8-FOOT TRANSONIC TUNNEL 1953 INSPECTION

INTRODUCTION

You are now visiting the Langley 8-Foot Transonic Pressure Tunnel. This tunnel is the NACA's newest transonic testing facility. At this visit we are going to give you a description of the two Langley 8-Foot Transonic Tunnels. In addition, you will inspect the data and control center, test chamber, and test section of the new 8-Foot Transonic Pressure Tunnel. Following this, your inspection will proceed to the first floor where you will hear about some of the research being conducted in transonic tunnels.

The Langley 8-Foot Transonic Tunnels are illustrated on this photograph and this planview drawing. The 8-Foot Transonic Tunnel is shown at the bottom and the newer 8-Foot Transonic Pressure Tunnel at the top. At the present time you are seated on the second floor at this approximate location adjacent to the data and control center. These two wind tunnels are companion facilities which utilize the same electrical drive-motor control equipment and switch gear. The common use of this equipment by both tunnels represents an important saving in capital investment since the cost of this equipment alone represents approximately 1/3 of the cost of the new tunnel. While one tunnel is in operation, model and test set-up changes are made in the other. The old tunnel was originally put in operation in 1935 as a 500 mph facility. You will recall that 500 mph was regarded as a very high speed in 1935. As a result of the demand of the industry and military services for aerodynamic information at transonic speeds, this tunnel was converted by 1950 to a transonic tunnel and at approximately this same time plans were begun for the construction of the new pressure tunnel. The conversion of the old tunnel was made by the installation of a transonic test section and by an increase in drive-motor power. This tunnel is a single-return atmospheric type in which the cooling is provided by an air exchange tower. The test section is a 12-sided polygon. The drive motor is rated at 18,000 hp for one hour's operation at 820 rpm. This motor drives a 16-foot diameter fan with 36 blades arranged in two rows.

The new 8-Foot Transonic Pressure Tunnel currently being placed in operation also enables us to conduct detailed investigations at transonic speeds. The square test section was selected to facilitate construction and model handling, and because the wave reflection problem is less severe. The test section is located here in a 36-foot diameter chamber. The contraction area ratio from the settling chamber ahead of the test section to the test section is 20-1/2 to 1.

Compared to the older tunnel, the 8-Foot Transonic Pressure Tunnel is more flexible in operation. In this tunnel, it is not only possible to vary the tunnel speed but also the pressure or density. This variation in pressure is accomplished by the use of auxiliary equipment which can pump the tunnel to two atmospheres of pressure or evacuate it to 1/4 atmosphere. This auxiliary equipment is located on the floor below and consists of a 10,000 cfm compressor, a 3,000 cfm compressor, and pressure control valves. In addition, there are silica-gel dryers and a refrigeration system which are used to control the humidity in order to avoid condensation shocks in the test section. The use of this equipment for a combined pumping and drying operation is illustrated schematically on this chart. Air enters the atmospheric intake valve, is water pre-cooled, and pumped by the 10,000 cfm compressor at a pressure ratio of 4 to 1 through a water aftercooler which reduces the air temperature to 100°F; it then enters a refrigerated freon aftercooler which further reduces the temperature to 40°F. This air then circulates through silica-gel dryers, and enters the tunnel through turning vanes with open trailing edges in this corner of the tunnel. When the tunnel has reached the desired pressure, the atmospheric inlet valve is closed and the tunnel valve, located in this part of the tunnel, is opened and the tunnel air is then re-circulated through the drying system until the desired dryness is reached. At this time the 10,000 cfm compressor is shut off. It is then used only as needed later to maintain a constant tunnel pressure, thereby compensating for leaks around hatches and access doors. The 3,000 cfm compressor then continuously recirculates the tunnel air through this system to maintain constant air dryness. Therefore, by the use of this auxiliary equipment, it is possible to accurately control the tunnel air conditions required for a specific investigation.

The tunnel is powered by an induction motor rated at 25,000 hp at 800 rpm. Accurate speed control is obtained by the use of a Kramer-Scherbius system. This power is absorbed by a 17-foot diameter fan of 32 blades followed downstream by two rows of stators. The fan drives the air in this direction.

In order to keep the air temperature within practical limits in the closed wind-tunnel circuit, the air is cooled by water pumped through a finned tube type heat exchanger located in this corner ahead of the test section. With this heat exchanger, it is possible to keep the tunnel air stagnation temperature below 125°F for maximum power conditions.

Now, we will continue our tour of this facility by an inspection of the data and control centers. Mr. will describe the operation of the auxiliary equipment and tunnel controls.

INSPECTION OF DATA AND CONTROL CENTERS

This is the data and control center. All of the controls for operating the tunnel and recording the data are at this location. On my left is the control panel for the auxiliary pumping, drying, and cooling equipment. The equipment, piping, and valves in the pumping circuit are laid out schedmatically on this panel to represent the system just described to you. This represents the 10,000 cfm compressor, this the 3,000 cfm compressor, this the refrigerating system, this the dual tower dryer, and these lines the air piping. This equipment has been seen or will be seen on the first floor. Push buttons remotely control the operation of butterfly valves which are in these lines.

The operation of this tunnel is illustrated as follows: The auxiliary equipment is put into operation from the first floor. The test engineer here will pre-set the temperature, pressure, either sub-atmospheric or super-atmospheric and the humidity controls to give the conditions he wishes to maintain during this test. He also selects the necessary pumping circuit and operates the valves in the selected circuit. The humidity is controlled and recorded here. The auxiliary equipment then operates automatically to produce and maintain the desired tunnel air condition during the test. If, during the test, it is desired to change the tunnel conditions, this can be done at any time.

The test engineer then directs the operator at the main drive motor control panel to start the tunnel. He does this and controls the motor speed to give the desired Mach number which is indicated on the Mach meter above. The data will then be recorded by cameras which are operated from here. These cameras are in the test chamber for taking schlieren pictures and here to photograph pressure distributions shown on these manometers. This screen is part of the schlieren system and is used for visual observation of the shock wave in the flow field around the model. The schlieren system is one of the items which makes possible the information on flow phenomena. This particular system allows the test engineer to keep the flow field under constant observation throughout the course of a test. These controls are used to select the portion of the flow field under observation, and the flexibility of the system allows the engineer to view any part of the flow field and from various angles. More details on the schlieren system will be given when you inspect the tunnel test chamber. The test engineer can keep the flow field under constant observation during the course of a test. The angle of attack of the test model is controlled remotely from this panel. Special NACA recording potentiometers will be located in this panel to measure the forces and moment on the model being investigated. The operation of all of these controls and the taking of data are accomplished by three people, the tunnel operator and two test engineers.

The aerodynamic load distribution on various components of a model are investigated by measuring the local static pressures on the components. These static pressures are measured by manometer boards. These manometer boards are back-lighted by neon-type lamps so that the combination of green light and the purplish manometer liquid results in greater resolution of detail on the photographic negative. This type of lighting is desirable so that the reading of the pressures recorded on the photographic negatives can be easily done by a special tele-reader. This tele-reader is a semi-automatic film-reading device which converts the photographic record to pressures and records the data on IBM cards which are then run through

IBM electronic computers. These computers convert the pressures to coefficients needed for final reduction of the data. During a pressure-distribution investigation, the amount of data taken during one run would require 40,000 data entires. The use of the IBM electronic computers has simplified the handling and reduction of data making the final information available in a short time. This concludes the inspection at this point. You will now follow me and we will continue the inspection in the test chamber.

