GENERAL DESCRIPTION OF THE GAS DYNAMICS LABORATORY AND OUTLINE OF SOME SUPERSONIC RESEARCH PROBLEMS

General Remarks

The considerable effort of the NACA devoted to study of the problems of flight at supersonic speeds utilizes a variety of facilities and techniques to carry out the experimental part of the research on these problems. These facilities and methods, briefly, include supersonic tunnels of the continuous-operation type ranging in sizes up to 8 by 6 feet and extending to Mach numbers just above 2.0, free-flight rocket vehicles covering a Mach number range up to 4.0, piloted research aircraft flying well into the supersonic speed range, and various intermittent tunnels and jets extending the Mach number range to 9. Special laboratory ballistic techniques have been used to obtain Mach numbers approaching 20, but only limited data were obtained. It is clear that each of these methods is necessary for research in special categories and it takes all of them to study the large number of problems related to aircraft and missiles which confront us, both aerodynamic and structural.

At previous inspections, some of the larger continuous-operation supersonic tunnels have been described and demonstrated. At this time it appears desirable to show you one of our newer facilities which is just now going into operation and which is somewhat unique and different. This facility is the Gas Dynamics Laboratory. The words "Gas Dyhamics" has been adopted over the years and simply refers to the basic dynamical properties of gases as related to high-speed gas flows. This laboratory was created for the purpose of exploring the problems of flight at very high speeds and altitudes, in other words, the laboratory is to be used for study of the fundamental aspects of these problems. To this end, a so-called "blowdown" facility was envisioned and constructed so that air at high pressures and high temperatures could be supplied to a number of moderate-sized separate jets or supersonic test setups. High pressures were necessary to achieve full-scale flight conditions (high Reynolds numbers) with small models, and high temperatures were necessary to avoid air liquefaction at high test Mach numbers and to provide a means for studying heat transfer. More about the high temperatures later. - - The resulting laboratory (which was started in 1950 and is just now going into operation) consists essentially of a high pressure air supply and storage system, an air heating system, and a series of pressure and temperature regulating devices so that flow can be delivered to any of a number of outlets at the desired conditions at each outlet.

Description of Air System

Chart 1 shows, schematically, the air supply and heating system and plan view of the laboratory building housing the various test jets. Rather than describe the system in detail, it is sufficient to note the pertinent points. - Air is compressed to 5000 pounds per square inch, cleaned and dried, and stored in tanks. The tanks have a total volume of 20,000 cu ft and the weight of air stored is 250 tons. Incidentally, as a matter of interest, air at this pressure and ambient temperature is about 1/3 the density of water. Air from the tanks is piped into the building and may be fed through a system of pressure (symbol P) and temperature (symbol T) controls in such fashions that a range of temperatures and pressures are available at the various outlets. The heaters can raise the temperature to 1040° F - heating is done in two stages, the first stage consisting of a vapor-to-air heat-exchanger, and the second stage, electric heaters. By suitable routing of the air through the pressure regulators, pressures can be varied between atmospheric and 3000 lbs. So-called "cold" piping (this pipe is carbon steel) carries air at moderate temperatures and pressures (up to 100° F and 500 lbs.) to a series of jets in which basic aerodynamic research is done. Hot air (up to 1040° F and 500 lbs.) is carried in stainless-steel piping to another group of jets in which heat-transfer research is done. Another pipe carries air at 1040° and 3000 lbs. to a high-pressure hypersonic jet. For the jets in which very low densities are desired (corresponding to high altitudes), a vacuum on the downstream end of the jet is required, thus the large vacuum sphere shown in the aerial photograph is connected to this jet.

Outline of Primary Problems and Work in Laboratory

Toward the aim of doing fundamental, exploratory work, the jets are small, ranging in size from 9 to 20 inches square. It should be explained that by "jet" we mean an apparatus on the end of a pipe that operates intermittently, i.e., in bursts of only several minutes duration, exhausting the flow to the atmosphere. The continuous tunnel, on the other hand, is a closed circuit, and air is continuously pumped around this circuit, thus the flow can continue indefinitely. It is important to note that the large number of small jets carry out work that saves costly time in large tunnels. Furthermore, the "smallness" of the jets permits great versatility in their use. Types of aerodynamic research done in the various jets here include basic aerodynamics, that is, studies of flows around bodies and wings, stability of aircraft and missiles, air inlets for supersonic aircraft, and, most important at the present time, studies of skin friction and heat transfer or aerodynamic heating. Mach numbers extend to 9, and simulated altitudes approach 200,000 feet. Other fundamental work carried out in the laboratory related to very high-speed flight, but not necessarily utilizing the jets, include research on turbulence, fundamental studies of the effects of dissociation and ionization of air at very high temperatures, and fundamental work on moving shock waves.

Brief Outline of Missile Heating

In order to illustrate this phenomena called "aerodynamic heating", let us take, as an example, a missile flying at supersonic speeds. First, as the air flow moves past the body there is a thin layer near the body through which the air is brought to rest at the body surface by the action of shear between adjacent layers of air. This shear layer is called the "boundary layer". When the air moves over the body at low speeds the action of skin friction or surface shear is small, thus the surface "scrubbing" does not appreciably heat the surface. As the speeds increase into the supersonic region, the scrubbing action at the surface heats the skin to higher temperatures, in fact, the skin temperatures increase with increasing speed up to a high limit. These high skin and structure temperatures are a serious problem to the aircraft and missile designer, and he needs to be able to calculate, for all flight conditions, the temperature distribution over the missile. Perhaps the most important flight case is that in which the missile starts its flight at low temperature and is rapidly accelerated to supersonic speed. For the case shown in chart 2, the temperature along the missile body was very low initially, then the body was accelerated very rapidly to a Mach number of 4.0 at 40,000 feet so that the skin is at 100° F at the start of $M = 4.0$ flight. As it continues to fly at $M = 4.0$, the skin gets hotter so that at the end of 10 minutes and 440 miles, the temperature distribution is like this. If allowed to continue indefinitely at these flight conditions, the final temperature distribution along the skin would be as shown by the top curve. The important fact in the example shown is that the missile is becoming <u>hotter</u> with time, that is, heat is flowing into the skin.

