobs wrote, "the flow that is observed at a given Reynolds number…corresponds to the flow that would be observed in a turbulence-free stream at a higher value of the Reynolds number. The observed coefficients and scale effects likewise correspond more nearly to a higher value of the Reynolds number in free air than to the actual test Reynolds number in the free stream." Jacobs then suggested that the name *effective Reynolds number* should be used to refer to "this higher value of the Reynolds number at which corresponding flows would be observed in free air." All applications of wind tunnel and comparisons of data, whether between the VDT and FST or any other two tunnels, should be made at the adjusted value of the Reynolds number.

[13] Abbott, "Airfoils," p. 23.

Although "probably better than nothing," Jacobs's concept rested, in the words of his own colleagues, "on an inadequate theoretical foundation and on slender correlations" of VDT results with results from other tunnels.[14] Jacobs figured the effective Reynolds number by multiplying the test Reynolds number by the tunnel's "turbulence factor," another NACA invention. For the VDT, the turbulence factor was 2.6, the highest of all Langley tunnels. Despite its limitations, all NACA wind tunnel sections started using the concept of effective Reynolds number, in particular to show the effects of Reynolds number on maximum lift. Some way to compensate for tunnel turbulence was better than no way at all.

The factor of scale and the corrupting effects of turbulence on aerodynamic measurement stimulated thinking in the aeronautical community worldwide in the 1930s, but certainly no group gave it more thought than Jacobs and his cohorts at Langley, given that it was their pioneering research taking the heat. Close scrutiny of their work sparked the ingenuity of the VDT team in important ways. For example, they began to look more carefully at basic flow phenomena that might be the source of the consistent errors in their results. In particular, they examined the "boundary layer," the thin stratum of air very close to the surface of a moving airfoil in which the impact pressure (i.e., the reaction of the atmosphere to the moving airfoil) was reduced because of the air's viscosity. In this layer, which was separated from the contour of the airfoil by only a few thousands of an inch, the air particles changed from a smooth "laminar flow" near the leading edge to a more turbulent flow toward the rear of the airfoil. To visualize the nature of the airflow around airfoils and other objects, the Langley group constructed a small low-turbulence smoke tunnel next to the other equipment in the VDT building.. Photographs of the smoke flowing around test models facilitated study of the boundary layer's conditions as they changed from low-friction laminar flow to high-friction turbulent flow. The NACA engineers accelerated their pursuit of a means to remove air from the boundary layer through slots or holes in the wing surface. This effort dated back to 1926 and was intended to decrease drag and increase lift by postponing "transition" from laminar to turbulent flow. Work in the smoke tunnel eventually led them to the conclusion that two of the critical factors causing transition, and thus high skin-friction drag, were surface roughness (the rivet heads, corrugations, and surface discontinuities then common in manufactured airplane wings) and pressure distribution on the wing surface.

It was not just airfoil experts working with Jacobs in the VDT who promoted boundary-layer research. On 18 April 1932, a young engineer in Langley's small Physical Research Division, Hugh B. Freeman, sent a memo to the engineer-in-charge in which he declared that the field of boundary-layer control offered "greater possibilities for the improvement of aircraft performance and safety than any other."

[14] Abbott, "Airfoils," p. 23.

Freeman's three-page memo, included in this chapter's documents, concluded with an outline of the advantages of conducting the boundary-layer tests at full scale in the Propeller Research Tunnel rather than on scale models in the VDT. Although Freeman's tests faced some insurmountable problems and led to no breakthroughs, they represented a major new line of aerodynamic inquiry that the NACA would pursue vigorously in the years to come, not originally so much through laminar flow control (later termed *LFC*) by blowing or suction applied to the entire wing, as Freeman suggested, but by designing for natural laminar flow from wing section shape alone. The latter became Eastman Jacobs's greatest ambition, and for roughly the next 10 years, he pursued it with crusading zeal, like a knight in search of the Holy Grail.

His quest for laminar-flow airfoils began with a campaign to build a new and larger variable-density tunnel with a significantly smoother airstream, for it would be impossible to design them without one. As noted earlier, Jacobs had started to call for a new low-turbulence tunnel as early as 1932, but he did not get the ball rolling until an April 1935 memo to Langley's engineer in charge (a document not included in our collection) in which he reported the results of a staff conference on ways to increase the speed of airplanes. A low-turbulence pressure tunnel, he urged, would greatly enhance the two related lines of research that the VDT team had long been pursuing: development of new airfoils and better understanding of the basic aerodynamic relationship between airstream turbulence, boundary-layer flow, and wing performance. Although Jacobs believed that the existing VDT could still provide useful design data, he warned that the "airstream necessary for the continued investigation of the fundamental characteristics of large scale airflows cannot be obtained in the existing tunnel." Turbulence in the old tunnel did not completely invalidate results for airfoils like those of the 4- and 5-digit classes, but accurate experiments with airfoils and other bodies that might enjoy low-friction laminar flow could not be expected in the existing facility.[15]

Within two weeks after receiving a copy of Jacobs's proposal for comment, two of Langley's most influential division chiefs sent memos to their superiors elaborating their reasons why the NACA should reject Jacobs's idea. Smith J. DeFrance, head of the Full-Scale Tunnel, questioned whether the knowledge to be gained from the new equipment would warrant the expenditure of money.[16] But it was Dr. Theodore Theodorsen, the brilliant head of the small Physical Research Division, who expressed the most vociferous, historically significant, but ultimately incorrect objections to the facility Jacobs had in mind: "I think the variable-density tunnels

[15] Eastman Jacobs to Engineer-in-Charge, "New Variable-Density Tunnel," 26 Apr. 1935, Langley Correspondence Files, Code A206-1, in NASA Record Group 255, National Archives, Mid-Atlantic Region, Philadelphia, Pa.

[16] Smith J. DeFrance to Chief, Aerodynamics Division, "Mr. Jacobs' memorandum on Proposed New Variable-Density Tunnel," 4 May 1935, A206-1, RG 255, National Archives, Philadelphia.

have outlived themselves. I do not think that the variable-density tunnel has led to any fundamental discoveries. They contain a very large amount of turbulence in the airstream, a condition that cannot be avoided." "What is a new variable-density tunnel to be used for?" Theodorsen asked. "Several years will be required to investigate the tunnel, and then what?" There was "no more need for airfoil testing," the physicist boldly declared, except possibly in connection with some questions about flow conditions in the boundary layer better answered by theoreticians.[17]

Dr. Theodore Theodorsen, a Norwegian immigrant who came to work for the NACA in 1929, contributed at least as much to aerodynamics as did Eastman Jacobs. His long list of accomplishments included an improved thin-airfoil theory, a theory of arbitrary wing sections, a basic theory of flutter, and numerous improvements made to the form of engine cowlings and propellers. NASA Image #L-47543-B (LaRC)

While Jacobs and his VDT staff had been developing the 4- and 5-digit airfoils using a systematic experimental approach, Theodorsen had been tackling various airfoil problems from the theoretical angle. Although perhaps his greatest contribution during this period was his theory of oscillating airfoils with hinged flaps, related closely to the problem of flutter, Theodorsen also provided some very arresting insights into the relationship between pressure distribution and boundary-layer flow, and hence, on wing section characteristics.

In NACA *TR 41,* published in 1931, Theodorsen described a "Theory of Wing Sections of Arbitrary Shape," which made it possible, as long as the airflow did not separate from the airfoil, to predict the pressure distribution of an airfoil. Starting with an arbitrary airfoil, one changed the closed two-dimensional shape through a conformal transformation almost into a circle; then, by using a rapidly converging series, one transformed the bumpy circle into a true circle about which the flow was known. Although no one at the time thought it was reasonable to apply this theory for the purpose of a practical design, the knowledge of the pressure distribution made possible by this clever double transformation later suggested the answer to the riddle of how to shape a laminar-flow airfoil.

The introductory section of *TR 411* is reproduced in this chapter's documents. In it, Theodorsen suggested that Langley's airfoil research had reached an experimental impasse. It could only move ahead with the help of some new theory, which he hoped he had provided in his report.

[17] Dr. Theodore Theodorsen to Engineer-in-Charge, "Comments on Mr. Jacobs' Memorandum Regarding New Variable-Density Wind Tunnel," 4 May 1935, A206-1, RG 255, National Archives, Philadelphia.

It is difficult to pinpoint just when Eastman Jacobs first considered controlling the boundary layer through body shape—or, more accurately, through control of the pressures acting along the body surface. The idea seems to have been germinating in Jacobs's mind at least as early as 1929. In a memorandum on airfoil scale effects dated 13 November 1929 (included in our collection), Jacobs discussed the importance of the relationship between transition and airfoil drag and mentioned the dependence of the transition point on airfoil shape. At the time, he expected that "the possible large drag reductions through prolonging of laminar boundary layers" (i.e., through prolonging transition to turbulent flow) would become apparent "as the result of the systematic tests of airfoil shapes." By 1935, however, he definitely knew this process of discovery could not happen without new turbulence-free testing equipment.

The equipment was not immediately forthcoming. The NACA turned down Jacobs's request on two grounds. First, other important projects, including the construction of an expensive new tunnel for high-speed propeller research (eventually built at Langley as the 19-Foot Pressure Tunnel but which was not really used much for propeller research) were awaiting funding. Second, Congress would not understand the desirability of low turbulence in wind tunnels. It would take two more years of persuasion on Jacobs's part before the low-turbulence tunnel would be authorized, and, as will be seen, even then the NACA only got its way by calling the tunnel something else.

In late 1935, Jacobs returned to Langley after representing the NACA in Rome at the Fifth Volta Congress on High-Speed Aeronautics. Now more than ever, he was convinced that Langley had to have a low-turbulence pressure tunnel. During his trip, he had visited most of the larger aeronautical research laboratories on the continent, whenever possible examining new experimental facilities and discussing current work. He found the European nations to be in keen competition, spending "large sums of money building up their research establishments." While concluding that America's "present leading position" in aeronautical research and development was "not seriously menaced at this time," Jacobs warned that "we certainly cannot keep it long if we rest on our laurels" and fail to modernize our test equipment. At the end of his trip report (which is not included in our documents), the Langley engineer returned to an old theme: "It is again urged that modern variable-density tunnel equipment be built in this country capable of testing at full dynamic scale for modern aircraft."[18]

Jacobs also brought back some new insight into the nature of the boundary layer. While in England, he had spent a weekend at the home of Sir Geoffrey I.

[18] Jacobs to Engineer-in-Charge, "Trip to Europe," 11 Nov. 1935, E32-12, RG 255, National Archives, Philadelphia.

Taylor, professor of physics at Cambridge University, who had presented a paper on high-speed flow at the Volta Congress. In long private conversations, Taylor described for Jacobs the substance of his recent work in the statistical theory of turbulence. This theory seemed to indicate "the transition from laminar to turbulent flow was due to local separation caused by the pressure field."[19] By implication, this result said that transition could possibly be delayed or perhaps avoided by preventing laminar separation (i.e., by using a falling pressure gradient). As it turned out, this would, in fact, become the mechanism eventually used by Jacobs in his design of laminar-flow airfoils.

Jacobs also had the chance at Cambridge to talk at length with Professor B. Melville Jones (1887-1975), England's leading aerodynamicist (later knighted). Jacobs had followed Jones's work for years very closely. Jones's 1929 article on "The Streamline Airplane" (reproduced in chapter 3) had, in fact, stimulated Jacobs's early interest in laminar-flow airfoils by showing how perfect streamlining would eliminate pressure drag caused by flow separation. If Jones's ideal streamlined airplane could be realized, and if the best aircraft of the 1930s were coming closer and closer to it, the only fundamental airfoil problem left for the aerodynamicist to solve would be reducing skin-friction drag. And every expert in the field knew from Prandtl's early boundary-layer research that skin-friction drag was significantly less in laminar flow than in turbulent flow.

During Jacobs's visit, Jones reported that recent British flight work showed considerable laminar flow over the forward regions of very smooth wings where there were favorable pressure gradients. This finding encouraged Jacobs greatly, because he knew it pointed to the possibility that drag levels achieved by well-designed advanced aircraft could soon, in fact, be down to the value of skin friction. The only remaining opportunity for reducing drag would lie in encouraging laminar flow.

Melville Jones later presented his thoughts on the subject in the First Wright Brothers' Lecture, which he presented before the Institute of the Aeronautical Sciences at Columbia University in New York City, on 17 December 1937, on the occasion of the 34th anniversary of the Wrights' first flight. The *Journal of the Aeronautical Sciences* published Jones's talk in its January 1938 issue, a complete copy of which is reproduced in this chapter.

How Jacobs's own thinking evolved after his 1935 trip to Europe, to the point of going hard after laminar-flow airfoils, is not totally clear. The most explicit document in which Jacobs personally dealt with this process was written on 27 December 1938 and was entitled "Notes on the History of the Development of the Laminar-Flow Airfoils and on the Range of Shapes Included." This is one of the essential documents to examine in this chapter, but unfortunately, it provides only a skeleton narrative of when, where, and how Jacobs got his ideas about laminar-flow airfoils.

[19] Jacobs to Engineer-in-Charge, "Trip to Europe," 11 Nov. 1935, E32-12, RG 255, National Archives, Philadelphia.

One thing is certain: Jacobs returned to the U.S. in late 1935 convinced that airfoils could be designed to maintain laminar flow "simply by shaping them to have large running lengths of decreasing pressure along the surface."[20] At a laboratory conference on boundary-layer control in July 1936, he argued that "direct control through shape should be placed first on our program" and again urged his colleagues to support his idea for the construction of suitable turbulence-free testing equipment.[21] In the fall, he wrote an article on "Laminar and Turbulent Boundary Layer as Affecting Practical Aerodynamics." As readers will see when they examine it in our documentary collection, his article was, in essence, another plea for the new tunnel.

Achieving laminar flow through proper shaping for favorable pressure distribution was "a nice idea" but far more difficult to implement than Jacobs and his colleagues imagined. As Anderson has explained, "recent advances in airfoil theory had been oriented toward calculating the pressure distribution for a given airfoil shape," but what Jacobs needed to do was "turn that theory inside out and design an airfoil for a given pressure distribution."[22] Such theoretical manipulation was not Jacobs's forte.

Still, in a brilliant piece of creative engineering science, Jacobs managed to do it. With his commitment to the design of laminar-flow airfoils now overshadowing all of his other work, Jacobs disappeared from Langley Field for a few days to unravel the mysteries of Theodorsen's 1931 airfoil theory and to explore ways of reversing its procedure, which had been designed to predict the pressure distribution from a given shape. He called over to his house a friend, Robert T. Jones, a highly intuitive NACA researcher who had taken a few classes at Catholic University taught by Max Munk. Together, Jacobs and Jones decided that Theodorsen's method could not be used in the way desired without adding to the theory. Jones proposed an extension of the theory derived from Munk's thin-airfoil work that seemed to be a way of calculating a shape that would give a desired sequence of pressures, but this idea also proved too inaccurate.[23]

When Jacobs returned to the laboratory from his short working vacation, he challenged his staff to apply Theodorsen's theory in design. H. Julian "Harvey" Allen, one of the brightest members of the VDT staff, came up with one means of inverting the theory based on a linearization that started from a thin Joukowski airfoil. Applicable only to thin sections, Allen's way proved too inaccurate near the

[20] Anderson, *A History of Aerodynamics*, p. 349.

[21] Jacobs to Chief, Aerodynamics Division, 20 July 1936, AV400-1, RG 255, National Archives, Philadelphia.

[22] Anderson, *A History of Aerodynamics*, p. 349.

[23] Robert T. Jones, "Recollections from an Earlier Period in American Aeronautics," *Annual Review of Fluid Mechanics* 9 (1977): 10-11.

leading edge for prediction of local pressure gradients.[24]

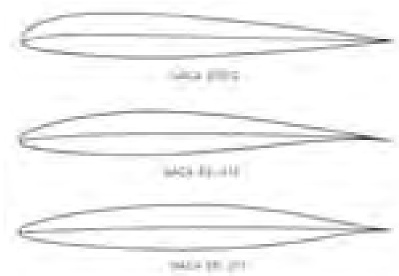
No one in the VDT section had any special training in advanced mathematics of the sort required, which prompted a few of the men to approach Theodorsen's Physical Research Division for assistance. According to Ira Abbott, another key member of Jacobs's staff: "We were told that even the statement of the problem was mathematical nonsense with the implication that it was only our ignorance that encourages us."[25] Theodorsen himself went to the trouble of showing that the shapes likely to result from an inversion of his theoretical method would be "unreal," things that looked like figure eights and surfaces that crossed over one another. Encouraged now by hearing this negative peer response, Jacobs stubbornly persisted in directing an all-out effort to devise a satisfactory inversion of the Theodorsen method.

The breakthrough came in the spring of 1938. The inversion, which Jacobs later said he modeled after Isaac Newton's clever method of approximating a square root, consisted essentially of changing a function in small increments in the conformal transformation of Theodorsen's theory. By taking an ordinary wing section like N.A.C.A. 0012 and "running it backwards," that was, designing its nose features according to the shape principles of the tail and its tail features according to the nose, Jacobs's team was able to arrive at an approximate shape that had falling pressures over most of the surface. It is impossible to document whether this single spectacular inversion ever took place; the inversion procedure may in fact have been a gradual refinement. Jacobs's role is not in dispute, however; he was the inspiration and driving force behind the entire laminar-flow program.

Something else essential to the development of laminar-flow airfoils also happened in 1938: a prototype of the low-turbulence tunnel that Jacobs had been seeking desperately for several years was finally built. Strangely, it was called "an icing tunnel." George Lewis continued to feel that NACA could not justify the expense of a new tunnel at Langley solely for the development of low turbulence. Congressmen simply would not understand the urgency. Lewis could, however, sell it on the basis of aircraft icing experiments. Many aircraft crashes traced to icing problems were attracting public attention in 1937, and the airlines were clamoring for useful information on the subject. Even if it meant stretching the truth more than a little bit, here was a way for the NACA to kill two birds with one stone. A perfunctory series of icing experiments was conducted during the summer of 1938, but immediately thereafter, the $103,000 pilot facility was converted into a low-turbulence tunnel for low-drag airfoil studies. It would be used in this way for three years, until the fall of 1941 and the eve of Pearl Harbor, when a new, more sophisticated, and even less turbulent machine, the Two-Dimensional Low-Turbulence Pressure Tunnel (cost, $611,000) was ready for operation.

[24] See H. Julian Allen, "A Simplified Method for the calculation of Airfoil Pressure Distribution," NACA *Technical Note 708*, 1939.

[25] Abbott, "Airfoils," pp. 23-24.



N.A.C.A. 23012 (top) was a conventional airfoil designed during the early 1930s. The other two airfoils, from the 6-digit series, were designed to maintain laminar flow over specified percentages of the chord. Comparing the top with the bottom two airfoils, one sees how the NACA designed "laminar-flow" sections by pushing the point of maximum thickness farther aft along the chord. The aft location was primarily due to the need to achieve a particular type of airfoil-surface pressure distribution that would enhance laminar flow. Laurence K. Loftin, Jr, *Quest for Performance*, Fig. 5.1, p. 105. http://www.hq.nasa.gov/office/pao/History/SP-468/ch5-2.htm

A June 1938 memorandum from Jacobs on the optimistic results of the preliminary round of tests with his first tentative laminar-flow model in the "ice tunnel" is included in this chapter's documentary collection. From it, he and his colleagues quickly designed a completely new family of airfoils for systematic testing: the NACA 6-series. By December of that year, he had grown so confident in the ultimate success of his work, that he composed his previously mentioned "Notes on the History of Development of the Laminar-Flow Airfoils."

In the summer of 1939, on the eve of Nazi Germany's invasion of Poland and the start of World War II, the NACA distributed two advance confidential reports prepared by Jacobs and his associates covering their new laminar-flow airfoils and their advantages over conventional airfoils.[26] The reports immediately caused a major buzz in the aircraft industry and military air services. By 1941, comments about the NACA having developed "a secret airfoil" were even popping into college aerodynamic textbooks, such as Newton H. Anderson's *Aircraft Layout and Detail Design*, a chapter of which is included in our documents.

The NACA tried to keep a lid on the laminar-flow developments but could not stop itself from at least previewing what its leaders were coming to regard as a truly major breakthrough. On the first page of its *Annual Report* for 1939, the NACA hinted: "Discovery during the past year of a new principle in airplane-wing design may prove of great importance. The transition from laminar to turbulent flow over a wing was so delayed as to reduce the profile drag, or basic air resistance, by approximately two-thirds." Though admitting that it was still too early to appraise this achievement, the NACA nonetheless suggested that its continued wing research should in the near future "increase the range and greatly improve the economy of both military and commercial aircraft."

With this long and very detailed technical story covering the conceptual genesis of the laminar-flow airfoils at Langley in mind, the reader is now ready to peruse a string of documents showing how the American aircraft industry, in association

[26] These reports were "Preliminary Report on Laminar-Flow Airfoils and New Methods Adopted for Airfoil and Boundary-Layer Investigations," June 1939, by Jacobs, and "Preliminary Investigation of Certain Laminar-Flow Airfoils for Application at High Speeds and Reynolds Numbers," August 1939, by Jacobs, Ira H. Abbott, and A.E. von Doenhoff.

In the spring of 1941, NACA Langley installed an experimental low-drag test panel on the wing of a Douglas B-18 airplane. The panel was fitted with suction slots and pressure tubes for a free flight investigation of the transition from laminar to turbulent flow in the boundary layer. Liquid manometers installed in the fuselage measured the pressure at each tube. NASA Image #L-25332 (LaRC)

with the NACA and the military air services, applied the new wings to World War II aircraft.

One of the major emphases of the documentary record is the experience of the North American P-51 Mustang, one of history's most remarkable airplanes and the first aircraft to employ a NACA laminar-flow airfoil. More than any other case study, the Mustang's performance in the war demonstrates how the NACA's laminar-flow airfoils proved to be a success, despite also being a failure. The record of this magnificent fighter plane confirmed expectations of appreciable improvements in speed and range as a result of the low-drag design, but practical experience with this and other aircraft using advanced NACA sections in the 1940s also showed that the airfoil did not perform as spectacularly in flight as in the laboratory. Manufacturing tolerances were off far enough, and maintenance of wing surfaces in the field were careless enough, that some significant points of aerodynamic similarity between the operational airfoil and the accurate, highly polished, and smooth test model were lost. Because the percentage drag effect of even minor wing surface roughness (e.g., dirt, dead bugs, and the dusty footprints of airplane crewmen) increased as airfoils became more efficient, laminar flow could be maintained in actual flight operation only in a very small region near the leading edge of the wing.

NACA Airfoil Sections Employed by the North American P-51 Mustang*

Aircraft	Wing root airfoil	Wing tip airfoil
NA-101 XP-51B Mustang	NAA/NACA 45-100	NAA/NACA 45-100
NA-102 Mustang III	NAA/NACA 45-100	NAA/NACA 45-100
NA-102 P-51B Mustang	NAA/NACA 45-100	NAA/NACA 45-100
NA-103 P-51C Mustang	NAA/NACA 45-100	NAA/NACA 45-100
NA-104 P-51B Mustang	NAA/NACA 45-100	NAA/NACA 45-100
NA-105 Mustang V	NACA 66-(1.8)15.5	NACA 66-(1.8)12
NA-105 Mustang VI	NACA 66-(1.8)15.5	NACA 66-(1.8)12
NA-105 XP-51F	NACA 66-(1.8)15.5	NACA 66-(1.8)12
NA-105 XP-51G	NACA 66-(1.8)15.5	NACA 66-(1.8)12
NA-105 XP-51J	NACA 66-(1.8)15.5	NACA 66-(1.8)12
NA-106 Mustang IV	NAA/NACA 45-100	NAA/NACA 45-100
NA-106 P-51D Mustang	NAA/NACA 45-100	NAA/NACA 45-100
NA-107 P-51C Mustang	NAA/NACA 45-100	NAA/NACA 45-100
NA-109 Mustang IV	NAA/NACA 45-100	NAA/NACA 45-100
NA-109 P-51D Mustang	NAA/NACA 45-100	NAA/NACA 45-100
NA-110 Mustang IV	NAA/NACA 45-100	NAA/NACA 45-100
NA-110 P-51D Mustang	NAA/NACA 45-100	NAA/NACA 45-100
NA-111 Mustang IV	NAA/NACA 45-100	NAA/NACA 45-100
NA-111 P-51C Mustang	NAA/NACA 45-100	NAA/NACA 45-100
NA-111 P-51D Mustang	NAA/NACA 45-100	NAA/NACA 45-100
NA-111 P-51K Mustang	NAA/NACA 45-100	NAA/NACA 45-100
NA-112 Mustang IV	NAA/NACA 45-100	NAA/NACA 45-100
NA-112 P-51D Mustang	NAA/NACA 45-100	NAA/NACA 45-100
NA-117 P-51H Mustang	NACA 66-(1.8)15.5	NACA 66-(1.8)12
NA-122 Mustang IV	NAA/NACA 45-100	NAA/NACA 45-100
NA-122 P-51D Mustang	NAA/NACA 45-100	NAA/NACA 45-100
NA-124 Mustang IV	NAA/NACA 45-100	NAA/NACA 45-100
NA-124 P-51D Mustang	NAA/NACA 45-100	NAA/NACA 45-100
NA-126 P-51H Mustang	NACA 66-(1.8)15.5	NACA 66-(1.8)12
NA-127 Mustang IV	NAA/NACA 45-100	NAA/NACA 45-100
NA-127 P-51D Mustang	NAA/NACA 45-100	NAA/NACA 45-100
NA-138 Mustang IV	NAA/NACA 45-100	NAA/NACA 45-100
NA-138 P-51D Mustang	NAA/NACA 45-100	NAA/NACA 45-100
NA-139 P-51H Mustang	NACA 66-(1.8)15.5	NACA 66-(1.8)12

*Adapted from David Leidner, "The Incomplete Guide to Airfoil Usage," Analytical Methods, Inc., Redmond, WA 98052.

Still, the Mustang's airfoil section turned into an excellent wing. Ironically, this development was due to its high-speed performance rather than its low-drag. In "one of those rare instances in the history of technology in which a system becomes a success because it unexpectedly excels at something for which it was not originally designed," a decade of dedicated airfoil research by the NACA resulted, not in what Eastman Jacobs and his colleagues were after, but in something else, almost as good.[27] Not only were the NACA's 6-series laminar-flow airfoils used with great success on the Mustang, they were also to be employed on just about every other high-speed airplane that came after it, up to the time that sophisticated computer-aided design took over and started customizing advanced airfoil shapes in the 1980s.

The delineation of the so-called laminar-flow airfoils was thus a great contribution by the NACA, even if not exactly in the way, or to the degree, advertised. The last document in this chapter is an excerpt from Theodore von Kármán's 1945 publication for the Army Air Forces command, *Where We Stand.* There can be no question after reading this except that von Kármán, too, was a fan of the laminar-flow airfoil—just as were the German military pilots who confronted it over the battlefields of Europe and the German aeronautical engineers who wondered over its design features. But more than any other group, the people that valued the P-51's performance the most were the Allied bomber pilots who flew dangerous missions deep into the German heartland when no previous fighter could fly far enough to escort them all the way to their targets and back. Thanks in part to its highly efficient aerodynamic design, the P-51 could fly all the way to Berlin and back, saving innumerable lives of Allied crewmen. General Henry H. "Hap" Arnold, commanding general of the U.S. Army Air Forces in Europe at the time, called it "one of the great miracles of the war," the appearance of the long-range fighter escort "at just the right moment in the very nick of time to keep our bomber offensive going without a break."

This chapter's account of the laminar-flow story ends in 1945, but the quest for true laminar-flow wings most certainly did not end there. Aerodynamicists around the world have never really given up their quest for the pot of gold at the end of the rainbow, primarily because "the attainment of practical laminar flow may well represent the final breakthrough to which pure aerodynamics can lead us."[28] Immediately after the war, the British built a laminar flow flight-test aircraft, the Armstrong Whitworth A.W. 52, a swept-wing, twin jet flying wing. Unfortunately, the aerodynamic performance of this bold prototype also proved much worse in actual flight than what wind tunnel tests predicted—primarily in this case because the introduction of the swept wing "added confounded elements to the laminar flow equation." As "today's Eastman Jacobs" at Langley, Fayette S. Collier (head of NASA's Laminar Flow Control Project Office at Langley Research Center), pointed out in the mid-1990s, "Aerodynamicists had to go through a whole new learning process when swept wings came in, because now they were dealing with cross-flow disturbances as well as chordwise disturbances."[29]

[27] Anderson, *A History of Aerodynamics*, p. 352.

[28] Stephan Wilkinson, "Go With the Flow," in *Air&Space Smithsonian* 10 (June/July 1995): 33.

Other laminar-flow projects followed, most of them involving efforts not just for shaping wing sections but for actively controlling the flow over the wing via sucking and blowing, ideas going back to the late 1920s. Following the failure with the A.W. 52, in 1955, the British initiated an active LFC project using three de Havilland Vampire jet fighters. Porous surfaces were given to the planes' wings through which the turbulent air in the boundary layer was sucked away. Aerodynamically, the tests showed promise, but structurally, the holes and additional elements required by the experimental system caused the airplanes some serious problems. Later, NASA tried something similar with a Lockheed F-94 interceptor, with roughly the same mixed results. In 1966, Northrop and the U.S. Air Force put together what has been called "the biggest laminar flow project ever attempted." Two Douglas WB-66 jets, previously used for weather reconnaissance, were converted into X21As by equipping their wings with a convoluted array of laminar-flow control apparatus: "thousands of razor-thin slits that were in turn perforated with over 815,000 minuscule holes, each of which sucked away turbulent air into a vast internal network of nearly 68,000 ducts, all leading to a pair of high-pressure pumps under the wings."[30] The X21A program proved that such a complex system was technically feasible but prohibitively expensive and a maintenance nightmare. The energy crisis of the 1970s rejuvenated NASA's laminar flow efforts, with different LFC test programs eventually being designed for the F-111 swing-wing at transonic speeds, the Grumman F-14, the Lockheed Jetstar, and a specially outfitted NASA Boeing 757, almost all of them somehow involving the potential aerodynamic benefits of active systems using pinholes in the wings, ducts, and pumps. Today, there are aerodynamicists at NASA and elsewhere who are still as enthralled with the idea of a laminar-flow wing as Eastman Jacobs and his associates were 60 years earlier. Whether the "miraculous" benefits of the technology will ever truly be realized is also still an open question.

[29] Quoted in Wilkinson, p. 35.

[30] Wilkinson, p. 36.

One of the reasons why many experts feel the North American P-51 Mustang represents the highest level of technical achievement ever accomplished in a propeller-driven fighter aircraft is its highly effective low-drag wing. The XP-51 was the first aircraft to incorporate an NACA laminar-flow airfoil. This is the second XP-51, which arrived at Langley in March 1943. NASA Image #L-34304 (LaRC)

The NACA put the Mustang through systematic drag reduction tests in Langley's Full Scale Tunnel. NASA Image #L-34590 (LaRC)

In the early 1950s the British conducted a wide-ranging program of laminar-flow control (LFC) research using three De Havilland Vampire jet fighters. The DeHavilland Vampire was Great Britain's second jet fighter of World War II, first flown in 1943. Powered by Rolls Royce engines, it could fly at 540 miles per hour, about 100 mph faster than the Spitfire. National Air and Space Museum, Smithsonian Institution (SI 2001-1109)

On the wing of the Vampire, researchers experimented with various porous surfaces in hopes of achieving prolonged laminar flow. The British tried a rolled metallic cloth on the wing surface, a technique that did not work, as roughness picked up in the mesh caused a premature transition to turbulent flow. They later used a perforated sheet metal. Eventually, they succeeded in providing a very smooth surfaces back to 25 percent of the chord. National Air and Space Museum (NASM 1A35429), Smithsonian Institution

In the early 1960s, NASA flew the Lockheed F-94 interceptor in a program designed to investigate the possibility of active laminar-flow control, but as with all other efforts in this area up to this time it achieved mixed results. Developed from the single-seat F-80 Shooting Star, America's first operational jet fighter, the two-seat F-94 Starfire served as the U.S.'s first operational jet all-weather interceptor. The first U.S. jet equipped with an afterburner, it could reach speeds as high as 630 miles per hour, cruising at 520 mph. National Air and Space Museum (NASM 1B13663), Smithsonian Institution

The F-94, which NASA flew in the early 1960s as part of a program of active laminar-flow-control experiments, employed a NACA 65-213 low-drag airfoil section. National Air and Space Museum, Smithsonian Institution (SI 82-14064)

Co-sponsored by the U.S. Air Force, the Northrop Corporation, and NASA, the biggest laminar flow project ever attempted began in 1963 and involved flight tests with the X-21, an experimental aircraft derived from the Douglas WB-66. Although the program encountered a number of serious difficulties, by the mid-1960s flights were attaining a significant degree of laminar flow over the X-21's wings. Some of the researchers involved believe that this program was terminated prematurely and could have produced even more helpful data. National Air and Space Museum (NASM 1B29178), Smithsonian Institution

The energy crisis of the early 1970s rejuvenated NASA's efforts to investigate new methods for laminar flow control. Its major research effort into the early 1980s involved a flight program with the General Dynamic F-111, a variable-sweep wing fighter developed in the 1960s under Secretary of Defense Robert S. McNamara as a plane that could be used by both the air force and the navy. One of the goals of the F-111 laminar flow program was to quantify the adverse effects of cross-flow instability due to wing sweep. The flight tests were carried out at Dryden Flight Research Center in 1980. Researchers installed airfoil "gloves" on the F-111, which was re-designated as the F-111/TACT (Transonic Aircraft Technology) airplane, and they tested it through a range of sweep angles. Results were again mixed, but provided the basis for a follow-on program involving another variable-sweep aircraft, the F-14. National Air and Space Museum (NASM 00049124), Smithsonian Institution

A navy fighter with very high performance and great operational versatility (and modified NACA 64A209.65 airfoil), the Grumman F-14 with its variable-sweep wing participated in laminar-flow flight experiments at NASA's Dryden Flight Research Center from 1984 to 1987. Results proved inconclusive, but the F-14 transition data did lead to an improved understanding of a number of complex aerodynamics effects related to wing cross-sectional shape, wing sweep, and the potential of boundary-layer suction (even though suction was not used on the F-14). National Aeronautics and Space Administration (NASA) via National Air and Space Museum (NASM 9A00420), Smithsonian Institution

The Documents

Document 4-1

Edgar S. Gorrell and H.S. Martin, "Aerofoils and Aerofoil Structural Combinations," NACA *Technical Report* 18 (Washington, 1918).

It will be obvious to the reader even from this brief early NACA report on airfoils that American engineers late in World War I were still oblivious to the significant theoretical work on which the Germans were basing their achievement of superior wing shapes, and were relying totally on empirical findings for further improvements. Gorrell and Martin make references to experimental programs at the National Physical Laboratory in England and at Eiffel's laboratory in France but no where in the article do they refer to the wartime work of Ludwig Prandtl and others at the University of Göttingen, which they will not know anything about until after Armistice Day.

Document 4-1, Edgar S. Gorrell and H.S. Martin, Aerofoils and Aerofoil Structural Combinations, NACA* Technical Report *18 (Washington, 1918).

REPORT NO. 18.
AEROFOILS AND AEROFOIL STRUCTURAL COMBINATIONS.
By Edgar S. Gorrell and H.S. Martin.
INTRODUCTION.
FORMULAE NOTATION.
(Pounds, square feet, miles per hour units.)

A = Area of aerofoil in square feet. The brass model aerofoils were 18 by 3 inches.

C.P. = Center of pressure; i.e . the point of intersection of the resultant vector of forces with the plane of the aerofoil's chord.

D = Drag of the aerofoil as given by $D\text{-}K_xAV^2=D_1\text{-}D_0\text{-}D_s$.

Density = Density of standard air; i.e . 0.07608 lbs./cu. ft.

D_0 = Drag of the aerofoil when $V=0$.

D_1 = Drag of the aerofoil at the correct V for the test.

D_s = Drag of the spindle used as a spindle correction.

i = Angle of incidence; i.e . angle of wing chord to the wind.

K_x = Drag coefficient used in the standard formula $D=K_xAV^2$.
K_y = Drag coefficient used in the standard formula $L=K_yAV^2$.
L = Lift of the aerofoil as given by $L=K_yAV^2=L_1=L_0$.
L/D = Ratio of lift to drag.
L_0 = Lift of the aerofoil when $V=0$.
L_1 = Lift of the aerofoil at the correct V for the test.
M = Moment of resultant vector$=(M_1-M_0)/3.65$ for M.I.T. balance.
M_0 = Moment of resultant vector when $V=0$.
M_1 = Moment of resultant vector at the correct V or the test.
V = Velocity of the wind; i.e., 30 miles per hour for these tests.

Mathematical theory has not, as yet, been applied to the discontinuous motion past a, cambered surface using the term cambered as generally understood in aeronautics. For this reason, we are able to design aerofoils only by consideration of those forms which have been successful, by applying general rules learned by experience and by then testing the aerofoils in a reliable wind tunnel. A great many aerofoils have from time to time been tested and from them we know general rules which must be observed concerning camber and the variations of camber on the upper and lower surfaces, if we are to expect to attain even fair results. Results better than the ordinary are only attained when these general rules are observed, and patience and good fortune are combined. There are equations of curves which are very much like some aerofoils but they are not deduced from mathematical knowledge of the flow past an aerofoil but rather from the knowledge of the shape of these curves, and a good idea of the shape of a satisfactory aerofoil. It seems possible that eventually we shall know mathematically the best form for speed and climb, but the practical application of this knowledge may be more difficult than the present method of designing.

OBJECT OF THE TEST.

Although a great many aerofoils have been tested, many are useless from a practical point of view. It seems safe to assert that in this country nearly every aerofoil used is either one of the best five or six tested by M. Eiffel near Paris or by the National Physical Laboratory at Teddington, England, or based upon them, with some slight modifications. As will be seen from the results of these tests apparently slight modifications may make considerable differences.

We are thus limited to a few aerofoils, and some of these lack certain desirable characteristics as to the depth of wing spars combined with aerodynamical efficiency. It would seem of advantage to have the following results of the tests made upon the six structurally excellent and heretofore aerodynamically unknown aerofoils designed by the Aviation Section Signal Corps, United States Army. This constitutes the largest single group of aerofoils, excepting those of the N.P.L. and M. Eiffel, which has been tested and published.

DESIGN OF THE AEROFOILS

U.S.A.1 is a modification of the Clark aerofoil to receive a deeper rear spar. It was designed to be a good high-speed wing, with good L/D, having at the same time sufficient rear spar depth.

Depth of front spar = 0.0584 chord.
Depth of rear spar = 0.0497 chord.

U.S.A. 2 is a combination of the good characteristics of both R.A.F. 3 and R.A.F. 6. It is an aerofoil designed for use in a biplane combination as follows: The depth of the front spar measured along a line making an angle of 10° 45' (angle of stagger) with the vertical is 0.875 that of R.A.F. 6. The depth of the rear spar is 0.88 that of the front spar of U.S.A. 2. The center of the front spar is 0.12 of the cord, and the center of the rear spar is 0.70 of the chord, from the leading edge. The curve of the upper surface is R.A.F. 3 and that of the lower surface is R.A.F. 3 lowered and modified to take the deeper spars.

U.S.A. 3 has the same structural features of U.S.A. 2. The nose is moved *forward* 3/8 inch and the ordinates are measured and calculated as a ratio of a 30-3/8-inch chord. These ordinates are then transposed to a 30-inch chord. The rear 0.8 of U.S.A. 3 is identical with the rear 0.8 of U.S.A. 2 and the changes necessitated occur in the leading 0.2 of the aerofoil.

U.S.A. 4 was designed as indicated for U.S.A. 3 except that the nose was moved 3/8 inch backward instead of forward as in U.S.A. 3.

U.S.A. 5 is not based upon any particular wing section but upon a general consideration of the factors necessary to result in an aerodynamically and structurally efficient aerofoil.

U.S.A. 6 is designed from the basic principles of a certain foreign aerofoil that has rendered particularly good results in the European conflict.

DISCUSSION OF THE RESULTS.

The results in no way contradict any of the known general principles regarding the effects of changing variations in the camber of aerofoils. There are rules for determining the relative value of different wing sections. The lift-drift ratio, which is a measure of the efficiency of an aerofoil, gives information as to the value of the wing. The qualities desired in a good aerofoil are high speed, or low resistance, great climbing ability, and excellent weight carrying capacity. Any one of these characteristics may be secured, but only at the expense of the other two to a certain extent. In a pursuit machine, where compromises are made to secure both high speed and excellent climbing ability, weight carrying is sacrificed. In a bombing machine weight carrying ability is desired to the partial sacrifice of speed and climb. In a training machine all three characteristics are desired, but in moderation. A machine designed for high speed alone has only a limited practical application.

It is generally conceded that there is no "best" aerofoil, for all have different characteristics and perform different functions. The selection of a desirable section depends on the performance required of the airplane desired.

All of the U.S.A. aerofoils have the fundamental quality of being structurally sound, permitting the use of sufficiently deep wing spars.

As suggested in Mr. Alexander Klemin's "Course in Aeronautics," the U.S.A. aerofoils are considered under the following headings:

(a) The maximum value of L/D, the angle at which it occurs, and the corresponding Ky. – The reason for this comparison is that an airplane in normal horizontal flight will generally be navigated at the angle giving the best L/D ratio, which is therefore important from an efficiency point of view. The value of the lift coefficient at the best L/D ratio is important because the greater the lift at this ratio the smaller the area of the wing surface required for the load. With a heavy machine a big lift coefficient is desirable. With a pursuit or racing machine a good L/D at small angles is desirable, so that with a sufficiently powerful motor a great speed may be obtained.

(b) The maximum Ky, the angle at which it occurs, and the corresponding L/D ratio. – The maximum Ky is a very important characteristic. The greater the maximum Ky, the slower is the speed at which a machine may fly and land. If large values of Ky are accompanied by good L/D ratios, then the machine will be efficient in climbing, though the best angle of climb is by no means at that of the maximum Ky. If the maximum Ky occurs at a high angle, then there are possibilities of good speed range.

(c) The shape of the burble point. – If the lift past the burble falls off very rapidly, the airplane can be quickly stalled. On the other hand, a wing with a flat lift curve at the burble point will avoid quick stalling. In all the U.S.A. aerofoils the shape of the curves at the burble points is sufficiently flat to be satisfactory.

(d) The L/D ratio at small angles of incidence and small values of Ky determine whether or not the aerofoil is really suitable for high speeds. We conform to Mr. Klemin's comparison value of Ky = 0.00086.

(e) Movement of center of pressure at low angles. – The importance of this fact is readily apparent from consideration of stability. In all the U.S.A. aerofoils the movement of the center of pressure is not prohibitive or unsatisfactory.

(f) Structural considerations are satisfactory in such aerofoils.

(g) Subheads (a), (b), and (d) are tabulated herewith for convenience of reference.

U.S.A. 1, its maximum L/D of 17.8, the highest of any U.S.A. aerofoils, occurs at 3.0°, at which point its center of pressure motion is fairly rapid but not so rapid as to make the aerofoil undesirable. This aerofoil would be undesirable as the wings of a very heavy machine, but it is very desirable as the wings of a fast pursuit machine. Its maximum Ky is sufficiently large to warrant a reasonable landing speed. Its L/D at small values of Ky is excellent and usually better than any of the other U.S.A. aerofoils. Because of its slow-landing speed and its great high speed and its burble point occurring at 15°, U.S.A. 1 would make the most satisfactory pursuit machine wing of all U.S.A. aerofoils with the greatest speed range of any U.S.A. aerofoils. Structurally it is excellent.

U.S.A. 4, with its large Ky of 0.00364, would be suitable and very desirable for heavy machines and for machines in which the designer is attempting to obtain a very slow landing speed. It is unsuitable for high speeds because of its low L/D values at small values of Ky. Structurally it is excellent.

U.S.A. 6 has a maximum L/D of 17.4, being second in this particular only to U.S.A. 1, of which the maximum L/D is 17.8. On both U. S.A. 6 and U.S.A. 1 the maximum L/D occurs at 3°. In each the maximum Ky is only fair. The maximum Ky of U.S.A. 1 is better than that of U.S.A. 6, so pursuit machines using U.S.A. 1 could be designed to have a slower landing speed than those using U.S.A. 6. It would appear, judging from the tabulation U.S.A. aerofoils just given, that U.S.A. 6 has better L/D values than has U.S.A. 1 for small values of Ky. However, when we examine this characteristic for many points, it is found that U.S.A. 1 has usually better L/D values for small values of Ky than has U.S.A. 6. Thus it seems that U.S.A. 1 is better than U.S.A. 6 for a pursuit machine. However, U.S.A. 6 could be used on a high-speed machine that is only a trifle slower than the machines using U.S.A. 1, but the machine using U.S.A. 6 would land much faster than the one using U.S.A. 1. At 3°, the angle of maximum L/D for both U.S.A. 1 and U.S.A. 6, the center of pressure movement of U.S.A. 6 is better than that of U.S.A. 1. U.S.A. 6 is undesirable for use on a heavy airplane. Structurally it is satisfactory.

U.S.A. 2 is next best to U.S.A. 4 for heavy machines or machines designed for slow speeds. It is unsatisfactory for a pursuit airplane. Structurally it is satisfactory.

U.S.A. 3 and U.S.A. 5 are above the average of aerofoils.

An off-hand estimate of the U.S.A. aerofoils would arrange them in order of merit as follows, but actual calculation might change this order.

U.S.A aerofoils arranged in order of preference	For carrying heavy loads or for slow landing speeds	For pursuit airplanes
Best	U.S.A. 4	U.S.A. 1
Second best	U.S.A. 2	U.S.A. 6
Third best	U.S.A. 5	U.S.A. 5
Fourth best	U.S.A. 3	U.S.A. 3
Fifth best	U.S.A. 1	U.S.A. 2
Sixth best (worst)	U.S.A. 6	U.S.A. 4

The general rules we have do not permit us to choose between two aerofoils of nearly the same characteristics, so a designer should actually go through the necessary computations, using each of the several possible aerofoils in order to ascertain which aerofoil is the best for the purposes of his design. As a matter of interest rough calculations are here given for a pursuit machine, and designers can follow the general method used herein for any type of airplane they may happen to be designing.

Among the U.S.A. aerofoils it seems apparent that U.S.A. 1 or U.S.A. 6 is best for a pursuit machine. For reasonable comparisons, the weight, horsepower available, and the parasite resistance should be the same for both machines. The weight will be assumed as 1,200 pounds, the parasite resistance as being represented by 0.025 V2 in pounds per square foot per mile per hour units, and the propeller efficiency as given by the following table, though such a propeller might be difficult to obtain in practice:

V in *mpt*	50	60	70	80	90	100	110	120
Efficiency	50	55	60	65	70	75	70	60

The horsepower available curve and the parasite resistance curve can then be plotted, the brake horsepower of the motor being assumed as 150. We may either assume a constant wing area and ascertain which wing section gives the best performance or we may prescribe certain performances and see which aerofoil section will come closer to or better the performances. This will result in variations in wing area and minor changes in weight which can be neglected. A low speed will be taken as 55 miles per hour. This will determine the area. The high speed and climb are to be the best obtainable under the assumed conditions.

Using the equation W = Ky AV2 we have 1200 = Ky A(2/55). The highest Ky of U.S.A. 1 is .00318 and of U.S.A. 6 is .00298, giving as areas required if U.S.A. 1 is used 124.5 square feet; if U.S.A. 6 is used 133.5 square feet.

$$1200 = (Ky)\ (124.5)\ (V^2) \text{ or}$$
$$Ky = 1200/(124.5)\ (V^2)$$

Thus we see that actual calculations demonstrate that U.S.A. 1 is better than U.S.A. 6 for a pursuit machine, considering speed above, for it has a greater high speed.

The best climb of U.S.A. 1 is 1,450 feet per minute at 70 miles per hour and for U.S.A. 6 it is 1,480 feet per minute at 60 miles per hour. Although U.S.A. 6 can climb 30 feet per minute faster than U.S.A. 1, yet the speed of U.S.A. 6 at which best climb occurs is 10 miles per hour less than the speed for tile best climb of U. S. A 1. We believe that the climbing ability of U.S.A. 1 is better for a pursuit machine than is that of U.S.A. 6. Hence U.S.A. 1 excels U.S.A. 6 in both speed and climb characteristics.

The above process should be pursued whenever there is any doubt between the relative desirability of two or more wing sections for specific purposes.

It would seem that Dr. Hunsaker is a trifle low in his estimate wherein he states that an increase in camber above 0.08 for the upper surface is disadvantageous since

four good U.S.A. aerofoils are cambered as follows:

U.S.A. 2 has a camber of 0.088 per cent of the chord.
U.S.A. 3 has a camber of 0.0868 per cent of the chord.
U.S.A. 4 has a camber of 0.089 per cent of the. chord.
U.S.A. 5 has a camber of 0.085 per cent of the chord.

It is generally conceded that the angle of no lift has no connection with the characteristics of an aerofoil. As a matter of interest the angle of no lift occurs in the U.S.A. aerofoil as follows:

Aerofoil	Angle of lift
U.S.A. 1	-2.5
U.S.A 2	-3.25
U.S.A 3	-2.9
U.S.A 4	-3.6
U.S.A 5	-3.05
U.S.A 6	-2.9

Aerofoils arranged in order of maximum negative angle of no lift	Aerofoils arranged in order of preference in order of preference as weight carriers or slow-speed qualities
U.S.A. 4	U.S.A. 4
U.S.A. 2	U.S.A. 2
U.S.A. 5	U.S.A. 5
U.S.A. 3 and U.S.A. 6	U.S.A. 3
U.S.A. 1	U.S.A. 6

From the above table it appears that perhaps at some future date it might be desirable to investigate whether or not the aerofoil with the greatest negative angle of no lift is also the best aerofoil for heavy aeroplanes or aero lanes designed for slow speeds.

Since the lowest value of Ky in the U.S.A. aerofoils occurs in U.S.A. 6, a designer designing for high speed only with no thought of other considerations, could probably obtain a higher speed with U.S.A. 6 than with any of the other U.S.A. aerofoils.

In order to check the values that we have obtained in the tests of the U.S.A. aerofoils, [a] R.A.F. 6 section made of wood was tested and found to conform to former tests which are known to be satisfactory.

An examination of all the published curves of the R.A.F. sections tested at the M.I.T. tunnel show the maximum L/D obtained varied between a little less than 16 to a trifle above 17. Our maximum L/D is equal to 16.78. On page 41 of "Reports on Wind Tunnel Experiments in Aerodynamics," Dr. Hunsaker says "It appears that undetected differences in workmanship and finish between two models may cause a change in coefficients of not more than 3 per cent." Let us assume for all R.A.F. sections tested at the M.I.T. tunnel L and D are correct within 3 per cent.

Possible error in L/D – (L+.03L)/(D-.03D) = L(1.03)/D(.97) = L/D (1.06)
or if the error be at the other extreme

Possible error in L/D = (L-.03L)/(D+.03D) = .97L/1.03D = L/D (0.94)

It is thus seen that all published results of the M.I.T. on tests of R.A.F. 6 are correct within the limits of workmanship and finish and that our test gives a result about the mean of all such tests.

It is suggested that it might be well if the United States Government owned standard brass aerofoils of the R.A.F. and Eiffel types constructed with absolute accuracy and which could be available for use on wind tunnels like the one at the M.I.T. for ecking the accuracy of the tunnel whenever desirable. The Government has standard weights and measurements. Why not apply this same idea to aeronautics?

In British Reports, 1912-13, No. 72, figure 14, the National Advisory Committee for aeronautics in England has suggested a method of corrections for LV. U.S.A. aerofoils were tested at an LV of 11 while R.A.F. 3, 4, 5, and 6 were tested at an LV of 6.3. Making the proper LV correction for the English tests of the R.A.F. 6, we find the N.P.L. results and our results for tests on the R.A.F. 6 give the same maximum L/D thus checking the accuracy of our series.

		Maximum $\frac{L}{D}$.			Maximum K_y.			$K_y = 0.0020$.	
Wing.	LV—chord of wing in feet X relative wind in feet-seconds.	Angle in degrees.	K_y.	$\frac{L}{D}$	Angle in degrees.	K_y.	$\frac{L}{D}$	Angle in degrees.	$\frac{L}{D}$
U. S. A. 1	11	2.0	0.00153	17.8	13.0	0.00315	8.4	0.62	12.8
U. S. A. 2	11	4.0	.00180	14.2	15.5	.00327	6.3	.0	8.8
U. S. A. 3	11	4.0	.001706	16.4	15.4	.00343	6.5	.3	13.4
U. S. A. 4	11	4.0	.00177	15.20	14.0	.00304	8.1	.85	9.1
U. S. A. 5	11	3.0	.001565	16.81	14.0	.00296	5.21	.15	11.6
U. S. A. 6	11	2.0	.001455	17.4	14.0	.0028	7.37	.1	12.3

Document 4-2

Colonel Virginius E. Clark, "Design Your Own Airfoils," *Aviation* (Oct. 1927): 944-46.

When asked to disclose the secret of designing efficient airfoils, Clark responded in this published interview in 1927, "I would gladly tell you the secret if I knew of any. The airfoil sections just seem to lay themselves out and, when good luck attends, fair results are attained." Perhaps nowhere in history is there a more explicit expression of the role of cut-and-try empiricism and personal intuition in technological design than Clark's statement. By the time this interview was published, however, the intuitive stage of airfoil design was on its last leg. A much more advanced way of shaping airfoils had emerged, epitomized by the scientific engineering methods being developed at NACA Langley laboratory in Virginia.

Document 4-2, Colonel Virginius E. Clark, "Design Your Own Airfoils,"* Aviation *(Oct. 1927): 944-46.

Design Your Own Airfoils
An Interview With
COL. V. E. CLARK

Most of the successful airplanes in the United States use airfoils designed by Colonel Clark. Among the most commonly used of his airfoils are the U.S.A. 27 and the Clark V and Clark Y series. These have been used on Colonel Lindbergh's Ryan monoplane and many others of the more successful commercial strut-braced monoplanes; the Vought "Corsair", the Navy PN-10, and the Wright "Apache", all holding world's records; the Glenn Martin Navy T3M, the National Air Transport Mail and Express Planes, the Douglas "Round-the-World" cruisers, the Curtiss Pursuit and the Curtiss Observation, the Consolidated Army and Navy training planes, the Consolidated "Courier", the Douglas Mail Planes and the Douglas Observation and Transport planes, and on many other successful Military, Naval and commercial planes. The characteristics of the Clark Y as obtained by high pressure tests at the National Advisory Committee wind tunnel at Langley Field are shown in Fig. 1.

We asked Colonel Clark one day: "How do you go about designing these airfoils of yours? What is the secret?"

He laughed and said: "I would gladly tell you the secret if I knew of any. The airfoil sections just seem to lay themselves out and, when good luck attends, fair results are attained.

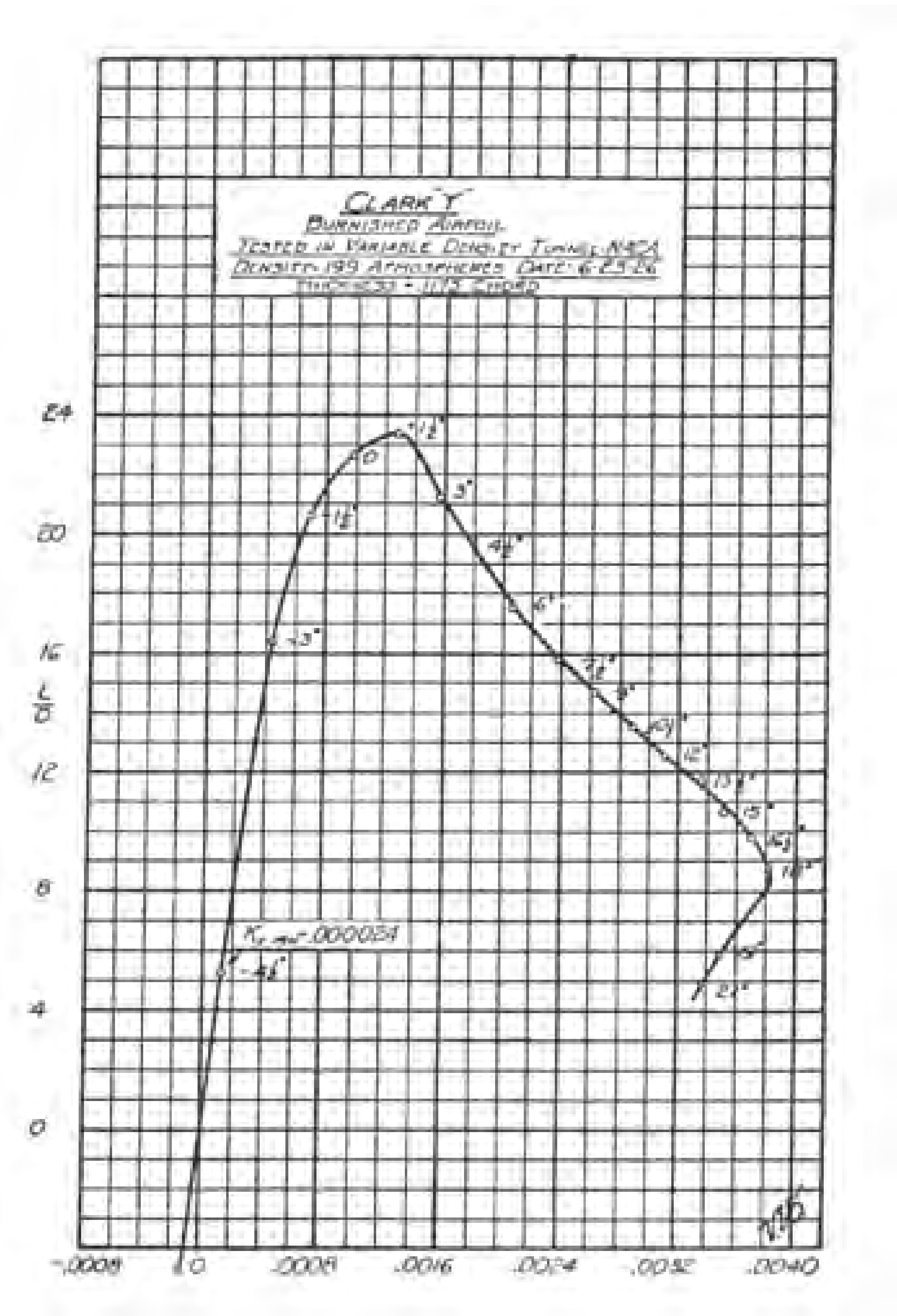


Figure 1

"My airfoils have been selected for various airplanes because comparative tests made in wind tunnels at a low value of the Reynolds number have indicated that they are fairly good. But, you know, the tests made by our National Advisory Committee for Aeronautics in their variable pressure tunnel at Langley Field prove, if we are to accept them, that even for the purposes of comparison, tests made at a low value of the Reynolds number are useless and misleading in many ways. Airfoil A may appear far superior to Airfoil B for a particular purpose when tested with a pressure of one atmosphere, whereas, when tested at twenty atmospheres, Airfoil B appears to be much better than Airfoil A for the same purpose. And no one has traced sufficient consistency in scale variations to justify a reliable system of rules for the prediction of full scale characteristics, having given the test results at low scale.

IMPOSSIBLE TO TEST EVERY AIRFOIL

"Therefore it is fair to assume that there are many airfoils which would be more popular than mine if they had been tested with twenty atmospheres' pressure.

"Unfortunately, the Advisory Committee, having taught us to be skeptical of low scale tunnel test results, cannot help us to reassure ourselves as to the merits of airfoils not tested at high value of the Reynolds number. Obviously, it would be impractical, and perhaps a grave misdirection of government funds, for the Langley Field laboratory to undertake to test every airfoil presented to it.

"Hence, if low scale tests are not indicative of comparative merit, and since high scale tests are unattainable, and as the mathematics have not yet been developed for the precise prediction of practical airfoil performance without supporting experimentation, it may be that we must, for a while forget about wind tunnels, for this particular purpose, and, as each new design problem arises, design an airfoil as we think it should be to best meet the requirements of the particular case;—build our wings accordingly, and hope for the best in full flight results.

"With this in mind, an 'adjustable' airfoil section, upon which you have the data has been laid down. By manipulation of the basic section,—changing the thickness or curvature of the median line, or both, according to the methods outlined,—an indefinite number of sections may be obtained, which are affinal. Airfoils tapered in plan, or in thickness ratio, or both, may also be constructed and their characteristics predicted. The thickness may be changed to meet structural demands (wing beam, depth, etc.) and the curvature of median line varied to obtain Maximum Lift Coefficient or Minimum Drag Coefficient to meet the 'performance' requirements of a particular design problem. These latter two important characteristics,—important not only of themselves,—but also because they usually constitute an index of merit for all-around applicability—have been 'predicted'. It takes a deal of temerity to venture such 'predictions', but, after all, for the reasons stated, these predictions of full scale characteristics probably will neither be confirmed nor contradicted. If they are no more inaccurate for the purpose of full flight performance calculations than low scale tunnel tests, as judged by the National Advisory Committee tests, the adjustable airfoil series may be useful.

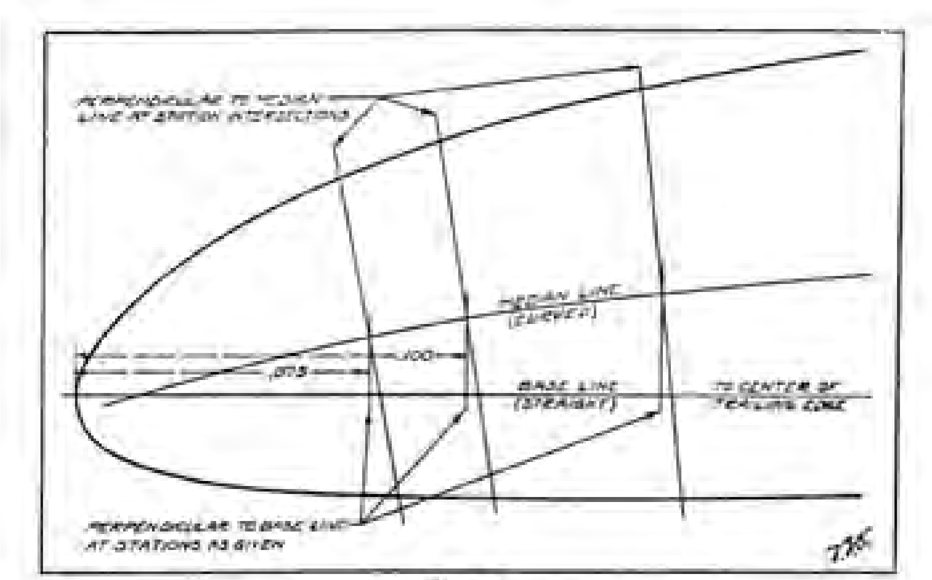


Fig. 2

"The layout method is shown in Fig. 2. A straight 'base' line is first drawn with the desired chord length. Perpendiculars to this line are erected at the stations indicated in Table I and the curved median line plotted and drawn. Then straight lines are drawn, in each case perpendicular to the median line at its point of intersection with the 'base' ordinate line, and the contour of the section plotted from Table II and drawn.

"If a higher maximum lift coefficient than that for the basic section is desired, at the sacrifice of drag and center-of-pressure travel, the ordinates from Table I are multiplied by a *constant* factor greater than unity, to give the desired characteristics 'predicted' in the chart, Fig. 3. Conversely, if low drag or center-of-pressure travel is more important than high lift for the particular case, the constant factor may be less than unity.

"If deeper spars than may be contained within the basic section are required for strength and rigidity, at the sacrifice of drag, the ordinates from Table II are multiplied by a *constant* factor greater than unity, a 'fatter' section drawn around the median line selected, and its characteristics predicted from Fig. 4.

"The combinations of camber and thickness are infinite."