INSPECTION OF TEST CHAMBER AND TEST SECTION

You are in the test chamber which is a part of the tunnel pressure shell. The heavy construction of the test chamber and test section was necessary because of the pressure under which the tunnel operates. This is the transonic test section. Installed in the test section is a static pressure tube for measuring the Mach number distribution throughout the entire length of the test section. This survey tube is attached to a large circular arc strut which is also used to support sting-mounted models. Thus strut is motor driven to change the angle of attack of a sting-mounted model. With this sytem, the model would stay in the center of the tunnel while the angle of attack is being changed. All pressure leads in the model pass through the strut and out the test section through manifolds in the test chamber walls and then to the manometer boards. The electrical leads are conducted in a similar manner to the recording potentiometers. For flow visualization, the test section side walls contain a number of optical glass windows for complete coverage by a schlieren system of the flow fields produced by a model. The schlieren system is suspended from this overhead crane. The main frame is a large very rigid inverted U. Light sources, parabolic mirrors, knife edges, and camera will be mounted on these pivoted platform. For visual observation, a periscope is used to reflect the schlieren picture to the viewing screen which you just saw. In order to observe the three-dimensional flow field around the model, it is necessary to move the schlieren apparatus during the test. The schlieren system is remotely controlled from the data and control center and provides longitudinal, vertical, and skewed motion of the optical axis. The flexibility of this system makes it possible to use a relatively inexpensive schlieren apparatus to perform the functions of a much more expensive system using large mirrors.

This concludes your inspection of the 8-Foot Transonic Pressure Tunnel facility.

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(a) Description of the 8-Foot Transonic Pressure Tunnel

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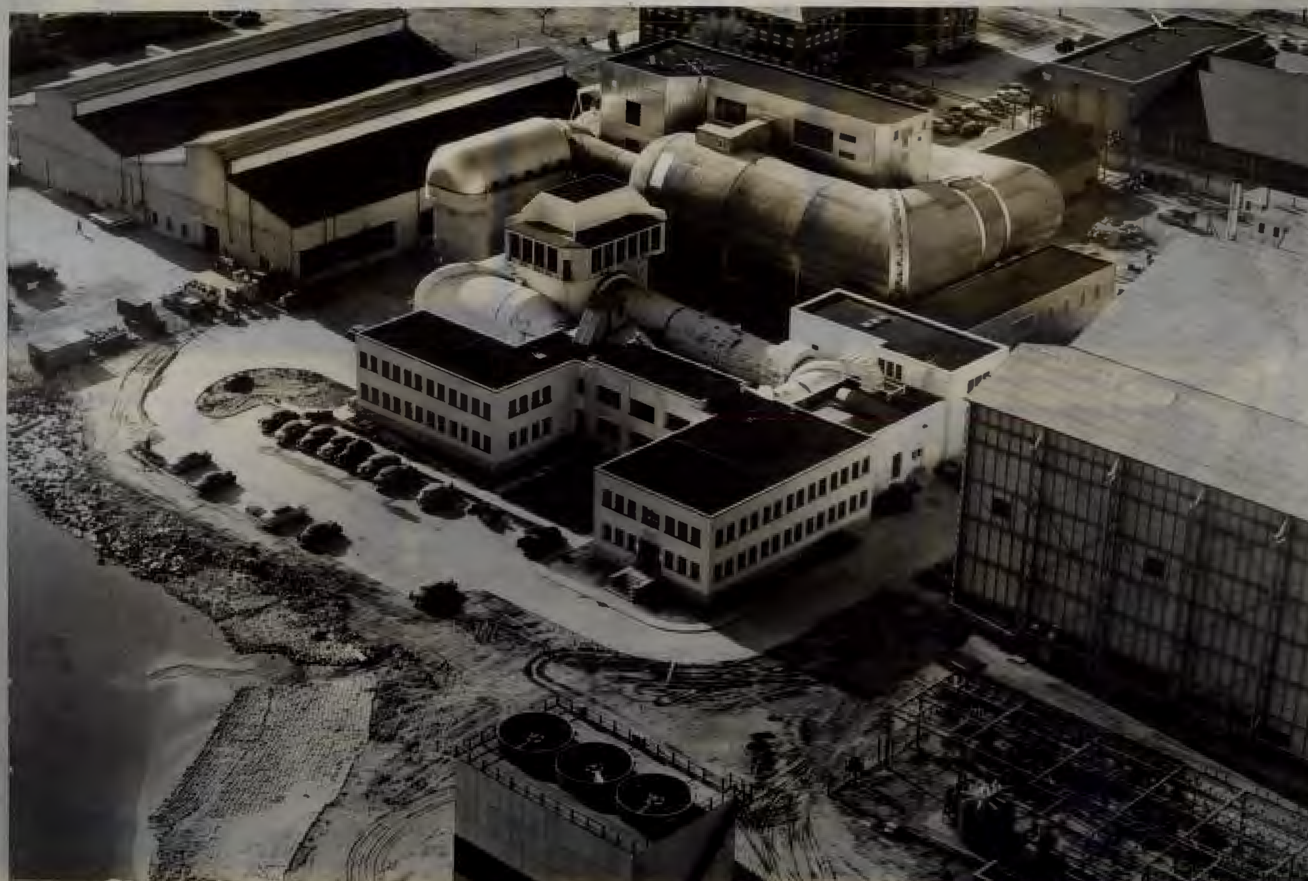
8-FT. TRANSONIC TUNNELS