TABLE I

Ordinates of Median Line. For Maximum Displacement .0400 Chord Length

[A] Distance from Leading Edge expressed in Terms of Chord Length
[B] Ordinates of Median Line in Terms of Chord Length

[A]	.025	.05	.075	.10	.15	.20	.30
[B]	.00224	.00911	.01491	.01951	.02671	.03183	.03760

[A]	.40	.50	.60	.70	.80	.90	
[B]	.03995	.03872	.03471	.02877	.02086	.01114	

Ordinates at Leading and Trailing Edges are 0

TABLE II

Half-thickness Ordinates. For Maximum Thickness .10 Chord Length

[C] Distance from L.E. in terms of Chord
[D] Half-thickness in terms of Chord

[C]	.025	.05	.075	.10	.15	.20	.30
[D]	.02191	.03016	.03582	.03996	.04539	.04856	.04985

[C]	.40	.50	.60	.70	.80	.90	1.00
[D]	.04722	.04236	.03593	.02836	.01988	.01077	.00119

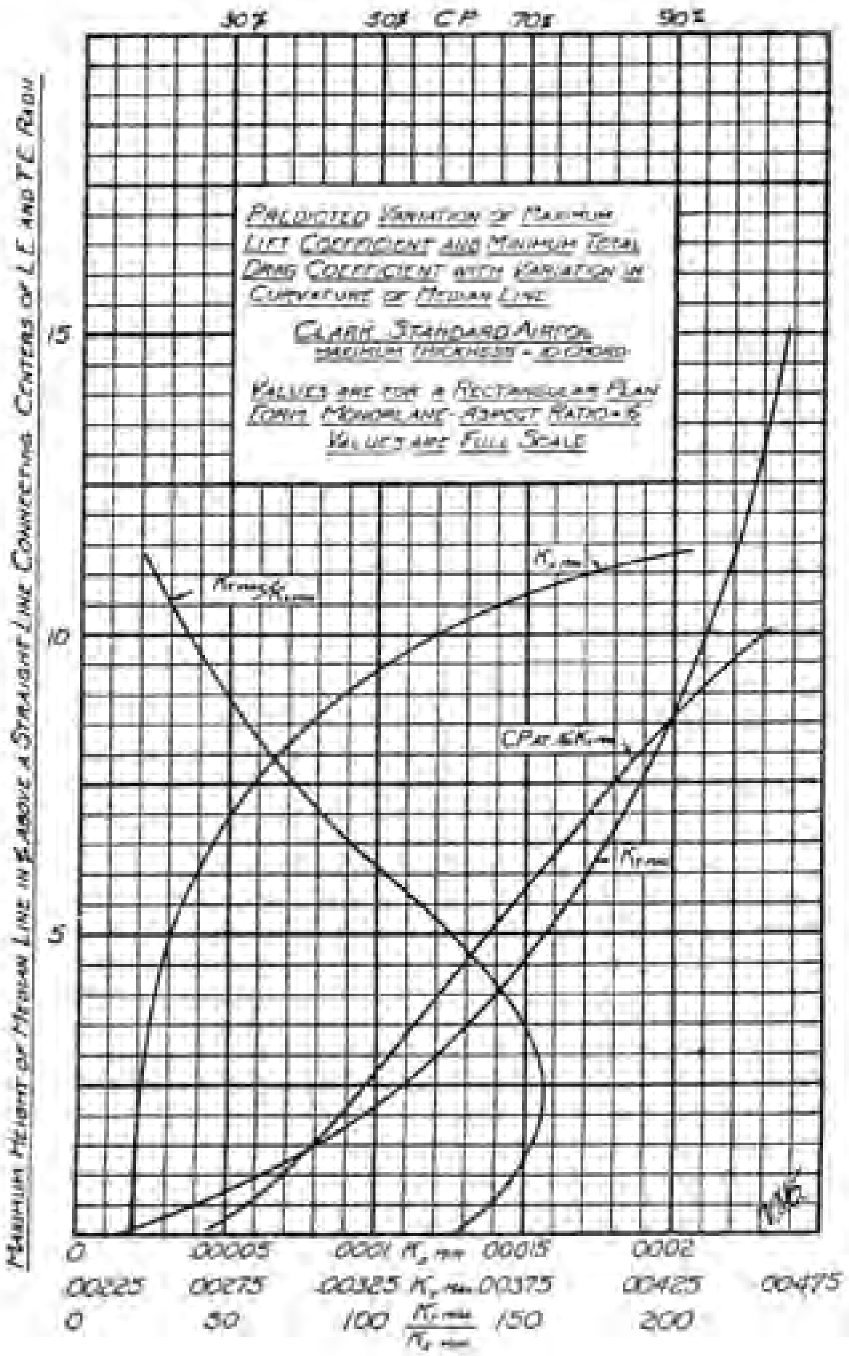

30%
50% C.P. 70%
90%
Predicted Variation of Maximum Lift Coefficient and Minimum Total Drag Coefficient with Variation in Curvature of Median Line
Clark Standard Airfoil
Maximum Thickness = 10% Chord
Values are for a Rectangular Plan Form Monoplane - Aspect Ratio = 6
Values are Full Scale
Maximum Height of Median Line in % above a Straight Line Connecting Centers of L.E. and T.E. Radii
15
10
5
0 .00005 .0001 Kx min .00015 .0002
.00225 .00275 .00325 Ky max .00375 .00425 .00475
0 50 100 150 200

Fig. 3

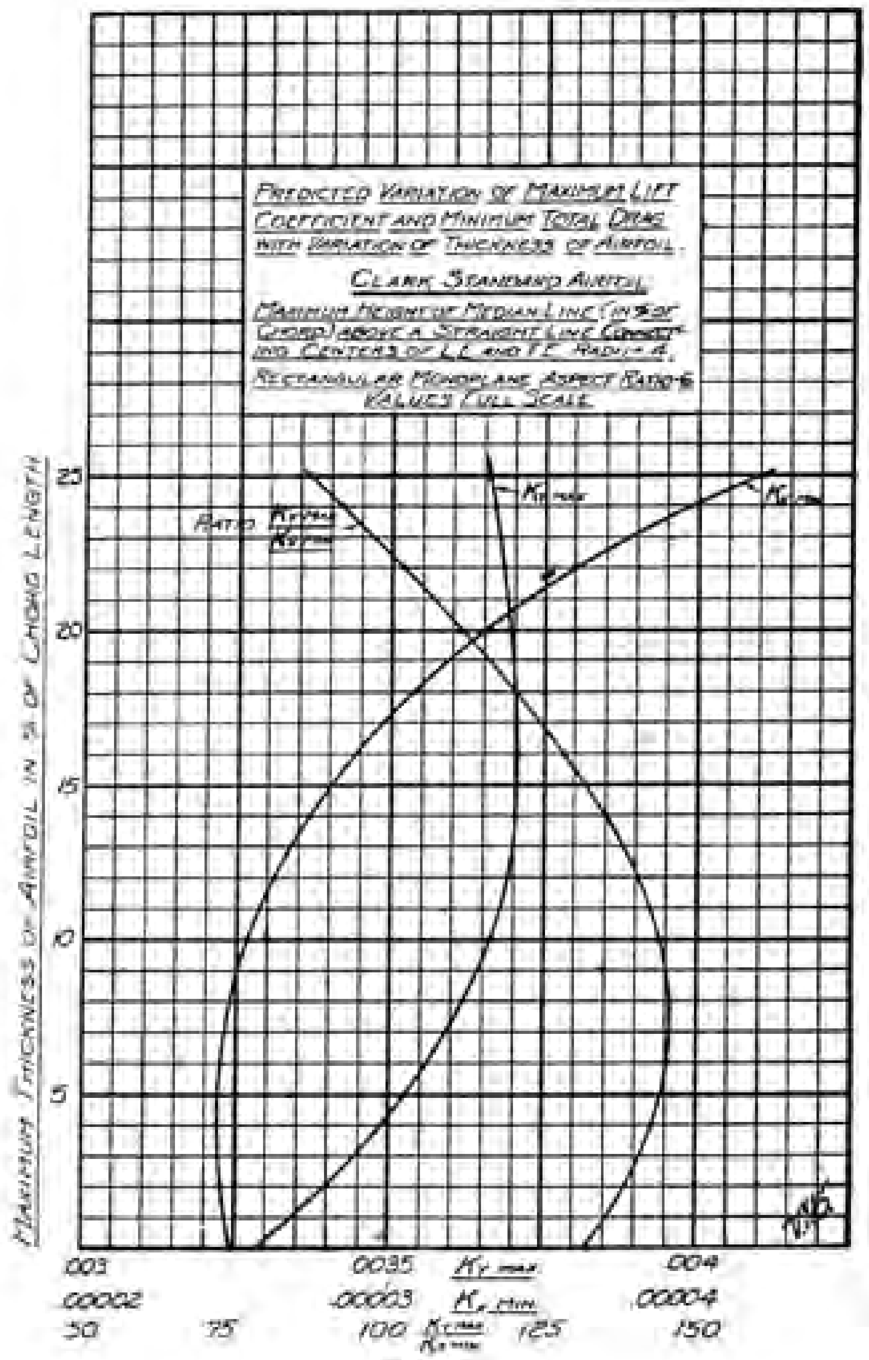

Predicted Variation of Maximum Lift Coefficient and Minimum Total Drag with Variation of Thickness of Airfoil.
Clark Standard Airfoil
Maximum Height of Median Line (in % of Chord) above a Straight Line Connecting Centers of L.E. and T.E. Radii = 4.
Rectangular Monoplane Aspect Ratio = 6
Values Full Scale
Maximum Thickness of Airfoil in % of Chord Length
25
20
15
10
5
.003 .0035 Ky max .004
.00002 .00003 Kx min .00004
50 75 100 125 150

Fig. 4

Document 4-3

Max M. Munk, "General Theory of Thin Wing Sections," NACA *Technical Report* 142 (Washington, 1922).

Munk's thin-airfoil theory was a turning point in the history of aerodynamics and therefore must be included in this chapter in spite of its highly arcane mathematical character. Mark Levinson, a retired professor of mechanical engineering at the University of Maine, has claimed in an unpublished history of early airfoil development ("Airfoil Profiles: Eyeballing, Design, and Selection, 1880-1922," March 1985, pp. 28-9) that TR 142 ushered in "the modern history of airfoil profiles." "All previous work, whether theoretical, experimental, or merely cut-and-try, may be considered as belonging to the pioneer period of that history." Munk's thin-airfoil history is to airfoil design what "the Euler-Bernouilli beam theory is to any of the modern, sophisticated theories of elastic rods or what lumped-parameter electric-circuit theory is to the full equations of electromagnetic field theory"—it is a theory of the "first order." Such theories prove "quite adequate for the purposes of engineering design: the good engineer understands the limitations of such approximate theories and knows when not to use them."

Walter G. Vincenti, a former NACA aerodynamicist at Ames Aeronautical Laboratory in California and professor emeritus of aerospace engineering at Stanford University, has written in *What Engineers Know and How They Know It: Analytical Studies from Aeronautical History* (p. 36) that "Munk theory's provided a new and illuminating way to think about airfoils and caused a basic shift in airfoil design." Before Munk, airfoil designers used "experience and judgment" to draw an airfoil shape, hoping that the lift and drag would be favorable; after Munk, they could "synthesize profiles with approximately predictable lifting characteristics."

A reader untrained in aerodynamics will no doubt have a difficult time understanding the paper, in part because of Munk's own challenging composition style. Still, we encourage at least a brief examination of the paper, both to ascertain the highly mathematical nature of most aerodynamic theory, Munk's and others, and to gain some general insight into the contents of one of the most historic papers in the history of aerodynamics.

Document 4-3, Max M. Munk, "General Theory of Thin Wing Sections," NACA Technical Report *142 (Washington, 1922).*

REPORT No. 142.
GENERAL THEORY OF THIN WING SECTIONS.
By Max M. Munk.

TABLE 1.

Per cent of chord	2.5	5	10	15	20	25	30	35	40	45	50
Factor	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
Per cent of chord		55	60	65	70	75	80	85	90	95	97.5
Factor		[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

TABLE 2.

Shape.	Equation of shape.	α_0	C_{M_0}
	$y=0$	0	0
	[illegible]	$-p$	0
	[illegible]	[illegible]	0
	[illegible]	[illegible]	0
	[illegible]	[illegible]	[illegible]
	[illegible]	[illegible]	[illegible]
	[illegible]	[illegible]	[illegible]

RÉSUMÉ.

The following paper contains a new, simple method of calculating the air forces to which thin wings are subjected at small angles of attack, if their curvature is not too great. Two simple integrals are the result. They contain only the coordinates of the wing section. The first integral gives the angle of attack at which the lift of the wing is zero, the second integral gives the moment experienced by the wing when its angle is zero. The two constants thus obtained are sufficient to determine the lift and moment for any other angle of attack. This refers primarily to a two-dimensional flow in a nonviscous fluid. However, in combination with the theory of the aerodynamical induction, and with our empirical knowledge of the drag due to friction, the results are valuable for actual wings also. A particular result obtained is the calculation of the elevator effect. The following is an outline of the subject as treated in this report:

I. INTRODUCTION.

By changing the angle between the stabilizer and the elevator the wing section formed by the combination of stabilizer and elevator is altered, and this alteration gives rise to new aerodynamical forces. It is useful to discuss this phenomenon from the theoretical point of view, however imperfect the result may be as a consequence of neglecting the viscosity of the air. A theoretical investigation at least gives the limit of what to expect. It enables the investigator to survey and keep in mind a great number of isolated experiences, whether the agreement between theory and experience be more or less close. It induces him to reflect on the phenomenon and thus becomes a source of progress by guiding him to new observations and experiments. It has often occurred even that some relation was thought to be confirmed by experience till the progress of theory made their relationship improbable. And only then the experiments confirmed the improved relation, contrary to what they were supposed to do before. A very conspicuous example of this is the discovery of differences in the atomic weight of certain elements. But is it really necessary to plead for the usefulness of theoretical work? This is nothing but systematical thinking and is not useless as sometimes supposed, but the difficulty of theoretical investigation makes many people dislike it.

In this first section I wish to give a short summary of the theory which I am going later to apply and expand. This theory deals with the relation between the shape of a wing section and the air forces applied to it by a nonviscous fluid. Only the two-dimensional problem is considered. The theory between forms the completion of the theory of the induced drag, in which latter the three-dimensional arrangement of the wings and the lift produced by them alone is considered, without paying attention to the details of producing the lift. The value of the induced drag and the effective angle of attack of every part of the wings result from the calculation. The theory of the wing section, however, gives no drag at all, for the drag additional to the induced drag is due to viscosity. Nor does the theory of the wing section give the true value of the maximum lift. It can be stated, therefore, that the theory of the wing section in its present state gives no indication whatsoever of the practical value of the wing investigated. Still there remain three important pieces of information which can be derived from the theory, all more been or less agreeing with the real phenomenon. These are the relation between the angle of attack and the lift, in particular the angle of attack for lift, the travel of the center of pressure, and the distribution of pressure. It has to be kept in mind that the angle of attack thus calculated for a particular lift coefficient is not yet the true angle of attack of a finite wing. The induced angle of attack has to be added.

We are indebted for the theory of the wing section to Kutta . He showed how the method of the two-dimensional potential can be used to calculate the flow around between sections and hence to deduce the resulting air forces. He confined himself to the straight line and simple circular segments. His idea is to pick out among the multitude of possible potential flows that particular one around the wing section, which at great distance degenerates into parallel flow and which leaves the wing section at the rear edge. His results are simple and important. The direction of the air flow in the case of zero lift of a circular segment of small curvature is parallel to the line dividing into equal parts the angle between the chord and the tangent at the rear end. The lift is proportional to the sine of the angle of attack. The slope of the curve of the lift coefficient plotted against the angle of attack is almost independent of the shape and it is 2π (the angle being measured in arc and the lift coefficient being formed by dividing of the lift per unit of the area by the dynamical pressure). That is, for small lift:

$$L = 2\pi\, S\, \alpha_1\, V^2\, (\rho / 2)$$

Joukowsky extended the theory, and investigated sections which at their rear end almost coincide with a circular segment, having there a common tangent for the upper and lower side. The entire form is generated from the circle, a circular segment forming as it were the skeleton of a Joukowsky section. Considering the connecting line between the rear edge and a pole near the center of curvature of the leading edge as the theoretical chord, the rule for the direction of the zero lift remains as before. The slope of the lift curve is hardly changed; the lift is proportional to the sine of the angles as before.

Karman replaced the circular segment in the Joukowsky section by one formed by two circular segments. This is already mentioned in the second paper of Kutta. These sections have two different tangents at the rear end, and the line which divides the rear angle into two equal parts determines the direction of the zero lift together with the theoretical chord as before. The law for the lift is the same again as for the circular segments of Kutta. Mises discusses in a general way how to obtain even more general sections and proves some general theorems concerning them. The most important is the theorem that the slope of the lift curve plotted as before is never smaller than 2π, and is always exactly 2π if the section is thin and the curvature small. So far it can be stated that only sections are investigated, the medial line of which is a circular segment. If the section is only moderately thick and if the curvature is moderate, too, the lift agrees with that of the segment according to the law found by Kutta.

Document 4-4

Max M. Munk and Elton W. Miller, "Model Tests with a Systematic Series of 27 Wing Sections at Full Reynolds Number," NACA *Technical Report* 221 (Washington, 1925).

This report by Dr. Max Munk announced the first airfoil family in NACA history. Coauthored by Langley engineer Elton W. Miller, it declared that wind-tunnel tests in the VDT showed "remarkable agreement" with Munk's thin-airfoil theory and resulted in the design of several sections (especially the M-6 and M-12) with excellent characteristics. Their most distinctive feature was an S-shaped mean camber line giving a reflexed (or folded back) trailing edge and a stationary center of pressure. The NACA, delighted that its VDT was establishing itself as the primary source worldwide for aerodynamic design data at high Reynolds number, even took the unusual step of naming the members of this experimental series the "M sections" after Munk, the same person who would be forced to resign from the organization two years later.

Though some were adopted for use, the "M section" airfoils never became tremendously popular with airplane builders, perhaps in part because of Munk's stormy departure from the NACA only a short time after their publication. More significantly, the research had not really been directed well enough to the production of airfoils suited to the needs of the time. Munk's method produced some effective shapes but not the optimum airfoils for the wings required by the higher performance, thicker-winged internally braced airplanes coming along in the late 1920s.

Document 4-4, Max M. Munk and Elton W. Miller, "Model Tests with a Systematic Series of 27 Wing Sections at Full Reynolds Number," NACA* Technical Report *221 (Washington, 1925).

REPORT No. 221

MODEL TESTS WITH A SYSTEMATIC SERIES OF 27 WING SECTIONS AT FULL REYNOLDS NUMBER

By, MAX M. MUNK and ELTON W. MILLER

SUMMARY

A systematic series of 27 wing sections, characterized by a small travel of the center of pressure, has been investigated at 20 atmospheres pressure in the variable density wind tunnel of the National Advisory Committee for Aeronautics.

The results are consistent with each other, and indicate that for such "stable" sections a small effective camber, a small effective S-shape and a thickness of 8 to 12 per cent lead to good aerodynamic properties

PURPOSE OF THE INVESTIGATION

This report contains the results of the investigation of the first systematic series of wing sections, 27 all together, made in the variable density wind tunnel of the National Advisory Committee for Aeronautics at about 20 atmospheres pressure. It was desired to obtain information about those aerodynamical properties of the wing sections which can not be computed. Those are the drag at several angles of attack, and the two values of the lift coefficient when (a) the lift coefficient has its maximum and (b) when the air forces change irregularly, commonly known as the "burble point." Without additional work, there was also obtained a check on the aerodynamic properties open to computation, namely, the lift and the moment.

PROGRAM OF THE INVESTIGATION

In this first systematic series the measurements were confined to one tank pressure, about 20 atmospheres. This gives approximately a full size, Reynolds number, for the model scale is about one-tenth, the velocity about one-half of the actual velocity.

The investigation was confined to such wing sections as have a very small travel of the center of pressure. The rate of the travel of the center of pressure is certainly an aerodynamic property of great practical importance, affecting the usefulness of the section for design purposes; it is not wise to compare the performance of several wing sections without taking the different rates of travel of the center of pressure, if any, into account. Within the useful range of the angle of attack, the wing sections described in this report have their center of pressure at about 25 per cent of the chord. Their rate of travel of the center of pressure is accordingly small, and the comparison of their performance is all that remains to be done. Wing sections with a larger rate of travel of the center of pressure may be taken up in a later research.

ARRANGEMENT OF THE TESTS

The 27 models were made of duralumin and were rectangular and not warped. The span is 30 inches; the aspect ratio is 6. The 27 wing sections form a systematic series. The series begins with three symmetrical sections of different thicknesses, Ml, M2, and M3. The curves are affine—i.e., the three sets of ordinates can be obtained from each other by multiplying each ordinate by a constant. Three more sections are then obtained by adding to each of the sets of ordinates M1, M2, M3

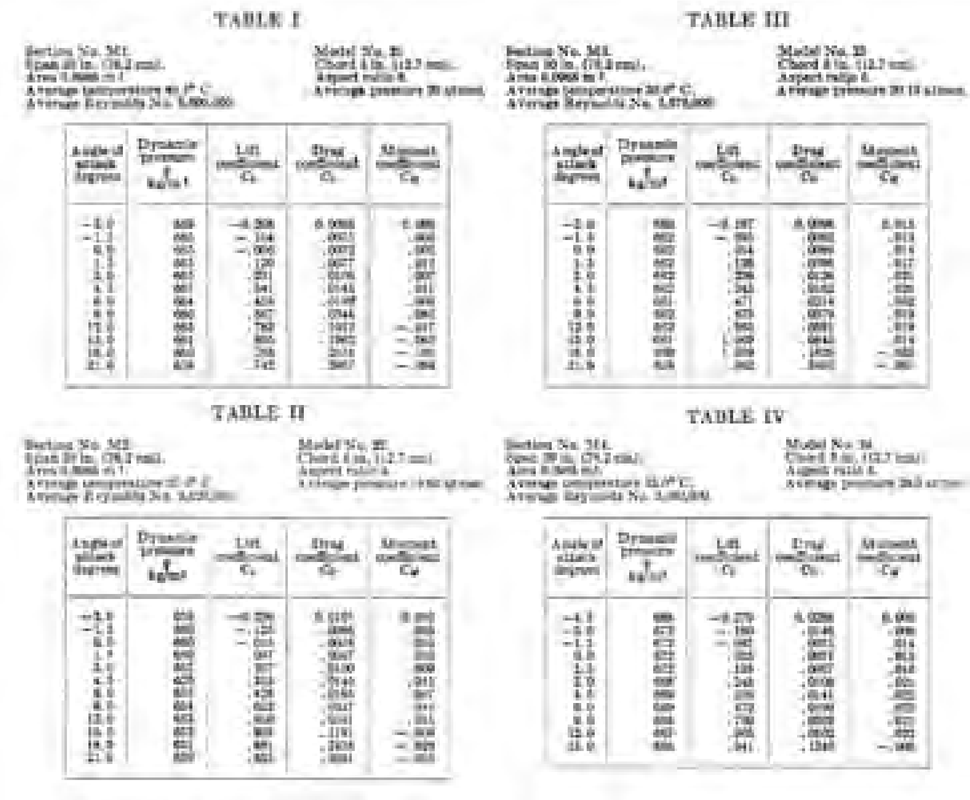


TABLE I

Angle of attack degrees	Dynamic pressure q kg/m²	Lift coefficient C_L	Drag coefficient C_D	Moment coefficient C_M
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

TABLE III

Angle of attack degrees	Dynamic pressure q kg/m²	Lift coefficient C_L	Drag coefficient C_D	Moment coefficient C_M
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

TABLE II

Angle of attack degrees	Dynamic pressure q kg/m²	Lift coefficient C_L	Drag coefficient C_D	Moment coefficient C_M
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

TABLE IV

Angle of attack degrees	Dynamic pressure q kg/m²	Lift coefficient C_L	Drag coefficient C_D	Moment coefficient C_M
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

the set of ordinates of a certain camber line, say "a," so chosen that theoretically its center of pressure does not travel. The series is further increased by substituting double the ordinates, 2a for a; then another camber line "b," with the same stability characteristics, and then combinations of the two camber lines. The camber lines "a" and "b" will be most easily recognized in wing sections M4 and MI0. This process of obtaining the shapes of the wing sections leads to their classification in Table XXVIII. The ordinates of the sections are given in Table XXIX in per cent of the chord. Each figure contains a drawing of the section.

Each airfoil was exposed to the air stream of the variable density wind tunnel of the National Advisory Committee for Aeronautics. It was fastened by thin wires to the balance of this tunnel. Moreover, a skid rigidly fastened to the airfoil was hinged to a vertical bar, forming a part of the balance. This bar extends across the air stream in rear of the model; it is shielded from the air stream and can be moved up and down. When moved thus, the angle of attack of the airfoil is changed. After the airfoil was put in, the tank was closed and the air pressure increased up to about 20 atmospheres. The air forces of the airfoil were then determined over a range of several angles of attack. The drag of the wires and of other attachments were determined in a separate test under the same conditions of flow. The measured drag has been corrected for this drag of the fastening parts in the usual way.

RESULT OF THE TESTS

The results of the tests are given in Tables I to XXVII and are illustrated in the 27 figures. The angle of attack always refers to a line fixed with respect to the section as shown in each diagram. In the tables the air forces are represented by the lift coefficients, the drag coefficients, and the moment coefficients. The lift and drag coefficients are obtained by dividing the lift or drag by the wing and by the dynamic pressure V^2 ($\rho/2$) where V denotes the velocity of the air stream and ρ the mass density of the air. The diagrams are so-called polar curves. The lift coefficient is plotted vertically up, and against it to the right, the drag coefficient, and to the left the moment coefficient. This latter refers to the moment of the air forces with respect to a point of the chord, one quarter chord from the leading edge. This point is chosen because it gives the least variation of the moment coefficient. The moment is divided by the wing area, by the dynamic pressure, and by the length of the chord. The Reynolds number is computed with the chord as the characteristic length.

The parabola of the induced drag coefficient for the aspect ratio 6 has been inserted in each diagram. No correction has been made for the influence of the tunnel walls, which may be perceptible, as the wing span is half the tunnel throat diameter. This question is not yet sufficiently cleared up.

In Table XXX, a survey of the series and of the results obtained is given. The first column gives the number of the wing section. The next three columns contain the minimum drag coefficient, the lift coefficient at the "burble point," and the maximum lift coefficient, if any. The last column gives the average moment coefficient, which is always small for the wing sections considered in this investigation.

DISCUSSION OF THE RESULTS

The main results of this test lie in the presentation of new information about the properties of the wing sections given in the tables and in the diagrams.

It seems that a small travel of the center of pressure is generally combined with a smaller maximum lift coefficient. Good sections are in the neighborhood of M6.

The test charts show that at full size Reynolds number, the minimum drag is much smaller than we are accustomed to obtain in the ordinary atmospheric wind tunnel. The maximum lift is not necessarily larger at a larger Reynolds number.

MODEL TESTS

One remark concerning the results seems pertinent. As shown by mathematical reasoning in Technical Report No. 191 of the National Advisory Committee for Aeronautics, the moment curves in the diagrams should theoretically be straight vertical lines. Most of them have approximately this shape, but not all of them. The small discrepancies can often be explained by taking the second approximation of the computations into account. For instance, with actual sections of a finite thickness, the theoretical leading edge is situated halfway between the actual one and the center of curvature of the leading edge, giving a shorter effective chord than the actual one. A very thick section, besides, is slightly more stable than a thin section of the same mean curve. Quite irregular moment curves can only be explained by sudden changes of the character of the flow just as at the burble point.

CONCLUSION

Looking at the results obtained in the variable density tunnel (including Technical Report No. 217) from a broader point of view, it is now established that the results obtained at the full size Reynolds number do not agree with the results at a diminished Reynolds number. Furthermore, tests now under way show that the variable density tunnel operated at one atmosphere gives results with a given wing section similar to the results obtained in other wind tunnels.

We conclude from these facts that the results obtained at full size Reynolds number will give better information to the designer than tests run at largely reduced Reynolds number. The information from the new tunnel will become more and more useful in the same degree as more results are obtained from it, so that results of new tests can be compared with results of similar older tests made under the same conditions.

Document 4-5(a-f)

(a) Ralph H. Upson to the NACA, "Attention: Mr. Victory," 19 November 1928, in Research Authorization (RA) file 270, Historical Archives, NASA Langley Research Center, Hampton, Va.

(b) Eastman N. Jacobs to Elton W. Miller, "Suggestions from Mr. R. H. Upson," undated (ca. 1 February 1929), RA file 270.

(c) Ralph H. Upson to Dr. G.W. Lewis, 19 March 1929, RA file 270.

(d) Eastman N. Jacobs to Elton W. Miller, 4 April 1929, RA file 270.

(e) G.W. Lewis to Langley Memorial Aeronautical Laboratory, "Airfoil tests suggested by Mr. R. H. Upson," 22 April 1929, RA file 270.

(f) Eastman N. Jacobs to Chief of the Aerodynamics Division (Elton W. Miller), "Airfoil testing program in Variable Density Tunnel," 17 May 1929, RA file 270.

This string of six short documents provides insight into the genesis of the NACA's famous four-digit airfoil series and suggests that an engineer outside of the NACA, Ralph H. Upson of the Aeromarine Klemm Corporation in New Jersey, played an important role in stimulating the research.

The story told by the documents begins on 19 November 1928 when Klemm sent a letter to the NACA (a) asking them to conduct a test that would provide additional information on the effect of median curvature of profile drag, and on the drag of squared ends of rectangular airfoils. Asked to respond to Upson's request, Langley engineer Eastman N. Jacobs, head of Langley's Variable-Density Tunnel section, sent an undated handwritten memo (b) to his boss, Elton W. Miller, chief of the aerodynamics division. In it, Jacobs explained how Upson's concerns could be addressed by further tests in the VDT. After receiving a response to his inquiry (a document not included) from Dr. George W. Lewis, the NACA's director of research in Washington, Upson again wrote the NACA (c), on 19 March 1929, this time directly to Lewis, with some additional questions, along with the assertion that "Of the many things that affect the so-called profile drag, thickness is surely the most important and fundamental." In particular Upson expressed serious concern that the wind-tunnel tests being proposed by the NACA engineers at Langley would fail to clarify the effect of thickness on the profile drag of an airfoil specifically; the tests, he felt, would mix up the three variables of thickness, median curvature, and type of curvature, and only leave him and others confused as to what wing section variable was actually causing a particular effect.

Again, Jacobs was brought into it, and his response (d) indicated agreement with Upson's criticism of the wind tunnel program as planned: it would be "better from both the theoretical and practical design standpoints" to treat the thickness variation and mean camber line shape as the fundamental properties of an airfoil, rather than variations in the shape of the upper and lower surfaces, the traditional emphasis.

On 22 April 1929, George Lewis approved (e) adding the testing recommended by Upson and endorsed by Jacobs to an existing NACA research authorization (no. 217), three months later to be superseded by a brand new RA (no. 290), calling for an "Investigation of Thickness and Mean Camber Line Shape on Airfoil Characteristics." In the meantime, on 17 May 1929, Jacobs, in a two-page memo to Elton Miller (f), laid out the desired airfoil testing program in the VDT and the engineering rationale for it. The last paragraph of this memo stated that the number of airfoils covered by the program might be as high as 80, the largest number of related airfoil shapes ever to be considered for a single test program up to that time.

As we will see later, Jacobs would be wrong, but only by two: the actual number would turn out to be 78.

Document 4-5(a), R.H. Upson to the NACA, "Attention: Mr. Victory," 19 November 1928, in Research Authorization (RA) file 270, Historical Archives, NASA Langley Research Center, Hampton, Va.

Aeromarine Klemm Corporation

Factory
New York Office
Keyport, N.J.
Paramount Building

Keyport, New Jersey
November 19, 1928.

National Advisory Committee for Aeronautics
3841 Navy Building
17th & B Streets, N.W.
Washington, D. C.

Attention: Mr. Victory.

Dear Sirs:

I have been making a little study on wing proportions which gives indications of very unusual value for practical design purposes. To prove the correctness of my premises, however, I am in very great need of some further information that so far I have been unable to get, viz.,

1. The effect of median curvature on profile drag.
2. The drag of square ends of rectangular airfoils.

R. & M. Report #946 gives evidence that the minimum profile drag is practically independent of median curvature, providing the latter is smooth and continuous. The experiments are confined to relatively thin sections at relatively low Reynolds numbers, however. Do you know of any experiments that have been made on thick sections at high Reynolds numbers?

I wrote once before on the subject of end drag, and also consulted Lieut. Diehl on the subject but apparently at that time there was little information available and none apparently on thick sections at high Reynolds numbers. The latter information, Item 2, is much more important than item 1. If you haven't as yet any accurate information available, don't you think that it would be worth while running a few simple experiments in the variable density tunnel to get at least two or three points on the curve? If you don't appreciate the importance of it, I think I can readily convince you of it the next time I am in Washington.

Yours very truly,
R. H. Upson

Document 4-5(b), Eastman N. Jacobs to Elton W. Miller, "Suggestions from Mr. R. H. Upson," undated (ca. 1 February 1929), RA file 270.

Memo to: Mr. Miller
Subject: Suggestions from Mr. R. H. Upson.
Reference: NACA Letter Jan. 23, 1929.

1. The two questions which Mr. Upson asked are both ones which should be answered by conducting tests in the variable density wind tunnel. The second question in regard to the drag of square ends of airfoils is, I think, the most important and also the easiest to investigate. It is likely that the scale effect on the drag of airfoil tips will be found to be large.

2. The first question, in regard to the effect of median curvature on profile drag, is a part of the more general problem of predicting airfoil characteristics. This question is considered and the existing data from tests in the variable density tunnel analyzed in the unpublished report on the above subject by G. J. Higgins. The tests to be carried out under R.A. 217 will be of value in answering such questions. Under this R.A. the upper camber is to be maintained constant and the lower varied from convex to concave thus changing the mean camber. It seems certain from disconnected tests made heretofore in the variable density tunnel, that there is a tendency for the profile drag to increase with mean camber for airfoils of a given thickness.

Eastman N. Jacobs
Assistant Aeronautical Engineer

Document 4-5(c), R.H. Upson to Dr. G.W. Lewis, 19 March 1929, RA file 270.

AEROMARINE KLEMM CORPORATION
Keyport, N.J.

March 19, 1929

Dr. G. W. Lewis
Director of Aeronautical Research
National Advisory Committee for Aeronautics
3841 Navy Building
Washington, D. C.

Dear Dr. Lewis:

In reference to your March 15th letter on profile drag, I hope you won't mind this further comment and inquiry on a subject to which I have given considerable thought, especially as my previous letter may not have been quite clear.

Of the many things that affect the so-called profile drag, thickness is surely the most important and fundamental, on account of its direct bearing on structural weight, stiffness and aspect ratio. For airfoils otherwise similar in type of curve, the thickness equation could probably be evaluated with fair accuracy from tests already made, if we knew the effect of median curvature, and the end drag for thick sections. But if several series of tests can be made it would seem better yet to make thickness the sole variable in any one series, minimizing the end drag by simply rounding the tips <u>in front elevation</u>. The end drag could also be checked by a test at small aspect ratio (say, half span).

In the tests you propose it is hard to see how you can avoid mixing up the three variables of thickness, median curvature and type or family of curve, all in one series. Of course, I appreciate that you have other objects in mind than the testing of profile drag. But for maximum utility in this respect I would strongly urge that the various tests be made susceptible of classification into groups and cross-groups, each of which involves the fewest and simplest possible variables, and that sections be included with a thickness ratio up to at least 25%.

The M series of airfoils (Report #221) are a good illustration of what I mean. The system in principle could hardly be improved upon; but unfortunately the range of thickness is insufficient for the range of modern design, and the cambers are confined to the relatively complex (though useful) reflex type.

I am trying to get off my chest here everything that might savor of criticism before the coming conference; not that you appear to mind, however, for you have always been wonderfully receptive to new ideas; also, if I am wrong I stand to be corrected.

With much appreciation for your interest.

Most sincerely,
R. H. Upson

Document 4-5(d), Eastman N. Jacobs to Elton W. Miller, 4 April 1929, RA file 270.

April 4, 1929

MEMORANDUM For Mr. Miller,

1. I am inclined to agree with Mr. Upson that the program as planned is not as good as the one which he has suggested. In our proposed program we vary the shape of the upper and lower surfaces of the airfoils, as these were once considered the fundamental properties of an airfoil. More recently there is a tendency to abandon this conception in favor of treating the thickness variation and mean camber line shape as the fundamental properties. This view is better from both the theoretical and practical design standpoints, because the shape of the mean camber line determines the angle of zero lift and pitching moment characteristics, and the thickness determines, almost independently, the drag, structure, desirable aspect ratio, structural weight, etc.

2. As previously stated, I agree with Mr. Upson about the importance of investigating the effect of tip shape and also its variation with thickness. It is reasonable to suppose that rounding the airfoil tips in front elevation will reduce the drag caused by the eddies produced by the sharp angles at the ends of the wing. This increment of profile drag should probably not be charged against the thick airfoils.

Eastman N. Jacobs,
Assistant Aeronautical Engineer

Document 4-5(e), G.W. Lewis to Langley Memorial Aeronautical Laboratory, "Airfoil tests suggested by Mr. R.H. Upson," 22 April 1929, RA file 270.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS
3841 NAVY BUILDING, 17TH AND B STREETS NW.
WASHINGTON, D. C.

April 22, 1929.

To: Langley Memorial Aeronautical Laboratory.
Subject: Airfoil tests suggested by Mr. R. H. Upson.
Reference: (a) L.M.A.L. letter EWM.B, April 4.
(b) Research Authorization No. 217.

1. Reference (a) suggested that it would be desirable that some airfoil tests be conducted under research Authorization No. 217 on the basis of varying mean cambers instead of variations in the upper and lower surfaces, the tests to be carried out in the variable-density wind tunnel.

2. It is suggested that you forward to this office your suggestions as to revision of the wording of "Brief Description of Method" of Research Authorization No. 217 to embody the modifications in the program which you suggest in reference (a).

G. W. Lewis,
Director of Aeronautical Research

Document 4-5(f), Eastman N. Jacobs to Chief of the Aerodynamics Division (Elton W. Miller), "Airfoil testing program in Variable Density Tunnel," 17 May 1929, RA file 270.

May 17, 1929.

MEMORANDUM For Chief of Aerodynamics Division.
Subject: Airfoil testing program in Variable-Density Tunnel.
Reference: (a) NACA Let. Apr. 22, 1929.
(b) Research Authorization No. 217.

1. In accordance with the request of reference (a), a revised program of airfoil tests under Research Authorization No. 217 has been considered. An examination of several airfoils which have good aerodynamic characteristics indicates that, when they are reduced to the same maximum thickness, the variation of thickness along the chord is very nearly the same. For the Göttingen 398, Clark Y, C-62, and the R.A.F. 31 the deviation of the thickness variation from the mean is less than ±2 percent. It will, therefore, be necessary to study only various combinations of maximum thickness, maximum mean camber, and mean camber line shape.

2. My suggestions as to a suitable program are embodied in the following rewording of "Brief Description of Method" of Research Authorization No. 217. If it should be considered to change research Authorization No. 217, it is suggested that a program similar to this should be carried out under a new research authorization. However, it is believed that the research here outlined is so extensive that the one outlined under Research Authorization No. 217 is unnecessary, especially in view of the similar research, already completed, on the Navy propeller sections.

3. A family of airfoils is to be developed, all having the same relative variation in thickness along the chord, but having five values of the maximum thickness: 6, 9, 12, 15, 18, and 21 per cent of the chord. The thickness variation is to be chosen so that it will be similar to that of the best airfoils which have been developed in the past. The airfoils are to be formed by thickening four types of mean camber lines as follows: a straight line, three circular arcs, six curves having their maximum ordinate at four-tenths of the chord behind the leading edge and three curves having their maximum ordinate at three-tenths of the chord behind the leading edge. These airfoils are to be constructed of metal and tested in the Variable Density Tunnel at 1 and 20 atmospheres. The results are to be analyzed with a view to establishing the relation of the thickness and mean camber line to the aerodynamic characteristics of an airfoil.

4. The program, as outlined above, requires the testing of about eighty airfoils. A job order to cover further study of the development of such a family has been requested. This study may indicate that it will be unnecessary to investigate all of the eighty airfoils.

Eastman N. Jacobs,
Assistant Aeronautical Engineer

Document 4-6

Eastman N. Jacobs, Kenneth E. Ward, and Robert M. Pinkerton, "The Characteristics of 78 Related Airfoil Sections from Tests in the Variable-density Wind Tunnel," NACA *Technical Report* 460 (Washington, 1933).

Besides providing data on a highly efficient series of new wing sections, *Technical Report 460* formally introduced the NACA's ingenious new way of numerically coding its airfoils. Devised by Eastman Jacobs with help from his closest associates, and patterned after a similar system used to identify the composition of steel alloys, the code literally enumerated an airfoil shape. Like all other aerodynamical laboratories, Langley until then had designated airfoils simply by numbering them in the sequence in which they had been tested (as in Munk's M-1, M-2, M-3, and so forth). In the new system, four numbers indicated the airfoil section's critical geometrical properties—thus the name the "four-digit" series. The first integer represented the maximum mean camber in percent of the chord; the second integer represented the position of the maximum mean camber in tenths of the chord from the leading edge; and the last two integers represented the maximum thickness in percent of the chord. Thus, airfoil "N.A.C.A. 2415" was a wing section having 2 percent camber at 0.4 of the chord from the leading edge, with thickness 15 percent of the chord. Zeroes were used for the first two integers when the section was symmetrical, as was the case of N.A.C.A. 0015.

In this simple graphic way, the NACA's numerical designation of wing profiles provided a wonderful shorthand statement of the values of the three critical airfoil parameters: the height and chordwise location of the uppermost point of the camber line and the magnitude of the maximum thickness. From the time TR 460 appeared in print, one could say, for instance, "N.A.C.A. 2415," and a complete airfoil shape would appear in any aerodynamicist's mind's eye. Reminding as much as instructing, the NACA's airfoil report complemented the coded information with graphic illustrations of two independent sets of curves. These curves communicated knowledge basic to an engineer's understanding of the relationships among an airfoil's variables. Pictorial representation of airfoil data – the outline of the physical shape reinforced by performance curves and the digital code – gave aeronautical engineers ready access to the wide range of parametric data necessary to their work. The NACA's digest gave them "a whole range of wings from which to choose, the way one might select home furnishings or automobile accessories from a catalog" (Alex Roland, *Model Research: The National Advisory Committee for Aeronautics, 1915-1958*, NASA SP-4103 (Washington, 1985) 2: 539-40).

Document 4-6, Eastman N. Jacobs, Kenneth E. Ward, and Robert M. Pinkerton, "The Characteristics of 78 Related Airfoil Sections from Tests in the Variable-density Wind Tunnel," NACA **Technical Report** *460 (Washington, 1933).*

REPORT No. 460
THE CHARACTERISTICS OF 78 RELATED AIRFOIL SECTIONS FROM TESTS IN THE VARIABLE-DENSITY WIND TUNNEL

By EASTMAN N. JACOBS, KENNETH E. WARD, and ROBERT M. PINKERTON

SUMMARY

An investigation of a large group of related airfoils was made in the N.A.C.A. variable-density wind tunnel at a large value of the Reynolds number. The tests were made to provide data that may be directly employed for a rational choice of the most suitable airfoil section for a given application. The variation of the aerodynamic characteristics with variations in thickness and mean-line form were therefore systematically studied.

The related airfoil profiles for this investigation were developed by combining certain profile thickness forms, obtained by varying the maximum thickness of a basic distribution, with certain mean lines, obtained by varying the length and the position of the maximum mean-line ordinate. A number of values of these shape variables were used to derive a family of airfoils. For the purposes of this investigation the construction and tests were limited to 68 airfoils of this family. In addition to these, several supplementary airfoils have been included in order to study the effects of certain other changes in the form of the mean line and in the thickness distribution.

The results are presented in the standard graphic form representing the airfoil characteristics for infinite aspect ratio and for aspect ratio 6. A table is also given by means of which the important characteristics of all the airfoils may be conveniently compared. The variation of the aerodynamic characteristics with changes in shape is shown by additional curves and tables. A comparison is made, where possible, with thin-airfoil theory, a summary of which is presented in an appendix.

INTRODUCTION

The forms of the airfoil sections that are in common use today are, directly or indirectly, the result of investigations made at Göttingen of a large number of airfoils. Previously, airfoils such as the R.A.F. 15 and the U.S.A. 27, developed from airfoil profiles investigated in England, were widely used. All these investigations,

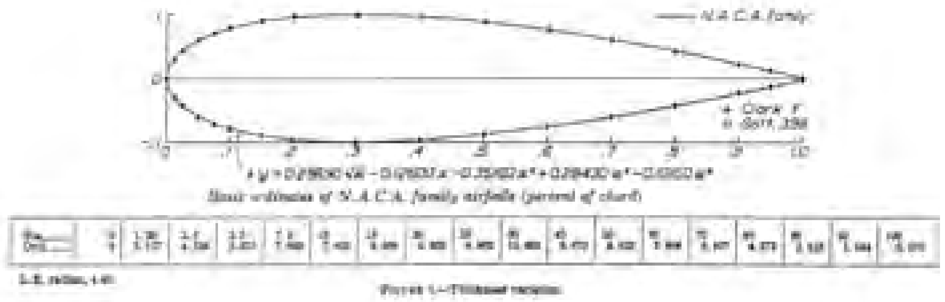
however, were made at low values of the Reynolds Number; therefore, the airfoils developed may not be the optimum ones for full-scale application. More recently a number of airfoils have been tested in the variable-density wind tunnel at values of the Reynolds Number approaching those of flight but, with the exception of the M-series and a series of propeller sections, the airfoils have not been systematically derived in such a way that the results could be satisfactorily correlated.

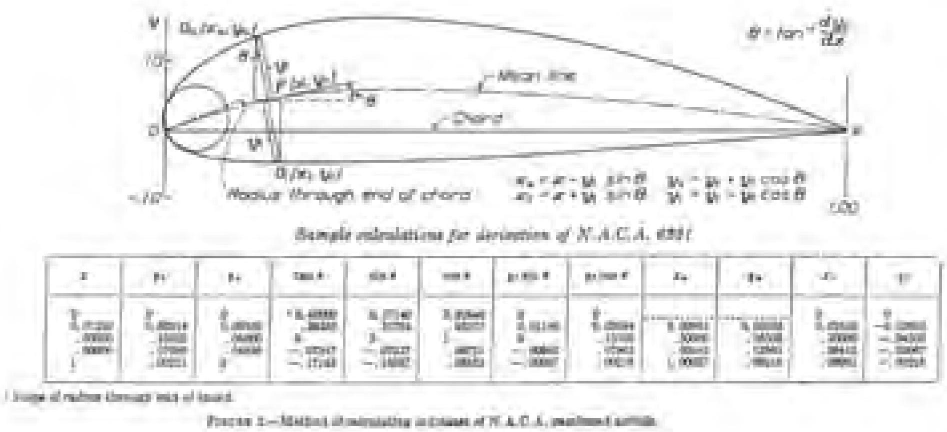
The design of an efficient airplane entails the careful balancing of many conflicting requirements. This statement is particularly true of the choice of the wing. Without a knowledge of the variations of the aerodynamic characteristics of the airfoil sections with the variations of shape that effect the weight of the structure, the designer cannot reach a satisfactory balance between the many conflicting requirements.

The purpose of the investigation reported herein was to obtain the characteristics at a large value of the Reynolds Number of a wide variety of related airfoils. The benefits of such a systematic investigation are evident. The results will greatly facilitate the choice of the most satisfactory airfoil for a given application and should eliminate much routine airfoil testing. Finally, because the results may be correlated to indicate the trends of the aerodynamic characteristics with changes of shape, they may point the way to the design of new shapes having better characteristics.

Airfoil profiles may be considered as made up of certain profile-thickness forms disposed about certain mean lines. The major shape variables then become two, the thickness form and the mean-line form. The thickness form is of particular importance from a structural standpoint. On the other hand, the form of the mean line determines almost independently some of the most important aerodynamic properties of the airfoil section, e.g., the angle of zero lift and the pitching-moment characteristics.

The related airfoil profiles for this investigation were derived by changing systematically these shape variables. The symmetrical profiles were defined in terms of a basic thickness variation, symmetrical airfoils of varying thickness being obtained by the application of factors to the basic ordinates. The cambered profiles were then developed by combining these thickness forms with various mean lines. The mean lines were obtained by varying the camber and by varying the shape of the mean line to alter the position of the maximum mean-line ordinate. *The maximum ordinate of the mean line is referred to throughout this report as the camber of the airfoil and the position of the maximum ordinate of the mean line as the position of the camber. An airfoil, produced as described above, is designated by a number of four digits: the first indicates the camber in percent of the chord; the second, the position of the camber in tenths of the chord from the leading edge; and the last two, the maximum thickness in percent of the chord.* Thus the N.A.C.A. 2315 airfoil has a maximum camber of 2 percent of the chord at a position 0.3 of the chord from the leading edge, and a maximum thickness of 15 percent of the chord; the N.A.C.A. 0012 airfoil is a symmetrical airfoil having a maximum thickness of 12 percent of the chord.





In addition to the systematic series of airfoils, several supplementary airfoils have been included in order to study the effects of a few changes in the form of the mean line and in the thickness distribution.

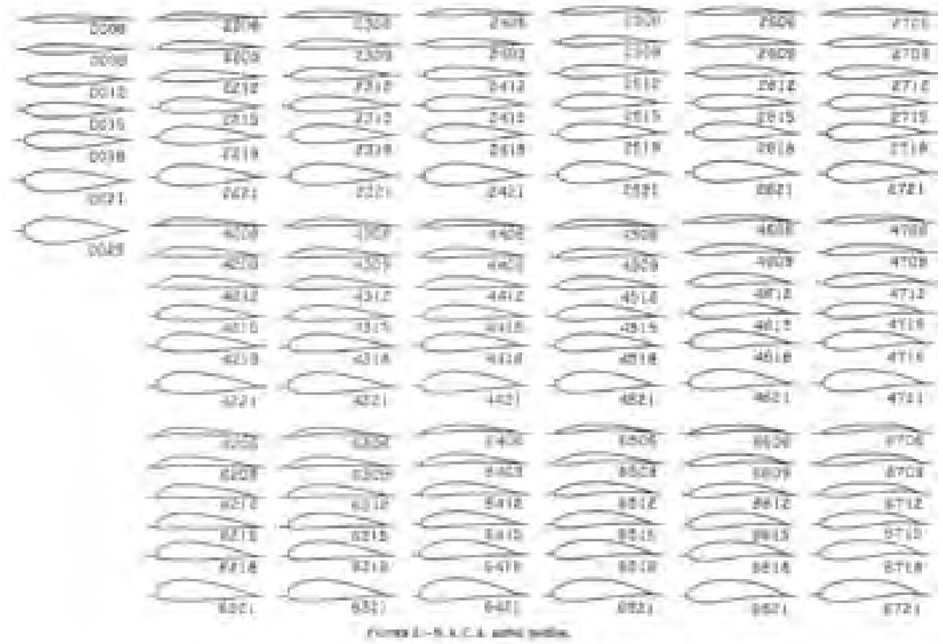
Preliminary results which have been published include those for 12 symmetrical N.A.C.A. airfoils, the 00 series and other sections having different nose shapes; and those for 42 cambered airfoils, the 43 and 63 series, the 45 and 65 series, the 44 and 64 series, and the 24 series.

The tests were made in the variable-density wind tunnel of the National Advisory Committee for Aeronautics during the period from April 1931 to February 1932.

DESCRIPTION OF AIRFOILS

Well-known airfoils of a certain class including the Göttingen 398 and the Clark Y, which have proved to be efficient, are nearly alike when their camber is removed (mean line straightened) and they are reduced to the same maximum thickness. A thickness variation similar to that of these airfoils was therefore chosen for the development of the N.A.C.A. airfoils. An equation defining the shape was used as a method of producing fair profiles.

When the mean lines of certain airfoils in common use were reduced to the same maximum ordinate and compared it was found that their shapes were quite



different. It was observed, however, that the range of shapes could be well covered by assuming some simple shape and varying the maximum ordinate and its position along the chord. The mean line was, therefore, arbitrarily defined by two parabolic equations of the form

$$y_c = b_0 + b_1 x + b_2 x^2$$

where the leading end of the mean line is at the origin and the trailing end is on the x axis at $x = 1$.

A family of related airfoils was derived. Seven values of the maximum thickness, 0.06, 0.09, 0.12, 0.15, 0.18, 0.21, and 0.25; four values of the camber, 0.00, 0.02, 0.04, and 0.06; and six values of the position of the camber, 0.2, 0.3, 0.4, 0.5, 0.6, and 0.7 were used to derive the related sections of this family. The profiles of the airfoils derived are shown collectively in figure 3.

For the purposes of this investigation the construction and tests were limited to 68 of the airfoils. Tables of ordinates at the standard stations are given in the figures presenting the aerodynamic characteristics. These ordinates were obtained graphically from the computed ordinates for all but the symmetrical sections. Two sets of trailing-edge ordinates are given. Those enclosed by parentheses, which are given to facilitate construction, represent ordinates to which the surfaces are faired. In the construction of the models the trailing edges were rounded off.

Three groups of supplementary airfoils were also constructed and tested. The derivation of these airfoils will be considered later with the discussion.

APPARATUS AND METHODS

A description of the variable-density wind tunnel and the method of testing is given in reference 8. [N.A.C.A. TR 416, not included herein. See chapter 2 for a description of the variable-density tunnel.] The models, which are made of duralumin, have a chord of 5 inches and a span of 30 inches. They were constructed from the computed ordinates by the method described in reference 8.

Routine measurements of lift, drag, and pitching moment about a point on the chord one quarter of the chord behind its forward end were made at a Reynolds Number of approximately 3,000,000 (tank pressure, approximately 20 atmospheres). Groups of airfoils were first tested to study the variations with thickness, each group containing airfoils of different thicknesses but having the same mean line. Finally, all airfoils having a thickness of 12 percent of the chord were tested to study the variations with changes in the mean line.

RESULTS

The results are presented in the standard graphic form (figs. 4 to 8) as coefficients corrected after the method of reference 8 to give airfoil characteristics for infinite aspect ratio and aspect ratio 6. Where more than one test has been used for the analysis, the infinite aspect ratio characteristics from the earlier test have been indicated by additional points on the figure. Table I gives the important characteristics of all the airfoils.

CONCLUSIONS

The variation of the aerodynamic characteristics of the related airfoils with the geometric characteristics investigated may be summarized as follows:

Variation with thickness ratio:

1. The slope of the lift curve in the normal working range decreases with increased thickness, varying from 95 to 81 percent, approximately, of the theoretical slope for thin airfoils (2π per radian).

2. The angle of zero lift moves toward zero with increased thickness (above 9 to 12 percent of the chord thickness ratios).

3. The highest values of the maximum lift are obtained with sections of normal thickness ratios (9 to 15 percent).

4. The greatest instability of the air flow at maximum lift is encountered with the moderately thick, low-cambered sections.

5. The magnitude of the moment at zero lift decreases with increased thickness, varying from 97 to 64 percent, approximately (for normally shaped airfoils), of the values obtained by thin-airfoil theory.

6. The axis of constant moment usually passes slightly forward of the quarter-chord point, the displacement increasing with increased thickness.

7. The minimum profile drag varies with thickness approximately in accordance with the expression

$$C_{D_0 min} = k + 0.0056 + 0.01t + 0.1t^2$$

where the value of k depends upon the camber and t is the ratio of the maximum thickness to the chord.

8. The optimum lift coefficient (the lift coefficient corresponding to the minimum profile-drag coefficient) approaches zero as the thickness is increased.

9. The ratio of the maximum lift to the minimum profile drag is highest for airfoils of medium thickness ratios (9 to 12 percent).

Variation with camber:

1. The slope of the lift curve in the normal working range is little affected by the camber; a slight decrease in the slope is indicated as the position of the camber moves back.

2. The angle of zero lift is between 100 and 75 percent, approximately, of the value given by thin-airfoil theory, the smaller departures being for airfoils with the normal camber positions.

3. The maximum lift increases with increased camber, the increase being more rapid as the camber moves forward or back from a point near the $0.3c$ position.

4. Greater stability of the air flow at maximum lift is obtained with increased camber if the camber is in the normal positions ($0.3c$ to $0.5c$).

5. The moment at zero lift is nearly proportional to the camber. For any given thickness, the difference between the experimental value of the constant of proportionality and the value predicted by thin-airfoil theory is not appreciably affected by the position of the camber except for the sections having the maximum camber well back, where the difference becomes slightly greater.

6. The axis of constant moment moves forward as the camber moves back.

7. The minimum profile drag increases with increased camber, and also with a rearward movement of the camber.

8. The optimum lift coefficient increases with the camber and for the highly cambered sections a definite increase accompanies a forward movement of the camber.

9. The ratio of the maximum lift to the minimum profile drag tends to decrease with increased camber (above 2 percent of the chord) and with a rearward movement of the camber (for the highly cambered sections).

Document 4-7(a-e)

(a) C.W. Howard, Major, Air Corps, and Chief, Engineering Section, Materiel Division, Office of the Chief of the Division, Wright Field, Dayton, Oh., 13 Jan. 1933, in RA file 290, LHA, Hampton, Va.

(b) Montgomery Knight, Director, Daniel Guggenheim School of Aeronautics, Georgia School of Technology, Atlanta, Ga., to George W. Lewis, NACA, Navy Building, Washington, D.C., 19 Jan. 1932, in Research Authorization (RA) file 290, LHA, Hampton, Va.

(c) Edward P. Warner, Editor, *Aviation*, New York, N.Y., to George W. Lewis, NACA, 20 Jan. 1933, in RA file 290, LHA, Hampton, Va.

(d) G.W. Lewis to Dr. Joseph Ames, Johns Hopkins University, Baltimore, Md., 10 June 1933, in RA file 290, LHA, Hampton, Va.

(e) "Possible Saving by Use of N.A.C.A. 2415 Airfoil," undated (ca. summer 1932), in RA file 290, LHA, Hampton, Va.

This string of five short letters from 1933 presents some of the extremely favorable immediate reaction to the NACA's release of the four-digit airfoil data. On 13 January 1933, Major C.W. Howard, chief of the engineering section of the Army Air Corps at Wright Field, wrote a letter to Lewis (a), after receiving a preliminary version of what will become TR 460, applauding its "great value to the designer." The same week, Professor Montgomery Knight, director of the Daniel Guggenheim School of Aeronautics at what was then called the Georgia School of Technology in Atlanta (Georgia Tech), wrote favorably to the NACA's George Lewis after reading advanced copies of two papers by Eastman Jacobs, one of them a preliminary version of what would become TR 460. In his letter (b) Knight, a former NACA Langley engineer, expressed with confidence that "the tests on this extensive series should bring in a new era into the choice of airfoils for different purposes." The next day Lewis received a letter from Edward P. Warner (c) in which the distinguished editor of *Aviation* magazine called the airfoil report "a perfectly marvelous job." By mid-year, Lewis had received so many congratulatory letters about the new airfoils, including several from the aircraft industry, that in sending out copies of the final report, he boasted (d) to Dr. Joseph Ames, NACA Chairman and physics professor at Johns Hopkins University, that "Mr. Jacobs' report is the most extensive and valuable report of this character that has so far been published."

Finally, in preparation for congressional hearings on the NACA's 1934 budget estimates, the NACA staff prepared a statement (e) enclosed in this chapter's documentary collection entitled, "POSSIBLE SAVING BY USE OF N.A.C.A. 2415 AIRFOIL." This brief item estimated that, by changing from the Göttingen 387 airfoil currently employed to the N.A.C.A. 2415, a single "typical" airplane such as the Fairchild FC-2W2 could, over the course of a million hours flying time, save as much as $630,000 per year.

The NACA bureaucrats who fought the political and budgetary battles in Washington certainly were not beyond gamesmanship and hype in calculating the value of NACA research contributions, but no one can dispute the unparalleled achievement of the airfoil work and how much it helped the American aircraft industry in its design of wings. In his 1941 textbook on the *Aerodynamics of the Airplane* (New York: John Wiley & Sons, Inc.), Dr. Clark B. Millikan, Caltech professor of aeronautics, past president of the Institute of the Aeronautical Sciences, son of distinguished scientist Robert Millikan, and an individual with not terribly strong connections to the NACA, asserted (p. 67) that, "Since about 1935 systematic families of airfoils developed by the N.A.C.A. have been almost universally used in this country," not to mention significant use of them abroad. Whether or not any particular airfoil application saved so much money for a single airplane, in retrospect it seems more than clear that the value of the NACA airfoil families to aerodynamic efficiency, all told, had to be worth countless tens of millions.

Document 4-7(a), C.W. Howard, Major, Air Corps, and Chief, Engineering Section, Materiel Division, Office of the Chief of the Division, Wright Field, Dayton, Oh., 13 Jan. 1933, in RA file 290, LHA, Hampton, Va.

WAR DEPARTMENT
AIR CORPS
Materiel Division
Office of the Chief of Division
Wright Field, Dayton, Ohio

January 13, 1933

Dr. G. W. Lewis
National Advisory Committee for Aeronautics
3841 Navy Building, 17th and B Sts., N.W.
Washington, D.C.

Dear Dr. Lewis:

The preliminary copy of your report "The Characteristics of 78 Related Airfoil Sections from Tests in the Variable-Density Wind Tunnel," by E. N. Jacobs, K. E. Ward, and R. M. Pinkerton, was studied with interest and the following comments are made.

This report is considered a very comprehensive systematic study of airfoil sections of rectangular planform. It is believed that an addition of similar discussions of effects, due to tapering in the plan view of the wing and due to interferences caused by locations of engine nacelles, fuselage, and the slipstream, which already have been published, would be of great value to the designer in search of the best wing combination for any particular case.

Very truly yours,

C. W. Howard, (Signed) A. J. Lyon
Major, Air Corps,
Chief, Engineering Section

Document 4-7(b), Montgomery Knight, Director, Daniel Guggenheim School of Aeronautics, Georgia School of Technology, Atlanta, Georgia, to George W. Lewis, NACA, Navy Building, Washington, D.C., 19 January 1932, in Research Authorization (RA) file 290, LHA, Hampton, Va.

GEORGIA SCHOOL OF TECHNOLOGY
Atlanta, Georgia
Daniel Guggenheim School of Aeronautics

January 19, 1932 (Note: This has to be 1933.)

Mr. George W. Lewis,
National Advisory Committee for Aeronautics,
Navy Building,
Washington, D.C.

Dear Mr. Lewis:

Thank you very much indeed for your kindness in sending the advanced copies of the two papers by Jacobs. I think he is to be complimented on his success in flattening the normal force curve for the Göttingen 398 by the simple expedient of sharpening the leading edge. It would be very interesting to see the results of autorotation tests or flight spinning tests with such a modified profile, particularly, to find out over how much of the span this modification would be necessary. I have an idea that improved efficiency could be obtained by having the sharpened leading edge extend inward from the tips, approximately to the inboard edge of the aileron. However, this is an obvious conclusion and I am sure it has occurred to you already. I shall look forward with interest to the results of further tests on this development.

The new N.A.C.A. 24-12 airfoil is a very good looking one and the characteristics are no less satisfactory. I am sure that the tests on this extensive series should bring about a new era into the choice of airfoils for different purposes. I, myself, intend to use this series almost exclusively in our work.

With kindest regards, I am

Yours sincerely,

MONTGOMERY KNIGHT
Director

Document 4-7(c), Edward P. Warner, Editor,* Aviation*, New York, N.Y., to George W. Lewis, NACA, 20 January 1933, in RA file 290, LHA, Hampton, Va.

AVIATION
330 West 42nd Street
New York, N.Y.

January 20, 1933

Dr. George W. Lewis,
National Advisory Committee for Aeronautics,
Navy Building,
Washington, D.C.

Dear George:

I have the report on The Characteristics of 76 Related Airfoil Sections, and my first observation has to be that it is a perfectly marvelous job. I have been waiting for it with eager anticipation for a long time, and the results fully justify my eagerness and expectancy. Please give my very warm congratulations to the authors.

With your leave I am going to do what I have so often done without anybody's leave, and keep this report in my file. I am quite unwilling to lose the opportunity of referring to the results of the research until such time as the Government Printing Office shall have put out the finished version.

While I haven't perhaps gone through the text of the report with the minute care that I have given to a few of them, a rather hasty examination leaves me thinking that it has been handled just about right. The curves in my copy were unfortunately rather badly printed, and are extremely difficult to study, but my only criticism at the moment would be that there has not been quite enough relative attention to the sort of presentation which the engineer without laboratory experience and without a profound knowledge of wing theory can use directly. I wouldn't want to sacrifice any of the curves that bear on the relationship between the laboratory results and the fundamental theory (such, for example, as Fig. 84), but I would like to suggest that there ought to be some more plotting of the characteristics that the engineer uses directly against the geometrical characteristics of the sections (such plotting as is done, for instance, in Figs. 85, 95 and 96). Aside from that I have nothing to suggest, but I do urgently hope the report can be brought out in the near future. I hope, also, that you can let me know at least a month ahead of time when it is to be published, and let me use either the present copy or a proof of the report in working up an article on airfoil characteristics in light of your studies. I should like to boil the whole thing down to a couple of pages, but obviously in order that such a presentation may be of interest it must come out practically simultaneously with the report itself, so that engineers will have a chance to look over the summary in the magazine before the report has come into our hands, been glanced at, and put aside.

Very sincerely,
E. P. W.
Edward P. Warner,
Editor

Document 4-7(d), G.W. Lewis to Dr. Joseph Ames, Johns Hopkins University, Baltimore, Md., 10 June 1933, in RA file 290, LHA.

June 10, 1933

Dear Doctor Ames:

I am forwarding herewith another report which I think the Aeronautical Research Committee will appreciate having. This is a report prepared by Mr. Jacobs on a group of seventy-eight related airfoils.

I noted in one of the British Reports that they are planning a very extensive investigation of airfoil characteristics in their new compressed-air wind tunnel. In connection with this proposed investigation the attached report will be of special interest. I personally feel that Mr. Jacobs' report is the most extensive and valuable report of this character that has so far been published. Certainly airfoils of the 2490 and 2200 series are now being used by manufacturers, especially the 2412 and 2212.

Sincerely yours,

G. W. Lewis
Director of Aeronautical Research

Dr. Joseph S. Ames,
Johns Hopkins University,
Baltimore, Maryland

Document 4-7(e), "Possible Saving by Use of N.A.C.A. 2415 Airfoil," undated (ca. summer 1932), in RA file 290, LHA.

POSSIBLE SAVING BY USE OF N.A.C.A. 2415 AIRFOIL.

Considering as typical the Fairchild FC-2W2, a drag or air resistance reduction may be accomplished by changing to a more efficient airfoil section. Applying the results of the N.A.C.A. airfoil tests, the N.A.C.A. 2415 section is found to be a more suitable section of the same thickness for this airplane than the present Göttingen 387 section. The direct reduction in drag due to the change in section is represented by the drag coefficient 0.0006. At the cruising speed of 100 m.p.h. this coefficient represents a drag saving of

0.0006 x 336 x 25.58

or approximately 5 pounds.

In addition to this drag reduction, the lower pitching moment of the N.A.C.A. 2415 allows a weight saving in the structure and smaller tail surfaces. From Technical Note 340, the reduced drag resulting from the use of smaller tail surfaces is estimated at 4 pounds. It is estimated that another reduction in drag of 3 pounds (20 pounds weight) would result indirectly from the weight saving, making the total reduction in drag of 12 pounds. This figure is considered a fair average for all airplanes.

The N.A.C.A. 2415 has a lower drag coefficient and a lower pitching moment than airfoils used on present day aircraft. Its substitution would result in a saving of approximately 12 pounds drag at 100 m.p.h. for most all airplanes flying during the past fiscal year.

$.035 =	cost of drag per lb. per hour				
428,930=	hours flown by commercial airplanes				
247,745=	"	"	"	Navy	"
371,254=	"	"	"	Army	"
1,047,929=	"	"	"	all	"

12 x .035 x 1,047,929 = $440,000.

Since cruising speed is higher than 100 m.p.h. in most cases and in the neighborhood of 120 m.p.h. we have

12 x (120^2 / 100^2) = 17.28 lbs. or

17.28 x .035 x 1,047,929 = $630,000.

Document 4-8(a-b)

(a) Charles H. Chatfield, Research Division, United Aircraft Corporation, East Hartford, Connecticut, to Dr. G.W. Lewis, NACA, 8 April 1937, in Research Authorization (RA) file 290, Historical Archives, NASA Langley Research Center, Hampton, Va.

(b) G.W. Lewis to Charles H. Chatfield, United Aircraft Corporation, 1 May 1937, RA file 290, Historical Archives, NASA Langley Research Center, Hampton, Va.

This pair of letters between the director of NACA research and head of United Aircraft's research division, provoked by the appearance of the NACA's new forward-camber airfoils, is highlighted by a brief but interesting exchange about the best way to make such a large volume of research data about different airfoils more useful and convenient to the airplane designer. Chatfield wanted the NACA to spotlight for industry only the more promising wing sections; Lewis believed that the requirements for wing sections varied so much that it would be impossible for the NACA to pre-select the airfoils that the industry would find useful.

Chatfield made specific reference to "N.A.C.A. 23012," certainly one of the best of the new forward-camber airfoils. Obviously, this was a <u>five</u>-digit airfoil. In this code, the first number still indicated the maximum camber in percent of the chord and the last two numbers still indicated the maximum thickness in percent of the chord; however, the middle two numbers indicated the position of maximum camber in percent of the chord rather than the previous single number in the four-digit series indicating maximum camber in tenths of the chord. Furthermore, the five-digit (and subsequent six-digit) series also indicated modifications like changes of the leading-edge radius or the position of maximum thickness by adding a suffix consisting of a dash and two more digits, as with N.A.C.A. 23012-64, one of the most outstanding sections in the popular 230-series, the family announced in 1935.

Eastman Jacobs explained this extended numbering system and summarized the advantages of the best new forward-camber airfoils in TR 610 of 1937, "Tests of Related Forward-Camber Airfoils in the Variable-Density Wind Tunnel," co-authored by Robert M. Pinkerton and Harry Greenburg.

Document 4-8(a), Charles H. Chatfield, Research Division, United Aircraft Corporation, East Hartford, Connecticut, to Dr. G.W. Lewis, NACA, 8 April 1937, in Research Authorization (RA) file 290, LHA, Hampton, Va.

UNITED AIRCRAFT CORPORATION
East Hartford, Connecticut
Research Division

April 8, 1937

Dr. G. W. Lewis
National Advisory Committee for Aeronautics
Navy Building
Washington, D.C.

Dear Dr. Lewis:

Thanks very much for the report "Tests of Related-Forward-Camber Airfoils in the Variable-Density Wind Tunnel" that you sent me with your letter of March 11th. It was interesting to see that the 23012 airfoil was not a freak case, but rather the best of a good strain. As a former stress analyst, I agree with the view expressed on page 11 of the report that these new airfoils are good structurally as well as aerodynamically. In the old days the sharp taper aft of the maximum ordinate was always troublesome.

Now that your extensive studies of airfoils have produced so many good ones, I think the time is approaching when a designer might lose sight of some of them. I, therefore, venture the suggestion that you issue a report giving the characteristics of the better of the airfoils in the various groups, so that a designer may have in one publication all the airfoils that he would be likely to consider seriously for any particular airplane. From the research point of view, it is certainly desirable to have available the characteristics of all the airfoils tested, but I doubt that the airfoils which were only steps in the development are of great interest to the practicing designer.

Yours very truly,
[Signed Chat]
Charles H. Chatfield

Document 4-8(b), G.W. Lewis to Charles H. Chatfield, United Aircraft Corporation, 1 May 1937, RA file 290, LHA.

May 1, 1937

Mr. Charles H. Chatfield,
United Aircraft Corporation,
East Hartford, Connecticut

Dear Chat:

I appreciate very much your letter of April 8 commenting on the report on forward-camber airfoils.

With reference to your suggestion about the desirability of presenting in one report data on the better airfoils in the various groups, the chief difficulty is the question of the method of selecting the airfoils to be presented. The determination of a basis for such a selection appears so difficult as to make the preparation of such a report of doubtful practical ability.

Of course the report on the forward-camber airfoils and the Committee's Technical Report No. 460 present in rather compact and easily usable form complete data on the airfoils used in the Committee's investigation.

There is now in preparation a report presenting the characteristics of a large number of miscellaneous airfoils and it is believed that these three reports will make available for ready use information on all the desirable airfoil sections investigated in the variable-density wind tunnel.

Sincerely yours,

G.W. Lewis
Director of Aeronautical Research

Document 4-9(a-b)

(a) G.W. Lewis, NACA, To The Members of the Board of Award, Sylvanus Albert Reed Award for 1937, 3 September 1937, in Research Authorization (RA) file 290, LHA, Hampton, Va.

(b) R.C. Platt, Memorandum for Dr. Lewis, "Airfoil sections employed for wings of modern airplanes," 2 September 1937, RA file 290, LHA.

Winning the Institute for Aeronautical Science's prestigious Sylvanus Albert Reed Award for 1937 represented not only the zenith of Eastman Jacobs' professional career but perhaps also the absolute highpoint of the NACA's reputation in aerodynamics—at least up to the point it won a trio of Collier Trophies for major aerodynamic breakthroughs (the X-1 breaking the sound barrier, the slotted-throat transonic wind tunnel, and the area rule) in the late 1940s and early 1950s.

Nominating a deserving NACA researcher for a prestigious national and international award was something that Lewis and other NACA officials started paying more attention to starting in the mid-1930s. The publicity surrounding the winning of scientific and engineering awards enhanced the NACA's reputation at just the time that the NACA was campaigning for significant funds for new construction. In particular, there was a growing concern over developments in Europe. In November 1935 the NACA's intelligence officer in Paris, John Jay Ide, reported that French had just completed a full-scale wind tunnel at Chalais-Meudon; the Italians had built an entire city, Guidonia, outside Rome, devoted to high-speed aeronautical research; and the Germans were in the midst of what appeared to be a major revitalization of their aeronautical resources. As a result of Nazi support, there would soon be five major regional stations for aeronautical research and development in Germany and a central establishment, the Deutsche Versuchsanstalt fur Luftfahrt (DVL) at Aldershof near Berlin. This news disturbed George Lewis so much that in the late summer of 1936 he crossed the Atlantic in the German airship *Hindenburg* in order to see for himself what the Europeans were doing. He visited England and France, but his real mission was to tour major aeronautical installations in Germany and Russia. In Germany he visited the Air Ministry in Berlin, the DVL, the Heinkel aircraft factory at Oranienbaum, and the University of Göttingen; in Russia, he concentrated on the operations of Moscow's Central Aerodynamic Institute.

Lewis came away alarmed over the warlike aspect of the expanded research programs in both Germany and Russia. The DVL at Aldershof looked to him "like a construction camp" being readied for experiments "with every conceivable device."

He estimated that between 1600 and 2000 well trained employees were working there, compared with only 350 at Langley. Although he still considered Langley "the single best and biggest aeronautical research complex in the world," he warned the government about the dangers of complacency (Lewis, "Report on Trip to Germany and Russia, September-October 1936," Langley Correspondence Files, Code E32-12, RG 255, National Archives, Philadelphia).

Lewis's were not the only warnings in those troubled times, but his were among those that paid off, preparing the way for the great expansion of NACA facilities undertaken in the years 1938 to 1941. By the time of the Japanese attack at Pearl Harbor, construction was nearing completion on two new separate NACA laboratories: Ames Aeronautical Laboratory near Palo Alto, California, and the Aircraft Engine Research Laboratory in Cleveland, Ohio (later renamed in honor of Lewis). Without these new laboratories, the NACA would never have been able to accomplish the tremendously increased workload brought on by the war.

Awards like Jacobs' were hardly enough to convince Congress that NACA merited more funding, but they certainly did not hurt. This is not to suggest that George Lewis nominated Jacobs or any other outstanding researcher for an award only so that the larger organization would get something out of it. Lewis truly enjoyed and admired "his boys" at Langley and wanted them to receive all the praise and reward they deserved—especially Jacobs, who he especially liked for his outstanding productivity and unselfish devotion to his work.

Interestingly, Jacobs' bold vision of what was possible technologically, a more and more freewheeling style, and a rather libertine personal lifestyle, would later try the patience of George Lewis to the breaking point. Lewis in the late 1930s not only had to order Jacobs to dismantle a primitive (and completely unauthorized) thermonuclear fusion reactor that Jacobs and an associate (Arthur Kantrowitz) had constructed in the VDT building, and later to keep a lid on his passion for a hybrid type of jet engine that he helped to design in the early 1940s, he eventually had to quietly encourage his resignation from the NACA. Along with his increasingly irresponsible and rebellious ways at the laboratory, Jacobs' personal life suffered from a series of scandals during the war that alienated many of his co-workers and finally convinced Lewis that NACA might be better off without him.

Seven years after winning the Sylvanus Albert Reed Award, the NACA's golden boy retired from government service to do independent consulting work back in his home state of California. He produced no technical work of any great worth after his departure from the NACA, and within the world of aeronautics turned into a shadowy figure of legend and mystery. Some stories had him running a hot dog stand at the beach in Malibu. Whatever actually happened to Jacobs is known only to his closest friends and family. But his demise was tragic, for at age 42 his outstanding career essentially ended.

Document 4-9(a), G.W. Lewis, NACA, To The Members of the Board of Award, Sylvanus Albert Reed Award for 1937.

September 3, 1937

To The Members of the Board of Award,
Sylvanus Albert Reed Award for 1937:

I respectfully submit for consideration for the Sylvanus Albert Reed Award for 1937, the name of Eastman Nixon Jacobs.

Mr. Jacobs has been responsible for a number of years for the operation of the variable-density wind tunnel of the National Advisory Committee for Aeronautics, and was responsible also for the initiation of research on the special problems of airfoil sections for use in the design of airplanes. As a result of this research, he has developed what are acknowledged to be the most efficient airfoil sections now in existence.

The importance of Mr. Jacobs contribution is evidenced by the fact that the most generally successful and widely used airplanes are those which employ airfoils of the later N.A.C.A. series, which have been developed under his direction. Among these airplanes the highly successful Douglas, Lockheed, and Sikorsky transports; the Northrop and Consolidated military airplanes; and the Beechcraft, Fairchild, and Cessna airplanes of the lighter commercial type.

In the military field the airplanes which are now being produced in large quantities, such as the Douglas bomber, which uses the N.A.C.A. 2200 series, and the Boeing bomber, employ airfoils of N.A.C.A. series.

Mr. Jacobs' contribution in the development of more efficient airfoils is one of the outstanding factors that has made possible the superior performance of large present-day airplanes, both military and commercial.

Mr. Jacobs is unselfishly devoted to his work, and has not only contributed his own personal time and energies to his research activities, but has also given freely of his time in discussing the problems of airfoil design and in acquainting the engineers of the industry with the results of his investigations.

I am attaching hereto a list of present-day airplanes which use airfoils of N.A.C.A. series.

Respectfully,
G. W. Lewis

September 3, 1937

USE OF AIRFOILS OF N.A.C.A. SERIES
IN MODERN MILITARY AND COMMERCIAL AIRPLANES.

N.A.C.A. 22 series airfoils		
Army		
	Stearman	PT-13
	Consolidated	P-30
	"	PB-2A
	Curtiss	75
	North American	22
	"	BT-9
	"	BT-10
	Douglas	B-18
Navy		
	Curtiss	SBC-3
	"	SOC-1
	"	BF2C-1
	Douglas	XP3D-2
	"	R2D-1
	Great Lakes	BG-1
	"	XB2G-1
	Stearman	JRS-1
Commercial		
	Douglas	DC-2
	"	DC-3
	Sikorsky	S-43
	Arrow	F
	Fairchild	F-45
N.A.C.A. 23 series airfoils		
No military		
Commercial		
	Monocoach	
	Curtiss	A19-R
N.A.C.A. 24 series airfoils		
Army		
	Northrop	2J
	"	A-17
Navy		
	Curtiss	R4C-1
	Northrop	BT-1
Commercial		
	Spartan	7-W
	Cessna	C-34
	Luscomb Phantom	
	Fleetwings	F-401
N.A.C.A. 230 series airfoils		
No Army		
Navy		
	Vought	SB2U-1
Commercial		
	Beechcraft, all models (2 basic types)	
	Martin	156
	Taylorcraft	
	Barkley Grow	TSP-1
	Grumman	G-21
	Waterman Arrowbile	
	Lockheed	12
N.A.C.A. 00 series airfoils		
Army		
	Boeing	199
No Navy		
Commercial		
	Boeing flying boat	

Document 4-9(b), R.C. Platt, Memorandum for Dr. Lewis, "Airfoil sections employed for wings of modern airplanes," 2 September 1937, RA file 290, LHA, Hampton Va.

Washington D.C.

September 2, 1937

Memorandum for Dr. Lewis.
Subject: Airfoil sections employed for wings of modern airplanes.

1. Attached herewith is a list of the airfoil sections employed on modern army, navy, and commercial airplanes prepared by Mr. Helms and myself.

2. Of the 101 types listed, 66 employ N.A.C.A. airfoils, the other 35 do not.

3. It is interesting to note that of the whole list of the most generally successful and widely used airplanes are those which employ the later N.A.C.A. series airfoils. Among these are the highly successful Douglas, Lockheed, and Sikorsky transports; the Northrup and Consolidated military airplanes, and to the Beechcraft, Fairchild and Cessna of the lighter commercial types. Of particular note is the Cessna, which in competition with most representative of modern private-owner airplanes was adjudged the most efficient of the group. In the military field large orders have been placed for the new Douglas bomber employing the N.A.C.A. 22 series airfoil, succeeding the Martin B-10, which used Göttingen 398 airfoil. It is interesting to note that Martin in his more recent commercial types is adopting the latest improved N.A.C.A. airfoil, to-wit: the 230 series. Boeing likewise in their very large commercial airplanes, as well as in the highly successful 199 bomber, are using one of the more efficient N.A.C.A. airfoils - the 00 series.

Respectfully submitted,
R. C. Platt

Document 4-10

H.J.E. Reid, Engineer-in-Charge, Langley Field, to NACA, "Paper entitled 'A Few Present Problems in Aerodynamics,' by Dr. von Kármán," 8 February 1933, in RA file 290, LHA, Hampton, Va.

By 1933 Dr. Theodore von Kármán was already well on his way to becoming the dean of American aerodynamics. One of Ludwig Prandtl's most gifted protégés at Göttingen, von Kármán came to the United States in 1930 after being vigorously recruited by Dr. Robert Millikan to direct the new Guggenheim Aeronautical Laboratory at the California Institute of Technology (GALCIT) in Pasadena. Located on the West Coast in the years before the NACA had any presence there, von Kármán's ties to the NACA were never terribly strong. Although he certainly followed the results of NACA research and attended occasional aircraft engineering conferences at Langley, von Kármán never served on the NACA Main Committee or for that matter on any of its technical committees, not even its Committee on Aerodynamics.

In his paper "A Few Present Problems in Aerodynamics," not included in this chapter's documents, von Kármán pointed out a discrepancy between maximum lift data published by the NACA and results with the same airfoil in the GALCIT wind tunnel. In Langley's response to it, as expressed mostly by Eastman Jacobs, one detects something a bit more than a mere technical disagreement between two research groups. One also senses a feeling of rivalry between Langley and Caltech.

Certainly, by the late 1930s, a quiet rivalry existed. Caltech's own research program under von Kármán's supervision had been enriching the field of aerodynamics in some respects even more so than were Langley's. Caltech graduates held distinguished positions in colleges and universities across the country, but hardly any of them worked for the NACA. Perhaps most importantly, with the aircraft industry growing by leaps and bounds on the West Coast, the NACA recognized that it needed a stronger presence there, or the manufacturers and the military services would rely more on Caltech for advice and assistance than they already were. With this in mind, in 1939, the NACA chose Moffett Field in Sunnyvale, California, as the site for one of its two new research centers (what became the Ames Aeronautical Laboratory). At the same time, the NACA opposed federal spending for new wind tunnels at Caltech. NACA leadership, as southern California Congressman Carl Hinshaw complained in 1941, preferred to "retain a concentration of research facilities entirely within the NACA. They do not seem to be inclined to favor allowing these facilities to be spread out among the several qualified educational institutions. I do not just know whether it is the old question of professional jealousy or the old question of expanding bureaucracy or some other queer incomprehensible angle" (Congressional Record, 77/1, Vo. 87, Pt. 1, 1941, p. 416).

Dr. von Kármán felt much the same way about the NACA as did his congressman: that the NACA selfishly wanted the entire field to itself. He saw inherent dangers in an NACA monopoly, and, being an ambitious man and program builder, he wanted a much bigger piece of the pie for himself and Caltech. During the war von Kármán criticized the NACA for not getting more into rockets and jet propulsion, revolutionary technologies his own small Jet Propulsion Laboratory were pioneering. At the end of World War II, von Kármán strongly encouraged the Army Air Forces to establish bureaucratic structures and its own independent advisory groups and laboratories to conduct scientific research in the service of American military air supremacy. In essence, von Kármán advised the air forces (and the independent U.S. Air Force when it came to life in 1947) not to trust the NACA for the intensive research and development necessary to generate the ongoing technical advances required to keep the nation ahead of its enemies in terms of air power, but to build its own independent R&D establishment. By the early 1950s, he described the NACA as conservative and overly cautious and was literally in charge of a little aerodynamic research empire of his own.

Document 4-10, H.J.E. Reid, Engineer-in-Charge, Langley Field, to NACA, "Paper entitled 'A Few Present Problems in Aerodynamics,' by Dr. von Kármán," 8 February 1933, in RA file 290, LHA, Hampton, Va.

Langley Field Virginia,

February 8, 1933.

From LMAL
To NACA
Subject: Paper entitled "A Few Present Problems in Aerodynamics," by Dr. von Kármán.
Reference: NACA Let. Jan. 30, 1933, CW/NW

1. Dr. von Kármán's paper has been read by various members of the laboratory staff, as requested in letter of reference, and is desired to keep the paper a few days longer. It is therefore not being returned at this time.

2. With reference to the third paragraph of your letter, concerning the discrepancy of maximum lift of various airfoils as tested in the wind tunnel of California Institute of Technology and in our variable-density tunnel, the laboratory is not yet in a position to comment finally. The effects of turbulence and scale on airfoil characteristics have been under consideration for a long while at the laboratory, and it is appreciated that we should make every effort to bring into agreement the measurements from our various wind tunnels. To this end equipment is in preparation for making sphere drag tests in all the tunnels and we are accumulating information on the characteristics of the Clark Y airfoil in all the tunnels.

3. Mr. Jacobs offers the following comment with reference to Dr. von Kármán's comparison between the maximum lift of the 2412 airfoil as measured in the California Institute of Technology tunnel and in the variable-density tunnel:

The paper is too broad and general to justify much comment in detail. The parts dealing with the comparison of airfoil test results from the California Institute of Technology tunnel and from the variable-density tunnel should be considered.

To begin with, the paper contributes nothing new about the effects of turbulence on airfoil test results, except to present new test data for a tunnel that is relatively free from turbulence. The effects of turbulence were considered and conclusions, at least as accurate as von Kármán's, were reached by Stack in T.N. No. 364.

The sketches given by von Kármán to indicate the effects of turbulence on the breakdown of the flow over an airfoil are certainly misleading and inaccurate. He is mistaken about the position of the separation point "S" in the figure being independent of the value of the Reynolds number. We also question the statement to the effect that the separation that limits the lift can be avoided only when the transition point from laminar to turbulent flow in the boundary layer is ahead of the point "S." This separation may be only local, the flow at the boundary changing to turbulent soon after separation and closing in again, so that the effect on the lift may not be important.

"In regard to the presented test results, the Reynolds number range is rather limited and a poor method of plotting has been chosen. Plots against a logarithmic Reynolds number scale are considered preferable. His results for the 2412 airfoil are replotted in this form, together with our results for the Clark Y airfoil from 364 and from the full-scale tunnel for comparison. The results from the full-scale tunnel must be considered tentative, but all the results tend to indicate that the shape of von Kármán's curve "A" corresponding to the least turbulence is affected by something other than scale and turbulence. The air speeds in the California Institute of Technology tunnel may be so high that the maximum lift coefficients are influenced by compressibility."

H.J.E. Reid,
Engineer-in-Charge

Document 4-11

Eastman N. Jacobs, Memorandum to Engineer-in-Charge, "Scale effect on airfoils, conference," 11 April 1934," in RA file 88, Historical Archives, NASA Langley.

The concept of "effective Reynolds number" seems to have stemmed at least in part from an in-house conference on airfoil scale effects and turbulence held at Langley on 11 April 1934. Besides Eastman Jacobs, who authored the memo, others attending this meeting were Elton W. Miller, chief of the aerodynamics division; Fred E. Weick, assistant chief of aerodynamics; Carl Wenzinger, an engineer in the 7 X 10-Foot Atmospheric Wind Tunnel section who specialized in aerodynamic effects with reference to stability and control problems and to the lifting powers of wings; and Smith J. DeFrance, head of the Full-Scale Tunnel. (The exact identity of "Mr. Leisy," whose letter to the NACA is referenced early in Jacobs' memorandum, is unknown.)

"Scale effects' were a notorious difficulty with wind-tunnel testing, but they were especially plaguing to data coming from the Variable-Density Tunnel where $1/20^{th}$ scale models were used, and in a very turbulent and low-speed airstream. Results from VDT tests simply could not be extrapolated reliably to the performance of the actual airplane. This meant that "practical engineers" really did not know how to use VDT data for their flight applications. Rightfully so, the NACA considered this to be a major problem and something that needed to be corrected.

Jacobs' concept of "effective Reynolds number," which he invented not long after this conference, became the NACA's stop-gap way of correcting for scale effects. Although not an altogether satisfactory solution, it remains even up to today a standard way of correcting for the problem. A *Dictionary of Technical Terms for Aerospace Use*, published by NASA in 1965 (NASA SP-7), lists the term and defines it as "A fictitious Reynolds number applied to the flow of air about a body in a wind tunnel, equal to the free-air Reynolds number at which the effect obtained is the same as the effect obtained in the wind tunnel" (p. 93).

Document 4-11, Eastman N. Jacobs, Memorandum to Engineer-in-Charge, "Scale effect on airfoils, conference," 11 April 1934," in RA file 88, LHA, Hampton, Va.

L. M. A. L.

Langley Field Va.,
April 11, 1934

MEMORANDUM For Engineer-in-Charge.

Subject: Scale effect on airfoils, conference.

1. A conference attended by the following members of the laboratory staff was held to discuss scale effect: Miller, Weick, Wenzinger, DeFrance, and Jacobs. A letter from Mr. Leisy together with Mr. Jacobs' reply was first considered as typical of the viewpoint of the practical engineers who wish to know how to use the variable-density tunnel data for their flight applications. Their problem is to predict the characteristics of the airfoil section at any value of the Reynolds number within the flight range, say 1,000,000 to 30,000,000. This problem was discussed in relation to the full-scale tunnel tests of the Clark Y airfoil, and the desirability of preparing a publication at this time discussing the solution of the problem, was considered. The relation of the proposed sphere tests to the general problem was also discussed.

2. The discussion brought out the fact that there are two rather distinct problems toward the solution of which our research should be definitely directed. These are:

(a) The problem of correcting full-scale tunnel test results to flight.

(b) The problem of correcting the variable-density tunnel results to flight at any Reynolds number within the flight range.

As regards the aerodynamic characteristics of airfoils, the second problem must be considered the more important, but its solution is to a large extent dependent on the solution of the first because we cannot test full-scale airfoils directly in flight. For our basic flight characteristics for comparison, we must depend largely on predictions from our airfoil tests in the full-scale tunnel.

3. Considering first, therefore, the problem of correcting the full-scale tunnel results to flight, either we must say that the corrections are so small that the results apply directly with sufficient accuracy, or we must evaluate turbulence corrections. For the time being, the first course is probably the best, the justification for it resting on the reasonably close agreement that has been obtained between flight and tunnel tests of the same airplanes. It might be advisable also to make sphere drag tests in the tunnel and in flight. The consensus of opinion was that sphere test data would be of some immediate value if they showed small differences between the tunnel and flight characteristics for the spheres.

4. Sooner or later, however, methods of correcting the full-scale tunnel data to zero turbulence will be desired. Probably the methods we devise for correcting the variable-density tunnel data for turbulence will be applied to the full-scale tunnel data to extrapolate to zero turbulence as determined by the characteristics of complete airplanes and spheres as measured in flight. In this connection it might possibly be advisable at some later time to investigate the characteristics of some airfoils, airplanes, and possibly spheres in the full-scale tunnel with increased turbulence so that in certain cases for the extrapolation to zero turbulence the shapes of the turbulence-effect curves may be more accurately established.

5. In regard to the main problem, that of correcting the variable-density tunnel results to flight at any value of the Reynolds number within the flight range, the data we now have, except for a few airfoils, are all at one value of the Reynolds number. Therefore, even if we could correct our data for turbulence, we would still not be in a position to predict flight characteristics at any desired Reynolds number.

6. The tests most urgently needed at present, therefore, are those for which authority was requested in our letter of August 16, 1932. The results are required to give the desired scale-effect data for a group of related airfoils. We are starting these tests in the variable-density tunnel on authority of N.A.C.A. letter of August 17, 1932 to investigate the scale effect for the following N.A.C.A. airfoils:

Thickness series	Thickness series with camber	
0009	2409	
0012	2412	6412 Camber series
0015	2415	
0018		6712

7. The recommended plan is, then, after analyzing this scale-effect data, to select some of the above airfoils for testing in the full-scale tunnel in order to determine their characteristics corresponding to the reduced turbulence of the full-scale tunnel. This information should form the basis for correcting all the results to zero turbulence and to any Reynolds number within the flight range.

8. It was decided in the meantime to continue the investigation in the 7 by 10 foot tunnel of the sphere-pressure system of measuring turbulence and to make preparations for measurements in flight. This system, if it proves satisfactory, will be used as a measure of the turbulence in the full-scale tunnel. The possibility of checking the sphere-pressure system in the N.A.C.A. tank both in the air and in the water was also discussed.

9. As regards the publication of the report covering the full-scale tunnel tests of the Clark Y, the consensus of opinion was that it might be published now substantially as it is, but that it should be very carefully edited to assure that it will give the reader the correct picture of the scale-effect and turbulence problem and the relation of that work to the Committee's general work on the problem.

10. The point we must keep in mind in formulating our programs is that it lends weight to our conclusions if we can show a relation between the effect of turbulence on a sphere and on airplanes and airfoils, but that the effect on airplanes and airfoils can be evaluated independently of spheres and other turbulence-measuring devices.

Eastman N. Jacobs
Associate Aeronautical Engineer.

Document 4-12

G.W. Lewis, Director of Aeronautical Research, NACA, to Mr. J.L. Naylor, Secretary, Aeronautical Research Committee, National Physical Laboratory, Teddington, Middlesex, England, 13 April 1937, in RA file 290, LHA, Hampton, Va.

Bothersome questions about the reliability of the NACA's airfoil data persisted late into the 1930s and had to be answered responsibly. One document in our collection that clearly indicates the NACA's need to respond to potentially embarrassing findings involves a letter from George Lewis, director of research for the NACA, to J.L. Naylor, secretary of the Aeronautical Research Committee for Britain's National Physical Laboratory, dated 13 April 1937. (This was just shortly before Lewis nominated Eastman Jacobs for the IAS award.)

Lewis was responding to preliminary confidential data showing that tests in the NPL's own compressed-air tunnel gave quite different drag numbers for the N.A.C.A. 23012 airfoil. Although admitting the discrepancies appeared "at first a little disturbing," Lewis tried to make light of them. First, he stated that "From past experience we have learned not to expect too much from comparisons of wind tunnel results." Then, he argued that the use of effective Reynolds number "now brings some of the results into fair agreement and that the agreement of the drag results tends to improve as we approach the higher Reynolds numbers in which we are particularly interested." Finally, he emphasized that, irrespective of the bothersome discrepancies, the British results, in general, confirmed the NACA's most important conclusion—that the N.A.C.A. 23012 was superior to most commonly used airfoils.

Document 4-12, G.W. Lewis, Director of Aeronautical Research, NACA, to Mr. J.L. Naylor, Secretary, Aeronautical Research Committee, National Physical Laboratory, Teddington, Middlesex, England, 13 April 1937, in RA file 290, LHA, Hampton, Va.

April 13, 1937

Mr. J. L. Naylor, Secretary,
Aeronautical Research Committee,
National Physical Laboratory,
Teddington, Middlesex.

Dear Mr. Naylor:

The Committee appreciates very much the opportunity to examine the preliminary confidential data from your compressed-air tunnel on the N.A.C.A. 23012 airfoil, which you kindly sent with your letter on February 15, 1937. This information was particularly interesting to the members of our staff at Langley Field. I note that you do not at present plan to publish this report, and agree that the difference in drag shown in the results from our variable-density wind tunnel appears at first a little disturbing. I note however, that you conclude in general that your results substantiate the claims made for the airfoil.

From past experience we have learned not to expect too much from comparisons of wind-tunnel results. Dryden's analyses of previous data in particular have shown that differences in wind-tunnel turbulence may produce marked discrepancies in the results from different wind tunnels. The fact that the use of the "effective Reynolds number" now brings some of the results into fair agreement and that the agreement of the drag results tends to improve as we approach the higher Reynolds numbers in which we are particularly interested seems to us to be encouraging. That some discrepancies still remain indicates we do not yet fully understand the subject and that further work remains to be done.

At his request, I am transmitting the following comments by Mr. Jacobs of our laboratory staff:

"The failure of the results from this compressed air tunnel, variable-density tunnel and full-scale tunnel to show better agreement is in some respects disappointing. I think, however, that the difference is found are in the main to be attributed to differences in wind-tunnel turbulence, or, if you like, to our failure to completely correct it for these effects. In general, the results from the compressed air toddled appear about as we had expected from our comparisons of the results from the variable-density tunnel with those from the much less turbulent full-scale tunnel. Such differences were in fact predicted some time ago when we drafted our report on scale effect and before we had seen the results from the compressed-air tunnel on the N.A.C.A. 0012 airfoil. It now appears that the "turbulence factor" of the

compressed-air-tunnel is somewhere between 1.2 and 1.6, although our impression was that early sphere tests in the compressed-air tunnel showed a higher value than this.

"It is suggested that a sphere test of that type made here by Mr. Platt (Technical Report No. 558) would be of considerable value in the interpretation of the results from the compressed-air tunnel. This test is easily made but it is necessary that the sphere be smooth and steady, and that it be mounted from the rear. We would like to know whether any changes have been made in the compressed-air tunnel that might have changed the tunnel turbulence since the early sphere and airfoil tests were made."

"A few matters of secondary importance should also be mentioned. It is unfortunate that the complete scale-effect data for the N.A.C.A. 23012 airfoil as they will appear in our scale-effective report were not available in England when your report was prepared. The complete data extend to lower values of the Reynolds number and point the way to a better fairing of the experimental drag results. For example, the cross fairing indicates that our results at the next-to-the-highest Reynolds number (15 atmospheres) were somewhat high, owing to a slight roughness accumulated on the nose of the model during this and the lower pressure runs. The same result does not appear at the highest Reynolds numbers because the models, to avoid slight roughness effects that were known to be most critical at the highest Reynolds number, were carefully refinished and repolished before the final 20-atmosphere tests were made. It appears that the results from the compressed-air tunnel may be subject to similar roughness effects at the highest Reynolds numbers. The compressed-air-tunnel data shown in figures 2 and 4 indicate effects something like those found in the compressed-air tunnel with roughness on the N.A.C.A. 0012 (R. & M. No. 1708). Furthermore, the maximum lift coefficient for the N.A.C.A. 23012 is surprisingly low as compared with that of the chromium-plated N.A.C.A. 0012. We would suggest that the N.A.C.A. 23012 be highly polished and check-tested at the highest Reynolds number."

In regard to the scale-effect report referred to in the preceding comments by Mr. Jacobs, I am forwarding here with an advance confidential copy, as it contains our latest data from the variable-density tunnel on both the N.A.C.A. 0012 and the N.A.C.A. 23012 airfoils, together with rather complete discussions of various corrections now employed. We are now engaged, as the result of the various suggestions, in incorporating a few minor changes in the report. We hope that you will find this report as interesting and helpful as we have found your preliminary reports.

Sincerely, yours,

G. W. Lewis
Director of Aeronautical Research

Document 4-13

Hugh B. Freeman to Chief of the Aerodynamics Division [Elton W. Miller], "Boundary-layer research," 18 April 1932, in RA file 201, LHA, Hampton, VA.

An excellent popular account of "boundary layer" research from the year 1995 explains that there is really "no such thing as absolutely pure laminar flow, for there is always a very thin and relatively stagnant 'boundary layer' of air between the skin of an airplane and the free-stream air surrounding it." No matter how smooth the skin of an airplane wing may be, it has "microscopic irregularities that tickle the air going past it at high speed" forcing the closest molecules to stumble and lose speed. These molecules "impart their confusion" to some other molecules in the next outward layer, and they in turn cause similar chaos in the next. "Imagine a roaring river at flood stage; within an inch of its banks, the water barely moves, though it burbles and twists. Those banks are the equivalent of an airplane's wings experiencing non-laminar flow" (Stephan Wilkinson, "Go With the Flow," *Air & Space Smithsonian* 10 [June/July 1995]:33).

Aerodynamicists measure the thickness of the boundary layer from the surface to the point where the speed of the molecules is 99 percent of the stream velocity. In practical terms, this means that the boundary layer is never more than about an inch thick. But, as Hugh B. Freeman and other aerodynamic thinkers realized by the early 1930s, that one inch played all sorts of tricks on the aerodynamic efficiency of an airplane (or airship, for that matter). In particular, turbulence in the boundary layer created drag, the main retarding force acting upon a body in flight. The greater the turbulence, the greater the drag—and the greater the drag, the less efficient is the airplane. It will fly slower and not as far, or it will require more fuel, which added cost.

These basic facts of physics made boundary-layer research critical to the future of aerodynamic improvement. As Hugh B. Freeman declared in his memo to the engineer-in-charge in 1932, no field of research offered "greater possibilities for the improvement of aircraft performance and safety" than boundary-layer control.

In his memo Freeman proposed a program of investigation by the NACA aiming at laminar-flow control (LFC) rather than natural laminar flow. Natural laminar flow was based on the idea that laminar flows could be maintained farther back over the chord of a wing simply by designing the airfoil shape correctly. Freeman's idea, on the other hand, was to prolong laminar flow mechanically by using slots in the wing and a blower system to suck away some of the turbulent molecules on the wing's surface.

The idea was certainly not original to Freeman or to the NACA. In the mid-1920s Dr. Richard Katzmayr, another Prandtl student, who was serving as director of the Vienna Aerodynamical Institute in Austria, had come up with the idea of increasing lift by blowing compressed air over a wing surface. Curiosity about this promising line of new research led the NACA on 21 January 1929 to approve Research Authorization (RA) 201, "Investigation of Various Methods of Improving Wing Characteristics by Control of the Boundary Layer." Under this aurthorization, as Freeman's memo points out, Langley engineer Millard J. Bamber conducted wind-tunnel work in 1929 and 1930 on airfoil boundary layer control using "backward opening" slots. At roughly the same time, NACA Langley chief test pilot Thomas Carroll made test flights with a special wing incorporating an arrangement of sucking slots.

Freeman's own involvement in boundary-layer work began, interestingly enough, not with airplanes but with airships. Assigned to airship research upon reporting to work at Langley in 1931, Freeman somehow picked up quickly on the enormous potential of boundary-layer control for reducing airship drag. Looking back into what the NACA and others had been doing to better understand the boundary layer, he came across Katzmayr's promising results and the tentative preliminary experiments of Bamber and Carroll. Considering some careful iteration of Carroll's full-scale tests as the best way to go about gaining insights into boundary-layer control, young Freeman formally presented his proposal to engineer-in-charge Henry J. E. Reid. After consulting with key staff members who liked Freeman's plan, Reid approved the work for the Propeller Research Tunnel, where full-scale investigations on wings were possible.

In an extended analysis of RA 201 that appears in Vol. 2 of *Model Research: The National Advisory Committee for Aeronautics*, 1917-1958 (NASA SP-4103, 1985), historian Alex Roland underscored the historic significance of Freeman's April 1932 memo. "This was truly a new departure in the history of R.A. 201," Roland emphasized. "Previous efforts had sought for ways to delay separation and increase the velocity gradient within the boundary layer. Freeman would concentrate on delaying the transition from laminar to turbulent flow." The idea was "by no means original with him, but his work on airships and his reading of earlier NACA efforts convinced him that this was a promising line of research and one with which the NACA should be deeply involved." Subsequent developments proved him right. Following his lead, NACA researchers, notably Eastman Jacobs, "would make their greatest contribution to boundary-layer control, the laminar-flow airfoil" (*Model Research*, II: 538).

Document 4-13, Hugh B. Freeman to Chief of the Aerodynamics Division [Elton W. Miller], "Boundary-layer research," 18 April 1932, in RA file 201, LHA, Hampton, Va.

April 18, 1932

MEMORANDUM For Chief Aerodynamics Division
Subject: Boundary-layer research.

1. The purpose of this memorandum is to call attention to the lack of large-scale experimental data relative to the boundary-layer problem and to suggest a program of research which will provide information along these lines.

2. The field of boundary-layer control, in this writer's opinion, offers greater possibilities for the improvement of aircraft performance and safety than any other. This is because the control of the boundary layer influences every important aerodynamic characteristic of an aircraft. The three most important advantages offered by the control of the boundary layer are: (1) increase in lift, (2) an increase in the angle-of-attack range below the burble, and (3) a decrease in minimum drag. The first two advantages have been shown repeatedly by tests on small models, principally those of Oscar Schrenk in Germany, who obtained a maximum lift coefficient of $C_L = 5.0$ and an L/D ratio of 50 by the method of removing the boundary layer on the upper surface of an airfoil by suction. The possibility of realizing the third advantage is shown by Figure 1 (Original figures not included herein.) in which is plotted the drag of the ZRS-4 airship computed for laminar and turbulent boundary layers. From these curves it is seen that if a method of controlling the boundary layer could be devised which would force the flow to remain laminar instead of changing to turbulent, the drag (of even a well streamlined body) could be reduced to about 10 percent of its present low value. The same thing holds true for the drag of the wings and fuselage or an airplane, since the Reynolds number in this case is greater than those shown, and hence a greater portion of the boundary layer is turbulent. It is not expected that a reduction as great as that cited above will be obtained in practice, of course, but even a 10 or 20 percent reduction in the drag of an airship or of an airplane would certainly be worthwhile.

3. The only serious experimental work which has been undertaken at this laboratory on boundary-layer control is that of Mr. Bamber. His tests, while they showed substantial improvement in the lift coefficient, were never carried to a logical conclusion. As he pointed out in his report, if multiple slots had been tested and also suction slots (i.e., slots opening normal to the surface) even more favorable results would have been obtained. Small-scale tests are at a disadvantage, however, even under the best of conditions, because of the fact that the boundary layer on small models is so extremely thin that it is not possible to construct slots in the surface which are not out of all proportion to this thickness and which will not distort the flow.

4. The full-scale tests reported by Carroll, at this laboratory (file No. 1115.6/1), transmitted with L.M.A.L. letter September 15, 1927, unpublished, on boundary-layer control by the method of pressure slots was a step in the right direction. In these tests, however, only two slots (backward opening) were used; the position of these slots, in the light of more recent experiments, was not very good; only one wing pressure was used; this pressure was not measured, and only one wing of the biplane was fitted with the slots. In spite of these limitations, however, the results showed a 6 percent gain in the rate of climb and a considerable increase in the maximum angle of attack at which the plane could fly. If the slots had been changed to simple suction slots normal to the surface and the air had been sucked into the wing instead of being blown out, the results would no doubt have been much more favorable.

5. There are several advantages to be gained by conducting the tests at full scale. these are enumerated as follows:

(1) The elimination of scale effect which is especially great in the flow through slots.

(2) The boundary layer is much thicker than on small models and hence the slot structure may be built in proportion, offering a minimum of disturbance to the flow.

(3) The blower system and the apparatus for measuring the boundary layer may be installed inside the wing, greatly simplifying the method of testing and the accuracy of the measurements.

(4) The power expended in driving the blower and the efficiency of the blower and slot system may be determined by direct measurements.

Hugh B. Freeman,
Assistant Physicist.

Document 4-14

Theodore Theodorsen, Senior Physicist, to Engineer-in-Charge, "Boundary layer removal," 4 Feb. 1932, in RA 88, LHA, Hampton, Va.

Dr. Theodore Theodorsen was definitely someone to listen to, not just at Langley but in the American aeronautics community generally. The man possessed one of the country's most brilliant scientific minds and in the long run contributed at least as much to modern aerodynamics as Eastman Jacobs did, if not more.

Following an engineering degree from his homeland's Norwegian Institute of Technology at Trondheim in 1922, Theodorsen came to the United States and earned a Ph.D. in physics from Johns Hopkins University. Encouraged by Dr. Joseph Ames, Johns Hopkins' president and chairman of the Executive Committee of the NACA, Theodorsen came to Langley in 1929 as an associate physicist. Within a short time, the talented young man was made head of the Physical Research Division, Langley's smallest division. Because his administrative duties were light, he was able to concentrate on his own work.

The list of his accomplishments just during the 1930s is prodigious: he improved thin-airfoil theory by introducing the best angle of streamlining; devised an elegant theory of arbitrary wing sections; developed the basic theory of aircraft flutter; made improvements to NACA engine cowlings and ducted propellers; expanded propeller theory and developed scaling laws for propeller vibrations; performed the first NACA in-house aircraft noise research; worked on fire prevention in aircraft and on means of icing removal and prevention; made early measurements of skin friction at transonic and supersonic speeds, and much more. In fact, although historians to this date have not generally recognized it, a strong case can be made that Theodorsen was the most thoughtful and productive researcher at the NACA during his 18-year tenure there. He resigned from the NACA in 1947 in order to help administer a new aeronautical institute being organized in Brazil. He later served as chief scientist for the U.S. Air Force and chief of research for Republic Aviation Corporation. (See *A Modern View and Appreciation of the Works of Theodore Theodorsen, Physicist and Engineer*, ed. Earl H. Dowell (Washington: American Institute of Aeronautics and Astronautics, 1992.)

Theodorsen's idea of "feathers" for boundary layer control may not make sense upon first reading, but essentially what he was suggesting was the idea of a "slotted flap." The significance of the flap, a hinged airfoil in the form of a long narrow strip attached to the rear of a wing, was discussed in Chapter 3 in relation to the design revolution of the 1930s. Essentially, a flap provided higher lift than a wing without flaps could manage. The NACA did not invent the flap; the flap evolved continuously from the idea of "ailerons" invented by French airplane designer and pilot Henry Farman for improved lateral control of his 1908 airplane. The technology grew from there. Late in World War I, German pilot G.V. Lachman tried out the first "slotted wing," with its long spanwise slot located near the leading edge of the wing. Soon after the war, Britain's Handley Page was running wind-tunnel tests indicating that slotted wings improved lift by as much as a whopping 60 percent. In the 1920s, numerous wing flap modifications followed, including ones invented by Orville Wright and Harlan D. Fowler, an engineer working for the U.S. Army who subsequently designed effective flaps for the Glenn L. Martin Company. By the time Theodorsen conceived his notion of "feathers," airplane designers worldwide had embraced the idea of flaps. There was no more practical way for them to deal with increasing speeds and wing loadings of the modern airplane.

The NACA was involved in flap research in a major way. It conducted pioneering experiments on various types, including split flaps, slotted flaps, and later on spoilers, double- and triple-slotted flaps, slats, and many other types of wing appendages in various combinations on both the front and back of the wing, designed for high-lift. In a sense, what Theodorsen was calling for in 1932 with his idea of "feathers" was an experimental slotted-flap system. It was a rather odd expression of a good idea that would pay off when such high-lift systems were effectively incorporated, particularly on large airplanes where the aerodynamic advantages were the greatest.

Theodorsen's memo also demonstrates a few interesting points about the personality of one of the NACA's leading aerodynamicists. Being primarily a theoretician did not stop him from suggesting even the most direct analogy from nature (like the one he made here based on bird feathers), nor did it inhibit him from proposing and carrying out the simplest experiment. (The suggestion he made in this memo was approved by the engineer-in-charge; Theodorsen put together some simple tests in the VDT on a wing section with two or three "feathery" modifications.) The first line of his memo also suggests how a problem could absolutely monopolize his attention. One of his closest associates in the Physical Research Division, Isadore Edward Garrick, remembered Theodorsen's rapt method of working. Having decided that something like boundary layer control was worth his thinking about, "he would work on it during relatively short periods of intense concentrated activity, almost incommunicado, followed by periods of apparent desultory inactivity." And once he became convinced that he had a good idea, he expressed it, without fear of ridicule and hoping that others would take it seriously and talk to him about it in an open, friendly, and constructive way. Equally, he allowed his subordinates to develop their own talents and resources without fear of embarrassment or ridicule. Not that he was not a stern critic, for he most definitely was. But Theodorsen was always helpful when asked, even if a junior associate came to him with a raw or semi-finished product. (Isadore Edward Garrick, "Sharing His Insights and Innovations," in *Modern View and Appreciation of Theodore Theodorsen*, p. 21).

Document 4-14, Theodore Theodorsen, Senior Physicist, to Engineer-in-Charge, "Boundary layer removal," 4 Feb. 1932, in RA 88, LHA, Hampton, Va.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS
LANGLEY MEMORIAL AERONAUTICAL LABORATORY
LANGLEY FIELD, HAMPTON, VA.

February 4, 1932

MEMORANDUM For Engineer-in-Charge
Subject: Boundary layer removal

1. I have given the question of boundary layer removal an examination as regards its theoretical possibilities. The result is so interesting that I shall submit the essential conclusion for your immediate consideration.

2. To increase lift and high angle control it is necessary to equip the trailing and end edges of the wing with "feathers"; that is, flexible marrow plates with a rather rigid central core or stem. It is significant that the birds, in spite of the handicap of low Reynolds numbers, have resorted to this trick. The operating principle of the "feathers" is to suck off the boundary layer. The kinetic energy of the lower side air stream is utilized to pump the dead air from the upper side along the stem of each "feather" when bent down. The most efficient size of the "feathers" can be approximately predicted. The best shape may, however, differ to some extent because of the larger Reynolds numbers. It is possible to attach the "feathers" directly to the wing or to the ailerons. Incidentally, all guiding surfaces, for instance the rudder, should apparently be equipped with "feathers."

Theodore Theodorsen
Senior Physicist

Document 4-15

Theodore Theodorsen, Introduction, "Theory of Wing Sections of Arbitrary Shape," NACA *Technical Report* 411 (Washington, 1931).

Dr. Theodore Theodorsen was practical enough to realize that the "imperfect status" of wing theory required designers to make their airfoils "independent of theoretical restrictions." But more than anyone else at Langley he saw the need for the NACA's research staff to fertilize its experimental routine with a stronger dose of theory. In his opinion, to discover more advanced airfoils the NACA did not need a new wind tunnel as Eastman Jacobs was suggesting but rather better mathematical and physical understanding of the effects of the basic aerodynamic phenomena on wing performance. The implication of his argument was that the experimentalists at Langley had become too interested in and dependent upon equipment for their own good.

Jacobs disagreed totally with the idea that theoreticians could answer the remaining questions about airfoils better than could experimentalists; he also rejected the argument that it was unnecessary and impossible for the NACA or anyone else to build a pressure tunnel having low airstream turbulence, which was one of Theodorsen's points. But Jacobs, in principle, did not oppose Theodorsen's notion of theory's critical role in successful research. Nor did he disagree that many Langley researchers were weak mathematically, as that was common for engineers trained in American colleges and universities. An adventurous man with an expansive outlook on what was possible, Jacobs kept up with and understood the most current theory—though he did not devote much of his own time to its study—and valued its role in creating the fundamental but directly useful technological information expected of the NACA. Jacobs' problem with Theodorsen was more a battle over "turf." At Langley both men controlled fiefdoms, and because both men were so valuable, NACA officials had permitted the feudal arrangement to flourish. Usually the two men worked on completely separate activities, but occasionally they had to work together—and then they inevitably clashed.

The introduction to TR 411 is one of the most remarkable openings to any NACA or NASA technical report ever published. Few formal publications of the agency have ever expressed such thoughtful statements on research philosophy and almost none have involved criticisms of colleagues, no matter how indirect.

Document 4-15, Theodore Theodorsen, Introduction, "Theory of Wing Sections of Arbitrary Shape," NACA* Technical Report *411 (Washington, 1931).

REPORT NO. 411

THEORY OF WING SECTIONS OF ARBITRARY SHAPE

By THEODORE THEODORSEN

SUMMARY

This paper presents a solution of the problem of the theoretical flow of a frictionless incompressible fluid past airfoils of arbitrary forms. The velocity of the 2 dimensional flow is explicitly expressed for any point at the surface, and for any orientation, by an exact expression containing a number of parameters which are functions of the form only and which may be evaluated by convenient graphical methods. The method is particularly simple and convenient for bodies of streamline forms. The results have been applied to typical airfoils and compared with experimental data.

INTRODUCTION

The theory of airfoils is of vital importance in aeronautics. It is true that the limit of perfection as regards efficiency has almost been reached. This attainment is a result of persistent and extensive testing by a large number of institutions rather than of the fact that the important design factors are known. Without the knowledge of the theory of the airflow around airfoils it is well-nigh impossible to judge or interpret the results of experimental work intelligently or to make other than random improvements at the expense of much useless testing.

A science can develop on a purely experimental basis only for a certain time. Theory is a process of systematic arrangement and simplification of known facts. As long as the facts are few and obvious no theory is necessary, but when they become many and less simple theory is needed. Although the experimenting itself may require little effort, it is, however, often exceedingly difficult to analyze the results of even simple experiments. There exists, therefore, always a tendency to produce more test results than can be digested by theory or applied by industry. A large number of investigations are carried on with little regard for the theory and much testing of airfoils is done with insufficient knowledge of the ultimate possibilities. This state of affairs is due largely to the very common belief that the theory of the actual airfoil necessarily would be approximate, clumsy, and awkward, and therefore useless for nearly all purposes.

The various types of airfoils exhibit quite different properties, and it is one of the objects of aerodynamical science to detect and define in precise manner the factors contributing to the perfection of the airfoil. Above all, we must work toward the end of obtaining a thorough understanding of the ideal case, which is the ultimate limit of performance. We may then attempt to specify and define the nature of the deviations from the ideal case.

No method has been available for the determination of the potential flow around an arbitrary thick wing section. The exclusive object of the following report is to present a method by which the flow velocity at any point along the surface of a thick airfoil may be determined with any desired accuracy. The velocity of the potential flow around the thick airfoil has been expressed by an exact formula, no approximation having been made in the analysis. The evaluation for specific cases, however, requires a graphical determination of some auxiliary parameters. Since the airfoil is perfectly arbitrary, it is, of course, obvious that graphical methods are to some extent unavoidable.

Curiously enough, the theory of actual airfoils as presented in this report has been brought into a much simpler form than has hitherto been the case with the theory of thin airfoils. In the theory of thin airfoils certain approximations have restricted its application to small cambers only. This undesirable feature has been avoided, and the results obtained in this report have a complete applicability to airfoils of any camber and thickness.

The author has pointed out in an earlier report that another difficulty exists in the theory of thin airfoils. It consists in the fact that in potential flow the velocity at the leading edge is infinite at all angles except one. This particular angle at which the theory actually applies has been defined as the ideal angle of attack. In the present work we shall not go any further into this theory, since it is included in the following theory as a special case of rather limited practical importance.

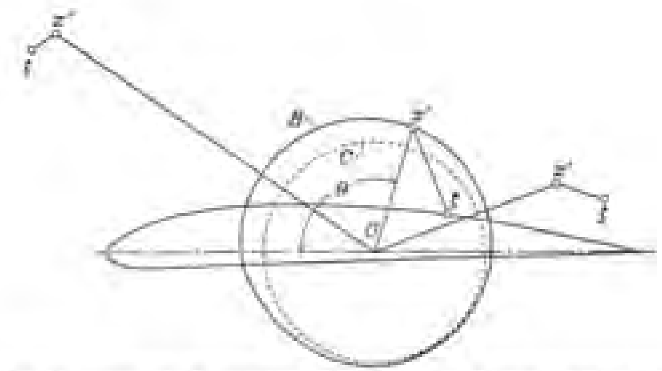


FIGURE 1.—Showing the transformation from a nearcircular curve B into an airfoil

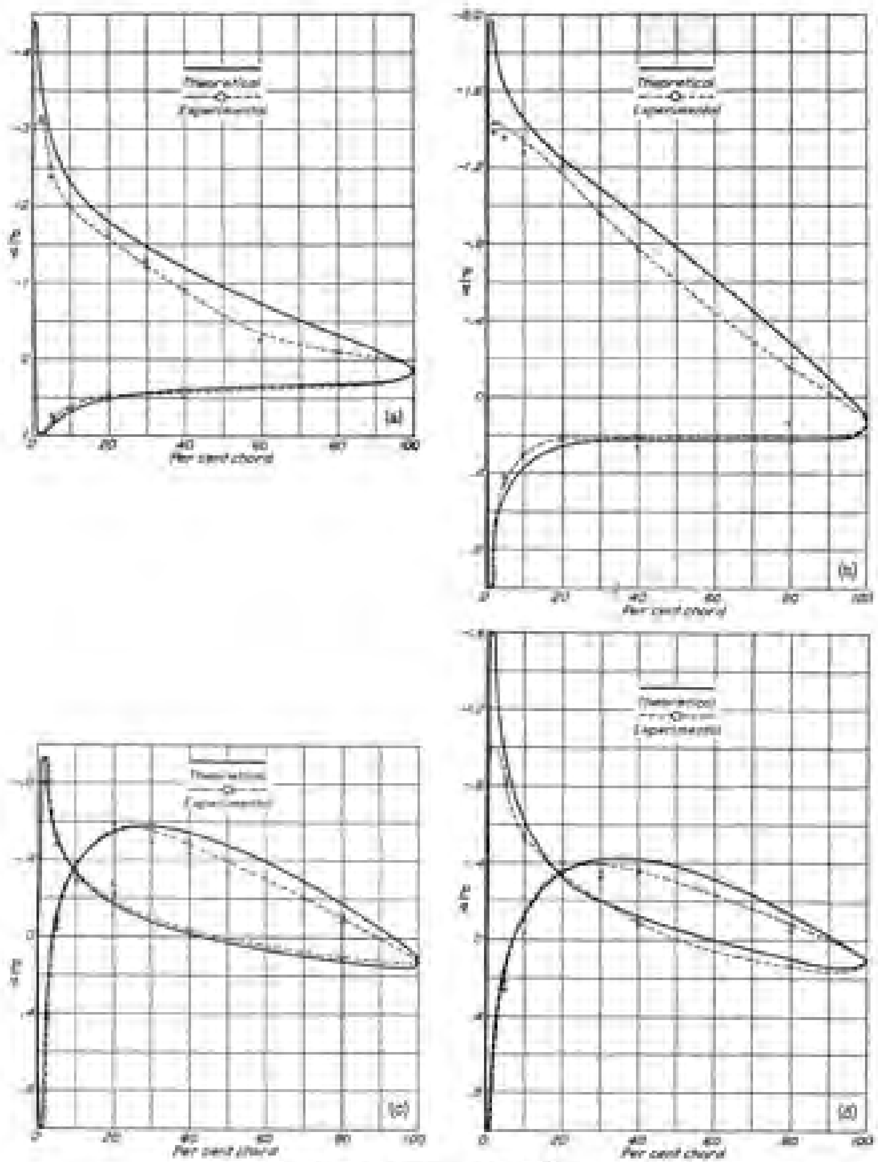


FIGURE 2.—Pressure-distribution curves along x-axis of Clark Y. $\frac{P}{q}$ against per cent chord. [illegible]

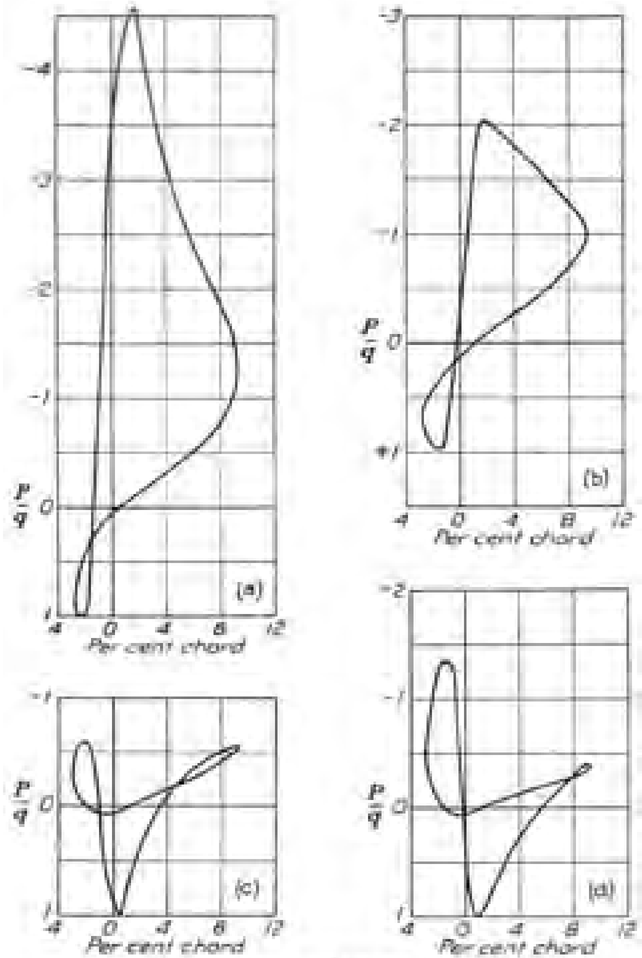


FIGURE 3.—Theoretical pressure distribution along x-axis of Clark Y
(a) α= [illegible], (b) α= [illegible], (c) α= $-1°$ [illegible], (d) α= $-3°$ [illegible]

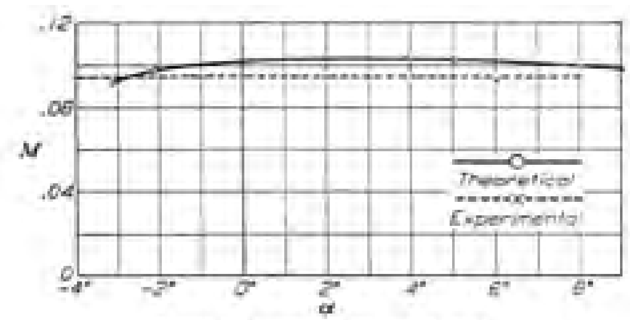


FIGURE 4.—Moment against angle of attack

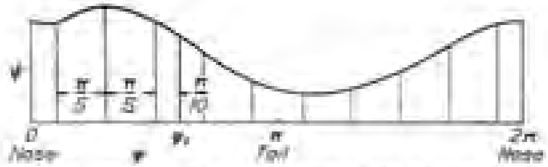


FIGURE 5.—The ψ against φ curve, illustrating method of evaluation of ϵ_t.

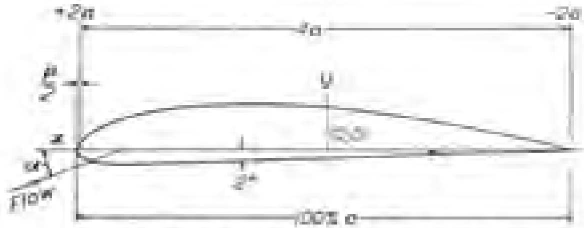


FIGURE 6.—Clark Y airfoil—showing system of coordinates.

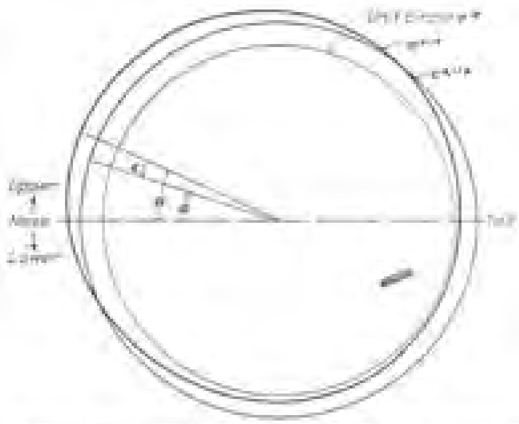


FIGURE 7.—The unit circle $z=e^{i\theta}$, the circle $z=e^{\psi+i\theta}$, and the corresponding curve $z'=e^{\psi+i\theta}$.

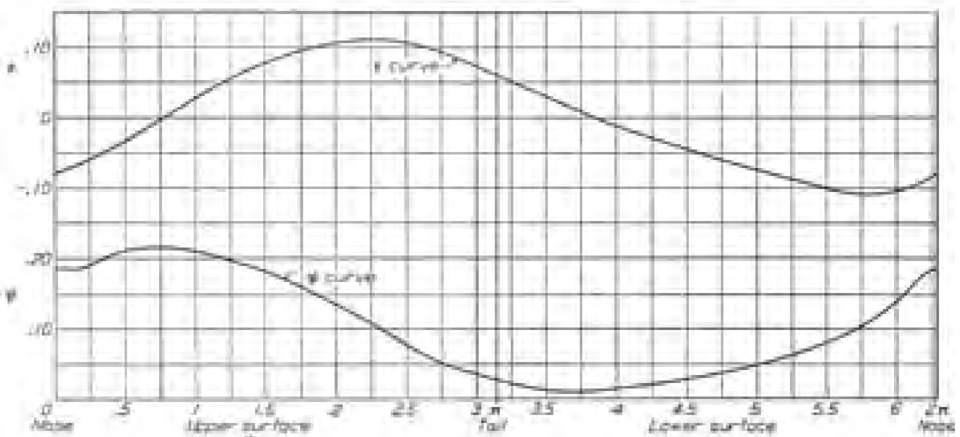


FIGURE 8.—(a) The ψ against φ curve for the Clark Y. (b) The ε against φ curve for the Clark Y.

TABLE I

CLARK Y

UPPER SURFACE

[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

LOWER SURFACE

[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

TABLE II

CLARK Y

[illegible]

Upper surface							Lower surface					
% c	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible] Formula (g)	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

Table II gives the numerical values for Figure 2a in detail as an example. See also Table I.

TABLE III

Figure	Geometric angle as measured from chord line experimentally α_0	Area of experimental pressure distribution curve A	[illegible] radians	Apparent angle measured from [illegible] degrees	Angle measured from chord line to be used as a basis for comparison with [illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

TABLE IV

Figure	M_x Moment about line x=2a per unit chord	M_y Moment about line y=0	M_p Moment about point x=2a per unit chord, y=0
[illegible]	[illegible]	[illegible]	[illegible]

Document 4-16
Eastman Jacobs, Memorandum to Engineer-in-Charge, "Program for study of scale effect on airfoils," 13 November 1929, in RA file 88, LHA, Hampton, Va.

Eastman Jacobs himself would later spotlight this memo as evidence of his early awareness of the possibilities "for controlling boundary layer directly through body shape or through control of the usual pressures acting along the body surface." (See Document 4-18.) It is interesting that Jacobs cited a conversation with George Lewis, the NACA's director for research, as the stimulus for his study of scale effects, which suggests that Lewis himself was no lightweight when it came to addressing fundamental aerodynamic problems.