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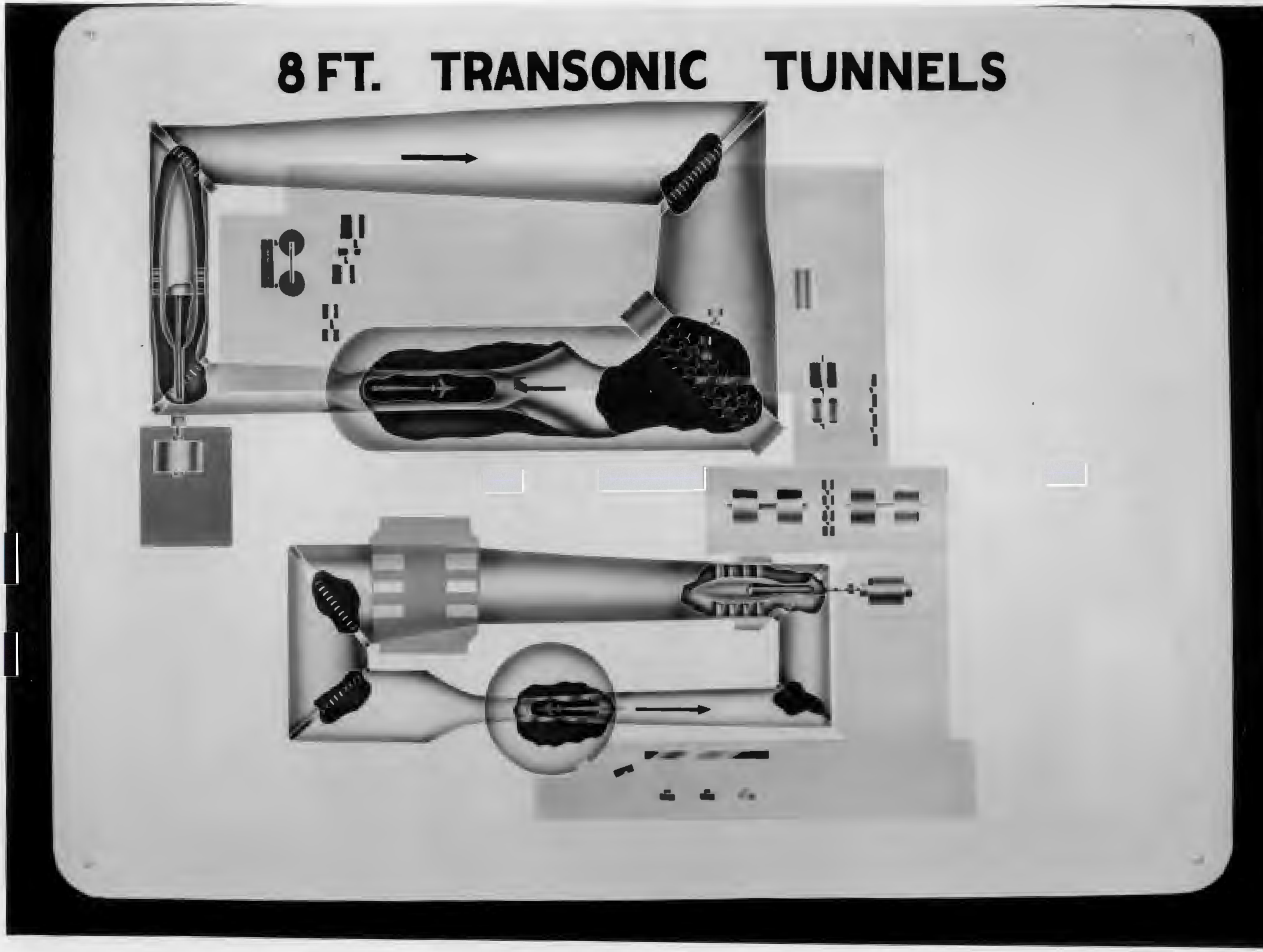

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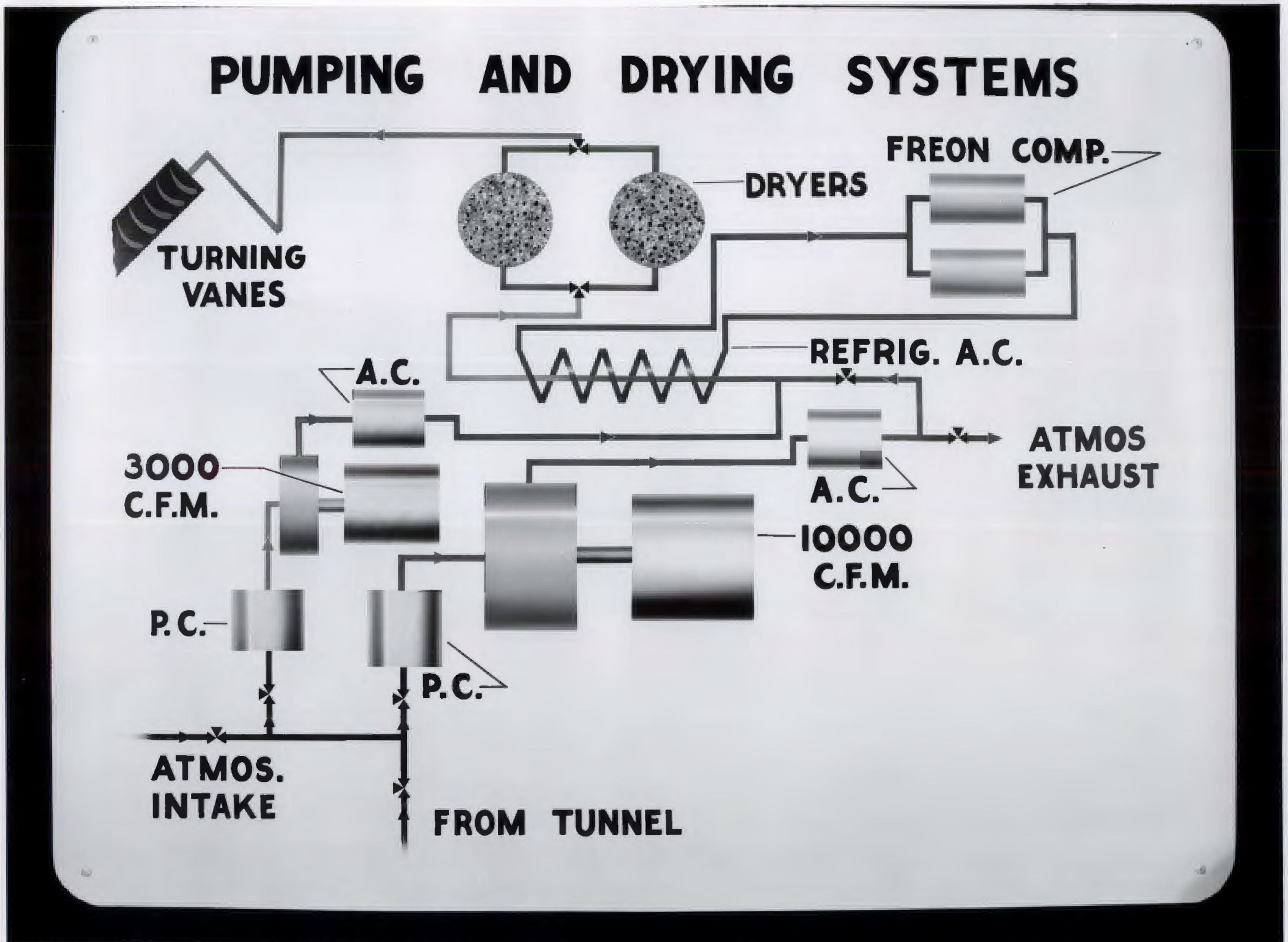
PUMPING AND DRYING SYSTEMS
FREON COMP.
DRYERS
TURNING VANES
REFRIG. A.C.
A.C.
3000 C.F.M.
ATMOS EXHAUST
A.C.
10000 C.F.M.
P.C.
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ATMOS. INTAKE
FROM TUNNEL