At the time, it is clear from this memo, Jacobs expected the VDT to provide the experimental data needed to arrive at the specific airfoil shapes that delayed transition and sustained laminar flow. It would not be long, however, before he became convinced that the turbulence in the existing Variable-Density Tunnel could never be resolved. Without a new low-turbulence pressure tunnel, no practical investigation of this type of boundary layer control for improved airfoils would be possible.

Document 4-16, Eastman Jacobs, Memorandum to Engineer-in-Charge, "Program for study of scale effect on airfoils," 13 November 1929, in RA file 88, LHA, Hampton, Va.

November 13, 1929.

MEMORANDUM For Engineer-in-Charge

Subject: Program for study of scale effect on airfoils.

1. A recent conversation with Mr. Lewis caused me to devote some more intensive thought and study to the mechanism and causes of scale effect on airfoils. Any one who has given the subject much attention realizes that the effect of the dynamic scale originates in the very thing and boundary layer along the surface of a body, for it is only in this layer that the viscous forces are of sufficient magnitude to have any immediate effect on the flow. This does not mean that only the flow in the boundary layer is altered by the scale, because changes in the boundary layer may, and do in some cases, radically alter the general flow, e.g., the Prandtl experiment of boundary layer control in a diffuser. What it does mean is that when the ratio of mass to viscous forces in a flow is changed by changing the density of the air in the tunnel, the immediate effects must be sought in the boundary layer where this ratio is sufficiently large to be of importance.

2. The character of the following in the boundary layer is important in determining the aerodynamic characteristics of airfoils for two reasons; first, because it influences directly the skin friction on the surface which in turn determines a considerable part of the profile drag of good airfoils, and second, because it controls the position of separation points of the fall and consequently the character of the entire flow under certain conditions. The character of the flow in the boundary layer may be described as either laminar or eddying (following Dryden's nomenclature). The direction of flow in either case may become reversed, thus producing a separation point, but the reversal is resisted to a greater extent by the eddying boundary layer. In general, the boundary layer over an airfoil is partially eddying and partially laminar, the amount of each type depending on the Reynolds number as well as on other conditions. Both the drag and the conditions under which reversed flow takes place depend on the Reynolds number of the boundary layer as well as on the character of the boundary layer, whether eddying or laminar, but according to different laws for the eddying end for the laminar boundary players.

3. The point of transition from laminar to eddying flow in the boundary layer is therefore of particular importance. It depends, of course, on the shade and attitude of the airfoil, but of greater interest from the standpoint of scale effect is it gets a variation with Reynolds number. Dryden and Kuethe have considered the relation of the transition point on airship forms to the Reynolds number and to the initial turbulence of the airstream. It is probable that any investigation of a similar nature, applied to airfoils in the Variable Density Wind Tunnel over a wide range of the Reynolds number, would throw light on the causes and mechanism of scale effect. The character of the airfoil surface might be expected to have a marked effect on the position of the transition point so that the effects of different surfaces should be studied as well as the effects of changes in initial turbulence and Reynolds number.

Part of the work in connection with the investigation could be done under Research Authorization Number 177, "Determination of Effect of Polish of the Surface on Airfoil Models," and the part dealing with the effect of turbulence could be done under Research Authorization No. 203, "Study of Characteristics of Very Thick Airfoil Sections," or it could be done under Research Authorization No. 88, "An Investigation in the Variable Density Wind Tunnel of Scale Effect on Airfoils." The results of the investigation would determine the advisability of preparing a special research authorization providing for an investigation of the mechanism of the scale effect on airfoils, and would indicate more clearly how such an investigation should proceed.

Eastman N. Jacobs
Associate Aeronautical Engineer

Document 4-17

B. Melvill Jones, "Flight Experiments on the Boundary Layer," *Journal of the Aeronautical Sciences* 5 (January 1938): 81-101. First Wright Brother's [sic] Lecture, Presented before the Institute of the Aeronautical Sciences at Columbia University, New York, 17 December 1937.

John Anderson has called B. Melvill Jones's First Wright Brother's Lecture of December 1937 "a fitting closure to one phase in the development of applied aerodynamics in the era of the advanced propeller-driven airplane" (*A History of Aerodynamics*, p. 354). This was the phase when the "call to action" for aeronautical engineers was to "Streamline" – to design aerodynamic bodies with the lowest possible form drag. Jones himself had ushered in the new age in 1929, when he delivered his famous address "The Streamline Airplane" to the Royal Aeronautical Society (see Chapter 3). An incredible amount of worthwhile streamlining for airplanes was done in the next eight years, including the NACA low-drag cowling, retractable landing gear, stressed-skin aluminum structures, flush riveting, more efficient airfoils, and much more. In 1937, Jones was telling his audience that the next hurdle was to reduce friction drag, the most significant major source of drag that remained—and the one that would be the hardest to do anything about.

Over 300 members of the Institute of the Aeronautical Sciences heard Jones's talk, including Orville Wright. Four days later, on 21 December 1937, he repeated it at an IAS meeting at Caltech. Both talks stimulated a major—and a very positive—reaction. A Douglas aircraft engineer who attended the West Coast talk by the name of Francis Clauser recognized what Anderson has called "the historical full-circle significance" (*A History of Aerodynamics*, pp. 354-5) of Jones's address: "It was a pleasure to hear from the man who provided the stimulation some years ago which has led to the practical elimination of unnecessary form drag in modern airplanes and it is reassuring that this same man is now engaged in research which may conceivably reduce the remaining skin friction to some fraction of its present value." The full account of Clauser's remarks can be read at the end of Jones's paper, in the commentary section.

Notably, the IAS also chose the NACA's George Lewis and Eastman Jacobs to be two of the select individuals commenting on Jones's historic paper. It is interesting that Lewis, from his special perspective as a research director, commented on "the many different problems of experimental technique," which required "the utmost ingenuity to solve." Jacobs, on the other hand, dealt squarely with the promise of laminar-flow airfoils implied by Jones's boundary-layer work. From his standpoint, "the outstanding result" of Jones's tests was that "he has definitely shown what we have suspected for a long time: that extensive laminar layers must be recognized as possibly existing on actual airplanes in flight." The outstanding question still needing an answer, in his mind, was "How much further can we go in maintaining these desirable low-drag laminar layers?"

Jacobs left his seat at Melvill Jones's talk determined to answer the question.

Document 4-17, B. Melvill Jones, "Flight Experiments on the Boundary Layer,"* Journal of the Aeronautical Sciences *5, January 1938.

JOURNAL OF THE AERONAUTICAL SCIENCES

JANUARY, 1938

FLIGHT EXPERIMENTS ON THE BOUNDARY LAYER

B. Melville Jones, Cambridge University, England

First Wright brothers' Lecture
Presented before the Institute of the Aeronautical Sciences at Columbia University, New York
December 17, 1937

FOREWORD

This lecture is to be the first of a series to be delivered annually in honor of those famous pioneers the Wright brothers. I am told that the lecture itself should be severely technical and should not deal in compliments, but it is right for me to record, before beginning, the deep admiration which I have always had for Wilbur and Orville Wright, ever since the time when they were the half mythical heroes of my school days. I am acutely aware of the honor which you have done me in asking me to inaugurate a series of lectures in their honor. I shall not discuss the work of the Wrights, which is familiar to all, and it would be an impertinence to attempt elaborate praise of men whose names will remain household words when I and the majority of those present have been long forgotten.

INTRODUCTION

The authorities of the Institute of the Aeronautical Sciences have decided, so I am instructed, that the Wright brothers' Lecture should deal with subjects upon which the lecturer is engaged at the time, rather than with a general survey of some wide branch of aeronautical knowledge. This decision has the advantage that the lecturer is actively interested in the subject about which he talks, but it leaves to chance the question whether he is in a position to end his lecture with simple and clear cut conclusions. I mention this because the problem upon which we are working at Cambridge, and about which I shall speak, is not yet solved and my lecture must, perforce, be confined to a discussion of aims and methods and of results so far obtained; it does not contain that simple statement of conclusions which is the ultimate aim of all good research. After this explanation you will not, I hope, be disappointed when the lecture ends on a note of interrogation.

The material from which the lecture has been constructed is drawn mainly from experiments made in flight at Cambridge, but in order to make it as complete as possible, results from Government Research Establishments with which we work in close cooperation are quoted. For permission to do this I have to thank the British Air Ministry and the various persons directly concerned. I have also to thank the British Aeronautical Research Committee for permission to use information which has been submitted to them but which, at the time of writing, has not been published.

The title of the lecture is Flight Experiments on the Boundary Layer and it deals more specifically with the transition of the layer from the laminar to the turbulent form. Everyone interested in modern aeronautics is of course well aware of the general field of knowledge surrounding this subject; that the resistance to motion, of modern aircraft arises mainly from the friction of air acting upon exposed surfaces; that this skin-friction, as it is called, is applied in a comparatively thin layer of air immediately overlying the exposed surfaces; that these "boundary layers" in the air take one of two forms, a smooth or "laminar" form near the front of the exposed surface, and a turbulent form towards the rear; finally that the friction of the laminar layer is much less than that of the turbulent layer, so that the mean friction coefficient of the whole exposed surface—the figure used by the designer in laying out his performance chart—depends upon the precise location on the surface of the line at which transition occurs.

The order of magnitude of the changes which occur in the drag of a smooth wing, when the point of transition from the laminar to turbulent flow moves forward or backward along the wing profile, is illustrated in Fig. 1. Here, ordinates represent the conventional profile-drag coefficient and abscissae relate to the mean distance—measured parallel to the wing chord—between the leading edge of the wing and the points where turbulence begins on the upper and lower surfaces, respectively. The drag coefficients were obtained in flight by the now well known method in which small pitot and static-pressure tubes are made to traverse the wing wake; the transition points were located by methods shortly to be described. The use of the mean transition point as a basis for plotting is open to objection because the velocity distributions on the two surfaces are not in general the same, but some sacrifice of precision is justified in order to bring the various results together on a simple diagram suitable for a preliminary survey of the situation. The precise positions of the points where turbulence was found to begin on each surface in various circumstances will be considered later.

The individual points in Fig. 1 relate to wings of various thickness, with lift coefficients between 0.3 and 0.4; the crosses relate to experiments made at Cambridge, and the other points to experiments made by the Royal Aircraft Establishment, Farnborough. To obtain the two points for the wing of thickness ratio 0.10 the point of transition on the upper surface was fixed at x/c = 0.24 and 0.07, respectively (see legend of Fig. 1 for definitions), whilst that on the under surface remained unaltered at about x/c = 0.16; the points on the upper surface were fixed by attaching wires of about 0.01 in. diameter transversely on the surface. To obtain the two more forward points for the wing of thickness ratio 0.30, the points of transition on both the upper and under surfaces were fixed by sticking thin paper sheets to the surfaces with their front edges in appropriate positions for which x/c was the same for both surfaces. For all the other points in Fig. 1., transition occurred spontaneously on surfaces which had been carefully smoothed and polished. The point corresponding to spontaneous transition for the wing of thickness ratio 0.10 has not been plotted because it lies very close to the more rearward of the two points for which transition was controlled.

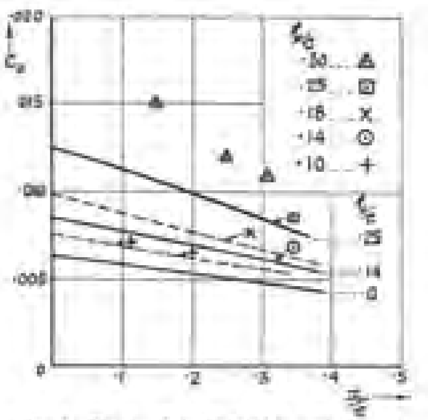


Fig. 1. Drag coefficient related to point of transition. Ordinates represent conventional profile drag coefficients. Abscissae show the mean values of the distances s from the leading edge of the points of transition from laminar to turbulent flow on the upper and under surfaces of various wings. The distance s is measured parallel to the wing chord which is of length c. The maximum wing thickness is represented by t. The drag coefficients shown by the curves are computed as described in the text. The figure relates to a wing Reynolds Number of seven millions and slight adjustments of the order 2 percent to 3 percent have in some cases been made to the observed drag values, to allow for the fact that they were obtained when the Reynolds Number had values ranging from five to eight millions.

The continuous curves in Fig. 1 are from computations made with certain simplifying assumptions. That for which t/c is zero relates to an ideally smooth thin flat plate and is built up from the Blasius solution for the laminar layer and the Prandtl-Karman logarithmic curve for the friction of the turbulent layer, the change from laminar to turbulent flow being assumed to occur suddenly and without change of momentum loss. The two continuous curves for which t/c is 0.14 and 0.25, respectively, are from calculations made at Farnborough by H. B. Squire and A. D. Young. In these the skin friction of the laminar layer was calculated step-by-step along the wing profile, using Polhausen's approximate method of representing the velocity cross-sections by fourth power polynomials; the friction of the turbulent part of the layer was computed by a similar process on the assumption that the velocity cross-section of the layer retains a constant form. Here also transition was assumed to occur suddenly without change of momentum loss. The

drag coefficients shown by these curves are greater than the resultants of the skin friction forces, because they contain some "form drag" which can be shown to be an inevitable consequence of skin friction acting upon a wing of finite thickness. The intermediate broken curves have been obtained by interpolation from the computed continuous curves and have been added merely to aid comparison with the experimental points for wings of corresponding thickness ratio.

Though the basis on which Fig. 1 is constructed is not such as to allow fine points of difference to be examined, the figure suffices to illustrate clearly the three main conclusions of practical importance which can be drawn from the experiments to be discussed. These are: (1) That when the transition points are known the profile drags of smooth fair-shaped wings of moderate thickness can be computed with sufficient accuracy for most practical purposes from the known values of skin friction on a smooth flat plate. (2) That at moderate values of the wing Reynolds number—five to ten million—transition can be postponed to distances greater than 0.3 chords from the leading edge, with a consequent reduction of drag of the order 30 to 35 percent of the drag with the layer wholly turbulent. (3) That very small roughnesses or imperfections of surface are sufficient to move transition points forward and so increase drag.

In relation to the third conclusion it may be mentioned that, in one instance at Cambridge, when the wing Reynolds number was about ten millions, a piece of tinfoil 0.002 in. thick, stuck down on the wing surface, appreciably influenced the position of transition. Again, at Farnborough, the drag of a smooth wing—measured by the pitot-traverse method—was appreciably increased when the aeroplane had flown through a cloud and this is considered to have been the result of a forward movement of the transition point, caused by mist drops deposited on the wing surface. Very small—barely perceptible—waves on the wing surface have also been shown materially to affect the point of transition and therefore the drag.

These considerations show the practical importance of knowing where upon the wings and other exposed surfaces of an aeroplane the boundary layer passes from the laminar to the turbulent flow, and they explain why the factors which influence the onset of turbulence are occupying the attention of many aeronautical research laboratories besides that at Cambridge.

METHODS OF EXPERIMENT

It is of course well known that the transition of the boundary layer from the laminar to the turbulent form is a gradual process, so that strictly one should speak of a transition region rather than of a transition point. The experiments of H. L. Dryden and others have shown also that the transition region itself does not remain stationary, but is subject to rapid to-and-fro movements, so that in strict accuracy one cannot speak of a transition point without first defining it in relation to the mean position of a fluctuating transition region. In the experiments to be described, however, the effects of rapid fluctuations are automatically meaned by the slow response of the apparatus, and its indication of transition is sufficiently sharp to define a point on the wing surface which for practical purposes can conveniently be called the transition point.

The hot wire anemometer which has been widely used for the study of transition in the laboratory is not a convenient instrument for use in flight, and an alternative method involving very small pitot tubes has therefore been developed. This method depends on the changes which occur during transition in the mean velocity cross-section of the boundary layer. Typical velocity cross-sections just before and just after transition are shown in the left-hand diagram of Fig. 2, whilst the right-hand diagram of that figure shows, in sketch form, the changes which would occur in the pressure registered by a very small pitot tube moved through the transition region along lines lying parallel to the wing surface, such as AA, BB, etc. in the left-hand diagram. It can be seen from this figure that a small pitot tube moved in the direction of flow at a constant distance from the wing surface, along a line such as AA which just before transition lies just outside the laminar layer, will register, as indicated by line A of the right-hand diagram, a small fall of pressure as it passes into the thicker turbulent layer. Pitot tubes moved in a similar manner along lines, such as BB or CC, situated closer to the wing surface will, on the other hand, register a rise of pressure as they pass through the transition region. The distance along the wing surface within which these changes of mean pressure occur varies of course with circumstances, but in the experiments to be described it was generally of the order four inches.

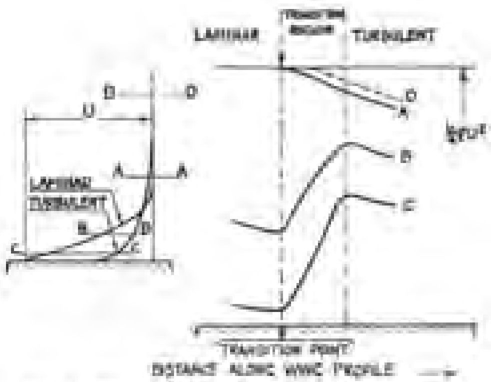


Fig. 2. Diagrams for explaining the methods used for locating the region of transition to turbulent flow. The left-hand diagram shows typical distributions of mean velocity in the boundary layer just before and just after the onset of turbulence. The right-hand diagram shows the variation of pressure in a pitot tube moved through the transition region along lines parallel to the wing surface at various distances from it.

The phenomena described above can obviously be used to detect transition. In wind tunnel experiments it is generally more convenient to use pitot tubes close to the wing surface, in which pressure rises as the pitot passes into the turbulent part of the layer, for this method gives, when the external stream is smooth, a very precise indication of the first onset of turbulence and, since the exploring pitot can be placed in actual contact with the solid surface, it is not necessary to make any but a rough estimate of the thickness of the laminar layer before beginning the experiment.

In flight experiments, on the other hand, it is generally more convenient to use a pitot tube which lies altogether outside the laminar layer and in which pressure falls as it moves into the turbulent layer. There are two reasons for this: one is that, in straight flight at altitudes where the air is steady, total-pressure is very accurately the same at all parts of the potential stream, so that an exploring pitot, coupled through a manometer to another pitot conveniently situated anywhere in the potential stream, enables minute defects in total-pressure to be easily detected: the other is that the bore of the outside pitot tube can be made much larger than that of the inside pitot. The latter is a great advantage in flight experiments because a small pitot-tube implies a large lag in the response of the manometer to pressure changes and a possibility of error due to change of pressure and temperature consequent upon an accidental or deliberate change of height. This latter consideration is not so important in laboratory experiments where the capacity of the connecting tubes can be kept small and the external pressure and temperature can be maintained more nearly constant.

At first sight it might be supposed that the method of the outside pitot suffers from the severe disadvantage that the thickness of the laminar layer must be accurately known before the experiment begins, but a rough consideration of the quantities involved is sufficient to show that this is not so, unless very great accuracy in location of the transition point is required. The slope relative to the wing surface, of the effective outside boundary of the layer in the transition region is of the order 1/25, so that a displacement of the outside pitot away from the wing surface by as much as 0.1 in. will shift the point where total-pressure loss is first detected by no more than 2.5 in. The effective thickness of the laminar layer just before transition upon a smooth wing at ordinary flight speeds is seldom much greater than 0.05 in. so that no great percentage accuracy in the estimation of its thickness is necessary in order to adjust the distance of the outside pitot from the wing surface so that it always lies outside the laminar layer, without being so far away from it as to make

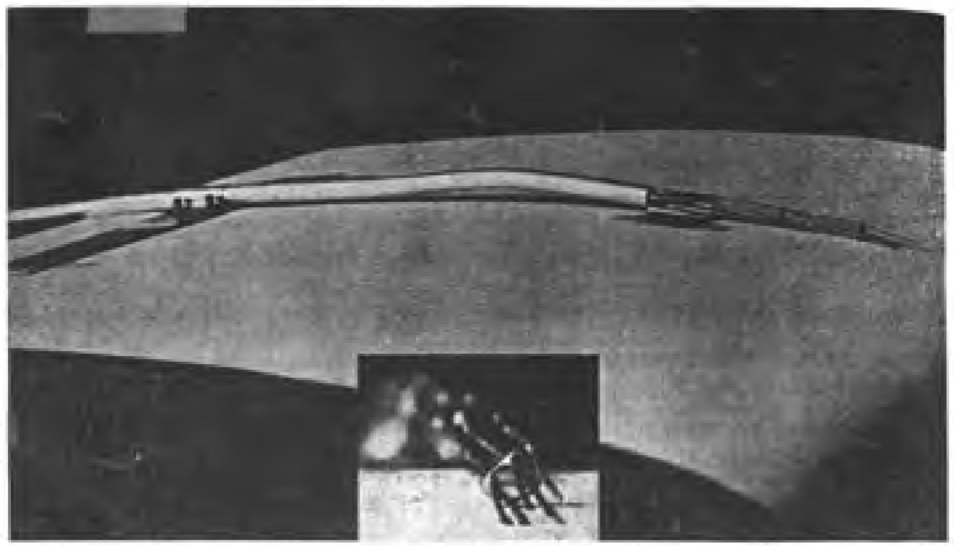


Fig. 3. Photographs of the five-tube head used for exploring the transition region in flight. The four tubes with orifices in the form of horizontal slits register total pressure at various distances from the wing surface, their outside dimensions at the front end are 0.012 in. deep and 0.064 in. wide. The tube with round closed end registers static-pressure.

the position of the transition point uncertain by more than 1 or 2 inches. For example, the pressure variations in a pitot moved along DD in Fig. 2 would be roughly as indicated by the broken curve D in the right-hand diagram, and the corresponding shift in the supposed position of the transition point would be no more than some 2 in. It is apparent from this consideration why much larger pitot tubes can be used by the outside method than by the inside method.

The most recent form taken by the pressure heads used in flight experiments at Cambridge is shown in Fig. 3. Here five tubes are used, each of 0.042 in. external diameter. One of these is a static-pressure tube by means of which the pressure distribution along the wing profile can be recorded. The other four are pitot tubes with flattened orifices of external depth (perpendicular to the wing surface) of 0.012 in. and width 0.064 in. One of these tubes, known as the surface pitot, is in actual contact with the wing surface, whilst the other three are situated at various distances from the wing surface, such that two lie within the laminar layer and the third lies outside it. A convenient method of using this group of tubes is to fix them in some chosen position on the wing and record the pressures in them when the aeroplane is flown at various steady speeds. The change of incidence and of Reynolds number consequent upon change of flight speed cause the transition region to move along the wing profile and, if the position occupied by the tubes lies within the travel of the region, the conditions under which it passes them can be determined.

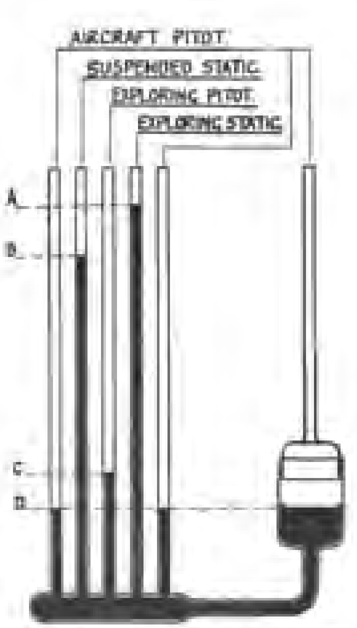


Fig. 4. Diagram illustrating manometer connections. [illegible]

The manometer used in these experiments was of the multiple "U" tube type illustrated diagrammatically in Fig. 4. The tubes contain alcohol and records were made of the shadow of the meniscus thrown upon a sensitized paper close behind the tubes. This manometer enabled pressure differences to be measured to within about 1 percent of the impact pressure at the lower flight speeds and of course with greater accuracy at greater speeds.

A typical record from this instrument is shown plotted on a lift coefficient base in Fig. 5. In this instance the transition point moved backwards on the wing with increase of lift coefficient, so that the right-hand size of the figure relates to a laminar boundary layer and the transition point coincided with the orifices of the tubes at lift coefficient 0.55. Judging from the rate of movement of the transition point with change of lift coefficient, estimated from this and similar experiments with the tubes fixed in different positions on the wing, the transition region illustrated in this figure occupied about 4 inches of the wing profile.

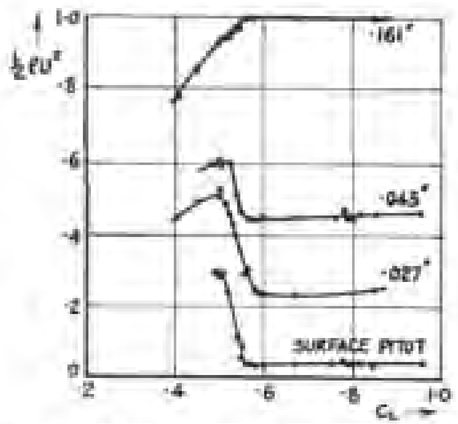


Fig. 5. Typical observations of total pressure in the transition region. [illegible]

The rather complicated arrangement of five tubes described above was devised to enable the form of the laminar layer just before transition and the character of the transition region itself to be investigated; but when no more is required than to determine approximately the point where turbulence begins, much simpler arrangements can be used. In the earliest flight experiments on this subject at Cambridge a single pitot tube only was used, with circular end about 0.05 in. external diameter. This was stuck on to the wing surface with adhesive tape and the front bent up slightly so that the orifice lay just outside the estimated thickness of the laminar layer. In this simple way it is easy to find the position of the transition point on a wing within about a couple of inches, but unless the position of the tube can be altered in flight or the position of the transition region is roughly known beforehand, two or three flights may be necessary.

A refinement is to provide a second pitot tube and manometer and to place this second tube about twice as far from the wing surface as the first. This enables the slope of the outer boundary of the layer to be roughly determined and, since in the transition region this slope is some ten to twenty times as great as that of the laminar layer just before transition, all doubt as to whether transition has really occurred can be removed. After a little experience this refinement becomes unnecessary because, in the transition region, the characteristic shape of the curve of total pressure against distance along the wing is easily distinguished from the shape which results when the pitot merely enters gradually into the thickening laminar layer.

By such simple devices transition points could, if desired, be quickly and easily determined in testing organizations such as those of manufacturing firms, where the object is rather to ascertain what happens to specific aeroplanes than—as in research organizations—to investigate the general character of new phenomena.

The majority of the experimental results which will shortly be discussed were obtained with apparatus of complication intermediate between the five-tube arrangement illustrated in Fig. 3 and the single fixed pitot tube mentioned above.

At Cambridge, for example, the greater part of the work has been done with a three-tube instrument, of which one tube was in the form of a static-pressure tube whilst the other two were circular-ended pitot tubes both situated outside the laminar layer. These three tubes were rigidly connected together so that their working ends formed a small triangle and the whole was mounted on the wing in such a way that the tubes could be pushed by a small electric motor forwards and backwards along the wing profile, the object of the movement being to allow a larger number of observations to be made in a single flight. At Farnborough a somewhat similar arrangement of four tubes—one static-pressure and three pitot—has been extensively employed.

RESULTS OF EXPERIMENTS

The experimental observations which are now to be considered were obtained in flight upon portions of wings of which the surfaces were carefully smoothed and polished. The coverings in some cases were considerably thicker than is usual in present practice, the object being to eliminate as far as possible the slight waviness which is often observed on wing surfaces and is the cause of local variations in the pressure gradients along the profile. Even so, the makers did not always succeed in eliminating every trace of surface waves, the effect of which on the observations were exceedingly interesting. The observations have all been made along profiles which lay between the body and inner ends of the ailerons, sufficiently far from either it is thought to avoid interference from those parts or from the airscrew slipstream.

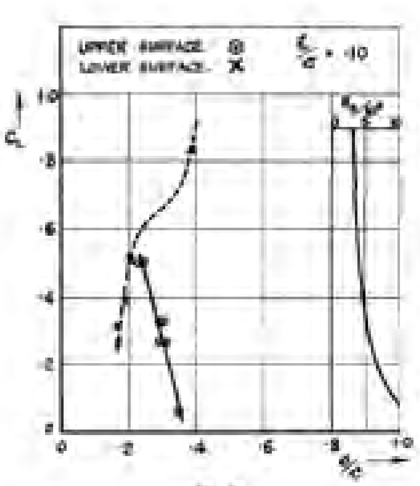
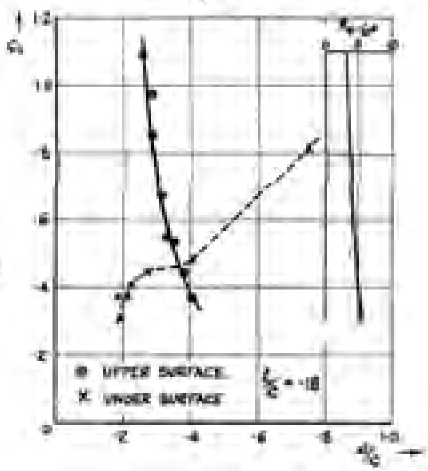


Fig. 6



The available information relating to the point of onset of turbulence on wings in flight is displayed in Figs. 6, 7, and 8, in which ordinates represent the lift coefficient of the aeroplane as a whole, and abscissae the distance of the transition point—measured round the wing profile—from the front stagnation point.

Fig. 6 shows the results of the first British experiments of this kind, which were made at Cambridge early in the present year upon the lower wing of a military biplane known as the Hart, the chord of which was 5 ft. in length. The approximate value of the wing Reynolds number appropriate to each lift coefficient is shown at the right-hand side of the figure. All the observations but one were made in level flight, with indicated flight speeds ranging from 60 to 120 m.p.h., at heights—round about 10,000 ft.—where the air was sufficiently calm to allow accurate observation. The exception is the point at lift coefficient 0.06; this point was obtained from observations made in long steep dives at an indicated air-speed of 240 m.p.h. The figure is built up from observations made in many different flights extending over several months.

On the upper surface of the wing the position of the transition point at each lift coefficient seemed to be very definitely established, but on the under surface, at lift coefficients round about 0.65, consistent results were more difficult to obtain, the position of the transition point being apparently very sensitive to air speed. A similar sensitivity at a different lift coefficient had been observed in some preliminary experiments made by very simple methods upon the same wing while it still had a standard fabric covering. Fig. 6 shows that at low lift coefficients the transition point on the under surface was much further forward than on the upper surface, but that as the lift coefficient increased the point on the under surface moved backwards whilst that on the upper surface moved forward.

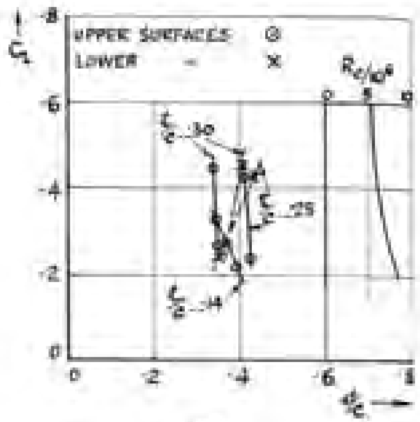


Fig. 8

Figs. 6, 7, 8.—Observations of the points of transition in the boundary layer of smooth wings in flight. Ordinates represent the lift coefficient for the whole aeroplane, abscissae the ratio to the wing chord (c) of distance—measured round the wing profile—between the front stagnation point and the point where turbulence first begins. The thickness-ratio t/c of the wing profile is shown in each figure. The approximate Reynolds Number R, of the wing at each value of the lift coefficient is shown by a curve on the right-hand side of the figure. Figs. 6 and 7 are from experiments at Cambridge and Fig. 8 from experiments at Farnborough.

Fig. 7 shows similar results obtained at Cambridge for a thicker wing of a small monoplane. In the place where the experiments were made the chord of this wing was 6.2 ft. in length; the maximum level speed of this aeroplane was, however, lower than that of the Hart, so that the Reynolds number realized in level flight was about the same. These experiments, like those on the Hart, involved many flights on different days. The two points shown by squares in the group relating to the upper surface of the wing came, in fact, from check experiments made six months after the experiments which gave the other points in the figure had been completed. As with the Hart, the upper surface points were relatively easy to obtain, whilst on the under surface there was a range of lift coefficients in this instance round about 0.45, within which the position of the point of transition moved rapidly with change of lift coefficient, and seemed sensitive to surrounding conditions. As with the Hart wing also, transition occurred relatively far forward on the under surface at small lift coefficients, and the point moved backwards on the under surface and forwards on the upper surface as lift coefficient increased.

An interesting feature of this and the previous figure is the accuracy with which the observation points fall upon definite curves, despite the fact that they were obtained from experiments on many different days; to the observers it seemed almost as though they were locating points actually fixed upon the wings. This surprising consistency suggests strongly that, in these experiments at least, the cause of transition is to be sought in the system of flow set up by the aeroplane itself and not in disturbances pre-existing in the atmosphere through which it flies; for it is unlikely that the nature and amount of atmospheric turbulence would remain exactly the same from day to day and from place to place. This deduction from flight experiment, if correct, is important, because wind tunnel experiments, upon which we have hitherto had to rely have shown transition to be strongly influenced by turbulence of the tunnel stream. Some years ago G. I. Taylor suggested that the rate of energy dissipation per unit volume of the atmosphere is too small for small scale turbulence of the kind which is known to influence boundary layer transition in wind tunnel experiments to be maintained, and it now seems probable, though not yet certain, that the atmosphere must from the present point of view be regarded as free from turbulence.

Fig. 8 shows the results of similar experiments at the Royal Aircraft Establishment, Farnborough, upon the wings of a small monoplane which was designed so that it could be reconstructed with wings of various thicknesses. The thickness/chord ratios of the wings examined were 0.14, 0.25, and 0.30, respectively. The two thinner wings were of conventional design tapering slightly from root to tip, but the profile of thickness ratio 0.30 was obtained by fitting a bulge on the 0.25 wing, which extended for about the half of one chord on each side of the measurement section. Previous wind tunnel experiments had shown that the boundary layer on a bulge of this kind is not seriously influenced by rapid changes of wing thickness on the sides of the bulge, but this matter is still under investigation. The main interest of these experiments lies in the fact that although the Reynolds number reached in level flight was appreciably higher than that in the Cambridge experiments the transition points were even further back on the wing and the marked forward movement, observed in the Cambridge experiments at low lift coefficients, did not occur, though there was still a slight tendency for the point to move forward on the under surface and backward on the upper surface as lift increased. It is not yet known whether the absence in the Farnborough experiments of this marked forward movement on the under surface was due to a better technique in laying on the wing covering or to a difference in the designed form of the wing profile; there is, as will be seen, some evidence of slight imperfections in the shape of the under surfaces of the Cambridge wings, although the surfaces were always well smoothed and highly polished. It is worth noting that Stuper in his experiments on the boundary layer of a wing in flight found transition points on the under surfaces in forward positions similar to those observed at Cambridge.

This, then, is the experimental evidence so far available in England upon the position of the point where turbulence begins in the boundary layer of smooth wings in flight. It shows conclusively that it is possible to retain a laminar layer over at least one-third of the whole wing surface, even when the Reynolds number is as high as eight millions. Experiments are now being made at Farnborough to carry observations of this kind up to larger Reynolds numbers, but conclusive results are not to hand at the time of writing. Drag experiments which have been already made at high Reynolds numbers by the pitot-traverse method suggest, however, that though the points of transition may move forward somewhat at the higher numbers, they certainly do not move right forward to the leading edge and the laminar form of the boundary layer can still be retained over a considerable proportion of the wing surface.

It remains to consider in rather more detail the circumstances in which transition occurred in the experiments which have been described, in order to see whether any light can be thrown upon the factors which influence it, but before this can be done it is necessary to give brief attention to some of the well known conclusions which can be drawn from a consideration of the dimensions of the quantities involved.

THEORETICAL CONSIDERATION

If the boundary layer is thin enough for Prandtl's well known approximation to the equations of motion to be applied, then it is easily shown that the shape of the velocity cross-section of the laminar layer at any given position on a wing of given shape at given incidence, is independent of the size of the wing, the speed of flight and the density and viscosity of the air. If δ be a linear dimension defining the thickness of the layer, then it can also be shown that

$$R_\delta^2 \propto R_c$$

where R_δ stands for $\delta\, U/v$ and R_c for $c\, U_0/v$ in which U_0 is the velocity relative to the wing of the undisturbed air at a great distance, U is the velocity of the air just outside the boundary layer, c is the wing chord, and v the kinematic viscosity of the air.

In analyzing experimental observations it is convenient to choose for the linear quantity δ the value known as the "displacement thickness" and to represent it by the symbol δ_* which is defined as follows:

$$\delta_* = 1/U \int_0^\infty (U - u)dy$$

where u stands for velocity within the layer at a point distant y from the wing surface.

For a flat plate in a flow field of uniform pressure, any of the well known mathematical solutions of Prandtl's approximate equations for motion within the laminar boundary layer give a relation which is very closely represented by

$$1/3\ R^2_{\delta^*} = R_z$$

where R_z stands for xu/v, in which x is distance from the leading edge.

The expression $(1/3)R^2_{\delta^*}$ therefore provides a convenient dimensionless quantity, or number, by which to define the thickness of the laminar layer at any position on a wing, for it has the properties that it varies directly with the wing Reynolds number (R_c) and is, for the flat plate, equal to the familiar x Reynolds number, by reference to which the already considerable amount of experimental data on transition is generally recorded. For brevity, this expression will be described as the *thickness number* and will be represented by the symbol N, so that

$$N = 1/3\ R^2_{\delta^*}$$

In considering the factors which may influence transition, the natural relation of the investigator is to examine first whether they can be defined in relation to the local conditions within and immediately surrounding the layer near the transition point. The extreme thinness of the layer in comparison with the size of the wing suggests that this may be so, and if it is so, and if it is unnecessary to take into account time fluctuations of the quantities involved suggests that transition should depend upon three parameters, N, λ, and C, of which N has already been defined, λ stands for $(\delta^2/v)(p'/\rho U)$, where p is the pressure gradient just outside the layer in the direction of flow and C stands for the ratio of δ_* to the radius of curvature of the wing surface in a plane parallel to the direction of flow.

It does not follow from these considerations that transition must depend solely, or even primarily, upon the three dimensionless parameters N, λ, and C; in fact the weight of evidence is, as will be seen, against this. The dimensionless analysis does no more than show that if, as seems at first sight probable, the phenomenon under consideration depends only on the variables from which the parameters are constructed, then it must be possible to express the conditions under which it occurs as functions of the parameters.

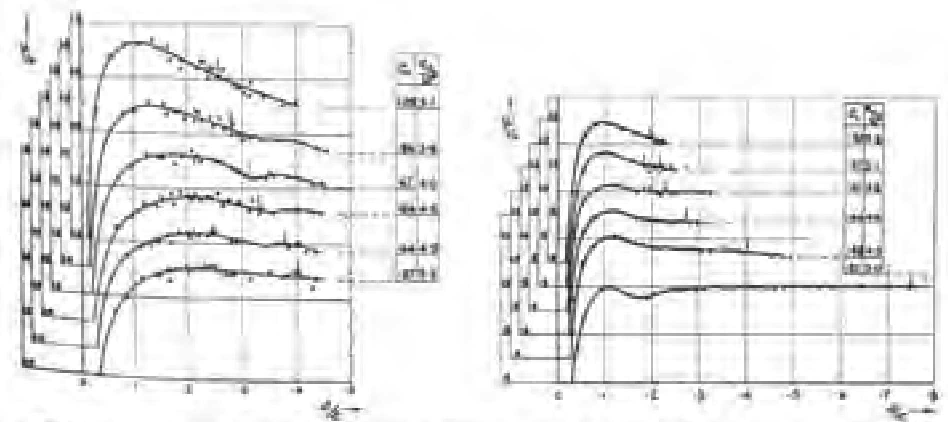
It is not impossible, of course, that the conditions which govern the onset of turbulence may include other factors, such for example as something which has occurred during the passage of the air from the front stagnation point to the point of transition, or even conceivably something which would happen in the laminar layer if it continued beyond the point where transition occurs in steady flight. It is conceivable that transition may occur first in some part of the boundary layer much

further back on the wing and that the turbulence then runs forward under the influence of the pressure gradients set up locally in the potential flow by the rapid increase of boundary layer thickness which accompanies transition. Again, it must be remembered that the quantities which have so far been measured in flight are time-means only, of quantities which may have been fluctuating and it is therefore possible that the primary cause of transition may ultimately be associated with fluctuations which have not been recorded in the flight experiments.

CONSIDERATION OF THE FACTORS WHICH MAY CONTROL TRANSITION

In order to examine whether the factors which control the transition from laminar to turbulent flow in the boundary layer can be simply expressed in terms of the three dimensionless parameters, N, λ, C, of the previous section, the values of these parameters have been estimated by step-by-step computation, starting from the front stagnation point and working backwards along the wing profiles to the observed transition points. In these computations the velocity U in the potential flow just outside the boundary layer and the pressure gradient p' along the profile were obtained from curves based on pressures actually registered in flight by the small static-pressure tubes previously described. Graphs, based on a typical series of such observations of pressure, are reproduced in Figs. 9 and 10 to show the order of accuracy attained in the experiment. The quantity actually plotted in these figures is not the pressure itself, but the velocity U just outside the boundary layer which can, of course, be deduced from the pressure by using the Bernoulli Theorem.

The computations of the values of the parameters were made by Polhausen's method in which velocity cross-sections of the boundary layer are represented as polynomials of the fourth order. This method is approximate only, and doubt has been expressed as to its accuracy when applied in circumstances where the param-



Figs. 9 and 10. Specimen curves from observations of pressure distribution measured in flight. Ordinates show ratios of the speed U relative to the wing of the air just outside the boundary layer to the speed U_0 of the air at a great distance from the wing, the values being obtained by means of the Bernoulli Theorem from measurements of pressure made with a small travelling static-pressure tube similar to that illustrated in Fig. 7. Abscissae represent the ratio to the wing chord c of distance s, measured round the wing profile, from the stagnation point to the observation point. The curves are fair curves drawn through the observation points and the short vertical marks upon them show where turbulence was observed to begin.

eter λ is negative. Some experimental check was therefore required to determine whether the method gives sufficiently accurate information in the circumstances in which it has been applied. For this purpose a very careful set of observations were made in flight with the five-tube arrangement illustrated in Fig. 3. These tubes were fastened at a fixed position on the surface of the wing of thickness ratio 0.18 and the pressures registered by them were recorded when flying steadily with various values of the lift coefficient. The velocity cross-sections of the boundary layer at the orifices of the tubes were then computed from pressure distributions obtained in previous experiments.

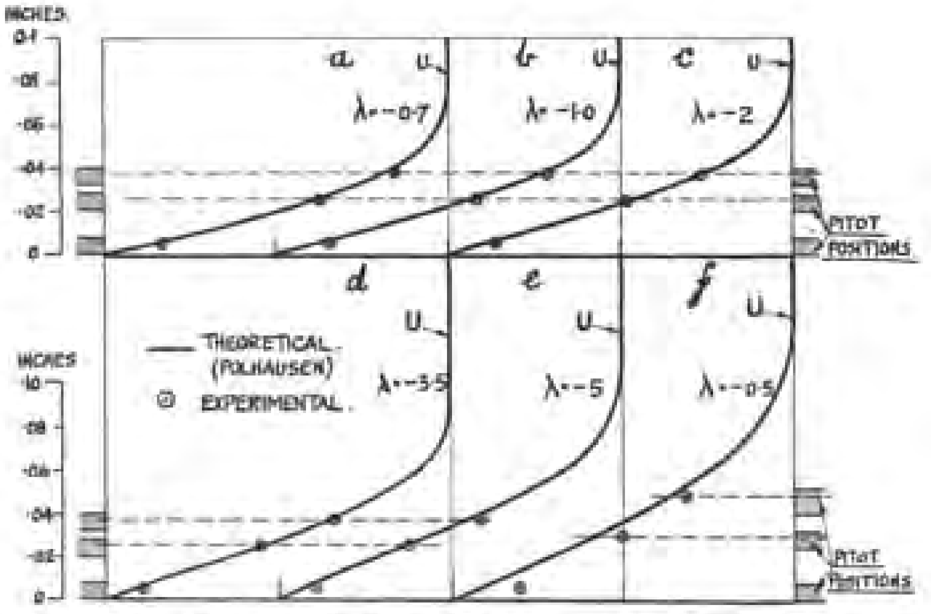


Fig. 11. Comparison between computed and observed velocity cross-sections of the laminar boundary layer. Ordinates represent distance from the wing surface and abscissae air velocity expressed as a fraction of the velocity outside the layer. The curves are computed by Polhausen's method and the points are from flight experiments with pitot tubes whose positions and depths are shown at the sides of the figure. The effective centres of the two outer pitots are taken to lie on the broken lines which are displaced from the geometrical centres by distances equal to 0.2 times the external depth of the flattened ends of the tubes, in accordance with the conclusions of footnote 11. The effective distance of the [illegible] from the wing surface are estimated from information in footnote 12.

A comparison between computed and observed values is shown in Fig. 11 (a) to (f), in which (a) to (e) relate to a position on the upper surface of the wing, 20 in. behind the stagnation point, measured along the surface, whilst (f) relates to a position on the under surface 26.5 in. behind the stagnation point. The remarkably close agreement shown in Figs. (a) to (e) may be to some extent accidental, for it is not considered that the distances of the pitot tubes from the wing surface were known with certainty to an accuracy greater than about 0.004 in., but the experiment shows that, in this instance, Polhausen's method gave δ_* values which agreed with the observed values within, say, 10 percent, even when the negative values of λ were as great numerically as five.

It is of some interest to consider the variation of the computed values of the parameters N, λ, and C at different positions along the wing profile, between the forward stagnation point and the point of transition and a few typical curves illustrating these distributions are therefore reproduced in Fig. 12. In this figure the parameter N is defined, not by its actual value, which varies with the wing's Reynolds number R_c, but by the ratio N/R_c, which, for any given position on a wing of given shape at given lift coefficient, is a constant independent of R_c. For a flat in a uniform pressure field, N is, as has been seen, equal to R, where s is the distance from the leading edge, hence the variation of N/R_c with s/c for a flat plate is represented in Fig. 12 by a straight line inclined at 45°.

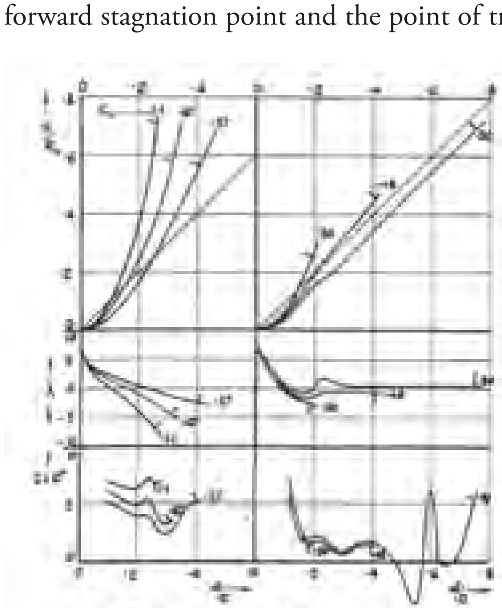


Fig. 12. Examples of the distribution of certain dimensionless quantities which define local conditions in the laminar boundary layer. The thickness/chord ratio of the wing to which these curves relate was 0.18. Lift coefficients are written against the curves, and the observed points of transition are shown by short cross-lines on each curve. Abscissae represent ratios to the wing chord c of distances s, measured along the wing profile from the front stagnation point. Of the other symbols, N defines the layer thickness, λ the pressure gradient within the layer, and C the curvature of the wing profile; they are more precisely defined in the text. Negative values of λ correspond to rising pressure gradients and separation of flow from the wing surface is generally assumed to occur when λ = −12.

The radius of curvature of the wing surface which is involved in the parameter C was obtained by an instrument of which two points, 6 in. apart, were pressed lightly on the surface, whilst a third, midway between them, actuated a micrometer dial. The measured radii were, therefore, mean values over 6 in. of the profile.

The curves in Fig. 12 are of some general interest, because they are fairly well typical of any wing of moderate thickness and conventional profile. Such curves can, of course, be obtained for any wing profile by computation alone, without experimental determination of the pressure distribution, since the latter can be obtained in the usual way from potential flow theory, but in the present instance it was thought preferable to employ the measured pressure distributions so as to include the effects of accidental variations of the actual wing profile from the designed profile.

The point where turbulence was observed to begin in the experiments from which the pressure distributions were obtained is marked on each curve and, where the curve is carried slightly beyond this point, the implication is that the laminar layer would have taken the computed form if turbulence had not, in fact, intervened. A difficulty of interpretation arises here on account of the characteristic kink which always occurs in the pressure distribution curves near the point of transi-

TABLE I

Data Relating to the Laminar Boundary Layer of Certain Wings, from Computations Based on Measured Pressure Distributions

			C_L	$R_c/10^6$	s/c	λ	$N/10^3$	$C \times 10^4$
CAMBRIDGE	t/c = .10	Upper Surface	.06	10.8	.35	−1.0	3.7	1
			.27	5.8	.32	−2.1	2.0	2
			.27	5.5	.30	−2.2	2.4	2
			.33	5.2	.30	−2.2	2.1	2
			.33	4.9	.30	−2.1	1.9	2
			.50	4.0	.23	−3.7	1.5	2
			.50	4.2	.25	−4.1	1.6	2
		Under	.27	5.1	.17	+0.1	0.8	1
			.32	4.8	.17	−0.4	0.8	1
			.38	4.4	.19	0.0	0.7	1
			.51	3.9	.21	+0.4	0.6	2
			.84		.40			
	t/c = .18	Upper	.37	5.3	.40	−2.2	3.0	5
			.44	4.9	.38	−2.5	2.8	5
			.54	4.5	.33	−3.5	2.4	4
			.67	4.0	.31	−4.6	2.3	4
			.85	3.6	.28	−5.2	2.2	5
			1.09	3.1	.26	−7.2	2.2	7
		Under	.30	5.6	.19	−4.4	1.0	1
			.37	5.8	.18	−3.1	1.1	1
			.37	5.1	.22	−4.2	1.5	1
			.41	4.8	.23	−2.1	1.2	1
			.44	4.6	.27	−0.8	1.3	1
			.48	4.3	.40	−1.0	1.9	1
			.82		.75	0.0	2.3	5
FARNBOROUGH	.14	Up.	.33	6.8	.35	−2.8	2.8	3
			.22	8.2	.40	−3.7	3.7	3
	.25	Up.	.43	5.7	.41	−6.9	4.3	5
			.24	7.7	.42	−5.3	4.8	4
		Un.	.43	5.7	.43	−4.9	2.4	4
			.24	7.7	.35	−3.1	3.3	3
	.30	Up.	.45	5.6	.34	−4.0	3.2	4
			.27	7.4	.35	−3.9	4.1	4
		Un.	.45	5.6	.40	−1.3	2.9	4
			.27	7.4	.37	−2.0	3.9	4

tion (see Figs. 9 and 10). In the present instance all quoted values of λ have been obtained from fair curves drawn through the points without regard to the kink, the reason for this procedure being that the kink is regarded as a consequence of transition itself, and the ultimate object of the research is to find how far it is possible to predict the point of transition from estimated information concerning the laminar layer in the absence of transition.

The N/R_c curves at the top of Fig. 12 show that near the front of a fairly thick wing the displacement thickness of the laminar layer at any given distance from the stagnation point is much less than it would be in corresponding circumstances at the same distance from the leading edge of a flat plate. They show, however, that as distance from the stagnation point increases the thickness of the layer begins to increase more rapidly than on a flat plate, until somewhere not far from the point on minimum pressure ($\lambda = 0$) the local Reynolds number of the layer thickness catches up the flat plate values and eventually rises considerably above them. Only in one curve, that relating to the under surface at very large lift coefficients, does the displacement thickness remain always below the corresponding flat plate values.

The λ curves, half way down Fig. 12, all start at the front stagnation point of the wing from Polhausen's figure, −7.05, and fall rapidly as distance from the stagnation point increases. They pass, of course, through zero at the point of minimum pressure, and on the upper surfaces the negative values thenceforward increase numerically up to the point where turbulence begins. The values for the under surface at first fall off more rapidly than those of the upper surface and, after passing through zero, rapidly rise, sometimes crossing the zero line again before they eventually settle down to a value not far different from zero. It seems probable that the more violent bends in these under surface λ curves are to be attributed to some imperfection in the shape of the surface itself, and it is not improbable that they are in some way associated with the relatively early onset of turbulence observed on the under surface of this wing. Similar double bends associated with forward positions of the transition point were also noticeable in the corresponding curves—not here reproduced—for the under surface of the wing of 10 percent thickness/chord ratio.

The curves at the bottom of Fig. 12 show approximately the variations of the parameter C. The violent fluctuations of the curve for the under surface between lift coefficients 0.5 and 0.8 are due to an imperfection of the wing surface in the form of a barely perceptible wave. It is of some interest to observe that this violent fluctuation of the C curve did not immediately cause transition to the turbulent regime.

The computed values of the three parameters N, λ, and C, relating to the laminar boundary layer at points where transition to the turbulent form had been observe in flight, are collected together in Table 1, and the values of N given in this Table are plotted against the appropriate values of λ in Fig. 13. It appears from the Table and figure that the λ values at transition all lie, speaking broadly, between 0 and −7, but that within this range there appears to be no simple relation between the two parameters.

The situation is not appreciably clarified when the values of the curvature parameter C are taken into consideration, for although Table 1 shows that the very low values of N and λ observed on the under surfaces of the Cambridge wings are associated with low C values, the very high values of N and λ found at Farnborough are associated with lower C than those of many of the Cambridge upper surface observations. Though the observations recorded in Table 1 do not preclude the hypothesis that surface curvature may have some influence on transition, they certainly show that in these experiments it did not exert the predominating influence.

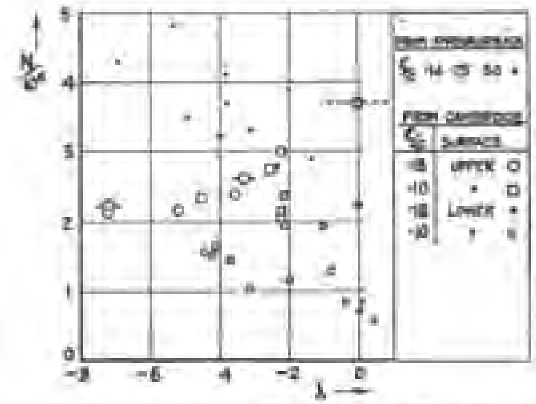


Fig. 13. Values of *N* and λ at transition for a number of different wings in flight. The parameters *N* and λ are defined in the text and further information relating to the circumstance in which each point was obtained is given in Table 1. The wide scatter of the points from different wings shows that these two parameters alone are insufficient to define the conditions which lead to transition in these experiments. The broken horizontal line through one of the small squares shows that in this instance the value of λ was not known with certainty within the limits shown by the line.

One feature which stands out clearly in Fig. 13 is that the two series of points representing observations on the upper surfaces of the wings examined at Cambridge fall very closely upon definite curves, though the curve is not the same for both wings. This consistency of the points for each wing taken alone is, of course, a direct consequence of the close functional relations between the lift coefficient and the position of the transition point which are revealed in Figs. 6 and 7; for, at any given point on a profile, both N and λ (as computed) are functions of lift coefficient. Bearing in mind the fact that the experiments in which these points were observed were made on many different days, sometimes with intervals of months intervening, it seems difficult to escape the conclusion already mentioned that the onset of turbulence was occurring under the influence of some dominating parameter whose origin is to be sought in the system of flow set up by the wings themselves. The wide scattering in Fig. 13 of the points for different wings shows, however, equally clearly that in these particular experiments the conditions leading to transition cannot be expressed simply in terms of the three parameters, N, λ, and C.

What the parameter which was controlling transition may be is still uncertain, but there are at the time of writing some indications from various sources as to its probable nature. It will be recalled that the apparatus used for observing the conditions of flow near the point of transition was of a kind which records time-means only of values which may in fact have been fluctuating. If therefore, keeping in mind the thinness of the laminar layer, we still retain the hypothesis that the onset of turbulence is determined by local conditions, we are almost forced to the conclusion

that the unknown parameter is to be sought in some form of fluctuation superimposed upon the mean flow.

H. L. Dryden using a hot wire anemometer within the laminar layer of a flat plate in a wind tunnel, has found just such fluctuations of surprisingly slow period and has published a figure which suggests strongly that in his experiments these fluctuations were in fact the primary cause of the ultimate break-down of the laminar flow. L. Schiller experimenting in similar circumstances has found similar fluctuations and had also shown that the point of transition can be controlled within a very wide range by slight alterations of the incidence of the flat plate, which caused relatively large displacements of the stagnation point on the rounded leading edge.

Working in collaboration with G. I. Taylor, we have recently constructed at Cambridge a small wind tunnel in which the working stream is remarkably free from turbulence, the root-mean-square of the longitudinal fluctuations of velocity being of the order of 0.1 percent of the mean speed, whilst that of the lateral fluctuations is about 0.2 percent of the mean speed. In this tunnel, using a flat glass plate 3/8 in. thick with rounded leading edge, we have observed the same phenomena as were observed by Dryden and Schiller; that is to say, hot wire examination has shown that relatively long period fluctuations of velocity occur in the laminar layer without being apparent in the potential flow outside the layer, and that the point of transition can be powerfully and very definitely controlled by slight alterations of incidence.

Consideration of results such as these from our own and other laboratories suggests that the final transition of the boundary layer to the fully turbulent condition may be the direct consequence of transient separation of the flow from the solid surface, brought about by these relatively slow fluctuations, but no experiment of which we are aware has, as yet, revealed with any certainty the origin of the fluctuations themselves. Whether for example their origin is to be found in minute fluctuations of the external stream, which become greatly magnified within the layer, or whether it is to be sought in some property inherent in the boundary layer itself, is a question which has yet to be determined, though the similarity of the fluctuations observed in the smooth flow of our new tunnel to those observed by Dryden in a stream of greater turbulence, lends some support to the latter view.

In the unusually smooth stream of our new tunnel we find, as was to be expected, that transition can be postponed until the value of N is considerably greater than the values usually obtained in tunnels of greater turbulence. Our new tunnel is not long enough to realize values of N much above three million, but this value has been reached without transition to the turbulent regime and it seems probable that in longer equally smooth tunnels it will be exceeded, even when the value of λ at transition is zero or slightly negative.

By modifying the shape of the tunnel walls one can produce, within limits, any desired sequence of pressure gradients along the surface of the plate, and we find, by this means, that the value of N associated with any given value of λ at transition can be greatly altered by changing the gradients through which the air has passed on its way to the transition point. Thus, the introduction of a sharp falling pressure-gradient, followed by a rising gradient, causes transition to be postponed to larger N values than when the rise of pressure is continuous from near the leading edge, and in this way we have succeeded in realizing N values greater than two million, even when the negative value of λ was as great as seven. These observations support the conclusion previously drawn from the flight experiments, that the conditions which lead to transition cannot be expressed solely in terms of the mean values of quantities in the immediate neighborhood of the transition point itself.

The introduction of artificially generated turbulence into the smooth air stream is found to have the well known consequence of causing transition to occur earlier than in the smooth stream and to cause the mean transition point—as located by slow-reading pitot tubes—to become much less clearly defined; no doubt because the amplitude of the to-and-fro movement of the transition region is greatly increased. This explains the remarkable precision and definition of the majority of experiments for the location of the point of transition in flight, in comparison with the results of previous experiments in wind tunnels. We find also that phenomena, such as the delicate dependence of the point of transition on the incidence of a flat plate, become much less apparent and definite when the turbulence of the tunnel stream is increased. It is now, in fact, becoming apparent that the more interesting features of the phenomenon of transition have in the past been masked by wind tunnel turbulence, and that the extension of the experiments to free flight coupled with the greater smoothness of flow in modern wind tunnels is opening up new fields for investigation.

It is yet too early to hazard an opinion as to what will be the final outcome of experiments of the kind we have been discussing. Whether or not the stream of information which comes in almost daily from aeronautic laboratories in many countries will ultimately reveal the possibility of controlling transition so as still further to reduce drag, or whether it will merely enable the point of transition to be predicted without enabling it to be controlled is a question which awaits an answer. For my part I am not able even to guess what the answer will be, but knowing that the problem is being intensively studied in the United States, I have, with the consent of my colleagues in England, told you exactly what we are doing and where we are as yet uncertain, in the hope that the discussion to follow will bring to light complementary evidence which, combined with our own, will enable a clearer picture to be formed of a phenomenon in which theorists and engineers alike are intensely interested.

DISCUSSION

Professor Jones' paper was presented at the Pupin Physics Laboratories of Columbia University on the afternoon of December 17, as a lecture which gave the principal results of the research work up to the time Professor Jones left Cambridge University two weeks before.

There were about 300 members and guests of the Institute present including Orville Wright.

Dr. Nicholas Murray Butler, President of Columbia University, welcomed the guest speaker, Mr. Wright, and members of the Institute. He spoke of his early interest in aviation and of the interest of Columbia University in the growth of the Institute, as all of its Annual Technical Meetings had been held in the Pupin Physics Laboratories. He expressed the hope that the Institute would continue to find the facilities of Columbia University helpful in its work. He praised the work of the Wright brothers and spoke of the great aeronautical development which their work inaugurated.

INTRODUCTION

Dr. Clark B. Millikan
California Institute of Technology

As you all know we are today privileged to be present at what will, I am sure, be remembered as an historic occasion: the first Wright brothers' Lecture. I shall not spend any time discussing the significance of the occasion since this will be done tonight. However, before introducing the speaker, I do wish to express our great appreciation to Columbia University, to Dr. Butler, and to Professor Pegram for their kindness in making this meeting place available and in welcoming us so warmly. Columbia saw the birth of the Institute, and all of our technical sessions have been held at it. It is, therefore, most happy for us that now the inaugural Wright brothers' Lecture is held under its auspices. Thank you again Dr. Butler and Professor Pegram.

I have been fortunate in knowing today's speaker for over seven years. Unfortunately the chances for personal contact have been few, but I have been able to follow his work fairly closely.

All of his contributions seem to me to have been marked by certain definite characteristics: (1) an interest in scientific problems which are closely related to the actual flight of aircraft; (2) a penetrating analysis of the problems attacked so that the simple fundamentals appear out of a maze of apparent complications; and (3) extreme ingenuity in developing simple experimental methods for studying the problems.

All of these characteristics appear strikingly in the work which the speaker has chosen for his subject today.

B. C. Boulton
The Glenn L. Martin Company

In this significant paper Professor Jones does not tell us how to control the boundary layer and hence vastly increase airplane performance, but like a true scientist he records basic phenomena and does much to lay a sound foundation on which others can build. Most of us shy off when the subject of boundary layer is mentioned and feel that it is too abstruse for the engineer; that it must be left to the scientist and mathematician. The present paper, however, does much to remove this barrier and gives us all a concrete realization and at least partial understanding of the subject. I will leave to my able colleagues in this discussion the more difficult task of commenting on the theory involved and will limit myself to emphasizing some aspects of special significance to the engineer.

In Fig. 1 it is of importance to note that the slope of the curves of decreasing drag with aft movement of the transition point is greatest for thick airfoils. With larger wings of higher aspect ratio the thickness ratios used are higher. In such cases the gain to be derived by using all means of maintaining laminar flow is great; also the corresponding loss. It may be remarked that except for thickness ratios of .30 and .10, the slopes of the curves are determined mathematically rather than experimentally. Another point that is rather amazing is that the forward transition point in the case of t/c = .3 was produced by the use of a thin sheet of paper. In another case a sheet of foil .002 in. thick affected the transition point. The sacrifice involved in a lap joint becomes painfully evident. This brings home to us the truth of the N.A.C.A. tests on the effect of roughness shown at the last annual conference. The fact which Professor Jones brings out that slight waves in an otherwise smooth surface also materially affect the transition point especially interests me. In the interests of weight saving we may use thin gauge material on a leading edge and feel, if the rivets are flushed, that all is right. If the material, because of its thinness, is wavy, all is not right, and on a long range aircraft the additional drag, so caused, may involve far more weight in fuel expenditure than that due to differences in gage. This is of significance structurally, and indicates we must count on the wing leading edge for structural strength since it must be reasonably heavy to secure low drag.

I believe it has been the general impression that as a wing becomes larger, little roughnesses due to skin laps and rivets become less important. Professor Jones points out the tendency with increasing Reynolds number for the transition point to move forward, though also holding out the possibility that with care the drag of wings on very large airplanes can be greatly reduced by keeping part of the flow laminar instead of entirely turbulent as would usually be assumed.

Of particular interest is the statement, verified by the constancy of the flight test results, that small scale turbulence does not exist in the atmosphere, and so for exploration of the boundary layer, flight experiments are more reliable than wind-tunnel tests. Turbulence in the latter case tends to mask the phenomena being investigated. As Professor Jones suggests it may be possible for an airplane manufacturer, using a single pitot tube just outside the laminar boundary layer, to make valuable investigations on the effect of different wing finishes and types of rivets as part of normal flight testing.

I would like to ask Professor Jones whether work has been done or is planned shortly, on exploring the boundary layer on hulls or fuselages. Since such elements constitute such a large proportion of the airplane drag, such work might prove to be of great value.

In recent flight tests at our plant we obtained a great reduction in drag through the use of cowl flaps on a radial engine to throttle to a minimum the cooling air at high speed. Part of the gain was due to increased pumping efficiency but we would like to enquire whether Professor Jones believes that the high speed air from the narrow gill slot could have acted as a sort of boundary layer control, thus further reducing the drag.

Dr. Hugh L. Dryden
National Bureau of Standards

The Institute is greatly indebted to Professor Jones for his very clear presentation of the practical importance to the airplane designer of an understanding of the nature of the flow in boundary layers and in particular of the factors controlling the transition from laminar to eddying flow. He and his colleagues honor us by making available in the Wright brothers' Lecture information of fundamental importance not hitherto published. Research workers in this country do have some additional information to add to that of Professor Jones, but must join him in admitting that the problem has not been solved and it is not known whether it will ultimately be possible to control transition to reduce drag. The more data obtained, the more complicated appear the phenomena.

In the experiments described by Professor Jones, a number of factors operating together influence the position of the point of transition and make analysis exceedingly difficult. Thus the increase of Reynolds number with decreasing lift coefficient tends to move the point of transition upstream, whereas the effect of the change in pressure distribution accompanying the decreasing lift coefficient may tend to move the point of transition downstream. It may therefore be possible to find some airplanes for which the point of transition is nearly stationary as the angle of attack is changed. The available theories are not adequate to describe the influence of the several factors and more experimental work is needed with conditions controlled to give but one variable factor at a time. Reynolds numbers and turbulence conditions corresponding to those in flight can however be obtained only in flight or in a full-scale wind tunnel of low turbulence.

The National Advisory Committee for Aeronautics has sponsored and financed a coordinated program of research on this problem at various university laboratories, at the National Bureau of Standards and in its own laboratories. The California Institute of Technology is studying the influence of curvature and roughness on transition and a progress report has been made available in Technical Note 613 of the National Advisory Committee for Aeronautics. The Massachusetts Institute of Technology is studying the influence of pressure gradient and the National Bureau of Standards the influence of wind-tunnel turbulence. The Langley Field Memorial laboratory is investigating the phenomena at large Reynolds numbers in the full-scale wind tunnel and in flight. By permission of Dr. G. W. Lewis, I am able to discuss the problem in the light of our work at the National Bureau of Standards, making use of some data not yet published by the Committee.

The use of a small number of total head tubes for determining transition is in general satisfactory and makes possible a fairly rapid accumulation of data. There is, however, a certain danger of missing an important phenomenon, namely, that of laminar or quasi-laminar separation, followed by a return of the flow to the surface, a phenomenon to which Professor Jones himself directed attention in his paper on Stalling in the Journal of the Royal Aeronautical Society, Vol. 38, p. 753, 1934. We have observed this phenomenon in recent experiments on the boundary layer near an elliptic cylinder.

The results of a detailed survey of the laminar boundary layer near this cylinder, in which laminar separation occurred, have been published. Comparison with theoretical calculations by Pohlhausen's method did not lend confidence in that method for computing separation, separation occurring for $\lambda = -5.4$ instead of Pohlhausen's value of -12. The theoretical speed distributions agreed fairly well with the observed distribution for values of the parameter λ between $+7$ and -5. Professor Jones' data in Fig. 11 are in agreement with this result. The limitations of Pohlhausen's method should, however, be clearly understood. The method developed by von Karman and Millikan is much better although requiring more tedious computations. (See Millikan, Clark B, A Theoretical Calculation of the Laminar Boundary Layer Around on Elliptical Cylinder, and Its Comparison With Experiment, Journal of the Aeronautical Sciences, Vol. 3, No. 3, January, 1936.)

Our recent measurements at the National Bureau of Standards were on the same elliptic cylinder at a higher speed (70 ft. per sec.) where transition occurred before separation. A contour map of the speed distribution as obtained by hot-wire measurements is shown in Fig. 1. Between 10.2 and 10.8 inches from the nose a region of reversed flow can be demonstrated by the use of smoke, lampblack, and oil on the surface, or any similar method. Since the hot wire is essentially a non-directional instrument, the reversal is not shown in the hot-wire measurements. From 10.8 inches to the second separation at 12 inches, the flow near the surface is in the direction of the main stream. The diagram is based on traverses at stations along the cylinder spaced 1 inch apart. By a method described later it can be demonstrated that a minimum in the local skin-friction coefficient occurs at 6.1 inches from the nose, although the phenomenon is not evident in the figure. We may therefore picture transition beginning at 6.1 inches, sufficient turbulence being produced to carry the layer some distance against the rising pressure but insufficient to prevent separation altogether in the region of rapidly rising pressure. The separation at 10.2

produces additional turbulence; the layer returns to the surface at 10.8 inches and separates again at 12 inches. Such a phenomenon might be missed by short-cut methods, but the apparatus of Professor Jones should detect it if the surface tube is included. If only the outer tube is used, the phenomena of separation and transition cannot readily be distinguished.

It may be noted that if transition is produced as a result of laminar separation, the point of separation is independent of Reynolds number only if the pressure distribution is independent of Reynolds number. We have observed that a laminar layer may separate, exist as a free laminar layer for some distance, and then become eddying. In such cases as the transition point in the free layer moves forward, the pressure distribution around the body is modified and the location of the point of laminar separation varies with Reynolds number.

We have used a hot-wire version of the surface tube, which has some advantages, and which we believe could be used in flight. The wire 0.00063 in. in diameter was mounted on a thin steel band 6 in. wide and 0.002 in. in thickness encircling the cylinder and capable of being moved around the contour of the cylinder so that the wire traversed the boundary layer at a small fixed distance from the surface (about 0.008 in.). While the actual measurement of speed is not easy with this arrangement, a simple electrical circuit permits the rapid location of the point where the speed is a minimum which corresponds to minimum skin friction on the surface or to the minimum total head shown by Professor Jones' surface tube. The device has been quite satisfactory in the laboratory and can probably be used in flight.

With this apparatus we have made rapid surveys of the effects of wind-tunnel turbulence which will be described in a forthcoming paper by G. B. Schubauer. A sufficient increase in the turbulence eliminates the first separation and region of reversed flow and the results on the location of transition check fairly well G. I. Taylor's formula for the relative influence of intensity and scale of turbulence.

Another interesting result is that when the turbulence is lowered beyond a certain point, further reduction does not move the point of transition farther from the nose, suggesting that the transition is then controlled by the pressure distribution system rather than by the turbulence of the external flow. A similar effect of Reynolds number is found at a suitable value of the turbulence, the transition point not moving aft of 6.1 inches as the Reynolds number is reduced.

Oscillograph records of fluctuations fail to show the sudden and intermittent transition observed on the flat plate. The intensity and frequency increase continuously with distance from the nose. There is a rather high maximum intensity at the transition much like that observed at the California Institute of Technology on the curved plate. Thus when pressure gradients and curvature are present the transition appears to be of a different character. There seems to be little doubt that the intensity and rate of fluctuations in the boundary layer are dependent on the sign and magnitude of pressure gradient and on the curvature as well as on the turbulence of the air stream.

As further results of the research sponsored by the National Advisory Committee for Aeronautics and of that conducted by Professor Jones and his colleagues become available, it is hoped that the picture may become clearer.

Dr. W. F. Durand
Stanford University

Referring first to the broad domain in which the work reported by Professor Jones finds its place, and to the wholly admirable character of the paper, both as to its content and as to the character of the treatment, I should like to emphasize what seems to me the profound importance of the activity which has characterized our study of the laws of fluid mechanics, in the broad sense, during the past ten or fifteen years.

I am not referring alone to the importance of this work in connection with its aeronautic applications, but rather to the broad significance which these laws play throughout the entire domain of nature. Matter, broadly speaking, is either fluid (liquid or gaseous) or solid, and throughout the entire domain of natural phenomena, the activities with which we are concerned and which touch our lives most closely involve, in an impressive degree the movement of a fluid, constrained or directed by a solid boundary; or more broadly, the relative movements of solids and fluids.

Thus the circulation of the blood in our veins and arteries; the movement of sap in trees; the flow of rivers and streams; the movement of the winds; the movement of water-borne and of air-borne craft—from such major class examples down to the most trivial actions as, for example, the agitation of a spoon in one's cup of coffee, in order to distribute more evenly the sugar in solution. These and thousands of other examples could be named, of activities or of natural functions, which touch our lives more or less closely and which all involve the phenomena attendant on the relative movement of solids and fluids; and thus mark the significance of the laws of what we call fluid mechanics.

And because of this, I hail with deep satisfaction the concentration of effort which these recent years have witnessed in the study of these phenomena, and in the better understanding of these laws which it has brought. And may not those who are especially concerned with the study of these laws in connection with the problems of air transport, take some measure of satisfaction in the thought that this study, on their part, will serve not only these immediate purposes, but also as a real contribution to a far wider domain of human activity; and that the refinement of laws which seems to be gradually developing out of such work as that described in Professor Jones' paper will find applications in domains of human interest far removed from those which have served as their immediate occasion. And may it not be said, that those who are thus serving the immediate interests of aeronautics, constitute, in effect, a service unit to all phases of human activity where a better understanding of the laws of fluid mechanics may enter as a significant factor.

Now with regard to Professor Jones' paper itself, I have been especially interested in his analysis of the conditions affecting the location of the transition point from laminar to turbulent flow, as expressible in terms of three non-dimensional parameters, and in the conclusion that three such parameters cannot include all the influential factors. In this connection the observations of Dr. Dryden regarding fluctuations in a laminar layer, and the suggestion of Professor Jones regarding the possible movement of the transition forward from the point of its initial formation, seem to me of special significance.

It seems quite clear that, as Professor Jones says, further observational work lies ahead—that until some adequate relation can be established between all further outlying essential phenomena and the basic conditions of the observations, we can hardly hope to be able to specify such additional parameter or parameters as may be required to give a complete account of the situation.

In the way of basic conditions affecting the observations, I have wondered regarding one characteristic of the air which, so far as I am aware, has not been usually taken into account as influential in connection with the study of these phenomena; and that is, the degree of ionization of the air. We have the physical characteristics of temperature, pressure, density, humidity, and viscosity; but may not, conceivably, the degree of ionization of the air be influential in connection with these obscure boundary phenomena? We know that it is influential in connection with other physical phenomena, as for example, the formation of rain droplets. Where we are approaching the ultimate in our study of this phenomena of the boundary layer, may it not be necessary to take some account of the electrical condition of the air as well as of its physical characteristics?

In closing, let me express my highest admiration for this paper and for the skill and resourcefulness with which the experiments were carried out. The researches here described belong to that supremely important type in which the locale is carried from the laboratory to the air, and to the condition of actual flight. And I am satisfied that, with this open door which has been provided by Professor Jones and his colleagues, these particular problems of the boundary layer will be carried through to some reasonably satisfactory and final conclusions.

Dr. J. C. Hunsaker
Massachusetts Institute of Technology

We are indebted to Professor Jones for a masterly survey of the state of knowledge regarding the breakdown of laminar flow, and in the true spirit of science he has given us the details of his own brilliant research results with an intimation of the direction in which they are leading him. Others who are working on the same problem are grateful for this disclosure and should respond in kind.

Professor Jones gives us evidence that the conditions for transition from laminar to turbulent flow cannot be expressed solely in terms of those parameters used in his experiments. However, I can add evidence that certain parameters are without doubt important.

At M.I.T., as a result of wind-tunnel research on flow separation and transition, we have good evidence that the transition point moves forward on a wing with increasing wind speed. Here the wing is held at a constant attitude and the only variable is the speed. Hence Reynolds number must be a controlling parameter.

We also confirm the fact that the thickness coefficient N of Professor Jones' notation must be important. Dr. Peters using a similar coefficient based on the momentum thickness found that transition took place always at about the same value of this coefficient for a flat plate and a wing and, furthermore, both on the upper and lower surfaces of the latter. This thickness coefficient appears to have a critical value.

In our work on hydraulic cavitation we have observed a periodic phenomenon associated with the flow through the throat of a Venturi nozzle, which suggests an analogy to periodic separation of air flow.

If we assume, as suggested by Professor Jones, that the laminar layer tends to separate and that transient separation actually takes place at breakdown into turbulence, we are led to believe that the instantaneous pressure gradient inside the boundary layer will differ from the mean value in the flow outside and should perhaps cause separation. Such separation will certainly break the flow down into turbulence. In general, turbulence is observed to be caused by separation or discontinuity.

A transient separation will cause a corresponding fluctuation in the local pressure gradient in the boundary layer, which in turn could cause a periodic fluctuation in the transition point as has been observed. The phenomenon of transient separation could then repeat itself in the manner observed for hydraulic cavitation.

When conditions are right the flow of water next to the walls of the throat separates to form vapor and turbulence. The vapor collapses, causes a sudden change in pressure in the throat with a return to continuous flow and a repetition of the cycle. The frequency of the repetition is a function of speed for given setting of the apparatus.

I do not wish to strain the analogy, but it seems to indicate that boundary flow phenomena are extremely sensitive to fluctuating pressures.

As to the scatter of the points on Professor Jones' final figure, I should like to point out that the pressure gradient in the laminar layer must be extremely sensitive to surface condition and must be extremely variable along any practical surface. Minute surface variations can very plausibly furnish a trigger action for transient separation, and may explain the scattering of results of tests made with different surfaces and attitudes. I, therefore, suggest that we do not worry about Fig. 13, nor try to draw general conclusions from it in view of the several sources of disturbance yet to be investigated.

It is most valuable to have the evidence of Professor Jones and his Cambridge colleagues spread before us in such clear form. American workers in this field will be greatly stimulated in continuing the attack on this fundamental problem of applied aerodynamics.

Dr. George W. Lewis
National Advisory Committee for Aeronautics

Before discussing the lecture proper I would like to express the sincere pleasure of the National Advisory Committee for Aeronautics on the selection of Professor Jones, a member of the British Aeronautical Research Committee, to present the first Wright brothers' Lecture before the Institute of the Aeronautical Sciences. I am sure that these lectures, carried on from year to year like those of the Royal Aeronautical Society, will be of great value in the advancement of aeronautical science, both in England and in this country. The National Advisory Committee has benefited often by the friendly and cooperative attitude of British research workers, so that I especially welcome this opportunity to share in entertaining a distinguished member of that group. We in this country who are primarily concerned with the scientific aspects of aeronautics look with great respect at the research staff of Cambridge University including, as it does, men of such eminence as Professor Jones and Professor Taylor.

Dr. Dryden has described to you the recent results obtained in our researches on transition in this country, so I wish to comment more on the engineering significance of Professor Jones' very interesting studies. Ever since the early days of flying, both manufacturers and research workers have been very much concerned with the problem of reducing drag, and it seems likely that this problem will continue to be important for some time in the future. It is of interest to note, however, that we are now entering into quite a new phase of the problem, and Professor Jones in his paper this afternoon has lived up to his reputation of being in the forefront of progress. We all remember his suggestion some years ago that the criterion of the aerodynamic cleanness of an airplane should be the closeness to which its drag approached the drag of a flat plate of the same wetted area. Up to the present time our efforts have largely been directed toward making the ratio as low as 1. Now, however, we appear to be entering a phase of investigation to bring this ratio below 1, actually to reduce the drag below what has sometimes been considered an irreducible minimum. To the designer of ten and fifteen years ago the increments of drag coefficient of which Professor Jones has spoken would have seemed insignificant, but on modern aircraft they represent a very appreciable and important part of the total. This is a real indication of recent aerodynamic progress.

To me one of the most remarkable features of Professor Jones' work is the apparent simplicity and directness with which he has obtained data applying to the case of particular interest, flight. I do not doubt that in the course of the investigation there arose many difficult problems of experimental technique which it required the utmost ingenuity to solve. It is noteworthy that Professor Jones has obtained data on very complicated phenomena with what appears to be very simple, understandable, and easily operable equipment.

I want to congratulate Professor Jones for presenting for the first time some definite figures on the dimension and location of the boundary layer. This will enable the designer to form some kind of mental picture of the factors that he must consider in attempting to secure minimum skin friction.

I. I. Sikorsky
Sikorsky Aircraft Division, United Aircraft Corporation

The lecture of Professor Jones was extremely interesting from a scientific standpoint. It is very probable that the ideas expressed will be very important in the near future, even from the standpoint of practical aeronautical engineering.

Ten or fifteen years ago the analysis of parasite resistance of an aircraft usually would have included some dozen items, such as struts, wires, wheels and their supports, separate power plant nacelles, and various other items. The profile drag of the wing in this case represented a minor item and the progress of design could best be achieved by simply reducing the number of protruding parts and improving the mutual interference of what was left. This work of cleaning up an airplane is nearly completed at the present time and in a well designed, modern airplane, particularly in a large, long-distance ship of the immediate future, we will usually find only three items of parasite resistance remaining, namely, the wing, the body, and the tail surface. This being the case, the resistance of the wing, even excluding the induced drag, becomes a major item that may approach one half of the total, while the skin friction of the whole airplane may eventually reach 75 percent of the total resistance. This being the case, it is important to study the methods which would permit controlling and, if possible, reducing this major item of resistance.

An important secondary problem in connection with this question appears to be a theoretical or experimental study of the question as to whether the skin friction of an ideal surface can be further reduced by artificial methods. By this we mean whether the remaining resistance is a basic figure similar to the induced drag figure which, we believe, cannot be reduced for the aircraft of given weight or span.

In line with this, it might be interesting to extend the study to birds flying. Some competent investigators believe that birds sometimes develop extremely high efficiency and L/D ratios that are far in excess of those obtained in aircraft. This factor is attributed to particular characteristics of wing feathers which permit the air to slip through, creating the effect of a boundary layer control. While I do not believe that birds really possess outstanding efficiency, yet further study may be of great interest. It is indeed extremely difficult because the tests that were made on wings of dead birds may not be identical to the conditions of a living bird.

Finally, we know at least two methods that would permit controlling the boundary layer and moving backward the point or region of transition from the laminar to the turbulent flow. These methods are the use of pressure or suction slots or the use of a sort of endless belt or built in rotors which would permit the surface of the wing to move backward at a velocity equal to that of the air stream. The latter method would probably eliminate entirely all separation. It appears important that aeronautical science should find out whether the weight and power expended on the operation of these or other artificial boundary layer control devices would be such as to leave room for improvement in performance of the aircraft.

In conclusion, I believe that we all owe sincere thanks to Professor Jones for discussing this very important question in such a remarkably clear and interesting way.

Eastman N. Jacobs
National Advisory Committee for Aeronautics

Owing to the lack of time I must forego the expressions I would like to offer of my admiration of Professor Jones' work, and pass on immediately to the subject of transition now under discussion.

From my standpoint the outstanding result of Professor Jones' lecture is that he has definitely shown what we have suspected for a long time: that extensive laminar layers must be recognized as possibly existing on actual airplanes in flight within the Reynolds number range commonly encountered. As wind-tunnel operators, many of us have hoped that we could escape this conclusion. When large possible movements of the transition point are encountered, corresponding uncertainties about our drag predictions are introduced. The outlook for us is gloomy. We remember Baker's experiments also made in England in which he found in towing airship models in water that the drag was practically indeterminate until the transition point was fixed by means of a small chord attached around the forward surface of the model. In fact, Reynolds' classic experiments exhibited this same uncertainty. One rather definite result was obtained, i.e., the lower critical Reynolds number at which turbulence started in the tube tended to be suppressed. The uncertainty appeared when it was attempted to determine the highest Reynolds number at which laminar flow was possible; the more care exercised in controlling experimental conditions, the higher were the Reynolds number values obtained.

The turbulence present in the variable-density tunnel has accomplished much the same result as Baker's cord and Reynolds' initial turbulence but the results must now be regarded as pessimistic as compared with the lower drags that Professor Jones has obtained in flight under carefully controlled conditions. In fairness to the variable-density tunnel, however, it should be emphasized that the results may still be employed as conservative. If you ask us to predict how much lower drag you may hope for in flight, however, it appears that we wind-tunnel operators are on the spot.

In order to predict actual airfoil drag coefficients, it is clear that something must be known about the position of the transition point. On this subject Professor Jones has provided some important data and a significant analysis, but the work impresses me mainly as providing the necessary material for a first class mystery.

The subject is evidently a complicated one and Dr. Dryden in his discussion has pointed out further complications encountered during his investigation of the flow in the boundary layer about an elliptic cylinder. A simple partial solution of the mystery should nevertheless be sought.

It is possible that the desired simplification may be reached through further complication? Dryden's investigations have shown, for example, that Pohlhausen's λ, which is described by the speaker as a non-dimensional measure of the local pressure gradient, is not an adequate parameter to describe the condition of the laminar boundary layer at least in regions of adverse pressure gradient where separation may be imminent. Von Karman and Millikan, however, have worked out a more complicated method of analysis which von Doenhoff, of our staff, showed would give satisfactory results when applied to Dryden's measurements on the elliptic cylinder. Moreover, G. I. Taylor's suggestions about the nature of transition together with certain experiments under carefully controlled conditions of vanishingly small turbulence, including some made at our laboratory to study transition on a plate in the presence of an adverse pressure gradient, have indicated that, within limits, transition may be very closely associated with laminar separation.

Such consideration suggests a limiting extent of the laminar boundary layer and consequently a limiting position of the transition point. Perhaps the approach to the problem may thus be simplified by first considering the extent of this range in which the transition point may be expected to lie near the point of laminar separation. On the one hand this position is an important one because more extensive laminar layers would not even be desirable. We have seen in the smoke tunnel at low Reynolds numbers the unfortunate result of a much more extensive laminar layer. The laminar layer separates, leaving a wide turbulent wake and a high form drag if transition does not occur shortly after laminar separation. On the other hand, less extensive laminar layers are also undesirable, owing to the drag increase. Consequently, this limiting position also represents the optimum one. Furthermore, one mystery of the subject paper would tend to be cleared up if this position were actually reached in the experiments under discussion; that is, why the point of transition frequently appeared to be so definitely fixed on the wing.

Time did not permit much quantitative consideration of Professor Jones' data but von Doenhoff kindly made the necessary calculation for the top velocity distribution curve of Fig. 9. To me it is significant that the calculated laminar separation point came out within approximately 2 percent of the chord from Professor Jones' tick indicating his measured transition point. The significance is that with sufficient care, smooth surfaces, and in flight in turbulent-free air, the range defining at least a close approach to the optimum transition position appears to extend to surprisingly high Reynolds numbers, that is, the values of several million reached in Professor Jones' flight experiments. The question I have previously asked still remains: How much further can we go in maintaining these desirable low-drag laminar layers?

T. P. Wright
Curtiss-Wright Corporation

In this lecture I was struck, as I have been when reading the previous lectures Professor Jones has given, with the scientific approach which he makes to his research investigations, striving constantly for the building up of facts and exhibiting the necessary caution in their interpretation.

We all remember, I am certain, the fundamental importance of Professor Jones' 1929 lecture wherein he developed the conception of the streamline airplane, outlining the wastefulness of design as it then existed, inasmuch as sixty-six percent of the engine power was wasted in overcoming drag due to turbulence in the wake of forms of non-streamline sections used throughout. He pointed out the possibilities as well as the advantages of designing to truly streamline shape and I know that personally, I was tremendously impressed by the lessons he taught in that lecture as no doubt, were many others.

In 1936 Professor Jones showed that much of his original idea, had been attained and that subsequently we must exert our energies toward producing a smoother surface which would reduce skin friction itself, thereby setting up a new ideal for which to strive. In that lecture he alluded to the possibility of moving the transition point so as to increase the proportion of laminar flow in the boundary layer at the expense of the drag producing turbulent flow.

In the present lecture, I am impressed with the scientific methods used by Professor Jones in obtaining and interpreting facts which he hopes will shed useful light on the factors governing the location of the transition point. (The following shows the parallel reasoning of scientists working on similar problems. Last Spring, Eastman N. Jacobs of our own N.A.C.A. gave a paper on the subject of laminar and turbulent boundary layer in which at one point he said: "The situation with regard to the airfoil drag is particularly serious because we have no equipment capable of studying the subject experimentally in the higher full-scale range of Reynolds Number in which we are at present most interested. Recourse therefore must be had to theory." The mechanical means (using the Pitot Transverse Method) which Professor Jones has so well described to us today has apparently filled this need.

Professor Jones has shown us clearly many facts pertaining to the effect on location of the transition point of smooth wings, of parameters which are determined by the Reynolds number of the boundary layer, the pressure gradient, and the radius of curvature. Although he arrives at the conclusion that it is some parameter whose origin is to be sought in the system of flow of the wings themselves, I think he rather regretfully concludes that the particular parameters investigated did not, of themselves, permit satisfactory determination of the transition point location. His allusion to the possibility that fluctuations superimposed on the main flow in the boundary layer may represent a basis for a parameter which will be of great importance in the transition point question should be noted and is a fact also alluded to by Dryden and Jacobs. There may possibly be some analogy between instability of these fluctuations which, through causes yet to be determined, seem to transform a laminar flow to a highly turbulent flow, with the instability of wings or control surfaces which at certain speeds may be subject to small vibrations without increase in disturbance but which at slightly higher speeds lose control, so to speak, and go into a phase of extreme flutter.

I trust I will be pardoned for again quoting from Jacobs' paper of last Spring wherein he says: "the present knowledge of wind tunnels makes it appear feasible to construct suitable equipment giving an airstream of effectively zero turbulence and capable of reaching the very large Reynolds Number for which engineers will very soon require reliable data." Here also, it appears that Professor Jones has in great part succeeded. The tunnel on non-turbulent flow described in his lecture represents a tool which should be extremely interesting and useful in continuing the general studies on the phenomena of transition point and the results of which studies, coupled with additional full-scale tests may, we hope, form the basis for another lecture by Professor Jones a year hence.

Quantitatively, it appears rather early to predict the order of performance improvement that may evolve from this research. Four or five percent speed improvement appears likely in a relatively short time, with some attendant increase due to the super-smoothness required to effect rearward transition point movement. Jacobs, I recall, was very optimistic.

With the streamline airplane closely approximated in our present air transports and with the growing appreciation of the importance of smoothness of surface in reducing skin-friction drag, it appears that the next ideal for which we should continue striving is the airplane surrounded in larger part by laminar flow, a goal toward the attainment of which Professor Jones has contributed so much. We have closely approximated the airplane of streamline form—now, (as Professor Jones has pointed the way by advancing our knowledge of it, and although it seems improbable and may prove impossible) let us strive for the airplane of laminar flow.

The repetition of the First Wright brothers' Lecture by Professor B. Melville Jones at the Athenaeum of the California Institute of Technology in Pasadena on Tuesday evening, December 21, was the occasion of the most outstanding meeting of the Institute of the Aeronautical Sciences ever held on the Pacific Coast.

Professor and Mrs. Jones were honored on the evening of December 20 by a reception held at the home of Dr. Theodore von Karman, and on the evening of the lecture, they were guests of honor at a dinner in the Athenaeum attended by over one hundred members and guests of the Institute. In addition to Professor and Mrs. Jones at the speakers' table there were: Robert A. Millikan, Theodore von Karman, E. P. Lesley, H. Bateman, Clark B. Millikan, Elliott G. Reid, Hall L. Hibbard, Arthur E. Raymond, A. L. Klein, Carleton E. Stryker and Norton B. Moore.

After the dinner and before the lecture, a short business meeting of the Los Angeles Branch was held. This Branch has applied for its charter on Founders' Day, and the meeting was its first since the charter had been granted. The Standard Form of By-Laws was adopted, and the following officers were elected for 1938: Chairman, Hall L. Hibbard; Vice-Chairman, Clarence L. Johnson; Recorder, Richard M. Mock; and Treasurer, E. E. Sechler. Another meeting of the Branch will be held on Friday, February 11.

At the lecture, a wire from Major Gardner was read: "Greetings from the Institute on occasion of holding first Wright Brothers' lecture on West Coast and organization of our most active Branch. Our sincere appreciation to Professor Jones for his willingness to give his important paper before our Southern California members."

Clark B. Millikan introduced Professor Jones, whose splendid lecture was enjoyed by well over two hundred listeners, and which was followed by a lively discussion entered into by Theodore von Karman, F. H. Clauser, H. Bateman, A. E. Lombard, and others.

Francis H. Clauser
Douglas Aircraft Company

It was a pleasure to hear from the man who provided the stimulus some years ago which has lead to the practical elimination of unnecessary form drag in modern airplanes and it is reassuring that this same man is now engaged in research which may conceivably reduce the remaining skin friction to some fraction of its present value.

One interesting point of the talk was the extremely small roughness necessary to precipitate transition. I wonder if Professor Jones has any data on either this effect or on the effect of roughness on the skin friction of the turbulent layer which might be compared with the permissible roughness given by current theory.

The speaker's remarks suggested the possibility of shaping the lower wing profile such that $dp/dx = 0$ at cruising velocities and thus preserving the laminar layer to great lengths over the surface. I wonder if he had attempted anything along these lines.

It would be of interest to know Professor Jones' ideas on the possibilities of drag reduction by boundary layer removal at the points of transition, thus having only laminar layers on the wing profile.

Lastly, what are Professor Jones' fondest hopes regarding large drag reductions by preserving the laminar layer throughout the entire length of the boundary layer? Here Professor Jones may cast scientific caution to the winds and speculate as in his fondest dreams.

LETTER TO THE EDITOR

December 17, 1937

Dear Sir:

In the April, 1937 issue of the Journal, R. S. Hatcher published an article on "Rational Shear Analysis of Box Girders." In the November, 1937 issue, George N. Mangurian published, in the form of a "Letter to the Editor" a set of formulae which is both more general and much simpler. For the benefit of younger students, who might become confused, it seems very desirable to say a few words on the physical meaning behind Mangurian's formulae.

Inspection identifies Managurian's basic formula for e as established by the oldest approximate method in existence, the so-called Centroid of Inertia method. This method was established in the days of wooden two-spar wings and gives very good approximations for true two-spar structures. For box-beams, however, it holds only if there action resembles that of two-spar wings. The indiscriminate application of the method to box-beams may lead to disastrous consequences, as shown in the following example.

Fig. 1 represents a fairly common type of construction, the nose and the tail furnishing no structural strength. Now it has been sometimes practice to pierce the rear shear web with large round holes to give access to the interior of the wing. This reduces the effective thickness t_2 practically to zero. Substituting $t_2 = 0$ in Hatcher's formula gives the elastic center location as H, while Mangurian's formula gives M. Reference to any good text book on strength of materials will show that Hatcher's location is the correct one.

Assuming a load applied at 50 percent of the wing chord (old L.A.A. case), Mangurian's formula gives only bending, no torsion, while Hatcher's formula gives correctly a very heavy torque added to the bending. This error is the more serious because the box considered is weak in torsion.

The distribution of stresses in a box beam is a statically indeterminate problem. For more than half a century it has been a recognized principle of engineering mechanics that such problems can be solved only by taking into account the elastic properties of the structure. It is not always possible to do this very completely, but any rule such as Mangurian's which violates a very important condition should be ear-marked very clearly as a sort of rule-of-thumb, very useful in the hands of a man who knows when not to use it, but a dangerous tool to put in the hand of a neophyte.

Paul Kuhn
National Advisory Committee for Aeronautics.

Document 4-18

Eastman N. Jacobs, Senior Aeronautical Engineer, to Engineer-In-Charge, "Notes on the history of the development of the laminar-flow airfoils and on the range of shapes included," 27 December 1938, in RA file 290, LHA, Hampton, Va.

Unfortunately, engineers often do not make good historians, at least when they are documenting the intellectual processes involved in their own current work. This short memo from Eastman Jacobs on the history of the development of the laminar-flow airfoils is not nearly as illuminating as one might hope it to be. Although Jacobs in this memo did a fair job of tracing his basic line of thinking back to earlier NACA reports, research authorizations, and even to some of his own memos, he did not provide a very introspective account of the history of his ideas (and others) leading to his concept of laminar-flow airfoils.

It is not known why Jacobs wrote this memo, or for whom. Such "Notes on the history" of any NACA development were extremely rare. They were usually provoked by a request from the NACA's Washington office for information that could be used for publicity purposes. It seems unlikely that this was the case with Jacobs' memo, though, as it was not written in the style usually seen when publicity was the goal.

Document 4-18, Eastman N. Jacobs, Senior Aeronautical Engineer, to Engineer-In-Charge, "Notes on the history of the development of the laminar-flow airfoils and on the range of shapes included," 27 December 1938, copy in LHA, Hampton, VA.

Langley Field
December 27, 1938

MEMORANDUM For Engineer-in-Charge.

Subject: Notes on the history of the development of the laminar-flow airfoils and on the range of shape included.

1. We have been familiar with the possible large drag reductions through prolonging of laminar boundary layers, particularly since the international airship model tests (1922-1923) were made in various wind tunnels for wind-tunnel standardization purposes. (See N.A.C.A. Technical Note No. 264, 1927.) It is difficult to state, however, just when I first considered plans for controlling the boundary

layer directly through the body shape or through control of the usual pressures acting along the body surfaces. Certainly the possibilities were clearly in mind in connection with our airfoil work before 1930 as shown by my memorandum of November 13, 1929; Research Authorization No. 88 on airfoil scale effect, which discussed the importance of transition on airfoil drag, and mentioned the dependence of the transition point on the airfoil shape. It was then expected that the gains would become apparent as the result of our systematic tests of various airfoil shapes. It is now known that little was found owing to the turbulence present in the variable-density tunnel and to the tunnel-wall and end effects present in the 24-inch high-speed tunnel. The long delay of almost 10 years may be largely attributed to these disturbing effects which tended to make the gains appear small or impractical. The ensuing work which finally disclosed and permitted the removal of the difficulties was, however, carried on continuously in the meantime.

2. Another line of attack is shown by a memorandum by Freeman dated April 18, 1932, which pointed out the possibility of drag reductions through boundary-layer control to delay transition on airships. (R.A. No. 201) In a laboratory conference on boundary-layer control (July 20, 1936) I compared the two methods and urged the necessity for new turbulence-free testing equipment as the primary necessity and emphasized that the direct control through shape appeared to be the most likely method and the one that should be placed first on our program before investigating the usual forms of boundary-layer control. The situation as it then existed is brought out in my S. A. E. paper: Laminar and Turbulent Boundary Layer as Affecting Practical Aerodynamics, March 12, 1937, which was a plea for suitable turbulence-free testing equipment. I remember deliberately withholding a disclosure of the details concerning the possible gains which I had definitely in mind at the time of the preparation of this paper, although I had disclosed earlier the possibilities of the new form of boundary-layer control in relation to airfoils as one of the most likely avenues of approach in my talk at the Manufacturers Conference on fundamental airfoil research and transition studies in May 1936. I wished to avoid building up too much hope for future advances without experimental verification, which seemed to require new testing facilities.

3. When the construction of the required new equipment was well under way, Pinkerton was asked, in

N. A. C. A. [illegible] AIRFOIL

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
Percent	Percent	Percent	Percent
0	0	0	0
[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]
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[illegible]	[illegible]	[illegible]	[illegible]
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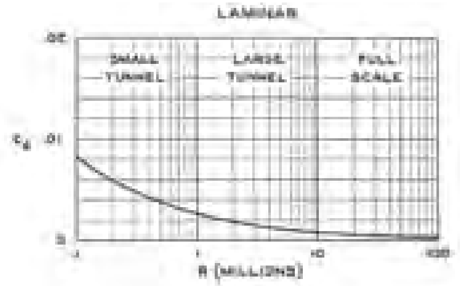
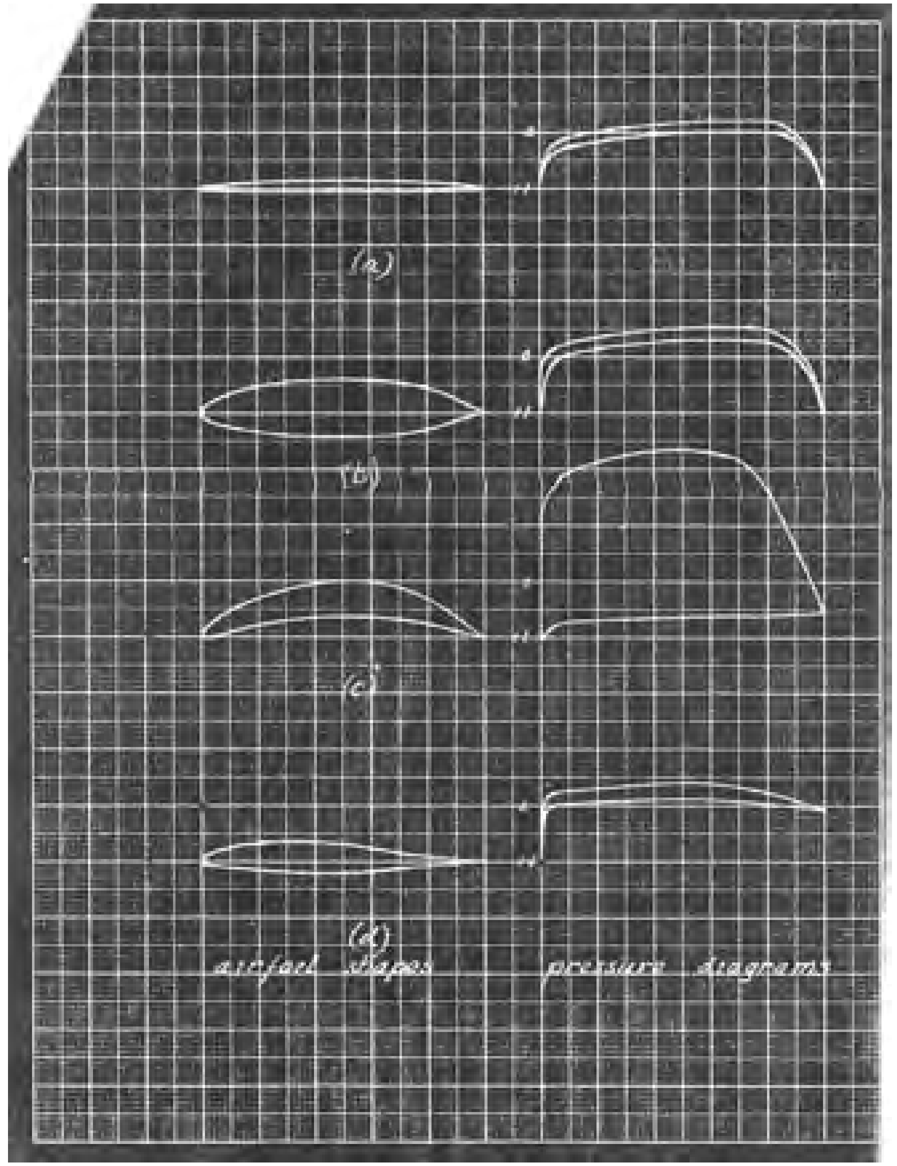
Fig. 1a (above) – Representation of Laminar Boundary Layer on the Upper Surface of an Airfoil
Fig. 1b (right) – Variation of Laminar Skin-Friction Drag Coefficient with Reynolds Number





December 1937, to seek airfoil shapes of the type required. A first approximation to a suitable shape was soon found, the pressure distribution verified theoretically, and a model constructed for tests. The large gains possible were first definitely established experimentally when this model was tested on June 23, 1938, in comparison with a conventional airfoil. The new airfoil showed a drag of the order of one-half that of the conventional airfoil. These tests were conducted as soon as possible after the new equipment had been put into operation and the test work and development have been pursued continuously and diligently ever since that time except for a forced short interruption for some icing investigations.

4. In order to indicate further the scope of the "laminar-flow" airfoils, the enclosed sketch has been prepared to suggest the range of shapes included. The form (a) is that which tests have shown to have the lowest drag of any investigation. This shape is designated N.A.C.A. 381-204 and a table of its ordinates is attached. Form (b) shows how the section thickness may be increased when required from other considerations. Form (c) shows how the desired laminar flow characteristics may be retained even at high lift coefficient. The purpose is accomplished by the use of a suitable, greatly exaggerated mean-line curvature. Form (d) is arrived at through somewhat different considerations. The forward part of the airfoil is derived to give the laminar-flow form but the rear portion is designed to give an easy and small pressure recovery as shown by the pressure-distribution diagram at the right of the figure. This character of airfoil is designed for the combined use of slot suction methods of boundary-layer control on the rear portion with the hope by this means of maintaining laminar flow over the entire body surface. Work is now going forward on this project but no test results are yet available.

Eastman N. Jacobs,
Senior Aeronautical Engineer.

Document 4-19

Eastman N. Jacobs, "Laminar and Turbulent Boundary Layers as Affecting Practical Aerodynamics," *Journal of the Society of Automotive Engineers* 40 (March 1937): 468-72.

Eastman Jacobs presented this paper at the National Aeronautic Meeting of the Society of Automotive Engineers in Washington, DC, on 12 March 1937. Supplementing his talk with slides and motion pictures, Jacobs described the general nature of boundary-layer phenomena and emphasized the lack of knowledge concerning the transition from laminar to turbulent flow.

This paper was especially significant to the aeronautics community of the late 1930s in showing "not what is known, but rather to emphasize that which is not known." Interestingly, one person in the audience who came to his feet afterwards to comment on Jacobs' paper was Dr. Max Munk, Jacobs' controversial predecessor as the head of airfoil research at NACA Langley and the father of the VDT. Munk's reaction to it, and to the whole idea of pursuing a laminar-flow airfoil, was very positive. "Expressing some concern as to whether Mr. Jacobs was optimistic or pessimistic in regard to the promise of future aerodynamic gains, Munk assured the session of his own optimism in this respect. He urged a continuation of this research as holding forth worthwhile promise." In reply, Jacobs reassured Munk of his own optimism and pointed out "the importance of reproducing flight conditions for the proper solution of the problem. He then went on to make a pitch for a new NACA low-turbulence tunnel. "This method necessitates equipment," he said, "whereby full-scale Reynolds numbers and low turbulence can be obtained" ("Two Aerodynamic Problems Debated," *Journal of the Society of Automotive Engineers 40* (April 1937): 26.

Document 4-19, Eastman N. Jacobs, "Laminar and Turbulent Boundary Layers as Affecting Practical Aerodynamics,"* Journal of the Society of Automotive Engineers *40, March 1937.

LAMINAR AND TURBULENT BOUNDARY LAYERS AS AFFECTING PRACTICAL AERODYNAMICS

BY EASTMAN N. JACOBS

The main part of this paper deals with one of the unsolved problems that impedes further progress in the aerodynamics of airfoil sections in relation to further research. In studying laminar and turbulent flow, special consideration is given to determining where the transition from one to the other takes place along the airfoil surface.

With no equipment capable of studying the subject experimentally in the higher full-scale range of Reynolds numbers, the problem has been attacked theoretically by two methods: According to the first method, the laminar boundary layer is supposed to become unstable.

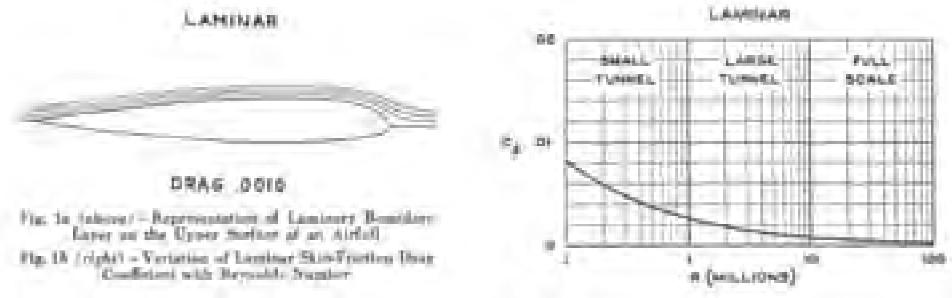
With the second method of attack the mechanism of transition is supposed to be something like separation. This comparison has the advantage that the separation phenomenon is comparatively well understood and can be dealt with quantitively by means of existing theory. Separation and its relation to the transition phenomenon are therefore considered, and the actual behavior of the flow during its change from laminar to turbulent is illustrated.

The final conclusion reached, however, is that we do not know but should find out whether theoretical gains indicated are possible. Such investigation will require suitable equipment capable of reaching these very large Reynolds numbers.

Recent progress in the most important field of practical aerodynamics, the flow about wing section, is due to an appreciation of the character of the flow as affected by variations of the section shape, the scale or Reynolds number of the flow, and the turbulence of the air stream. This progress has resulted in the development of improved wing sections, greater accuracy in the derivation of airfoil *section* characteristics from the usual airfoil tests, improved methods of predicting the section characteristics to be expected in flight at other Reynolds numbers and other conditions of turbulence than those under which the characteristics were measured and, finally, improved methods of predicting complete wing characteristics from the basic section characteristics.

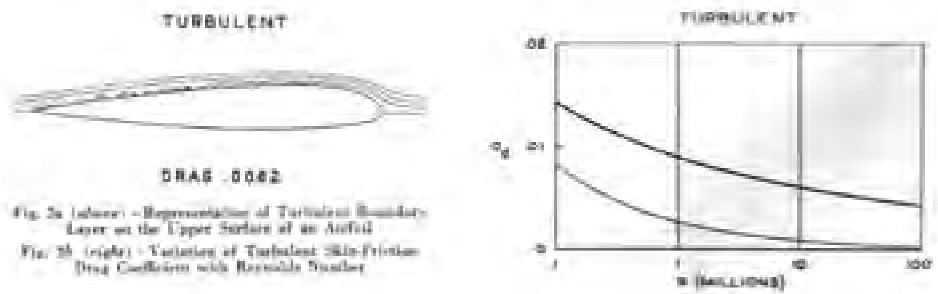
This paper, however, deals with one important unsolved problem that stands in the way of further progress. Our lack of knowledge about the boundary layer constitutes the main difficulty. Two types of boundary layer are encountered which,

owing to their entirely different character and behavior, markedly influence the final practical aerodynamic characteristics of airfoil sections. These two types of boundary layer are known as laminar and turbulent. Figs. 1(a) and 2(a) show the two types as they might be imagined to occur on the upper surface of an airfoil section in flight (inasmuch as the complete laminar boundary layer could not actually exist but would separate from the airfoil surface). The very low resistance to separation as compared with the turbulent boundary layer is, in fact, one important characteristic



of the laminar layer. Both types of boundary layer are shown as they would develop on a flat plate where separation would not be involved. The other important respect in which the two boundary layers show a marked difference is indicated by the numbers in Figs. 1(a) and 2(a) comparing the drag coefficients and by the skin-friction-drag variation with Reynolds number shown in Figs. 1(b) and 2(b).

A very large scale or Reynolds-number range is encountered in practical aerodynamics. Most of the figures presented herein such as Fig. 1(b) show a thousand-fold range of the Reynolds number, that is, from 100,000 to 100,000,000. The three main divisions shown on the plots, each representing a ten-fold range, have been indicated in Fig. 1(b) as small tunnel (100,000 to 1,000,000); large tunnel (1,000,000 to 10,000,000, which also covers the lower full-scale range including the landing conditions for existing transport airplanes); and full scale (10,000,000 to 100,000,000, which corresponds to the large future airplane or flying boat having a wing chord of 40 ft. and flying at 260 m.p.h.). The important result shown in Fig 2(b) then is the greatly reduced drag, corresponding to both types of boundary layer as these higher full-scale values of the Reynolds number are approached, and the



increasing difference between the two, the laminar drag becoming almost insignificant in the higher full-scale range. In this range it obviously makes a great deal of difference in the drag whether the boundary layer is laminar or turbulent.

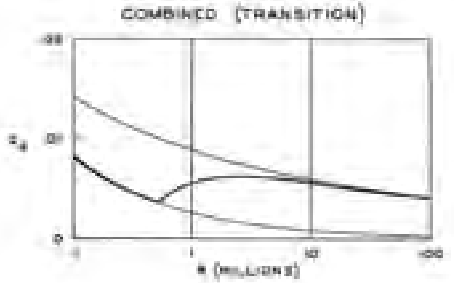


Fig. 3a (above) – An example of the Combined Laminar and Turbulent Boundary Layers on the Upper Surface of an Airfoil Showing the Transition Point

Fig. 3b (right)—An Actual (Combined Laminar and Turbulent) Variation of Skin-Friction Drag Coefficient on a Flat Plate

In general, both types are observed; the laminar appears over the forward part of the airfoil and changes to the turbulent somewhere along the airfoil surface at the so-called transition point (Fig. 3(a)). Owing to the difference in the character of the laminar and turbulent boundary layer, it is clearly essential to consider where the transition from one to the other takes place.

The classic studies by Osborne Reynolds of the flow in pipes showed that transition occurs at a certain value of the ratio we now know as the Reynolds number, dependent on the steadiness of the flow entering the pipe. When the transition occurs in the boundary-layer flow along a flat plate at a given Reynolds number (based on either the boundary-layer thickness or the distance of the transition point from the leading edge of the plate), the actual variation of the skin-friction drag with scale is presumably something like that shown in Fig. 3(b). Likewise, as Reynolds found in his pipe experiments, the transition occurs earlier or at a lower Reynolds number, if the air stream flowing over the plate is unsteady or turbulent. The effect of this early transition on the skin-friction drag coefficient, cd, of the plate is shown in Fig. 4.

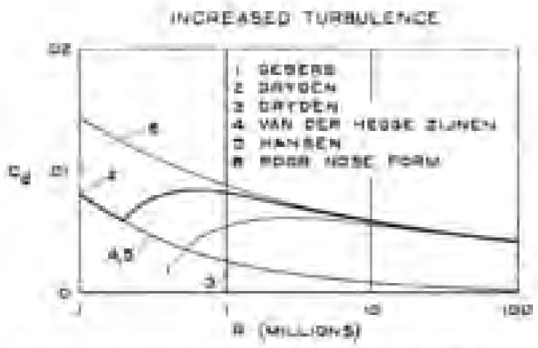


Fig. 4 – Skin Friction Variation with Reynolds Number on Flat Plate with Increased Turbulence – The Numbers Indicate Approximately Where Transition Was Encountered by Different Experimenters Under Various Conditions.

Experimental results in general confirm this view of the subject but, as shown by Dryden, who has obtained transition points ranging between those indicated by the numbers 2 and 3 in Fig. 4, pressure variations along the plate also have a very important effect. Roughness of the plate or a poor nose form may also introduce

turbulence and thus hasten transition, even to the extent indicated by curve 6 in Fig. 4, which corresponds to transition at the leading edge and shows no effect of a laminar boundary layer. Finally, when airfoils are considered, the study of the occurrence of transition is complicated further by the presence of large variations of pressure along the surface and, possibly, by the curvature of the streamlines. Nevertheless, thin airfoils, which are associated with small pressure variations and curvatures, may at least by compared with flat plates as in Fig. 5. The resemblance to the corresponding curve for the flat plate with increased turbulence (Fig. 4) is striking.

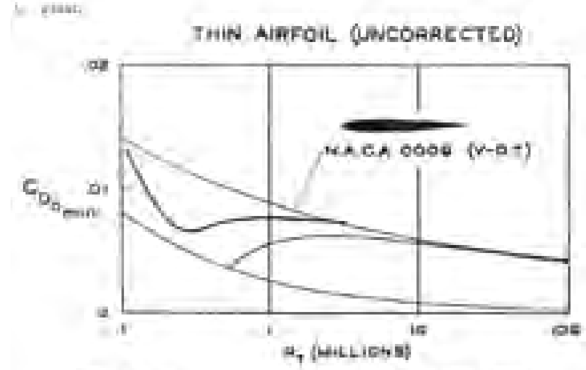


Fig. 5 – Comparison of Airfoil and Flat-Plate Drag

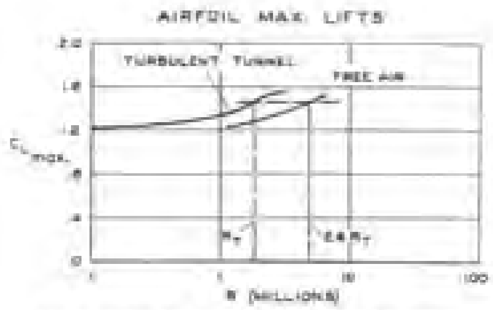


Fig. 6 – Airfoil Maximum Lift as Affected by Tunnel Turbulence

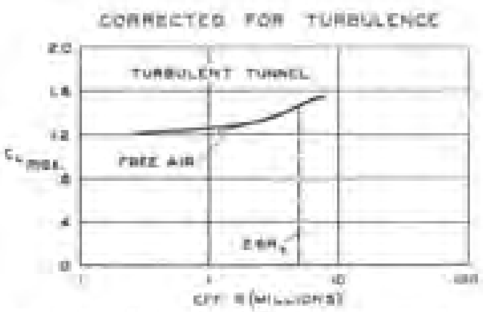


Fig. 7 – Same Measurements as Fig. 6 but Represented as a Function of Effective Reynolds Number

The effects of the increased turbulence in the variable-density tunnel may be taken into account on the basis of an effective Reynolds number approximately 2.6 times the test Reynolds number at which scale the corresponding transition conditions might be expected to occur in a turbulence-free stream. The turbulence factor, 2.6 for the variable-density tunnel, was determined as shown in Figs. 6 and 7 by a comparison of airfoil maximum-lift measurements in the tunnel and in free air. (The free-air results are actually inferred from tests in the N.A.C.A. full-scale tunnel). An interpretation of the drag on the basis of the effective Reynolds number with an allowance for the reduced skin friction at the higher Reynolds number results in a curve (Fig. 8) that is much like the well-known Gebers curve, representing the drag of a flat plate towed in water; moreover, thick airfoils (Fig. 9) have higher drag coefficients but appear to show similar variations with the Reynolds number. On this basis airfoil results from the variable-density tunnel may be extrapolated into the higher full-scale range as indicated by the dotted line in Fig. 9.

Up to this point the results, as corrected for turbulence, seem to be consistent and reasonable, but there remains the question: Do they apply accurately to flight conditions? The difficulty is that the turbulence factor and the effective Reynolds number are determined, in either sphere-drag or airfoil maximum-lift measurements, by the effects of turbulence on transition in a strong adverse pressure gradient in the neighborhood of the separation point, whereas Dryden's results have indicated that small changes of turbulence may produce large changes in the critical Reynolds number for flat plates. *In other words, the drag of a sphere or the maximum lift of an airfoil does not appear to be sensitive to small changes of turbulence as compared with the drag of a flat plate or an airfoil.* Consequently the usual turbulence correction when applied to the drag of an airfoil is likely to be too small. This expectation is supported by the comparison in Fig 10 of drag results for the N.A.C.A. 0012 airfoil from different tunnels. The rise in drag with increasing Reynolds number, probably associated with a forward movement of the transition point, is seen to occur too early in the more turbulent variable-density tunnel even after the turbulence effect has been

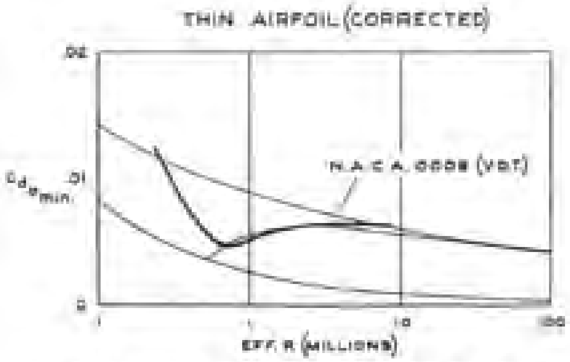


Fig. 8 - Comparison of Flat-Plate and Airfoil Drag Corrected to the Effective Reynolds Number

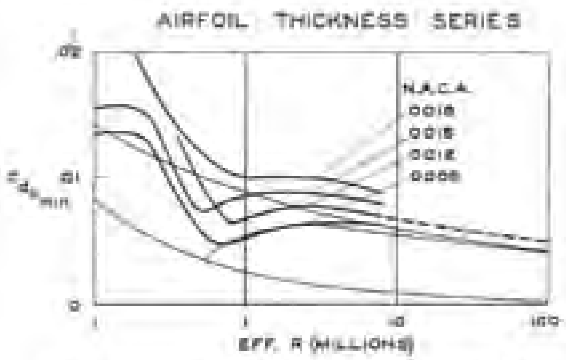


Fig. 9 - Variation of Drag with Reynolds Number for Symmetrical Airfoils of Varying Thickness

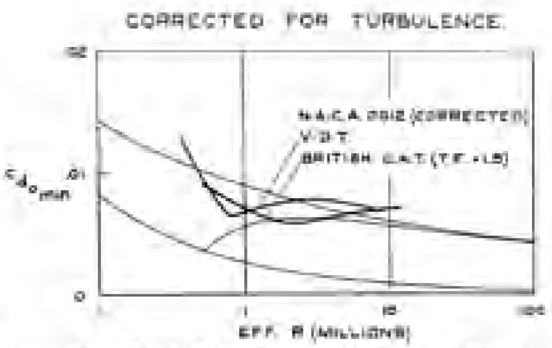


Fig. 10 - N.A.C.A. 0012 Airfoil Minimum Drag As Found from Different Tunnels

allowed for in the usual way by representing both sets of results at their effective Reynolds numbers. Furthermore, the rise is less abrupt in the less turbulent British compressed air tunnel so that it might be supposed that, in a turbulence-free stream or in free air, the rise would be still more gradual and would occur even later than indicated by the British compressed-air-tunnel results.

It thus appears that the interpretation of airfoil results on the basis of the effective Reynolds number, although it has proved in many instances to be a very useful engineering approximation, represents in reality an oversimplification. Unfortunately this conclusion leaves us without a reliable means of predicting airfoil-drag results, particularly in the higher full-scale flight range. In fairness to the results from the variable-density tunnel, however, it should be noted that drag values are often considered extremely uncertain, in fact sometimes almost indeterminate, in the range where a considerable movement of the transition point may occur. In practice slight roughness, vibration, or induced unsteadiness of the air flow near the airplane wing may bring about transition near the airfoil nose; thus this uncertainty concerning the drag may actually not appear in practice. On this basis the turbulence present in the variable-density tunnel accomplishes the same purpose. The results may thus be considered the most reliable available for *conservative* extrapolations into the higher full-scale range for aerodynamically smooth airfoils.

This consideration brings us, however, to the main subject of the paper. We know very little about why, how, or where transition occurs or, therefore, about the relative extent of the laminar and turbulent boundary layers. Finally, it follows that we have practically no certain knowledge about the two most important airfoil characteristics, $C_{l\max}$ and C_{do}, because they are both directly affected by the occurrence of transition.

The situation with regard to the airfoil drag is particularly serious, because we have no equipment capable of studying the subject experimentally in the higher full-scale range of Reynolds number in which we are at present most interested. Recourse, therefore, must be had to theory.

The theoretical problem has been attacked by means of two methods. According to the first, the laminar boundary layer is supposed to become unstable. Small disturbances that were damped out by the viscous forces at low values of the Reynolds number lose, at high Reynolds numbers, the damping necessary to prevent their growth into turbulence. Many prominent mathematical physicists have attacked this phase of the problem without obtaining very satisfactory results.

According to the second method of attack, the mechanism of the transition is supposed to be something like that of separation. This comparison has the advantage that the separation phenomenon is comparatively well understood and can be dealt quantitatively with by means of existing theory. The separation referred to may occur only locally, but any return flow tends to cause an accumulation of dead air over which the main flow must run. When a local dead-air region or bump is overrun by the main flow, reduced pressures are created which tend to draw in additional dead air, thus augmenting the disturbance. The turbulence may be considered as the final result of the building up of the bump until its top is carried or curled over by the main overrunning flow and thus moves downstream to form a distinct eddy.

The details of the transition have been observed and photographed at moderate Reynolds numbers on a flat plate in the N.A.C.A. smoke tunnel. When the transition is not brought about prematurely by slight surface roughness which also may cause transition by first promoting separation, the normal transition was observed to be closely associated with laminar separation. In general, when the turbulence and roughness were both practically zero, the transition was never observed to occur appreciably forward of the point at which laminar separation normally occurred. Furthermore B. M. Jones at Cambridge reports that he and his associates have found in flight laminar boundary layers on very smooth airplane wings sufficiently extensive to approach the laminar separation region. The fact that such extensive laminar boundary layers are not ordinarily observed at high Reynolds numbers in wind tunnels may be explained as the result of the airstream turbulence. The turbulence tends to produce localized pressure gradients along the airfoil surface that combine with the general pressure gradient to produce local separation and hence, by this theory, also to produce transition at points farther forward than the usual separation point. In fact, this second method recently has gained much prestige owing to the fact that G. I. Taylor employed equivalent concepts to make quantitative predications about the results of sphere-drag tests in turbulent wind tunnels.

These concepts may now be extended to account, in a general way, for the difference between the two drag curves in Fig. 10. In the variable-density tunnel, where the pressure gradients associated with the turbulence are relatively large in relation to those along the airfoil surface, they may combine to produce an adverse gradient of sufficient intensity to start local separation, even in the generally favorable gradient field near the airfoil nose. A relatively early and rapid forward movement of the transition point, as indicated by the rising drag curve, is then obtained. In the British compressed-air tunnel, however, where the pressure gradients associated with the turbulence are relatively less, the transition point is more reluctant to pass forward into the generally favorable pressure field; hence the later and less rapid increase of the drag coefficient.

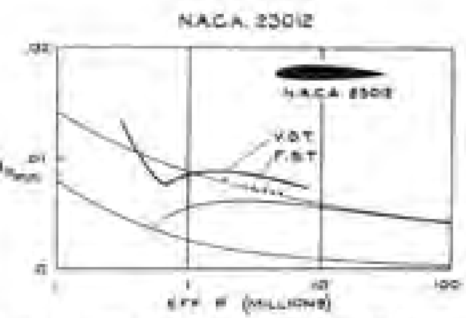


Fig. 11 – N.A.C.A. 23012 Airfoil Minimum Drag As Found from Different Tunnels

The few points shown in Fig. 11 for the N.A.C.A. 23012 airfoil and obtained from tests in the still less turbulent N.A.C.A. full-scale tunnel show, with increasing Reynolds number, little if any rise in drag that may be attributed to a forward movement of the transition point. On the other hand, the failure

of the drag points to fall much above the turbulent skin-friction curve indicated the presence of a rather extensive laminar layer. Otherwise the increased velocities over the airfoil, as compared with the flat plate, and the pressure drag would cause the airfoil drag to be considerably higher. This result may be associated with the theory that the forward movement of the transition point is caused by local pressure gradients associated with the tunnel turbulence, so that its movement is very slow when the turbulence is small; at least this theory seems tenable for smooth airfoils in the lower full-scale range. Now consider an extension of the same theory.

If the turbulence is zero, as it sometimes is in free air, the theory, carried to its logical conclusion, seems to indicate that the transition point will not move forward toward the leading edge of the airfoil as it does in the wind tunnel. If this supposition is true and other disturbances, such as turbulence originating near the nose or due to surface roughness, do not alter the situation, such a conclusion has considerable practical significance. A practical result is indicated in Fig. 12 by the dotted line showing how extrapolations should be made on this basis. The rise in drag associated with the forward movement of the transition point is assumed not to occur in free air at zero turbulence, the transition point remaining near the laminar separation point. It is then noted that the drag reached a surprisingly low figure at a Reynolds number of 100,000,000.

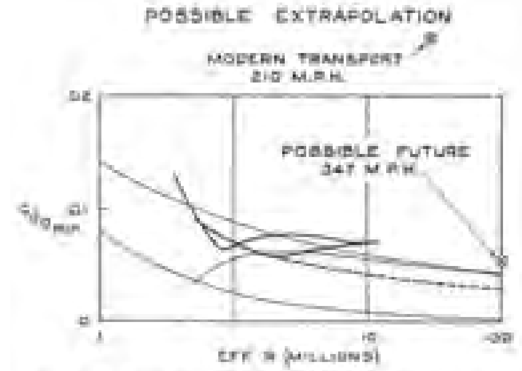


Fig. 12 – Possible Extrapolation of Tunnel Test Data into the Higher Full-Scale Range of the Reynolds Number

The importance of this possible result is further brought out by the comparison shown in Fig. 12 between two airplanes. The upper point ($C_D = 0.0246$) represents a conventional modern transport airplane having a speed of 210 m.p.h. The lower point represents a large hypothetical airplane having the same power and wing loading but designed to have very smooth surfaces, pusher propellers, and the other requirements necessary in order to take full advantage of the possible laminar flows over its forward surfaces. The combined wing, fuselage, interference, and tail-surface drag is based on actual tests of complete models in the variable-density tunnel but the extrapolation to 100,000,000 is based on the assumption that laminar boundary layers on the forward portions of the surfaces may be realized. The comparison is shown primarily to indicate that there is no necessity for pessimism concerning possible future aerodynamic improvement. Incidentally, even further drag reductions may be possible. For example, most of the fuselage and part of the tail-surface drag might have been eliminated by using the flying wing; furthermore, the possibilities of boundary-layer control have not even been considered. Nevertheless, the possible drag reductions considered allow a speed increase from 210 to 347 m.p.h.

The final conclusion, however, is that we do not know whether or not such gains are possible, but it is evident that the possible gains are large enough to justify immediate and careful investigation. Unfortunately, the necessary investigations require equipment that is not available. The present knowledge of wind tunnels makes it appear feasible to construct suitable equipment giving an air stream of effectively zero turbulence and capable of reaching the very large Reynolds numbers for which engineers will very soon require reliable data.