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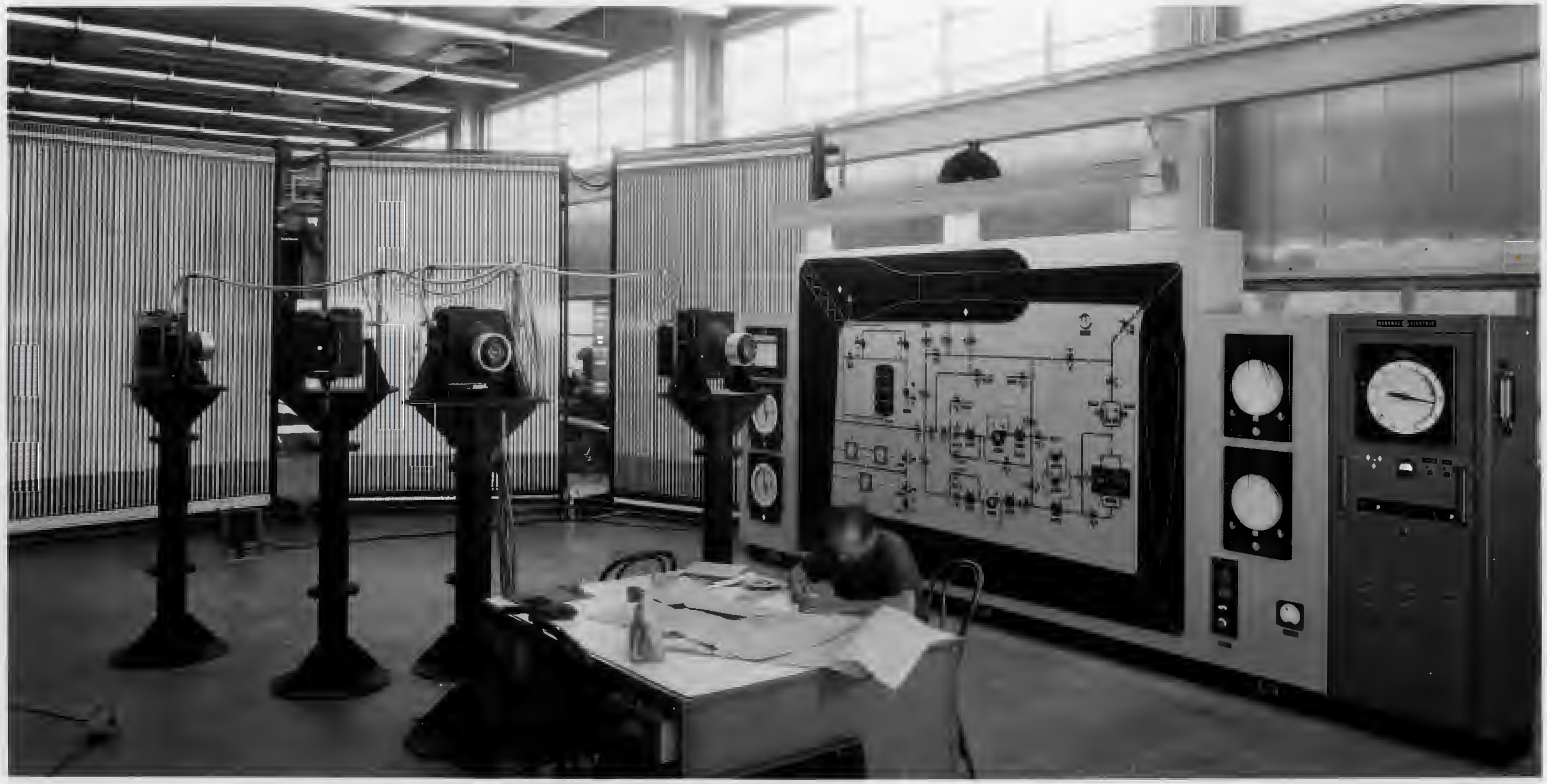

(b) Control and Data Center of the 8-Foot Transonic Pressure Tunnel

LANGLEY BIENNIAL INSPECTION (MAY 1953)

PRESSURIZED 8-FOOT TRANSONIC TUNNEL GROUND FLOOR

INTRODUCTION

Speakers: T. V. Bollech, H. N. Kelly, and G. W. Jones, Jr.

The talks at this stop are concerned with wind-tunnel research at transonic speeds. In this speed range, as you know, the air flow about the aircraft is partly subsonic and partly supersonic, with shock waves whose precise location is extremely sensitive to small changes in speed, angle of attack, aileron deflection, the exact shape of the body, and so on. The phenomena involved are far too complicated for theory to contribute much besides broad generalizations, and so we have to rely mainly on experimental research for our design information; and, the recent development by the NACA of transonic wind tunnels has aided enormously in getting these necessary data.

At the Langley Laboratory we have three major transonic wind tunnels. This first chart shows a photograph of the two 8-foot transonic tunnels. The older tunnel, which was converted for transonic operation in 1950, was the first transonic tunnel used for aeronautical research. The new pressure tunnel, where we are now, (which you have just inspected upstairs), is the most recently commissioned Langley wind tunnel. (You will inspect this tunnel just after leaving here.) The pumps and piping are part of the auxiliary equipment. In the new tunnel the air pressure can be varied from one-quarter atmosphere to two atmospheres which makes possible the study of Reynolds number and Mach number effects independently. Here is an exterior view of the 16-foot transonic tunnel which is located on the other side of Langley Field. The 16-foot tunnel is the largest transonic wind tunnel in this country and is the one which you toured at the last Langley inspection.

(Next chart) Here we have the cross-sectional shapes and relative sizes of test sections of these wind tunnels. The old 8-foot tunnel has a twelve-sided throat; the new pressure tunnel is square; and the 16-foot tunnel test section is octagonal. These wind tunnels are used for investigation of performance stability, control, flutter, and buffeting of aircraft configurations. We also study the performance of air inlets and jets, and determine load characteristics by measurement of the pressure distribution on models. The 8-foot tunnels are well suited for investigating the over-all characteristics of complete model aircraft configurations. Furthermore, these same models can be used in the 4- by 4-foot supersonic tunnel and in the 7- by 10-foot wind tunnels, thus providing a coordinated research program on a given model through its entire speed range. The use of the same model in all three tunnels results in an important saving in model cost. In addition, major reductions in operating costs are obtained by conducting low-speed parts of the test programs in the relatively low-cost 7- by 10-foot tunnels rather than in these complicated and expensive 8-foot transonic tunnels.

The 16-foot tunnel, because of its large size can accommodate relatively large models. The mock-up here represents the size of model utilized for many of the research programs. These relatively large models make possible extensive instrumentation of the small components such as ailerons, fences, spoilers, and leading-edge extensions. Most model wings are instrumented with about 300 pressure orifices, indicated by the yellow dots on this mock-up, with which the air loads are measured. The aerodynamic and structural characteristics of full-scale supersonic propellers are also investigated in the 16-foot tunnel.

On nearly all models, we measure the forces and the pressure distributions. The tunnels also have schlieren equipment, for visualizing shock formations. In addition, we frequently use the liquid-film technique and tufts for showing the air flow in the boundary layer right on the surface of the model.

The next speaker will have more to say about these flow-study techniques. Let me introduce Mr.____________, who will discuss the nature of transonic flow on wings, and the related loads and stability problems.

TRANSONIC FLOW STUDIES PERTAINING TO LONGITUDINAL STABILITY AND AERODYNAMIC LOADS

Speakers: J. M. Hallissy, A. E. Allis, and G. Hieser

By coordinating the information obtained from schlieren pictures, tuft observations, liquid film studies, and pressure distributions, we have achieved a fairly good understanding of the air flow at transonic speeds, and of the fundamental reasons for the stability, loads, and performance problems encountered in transonic flight. This understanding provides a direct and rational basis for the solution of these problems.

Throughout the subsonic speed range a leading-edge vortex flow exists on thin swept wings at moderate and high angles of attack as shown here (left side of next chart). This vortex increases in size toward the wing tip and, at a high enough angle of attack, causes separation of the flow over the tip portions of the wing. As the speed is increased into the transonic range, the vortex flow disappears; however, separation of the flow at the wing tips still exists at moderate and high angles of attack. The tip separation is now caused by an entirely different flow phenomenon; that is, the interaction of shocks with the boundary layer. This diagrammatic sketch shows the major shock formations. One shock originates at the wing leading-edge fuselage juncture and sweeps across the wing; a second shock originates at the wing trailing-edge body juncture. These two shocks reinforce one another at their intersection over the outboard portions of the wing causing the flow to separate behind the shocks in this region. At some Mach numbers and angles of attack, a shock stands out from the fuselage and further intensifies the separation of flow at the wing tips. This phenomenon is an interesting case in which the adverse effects of interference are manifest in a region remote from the source of the interference.

Here are some boundary-layer flow pictures obtained by the liquid-film technique which illustrate what I have just been discussing. A colored liquid is introduced through small orifices on the wing upper surface near the leading edge and the fluid follows the flow in the boundary layer. These pictures were made at a Mach number of approximately 0.95. The first photograph shows that at zero angle of attack the flow is essentially undisturbed over the wing except for some spanwise flow near the trailing edge at the tip. This spanwise flow exists in the boundary layer only and is caused by the normal spanwise pressure gradient. When the angle of attack is increased to 5° a disturbance across the entire span of the wing appears near the trailing edge. This is caused by the wing-body trailing-edge juncture shock which I showed here. (Previous chart.) The flow near the wing tip has become somewhat rough. At an angle of attack of 9° the disturbance caused by the wing leading-edge body-juncture shock can be seen, and the rough flow near the tip has become more intense. The beginning of flow separation is evident from the fact that some of the flow in the boundary layer is moving upstream. At 13° angle of attack complete flow separation has occurred over the outboard wing panel.

As a consequence of wing-tip flow separation, a loss in loading over these outboard portions of the wing results, along with a simultaneous increase in loading over the inboard wing panels. This phenomenon is one of the chief factors contributing to the dangerous stability characteristics called pitchup. (Use model) At small angles of attack the load is properly distributed over the entire wing, and the total lift on each wing panel can be represented by a resultant vector applied at the center of load. When the angle of attack is increased to produce moderately high values of lift coefficient, a condition encountered during take-off, landing, and maneuvers, the loss in lift at the tips causes the center of load to shift inboard, and since the wing is swept the load also shifts forward. The forward shifting of the load produces a nose-up moment, which further increases the angle of attack, and under some circumstances can overload and destroy the wings. In practical aircraft, this difficulty has been brought under partial control, for example, by use of fences and leading-edge devices, but pitchup remains a serious problem at transonic speeds for some configurations.

The next talk concerns airplane performance and will be given by Mr. ____________.

TRANSONIC PERFORMANCE PROBLEMS

Speakers: R. R. Howell, G. J. Bingham, and J. R. Mililllo

The airplane's performance is determined principally by the thrust and efficiency of the engine, and by the drag of the airframe. All parts of the performance problem are under continuous study by the NACA. We would like, at this time, to present results on engine installation and drag problems of "transonic" airplanes.

At transonic speeds the engine air inlet is usually the most critical component of the engine installation. The "transonic" air inlet suitable for use with a transonic airplane is superficially similar to the subsonic air inlet,but has a number of additional requirements. These requirements are: low transonic drag rise; and low sensitivity of both the drag and pressure recovery to variations in Mach number and internal-flow rate.

On your right are some of the transonic inlet models that have been studied. The large model was investigated in the 8-foot transonic tunnel. It has a forward-underslung inlet similar to that used on the F-86D airplane. This nose inlet was also investigated on the same body. Such an inlet would be applicable to either a fighter fuselage or a bomber nacelle. The third model has a wing-root air inlet also suitable for either a fighter or bomber. This particular model was investigated in a small transonic blowdown tunnel.

The envelope performance of these three inlets is shown in the next chart. The ordinate which defines the performance is an effectiveness factor which takes into account both the pressure recovery and drag of the inlets. Essentially, it is the ratio of the net propulsive thrust actually provided by the engine as installed, to the ideal engine thrust as specified by the engine manufacturer. As is indicated by the use of a single line, it has been found possible to obtain very nearly the same envelope performance from these three inlets.

I want to emphasize that this is an envelope curve -- by which I mean that for any chosen Mach number and angle of attack, I can design an inlet that has this efficiency at that condition, but not necessarily this efficiency over its entire flight range of Mach numbers and angles of attack.

The magnitude of the losses incurred in the "off design" flight conditions is of very serious concern to the designer. Consider, for example, that the inlet is designed to be optimum at a subsonic cruise Mach number. The inlet performance obtained in level flight then varies throughout the Mach number range in the manner shown by this curve. This is the design point, and significant losses are seen to occur at speeds greatly different from the design value.

This curve shows what happens if the inlet is designed for maximum efficiency in the high-speed rather than in the cruise condition. In this case, the losses in engine performance at the subsonic cruise speed are very large. Most actual airplanes designed in this manner require auxiliary air intakes or other means of varying the inlet geometry in order to minimize these losses.

One other point must be made. The inlet performance values given by these overlay curves are for the level-flight condition only. Additional losses, as large as 20 percent of the ideal thrust, occur at the higher angles of attack required in climb and in maneuvers.

The present status of transonic air-inlet research, in brief, is that enough is known about the simpler inlet types so that they can be designed for high efficiency in one particular flight condition. The goal of current research is to clarify the design rules for the more complicated types of inlets, and to determine means for obtaining more satisfactory performance throughout the flight range.

When optimum engine performance has been achieved, the only further means by which the speed of an airplane can be increased is by reduction of the drag. For a long time now, all jet and propeller-driven aircraft have been limited to subsonic speeds in level flight because of the large increase in drag that occurs near a Mach number of 1.0. Recent NACA research, however, has shown that a large part of this drag increase usually is caused by aerodynamic interference between the various components of the airplane. The basic understanding needed for the reduction or elimination of this element of the drag also has been obtained and the results of this research are now being applied in the design of nearly all of the new airplanes. This advance of the art is regarded as one of the major keys to supersonic flight.