THE THROTTLE-STOP IN LIGHT FLEET UNITS

There is one item of which I have not spoken heretofore but which, to my mind, has been of such importance that I want to take the time to make special mention of it. I am referring to that little gadget known as the throttle-stop and I take a personal pride in the fact that our organization had a part in its development. The average fleet owner does not need the performance and top speed which is being built into the light automobiles of today. Economy of operation is far more desirable for our purposes, and you know that economy and performance do not travel together. We experimented with stopping our throttle at a point where it was only about one-third open, at which point the cars had a top speed of from 55 to 60 m.p.h. This we considered a sufficient rate for business purposes. The car was a little slower in acceleration and had to go into gear on climbing some hills, but it did have economy of operation. There was some growling from the drivers, as they are only human and, even as you and I, would like to have the best, be it in performance, speed, or appearance, but that growling disappeared when drivers were educated to the necessity for economy.

We formerly made our own throttle-stops and did our own work on the engines, but today we can all buy economy models with throttle-stops built into the car by the manufacturer.

The first paper, Flexible Exhaust-Valve Seats, by S. D. Heron, Ethyl Gasoline Corp., and A. L. Beall, Wright Aeronautical Corp., was read by Mr. Heron. He stated that, although some cylinder designs are attractive due to compactness and ease of securing large valve area, they are known to be subject to exhaust-valve-seat distortion. The investigation reported in the paper was carried out to determine whether the difficulties resulting from distortion can be overcome by flexibility in the valve head or valve seat, he announced.

In discussion following, A. T. Colwell, Thompson Products, Inc., stated that shallow and wide valve seat inserts gave poor results, whereas narrow and deep inserts gave good results. It also was stated that seats faced with Stellite gave good performance owing to their non-corrosiveness.

Flexible valve seats, according to Arthur Nutt, Wright Aeronautical Corp., may give four-valve cylinder heads a new lease on life. Two valves, he added, were used in this country mainly on account of seat distortion accompanying the four-valve type. The two-valve type of heads give from 400 to 500 hr. of satisfactory service provided no lead is used, he specified.

A. G. Elliot, Rolls-Royce, Ltd., criticized the flexible seats on the basis of: (1) increased heat-absorption area in the combustion-chamber; (2) restriction of the valve diameter; (3) increased heat flow up the valve stem which would cause carbon formation on the valve stem and springs.

In his conclusion, Mr. Heron remarked that there was nothing to be gained by using flexible valve-seat inserts in cases where distortion did not occur. Furthermore, when using flexible valve seats it is necessary to obtain all the internal valve cooling possible, he cautioned.

Other discussers were: C. D. Waldron, National Advisory Committee for Aeronautics; Harold Caminez, and A. L. Beall, the co-author.

In introducing his paper on The Determination of Ratings for Transport Aircraft Engines, R. F. Gagg, Wright Aeronautical Corp., stated that the objective in choosing an engine rating is to establish the limiting values in the operating procedure which permit a maximum of utility in power output and economy of fuel consistent with requirements for safety and durability.

In his paper Mr. Gagg pointed out the importance of design calculations, of single-cylinder laboratory tests, and of dynamometer calibrations of the engine performance.

Furthermore, he stated, after the engine ratings are determined on the basis of stress values, it becomes necessary to recheck the fuel consumption and detonation performance of the engine with the fuel tentatively selected for use.

Among those who discussed Mr. Gagg's paper were: F. C. Mock, Bendix Products Corp., Harold Caminez, and Charles Froesch, Eastern Airlines, North American Aviation, Inc., who submitted a written discussion which was read by A. L. Beall.

The need for controlled oil circulation governed by the viscosity rather than by the oil temperature was pointed out by Weldon Worth, U.S. Army Air Corps, in his paper "Lubrication and Cooling Problems of Aircraft Engines." Control of the oil temperature by means of the oil flow was more satisfactory than by the use of shutters on the radiator, he reported.

Mr. Worth described the oil-dilution system used for facilitating the starting of cold engines and which is now undergoing tests by the Air Corps. In this system he explained how provision is made for thinning the lubricating oil by the addition of gasoline prior to stopping the engine.

Among those who discussed this paper were: W. H. Robotham, Rolls-Royce, Ltd., R. M. Hazen, Allison Engineering Co., S. D. Heron, R. F. Gagg, Kenneth Campbell, Wright Aeronautical Corp., L. P. Saunders, Harrison Radiator Corp., A. G. Elliot, and Arthur Nutt. In reply to their questions, Mr. Worth stated that the hazard created by gas in the crankcase was not serious, as the mixture normally found in the crankcase was over-rich for combustion. There apparently was no undue wear caused by lead in the oil or caused by using a small quantity of oil very severely for a short time rather than a large quantity less severely for a longer period, he added. In normal operation, he concluded, about 2 qt. of gasoline are used for oil dilution, and the progress in diluting the oil can be observed on the oil pressure gauge.

TWO AERODYNAMIC PROBLEMS DEBATED

Pointing out that the purpose of his paper was to show not what is known, but rather to emphasize that which is not known, a resume of the researches leading to the realization of the importance of the boundary-layer phenomena, especially the transition from laminar to turbulent flow, was presented by E. N. Jacobs, National Advisory Committee for Aeronautics, in the first paper at the Practical Aerodynamic Problems Session, Laminar and Turbulent Boundary Layers as Affecting Practical Aerodynamics. Supplementing his talk with slides and motion pictures of the boundary layer over airfoils and flat plates, he indicated the general nature of the phenomenon and emphasized the lack of knowledge concerning the transition from laminar to turbulent flow. Peter Altman, University of Detroit, presided.

Expressing some concern as to whether Mr. Jacobs was optimistic or pessimistic in regard to promise of future aerodynamic gains, Dr. Max M. Munk assured the session of his own optimism in this respect. He urged a continuation of this research as holding forth worthwhile promise. T. P. Wright, Curtiss-Wright Corp., presented some rough figures on the possible gains to be expected and expressed an optimistic outlook. C. H. Chatfield, United Aircraft Corp., asked about the importance of roughness over the aft portion of airfoil.

In reply, Mr. Jacobs reassured Dr. Munk of his optimism and pointed out the importance of reproducing flight conditions for the proper solution of the problem. This method necessitates equipment whereby full-scale Reynolds numbers and low turbulence can be obtained, he indicated. Mr. Altman raised the question of double-peak lift curves and the influence of the type of lift-curve peak in design.

H. D. Fowler, Glenn L. Martin Cp., discussed the merits of the flap bearing his name and urged its use as a solution of the difficulties in present-day design in the session's second paper: The Practical Application of Fowler Flaps. He urged also that it not be discarded because of mechanical difficulties and emphasized the need of allowing for the flap in the basic design rather than the arbitrary application of a flap to an already established design. He discussed at some length the merits of this particular flap in the performance of its several functions.

F. E. Weick, Engineering and Research Corp., opened the discussion by raising a question as to the exclusive merit of the particular flap under discussion, indicating he did agree that in many functions it was superior to other types of flaps. T. P. Wright, reading from written discussion, emphasized the author's warning that a design should not be discarded because of mechanical difficulties.

Document 4-20

Eastman N. Jacobs, Senior Aeronautical Engineer, "Investigation of low-drag airfoil sections having extensive laminar boundary layers," undated (but typed 27 June 1938), in RA file 290, LHA, Hampton, Va.

In this memorandum to the engineer-in-charge, Eastman Jacobs presented the "significant findings" resulting from the first round of airfoil tests in what the NACA had rather mischievously called its "icing tunnel." In short, Jacobs reported that his preliminary experiments "have more than justified our hopes for low-drag airfoils."

The test program that followed led to the design of what would come to be known as the NACA "laminar flow airfoils." There were, in fact, several different families of these airfoils developed in the next eight to ten years.

Given the number and variety of NACA airfoil families created by the end of World War II, it is not easy for anyone but the true airfoil specialist to keep them straight. To summarize, the NACA program began with the "M series" developed by Max Munk in the mid-1920s. Next came the four-digit series developed by Jacobs and his colleagues. Then, in the mid-1930s, the NACA designed a five-digit series, but continued to work on modifications leading to better airfoils in the four-digit series. One of the best five-digit airfoils was "N.A.C.A. 23012," the most famous member of the celebrated "230" family, first announced in 1935. By 1939, "230" wings were the most widely used wing sections in the world, primarily because of their superiority in lifting.

With the emergence of interest in laminar flow came several new airfoil families, designated Series 1, 2, 3, 4, 5, and 6; all of them appeared by the end of World War II. Series 1 represented "the first attempt to develop sections having desired types of pressure distributions" and were "the first family of NACA low-drag high-critical-speed wing sections" (Ira H. Abbott and Albert E. Von Doenhoff, *Theory of Wing Sections* [New York: McGraw Hill, 1949], p. 118). Series 1 airfoils were designated not by one digit, as one might think, but by a five-digit number with a dash between the second and third numbers as in "N.A.C.A. 16-212." (The first integer represented the series designation; and the second integer represented the distance in tenths of the chord from the leading edge to the position of minimum pressure for the symmetrical section at zero lift. The first integer following the dash indicated the amount of camber expressed in terms of the design lift coefficient in tenths; and the last two numbers together indicated the thickness in percent of the chord.)

It was a wing of the Series 4 family that North American Aviation, Inc., selected in 1940 for its high-performance fighter plane, the P-51 Mustang. The Series 5 forms, which had a blunter nose than any previous NACA airfoils, proved impracticable for wings. But the Series 6, with its yet more favorable distribution of pressure over the chord, soon became the standard low-drag wing. By the end of 1944, Series 6 wings (which were designated by a six-digit number but also with a statement showing the type of mean line used, as in "N.A.C.A. 65,3-218, a= 0.5) were in use not only on the last version of the Mustang (the P51H) but also on the Bell P-63 Kingcobra, Douglas A-26 Invader, and jet-propelled Lockheed P-80 Shooting Star and Bell P-59 Airacomet. In comparison with conventional wing sections, the Series 6 airfoils looked quite different in that their maximum thickness was much farther back from the leading edge. Late in the war, NACA Series 7 wing sections also came to life and were characterized by a greater extent of possible laminar flow on the lower than on the upper surface.

Document 4-20, Eastman N. Jacobs, Senior Aeronautical Engineer, "Investigation of low-drag airfoil sections having extensive laminar boundary layers"

Langley Field, Va.
Undated.

MEMORANDUM For Engineer-in-Charge.

Subject: Investigation of low-drag airfoil sections having extensive laminar boundary layers.

1. Preliminary experiments in the ice tunnel have more than justified our hopes for low-drag airfoils through design to produce extensive laminar layers. We can now conclude definitely that the most likely form of boundary-layer control to reduce drag is through the use of the flow conditions and pressures ordinarily attainable over the section through changes of the section shape to provide the desired control to maintain laminar flow.

2. The significant findings are:

a. That a low-turbulence tunnel is required for this advanced type of airfoil testing. This conclusion is justified by the fact that the variable-density tunnel tests failed to show an unusually low drag for the same airfoil that showed a startlingly low drag in the ice tunnel.

b. That low drags, under conditions approximating those of flight, are readily attainable. Wake measurements in the ice tunnel for the first trial airfoil indicate as compared with the N.A.C.A. 0012, measured in the same

tunnel in exactly the same way, a reduction in drag of something like 30 percent.

c. That boundary-layer measurements on the upper surface of the same airfoil indicate laminar flow at a position 75 percent of the chord behind the leading edge on the 5-foot airfoil at a speed of 147 miles per hour.

3. These results are of such marked significance that we must at once give careful consideration to the course of future work. We now have an airfoil that may be expected to give unprecedentedly { low drags when applied to light airplanes or gliders at one certain lift coefficient (C_L = 0.2 approximately). Our investigation must now proceed immediately to include obvious extensions and improvements. These extensions will include derivation of modified airfoils having:

Longer laminar layers.
More or less favorable pressure gradients.
More or less thickness.
Higher lift coefficients for minimum drag.

In addition means, including flaps and boundary-layer control by suction, should be investigated for improving the relatively low maximum lift coefficients of these airfoils and one or more of the best sections should be investigated to higher speeds and Reynolds numbers in the 8-foot high-speed tunnel and to still higher Reynolds numbers in the pressure tunnel when it is available.

4. Authority to proceed with this work (building the models and tests in the ice tunnel) is requested and a job order request is attached to cover calculations required immediately to develop the series of the 12 or 15 sections required.

Eastman N. Jacobs
Senior Aeronautical Engineer.

(Typed 6/27/38)

Document 4-21

Eastman N. Jacobs, Senior Aeronautical Engineer, to Chief of the Aerodynamics Division, "Patent on airfoil developments," 9 December 1938, in RA file 290, LHA, Hampton, Va.

In this rather surprising memorandum, Jacobs suggested that the NACA initiate a patent application for the low-drag airfoils. Unfortunately, no other documents have been found in the NACA records to illuminate what happened to the idea. One of the most surprising aspects of the idea is that Jacobs began his memo by saying that Dr. George Lewis, the NACA's director of research, suggested the idea for a patent application.

The NACA was normally not in the business of patenting its research results. However, George Lewis's thinking might have been different on this issue as it related to laminar-flow airfoils, or such a patent application would apparently never be made. Perhaps this was for national security reasons. No doubt it was due in part to the fact that NACA research was paid for by the American taxpayer and needed to be available as freely and widely as possible in order to advance the cause of American aeronautics.

The whole issue of patents for Federal employees is very problematic, from both the legal and historical perspectives. No NACA or NASA history has dealt with the matter explicitly, and records related to the NACA policy are particularly hard to find. Some light on the issues raised by Jacobs' memo might be shed by reference to other Federal agency policy on patents by employees. In a long and detailed footnote to his *Measures for Progress: A History of the National Bureau of Standards* (Washington: U.S. Department of Commerce, 1966), Rexmond C. Cochrane clarified the policy for the NBS: "Traditionally, the Government retained rights to the use of inventions of Federal employees but otherwise left title to them with their inventors."

But not all government agencies followed this policy—for example, the NBS. For 20 years under the direction of Dr. Samuel Stratton, head of the NBS, "it was understood that any innovations or invention of a Bureau staff member was to be patented in the name of the Government for the use of the public." In 1921/22 this was challenged by NBS researchers Percival D. Lowell and Francis W. Dunmore, who claimed that one of their inventions was only remotely related to the Army Air Corps project on which they were working, and that they were deserving of a patent application of their own. It took ten years for the US District Court (in Delaware) to hand down a judgment that went for Lowell and Dunmore against the government. The ruling was appealed and eventually the matter made it to the U.S. Supreme Court. In 1933, the Supreme Court ruled, with only one judge dissenting, that "in the absence of a specific contractual agreement, all commercial rights to patents belonged to the inventor, whether or not the work was performed on Government time" (Cochrane, *Measures for Progress*, p. 348n).

But this does not mean that, even after 1933, all Federal agencies abided by the ruling in favor of the private rights of their inventors. As Cochrane explained, the policy put into effect by Stratton at the NBS "continued in force until modified in 1940, when [NBS] patents were procured by the Justice Department and assigned to the Secretary of Commerce for licensing under terms he prescribed" (Cochrane, *Measures for Progress*, p. 349n). Ten years later, the Federal policy of permitting employees to retain title to their own inventions (the policy that the NBS, for one, had not been followed) came to an end. Executive Order 10096, issued by President Harry S. Truman on 23 January 1950, declared that "all rights to any invention developed by a Government employee in the course of his assigned work belonged to the Government." (Cochrane, appendix C, p. 547.) This was later amended by Executive Order 10930. In October 1988 Congress issued a rule (53 FR 39734) establishing a "Uniform Patent Policy for Domestic Rights in Inventions Made by Government Employees [Docket No. 80627-8127]. This rule transferred the provisions of Executive Order 10096 (as amended by Executive Order 10930) from the Commission of Patents and Trademarks to the Under Secretary for Economic Affairs in the Department of Commerce. This final rule also established 37 CFR Part 501, which set forth this delegation of authority to the Under Secretary. In addition, it authorized each Government agency on its own to determine "whether the results of research, development, or other activity within the agency constitute an invention with the purview of Executive Order 10096, as amended by Executive Order 10930, and to determine initially the rights therein" in accordance with the provisions of the appropriate sections of Federal law.

From this short review, it should be clear that there was no uniform patent policy across the Federal government in the late 1930s. The only way, then, to know what the NACA's own policy amounted to at the time of Jacobs' memo of 1938, or at any other time, is to examine NACA policy and patent history, specifically.

One might suggest from some of the circumstantial evidence that the NACA probably handled patent matters similarly to NBS. Besides serving as the bureau director through the 1920s and 1930s, Stratton was also a prominent, and original, member of the NACA. He served as secretary of the Main Committee from 1917 to 1923 and chaired its Executive Committee during that same time. He remained on the Committee until October 1931. Given his early influence on the NACA, one might think that NACA patent policy would have been similar to that which Stratton directed at the NBS. However, that conclusion must remain hypothetical until historical research confirms it.

Unfortunately, Jacobs' memo does not help to clarify the picture. It called for a patent application, but it did not directly address the issue of who was making the application. The clear inference is that it should be the NACA itself, not Jacobs and his associates, applying for the patent. But why the NACA should apply for a patent for the new laminar-flow airfoils when it had not applied for patents for any of the earlier airfoil families, or for any other research results up to that point, is unclear.

Document 4-21, Eastman N. Jacobs, Senior Aeronautical Engineer, to Chief of the Aerodynamics Division, "Patent on airfoil developments," 9 December 1938.

Langley Field, Va.,
December 9, 1938.

MEMORANDUM For Chief Aerodynamics Division.

Subject: Patent on airfoil development.

1. As suggested by Dr. Lewis, it seems desirable to initiate a patent application covering our recent airfoil design advances. Suggest something as follows:

2. Previously, nearly all possible airfoil shapes have been tried. Various advantages have been claimed for various shapes, but the developments have in general led to airfoil section shapes of the so-called "streamline" form, that is, shapes having well faired contours, and <u>fine tail forms.</u> Such forms have been generally considered to represent the ultimate in low-drag wing-section shapes. Various forms of boundary-layer control have also been proposed for these "streamline" forms with the object of reducing the drag. We, however, know of no previous attempt to secure boundary-layer control through an intelligent development of new forms, not necessarily of the "streamline" type but intended primarily to control the boundary layer with the primary object of obtaining unusually low drag. By providing shapes to produce falling pressures downstream along the surface, the boundary layer is effectively controlled in such a way as to delay the transition. The invention consists of wing-section forms shaped primarily to control the boundary layer along a major part of the surface in such a way as to yield particularly desirable aerodynamic results.

3. Figure 1 represents the pressure distribution on a typical "streamline" form or nonlifting airfoil section. The minimum pressure occurs at A. From this point aft a pressure recovery begins, which might continue to 3 under favorable conditions without a breaking down of the associated boundary layer along the airfoil surface from the low-drag laminar form the higher-drag turbulent form, although the rising pressure tends to promote such a flow breakdown. At the point B, however, which represents the laminar-separation point, the flow <u>must</u> shortly thereafter either break down or lead to even more unfavorable separation effects. In any event, in the practical case an unnecessarily large portion of the surface is exposed to the drag-producing scouring effect of the turbulent boundary layer.

4. In figure 2 the form is altered in order to control the boundary layer. An altered distribution of thickness and relatively fine leading edge may be chosen to give a pressure distribution of the type shown having a gradual falling pressure over

the major part of the surface to the point A. In the practical case the transition to the turbulent flow again occurs between A and some point near B but only a relatively small proportion of the surface near the trailing edge is exposed to the scouring action of the turbulent flow.

5. In figure 3 the new form is further changed by curving the mean line in such a way that the airfoil may develop lift. A mean-line curve is chosen so that the lift is distributed in a way that does not affect the desirable character of the pressure distribution on either surface.

6. In figure 4 is indicated how the new airfoil form may be extended to provide a larger wing area or to permit better flow conditions at the trailing edge. A good compromise form, arrived at through tests, is shown in figure 5.

7. At large Reynolds numbers (large wings and high speeds) the boundary-layer control exerted by the falling pressures may become insufficient to delay the transition to the point A. This difficulty is overcome by removing a part of the boundary layer through slots in the surface or through a porous material forming all or part of the wing surface, and into the hollow wing interior where it is removed and discharged by means of a blower or other suitable device.

8. I claim:

(a) Wing-section shapes differing from the usual streamline forms which are altered from the conventional form with the object of controlling the boundary layer along a major part of surfaces in such a way as to yield particularly desirable aerodynamic results.

(b) Wing-section shapes designed to avoid rising pressures in the downstream direction over an extensive part of the airfoil surface with the object of reducing the drag.

(c) Wing-section shapes having the minimum pressure on both surfaces well aft.

(d) Wing section shapes having the laminar separation point on both surfaces well aft.

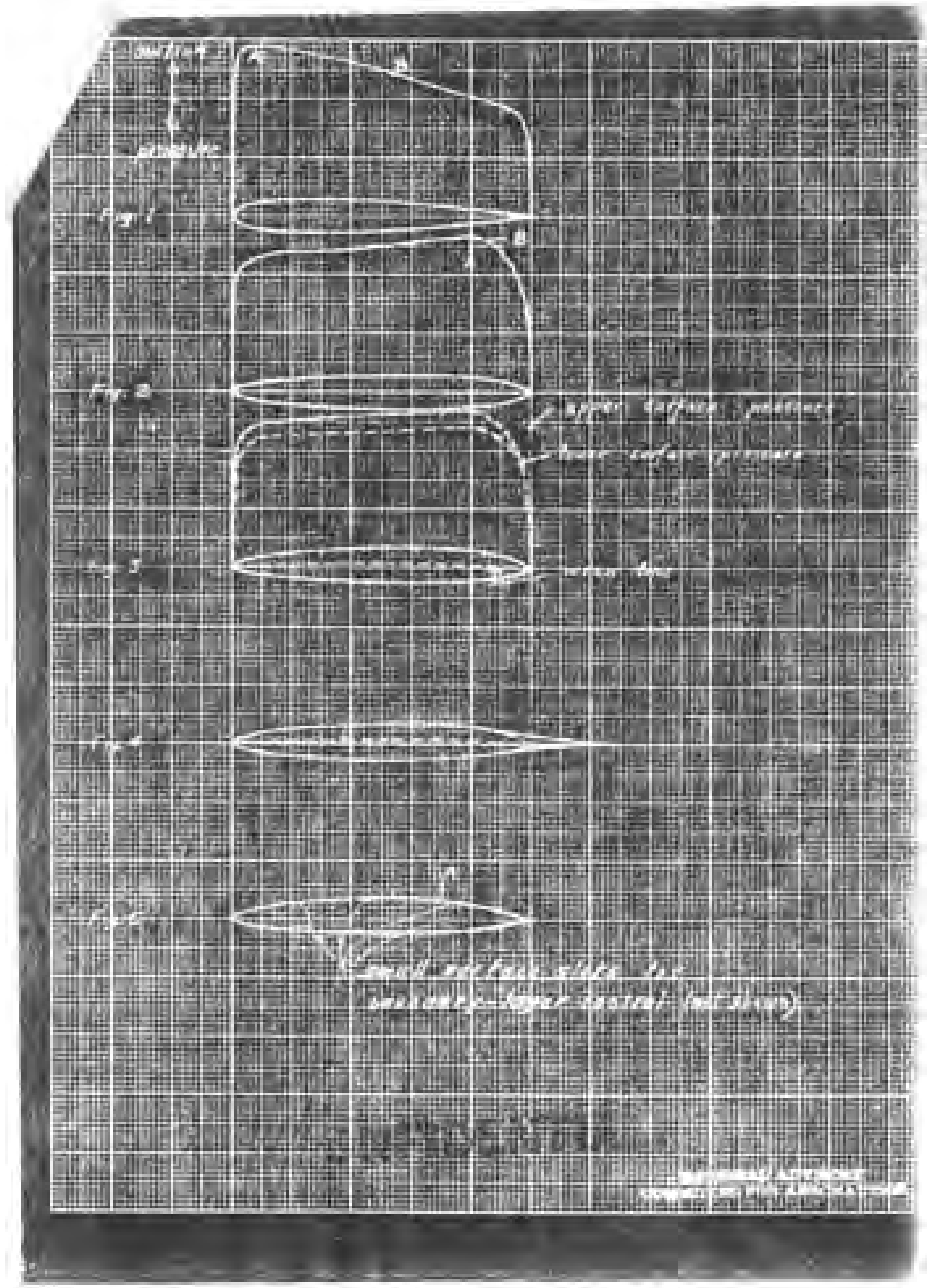
(e) The above-mentioned shapes further altered to permit carrying lift without a sacrifice of other desirable characteristics.

(f) The above-mentioned shapes further altered to give reflexed contours behind the minimum pressure point.

(g) The above-mentioned shapes further altered or supplemented by the addition of the usual forms of boundary-layer control.

(h) The above-mentioned shapes combined with boundary-layer removal through a porous material forming all or part of the wing surface.

Eastman N. Jacobs,
Senior Aeronautical Engineer.



Document 4-22

Newton H. Anderson, *Aircraft Layout and Detail Design* (New York and London: McGraw-Hill, 1941), pp. 83-89.

Newton H. Anderson was a Douglas Aircraft Company engineer who served as program director of the education department of the Institute of Aeronautical Sciences. Chapter 3 of his 1941 textbook was dedicated to "Airfoils," and nowhere in the engineering literature can one find a more instructive testimony to the value of NACA research in general and its systematic airfoil program in particular. Section 3.1 of his text supplies a good indicator of how influential the NACA airfoil work was to American industry. But it is Sections 3.5 and 3.6 that are especially remarkable, in that the whole method of airfoil design prepared by Anderson was based on NACA research.

Section 3.8 is based on a popular article, "From the Wind Tunnels of Langley," which appeared in the March 1941 issue of *Fortune* magazine. It provided an excellent and easily understood summary of the NACA's crucial role in airfoil development up to the start of World War II. Readers will want to look in particular for the paragraph that begins, "The N.A.C.A. has developed a secret airfoil," which could do no more than hint at the confidential laminar-flow development given how little public information has been released about it.

Document 4-22, Newton H. Anderson,* Aircraft Layout and Detail Design*, 1941.

AIRCRAFT LAYOUT AND DETAIL DESIGN
By:
NEWTON H. ANDERSON, B.S.

3.8 The N.A.C.A. It has been shown in this chapter what an important part the National Advisory Committee for Aeronautics has played in the development of airfoils. However, the work of the N.A.C.A. is not limited to airfoils, but covers practically all branches of the aeronautical sciences. Whether the layout man realizes it or not, practically all of his design has been affected in some manner by the findings of this committee.

It was in 1915 that Congress established the N.A.C.A. with an appropriation of $5000 a year to "supervise and direct the scientific study of the problems of flight." As a measure of its growth and of the importance of its work, it should be noted that the appropriations for 1941 exceed $11,000,000.

The President appoints the committee of fifteen to serve without pay. Six are specially qualified civilians, two each from the Army, the Navy, and the Civil Aeronautics Administration, and one each from the Weather Bureau, the Smithsonian Institution, and the National Bureau of Standards. There are four major technical committees: (1) Aerodynamics, (2) Power Plants, (3) Aircraft Materials, (4) Aircraft Structures, whose titles define the scope of their work. The actual work is done by a staff of nearly a thousand employees.

The original laboratories are at Langley Field, Va., where practically all the research has been carried on in the past. However, a new engine research laboratory is being constructed at Cleveland, Ohio; at Sunnyvale, Calif., a new aerodynamic laboratory, the Ames Aeronautical laboratory, is being built at the old Navy dirigible base.

Fortune Magazine in its article on the N.A.C.A. stated that a perfect definition of the aircraft designer's goal is "the reduction of drag." This statement is no exaggeration for the aircraft designer must strive for better performance and more speed, and there are many things he may do to help attain that goal.

Since the reduction of drag is so important, the various types of drag and their meaning should be discussed. "Drag" is the term applied to those forces which resist the forward movement, or the lifting capacity, of an airplane. Probably everyone has tried holding his arm out of the window of a moving automobile and has experienced that invisible force of the air trying to push the arm back. As an airplane flies, the air is resisting its forward movement, juts as the air resisted the forward movement of the autoist's arm. The unfortunate thing about drag is that it increases as the square of the speed. An airplane flying at 200 m.p.h. will have four times as much drag as at 100 m.p.h. At 300 m.p.h. it will have nine times the drag of 100 m.p.h.

Some drag can be reduced, some cannot. Induced drag is the inevitable price paid for lift and cannot be reduced except very slightly. It is extremely difficult to define "induced drag" and to show clearly why it cannot appreciably be reduced. A simple analogy of an object moving through water may be drawn: the water is parted by the moving object and flows together in its wake, which sets up eddies. Energy, which can never be recovered, was imparted to the water to set it in motion. The airplane as it passes through the air moves this air; energy was expended to set it in motion. The imparting of this motion to the air manifests itself in the form of lost energy which is known as induced drag.

Parasite drag included the various kinds of drag that can be reduced. "Profile drag," one form of parasite drag, is a term applied to a wing to describe the effect of turbulence in the thin layer of air close to its surface and in its wake. Another form of parasite drag is interference drag which is due to eddies set up by the proximity of two structural members. Skin friction, another type of parasite drag, results when particles of air are forced along the exterior surface of the wing. When any liquid flows through a tube or pipe, the friction between the liquid and the walls of the tube resists the flow of the liquid. Since air is a liquid, there is a similar action between the air and the skin of the airplane.

The N.A.C.A. has led the world in the attack on parasite drag. Some of the more important contributions may be listed as: airfoils, engine cowling, location of nacelles, location of the wing in relation to the fuselage, and a general cleanup of struts, landing gear, etc.

Until 1924, the Clark Y was considered a very good airfoil, but in that year the N.A.C.A. brought forth their "twenty-three-oh-twelve" (23012) which proved so superior to previous airfoils that approximately three-fourths of the world's airplanes today are using versions of this scientifically famous airfoil. Among other things, it is distinguished for its high ratio of lift to drag. This ratio (usually expressed as L/D or in conversation as "L over D") is similar to an efficiency factor; the higher the ratio, the greater the efficiency. In the 23012 airfoil, L/D goes up to 24, which is considered very good. This airfoil has an unfortunate tendency to stall suddenly, that is, it suddenly loses its lifting power. However, ways have been discovered to alleviate this tendency to stall, which will be discussed later.

Until about 1931, air-cooled engines had no cowling; the radial cylinders were fully exposed to the air stream. The N.A.C.A. developed a cowling that has been universally adopted and it is known as the N.A.C.A. cowling. It added approximately 20 m.p.h. to the first plane built with it, Frank Hawks's Lockheed. Although the engine is apparently closed in by the cowling, it directs the air stream so that cooling is more effective with the cowling than when the cylinders were fully exposed.

An exhaustive series of tests were performed in order to determine the most efficient location of the engine with respect to the wings for multi-engined airplanes. As a result of these tests started in 1929, the location of the engine nacelles in the leading edge of the wing is now standard, as in the Douglas transports. It has also been determined that a pusher propeller is more efficient than a tractor because it does not disturb the air flow over the wing. However, the pusher propeller has not received very wide use because of practical difficulties such as structural support, and equipment already in the wing as tanks, guns, landing gear, etc.

In the N.A.C.A.'s studies on the location of the wing with respect to the fuselage, it has been found that the mid-wing offers a minimum of interference drag, the high wing next, and the low wing a maximum. All three positions are used in today's airplanes; each position having certain advantages and other disadvantages. Although the mid-wing is most efficient from an aerodynamic standpoint, the spars passing through the fuselage may very effectively block any freedom of movement of crew or passengers in the fuselage itself. The high wing affords splendid vision for pilots and passengers; yet the problem of retracting the landing gear may become extremely complicated because of the large struts necessary. The low wing, although least desirable from an aerodynamics and visibility standpoint, has definite structural advantages. The spars through the fuselage are usually below the cabin floor level, and the shorter struts for the landing gear enable the wheel and mechanism to be neatly retracted into the wing. The logical answer then is the large fillets between the wing and fuselage which are so familiar on low-wing monoplanes today.

One has only to compare the latest designs with airplanes of a dozen years ago to appreciate the N.A.C.A.'s general cleanup of struts, wires, landing gear, landing lights, etc. The net result is that a present-day four-engined transport has but 40 percent of the drag of a single-engined monoplane of a dozen years ago.

There are but two ways to reduce parasite drag: smooth out the surfaces and change the contour. It is interesting to note that the N.A.C.A. found that a transport at 225 m.p.h. with lap joints and 3/32 in.-diameter brazier-head rivets required 182 hp. to pull them through the air. When flush joints with 1/16 in.-diameter brazier-head rivets were substituted, only 82 hp. was required. Even a coat of spray paint added 91 hp. more than the polished metal surface. Every layout man has had experience with flush joints and known the problems that arise at times to maintain a perfectly smooth exterior surface.

The designer can do little toward changing the contour, but the N.A.C.A. has done much, as will be shown. One common affliction of existing airfoils is turbulence in the boundary layer. This boundary layer is the thin sheet of air in contact with the surface and may vary in thickness from a few thousandths of an inch to 1½ in.

The air starts in at the leading edge with a smooth sliding flow called "laminar flow," and at the transition point, usually close to the leading edge, it breaks into turbulence. Everyone is familiar with the behavior of water flowing in a brook as it hits a projecting rock. The water flows around the rock and breaks into a series of eddies and becomes turbulent. Air behaves in the same way, although it is difficult to see and must be studied in smoke tunnels. This turbulence imposes profile drag which was mentioned earlier in this article.

The N.A.C.A. has developed a secret airfoil, commonly known as the "laminar wing," in which the transition point is almost at the trailing edge. Very little can be told about this wing except that its leading edge is rather thin and that the maximum camber (thickest portion) is father aft than usual. A glove-like structure having this new airfoil was built on a wing of a Douglas B-18, and the N.A.C.A. was able to check its flying characteristics. The one main objection to the laminar flow wing is that the surface must be as smooth as an automobile fender. A piece of scotch tape on the leading edge will make it break into turbulence, which explains the reason for such care being given to problems of flush joints on laminar flow wings. Laminar flow airfoils are being used in the propellers on many of today's aircraft, increasing the propeller's efficiency.

When parasite drag is fully under control, the only limitations on aircraft design will be those placed by nature: the height of the atmosphere (approximately 50 miles) and the speed of sound. At the speed of sound (approximately 1100 ft. per second or 750 m.p.h.), there occurs a peculiar form of drag known as "compressibility burble" which imposes a prohibitive drag, as far as efficient aircraft design is concerned. If it were not for compressibility burble, the range of gun-fired projectiles would be greatly increased; in this case, by increasing the powder charge, enough additional energy can be imparted to the projectile to overcome this drag. Obviously, such methods are impossible with aircraft, so that, unless some new

means of propulsion is devised or unless means of controlling compressibility burble is discovered, the speed of an airplane will be limited by the speed of sound.

The N.A.C.A. has done much work on lift. The present-day flaps for landing which are standard on all airplanes result from N.A.C.A. tests and research. When the flaps are lowered at relatively slow speeds, they increase both the lift and drag, which not only slows the speed of the airplane, but also enables the airplane to remain aloft at a lower rate of speed. At times, in order to provide more flap area, the flaps are extended to the wing tips. This presents a problem of where to place the ailerons. One solution is to mount them on top of the wing on short vertical masts, in which case the aileron is known as a "spoiler." Other work of the N.A.C.A. relates to stall—that peculiar characteristic of a wing where, at certain attitudes of flight, the lift suddenly drops. This stalling characteristic of the N.A.C.A. 23012 and related airfoils was mentioned previously in this article. To alleviate this tendency to stall suddenly, the N.A.C.A. recommends giving the wing tips "wash out" which is no more than twisting the wing so that the incidence at the tip is less than that at the root. This explains why in the original design of a wing, the incidence so often decreases toward the tip. (See Fig. 3:3 where the incidence changed from 5 deg. at the root to 3 deg. at the tip.)

Although the model to be tested in a wind tunnel is usually mounted in the air stream on delicate measuring devices, the N.A.C.A. may use their free flight tunnel in which an exact model correctly balanced to agree with the actual airplane is actually flown. A tiny electric motor usually drives the propeller while the operator, by means of fine trailing wires, may operate the various control surfaces. Another operator controls the speed of the air stream so that the model will not smash itself against the walls of the tunnel.

The free spin tunnel may be used if the spinning characteristics are being investigated. Here the model, controls set for a spin, is tossed into a vertical tunnel having an uprushing blast of air. Then the operator moves the controls by means of the fine trailing wires and the recovery is studied. Before the N.A.C.A.'s spin tunnel, it was necessary for a pilot to take the airplane aloft, throw it in a spin and then hope for the best. If it did not come out of the spin, it was necessary for the manufacturer to build another airplane, making whatever changes were deemed advisable, and try it again. Now it is possible to make these changes in the model, thus saving much time and expense. The N.A.C.A. has found that the design of the tail is nearly everything in spin recovery. The vertical stabilizer and rudder should be large, with the horizontal stabilizer and elevator mounted rather high so as not to blanket the vertical surfaces while the plane is spinning.

Everyone who has ridden in an airplane is familiar with bumps" which are usually caused by uprising drafts of air. If the N.A.C.A. is investigating the behavior of an airplane in these vertical drafts, they use their gust tunnel. Here a model is catapulted into uprising currents of air, and its behavior is studied by means of cameras, etc. They have developed what is known as "V-G recorders," measuring instruments that record the severity of the bumps. They are carried on many planes in flight to measure actual flight conditions; very small ones are also carried by the models in the gust tunnel. There are many other tunnels and laboratories used by the N.A.C.A. in their research; however, it is impossible to discuss them all in this text. From the foregoing, the student can gain some idea of the magnitude and the complexity of the work. Much investigation has been done on materials, and many of them have been improved as a result of N.A.C.A. research and tests.

One of every service model of an Army or Navy airplane automatically goes to Langley Field for test, and invariably leaves 20 to 60 m.p.h. faster. The Bell Airacobra gained approximately 60 miles by changing air scoops, supercharger location, wheel wells, etc. When the manufacturers submit designs of a new model to the N.A.C.A., an opportunity is offered to eliminate some problems before they arise. The new Vought Navy fighter had the air ducts for the oil cooler changed, and the list could go on indefinitely.

Document 4-23(a-d)

(a) Eastman N. Jacobs to Engineer-In-Charge, "Conversation with Dr. Lewis regarding application of laminar-flow airfoils," 3 February 1939, RA file 290, LHA, Hampton, Va.

(b) H.J.E. Reid, Engineer-in-Charge, LMAL, to NACA, "Aerodynamically smooth finishes for airplanes—information for Vultee Aircraft, Inc.," 28 November 1940, in RA file 290.

(c) G.W. Lewis, Director of Aeronautical Research, to the Chairman of the NACA (Dr. Joseph Ames), "Investigation of laminar-flow low-drag wings," 27 November 1939, RA file 290.

(d) G.W. Lewis, Director of Aeronautical Research, to Langley Memorial Aeronautical Laboratory, "Investigation in flight of laminar-flow low-drag wings," 31 January 1940, RA file 290.

This quartet of documents involves NACA plans for flight-tests of laminar flow-wings in 1939 and 1940 and concern for the smoothness of wing finishes.

The first document concerns tests with a Douglas B-18 airplane, which would eventually be conducted at Langley in the spring of 1941. Langley installed an experimental low-drag test panel on the wing of the bomber and fitted the panel with suction slots and pressure tubes for a free-flight investigation of the transition from laminar to turbulent flow in the boundary layer. The pressure of each tube was measured by liquid manometers installed in the fuselage. Most significantly, an extraordinary amount of care had to be given to the finish of the laminar-flow test panel in order to make its surface as smooth and fair as possible. No less than 48 coats of paint and lacquer were applied to the laminar-flow test panel, and rubbed down with weather-dry paper after the twelfth and twentieth coats. (Normally, an airplane wing had only two coats of paint.) In addition, a proxlyn-glazing putty was also used to fill in surface depressions. Obviously, actual wings on operational aircraft were never going to be as smooth as the laminar-flow test panel, a fact that eventually compromised the aerodynamic performance of the NACA low-drag airfoils to a significant degree.

The second document in the string provides the NACA's response to a request from Vultee Aircraft, Inc., to know what finishes had been found to be the smoothest for wings.

The other two documents show that the NACA was interested in designing a small, laminar-flow "research airplane." If this had been done, it would have represented the first specially-built NACA research airplane in the organization's history, predating its involvement in the XS-1 transonic research airplane program by some five years.

The final document in the string reveals that the Committee on Aircraft Structures (chaired by Dr. Lyman J. Briggs of the National Bureau of Standards) approved of the idea on November 1939. The laminar-flow research airplane was never built, though scale-models of the concept were conducted in wind tunnels into early 1941. The airplane was a high-wing, pusher monoplane with exceptionally small tail surfaces. Pusher propellers were used to eliminate the undesirable effects of the slipstream on the flow over the wing. The tail surfaces were designed to give low static stability on the premise that a smoother riding airplane could be achieved inasmuch as gusts would produce straight sideways or vertical displacements rather than yawing or pitching moments.

Early in 1941, the NACA decided to push the development of its laminar-flow airplane as a large long-range bomber; however, it soon abandoned the project when the army became interested in the low-drag airfoil and it became evident that the P-51 would be flying long before the laminar-flow airplane could be put into the air.

Document 4-23(a), Eastman N. Jacobs to Engineer-In-Charge, "Conversation with Dr. Lewis regarding application of laminar-flow airfoils," February 1939.

Langley Field, Va.
February 3, 1939.

MEMORANDUM For Engineer-in-Charge.

Subject: Conversation with Dr. Lewis regarding application of laminar-flow airfoils.

1. In a conference January 30, 1939, attended by Dr. Lewis, Messrs. Reid, Miller, and Jacobs, the subject of our laminar flow investigations was discussed. First the fundamental work on transition was discussed and then further consideration was given to applications in flight of the new airfoils. It was first agreed that further fundamental flight investigations is essential to determine beyond question the limiting extent of the laminar boundary layer.

2. The application of the new airfoils to military airplanes was then considered. In reply to Dr. Lewis' questions concerning my recent memorandum requesting the loan of a B-18 from the Army, I indicated that the suggested procedure does not differ essentially from the plan he had suggested in an earlier conference when it was agreed that we should aim at the eventual application of the low-drag airfoil with boundary layer control to a P-39 airplane specially modified for the purpose. I pointed out that the suggested investigation with a glove on the B-18 as well as the high-speed-tunnel tests of the N.A.C.A. 472-212 will supply design data desired for the application to the P-39. He thought, however, and I agree also, that the 8 or 9 month period estimated for the B-18 tests indicated an excessive delay in the P-39 project. He was also reluctant to ask for the B-18 for such a long period. It was finally agreed, therefore, that the B-18 tests should be planned so that they can be completed much more quickly.

3. It is therefore recommended that design work on the B-18 be started immediately, and whole project expedited as much as possible.

Eastman N. Jacobs,
Senior Aeronautical Engineer.

Document 4-23(b), H.J.E. Reid, Engineer-in-Charge, LMAL, to NACA, "Aerodynamically smooth finishes for airplanes—information for Vultee Aircraft, Inc.," November 1940.

Langley Field, Va.
November 28, 1940.

From LMAL
To NACA

Subject: Aerodynamically smooth finishes for airplanes-in-formation for Vultee Aircraft, Inc.

Reference: NACA Let. Nov. 14, 1940, MMM AMJ ldl, enc.

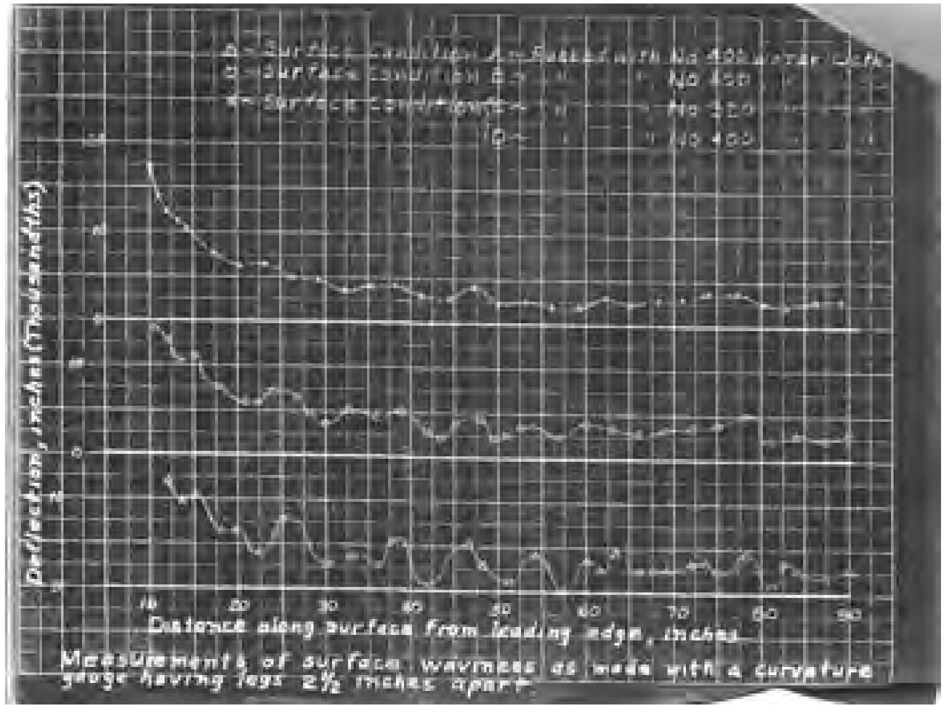
1. The Vultee Aircraft Company's letter dated November 7, 1940, which was forwarded by your office to the Laboratory and in which information is requested regarding finishes for airplanes both from the standpoint of obtaining smoothness and also means of determining the degree of smoothness obtained, has been brought to the attention of Messrs. E. N. Jacobs, Ernest Johnson, and members of the flight section.

2. Based on his experience with finishing models for laminar-flow studies, Mr. Jacobs refers to pages 8 and 9 of his Advance Confidential Report entitled "Preliminary Report on Laminar-Flow Airfoils and Methods Adopted for Airfoil and Boundary-Layer Investigations." Commenting further, Mr. Jacobs states that

> The work we have done since that time indicates that somewhat greater local roughness may be acceptable as aerodynamically smooth under some conditions. Our usual requirement, that no surface imperfection that can be felt should be accepted as aerodynamically smooth, thus seems, at least in some instances, to be somewhat too severe. On the other hand, bumps or depressions producing unfairness of a relatively long wave length appear to produce small adverse effects on transition, even though they are very difficult to detect without the use of a curvature gauge. We think at present that this type of surface defect or unfairness should in general be held sufficiently small so that it will produce no noticeable effects on the pressure distribution on the wing surface.

It is admitted that these requirements are rather indefinite, but there can be no assurance that very definite requirements may ever be reached, except in relation to specific situations. Although we have started a general investigation of these requirements of aerodynamic smoothness, it is considered better not to wait for any general results but to consult us about specific cases.

3. For the information of your office, it might be said that Mr. Jacobs makes the suggestion that since in two instances we have offered to investigate in the low-turbulence tunnel models to be supplied by manufacturers, and, further, since due to congestion in the Laboratory shops occasioned by the current which has handicapped somewhat our supply of models for this tunnel, pressure of work, if in agreement with the policy of your office, it might be deemed desirable to request them to supply a built-up test specimen for testing in this tunnel. It is believed that such a procedure would yield results which will mutually benefit the Laboratory as well as the manufacturer. If such a procedure is followed, a small portion of an actual wing or a built-up test specimen, having a spanwise length of 35 ¾ inches and any chord less than 90 inches, could be conveniently tested.



4. For the information of the Vultee company, the procedure followed in preparing for testing an experimental laminar-flow wing panel for flight tests on a B-18 airplane is given below by Mr. John A. Zalovcik, who is one of the engineers connected with this project. It will be noted in these comments that 48 coats of paint were applied to the wing surface, and it should be explained that at least 10 of these were necessary as a base for the wooden model used in order to waterproof it prior to the subsequent finishing process requiring the procedure of rubbing down with water cloth, and also that several coats were applied simultaneously during the various paint applications. It should suffice to say that with the condition represented by surface condition C and D, shown on the accompanying figure, good results

from the standpoint of drag will be obtained. The procedure mentioned above was as follows:

> The laminar-flow test panel mounted on the B-18 airplane was alternately sprayed with paint and sanded down with various grades of water cloth. The surfaces were first given about 12 coats of grey paint and then sanded with blocks using No. 280 water cloth. The panel was given 8 more coats of grey paint and sanded again. The surfaces were then given 14 coats of white lacquer (with white pigment) and sanded first with No. 320 water cloth and finally with No. 400. An index of surface waviness of the panel up to this point was obtained by making surface measurements with a curvature gage, having legs 2 ½ inches apart. The results of these measurements along the center line of the panel are given in the accompanying figure and are indicated as surface condition A. Amplitudes of the deflections of the curvature gage of about 0.005 inch magnitude were found over a number of sections of the panel surface. To improve this condition, the high places were rubbed down and the low places were filled in with grey paint, and then the entire surface was given 14 coats of grey and rubbed down with No. 320 water cloth. In this latter case the sanding was done with a bow 30 inches long with a strip of aluminum 6 inches wide and 1 1/16 inch thick stretched across ends and to which water cloth was cemented. The aluminum strip was weighted down with shot bags to give necessary pressure. Measurements were again made with the curvature gage and waves of 0.002 inch amplitude were found. (See surface condition B in the figure.) Sanding with the bow was continued until the condition C was obtained. The final surface condition was obtained by adding 8 coats of white lacquer and sanding with No. 400 water cloth. It will be noted that the final coating did not alter the surface waviness but only affected the texture. It may be added that the change in surface condition from that represented in the figure as condition A to that of condition D resulted in extending the run of the laminar-boundary layer about 10 percent of the chord.

5. It may be stated that the paint and lacquer described in the foregoing procedure is what is known as primer surfacer and any good grade of lacquer. In addition to these two finishers, a proxlyn-glazing putty is also used to fill in relatively large depressions in the surface during the process of finishing. It may be stated that the Laboratory has obtained the best results from these paints when thinners used with them are made by the same manufacturer. The putty has been obtained from the Acme White Lead Company, and the paint and lacquer from any of several companies, including Sherwin-Williams, Pittsburgh Plate Glass, and Egyptian Lacquer Company. The water cloth mentioned is commonly known as weather-dry paper.

H. J. E. Reid,
Engineer-in-Charge.

Document 4-23(c), G.W. Lewis, Director of Aeronautical Research, to the Chairman of the NACA (Dr. Joseph Ames), "Investigation of laminar-flow low-drag wings," November 1939.

November 27, 1939.

MEMORANDUM for the Chairman.

Subject: Investigation of laminar-flow low-drag wings.

1. During the past year the Committee has had under investigation in the two-dimensional low-turbulence wind tunnel certain laminar-flow airfoils. This investigation has been conducted under research authorization No. 290, "Investigation of Effect of Thickness and Mean Camber Line Shape on Airfoil Characteristics."

2. The investigation has proved successful. Certain wing forms have been developed having a profile drag of one-third that of the best wings now in use, but at relatively low Reynolds number. Since June, 1939, the investigation has been extended to forms having low drag at higher Reynolds number. Through the cooperation of the Army Air Corps a portion of a wing of the new design is being investigated on a Boeing B-18 bomber.

3. The project has reached a stage when it is necessary to make rather extensive flight tests with certain of the laminar-flow wings. The construction of a wing for laminar-flow presents certain structural problems, since it is necessary that there be no deformation of the wing under load and that the surface remain free of roughness and also free from wrinkles and other deformation.

4. The subject was discussed at the last meeting of the Committee on Aircraft Structures held on November 8. The matter was thoroughly considered and received the approval of that committee.

5.It is therefore recommended that the Committee proceed with the flight tests and that for this purpose a small airplane of approximately 2300 pounds be constructed with a wing incorporating the low-drag laminar-flow design. This airplane would be considered as research equipment for the investigation of laminar flow and not as a particular airplane development of the Committee.

G. W. Lewis,
Director of Aeronautical Research

Document 4-23(d), G.W. Lewis, Director of Aeronautical Research, to Langley Memorial Aeronautical Laboratory, "Investigation in flight of laminar-flow low-drag wings," January 1940.

Washington, D. C.
January 31, 1940.

From NACA
To LMAL

Subject: Investigation in flight of laminar-flow low-drag wings.

Reference: LMAL letter October 30, 1939, WHH. MHY. EWM.

1. At the meeting of the Committee on Aircraft Structures, held on November 8, 1939, there was discussion of the proposal that the Committee's laboratory construct a small airplane of approximately 2300 pounds, with a wing incorporating the low-drag laminar-flow airfoil, this airplane to be considered as research equipment for the investigation and not as a particular airplane development of the Committee. The discussion of this subject in the meeting of the Structures Committee occurred following the discussion of the proposal of the University of Maryland for an investigation of the strength of high-speed wing structures.

2. At the meeting of the Structures Committee the following resolution was adopted:

> "RESOLVED, That the Committee on Aircraft Structures recommends to the Executive Committee that approval be given to the Langley Memorial Aeronautical Laboratory to design and construct, or to have constructed, an experimental airplane incorporating laminar-type airfoil wing or wings to have places for two to four people and to fly at approximately 200 miles per hour with a useful flight range."

3. I have taken up with the Chairman of the Committee, Dr. Bush, the desirability of expediting this project, and the project has received his approval.

4. A new research authorization will be presented to the Executive Committee to cover the investigation in flight of a laminar-flow low-drag airfoil. The number 720 is tentatively assigned to this research authorization.

5. A separate letter will be written to the laboratory regarding the proposed research authorization to which the number 720 was previously assigned.

G. W. Lewis,
Director of Aeronautical Research

Document 4-24(a-e)

(a) Robert J. Woods, Chief Design Engineer, Bell Aircraft Corp., 2050 Elmwood Avenue, Buffalo NY, to NACA, "Attn: Dr. George W. Lewis," 25 March 1940, RA file 290, LHA, Hampton, Va.

(b) G.W. Lewis, Director of Aeronautical Research to Langley Memorial Aeronautical Laboratory, "Request for information on new airfoil sections—Bell Aircraft Corporation," 26 March 1940, RA file 290.

(c) H.J.E. Reid, Engineer-in-Charge, to NACA, "Request for information on new airfoil sections—Bell Aircraft Corporation," 27 March 1940, RA file 290.

(d) G.W. Lewis, Director of Aeronautical Research, to Robert J. Woods, Chief Design Engineer, Bell Aircraft Corporation, 29 March 1940, RA file 290.

(e) Eastman N. Jacobs to Engineer-in-Charge, "Visit of Mr. Robert J. Woods of Bell Aircraft to LMAL, February 5, 1941," CONFIDENTIAL, in RA file 290.

It did not take long for American industry to seek help from the NACA on the application of low-drag airfoils to the design of new aircraft. This string of five documents exemplifies how such an interaction in one case began. Robert J. Woods, Bell Aircraft's chief designer (and former NACA Langley researcher, 1928-29), contacted the NACA by letter on 25 March 1940, stating that "we have heard that your laboratory are [sic] conducting tests on laminar flow airfoils." Woods wanted up-to-date information that might help his company design the wing for its new airplane, the P-39 Airacobra, and he wanted the help as soon as possible.

From the back-and-forth within the NACA and from the response Woods received four days later from George Lewis, director of research for the NACA, it is clear that the NACA planned to proceed with caution in releasing data about the new airfoils. Lewis hoped that he could give Woods what he wanted in about a month.

The most interesting comment in the last document in this section is perhaps Eastman Jacobs's comment about the effect of dusty wings on the aerodynamic efficiency of the laminar-flow airfoils. (In actual performance during the war and afterwards, many aeronautical engineers in the U.S. would express disappointment in the NACA's low-drag airfoils because operational aircraft failed to achieve the very low drags measured in the wind tunnel because the surfaces of their wings could not be kept clean and smooth.) In this memo Jacobs noted that dusty wings "revert to low drag" as an airplane reaches high speed, "thus blowing the dust off the wing." To some extent, this proved to be the case, but the overall maintenance problems involving operational wings that had to be kept extremely smooth and fair meant that the problem would not be mitigated as much as Jacobs thought.

The identity of the "Colonel Green" mentioned in document 24-e should also be made clear. Carl Greene served in the late 1930s for the U.S. Army as chief of the engineering division of the Air Service Technical Command. In March 1939 he moved from Wright Field to Langley. His new job was to provide more regular liaison between the applied research and development activities of the Air Corps and the more basic research of the NACA. Besides funneling information to appropriate Air Corps offices, the occupants of "Greene House" across from the LMAL administration building enabled the army to keep up better with the detailed requirements of NACA Langley's research methods, facilities, programs, and personnel. To complete the conduit, the NACA later created its own liaison office at Wright Field.

Document 4-24(a), Robert J. Woods, Chief Design Engineer, Bell Aircraft Corp., 2050 Elmwood Avenue, Buffalo NY, to NACA, "Attn: Dr. George W. Lewis," March 1940.

BELL AIRCRAFT CORP.
2050 Elmwood Avenue
Buffalo, N. Y.

March 25, 1940

National Advisory Committee for Aeronautics
Navy Building
Washington, D. C.

Attention: Dr. George W. Lewis

Dear Dr. Lewis:

We have heard that your laboratory is conducting tests on laminar flow airfoils of the low drag at high speed angle of attack variety, at full scale Reynolds number values, both in the laboratory and in free flight tests.

We would like to ascertain at this time if any of this work has progressed to the point where you may send us data or information on your test results. Our present problem is for an airplane of 80 inch root chord, 50 inch tip chord. Symmetrical root airfoil section with a 15 percent thick maximum ordinate. Tip airfoil section N.A.C.A. 23009 with minor modifications to the under surface at the nose. The design velocity is approximately 475 m.p.h. at 20,000 ft. altitude, which corresponds to a Reynold's number value of approximately 18,000,000.

To be of use to us on our current design problem we must have any data you may be able to send us on or before April 3, 1940. We will greatly appreciate your comments on our problems and any help you may be able to give us.

With sincere best regards,

Very truly yours,

Robert J. Woods
Chief Design Engineer.

Document 4-24(b), G.W. Lewis, Director of Aeronautical Research to Langley Memorial Aeronautical Laboratory, "Request for information on new airfoil sections—Bell Aircraft Corporation," March 1940.

Washington, D.C.
March 26, 1940.

From NACA
To LMAL

Subject: Request for information on new airfoil sections—Bell Aircraft Corporation.

1. There is attached herewith a copy of a letter from R. J. Woods of the Bell Aircraft Corporation. You will note that they are interested in receiving information on new airfoil sections and that the information to be of use on their current design problem must be received on or before April 3, 1940.

2. I discussed with Mr. Jacobs on my last visit to Langley Field the current development of new airfoil sections, under his direction. I am of the opinion that the Committee should be careful in releasing new airfoil section data until thorough tests have been made and we are very sure of our grounds. Please advise me as soon as possible whether the information discussed with Mr. Jacobs can be released on or before April 3.

G. W. Lewis,
Director of Aeronautical Research.

Document 4-24(c), H.J.E. Reid, Engineer-in-Charge, to NACA, "Request for information on new airfoil sections—Bell Aircraft Corporation," March 1940.

Langley Field, Va.,
March 27, 1940.

From LMAL
To NACA

Subject: Request for information on new airfoil sections—Bell Aircraft Corporation.

Reference: NACA Let. Mar. 26, 1940, L:CMM, Enc.

1. Your letter of reference, together with Mr. Woods' letter of March 25 accompanying it, has been discussed with Mr. E. N. Jacobs, and he states that it will be quite impossible to prepare the airfoil information, which he discussed with Dr. Lewis on his last visit, for release in report form by April 3.

2. Mr. Jacobs states furthermore that it would not be possible, with the information contained in Mr. Woods' letter, to make any suggestion which would be helpful to him in choosing one of the newer airfoils, but he thinks if Mr. Woods cared to visit the Laboratory for a discussion that he might be able to suggest one of the newer sections on which information has been released.

H. J. E. Reid,
Engineer-in-Charge.

Document 4-24(d), G.W. Lewis, Director of Aeronautical Research, to Robert J. Woods, Chief Design Engineer, Bell Aircraft Corporation, March 1940.

March 29, 1940.

Mr. Robert J. Woods,
Chief Design Engineer,
Bell Aircraft Corporation,
2050 Elmwood Avenue,
Buffalo, New York.

Dear Mr. Woods:

I have your letter of March 25, and before replying referred your letter and your questions to the staff at Langley Field.

I have just received a reply. The letter has been discussed with Mr. E. N. Jacobs and others of the staff, and I regret very much that the Committee will not have for release any information on airfoil sections that can be used on or before April 3.

We are pushing the work with reference to new airfoil sections that can be used on high-speed aircraft as fast as we can. We, however, must be sure of the data. I hope that this information will be out within the next four weeks.

Sincerely yours,

G. W. Lewis
Director of Aeronautical Research.

Document 4-24(e), Eastman N. Jacobs to Engineer-in-Charge, "Visit of Mr. Robert J. Woods of Bell Aircraft to LMAL, February 1941.

Langley Field, Virginia
February 5, 1941

MEMORANDUM For Engineer-In-Charge.

Subject: Visit of Mr. Robert J. Woods of Bell Aircraft to LMAL, February 5, 1941.

1. After a general discussion of the Bell P-59 project with Bob Woods and several Army and NACA representatives, which will be reported separately, I had a discussion with Bob Woods in my office after lunch, concerning the selection of a laminar-flow airfoil for the P-39. The triad of such an airfoil on the P-39 is apparently Woods' own idea, although he discussed the matter with Colonel Greene before having the conference with me. We discussed what section thicknesses might be used, and although I said he might use an eighteen-percent thick section, he thought it would be safer to reduce it to sixteen, in view of the possibility that they might change to a higher-powered Continental motor within the next two years, so that the compressibility margin gained by the thinner airfoil could be more conservative. The recommendation was therefore approximately the same as for the P-59 project, for which he already has the required data. It was agreed, therefore, that he would use the NACA 66, 2-116 at the root and the NACA 66, 2-216 at the tip. He has the NACA 66, 2x-015 thickness ordinates, which may be scaled up to sixteen to produce the desired thickness distribution, and we gave him the necessary data on the a = .6 type mean line.

2. For studies of possible use as a tail surface section, we also gave him a copy of the attached thickness ordinates for the NACA 67, 1-015 airfoil.

3. For Mr. Bell's information it should be noted that dusty wings which show a high drag at low speeds have been observed to revert to low drag when the speed was increased, thus blowing the dust off the wing.

Eastman N. Jacobs
Principal Aeronautical Engineer

Document 4-25(a-g)

(a) Eastman N. Jacobs, Principal Aeronautical Engineer, Memorandum to Director of Aeronautical Research (George W. Lewis), "Application of new airfoil data to experimental military airplanes," 11 June 1940, NACA Langley Correspondence Files, Code 173-1, National Archives, Mid-Atlantic Region, Philadelphia, Pa.

(b) George W. Lewis, Director of Aeronautical Research, NACA, to Chief of the Bureau of Aeronautics, Navy Department, Washington, D.C., "Confidential memorandum report regarding the application of new airfoil-section data of the laminar-flow type for current and new airplane designs," 17 June 1940, RA file 290, LHA. (Also includes the attached confidential report, "Immediate Use of New Airfoil Sections of the Laminar-Flow Type," 14 June 1940.)

(c) W.H. Herrnstein, Aeronautical Engineer (LMAL), to Engineer-in-Charge, "Discussion between Dr. G. W. Lewis and members of Laboratory staff relative to airfoil selection problems," 15 July 1940, RA file 290.

(d) Edwin P. Hartman, Western Coordinating Officer of the NACA, Santa Monica, CA., to Coordinator of Research, "Visit to Ryan Aeronautical Company," 13 August 1940, in RA file 290.

(e) Eastman N. Jacobs, Principal Aeronautical Engineer, to Engineer-in-Charge, "Visit to the Buffalo Curtis plant at the request of Don Berlin, September 30, 1940," RA file 290.

(f) Arthur E. Raymond, Vice President of Engineering, Douglas Aircraft Company, Inc., Santa Monica, CA, to Dr. G. W. Lewis, Director of Aeronautical Research, NACA, 15 March 1941, RA file 290.

(g) Elton W. Miller, Chief of the Aerodynamics Division (LMAL) to Engineer-in-Charge, "Visit of Mr. L. C. Miller of the Brewster Company to the Laboratory on January 3, 1941," 4 January 1941, RA file 290.

The first six documents in this string testify to the extremely strong interest shown by many U.S. aircraft manufacturers in 1940 and 1941 for the NACA's low-drag airfoils. The first memo, from Eastman Jacobs on 11 June 1940, is fascinating for its commentary on the difficulties of NACA-army dealings and on how strongly the NACA engineers felt they needed to be in control of their own experimental programs. The name "Diehl" referred to in the first line of the memo was Walter S. Diehl, the U.S. Navy officer in charge of technical liaison with the NACA at the navy's Bureau of Aeronautics in Washington. A construction corps engineer who in his insistence on remaining a technical man refused throughout his career to pursue promotions via sea duty, Diehl was one of the NACA's strongest allies and most intimate associates from within the U.S. military. He was a regular visitor to Langley, and given that the NACA's Washington office was located in the Navy Building, Diehl interacted regularly with his friend and fellow engineer George Lewis.

The identity of "Mr. Deport" is not known. "Mr. J.A. Roche" was Jean Roche, who worked as a civilian aeronautical engineer for Col. Carl Greene in the Army's NACA liaison office at Langley Field.

The other documents in this section reflect not only the eagerness of industry to acquire detailed knowledge of the new laminar-flow airfoils but also the tenuous position the NACA was in concerning the release of information about them. On the one hand, the industry and the military services clamored for information, and the NACA engineers involved in the airfoil development, especially Jacobs, were convinced that it was going to be "easier than expected in practical applications to realize the low-drag properties of the new sections." NACA leadership, however, did not want the airfoils to promise too much; George Lewis in particular wanted to make sure that the NACA had enough solid information on the total performance of the new-type airfoils before turning them over to industry. The NACA prided itself on completely reliable research findings, and it did not want to mislead the country into moving down a technological path that might lead to mistakes and inferior fighting aircraft.

The final document, on the other hand, demonstrates that not everyone in the American aeronautics community was so excited about applying laminar-flow wings—at least not the Brewster Aeronautical Corporation. It also suggests that the U.S. Navy was "not supporting them as they might." This suggestion may shed light on the critical comment made by Jacobs about Walter Diehl in Document 25-a.

Document 4-25(a), Eastman N. Jacobs, Principal Aeronautical Engineer, Memorandum to Director of Aeronautical Research (George W. Lewis), "Application of new airfoil data to experimental military airplanes," June 1940.

Langley Field, Va.
June 11, 1940.

MEMORANDUM For Director of Aeronautical Research

Subject: Application of new airfoil data to experimental military airplanes.

1. Following up our discussion with Diehl in your office at which time you asked him to give further thought to possible applications, I asked him at the Laboratory if he had thought of anything yet. His reply was to the effect that he would rather have the Army apply the new wings first. Although I did not argue the point, his position must be considered technically unsound. No one application should be made first. On each experimental type, the best possible compromise should be reached in the choice of the particular wing section for that type in the light of our most recent technical data.

2. Following his comment, however, I contacted the Army Liaison Office here to find out how they feel about it. I discussed the matter with Mr. De Port at Mr. J. A. Roche's suggestion, because he happened to be here at the time, and also with Messrs. Roche and H. J. E. Reid. De Port seemed to get my point of view. In any event, it appears that the Army is willing to cooperate with us in the application of the new wings. We should appreciate that there is some chance of obtaining disappointing results unless the Army, the Committee, and the manufacturer are all behind the project. This requirement greatly complicates our procedure and makes it increasingly clear that we will eventually have to run an experimental airplane-construction shop under our direct control.

3. In the meantime, we must continue to work with the services and the manufacturers. The next step is to find out what experimental projects are possibilities. Roche considers the P-47 one of the best. I would like, therefore, to obtain authority to discuss the airfoil selection problem with Republic representatives.

4. Finally, I agreed with De Port that it would be desirable to give the Army a memorandum indicating the gains possible through the choice of better sections. I plan to prepare and transmit such a memorandum which might also be sent to manufacturers. In return, De Port agreed that he would try to keep us informed through their liaison office here of possible experimental types under consideration.

Eastman N. Jacobs,
Principal Aeronautical Engineer.

Document 4-25(b), George W. Lewis, Director of Aeronautical Research, NACA, to Chief of the Bureau of Aeronautics, Navy Department, Washington, D.C., "Confidential memorandum report regarding the application of new airfoil-section data of the laminar-flow type for current and new airplane designs," 17 June 1940.

June 17, 1940.

Lieutenant Colonel Carl F. Greene,
Air Corps, U.S.A.,
Liaison Officer at the N.A.C.A. Laboratory,
Langley Field,
Virginia.

Dear Colonel Greene:

There is attached hereto a copy of a confidential memorandum, prepared by Mr. Eastman N. Jacobs of our technical staff, on the consideration of airfoil sections of the laminar-flow type for current and new airplane designs. Letters have been written to the chief engineers of the following companies:

Bell Aircraft Corporation
Boeing Aircraft Company
Consolidated Aircraft Corporation
Curtiss-Wright Corporation
Douglass Aircraft Company, Inc.
Lockheed Aircraft Corporation
Glenn L. Martin Company
North American Aviation, Inc.
Republic Aviation Corporation
Vought-Sikorsky Aircraft

inviting them to send their engineers to Langley Field for conferences with members of our technical staff on the subject of the selection of airfoil sections. These conferences will be with individual companies, there being no group conferences.

If there are any other companies to which you wish invitations sent, please advise the Committee.

Sincerely yours,

G. W. Lewis
Director of Aeronautical Research.

Document 4-25(c), W.H. Herrnstein, Aeronautical Engineer (LMAL), to Engineer-in-Charge, "Discussion between Dr. G. W. Lewis and members of Laboratory staff relative to airfoil selection problems," July 1940.

Langley Field, Va.
July 15, 1940.

MEMORANDUM For Engineer-in-Charge.

Subject: Discussion between Dr. G. W. Lewis and members of Laboratory staff relative to airfoil selection problems.

1. On the afternoon of Monday, July 15, 1940, Dr. G. W. Lewis called a group to discuss, in the office of the Engineer-in-Charge, the future policy of the Committee regarding the type of airfoil information to be supplied, in view of the present national emergency. Those present were: Dr. Lewis, Messrs. E. N. Jacobs, J. Stack, C. J. Wensinger, I. H. Abbott, and R. G. Robinson.

2. Dr. Lewis called our attention to the fact that this country has set itself the task of building 25,000 fighting airplanes within the next two years, and that the Committee would have to help in this enterprise all they could, in spite of the fact that some of the airplanes to be built would not appear to members of the Laboratory staff as optimum arrangements. He also called attention to the fact that some manufacturers who had formerly been using the 230 series airfoil sections were attempting to develop new sections of their own, with the expectation that these new sections would be superior. One of his reasons in calling the group together was to see if they could not settle upon some course of action to supply the manufacturers with airfoil data that would show improvement over that for the old 230 series sections, but be a sort of compromise between them and the newer so-called laminar-flow types. In Dr. Lewis's opinion, the Committee does not have enough information on these newer type airfoil sections to turn them over to the industry with the expectation of the industry's making use of them. For instance, we know little about them, outside of the drag. Other information needed would be data regarding the maximum lift, pitching moments, and particularly stalling characteristics. This latter point was very strongly stressed.

3. Although Mr. Jacobs objected at first quite strongly to any sort of compromise, he agreed, and the rest of those present concurred, that it would be a very good idea to test some model airplanes in the 19-foot pressure tunnel, equipped with the new wing sections. In this way the best data at present procurable will be obtained for airplanes with the new wings as regards stalling characteristics and maximum lift. Further data will also be obtained concerning any possible peculiarities in the stability characteristics associated with the use of the new airfoils. Certain data on the airplane drag and the effects of additional changes, for which Dr.

Lewis agreed to give the aerodynamics group a reasonably wide latitude, will also be obtained. As regards the wing drag, Mr. Jacobs recommended deducting the value as measured by the wake method in the tunnel, and adding in suitable free air values for purposes of performance estimation. Dr. Lewis recommended that we test the following models in the order given:

XP-41, as modified by Dr. Theodorsen's division.
XP-46
XF4U-1

He further stated that he would get us some information on a couple of bomber models which he thinks we should test.

4. Dr. Lewis believes that the physical research division should be consulted freely on this program, especially so in regard to any possible changes on the after-body of the XP-46, in light of their experience with the XP-41.

5. Dr. Lewis also stressed the need for power plant reliability, and stated that the R-1830 engine, as used in the XP-41 airplane, is probably the most reliable power plant we have today for pursuit use.

W. H. Herrnstein,
Aeronautical Engineer

Document 4-25(d), Edwin P. Hartman, Western Coordinating Officer of the NACA, Santa Monica, CA., to Coordinator of Research, "Visit to Ryan Aeronautical Company," August 1940.

Santa Monica, Calif.
August 13, 1940.

MEMORANDUM For Coordinator of Research.

Subject: Visit to Ryan Aeronautical Company.

4. INQUIRIES AND REQUESTS: Airfoil Data—Mr. Boyd stated that he had heard something of the new high-speed airfoils being developed at the N.A.C.A. and was quite anxious to obtain information and design data on them. He said the Ryan Company was considering some rather high-speed designs, and information on the new airfoils would be extremely helpful even though the final development of the airfoils had not been reached by the N.A.C.A. I'm sure the Ryan Company will appreciate any information the Committee cares to send them on the new airfoils. They use the N.A.C.A. 24xx series on their trainer.

Edwin P. Hartman

Document 4-25(e), Eastman N. Jacobs, Principal Aeronautical Engineer, to Engineer-in-Charge, "Visit to the Buffalo Curtis plant at the request of Don Berlin, September 1940."

Langley Field, Virginia
October 3, 1940

MEMORANDUM For Engineer-in-Charge.

Subject: Visit to the Buffalo Curtiss plant at the request of Don Berlin, September 30, 1940.

1. Mr. Berlin met me for breakfast at the hotel and took me to the plant Monday morning. We discussed some general considerations of the application of the laminar-flow wing to the P4OD, some of the possible difficulties, and what wing area should be used. It was agreed that we should prefer to use the same wing area and plan form as the original airplane. Some general matters of performance requirements and production considerations were also discussed with the vice president, Mr. Wright. He read a telegram he had just received from Lord Beaverbrook offering the Curtiss Company warm congratulations for having met their delivery agreements on one of the first consignments of P40's.

2. Later I was given an opportunity of inspecting their production set-up, mainly on the P40. They have done some remarkable work on assembly line methods and their present large production seems to be moving smoothly. In spite of their greatly improved methods, however, I am more than ever convinced that research should be directed toward the elimination of much of the slow and costly riveting and spot welding.

3. The problems of the application of the low-drag wing were discussed with the project engineer and with Mr. Jenkins and Mr. Child. Mr. Fladder was also present but was interested more in a twin-motored bomber application. We have an inquiry concerning a wing for this project from the Liaison office, so I will review our discussion of it when I answer this inquiry.

4. In regard to the P4OD, I was shown their mock-up of the project and also the mock-up for the P46, which has changed considerably since the full-scale model was tested here. The P4OD is the more interesting airplane from the standpoint of immediate development. In fact, their production facilities have already been organized for it to such an extent that any change in the wing appears to be too late to work in on their first ships. It is therefore unfortunate that we did not get together earlier on a program.

5. The greatest difficulty in the application of the new wing was with space for landing gear retraction. A slightly larger wing area would have avoided the difficulty, but Mr. Berlin thought that the difficulties might be overcome. He also considered small blisters on the lower wing surface against which, I regret to admit, my protest was only weak.

6. The question of changing to a more conservative wing section near the fuselage, an N.A.C.A. 65 Series for example, in order to avoid danger of separation near the fuselage juncture, was also considered with them. Berlin, however, was not much afraid of this situation and considered it primarily a problem in filleting that could be worked out as an addition. He plans, nevertheless, to make some investigations at once of suitable fillets in their wind tunnel. We therefore agreed to go ahead with the original plan to carry the same section right in to the fuselage.

7. Another possibility considered was that of building an experimental wing using makeshift methods. I admire the bold stand Berlin takes on this possibility. Final plans should be made on the supposition that the wing will work out as expected without important changes. If successful, we are then much further advanced after the flight tests. I endorse such methods and strongly recommend that the governments support the project financially.

Eastman N. Jacobs
Principal Aeronautical Engineer

Document 4-25(f), Arthur E. Raymond, Vice President of Engineering, Douglas Aircraft Company, Inc., Santa Monica, CA, to Dr. G. W. Lewis, Director of Aeronautical Research, NACA, March 1941.

March 15, 1941

Dr. G. W. Lewis
Director of Aeronautical Research
National Advisory Committee on Aeronautics
Navy Building
Washington, D. C.

Dear Dr. Lewis:

Through the present preliminary design period of our new light bomber we have been particularly grateful for the assistance we have received from the N.A.C.A. with respect to various design recommendations, particularly for the invaluable aid rendered by your Mr. Eastman Jacobs in the selection of laminar flow airfoil sections.

You are familiar with the fact that we intend to construct a total of three laminar flow airfoil models for N.A.C.A. testing. These models are being made up according to the suggestions given by Mr. Jacobs. It would considerably accelerate our test program if we were to have the privilege of a short visit from Mr. Jacobs, at which time the following would be accomplished:

(a) A decision could be made concerning the satisfactory design of our laminar flow airfoil models with respect to suitability for testing in the new laminar flow tunnel. Such a procedure would obviate possible time delays due to design changes brought about by interchange of correspondence.

(b) A general discussion of the aerodynamic features of our new light bomber with Mr. Jacobs, during which his opinions could be obtained with respect to manufacturing tolerances which can be allowed, still maintaining laminar flow over the wing and high aerodynamic efficiency throughout the design.

In addition to our new light bomber design, we have several projects under way which incorporate laminar flow airfoils. As is usual in such newly developed features, many questions have arisen which are difficult to answer without a background of experience. Feeling that Mr. Jacobs has had this experience and can render us valuable aid on the two items described above, we respectfully request that he be permitted to spend a short time on the west coast in the immediate future if this will not interfere with your plans and schedule.

Very truly yours,

DOUGLAS AIRCRAFT COMPANY, INC.

A. E. Raymond
Vice President Engineering

Document 4-25(g), Elton W. Miller, Chief of the Aerodynamics Division (LMAL) to Engineer-in-Charge, "Visit of Mr. L. C. Miller of the Brewster Company to the Laboratory on January 3, 1941," 4 January 1941, RA file 290.

Langley Field, Va.
January 4, 1941.

MEMORANDUM For Engineer-in-Charge.

Subject: Visit of Mr. L. C. Miller of the Brewster Company to the Laboratory on January 3, 1941.

Regarding laminar-flow airfoils, Mr. Miller stated that he had been able to arouse very little interest on the part of his Company in applying the laminar-flow airfoils. He felt also that the Navy was not supporting them as they might. He wondered whether any further information had been obtained regarding the maximum lift characteristics of the laminar-flow airfoils. Mr. E. N. Jacobs informed him that tests in the 19-foot tunnel on one installation had shown the maximum-lift coefficient to be within about 0.1 of that obtained with a conventional wing. He suggested that if Mr. Miller had a particular application in view that he arrange for a model to be built for tests in the 19-foot tunnel where the lift characteristics will be reliably obtained. He suggested also that he might have a section of wing built for tests in the low-turbulence tunnel where the drag coefficient would be determined and where it would be possible to determine what degree of roughness may be tolerated without sacrificing the laminar-flow properties. An airfoil for this tunnel should have a span of 35-3/4 inches and a chord anywhere up to 100 inches.

Elton W. Miller
Chief Aerodynamics Division.

Document 4-26(a-d)

(a) George J. Mead, Director, Airplane and Engine Division, National Defense Council, Federal Reserve Building, to Aeronautical Board, Room 1907, Navy Building, 28 August 1940, NACA Langley Correspondence Files, Code E38-8, National Archives, Mid-Atlantic Region, Philadelphia, Pa.

(b) Ira C. Eaker, Lieutenant Colonel, Air Corps, For the Chief of the Air Corps, to Dr. George J. Mead, Director, Airplane and Engine Division, National Defense Council, Federal Reserve Building, September 1940, NACA Langley Correspondence File, Code A173-1, National Archives, Philadelphia, Pa.

(c) G.W. Lewis, Director of Aeronautical Research, NACA, to Edward J. Horkey, Aerodynamics Department, North American Aviation, Inc., Inglewood, CA, 1 November 1940, NACA Langley Correspondence Files, Code A173-1, National Archives, Mid-Atlantic Region, Philadelphia, Pa.

(d) Excerpts from North American Aviation, Inc., Manufacturing Division, Engineering Department, Inglewood CA, "Aerodynamic Load Calculations for Model NA-73 Airplane," North American Report NA-5041, 3 March 1941, RA file 290, LHA, Hampton, Va.

This string of documents sheds light on the genesis of the low-drag NACA airfoils that were to be used in the design of the North American P-51 Mustang, or what in the prototype phase was called Model NA-73. As North American engineer Ed Rees later recalled (see Destination Document, this chapter), this was the "design touchstone" of the Mustang: its novel high-lift, low-drag wing. Considered "too revolutionary" by many experts at the time, the North American designers grew totally devoted to it—and thus to the NACA research on which it was based. If the laminar-flow wing had proved a mistake, so, too, would have the Mustang. And the NACA's reputation for outstanding and reliable research might have been irreparably damaged. But the Mustang flew magnificently, in large part because of its wing. Many aircraft experts believe the P-51 represents the highest level of technical refinement ever achieved in a propeller-driven fighter aircraft. They do not get many arguments.

In *Frontiers of Flight: The Story of NACA Research* (New York: Alfred A. Knopf, 1948) author George W. Gray recalled how the airplane came to be. It had its origins, he wrote, in a series of conferences between North American Aviation, Inc., and a British airplane purchasing commission. "As the story goes, in April of 1940 [four months after Mead wrote his letter to the Aeronautics Board], the British gave the North American executives a list of the performance characteristics they wanted in an airplane, and specified that if the order was accepted the design must be completed and the prototype delivered for trial within 120 days."

Gray's story failed to give all the details. When the British commission arrived in the U.S., it meant to buy modified Curtiss P-40s and Bell P-39s. Because the assembly lines of the two companies could not produce all the airplanes Great Britain wanted, the commission also asked North American Aviation, Inc., to consider producing P-40s also. After thinking about it a while, NAA officials suggested to the British that it could offer a completely new and better airplane better suited to mass production, and that it could do it within the stipulated three-month deadline.

North American had by then received preliminary NACA reports on the low-drag airfoils, as had all the other companies building aircraft for the army and navy, and "its engineers were favorably impressed." Russell G. Robinson of NACA Headquarters, soon to be dispatched to Santa Monica, California, to organize the new NACA's West Coast Coordinating Office, helped North American to select the specific parameters for the laminar-flow airfoil shape for the experimental P-51 models, with significant input from Eastman Jacobs at Langley.

Although there was a great deal of concern within the company about selecting a brand new, untried type of airfoil, so much enthusiasm for the laminar-flow wings sprouted among its leaders that North American stuck its neck out and selected one of the NACA's Series 4 airfoils. "The margin of time available within the 120-day limit was so narrow that while the work of adapting the NACA low-drag wing was being rushed to completion by one group, other engineers were developing an alternative wing of conventional design in case the new idea failed to pan out successfully" (Gray, *Frontiers of Flight*, pp. 106-07). Even today, North American's decision to try the new wing seems a tremendous risk, in that the only data available was Jacobs' advance confidential report. The wing could have had poor stall or stability characteristics and any number of unknown problems. Fortunately, it did not.

The British approved North American's preliminary design in early May 1940 and by the end of the month ordered 320 of the aircraft. An XP-51 flew for the first time five months later, in October, and did so extremely well. It entered combat with the RAF in July 1942.

Of course, the wing only partly explained the Mustang's phenomenal success in the air. Later versions of the Mustang (there were several variants of which the P-51D was the most numerous and best known) had a remarkable Merlin engine, built by Rolls-Royce, capable of producing a then-amazing 1505 horsepower at an altitude of nearly 20,000 feet. (The Packard Motor Car Company built the engines under license in the United States.) Other vital statistics worth mentioning was the airplane's great range (1650 miles at a speed of 358 mph and altitude of 25,000 feet) and climbing ability (up to 20,000 feet in 7.3 minutes). It was also the only fighter plane of World War II to fly over three enemy capitals: Berlin, Rome, and Tokyo.

Most significantly from the aerodynamicist's point of view, the Mustang's coefficient of drag was a record low 0.0163, which meant that it was the "cleanest" airplane that had ever flown anywhere up to that time (and for quite a while thereafter). For those interested, Document 26-d provides the basic aerodynamic characteristics of the wing. The individuals involved in the design of the wing at North American were Edward J. Horkey, Irving L. Ashkenas, C. L. David, and H. J. Hoge. In cross-section their wing was slightly thicker than any of the "230" family airfoils, with maximum thickness farther back from the leading edge, nearer the center of the chord. Also, it had a cusped trailing edge.

Document 4-26(a), George J. Mead, Director, Airplane and Engine Division, National Defense Council, Federal Reserve Building, to Aeronautical Board, Room 1907, Navy Building, August 1940.

August 28, 1940

To: Aeronautical Board

Room 1907, Navy Building

From: George J. Mead

It has come to my attention that the North American Company are developing a fighter for the British, which is said to incorporate the new laminar-flow NACA wing. I should like to know, therefore, whether the Board has already approved an export license for this airplane. It does not seem desirable in the interests of national defense that this development be permitted to leave the country before our own aircraft are thus equipped.

Director,
Airplane and Engine Division

Document 4-26(b), Ira C. Eaker, Lieutenant Colonel, Air Corps, For the Chief of the Air Corps, to Dr. George J. Mead, Director, Airplane and Engine Division, National Defense Council, Federal Reserve Building, September 1940.

WAR DEPARTMENT
Office of the Chief of the Air Corps
WASHINGTON

September 5, 1940.

MEMORANDUM FOR: Dr. George J. Mead,
Director, Airplane and Engine Division,
National Defense Council, Federal Reserve Building.

Reference is had to your letter of August 28, 1940, addressed to the Aeronautical Board, requesting information regarding the new laminar-flow NACA wing.

Information has been received from the North American Aviation, Incorporated, that the wing sections being installed on the NA-73 type aircraft being manufactured for the British Government are based on NACA Report No. 411 (Wing Sections of Arbitrary Shapes) published in 1931 and Report No. 452 (General Potential Theory of Arbitrary Wing Sections) published in 1933. These reports are unrestricted publications. It is understood that this new Wing Section being developed by North American is equipped with slotted flaps and is not the Laminar-Flow NACA wing.

The NA-73, single seat pursuit type aircraft was officially released for export sale August 1, 1940, as a result of an Agreement signed by a representative of North American Aviation, Inc. and the War Department, and approved by the Assistant Secretary of War, May 8, 1940. This agreement specifies that the first NA-73 airplane built will be tested by the Air Corps and the fourth and tenth articles delivered to the Air Corps, which will furnish the Air Corps with complete information concerning any new Wing Sections and Flap Installations developed by the North American Company.

The Air Corps is conversant with this wing development and in accordance with the mutual agreement will receive full benefit of the engineering work being done without additional expense. It is believed that it will be to the best interest of the Air Corps to encourage the continuation of the research and development work being done by North American in connection with high speed wing sections for the NA-73 type airplane.

For the Chief of the Air Corps:

Ira C. Eaker
Lieut. Colonel, Air Corps,
Executive.

Document 4-26(c), G.W. Lewis, Director of Aeronautical Research, NACA, to Edward J. Horkey, Aerodynamics Department, North American Aviation, Inc., Inglewood, CA, 1 November 1940.

November 1, 1940.

Mr. E. J. Horkey,
Aerodynamics Department,
North American Aviation, Inc.,
Inglewood, California.

Dear Mr. Horkey:

In response to the inquiries contained in your letter of September 23, 1940, the following discussion has been prepared by our laboratory staff, and the replies are arranged in the same order as the questions in your letter:

I. WINGS

Mr. Jacobs has commented as follows:

A. NA-73X smoothness—In spite of the fact that your measurements show discontinuities at the rivets and joints of only 0.002 inch to 0.003 inch, it is believed that many of the irregularities shown are too large to permit the maintenance of laminar flow over them. Aside from the wrinkles, however, it is believed that most of these defects may be removed by means of points.

B. Machine guns—It will be very difficult to realize laminar flow over the part of the wing behind the machine gun blast tubes, and it is our experience that it cannot be done unless air is taken in the opening at the leading edge and unless the opening is located very near the front stagnation point. In this respect, the 50-caliber gun is located slightly too high and the outboard 30-caliber gun slightly too low. The other 30-caliber gun is much too low. To maintain laminar flow, the vertical height of the opening should be made as small as possible, say 3/4 inch. The length might be equal to the blast tube diameter, forming a spanwise slot faired out to a point at either end. The air flow discharge opening shown is considered satisfactory except that it should be restricted at the extreme back edge of the slot until the air leaving the slot at nearly flight speed will produce a flow of air into the intake openings at a speed of the order of one-third the flight speed.

C. New type section—Attached are ordinates for the NACA 65, 2-213.5 airfoil, which you might try for the tip. Its use will produce a slight discontinuity in the spanwise fairing lines where it joins the original wing at station 190.5. This defect is not considered serious, however, and the new section at the tip should produce a lower drag and a higher maximum lift. Of course, the section at station 190.5 cannot be altered, the new section fairing in from the 50-inch chord section at the tip to the old section. So little wing area is thus involved in the change that it is considered doubtful whether much change in the stalling characteristics will be observed.

D. Latest data on new wing sections—If you were designing a completely new wing, we would recommend the usage of the newer sections having larger leading edge radii and a somewhat more aft position of the minimum pressure. You could thus realize the possibility of obtaining lower drags, higher maximum lift coefficients, and somewhat increased critical speeds. If you contemplate building a new wing we will gladly recommend suitable sections.

E. Flaps—Slotted flaps have not been investigated on the newer types of laminar-flow airfoils, although some data have been obtained on split flaps. Because the behavior of the split flap on the new airfoils is about the same as its behavior on the old conventional airfoils, we expect the aileron type flap to behave similarly. Some tests of the aileron type flap in the high-speed range tend to confirm this belief.

Finally, we would greatly appreciate any information you can send us about the new airfoil as a result of your flight tests. We would be particularly interested in a comparison of the wind tunnel and flight stalling characteristics. It is understood that the first airplane will be delivered to the Army Air Corps. If it can be arranged with the Army, it seems desirable that the airplane should be brought to Langley Field so that we can make wing drag measurements on it in flight. Your cooperation on such a project would be appreciated.

II. POWER PLANT PROBLEMS.

Messrs. Silverstein and Biermann have prepared the following discussion:

A. WING DUCTS.

1. The duct leading edges on figures II-A-1-a and II-A-1-b appear satisfactory. If trouble is experienced on the model with early stalling it may be cured either by increasing the camber on the upper leading edge of the duct, or by lowering the duct-inlet opening. For the high-speed condition the symmetrical opening as shown in the photographs is slightly preferred.

2. Since sliding doors are liable to stick in operation due to wing deflections, dirt, etc., we have never worked much with them. We prefer regulation by means of a concealed flap such as shown in figure 6f of the advanced confidential report entitled “Full-Scale Wind Tunnel Investigation of Wing Cooling Duct” by F. R. Nickle and Arthur B. Freeman. A plain outward-opening flap may also be used; however, it is more costly in drag.

B. PROPELLER CUFFS.

Mr. Biermann has commented as follows:

1. Cuffs of a fineness ratio of 3.5 are about the same we have been testing and appear to be about as good as we can do at the present. Although the structural problem is not mine I doubt whether the cuffs will stay on if they are built according to the drawing. The centrifugal load must pass from the skin to the shank casting through four points as shown and I think the skin will tear out there. The problem is to distribute the load over a wide area of the skin.

We have calculated the angles recommended for the cuff settings based on our experience and tests of cuffs. Enclosed is a chart giving the angles computed.

III. FLIGHT TESTING.

A. AILERON CRITERIA

Mr. Gilruth has prepared the following discussion on this item:

1. Various pursuit types tested gave values of pb/2v ranging from 0.12 to 0.079, all of which were considered satisfactory by pilots.

2. With regard to the example given, it has been our experience that it is unwise to depend on aileron deflections greater than 20 degrees for additional control. With the BT-9, for example, aileron effectiveness tests show that the aileron did very little except produce additional yaw after 20 degrees up aileron was reached. Similar results have been observed on several other machines. In application of formula, therefore, it would seem advisable to use a max of 30 degrees rather than 42 degrees. In the actual airplane this would allow a considerable reduction of stick force (by permitting increased mechanical advantage) and still allow ample margin over the minimum satisfactory value since the pb/2v obtainable would be about 0.11.

3. A report describing the tests and analysis used in setting up this criterion should be available in a few weeks.

B. TRAILING BOMBS.

Mr. F. L. Thompson has commented as follows:

1. There are attached one copy each of drawing D-5391 and D-6375, showing the NACA suspended air-speed head and the total head meter that is used with it when the suspended head is used only for determining the static pressure. For a description of the method used by the Committee in making air-speed measurements, reference is made to NACA Technical Note 616.

2. Such calibrations as have been made to date for the suspended head have been confined to relatively low speeds and show the error to be less than one percent of the dynamic pressure. A calibration to cover the entire range of speeds over which this head might possibly be used is to be made at an early date but is not available at the present time. It is not anticipated, however, that there will be any appreciable variations in the calibration except possibly at very high speeds.

I trust that this information will be of assistance to you in solving your problems.

Very truly yours,

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

G. W. Lewis
Director of Aeronautical Research.

Document 4-26(d), Excerpts from North American Aviation, Inc., Manufacturing Division, Engineering Department, Inglewood CA, "Aerodynamic Load Calculations for Model NA-73 Airplane,"* North American Report NA-5041, *March 1941.

NORTH AMERICAN AVIATION INC.
MANUFACTURING DIVISION
INGLEWOOD, CALIF.

ENGINEERING DEPARTMENT

AERODYNAMIC LOAD CALCULATIONS
FOR
MODEL NA-73 AIRPLANE

DESCRIPTION

The N.A.A. Model NA-73 Airplane is a single-place, single engine, low wing monoplane.

The wing is of all-metal, stressed-skin, stringer construction and is fully cantilevered. It consists of two sections, tapered in both planform and thickness, joined at the centerline of the airplane. The airfoil sections which are used are of the laminar-flow type and were developed at N.A.A. as explained in the text. A two-degree structural twist is incorporated in the wing varying from one degree positive incidence at the root section to one degree negative at the tip. Self-sealing fuel tanks are mounted in the wing structure.

The all-metal fuselage is divided into two separable units for ease of repair. Both sections are of semi-monocoque construction covered with alclad. Two built-up cantilever beams form the engine mount.

Simple metal-covered flaps extend outward from the sides of the fuselage to the inboard end of the ailerons. All metal ailerons of the partially sealed type extend from the outboard end of the flaps to the tip sections.

The fixed tail surfaces are of all metal construction, while the movable surfaces have aluminum alloy frames with fabric coverings. The movable surfaces are statically and dynamically balanced.

The landing gear is of the full cantilever half-fork type and is retractable inboard into wells in the wings.

The engine used with this plane is a 12 cylinder, Prestone cooled, Allison V-1710 with a present military rating of 1000 H.P. at 2800 R.P.M. at sea level. The radiator is located aft of the pilot's cockpit and is provided with an air scoop having an adjustable inlet and outlet.

AERODYNAMIC CHARACTERISTICS AND DISCUSSION

The airplane characteristics used in this report are based partially on wind tunnel data and partially on theoretical calculations.

The airfoil section at the tip and at spanwise Station 50 were derived by performing a series of pressure distribution calculations until, at the design lift coefficient, the negative pressure reached a maximum at or near the 50% chord point with no adverse pressure gradient ahead of this point. The calculations were based on the method developed in Ref. (c). This method, based on potential-flow theory, was modified in accordance with Ref. (d) in order to minimize the discrepancies between the theoretical and measured results. The remaining wing sections were developed by linearly varying the ordinates between the two known sections.

The spanwise distribution of section characteristics will be based on theoretical calculations whenever possible and estimated from reference data when no means of theoretical treatment is known.

SPANWISE DISTRIBUTION OF AIRFOIL CHARACTERISTICS

The forces on an airplane wing may be considered as functions of the characteristics of the airfoil sections. Certain of these characteristics depend only on the section shape and may be computed mathematically. These include the horizontal and vertical locations of the aerodynamic center with reference to the airfoil chord, the pitching moment about the aerodynamic center and the angle of zero lift. Pages 16 to 67 contain the computations for the above characteristics at seven spanwise sections. The procedure used is that suggested and outlined in Ref. (c).

Since the remaining characteristics do not lend themselves to theoretical treatment, wind tunnel data must be utilized. From a perusal of available published data, including that of Refs. i, j, k and l, values of C_{lopt}, C_{domin} and C_{lmax} are assigned to the seven sections. These values are based mainly on variations in thickness and camber. Care is taken that the distribution of C_{lmax} results in the proper maximum lift coefficient for the wing.

The lift-curve slope is assumed to be constant along the span. Its value is obtained from the wind tunnel data, App. I, Page Z, corrected to infinite aspect ratio.

$$a_o = a \, / \, (1-18.24/n) \, a \, (1+ \tau)$$

where a_o = Lift-curve slope for infinite aspect ratio

a = Lift-curve slope for finite aspect ratio

= .0742 (App. I, Page z)

n = aspect ratio

$= 5.815$ (P.10)

τ = correction for shape of span-loading

curve = 0.18 (Ref. k Page 53)

$$a_o = \frac{.0742}{(1-18.24/5.815) \times .0742\ (1+.18)}$$

$= .1022$

The characteristics obtained by the above methods are tabulated in the left hand portion of Table XXV, Page 71 and plotted on Page 70. From these spanwise distribution curves, the remainder of Table XXV, Page 71 is completed.

AIRPLANE COEFFICIENTS

The full-scale airplane aerodynamic coefficients used in this report are based on the results of wind tunnel tests on a 1/4th scale model of the NA-73 Airplane. The tests were made at the Guggenheim Aeronautics Laboratory, California Institute of Technology, and at the University of Washington Aeronautical Laboratory.

The results of the tests performed at GALCIT are used in this report with the exception of the maximum negative lift coefficient taken from the UWAL tests. All of the test curves from which data were taken are reproduced in App. I.

The Reynolds number of the wind tunnel tests was approximately 1.8×10^6 while the full-scale Reynolds number at H.A.A. is about 16.7×10^6.

The airplane coefficients are listed in Table XXVII and plotted and extrapolated on page 83. The Reynolds number extrapolation for the positive maximum lift coefficient, Page 79, follows the trend of a similar extrapolation for the N.A.C.A. 23012 airfoil presented in Ref. k, Page 117. The negative maximum lift coefficient is extrapolated in a similar manner using a slightly smaller C_{lmax}.

The airplane drag curves are extrapolated to agree with the full-scale maximum lift coefficients of the wing. Due to a lack of reference data concerning the extrapolation with Reynolds number of the minimum wing drag of laminar-flow airfoil, no extrapolation is performed. The wings of the airplane are smooth and, therefore, it is considered that the increase in drag coefficient for surface roughness is approximately .0001. An increase in the drag coefficient of the fuselage of .0039 is assumed. This is due to the effect of the surface roughness, carburetor scoop, radiator scoop, exhaust stacks and wing-fuselage interference as determined from wind tunnel investigation.

In order to obtain the pitching moments over the entire flight range, the aerodynamic center of the wing alone and of the wing-fuselage combination is first calculated on pages 81 and 82. The constant pitching moment around each of these centers is also determined and used to compute the required pitching moments around the wind tunnel model C.G. position.

Lift, drag and moment coefficients over the entire flight range are listed in Table XXVIII, Page 84. The lift and drag coefficients are resolved into components perpendicular and parallel to the thrust line, and, with the moment coefficients, are transferred to the aerodynamic center of the wing alone. A correction factor, arising from a difference in the model M.A.C. and the M.A.C. calculated for the full-scale airplane, is used to slightly reduce the wind tunnel moment coefficient values prior to their transfer.

CALCULATION OF CRITICAL SPEED

In the investigation of the effects of compressibility phenomena on the characteristics of an airplane, it is important to know the free air velocity at which the velocity of the air over the wing reaches the local speed of sound. This velocity, referred to as the "critical speed," varies along the span and with the angle of attack of the wing.

The maximum critical speed will occur in the vicinity of zero wing lift, however, in this report, it is assumed that the maximum critical speed occurs at minimum airplane drag. In order to obtain an average value along the span, the critical speed calculations are based on the airfoil section at the M.A.C. of the wing, Station 97.67.

The air pressure, at that point on an airfoil over which the air velocity has reached the speed of sound, will have the highest negative value that exists over the airfoil surface. Therefore, the pressure distribution at the M.A.C. section, (Y/ b/2 = .4375) is calculated at minimum airplane drag.

From the curves, Page 83, it is seen that minimum airplane drag occurs at a wing C_L of .100. The calculation of the section lift coefficient c_{lo} follows:

$c_{la1} = 1.060$ (P .75) $F_1 = .006$

(Ref. f, Page 15)

$c_{lb} = .0125$ (P .76) $F_2 = .002$

$\Delta c_L = F_1 \times F_2 = .006 \times .002 = .0001$

$c_L" = .1000 - .0001 = .0999$

$c_{la}" = .0999 \times 1.060 = .1059$ (Ref. f, Page 18)

$c_{lo} = .1059 + .0125 = .1184$

$= C"_{2a} + C_{2a} + C_{2b}$

From Page 14, the slope of the lift curve at wind tunnel speed is .1022. For use in the pressure distributions for critical speed, this slope is modified by the slope correction factors (P. 90) to agree with the high angle of attack speed.

$$a_o = .1022 \text{ X } (1.060/1.006) = .1077$$

$$\text{where } F_m \text{ at } 115.4 \text{ M.P.H.} = 1.006$$
$$F_m \text{ at } 260 \text{ M.P.H.} = 1.060$$

From the computations on page 86, it is evident that variations in the value of the slope of the lift curve has little effect on the value of the maximum negative pressure coefficient.

With the above basic data and the method detailed in Ref. d, the maximum negative pressure coefficient (P/q) is calculated on Page 86 to be -.5299. The corresponding Mach number, taken from Ref. b, Page 51, is .715. In a similar fashion, the critical Mach numbers over a range of angles of attack are determined in Table XXXI, Page 87 and plotted on Page 88.


North American Aviation, Inc.
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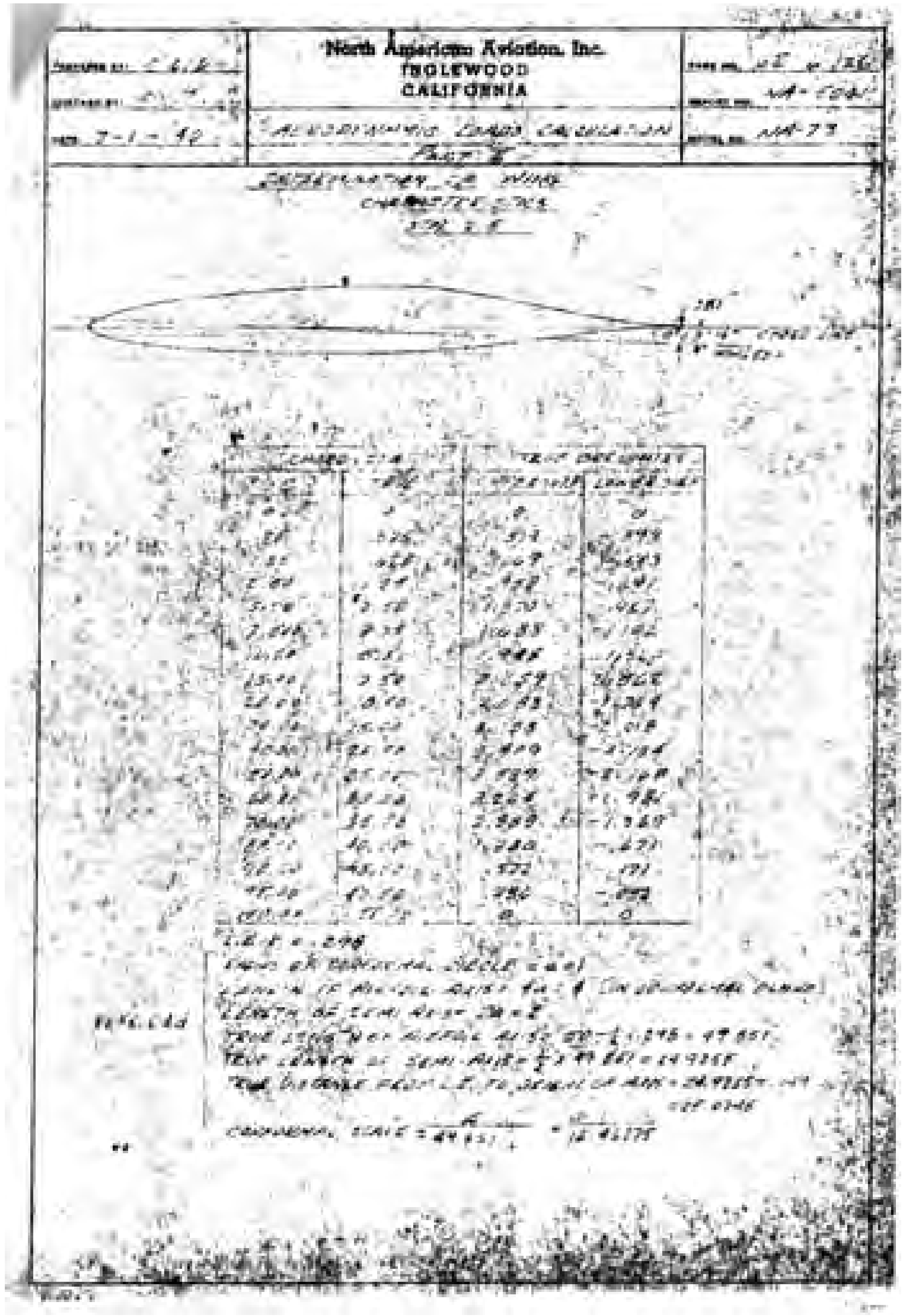
DETERMINATION OF WING CHARACTERISTICS


North American Aviation, Inc.
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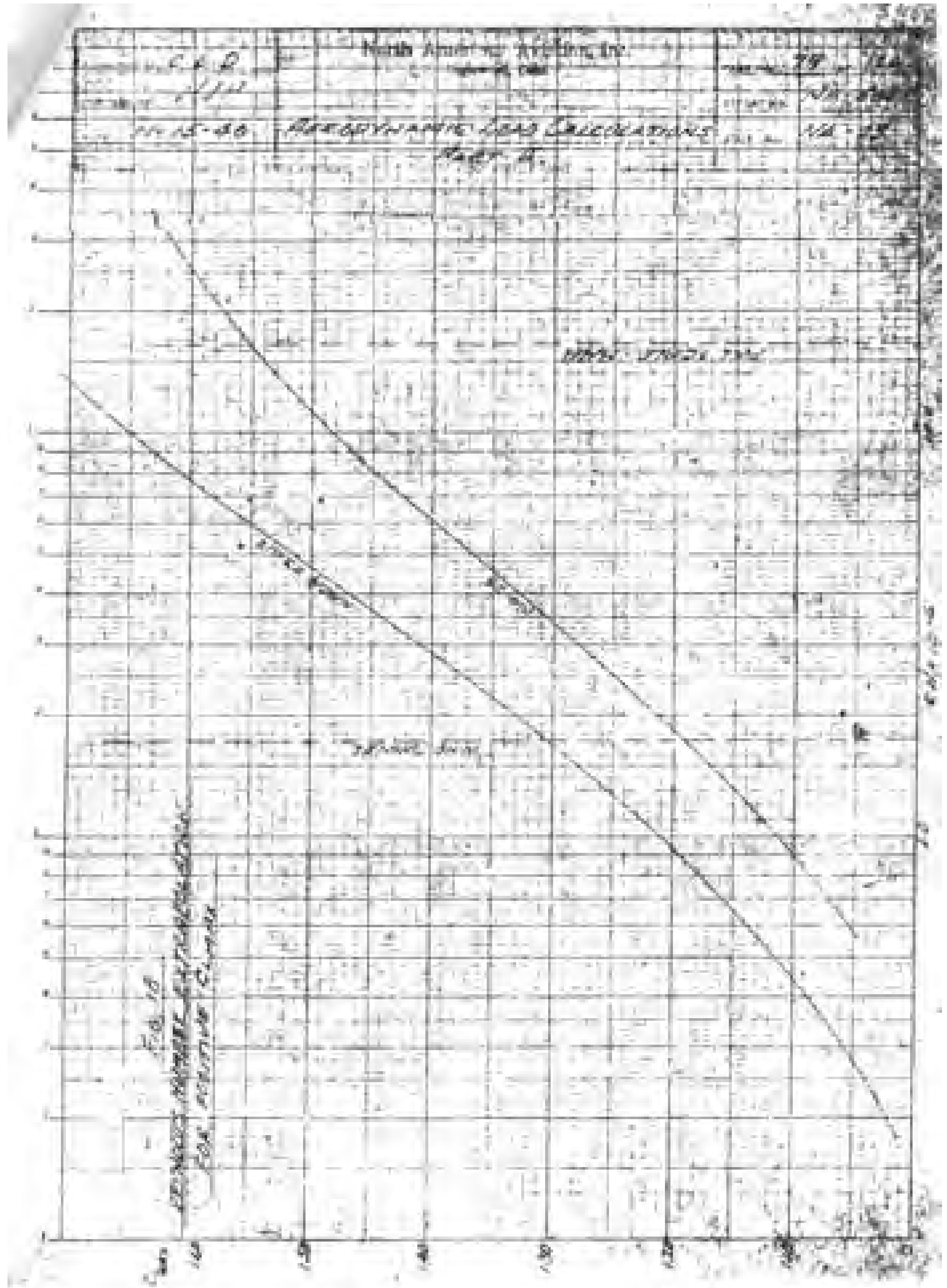
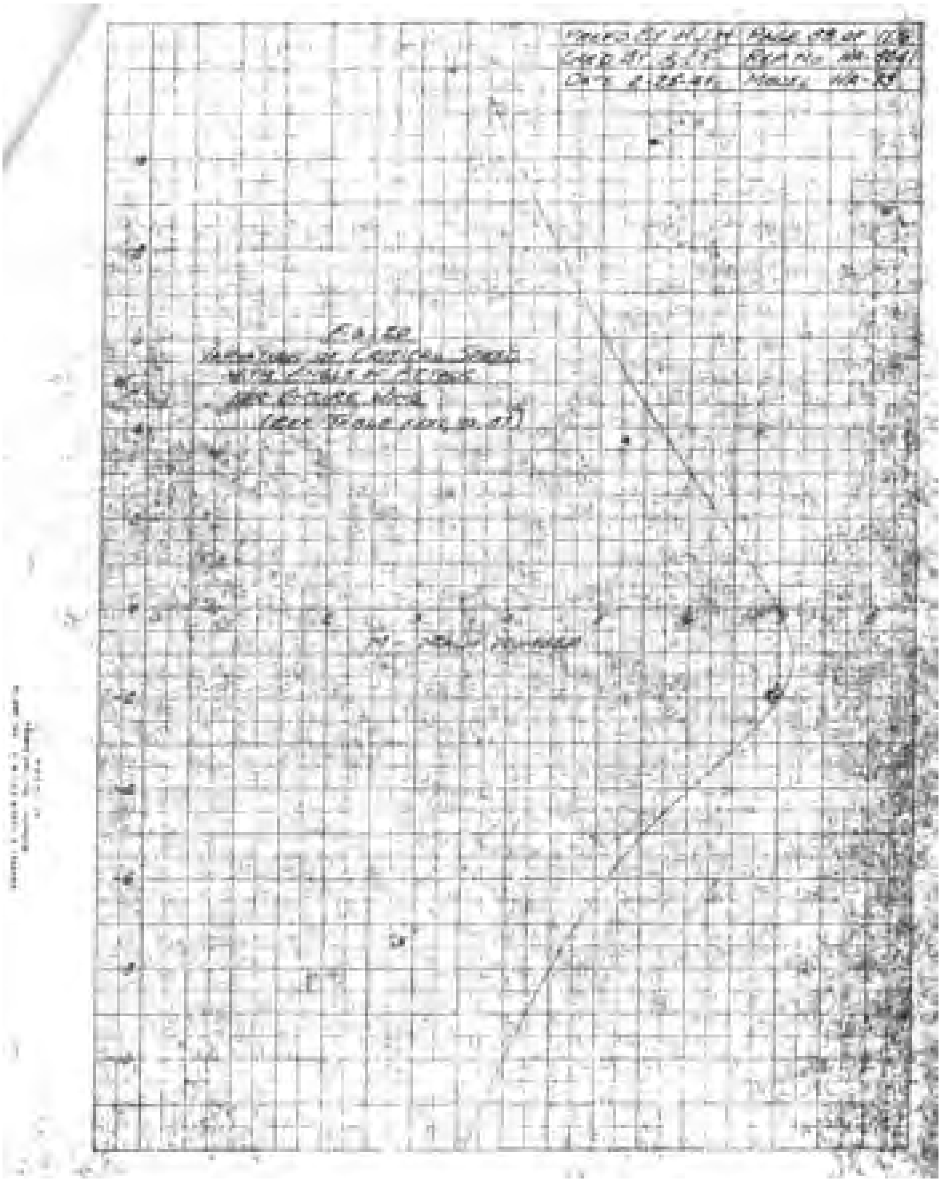
AERODYNAMIC LOADS CALCULATION
PART II


North American Aviation, Inc.
AERODYNAMIC LOAD CALCULATIONS
PART II



Document 4-27(a-d)

(a) Edward Warner, Civil Aeronautics Board, Washington, D.C., to Eastman N. Jacobs, Principal Aeronautical Engineer, NACA Langley, 12 October 1940, RA file 290, LHA, Hampton, Va.

(b) Eastman N. Jacobs, Principal Aeronautical Engineer, NACA Langley, to Dr. Edward Warner, NACA, Navy Building, Washington, D.C., 24 October 1940, Langley Correspondence Files, Code E38-8, National Archives, Mid-Atlantic Region, Philadelphia, Pa.

(c) Eastman N. Jacobs, Principal Aeronautical Engineer, to Dr. Edward P. Warner, Chairman, Committee on Aerodynamics, NACA, Navy Building, Washington, D.C., 27 November 1940, RA file 290.

(d) G.W. Lewis to LMAL, "Report by Mr. Jacobs on present status of laminar-flow wing development, for next meeting of the Aerodynamics Committee, to be held the latter part of January," 14 December 1940, RA file 290.

Additional insight into the NACA's early treatment of its laminar-flow airfoil development can be gained from a review of the following four documents. They refer mainly to a possible laminar-flow wing application for the Curtiss P-40D. In Document 27-B, Eastman Jacobs declared what can only be termed one of the grossest exaggerations of the war, that the laminar-flow application on the P-40 was "the most important single technical project in the United States today."

George Lewis's memo dated 14 December 1940 is interesting for its concern over circulating among the NACA Aerodynamics Committee proprietary information about a possible laminar-flow application by Curtiss for the P-40.

Document 4-27(a), Edward Warner, Civil Aeronautics Board, Washington, D.C., to Eastman N. Jacobs, Principal Aeronautical Engineer, NACA Langley, October 1940.

CIVIL AERONAUTICS BOARD
WASHINGTON

OCTOBER 12, 1940.

Mr. Eastman N. Jacobs,
Principal Aeronautical Engineer,
National Advisory Committee for Aeronautics,
Langley Field,
Hampton, Virginia.

Dear Mr. Jacobs:

I was very much interested in your letter of October 4^{th} about the supersonic tunnel and the possibilities of increasing the Mach Number at Langley Field. I take it from your longhand postscript that you have now had the desired talk with Dr. Dryden, and I am asking that he let us have his comments in such time and fashion as may be convenient for him.

I was also interested in what you said about the usefulness of committee discussions in connection with the development and application of the new airfoils. I hope that you will, at your convenience, let me have something more specific on that for possible distribution in advance of the next meeting of the committee. To make these meetings as useful as possible, we want to circulate a maximum of preliminary information in advance. To get the members fully informed in advance is the best possible preparation for a really useful discussion.

Sincerely,

Edward Warner

Document 4-27(b), Eastman N. Jacobs, Principal Aeronautical Engineer, NACA Langley, to Dr. Edward Warner, NACA, Navy Building, Washington, D.C., October 1940.

October 24, 1940

Dr. Edward Warner
National Advisory Committee For Aeronautics
Navy Building
Washington, D. C.

Dear Dr. Warner:

Your last letter of October 12, showing so much interest in the application of the new airfoils, and your interest in the usefulness of the committee in the matter of course pleased me very much. Perhaps I may be expected to be over-enthusiastic about the project, but I honestly think that the immediate application of our recent airfoil research in the form of a new wing for the Curtiss P-40D airplane is the most important single technical project in the entire United States today. This immediate application you will remember was suggested by Mr. T. P. Wright at the last meeting of your committee and I agreed that it was an excellent suggestion and that we should all work together on it.

The enclosed letter will indicate about what has happened since then. As far as I know, Curtiss are going ahead on the project with reasonable speed, although they had not yet gone far when I visited the Buffalo plant on September 30. I have wondered if it would be desirable to suggest that the Army representative on our committee check on all phases of the project and report to us from the Army's standpoint on the progress, at our next committee meeting. I can review the progress from the research standpoint.

I hoped to write you giving rather complete details, but so far have had trouble finding the necessary time. This brief note is being written before I leave on a short trip to discuss the supersonic tunnel blower with Allis-Chalmers in Milwaukee. On my return I will try to find time to prepare something that might be circulated to the committee members. In the meantime I would be glad to receive your suggestions.

Sincerely yours,

Eastman N. Jacobs
Principal Aeronautical Engineer

Document 4-27(c), Eastman N. Jacobs, Principal Aeronautical Engineer, to Dr. Edward P. Warner, Chairman, Committee on Aerodynamics, NACA, Navy Building, Washington, D.C., November 1940.

Langley Field, Virginia
November 27, 1940

Dr. Edward Warner
Chairman Committee on Aerodynamics
National Advisory Committee for Aeronautics
Navy Building
Washington, D. C.

Dear Dr. Warner:

In accordance with the suggestion in your letter of October 12, I have prepared the attached outline of the problems encountered in the use of the laminar-flow airfoils and the progress being made in solving these problems. I hope that you will find this draft of the subject suitable for circulation to the members of the Aerodynamics Committee, so that it can be brought up for discussion at the next meeting of the committee.

Very sincerely yours,

Eastman N. Jacobs
Principal Aeronautical Engineer

LOW-DRAG AIRFOILS

It has been established that airfoil drags may be greatly reduced through the use of wing sections of suitable shapes and with smooth and fair surfaces which permit the use of the low-drag properties of extensive laminar boundary layers. It has also been shown that the new laminar-flow sections may be designed to give results little different from conventional sections when used in a rough condition or outside their design range of low drag. There seems to be no reason, therefore, why they should not be employed at once on new types of military airplanes in place of the old conventional sections.

Actually, some applications on new military types are going forward through the cooperation of the services and certain manufacturers. In addition, the Committee has a project in the 19-foot pressure tunnel investigating a complete model of an air-cooled pursuit airplane with a laminar-flow wing, and a Navy fighter model with a wing of the new type substituted for the original wing. It would appear at first sight therefore, that the development and application of the new wings is progressing satisfactorily. On the other hand, many of the projects are in a preliminary form and much preliminary lay-out work and wind tunnel testing remain to be done before the actual wing design is even started. It may be years before flight test checks are obtained on some of these applications.

In the meantime, we need actual flight checks on the characteristics of the new wings, because the wind tunnel results are not complete and entirely reliable. Furthermore, the most important research remaining to be done concerning the laminar-flow wings has to do with their practical construction and maintenance under service conditions. These phases of the research and development will necessarily tend to become cooperative projects with the builders and operators and will finally pass almost entirely to them. If we had the necessary shop facilities, we might get on with this research more quickly by building and testing some practical wings here at the Committee's Laboratory, but for military types under existing conditions it seems advisable to pass at once to the cooperative phase of the research. At this point the Aerodynamics Committee seems to me to become an agency of vital importance. It should serve the important functions of guiding and coordinating the required activities of the N.A.C.A., the services, and the manufacturers. The committee has, in fact, started to function in this capacity.

You will remember that I mentioned at the last meeting on August 8 some of the developments in the laminar-flow airfoils and that Ted Wright suggested that thicker airfoils would be desirable on modern pursuit types in order to give internal space for more and larger armament. The discussion thus led to the conclusion that it would be desirable to investigate a wing of the new type as soon as possible for the Curtiss P-40-D airplane. Within the next few days a request came through Colonel Greene's Army Liaison Office here, from Curtiss, for some of the new and thicker airfoils for lay-out purposes on pursuit airplanes. We replied on August 13, giving them ordinates of the NACA 66, 2-018 airfoil, which we considered should be suitable for lay-out purposes and should give a drag coefficient of .0036 or less, below R = 20,000,000.

Following this, on August 21, Mr. Don Berlin visited the Army Liaison Office and I was called over to discuss possible new sections for the P-40-D airplane. After some discussion of the possibility of model tests in the 19-foot pressure tunnel, I suggested that we should all try to get together on some definite program. In discussing various possible programs, we finally agreed that tunnel tests would be unnecessary if, as the Army representatives (Colonel Greene and Mr. J. A. Roché) agreed, the maximum lift coefficient and stalling characteristics were not required to be known in advance. We all agreed, therefore, to cooperate toward the construction of a wing for flight tests as soon as possible. I suggested, however, that we might help by testing in the low-turbulence tunnel a wing sample to be built by Curtiss, to investigate the effects of construction imperfections, effects of camouflage paint, etc., although any such tests were not to hold up construction of the wing. Later, a suitable camber for the basic thickness form I had already sent to Curtiss was discussed. No final conclusion was reached, and it was agreed that we should look into the matter and make a definite recommendation later.

Fortunately, I had an opportunity of going over the question with Mr. T. P. Wright the next morning. After going over most of our more recent airfoil data, we agreed that a small amount of camber seemed desirable, and I made recommendations to the Curtiss Company accordingly on August 23.

In discussing with Mr. Wright the question of obtaining the desired surface conditions on the wings in practice, we used our surface curvatures gauge to check the surface conditions of a service P-40 against some commonly used wing surfaces, and found the P-40 wing, in some regions at least, to be of a different order of fairness. In fact, it appears that the construction methods now employed might not require drastic changes beyond the use of carefully applied butt joints between the wing cover plates. Furthermore, Mr. Wright informed me that additional improvements had already been made on the Curtiss-Wright transport, so that they were familiar with the problem of producing an improved wing surface.

Mr. Berlin wrote the Army Liaison Office on September 5 concerning some possible control surface difficulties, which I think we were able to clear up. In this letter he stated that they were proceeding with the design of a laminar-flow wing for the P-40-D airplane, in accordance with the information we had furnished them on August 21, stating "We expect to proceed with all possible effort in order to get a wing of this type on an airplane at the earliest possible date."

On September 30, I visited the Curtiss plant with the hope that I might aid in speeding up the project. I was much impressed by the P-40-D mock-up, and became more than ever convinced that this is the airplane on which to continue the investigations of the new type of wing. In fact, it seems to be the most interesting pursuit airplane available, from the standpoint of immediate development, although the production facilities have already been organized for it to such an extent that it may be too late to work in a wing change on their first production airplane. It is unfortunate, therefore, that we did not get together earlier on the program.

The greatest difficulty with the application of the new wing was the space for landing gear retraction. A slightly larger wing would have avoided the difficulty, but Mr. Berlin thought that it might be overcome.

The possibility of changing to a more conservative wing section near the fuselage, an NACA 65-series, for example, in order to avoid the danger of separation near the fuselage juncture, was also considered with them. Berlin, however, was not much afraid of this situation and considered the problem to be primarily one of filleting that could be worked out as an addition. We therefore agreed to go ahead with the original plan to carry the same section right into the fuselage.

The possibility was also discussed of building an experimental wing using makeshift methods. I concur in the stand Berlin takes on this possibility. "Final plans should be made on the supposition that the wing will work out as expected without

important changes; if successful, we are then much further advanced after the flight tests." I certainly favor such methods and strongly recommend that the Government support the project financially. Since that time I have picked up a little information about the progress of the project from time to time, and had a few minutes to discuss it here with Mr. Berlin within the past two weeks, although most of the day of his visit was spent in going over the details of nearly all our airfoil results with Mr. Child, who is in charge of the Aerodynamics Department at Curtiss. As a result of these discussions, it appears that the project had not moved forward much in the preceding month since my visit to Curtiss, and it appears that they have considered the application of the wing as applying now to a new pursuit airplane development. Whether or not this pursuit may be considered satisfactory will depend on possible delays involved. If the new pursuit type is like the others, so that it will require an extended program of wind tunnel investigations, the application of the new wing to it does not meet our requirements for a flight application on which we can continue our investigations within a reasonable time. On the other hand, if it represents only a development of the P-40-D which can go forward at once, it may be considered satisfactory.

It thus appears that these questions may now be considered to advantage by the Aerodynamics Committee. For that reason, the preceding outline of this work has been prepared for circulation to the members. It is hoped that the Army representatives in particular and Mr. T. P. Wright may find time to investigate the questions brought up and the present status of the project, so that they may report on and discuss our progress at the next meeting of the Committee.

Document 4-27(d), G.W. Lewis to LMAL, "Report by Mr. Jacobs on present status of laminar-flow wing development, for next meeting of the Aerodynamics Committee, to be held the latter part of January," December 1940.

Washington, D. C.
December 14, 1940.

From NACA
To LMAL

Subject: Report by Mr. Jacobs on present status of laminar-flow wing development, for next meeting of the Aerodynamics Committee, to be held the latter part of January.

1. Doctor Warner has taken up with me the memorandum report prepared by Mr. Jacobs, on low-drag wings, and submitted to Doctor Warner with a letter dated November 27. Doctor Warner requested my advice as to whether it would be desirable to send a copy of this memorandum to all the members of the Aerodynamics Committee.

2. I told Doctor Warner that I did not think it desirable to do so, as the subject of Mr. Jacobs' memorandum had to do largely with the application of a modified laminar-flow wing to the P-40 airplane, and with relationships between the Committee, the Army Liaison Office, and the Curtiss Company.

3. I suggested that Mr. Jacobs prepare a more general review of the subject for confidential circulation to the members of the committee in advance of the meeting of the Aerodynamics Committee which is to be held immediately following the annual meeting of the Institute of the Aeronautical Sciences in New York. It is requested that Mr. Jacobs prepare a general review of the present status of low-drag airfoils.

G. W. Lewis,
Director of Aeronautical Research.

Document 4-28(a-h)

(a) Vannevar Bush, Chairman, NACA, to Brigadier General C. L. Lindemann, Air Attache, British Embassy, Washington, D.C., 21 December 1940, RA file 290, LHA, Hampton, Va.

(b) G.W. Lewis, Director of Aeronautical Research, NACA, to Sir Henry Tizard, Chairman, Aeronautical Research Committee, c/o Director of Intelligence, Air Ministry, London, W.C. 2, England, 2 January 1941, RA file 290.

(c) H.T. Tizard, Ministry of Aircraft Production, Millbank, to Dr. George Lewis, Director of Research, NACA, 12 February 1941, RA file 290.

(d) G.W. Lewis to Tizard (via British Embassy for Diplomatic Pouch), 25 March 1941, RA file 290.

(e) Edward Warner, American Embassy, 1 Grosvenor Square, London, to Dr. George W. Lewis, NACA 1500 New Hampshire Ave., Washington, D.C., 25 August 1942, Langley Correspondence Files, NASA Record Group 255, Code E38-8, National Archives, Mid-Atlantic Region, Philadelphia, PA. (Copy also in Milton Ames Collection, Box 4, Files 51-2, Historical Archives, NASA Langley Research Center, Hampton, Va.).

(f) Eastman N. Jacobs, Principal Aeronautical Engineer, NACA Langley, to Engineer-in-Charge, "Low-drag airfoils in England," 22 September 1942, Langley Correspondence Files, RG 255, Code E38-8, National Archives, Mid-Atlantic Region, Philadelphia, Pa. (Copy also in Ames Collection, Box 4, Files 51-2, LHA, Hampton, Va.)

(g) Ivan H. Driggs, Aviation Design Research Branch, Bureau of Aeronautics, Navy Department, Washington, D.C., to Experiments and Developments Branch, BuAer, "British Views upon Airfoils for High Speed Aircraft," 16 August 1943, in Langley Correspondence Files, RG 255, Code A173-1, National Archives, Mid-Atlantic Region, Philadelphia, Pa.

(h) Ira H. Abbott, Senior Aeronautical Engineer, to Engineer-in-Charge, "British views upon airfoils for high-speed aircraft," 13 September 1943, in Langley Correspondence Files, RG 255, Code A173-1, National Archives, Mid-Atlantic Region, Philadelphia, Pa.

The longest string of documents in our volume up to this point tell an illuminating story of the critical British reaction to the NACA laminar-flow airfoils and the equally critical response of the NACA's airfoil experts to it. In essence, the British were quite skeptical of the American data because it derived from aerodynamically clean test models that could not possibly perform as well in actual operation. They felt, as Edward P. Warner reported back from London in August 1942 (28-e), that "the establishment of laminar flow can have only a relatively small effect," and that the outstanding performance of the P-51 Mustang, with which they were greatly impressed, had "little to do with" the laminar-flow wing.

The NACA airfoil experts thought the British thinking about laminar-flow airfoils was generally wrong. No one in the United States was saying that the new wings could perform as predicted unless the aircraft operators, i.e., the Allied air forces, learned to maintain the highly polished and clean wing surfaces that were required. Part of the reason for the mistaken British opinion also involved differences in how aerodynamic data was gathered and the fact that the British lacked the same low-turbulence equipment for testing.

On the other hand, it is obvious Warner was seriously concerned that the NACA laminar-flow results might in fact be too promising, and that skeptical British reaction was in fact on the mark. In his P.S. to George Lewis, he wrote, "Needless to say, I hope that the optimism on this subject which prevails in the United States will prove to be fully justified."

In key respects, the British analysis of the limited prospects of the laminar-flow wing proved quite correct, as Warner feared. R.A.E. engineers realized as early as 1942 what NACA engineers would not be willing to concede until later in the war: that the greatest advantage of so-called laminar-flow airfoils was not really low drag, as had been billed, but their excellent high-speed characteristics, which reduced compressibility problems to a minimum.