The significance of this work is illustrated qualitatively on this final chart, which presents the variation of thrust and drag with Mach number for two different combinations of aircraft components for the condition of level flight at constant altitude. The angle of attack of the airplane varies from a high value at the left of the chart to a very low value here. For a specified turbo-jet engine, the thrust available over the speed range is indicated by the dotted curve. At the top of the chart are shown the major individual components from which most aircraft are built up: wing, fuselage, tail, pilot canopy, nacelles, and external stores. Combination A is a typical case in which the combination of these bodies in a conventional manner has resulted in a severe drag rise in the transonic speed range. Maximum speed in level flight is reached, of course, at the point where the thrust and drag are equal. In this case the maximum speed falls slightly below Mach number 1.0. On the basis of our better understanding of interference drag at transonic speeds, the designer can rearrange these same components into an airplane design having a much lower transonic drag rise, and reduced drag at supersonic speeds, as indicated by the drag of combination B. For the case shown, the thrust available always exceeds the drag and it becomes possible to attain supersonic speeds without diving the airplane.

In this review of transonic research we have touched briefly on the development of transonic wind tunnels and research techniques, and have indicated significant accomplishment toward improving the performance of transonic aircraft. Although these findings are quite recent, the information is already being incorporated in the designs of future aircraft, and as modifications to some already in operation.

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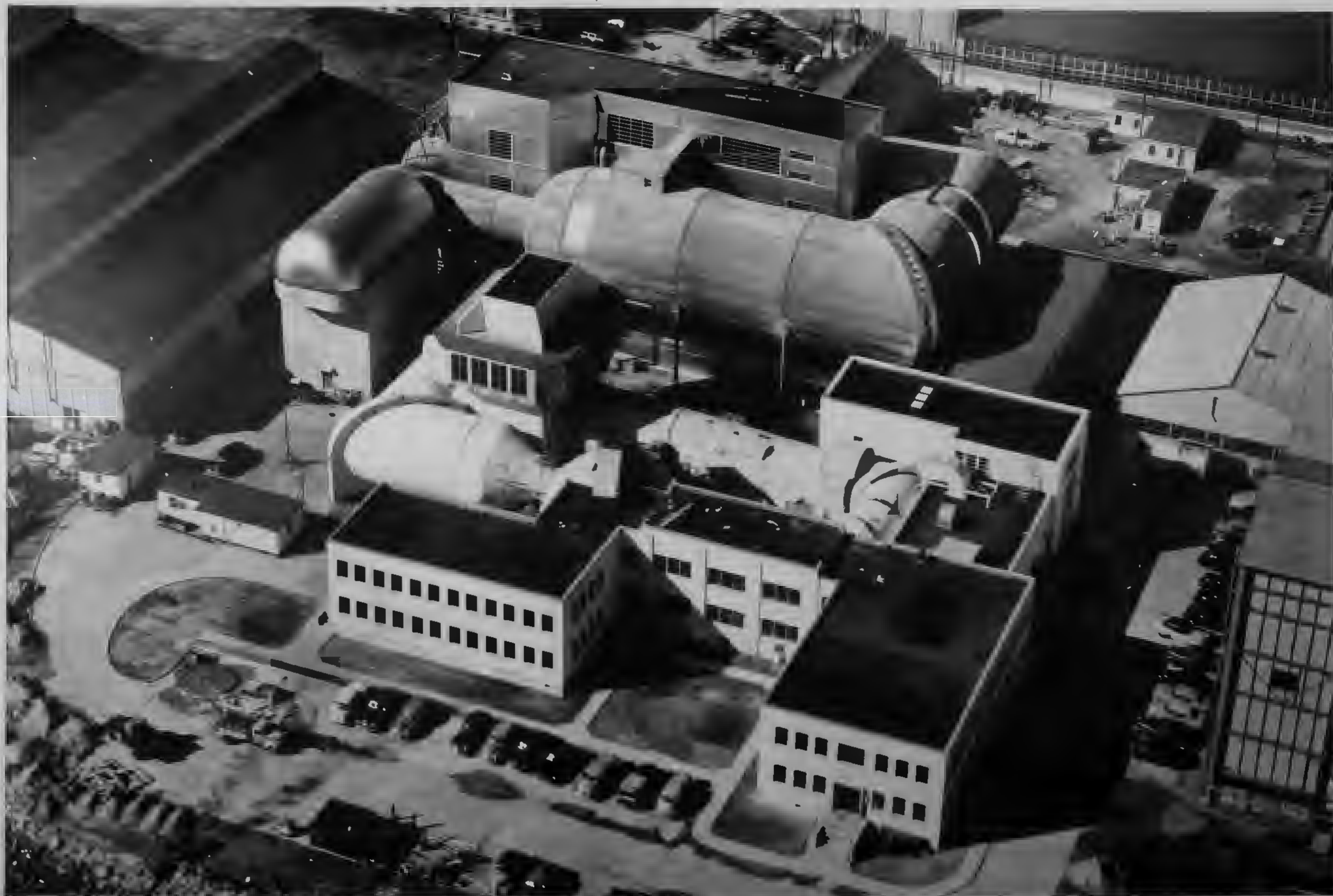
Display for Transonic Research