Document 4-28(a), Vannevar Bush, Chairman, NACA, to Brigadier General C. L. Lindemann, Air Attache, British Embassy, Washington, D.C., December 1940.

December 21, 1940.

Brigadier General C. L. Lindemann,
Air Attache,
British Embassy
Washington, D. C.

Dear General Lindemann:

Doctor Lewis informs me that the British Air Ministry would like to obtain such information as we have released on the so-called "laminar-flow airfoils."

The first confidential report released by the Committee was entitled "Preliminary Report on Laminar-Flow Airfoils and New Methods Adopted for Airfoil and Boundary-Layer Investigations." This is a general theoretical discussion with some preliminary results obtained in the low-turbulence wind tunnel at Langley Field. Subsequent to that, two reports have been released, entitled "Preliminary Investigations of Certain Laminar-Flow Airfoils for Application at High Speeds and Reynolds Numbers," and "Wind-Tunnel Investigation of the Lift Characteristics of an NACA 27-212 Airfoil Equipped with Two Types of Flap." The Committee has not released any reports on laminar-flow airfoils which contain the necessary information required by the aircraft designer. This is because we do not have a low-turbulence wind tunnel of a size adequate to investigate laminar-flow airfoils at high Reynolds Numbers.

To correct this situation and to furnish the best information that we have to American aircraft designers, we have invited to Langley Field representatives of the aircraft companies to discuss particular designs with members of our technical staff.

The Committee would be very pleased to have you send to Langley Field a representative of the British Air Ministry to discuss particular designs with the members of our staff.

Sincerely yours,

V. Bush,
Chairman.

Document 4-28(b), G.W. Lewis, Director of Aeronautical Research, NACA, to Sir Henry Tizard, Chairman, Aeronautical Research Committee, c/o Director of Intelligence, Air Ministry, London, W.C. 2, England, January 1941.

January 2, 1941.

Sir Henry T. Tizard,
Chairman, Aeronautical Research Committee,
c/o Director of Intelligence, Air Ministry,
London, W.C. 2, ENGLAND.

Dear Sir Henry:

For your information, I am attaching hereto a copy of a letter to General Lindemann, transmitting such information as we have prepared on the so-called "laminar-flow airfoils."

You will recall that when you were here I stated that we were working on the laminar-flow type airfoil but were not in a position to make a definite recommendation as a result of actual use of the airfoil in flight.

The situation has not changed. As yet we have not been able to have constructed a wing using the laminar-flow airfoil. A project for constructing such a wing for the Curtiss P-40 airplane has been under way for a few months, but very little progress has been made.

The nearest approach to the use of the laminar-flow airfoil is the wing that is used on the North American XP-51, designated by the British as the "Mustang". In cooperation with the Committee, the engineers of the North American Aviation Company have used a modified laminar-flow wing. The airplane has been flown, but recently crashed as a result of engine failure near the ground which necessitated a forced landing in a rough field.

I have talked with the pilots, and they have advised me that the stalling characteristics and the control characteristics of the airplane are very satisfactory. We do not have any indication as to the drag characteristics of the wing or as to the performance of the airplane with all-out power, as the engine was never operated at more than sixty per cent of its rated power before the crash.

With kind regards and best wishes for the coming year,

Sincerely yours,

G. W. Lewis,
Director of Aeronautical Research

Document 4-28(c), H.T. Tizard, Ministry of Aircraft Production, Millbank, to Dr. George Lewis, Director of Research, NACA, February 1941.

12th February 1941.

Dear Dr. Lewis,

I was much interested to have your letter of January 2nd, but sorry to hear that your Curtiss P.40 aeroplane with the new wings had made very little progress so far. We hope you will keep us informed about the progress of these experiments as we attach a great deal of value to them.

It may interest you to know that we have had an airfoil .20c thick with maximum thickness at .50 of the chord in flight, and have achieved transition points on the upper and lower surfaces as far back as .6 of the chord; so we feel that we have already made the first step from the laboratory stage into the field of practical design.

I am hoping that Mr. Relf, whom you know, will be able to pay a visit to America this Spring, and before he goes Dr. Darwin, the present Director of the National Physical Laboratory, will be going to Washington as the Head of our Scientific Mission there. Darwin is a member of the Aeronautical Research Committee, although he would not pretend to be an expert on aerodynamics. I am sure, however, that he will be anxious to get into touch with you.

Yours very sincerely,

H. T. Tizard

Dr. G. W. Lewis,
Director of Aeronautical Research,
National Advisory Committee for Aeronautics,
Navy Building,
Washington, D. C.

Document 4-28(d), G.W. Lewis to Tizard (via British Embassy for Diplomatic Pouch), March 1941.

March 25, 1941.

Sir Henry T. Tizard,
Ministry of Aircraft Production,
Millbank, S. W. 1,
London,
ENGLAND.

Dear Sir Henry:

Thank you very much for your letter of February 12, 1941. I am indeed very pleased and interested to learn that you have constructed and flown a type of low-drag airfoil .20c thick with maximum thickness at the 50 percent point. The results that you have obtained in achieving transition points on the upper surface as far back as .6 of the chord are most encouraging.

Unfortunately, the necessity of expediting the production program of present types of aircraft has made it impossible for us to obtain flight tests of the low-drag wing. The nearest we have come to it is, of course, the wing used on the North American "Mustang." I have talked to the pilots who have flown this airplane and they advise me that it has good flying characteristics and good stalling characteristics, and the high-speed performance of this airplane exceeds by some eleven miles the expected performance. All of the increase cannot, of course, be attributed to the low-drag wing. A part of it, no doubt, results from the clean design.

The delay in applying a low-drag wing to the P-40 was caused by the fact that an entirely new landing gear would have to be designed to eliminate the projection at the leading edge of the wing on the present P-40. The P-60 airplane, which is a modified P-40 using the low-drag wing, is now under construction and has been approved as a production type, although we have no flight tests.

Mr. Relf's proposed visit to America would be most helpful, and I shall be delighted to see him. I am pleased to learn that Doctor Darwin will be in Washington in the near future.

Mr. Taylor, who is working under the direction of Professor Parkin on the icing problem, together with two assistants, has been in Washington and spent a day at Langley Field. We are having a meeting of our Special Subcommittee on Deicing Problems about the 15th of April. Mr. Taylor will be invited to attend this meeting.

You will recall that we fitted the Lockheed 12 airplane with a heating system from the exhaust of the engine. This modification was completed some time ago, but unfortunately we have not been able to find any severe icing conditions on the West Coast. There is a great deal of interest in the use of heat as a means of deicing

the wings and possibly the tail surfaces. The Douglas company is at present designing a heating system using the Stewart Warner gasoline burners plus the heat from the engine oil radiators, with the idea of using this method of deicing on the DC-6 and the XA-26.

Any further information on the results you obtain in the flight tests with the low-drag wing will be greatly appreciated.

Sincerely yours,

G. W. Lewis
Director of Aeronautical Research

Document 4-28(e) Edward Warner, American Embassy, 1 Grosvenor Square, London, to Dr. George W. Lewis, NACA 1500 New Hampshire Ave., Washington, D.C., August 1942.

American Embassy
1, Grosvenor Sq.,
25 August 1942.

Dr. George W. Lewis,
National Advisory Committee of Aeronautics,
1500 New Hampshire Ave.,
Washington, D. C.

Dear George:

I have now had the opportunity of talking about laminar-flow wings at some length with Farren, Perring, Douglas, Stevens, and Squire, at the R.A.E., and have also discussed with them various designers in the industry. I am sure you are well informed on the general point of view of British aerodynamicists, but possibly I can add something to the information which you have previously received.

The general belief at the R.A.E. is that the establishment of laminar flow can have only a relatively small effect, and they are not satisfied with the direct measurement of drag in the wind tunnel in the case of the laminar-flow airfoils. They prefer the computation of drag, based on a direct study of the type of flow prevailing over the various portions of the wing, as being both more revealing as to the functioning of the wing and more accurate for practical use. The feeling continues, much as it was a year ago, that true laminar flow will in any case be restricted to a comparatively limited portion of the wing, and that the wings had better be designed in recognition of that fact; and that there cannot in any event be any maintenance of laminar flow in the slip-stream. I suppose there would be general agreement on the latter point.

Another point that occasions great anxiety at the R.A.E. is the effect of interferences, such as those of body-wing intersections, on the laminar-flow performance. Another is the determination of the maximum lift coefficient. They are very doubtful of the adequacy, or indeed the validity, of existing wind-tunnel techniques for this purpose.

In the specific case of the P-51, the performance of which has made a great impression, there is official conviction here that the laminar-flow wing has little to do with the performance. It was first remarked, when I discussed that airplane; "Well, it really hadn't much of a laminar-flow section;" that was subsequently modified to a suggestion that the wing design was such that laminar flow could only exist near the tips; and finally that a skin joint running parallel to the span was sufficient

to destroy any laminar flow beyond the first 25% of the chord. Mr. Squire, one of the younger men who seems to be specializing in this matter under the general direction of Douglas, had computed the effect of laminar flow on the P-51 performance as being 5 or 6 m.p.h. at most. The remainder of the good performance was attributed to a variety of other small points, including an exceptionally good surface finish and the position of the radiator.

One expression of the general position of the R.A.E. was that they felt relatively little interest in the laminar-flow section as such, but a great deal in compressibility; and that it fortunately happened that the airfoil form favorable to laminar flow was also one favorable to keeping compressibility to a minimum.

Designers here are not much excited about the idea. New airplanes are coming through without laminar-flow sections, and there is some disposition to feel that such sections will present their greatest benefit on some other fellow's airplane. One idea that I have encountered is that they are of substantial value only on machines for very long range, where most flights will be at very close to a fixed approximate angle of attack.

The Typhoon II is using a wing designed to have some laminar-flow characteristics, its maximum thickness being at 38% of the chord. The finish of the prototype is exceptional and makes even the P-51 (Mustang) seem crude by comparison. Several coats of pyroxylin lacquer are used, with a final high surface polish, and the construction is so smooth and the filling of cracks and depressions so well done that it looks like a plywood construction. Only the closest examination serves to show location of any of the rivets, or joints in the skin. I believe however that the wing, although freer from abrupt local discontinuity than the Mustang's, may not be quite so free from waviness. The exceptional smoothness of contour and the freedom from any ripples or distortions in the wing of the Mustang, helped by the thickness and the relatively generous support of the skin, are widely remarked. I think the Hawker people expect more gain in performance from the direct effect of a good surface finish, and from the fact that the wing thickness has been reduced by 20% as compared with that of the Typhon I, than from the change in airfoil section.

I was disturbed, some days after these conversations at the R.A.E., when I heard from Colonel Chidlaw, who accompanied General Echols here, that direct comparative tests of a laminar-flow wing and one of conventional section on the P-47 had shown very little benefit from the former. That would seem rather to confirm the British position. I hope nothing will interfere with Relf's trip to the United States, of the prospect of which you told me, as it ought to be most useful for Relf and Jacobs to get together and talk a lot of these matters out, and discuss the interpretation that is to be put on the data so far accumulated and on such as are yet to be secured.

Sincerely,

Ed Warner

P.S. I have tried in this letter to give a fair indication, by particular example, of the feeling that exists here with respect to laminar-flow wings. Needless to say, I hope that the optimism on this subject which prevails in the United States will prove to be fully justified. I think that in this particular case the British have rather resigned hope of large accomplishment before they have had full justification for doing so; but in saying that, I want to couple with it an expression of the greatest admiration for the general quality of the British research effort and the way in which it is being carried on in wartime. As was the case during my visit a year ago, I find the variety and the quality of the work being done at the R.A.E. and elsewhere most impressive.

Document 4-28(f) Eastman N. Jacobs, Principal Aeronautical Engineer, NACA Langley, to Engineer-in-Charge, "Low-drag airfoils in England," September 1942.

Langley Field, Virginia
September 22, 1942

MEMORANDUM For Engineer-in-Charge.

Subject: Low-drag airfoils in England.

Reference: Dr. E. P. Warner's let. To Dr. G. W. Lewis, Aug. 25, 1942.

1. It is unfortunate that the British are so pessimistic about realizing substantial gains through the use of extensive laminar flow on wings, and the situation is not improved by Colonel Chidlaw's premature comments on results obtained by the P-47. I had hoped that we could count on the British to show the Army how to maintain the desired wing surface conditions in service in case our military people fail in this important phase of the development. Sir Henry Tizard assured me that they would do so if we could give them airplanes showing substantial gains.

2. I will look forward to talking with Relf about our low-drag-airfoil work, but think nevertheless that I should go to England as soon as the situation here permits.

Eastman N. Jacobs
Principal Aeronautical Engineer

Document 4-28(g), Ivan H. Driggs, Aviation Design Research Branch, Bureau of Aeronautics, Navy Department, Washington, D.C., to Experiments and Developments Branch, BuAer, "British Views upon Airfoils for High Speed Aircraft," August 1943.

NAVY DEPARTMENT
Bureau of Aeronautics
Washington

16 August 1943

From: Aviation Design Research Branch.
To: Experiments and Developments Branch.

SUBJECT: British Views upon Airfoils for High Speed Aircraft.

1. During a recent trip to England an opportunity was afforded to discuss the subject problem with Captain Liptrot of the M.A.P., Dr. Goldstein and Mr. Relf of the N.P.L., Mr. Smelt of the R.A.E. and various engineers at Gloster Aircraft and de Havilland. With respect to compressibility all the people contacted were in complete accord but the Low Drag Airfoil as suggested by the N.A.C.A. meets with no uniform acceptance or approval.

2. All of the newer fighters and particularly those using jets are employing airfoils that have much lower thickness ratios than present American practice. The root sections are almost invariably 12 to 13% thick and the tip sections around 8 to 8 +%. An exception to this statement is the De Haviland "Ace" which uses a 15% airfoil at the wing root. The British are convinced that thickness ratio is the most important single variable affecting the critical Mach number of any airfoil; the shape, providing it is reasonably good, having less effect. The sections are being designed for as near a rectangular pressure distribution as possible at the design lift coefficient which is determined by the design speed of the airplane and the wing loading. Thus for any given C_L the least possible maximum negative pressure is obtained and consequently the highest critical Mach number. This principle is not the same as that used by the N.A.C.A. in the development of the L.D. sections, where the airfoil design is such that the negative pressure curve at first rises sharply and then at a less rapid rate to a point well back on the chord line, after which the pressure is recovered by a rapid decrease of negative pressure to the trailing edge. The sketches below illustrate the difference in principle between the two airfoil types.

3. Dr. Goldstein believes that the N.A.C.A. type airfoils have sacrificed something in critical Mach number in order to obtain more extensive laminar flow and a higher C_{Lmax} than he believes necessary. It is probable that the British type airfoil will have a slightly higher critical speed than the N.A.C.A. type at the same lift

coefficient due to a lower peak negative pressure. However, the positive slope to the pressure curve of the N.A.C.A. type helps to stabilize the otherwise unstable laminar boundary layer and therefore promote a lower profile drag at the design lift coefficient. It is probable that the British type airfoil will require much better surface conditions to obtain the same percentage of laminar boundary layer, but the British seem willing to accept this reduction to obtain higher critical speeds. Mr. Smelt was of the opinion that 35% laminar flow could be readily obtained in practice. Dr. Goldstein made the point that in the final analysis the operating and maintenance people will determine the amount of laminar flow that will be obtained on any wing, no matter what its shape or original condition when it left the manufacturer. He pointed out that accumulations of dried salt spray or hoar frost or the results of careless walking over the wing surface will probably upset the delicate balance in the unstable laminar boundary layer, resulting in performance that is inferior to that which would be obtained from an airfoil not so radically designed. Dr. Goldstein said that it might be that a change in airfoil section would be necessary along the span of a tractor airplane if it were desired to obtain the maximum possible L.D. effect, since an airfoil that would be most suitable outside the slipstream might be quite inferior when exposed to the turbulent air thrown back by the propeller.

4. Dr. Goldstein stated that he has developed means of calculating the airfoil shape required to produce any chosen pressure distribution curve. It was impossible to arrive at any understanding of his mathematical processes in one or two days so that no attempt was made to do so. Dr. Goldstein seemed most willing to cooperate but stated that it would take a trained mathematical physicist a number of months work at N.P.L. to become thoroughly familiar with all the processes. He offered to train any man that the Bureau of Aeronautics might desire to send to him to work in cooperation with his own staff for not less than 6 months. He also stated that he would further like to furnish the Bureau of Aeronautics with the ordinates of some of his airfoils for test in the new Carderrock tunnels if desired. He pointed out that as far as L.D. testing is concerned the tunnel turbulence must be very low. Not knowing just how far it was planned to reduce the turbulence in these tunnels no arrangements were made to obtain such ordinates for test while in England. Dr. Goldstein is of the opinion that the pressure distribution at high lift coefficients can be controlled to a reasonable extent so as to increase the critical Mach number of the wing during a high "g" turn at altitude. He believes this requirement to be of primary importance in maneuverability. A given airfoil may be greatly superior to another in giving a high C_{Lmax} when tested at low speed and would lead one to assume that greater maneuverability would result from its use. On the other hand the second airfoil might actually prove superior when executing a high speed turn at altitude since it might be designed to give a more favorable pressure distribution at the angle for C_{Lmax} and thereby be less affected by compressibility. Dr. Goldstein exhibited a drawing of the airfoil that he has designed for the new Spitfire (Griffon engine) on which he had tried out his theories. The section appeared perfectly normal with no concavity over the rear portion. The high speed tunnel tests had proven that a reduction in the drag of about 25% at a Mach number of .80 could be anticipated in comparison with the original Spitfire wing section. This airfoil is 12% thick.

5. Dr. Goldstein discussed items that he considered must be decided before the proper choice of an airfoil could be made for any airplane. These points are repeated below as given to the writer. It is to be noted that not all of the following list will apply to every airplane; for instance, a heavy low speed plane will not be concerned with compressibility.

1. The desired critical Mach number for wing and tail surfaces, taking interference into account.
2. The stowage room desired within the wing for guns, ammunition, fuel, landing gear and structure. This and point 1 largely determine the thickness ratios that can be used.
3. The lift coefficient at which the drag and the Mach number are to be the most favorable.
4. The wing geometry, taper ratio, aspect ratio, etc. This is required in order to control the position and extent of the stall.
5. The desired moment coefficient at zero lift coefficient in order to control the tail loads in a dive.
6. The desired C_{Lmax} and the flap type to be employed to obtain it.
7. The lateral control requirements as defined by the turning circle and time to bank.
8. The desirability of obtaining a small variation of the profile drag coefficient with C_L that is a high value of wing efficiency, e. This point is particularly important for the long range, slower airplanes.
9. The type of wing construction and the surface conditions the manufacture proposes. This point includes consideration of holes, cracks, doors and their fit slots and general surface fairness.
10. Whether the maintenance personnel are capable of keeping up the surface properly under operational conditions.
11. The tactical use of the airplane, whether fighter, intruder, bomber, etc.
12. The propeller location, whether pusher or tractor.

After Dr. Goldstein had enumerated the above items, it was very evident why he did not believe that airfoils should be "taken off the shelf" as he expressed it, but must be scientifically designed for each problem. This statement is probably true at this time but it would appear that as experience is obtained in the use of his methods families might be developed which would be suitable for various airplane types.

6. Although the point was not specifically discussed with Dr. Goldstein it appears to the writer that too much consideration is being given to airfoil section characteristics as measured in two dimensional flow and that not enough attention is being paid to what happens when the airfoil section is employed on a wing operating in a three dimensional flow. Extreme types of pressure distribution curves may be very unstable and degenerate into entirely different shapes under the action of strong lateral flows which may exist in actual practice. Present wind tunnel tests on N.A.C.A. type L.D. wings show such lateral flow at lift coefficients only moderately above the design C_L. It appears that there are many points that must be cleared up by thoughtful analysis and careful testing before all the airfoil theories can be accepted wholesale.

7. Dr. Goldstein was questioned in regard to two phenomena that had been observed in testing L.D. wings in the N.A.C.A. tunnels. The first was the extremely poor value of the wing efficiency, e, caused by an unusually rapid increase in C_D with C_L. He stated that their L.D. sections had been tested in two dimensional flow only, since they had no tunnels that gave sufficiently high R.N. at a turbulence factor low enough to make complete wing tests of any value. The second question concerned the discrepancy between the minimum drag of complete wings as determined on the wind tunnel balance and as found by the momentum survey method in the two dimensional tunnel and at the center of the span of a model <u>wing</u>. He stated that there should be complete agreement if <u>all</u> of the momentum changes were determined by the survey and integrated over the span. It was his belief that the survey should extend from tunnel wall to tunnel wall and should not be confined to one particular section of the flow aft the wing. In this connection, he pointed out that he did not agree with the N.A.C.A. with respect to the method used in determining the lift coefficient in the two dimensional tunnel. He stated that in order to determine the lift by measuring the pressure change on the tunnel walls, it was necessary to go to infinity in both directions and there might be considerable error in the extrapolation employed by the N.A.C.A. Dr. Goldstein prefers to measure the pressure distribution over the airfoil and integrate that. This latter method does appear to be the most direct and to serve another purpose as well, since pressure distribution curves are essential to the predication of critical Mach numbers from low speed tests.

8. Invariably the British feel that the P-51, Mustang is a remarkable airplane but there is no acknowledgment that its superior performance is due to a semi-low drag wing. Captain Liptrot of the M.A.P. stated that the drag of this airplane as delivered from the manufacturer is 50 lbs. at 100 ft./sec. while the Spitfire is about 61 lbs. at the same speed as received. He stated that when the latter airplane is faired up with tape over all cracks, etc., and the surface put into the same condition as that of the Mustang, the drag is reduced to the same figure of 50 lbs. It is his belief that the superior performance of the P-51 should be credited to the <u>Manufacturing</u> Division of North American for producing the airplane with no cracks or leaks, correct alignment of all doors, and cowlings, and in maintaining the proper surface conditions over the whole airplane in quantity production. It is his belief that a like treatment to any airplane which has a good basic aerodynamic form will produce like results. This appears to be the very realistic and eminently practical viewpoint of an experienced engineer.

Ivan H. Driggs

Document 4-28(h), Ira H. Abbott, Senior Aeronautical Engineer, to Engineer-in-Charge, "British views upon airfoils for high-speed aircraft," September 1943.

Langley Field, Virginia
September 13, 1943

MEMORANDUM For Engineer-in-Charge.

Subject: British views upon airfoils for high-speed aircraft.

Reference: NACA let. Aug. 27, 1943

1. The British views upon airfoils for high-speed aircraft as expressed in the memorandum by Mr. Ivan H. Driggs, enclosed with the reference letter (a), do not differ fundamentally in any serious way from our views. Several details of Mr. Driggs' memorandum, however, differ sharply from our views and deserve comment. It is particularly surprising that the views expressed differ in some details so greatly from the views expressed by Dr. Goldstein at the time of his last visit to this laboratory.

2. It is unfortunate that comparisons should be emphasized between British wings designed for extremely high-speed aircraft and NACA low-drag airfoils, which have been designed for aircraft operating at lower speeds. Our work on airfoils has been largely confined to such airfoils as are of immediate interest to the Army and Navy, and most such airfoils have been developed for application to airplanes with designed for high speeds not over about 450 miles per hour. Neither the Army nor the Navy has shown interest in airfoils for extremely high-speed airplanes. NACA low-drag airfoils developed for such airplanes would be very similar to the British airfoils although they might differ in detail.

3. It is our belief that the design principles of NACA low-drag airfoils are con-

sistent with high speed. It is true, of course, that if nothing else except freedom from compressibility at extreme speeds is desired, slightly higher critical speeds may be obtained by large sacrifices in other desirable qualities. We agree that thickness ratio is the most important single variable affecting the critical Mach number of any airfoil. Airfoils for extremely high speeds must, accordingly, be thin. The principle of designing for as nearly a rectangular load and pressure distribution as practical has been in use for a long time in the design of NACA low-drag airfoils. It should be pointed out that carrying this principle to an extreme, as apparently recommended in the memorandum by Mr. Driggs, results not only in inability to maintain extensive laminar flow and in a vanishingly small low-drag range, but also in a vanishingly small range of lift coefficients where high critical Mach numbers are obtainable. The low-drag range is also the range of high critical Mach numbers. The type of pressure distribution advocated by Mr. Driggs would peak near the leading edge at only small departures from the design lift coefficient, with serious loss in critical speed. We question the practicability of an airfoil design which would allow high critical speeds over such a small range of lift coefficients. Incidentally, the sketch showing the type of pressure distribution of NACA low-drag airfoils is not typical.

4. We have recently undertaken work to increase the critical Mach numbers of NACA low-drag airfoils beyond the point possible with rectangular load distributions. This work shows considerable promise, although it is not yet evident to what extent other desirable properties of the airfoils must be sacrificed to obtain appreciably higher critical Mach numbers for a given thickness of airfoil.

5. We agree that maintenance will determine the amount of laminar flow obtainable no matter what condition the wing was in when it left the manufacturer. With regard to the effects of roughness, the airfoil sections recommended by us will not have inferior performance to conventional sections when equally rough.

6. With regard to methods of deriving airfoils, it appears from what we know that Dr. Goldstein is rapidly approaching our methods. Exactly what method is now used by Dr. Goldstein is not known, but available information indicates he is following a line of development of processes very similar to ours.

7. We appreciate the importance of three-dimensional flow but feel that such effects have sometimes been exaggerated. We cannot escape the fact that good two-dimensional test equipment is in existence and has amply demonstrated its worth. Low-turbulence three-dimensional test equipment is not yet available. Pending the time such equipment is available, existing equipment should be used to its best advantage.

8. The determination of wing efficiency factors with low-drag airfoils has caused considerable trouble, principally because the fundamental concept is erroneous. Any airfoil with a low-drag range will show a more rapid rise of drag outside the low-drag range than will a conventional airfoil, even though the actual drag at any particular lift coefficient be no greater. It would be more nearly fair to say that the low-drag airfoil shows a sharper reduction in drag within the low-drag range. The picture has been further complicated by some tests of wings with poorly chosen sections and by tests made at too low values of the Reynolds number. Rapid scale effects on drag occur outside the low-drag range at all Reynolds numbers obtainable in any three-dimensional-flow tunnel.

9. We agree that drag values obtained by balances and wake surveys should check. Such wake surveys have been made on numerous occasions over the whole span, as suggested by Dr. Goldstein. It is my belief that the discrepancies shown by these measurements, when present, may be explained on the basis of inadequate corrections to the balance results.

10. Surprise is felt that Dr. Goldstein does not agree with us with respect to our methods of measuring lift in the two-dimensional tunnels. Dr. Goldstein went over this method with us and expressed satisfaction with it. Considerable error cannot be present in our method for the reasons stated because the so-called “extrapolation” amounts to only a few percent in most cases and is made as accurately as perfect fluid theory permits. In addition, several checks of the method against balance measurements and lifts obtained from integration of pressure distributions over the model have been made with excellent results. It should also be noted that pressure-distribution orifices in the model interfere with the flow over the model producing, in some cases, erroneous measurements.

11. It is agreed that excellent performance of any airplane will be obtained only by attention to detail in every particular. Any item of an airplane can ruin its performance, while no one item by itself can make a good airplane. It is considered highly improper to pervert this argument into one for accepting a mediocre treatment for one part of the airplane because of the effects of other details.

12. In view of this memorandum, it is thought that a visit by Mr. Driggs to the Laboratory to discuss airfoil problems would be mutually advantageous.

Ira H. Abbott
Senior Aeronautical Engineer

Document 4-29

Edwin P. Hartman, NACA West Coast Coordinating Office, to Chief of Research Coordination, NACA, "Information from the industry on the application of low-drag airfoils," CONFIDENTIAL, 29 July 1944, Langley Correspondence Files, RG 255, Code A173-1, National Archives, Mid-Atlantic Region, Philadelphia, PA.

By the summer of 1944, U.S. aircraft manufacturers had made enough different applications that it was hard for the NACA to keep up with them. In this memorandum Edwin P. Hartman of the NACA's Western Coordinating Office in Santa Monica, California, compiled a list of the airplanes designed on the West Coast to which NACA low-drag airfoils had been applied. The list turned out to be 27 airplanes long, with 13 of the airplanes currently flying.

But from the historical perspective this memorandum contributes much more than just a handy digest of World War II laminar-flow applications. The last part of Hartman's memo, starting with Paragraph 11, offers the most direct, introspective, and critical commentary on the U.S. aircraft's industry's judgment of the laminar-flow research than has ever been put into the historical record. To summarize, Hartman reports that airplane designers in industry by mid-1944, at Douglas Aircraft Company in particular (Hartman worked from a Santa Monica office and thus dealt with Douglas engineers on a regular basis), felt that the laminar-flow airfoils had "not performed as might have been expected from the NACA data." "It has been a costly experiment," Douglas engineers had told him, "not only for the companies but also for the nation." Reading Hartman's report, one is reminded of the British skepticism. expressed more than two years earlier.

Document 4-29, Edwin P. Hartman, NACA West Coast Coordinating Office, to Chief of Research Coordination, NACA, "Information from the industry on the application of low-drag airfoils," CONFIDENTIAL, July 1944.

Santa Monica, California
July 29, 1944

MEMORANDUM For Chief of Research Coordination

Subject: Information from the industry on the application of low-drag airfoils.

1. Following your suggestions, I have taken preliminary steps to obtain much

available, reliable data as the industry has regarding the application of low-drag airfoils. Inasmuch as almost all the low-drag airfoils used by the west coast aircraft companies have been tested in the Committee's wind tunnels, either as individual sections or as parts of complete models or airplanes, it is presumed that the committee has all necessary data concerning the designations and profiles of these sections. It is also assumed that the Committee is in a position to obtain drawings showing the actual wing construction of these airplanes from the Army and Navy. Indeed, it is gathered from your memorandum that the Committee wants only the small amount of reliable airfoil test data known to have been obtained in flight by the North American and the Douglas Santa Monica aircraft companies and any similar data obtained by other companies.

2. To the best of my knowledge, the following is a complete list of the airplanes designed on the west coast to which NACA low-drag airfoils or some modification thereof have been applied.

Airplane	Now Flying	Airplane	Now Flying
1. B-29	x	15. BTD-1	x
2. XPBB-1 (?)	x	16. XB-35	
3. XF8B-1		17. N9M	x
4. XP-58	(AAL w-t tests)	18. XP-54	x
5. XP-80	x	19. XA-41	x
6. XR60-1		20. XP-81	
7. XB-42	x	21. XB-36 (C-99)	
8. XB-43		22. XPB5Y-1 (?)	
9. XC-74		23. D-2	x
10. P-51A, B	x	24. HK-1	
11. XP-51F, G	x	25. XF-11	
12. P-51H		26. XFR-1	x
13. P-82		27. XBDR-1	
14. A-26	x		

3. For all of these airplanes that are now flying, there should be some information, either quantitative or qualitative, indicating the success of the airfoil application. I believe, however, that only in the cases of the P-51A and the KP-54 have any quantitative measurements of section drag been made. In the case of the XC-74, the

drag characteristics of the chosen section were obtained in flight by extensive tests of a glove mounted on an A-17 airplane. Tests of gloves representing sections for the XR60-1 and also, I believe, the XP-80 (one on each wing of a P-38 airplane) are soon to be started by Lockheed.

4. In the procurement of the available flight-test drag data on low-drag airfoils, requests have been made to:

(1) North American for data on the P-51A and P-51F airplanes. (Some data on C_{Lmax} for the P-51F (or G) will be available in about two weeks.) The North American data on the P-51A are already in the hands of the NACA, and IMAL has of course obtained extensive additional data on the airplane during recent flight tests.

(2) Consolidated Vultee for section momentum drag data on the XP-54.

(3) Douglas Santa Monica for data on tests of the XC-74 glove on an A-17 airplane. Dr. Klein, to whom I was referred for these data, said that, because of certain peculiarities in their organization, he was not sure that the data could be made available to the Committee. He will, however, take the matter up with Mr. Raymond.

5. The just-mentioned data requested of the aircraft companies are, I gather, all that the Committee wants me to obtain. I will, however, take steps to obtain any further qualitative or quantitative information concerning the other low-drag-airfoil applications that the Committee may suggest.

6. Examination of the rather vague suggestions made by Aerodynamics Committee members, which apparently led to the NACA's decision to prepare a summary report, indicated that further investigation into the matter would be desirable. Therefore, in order to obtain a clearer picture of the industry's wishes with regard to the content of the report, I discussed the matter with Mr. L. E. Root, of Douglas, who apparently was the Aerodynamics Committee member who originated the suggestion of writing the summary report. The general subject of low-drag airfoils and the summary report was subsequently discussed with Messrs. L. L. Waite and E. J. Horkey, of North American; Drs. W. B. Oswald and F. Clauser, of the Douglas Aircraft Company, Inc.; and Mr. Philip Colman, of the Lockheed Aircraft Corporation. The following paragraphs present the ideas and suggestions obtained from the discussions with these men.

7. MR. ROOT: Concerning his suggestion that the Committee write a summary report on low-drag airfoils, Mr. Root said he had actually had in mind the writing of either two or three reports: the first one, a collection of existing NACA wind-tunnel and flight-test data with a rationalized explanation of their use in future low-drag-airfoil applications; the second, a collection of qualitative and quantitative information from the industry concerning all the details of their experience in applying low-drag airfoils to their airplanes; and the third, a report showing how present available data could be used to obviate the troubles that the companies have encountered and providing the necessary additional material required for the successful application of the airfoils.

8. The data for the second report could, he said, be obtained by asking the aircraft designers such questions as:

a) What factors entered into the choice of the section used and what were the reasons for selecting a low-drag section?

(b) Did data from wind-tunnel tests confirm the basis for selection?

(c) What steps were taken to acquaint engineering and production groups in the plant with the problems of low-drag-airfoil application?

(d) Did flight-test data confirm the basis for the airfoil selection?

(e) Has the airfoil selected affected the aileron performance?

(f) Has the airfoil selected affected the stability and control of the airplane?

(g) Did any weight penalty result from the use of the section?

(h) Has the maximum lift coefficient obtained in flight been as much as expected at the time the airfoil was selected?

9. The Douglas aerodynamics group is under high pressure at the present time because of weaknesses displayed by the BTD-1 and the A-26 airplanes and is therefore inclined to be critical of factors (low-drag airfoils) which, they think, may have contributed to these weaknesses. I gathered that neither the BTD nor the A-26 developed the maximum lift coefficient that was expected when the airfoils were selected, and it is apparent from the performance of the airplanes that extensive laminar flow is obtained in neither case. I believe that Mr. Root also attributes certain stability difficulties of the BTD to the airfoil selection.

10. Mr. Root stated that, as long as Douglas El Segundo continues to build Navy airplanes, they will probably never again make use of low-drag airfoils, at least not until data are forthcoming from the NACA, or elsewhere, proving that a definite gain can be obtained with these airfoils and showing how a successful practical application can be made. Douglas's future use of conventional airfoils will be determined not only by the fact that the Navy believes low-drag sections to have no advantage over NACA 2400 series sections at high speeds and to have serious disadvantages at low speeds but also to the fact that the aerodynamics group has no evidence, of its own or from other companies, with which to prove to the Douglas engineering management that low-drag airfoils are superior or even equal to conventional airfoils.

11. Douglas engineers appreciate that the NACA cannot be expected to come out and build their wings for them and that the industry must be willing to take some chances in trying new things in order to improve the efficiency of its aircraft. They feel, however, that the industry has been willing to take these chances, as evidenced by the 20 or more low-drag-airfoil applications on west-coast airplanes, and that, on the whole, the airfoils so applied have not performed as might have been expected from the NACA data on those airfoils.

12. It has been a costly experiment, Douglas engineers think, not only for the companies but also for the nation. They feel that it is high time for a critical examination of all the applications to be made to find out why the failures have occurred and why, in none of the applications that Douglas engineers know of, has extensive laminar flow been obtained.

13. Mr. Root said that the industry had been encouraged directly by the NACA, as well as indirectly through recommendations made by the NACA to the services, to use low-drag airfoils. He feels, therefore, that the NACA has some responsibility in the success or failure of the application of low-drag airfoils to aircraft. He said he thought it would be to the interest of the NACA to do everything possible to ensure that its development of low-drag airfoils did not collapse because of unsuccessful applications which it might have helped prevent by taking a closer interest in the applications.

14. Douglas aerodynamics men had planned to measure the section drag of the A-26 and the SB2D-1 wings but were prevented from doing so by organizational difficulties. They feel that only the NACA is in a position to obtain wing-section drag data on airplanes on which low-drag-airfoil applications have been made. Measurements that the Committee could make on these airplanes would, Douglas engineers feel, be of tremendous assistance in future applications of low-drag airfoils.

15. Mr. Root pointed out that the success of a Navy airplane depended very greatly on its performance at low speeds. In this connection, he said he thought the designers of aircraft did not appreciate the dangers of obtaining low maximum lift coefficients through the use of low-drag airfoils. He also thought that the thicker wing-tip sections recommended for low-drag airfoils would have a distinctly deleterious effect upon lateral control, so important in Navy airplanes.

16. Douglas engineers believe that it would be highly desirable to make comparative tests on an airplane equipped, in the first instance, with a conventional, say NACA 2400 series, airfoil and, in the second instance, with a low-drag airfoil. If the airplane with the low-drag airfoil showed outstandingly improved performance, it would provide tremendous encouragement for designers to use low-drag sections in the future.

17. According to Douglas engineers, designers are now in a questioning, skeptical frame of mind. Many applications of low-drag airfoils have been made but there is no definite evidence that the performance of these airplanes is improved by the use of low-drag airfoils or is even as good as if conventional sections had been used. Considering the millions and millions of dollars that are going into these airplanes, Douglas engineers feel that it would be well worthwhile to make some experiments

comparing conventional and low-drag airfoils on different types of airplanes.

18. Mr. Root pointed out that the rush of the construction of a new military airplane was such that the company just did not have the time to make the necessary airfoil tests to properly select a low-drag airfoil for the airplane and thus was forced either to use a conventional section with known characteristics or to blindly select a low-drag airfoil from the far too meager supply of low-drag-airfoil data that are available.

19. Another handicap, Mr. Root said, was that in order to make a successful low-drag-airfoil application, it was necessary to reeducate engineering, tooling, and production groups throughout the whole plant in the problems associated with the application. Even after that was done, he said, in the rush of production and with the poor type of labor available, it was almost impossible to build airplanes with sufficiently smooth surfaces to obtain extensive laminar flow. It was his opinion, Mr. Root said, that the most rearward limit of laminar flow on a low-drag airfoil was the location of the first wing-skin joint, usually at the front spar.

20. NORTH AMERICAN COMMENTS: Messrs. Waite and Horkey agreed that a report collecting and rationalizing existing NACA wind-tunnel and flight-test data on low-drag airfoils would be helpful to designers, but they saw no use in the Committee's preparing a report of the type suggested by Mr. Root describing the experiences of other companies in the application of low-drag airfoils.

21. Although North American engineers are not too hopeful of obtaining extensive laminar flow on any of their low-drag-airfoil applications, they do feel that their low-drag-airfoil application in the case of the P-51B and P-51F has been successful. Their feelings in this matter are apparently based on the generally good performance of the P-51B and F airplanes, on the absence of serious compressibility difficulties, and on the good stalling characteristics and fairly low stalling speed of the airplane.

22. Mr. Waite said he would prefer to have the NACA go ahead and get fundamental data on the low-drag sections and leave the building of the wings to them. Mr. Horkey said he thought that all aircraft companies would like to have the NACA run tests on families of airfoils to fill in the vast gaps that exist at present in airfoil data. Despairing of getting such data from the NACA, aircraft companies were reported to be planning to obtain the data from tests in their own or other available wind tunnels. I gathered that North American engineers would also like to see comparison tests run on an airplane equipped alternately with a conventional wing and with a low-drag wing.

23. North American engineers said they would furnish the Committee with all the data they had from their tests of the wings on P-51 airplanes; however, it was their belief that the Committee already has all the reports they have prepared up to the present time. Mr. Horkey said, however, that they would supply the Committee with the results of some maximum-lift-coefficient determinations which they were making on the P-51F airplane and which should be available in a couple of weeks.

24. Concerning the optimum location for the wing-skin joint, Mr. Horkey stated that they had tried three different locations from the stagnation point aft and did not know which was best.

25. Mr. Waite stated that high-speed airplanes were being designed in this country having wing sections only 10 percent thick. He said he was afraid the companies would find themselves in trouble with airfoils of this thickness although there seemed to be no way of avoiding shock-wave formation with any thicker sections. Some other solution to the airfoil problem for high-speed airplanes is clearly needed, North American engineers believe. They also feel that the Committee should be developing airfoils for efficient-as-possible operation in the supercritical and supersonic realms of flight.

26. DOUGLAS SANTA MONICA COMMENTS: Dr. Oswald thought the summary report the Committee plans to write on low-drag airfoils would be of some help, but neither he nor Dr. Clauser thought that any report covering the experiences of the industry would be of much benefit or a feasible project for the NACA. Dr. Oswald believes, however, that the NACA should make a careful examination of all the airplanes with a low-drag-airfoil application to find out why none of them seem to be getting extensive laminar flow and to obtain some useful information that would be helpful to the industry in future applications and to the NACA in planning further research on low-drag airfoils. He feels that the NACA is in the best position to obtain and correlate flight-test data on airplanes with low-drag airfoils and that this should be an important part of the Committee's airfoil-development program.

27. Dr. Oswald pointed out that not nearly sufficient data exist to enable a designer to make a rational selection of a low-drag airfoil. The only alternative is for the company to run an extensive test program of its own for which there was little time available and which was difficult to sell to the company management. The real need of the industry, Dr. Oswald stated, is for some new airfoils that will have a much better chance than present ones of achieving the favorable characteristics which NACA test data credit to present low-drag airfoils.

28. Millions of dollars have been spent in low-drag-airfoil applications which apparently have been unsuccessful, Dr. Oswald said, and he thought it was time for something to be done about it. The hands of the Douglas aerodynamics groups are tied in this matter. They would like to examine their own low-drag-airfoil applications to find out what was wrong with them but, because of organizational difficulties, they are prevented from doing so. The help of the NACA is therefore needed and can best be supplied by careful examination and tests of existing applications and through further research to provide the necessary data for a successful practical application of these airfoils.

29. Dr. Clauser felt that new data were needed rather than a rehash of the existing data already available to the industry; he also thought that a collection of the opinions of the various companies with regard to the application of low-drag airfoils

would be of no great value to Douglas. Dr. Clauser emphasized the need for fundamental research on airfoils rather than just running tests on families of airfoils.

30. The suggestions that Drs. Oswald and Clauser made for needed data are included in the list at the end of this memorandum, which presents the suggestions for needed research obtained during my discussions with various men in the industry.

31. I inquired of Dr. Clauser concerning the availability of the data they obtained in the A-17 glove tests; he referred me to Dr. A. L. Klein, who was said to assist Mr. Raymond in matters of that kind.

32. LOCKHEED COMMENTS: My conversation on the low-drag-airfoil problem with Mr. Colman was very limited. Mr. Colman said he thought the survey report that the Committee was planning to write on low-drag airfoils would be helpful, but he also said he thought the opinions and experiences of other companies regarding the application of low-drag airfoils would neither be of much help to them nor make a suitable subject for an NACA report. He said that, although Lockheed had been slow in adopting low-drag airfoils, they had now been converted to the use of these airfoils and were very much pleased with the results of the application made on their pursuit airplane. He stated that, although they had no definite measurements indicating the extent of the laminar flow obtained on the wing of this airplane, they thought they were getting extensive laminar flow because, when tufts were added to the wing surface, a notable decrease in speed was observed. Mr. Colman made a suggestion for needed research work in the field of low-drag airfoils that has been put in the following list of suggestions received from various men in aircraft companies.

33. SUGGESTIONS FOR FURTHER RESEARCH ON LOW-DRAG AIRFOILS:

(1) The relative effects of waves and roughness at various points on the surface of the wings with different pressure gradients. In order to make the study fundamental, Dr. Clauser suggests that it be a study of the relation between waves and pressure gradients on the airfoil as affecting laminar flow and separation. Permissible waviness as a function of pressure gradient and boundary-layer thickness.

(2) Effect of different kinds of paint on laminar flow. Dr. Clauser suggests that this be defined as a study of micro-roughness on airfoils. Permissible roughness (size and shape) as a function of pressure gradient and boundary-layer thickness.

(3) More data at high Reynolds and Mach numbers. Lockheed particularly is interested in data at high Reynolds numbers. They feel that such data can best be obtained in flight.

(4) Data for complete families of low-drag airfoils.

(5) Effects of different locations of wing-skin joints in practical applications. Aluminum sheets are made only in certain widths, and these widths are sufficiently small that skin joints must be incorporated fairly close to the wing leading edge on both upper and lower surfaces.

(6) Data for choice of optimum nose radius for compressibility and C_{Lmax}.

(7) Determination of turbulent separation parameters.

(8) Effect of wing configuration on the successful application of low-drag sections.

(9) Fundamental study of airfoils to obtain a section that will provide high lift. This suggestion was made by Dr. Clauser, who stated that the Committee's work so far has been in the direction of low drag and feels that, with a similar type of approach, the Committee might be able to design an airfoil with a maximum lift coefficient, without the use of flaps, of as much as 2 or 3. A study of the factors that make for high lift, he feels, might result in the design of an airfoil that would be a far better compromise between high lift and low drag than the present low-drag airfoils.

(10) Design of optimum airfoils for use in propeller slipstream. Dr. Oswald pointed out that in many ships, as much as 60 percent of the wing is bathed in slipstream and that we do not know the best shape of airfoil to use in such locations.

(11) The effect of airfoil sections and thickness on aileron performance.

(12) Representative flap tests on all airfoils tested.

(13) Study of airfoils designed to get low drag at high lift coefficients, in other words, highly cambered sections. This is Lockheed's suggestion.

(14) Standardized wing-duct dimensions and exit design.

(15) Desirable internal structure and skin-joint designs.

(16) Further study of airfoils with reflexed camber line, suggested by Mr. Horkey, who asked the question, "Is a high C_{mo} serious at high Mach numbers?"

(17) Comparative tests of airplanes with conventional and low-drag wings showing how much can be gained through the use of low-drag wings to provide incentive for the use of low-drag wing sections.

(18) Development of permanent long-life finish for low-drag airfoils that will maintain laminar flow.

Edwin P. Hartman.

Document 4-30(a-b)

(a) Theodore von Kármán, "Laminar Flow Wings," in *Where We Stand* (written in 1945 and issued as an Army Air Forces Report in 1946); reprinted in *Prophecy Fulfilled: "Toward New Horizons" and Its Legacy* (Air Force History and Museums Program, 1994), ed. Michael H. Gorn, pp. 69-72.

(b) Von Kármán, "Aerodynamic Problems," in *Science, The Key to Air Supremacy* (originally published in 1945 as part of the multi-volume "*Toward New Horizons*," reprinted in *Prophecy Fulfilled*, pp. 108-09).

The delineation of the laminar-flow airfoils was a great contribution by the NACA, even if not exactly in the way, or to the degree, advertised. It was certainly not the NACA alone that promoted the value of the promising new technology. The last document in this chapter holds two excerpts from Theodore von Kármán, one from *Where We Stand* and another from *Science, The Key To Supremacy*, both written right at the end of the war. There can be no question after reading the excerpts that von Kármán, too, was a fan of the laminar-flow airfoil. So, too, were the German pilots who faced the P-51 in aerial combat and the German aeronautical engineers who marveled over what to them were the Mustang's mysterious design features.

But more than any other group, the people that surely valued the P-51's performance the most were the Allied bomber pilots who flew dangerous missions deep into the German heartland when no previous fighter plane could fly far enough to escort them all the way to their targets and back. Thanks in large part to its highly efficient aerodynamic design, the P-51 could fly all the way to Berlin and back, saving innumerable lives of Allied crewmen in the process. General Henry H. "Hap" Arnold, commanding general of the U.S. Army Air Forces in Europe at the time, called it "one of the great miracles of the war," the appearance of the long-range fighter escort "at just the right moment in the very nick of time to keep our bomber offensive going without a break."

Even von Kármán exaggerated the contributions of the laminar-flow wing to the P-51's overall performance abilities. But, as the second excerpt shows, in particular, as an aerodynamicist he was tremendously excited by the potential of the laminar-flow concept. The "initial successes of the laminar flow wing are so encouraging," he wrote in 1945, "that in future research we should strive to go the whole way." The P-51 was not the end of the aerodynamicist's quest for smooth air flow via boundary layer control, it was just a milestone along the way.

Document 4-30(a), Theodore von Kármán, "Laminar Flow Wings," in* Where We Stand *(written in 1945 and issued as an Army Air Forces Report in 1946).

LAMINAR FLOW WINGS
In this field we were far ahead of the Germans. In the following paragraphs, the German developed status will first be given, followed by our own.

GERMAN DEVELOPMENTS
According to the German dynamicist Schlichting, German work on laminar flow airfoils did not start until about the end of 1938. By 1940, Schlichting considered that the fundamentals were known. Drag coefficients as low as 0.0027 were reached at a Reynolds number of 5 X 10^6, but the German scientists were unable to retain the low drag at higher Reynolds numbers. They were handicapped by lack of suitable low-turbulence wind tunnels. On one occasion, Prandtl reported: "Suitable wind tunnels for the conduct of airfoil investigations at sufficiently high Reynolds number and at low turbulence are lacking in Germany. On the other hand, it is known that in the U.S.A. particular installations created for this purpose are working exceptionally vigorously in this field."

Tests were made on a Japanese laminar flow airfoil, on three airfoils derived from one member of an obsolete NACA Series 27215 (which was described in a captured French secret report), and on a few airfoils designed by Schlichting. The Germans also had some information on a Russian laminar flow airfoil obtained from a captured report.

The Germans never used laminar flow airfoils on aircraft. They were astonished and mystified by the performance of the Mustang and made many wind-tunnel and flight tests. They gave the following tabulation of wing profile drag coefficients (obtained by momentum method) for a number of airplanes at lift coefficient of 0.2:

He-177	0.0109
FW-190	0.0089
Ju-288	0.0102
Mustang	0.0072
Me-109B	0.0101

The German comment is: "The drag of this only foreign original airfoil tested up till now is far below the drag of all German wings tested in which it should be remembered that it was tested without any smoothing layer."
Another writer says: "A comparison of flight measurements shows quite unmistakably that the Mustang is far superior aerodynamically to all other airplanes and that it maintains this superiority in spite of its considerably greater wing area."

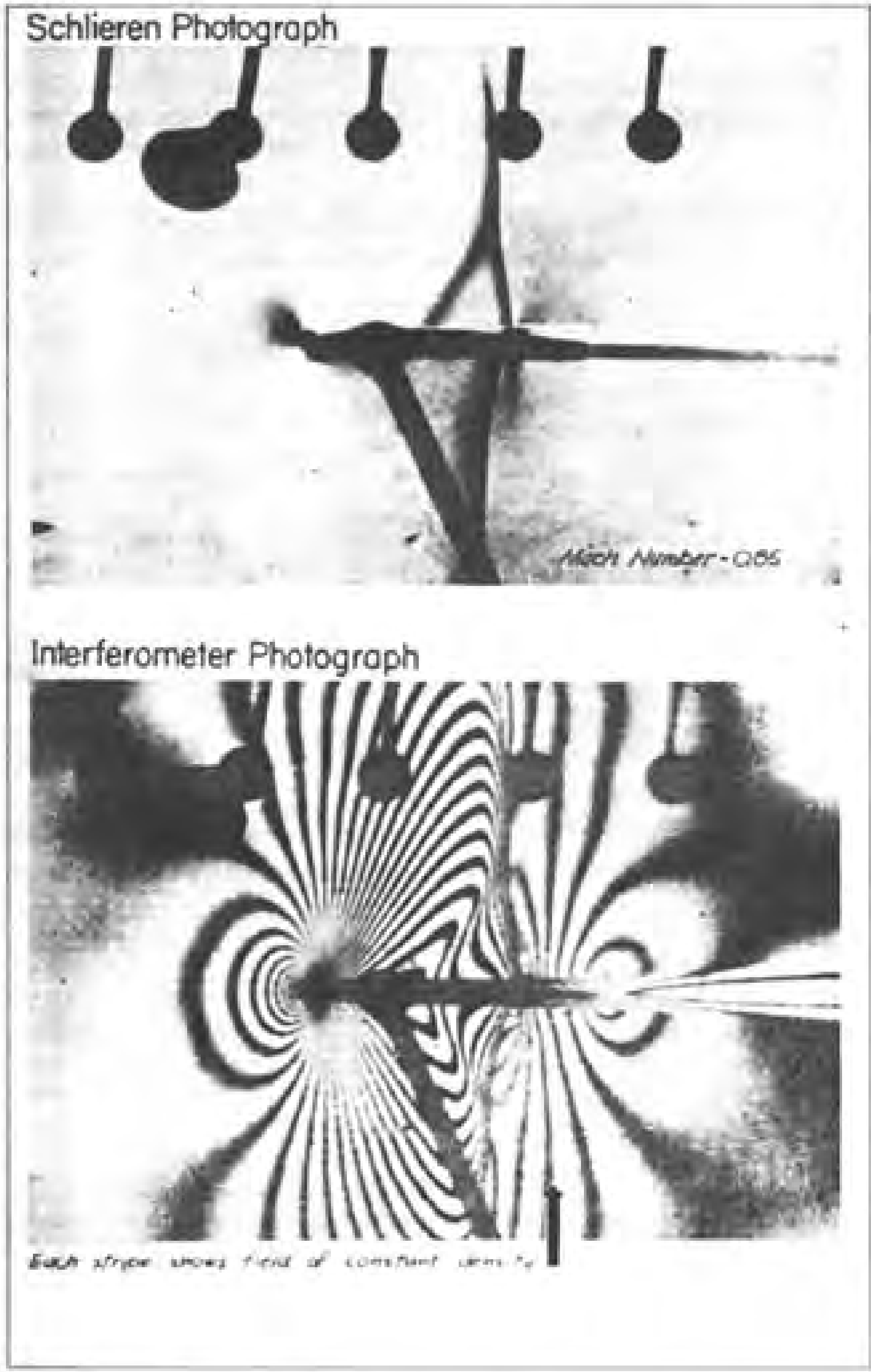


Figure 15–High-Speed Airflow Photographs

ALLIED DEVELOPMENTS

The NACA began investigations of laminar flow airfoils in a low-turbulence wind tunnel in the spring of 1938, and the encouraging nature of the results obtained (without details) were described in the Wilbur Wright Lecture of the Royal Aeronautical Society on 25 May 1939, and in the NACA Annual Report for 1939. In June 1939, an advance confidential report by Jacobs was released. A summary was published in March 1942 in confidential form. The most recent summary was released in March 1945, and this summary has been kept up to date by supplementary sheets.

As indicated in the summary of German developments, the Allies are far ahead in low-turbulence wind-tunnel equipment and in knowledge of laminar flow airfoils and their application to aircraft. Drag coefficients as low as 0.003 at a Reynolds number of 20 X 106 have been obtained.

A summary of the present state of knowledge is given in the NACA restricted report L5C05, "Summary of Airfoil Data," by Abbott, von Doenhoff, and Stivers, March 1945.

BOUNDARY LAYER CONTROL

In this field the Germans had an advanced start and had just about reached a practical state. A discussion of German and Allied developments follows.

German Developments. Considerable work was done on boundary layer control at AVA, Gottingen, starting in 1925. The first airplane with boundary layer control was built and flown in 1932.

From about 1942 on, work was intensified. Schwier obtained a maximum lift coefficient of 4.3. using pressure jet boundary layer control in wind tunnel tests. In July 1943, Stuper obtained a maximum lift coefficient 3.8 in full-scale flight tests with boundary layer control by suction. The maximum lift coefficient on his airplane without boundary layer control was 1.9. About the same time, a maximum lift coefficient of 3.4 with boundary layer control was reported in wind-tunnel tests of a four-motored airplane which was to be developed by Junkers. A unique suction and pressure-jet boundary layer control system was used. Air was sucked in over the inboard portion of the wing, just ahead of the flaps, and blown out over the outboard portion of the wing, just ahead of the ailerons. In November 1943, Wagner outlined work which was done at Arado, showing a maximum lift coefficient of 4.0 to be possible.

All German investigators noted that the internal wing ducting required and the power required to drive the boundary layer control equipment constituted serious obstacles to the successful, practical application of boundary layer control. However, it was felt that these obstacles could be successfully met. At the end of the war, an Arado transport airplane, having low landing and take-off speeds because of boundary layer control, was in service in the German Air Force.

UNITED STATES DEVELOPMENTS

An L-1 airplane was equipped with boundary layer control by suction. The maximum lift coefficient was 3.5 without boundary layer control and 3.6 with boundary layer control. The landing speed of the modified L-1 was considerably higher than that of the original airplane due to the weight of the boundary layer control equipment.

Boundary layer control has an important application in making low landing speeds possible on high-speed aircraft. It also appears that the potentialities of boundary layer control in the transonic speed range have never been systematically evaluated. We found that some interesting work was done by Ackeret at the Institute of Technology in Zurich, Switzerland. The Scientific Advisory Group recommends that an intensive research program on boundary layer control be undertaken by the Army Air Forces.

Document 4-30(b), von Kármán, "Aerodynamic Problems," in* Science, The Key to Air Supremacy *(originally published in 1945).

AERODYNAMIC PROBLEMS

2.20 Improvements in the lift-drag ratio proportionately increase the range of an airplane. Therefore, efforts should be concentrated to attain such improvements. In 1935, an eminent American aerodynamicist, who, ironically enough, later became instrumental in the development of the laminar wing, declared that in his opinion no more major progress can be expected in aerodynamic science. He referred to the fact that with the discovery of the wing theory, lift and drag became calculable quantities, and the performance of the airplane could be fairly exactly predicted. Also, the designer learned the rules of streamlining and methods of eliminating superfluous drag by "cleaning up" the airplane. By use of systematic and detailed wind-tunnel tests, this cleaning up process became almost perfect, so that further improvements can be expected only in exceptional cases. However, even in the fairly well explored subsonic speed range, new possibilities appeared with the discovery of the laminar wing section and the efforts to design an efficient flying wing.

2.21 The concept of the laminar wing is based on the fundamental fact that when the flow in the boundary layer of a surface moving in air is laminar, the surface friction is very much less than in the case when turbulent motion takes place in the same layer. The laminar wing sections which we are using in the present-day design, endeavor to keep the boundary layer laminar over a portion of the wing surface by means of an appropriate shape of the section. This method was applied in the design of quite a few of our modern airplanes, with considerable success. The proposal was first received with skepticism. Several objections were raised; that the expected effects of drag reduction could only be obtained if the wing surface is extremely smooth, and that the beneficial effect could only be attained for small values of the lift coefficient, thus restricting the benefit of the reduced friction to certain flight attitudes. Nevertheless, it appears that the initial successes of the laminar wing are so encouraging that in future research we should strive to go the whole way, i.e., to try to secure laminar flow in the boundary layer by positive measures along the entire wing and in a large range of angles of attack. It is known that theoretically this aim can be attained by the so-called boundary layer control. Results along this line are already available, for example, in the tests carried out by Professor J. Ackeret and his collaborators at the Technical University at Zurich. It is true that the process requires extremely smooth surfaces with relatively narrow slots extending spanwise along the wing. This might cause practical difficulties (for example, in the case of icing). However, looking into the future, extreme smoothness might be realized by materials now in the making, and it will certainly be worthwhile to put in a great amount of research work to eliminate other possible practical obstacles. There is even the possibility of eventual elimination of conventional movable control surfaces, by use of boundary layer control to effect changes in lift and moment.

2.22 The same principle can be applied also to reductions of the drag and airplane for example, bodies with circular cross sections. In the case of wings, it will be necessary to subdivide the wing into a number of compartments, with individually regulated boundary layer control. In the case of bodies, it might be sufficient to apply the control at a few critical cross sections.

2.23 The fundamental idea of the flying wing is the elimination of the parasite drag contributed by such parts of the airplane as do not produce lift. The tailless airplane is an even more controversial subject than the laminar wing. As does every unorthodox type, it introduces some new problems. The fact that the longitudinal control is placed in the wing involves control force characteristics which are different from those occurring in conventional airplanes. Much discussed problems are the proper method of securing directional stability, and the best arrangement for sweepback. As a matter of fact, the designs which have been produced up to now have not yet brought a final decision concerning the relative advantages and disadvantages of the flying wing and the tailless airplane. However, as the global character of aerial transportation, and especially aerial warfare, becomes more and more evident, it is apparent that our present airplanes are inadequate to meet the demand for range. Therefore, the two methods promising essential aerodynamic progress, namely boundary layer control and tailless design, should be explored with adequate facilities.

2.24 The large decrease in the value of the lift-drag ratio at the Mach number of about 0.8 is due to the rather sudden increase of the drag of the airplane. This increase is essentially due to the fact that the relative velocity of the air locally becomes larger than the velocity of sound. Simultaneously with the increase of the drag, difficulties are encountered, in most cases, in the stability and control of the airplane. Generally these phenomena are designated as compressibility effects; we prefer to use the designation "transonic problem." Obviously, in order to extend the speed limit of high-speed airplanes, a thorough investigation of the aerodynamic phenomena in the transonic range is needed. As a matter of fact, the aerodynamics of both the subsonic and supersonic ranges are better known than that of the transonic range, which extends approximately between the Mach numbers of 0.8 and 1.2. One reason is that the mathematical analysis is extremely difficult, since the flow around the airplane is partly subsonic and partly supersonic. Another great difficulty is caused by the unreliability of wind-tunnel tests in this range. Flight tests, dropping tests, and measurements on models carried by rockets are the main sources for experimental information.

2.25 Fighters and interceptors now in the making operate actually at the border of the transonic range. Hence, every method which is able to raise the limit of the rapid drag increase is of great importance. German scientists observed that increase of drag of the wing can be postponed to higher Mach numbers by sufficient sweepback. This method is generally used now in the design of fast fighters and interceptors. Designers are seeking means to reduce the excess weight and the difficulties in stability and control connected with the swept-back wing shape. However, this solution is not necessarily a final one. When our knowledge of aerodynamic phenomena in the transonic range has been more firmly established, we may find methods for eliminating the separation of the flow behind the shock wave, and the fundamental trouble, namely the occurrence of shock waves. In the subsonic range aerodynamic research brought rich returns. It can be expected that the same process will repeat itself and will lead to the solution of the transonic problem.

2.26 One of the main questions in the supersonic speed range is the feasibility of long-range flight. The supersonic airplane necessitates very high wing loading with small size of the wing. Hence, in most cases, the volume available in the wing for fuel or payload is very small, and a disproportion appears between the sizes of the wing and the fuselage. In other words, the resistance of the body in comparison with the resistance of the wing is much greater than in the case of the conventional subsonic airplane. It appears that the best solution is offered by a fuselage of large fineness ratio. A rather thorough investigation of the problem was made by the Scientific Advisory Group on this question. These investigations suggest that, assuming a given ratio between fuel and total weight and a certain space required in the fuselage, the range is essentially a function of the altitude at which the supersonic flight takes place. A preceding diagram shows an example of the variation of range with altitude. The ideal application of such a supersonic airplane is the pilotless bomber. Similar types of supersonic airplanes will serve as pilotless interceptors. The best speed range for the latter device may be between 1.2 and 1.5 times sound velocity.

2.27 The fact that in the case of the supersonic airplane, the body resistance contributes a relatively larger portion to the total drag than in the case of subsonic planes calls for study of an all-wing design. However, supersonic flight requires wings with small thickness-chord ratio. Hence, one can create sufficient space only by using a wing shape of very small aspect ratio. It is fortunate that, in the supersonic range, triangular-shaped wings give relatively high lift-drag ratios in comparison with other plan forms. Hence, for manned interceptors a series of all-wing airplanes should be tried, eventually with a small cockpit for a pilot. Such a series should extend from a tailless airplane similar to the Me-163 to pure triangular-shaped airplanes.

2.28 Besides the lift and drag properties, the questions of stability and control are the most important. The change of the flow regime introduces difficulties in the transonic range. But also in the pure supersonic range, very little is known about the efficiency of aerodynamic control surfaces and control forces. This field needs thorough exploration by all means available, starting with wind-tunnel tests and ending with flight tests. Possibly in addition to conventional means, displacements of weights or direct control of pressure distribution by modification of the flow, as in the case of boundary layer control, are necessary.

2.29 The difficulties of landing are much more serious for supersonic than for subsonic airplanes because of their high-wing loading. The wing loading decreases with altitude and supersonic airplanes designed for stratospheric flight may land without special devices. However, systematic investigations are necessary of high-lift devices suitable for use on the thin, sharp-nosed airfoils that are desirable for supersonic flight. This must include the problem of raising the maximum lift of triangular, low-aspect-ratio wings, and particularly of reducing the extremely large angles at which such wings now attain their maximum lift. In addition, devices such as rockets, which produce simultaneously deceleratory thrust and increase of lift for the short period of landing should be studied.

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Cover: *Fluid Dynamics*, Tina York. The study of fluid dynamics attempts to explain what happens to an object when it encounters the friction of atmospheric resistance (such as a plane encountering resistance as it speeds through the air). The artist has decided to depict the effect of air flow as a plane or other flying objects move through the air. NASA Image 95-HC-379.

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