LAL 80165

8-FT. TRANSONIC TUNNELS



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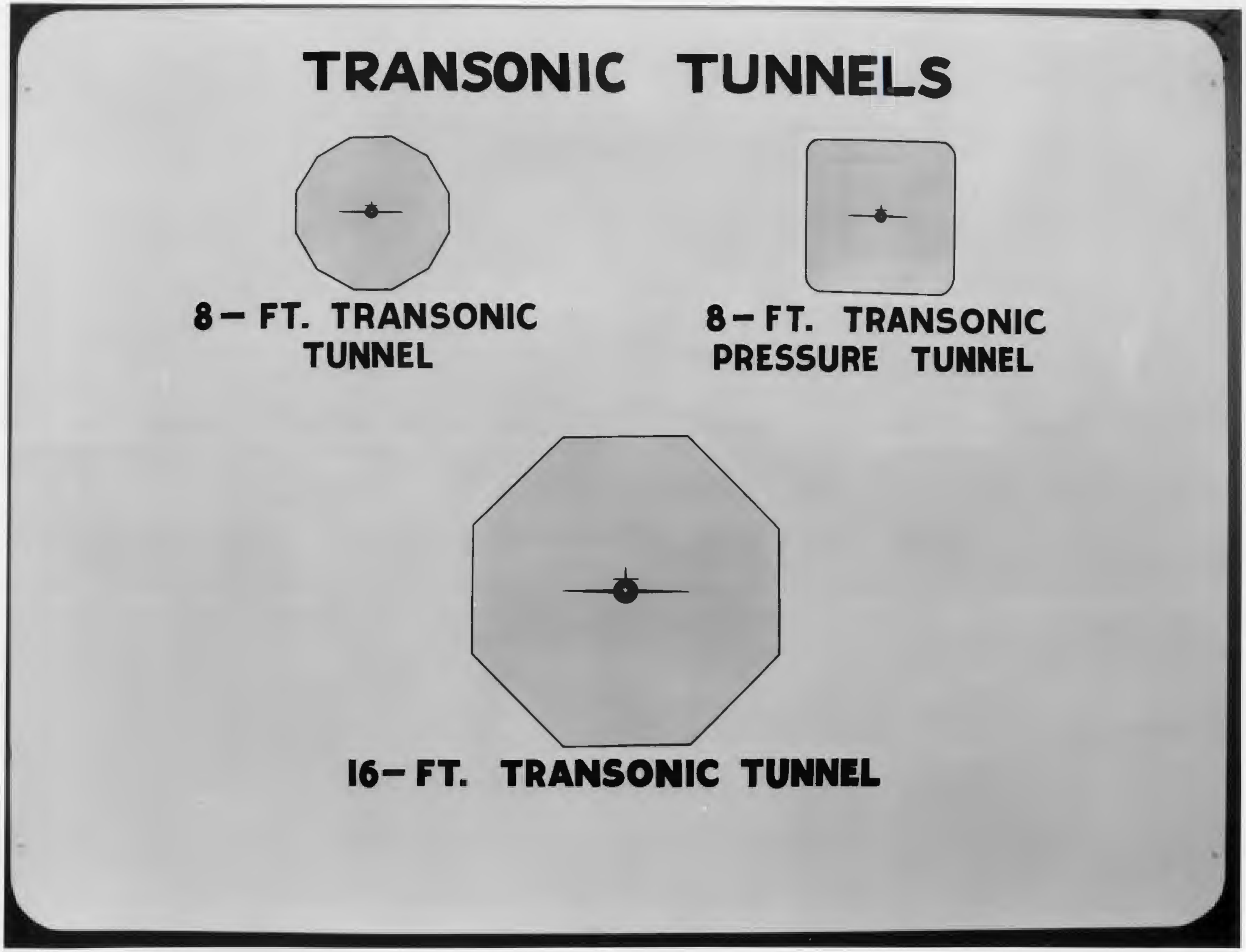
TRANSONIC TUNNELS
8 – FT. TRANSONIC TUNNEL
8 – FT. TRANSONIC PRESSURE TUNNEL
16 – FT. TRANSONIC TUNNEL

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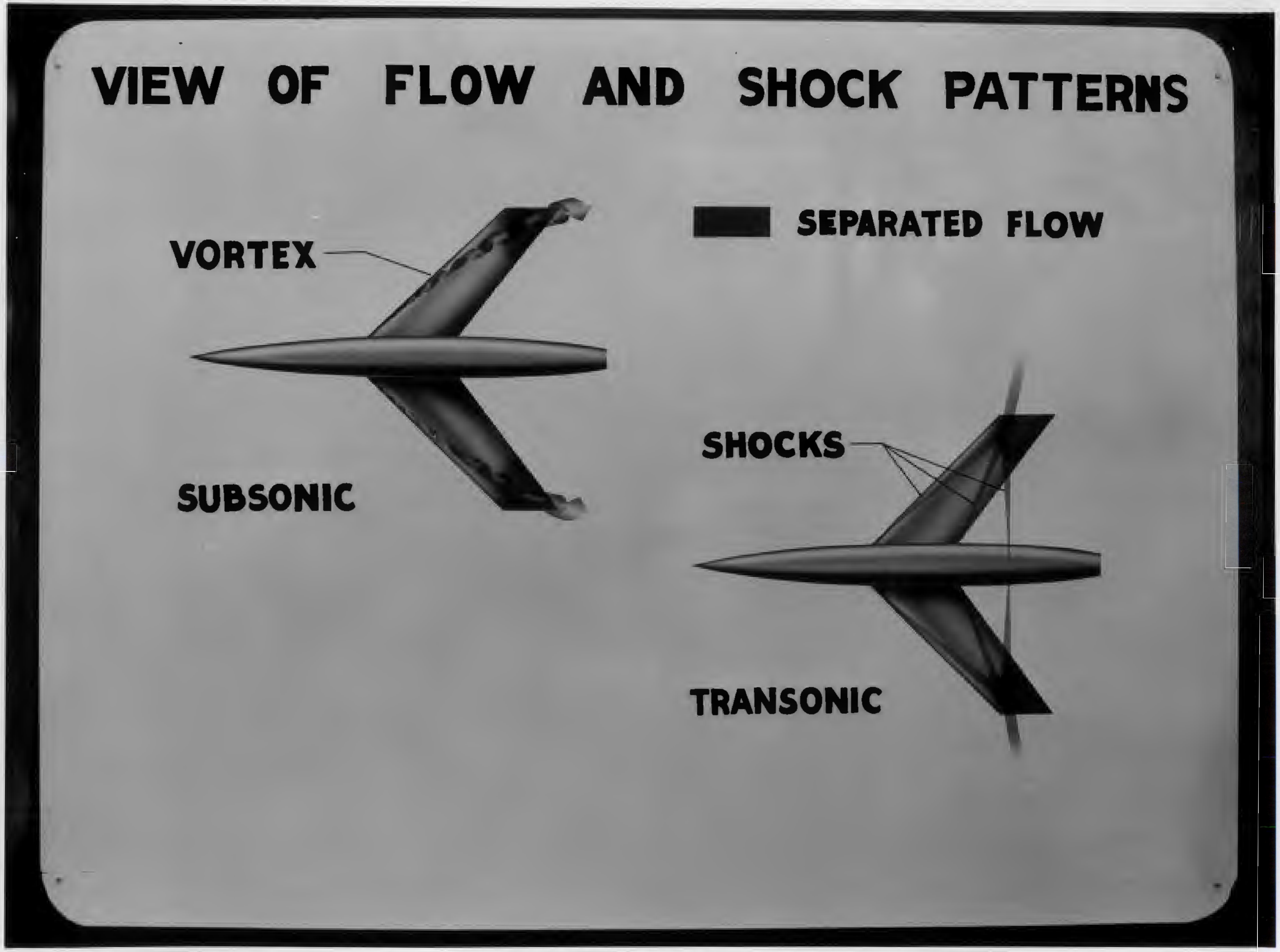
Mock-Up of a 16-Foot Transonic Tunnel Model

LAL 80160


VIEW OF FLOW AND SHOCK PATTERNS
VORTEX
SEPARATED FLOW
SUBSONIC
SHOCKS
TRANSONIC

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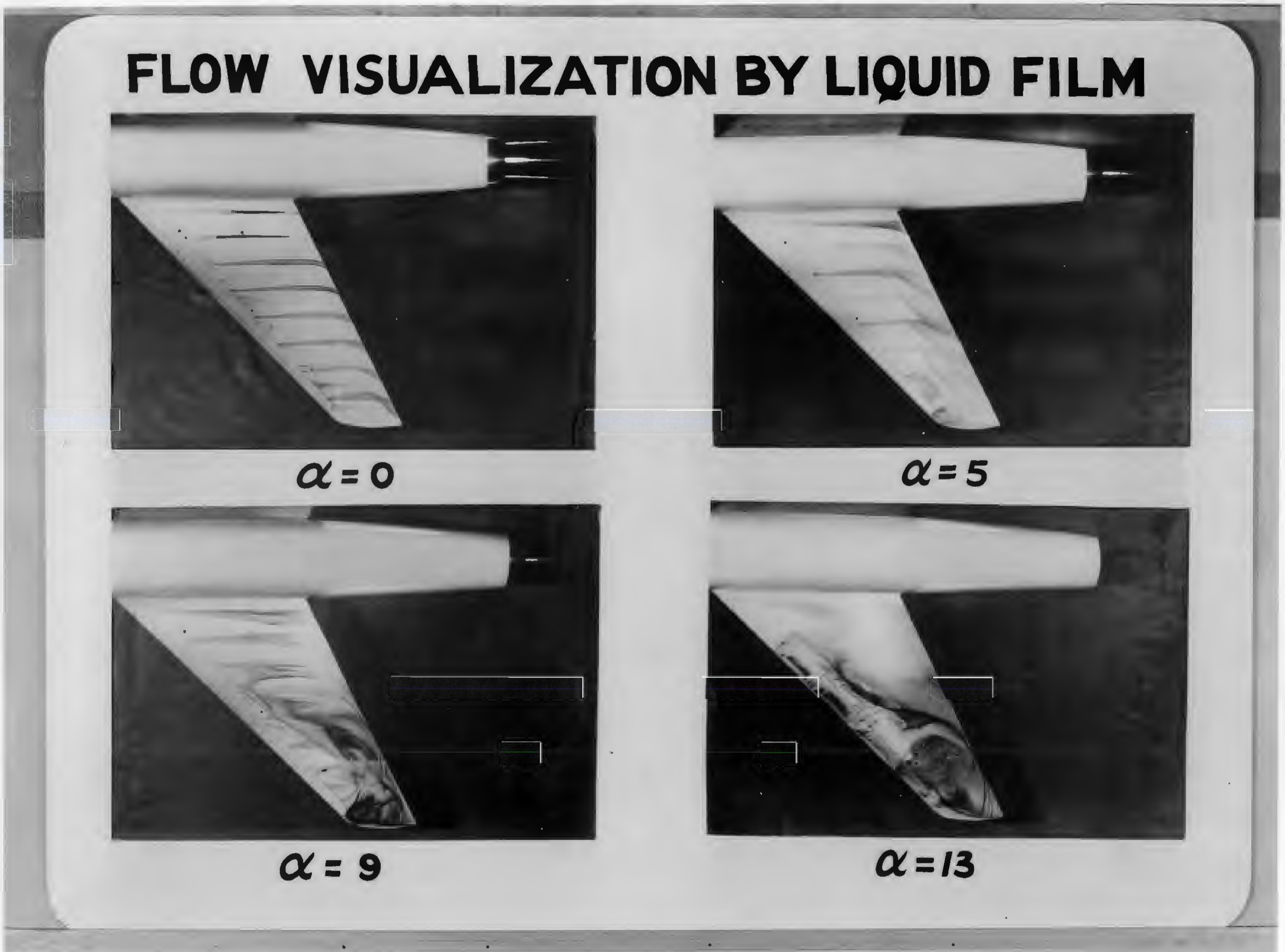

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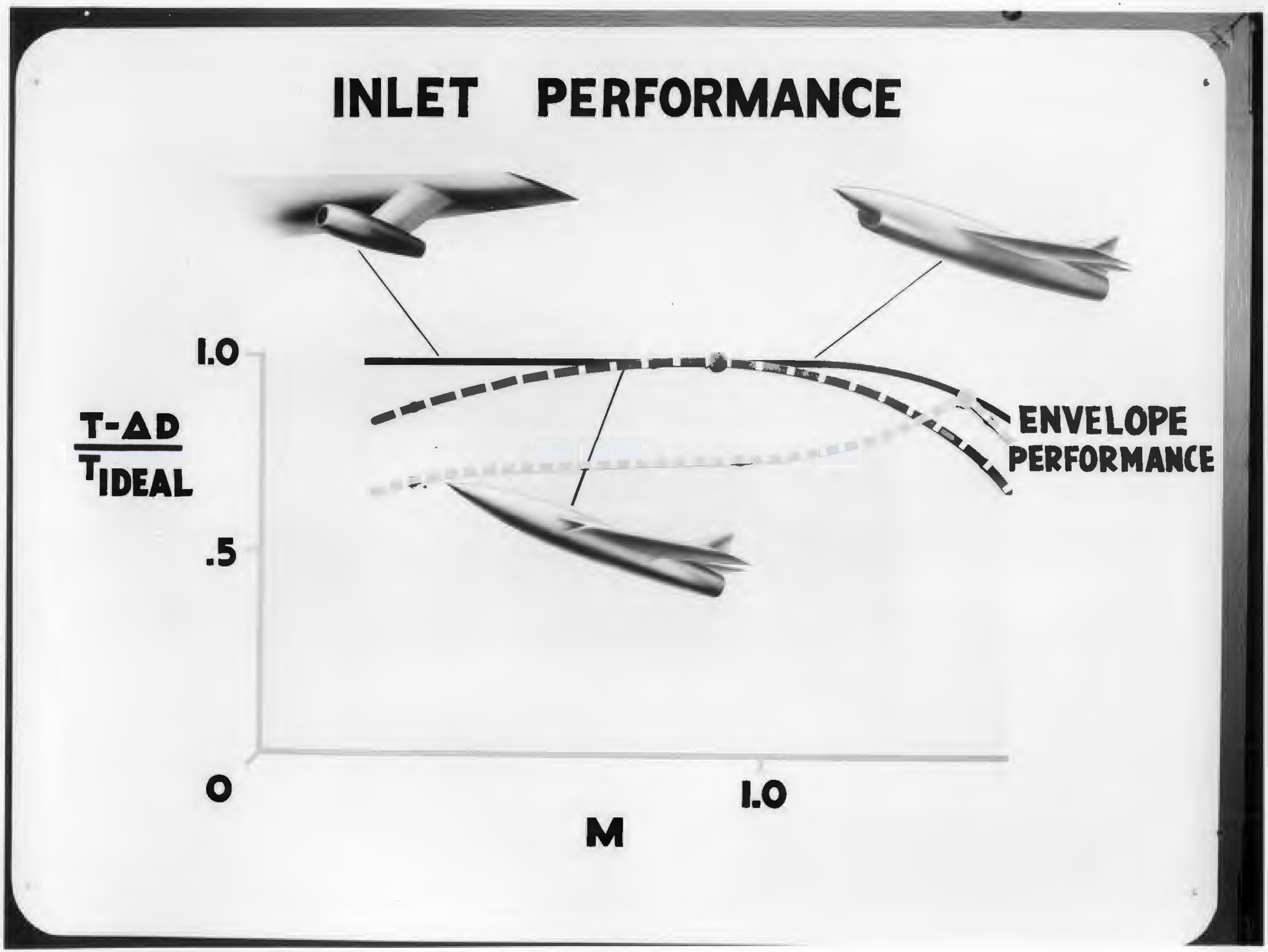
INLET PERFORMANCE
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