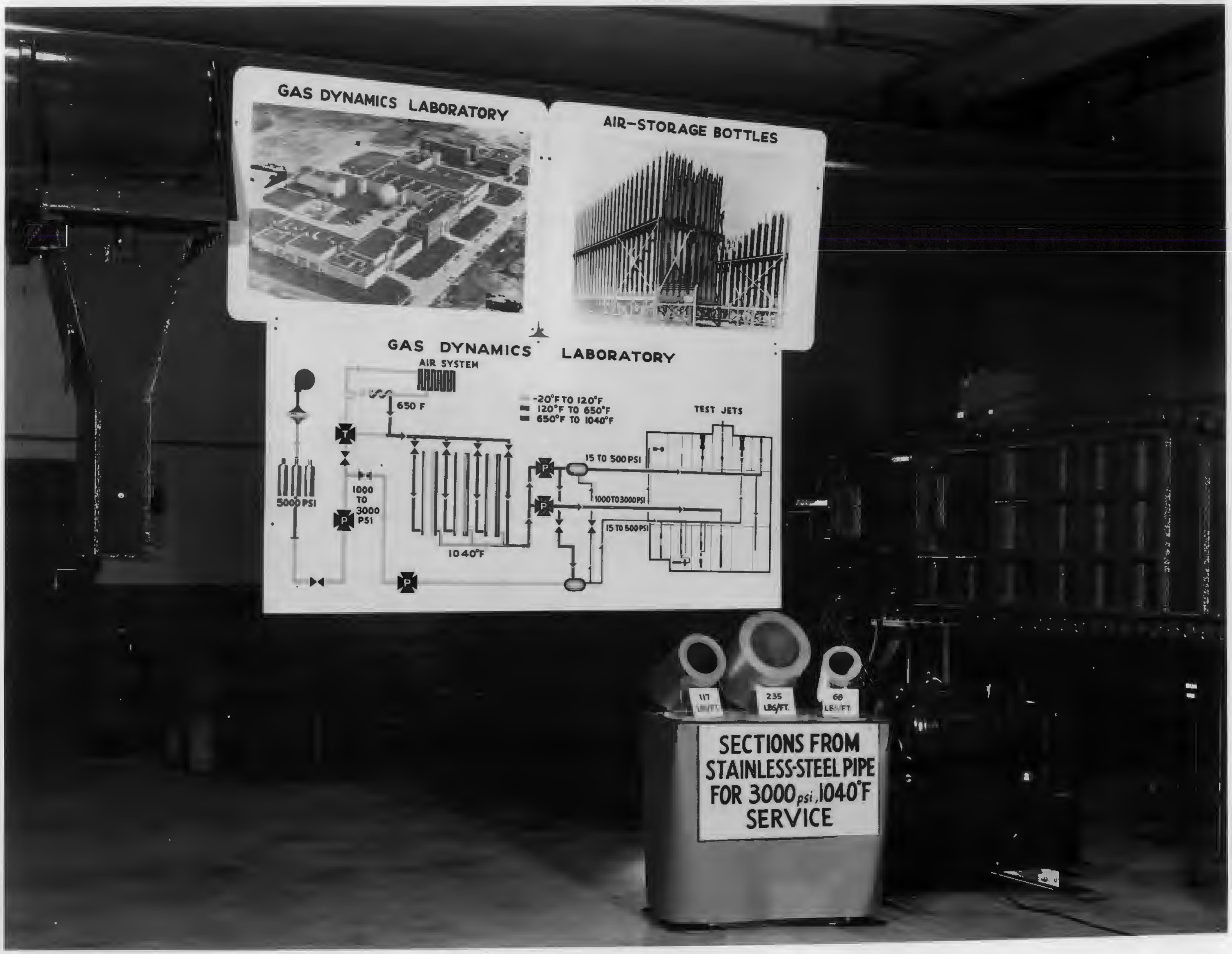
Temperature Distribution Along Nozzle

At this point it is appropriate to explain some of the experimental factors involved in the apparatus designed to study this phenomena of aerodynamic heating or heat transfer. Chart 3 shows the temperature distribution along a jet which generates a supersonic test flow. Look first at what happens to the temperature if we supply the jet with room temperature air. The air in the stilling chamber is expanded through a nozzle, shaped as shown, and reaches a temperature for the Mach number 4.0 flow, of -350° F, at which temperature the air is almost liquified. For higher Mach numbers with the same initial room temperature, the air would liquify and make tests impossible. The only solution is to initially heat the air and the top curve shows the temperature distribution along the nozzle flow when we start with air at 1000° F. The flow temperature in the nozzle for this case is about -70° F and we note that this is about the ambient temperature of the air at high altitudes and that we are far away from the liquid-air temperature. Now, if we suddenly immerse a cool model in this "hot" flow, the skin of the model starts to heat, so that at the end of about one-half minute, the temperature distribution along the model is as shown by the lower curve. If the model is left in the flow until the skin is at equilibrium temperature, the temperature along the skin is as shown by the top curve, in fact, about 900° F, almost duplicating the flight example of the previous chart. The important fact, however, is not the duplication of flight conditions, but the simulation of heat flowing into the skin with time. We make use of this fact to obtain heat-transfer data for the designer, that is, we suddenly immerse the model in the hot flow (by a technique to be shown at a subsequent stop here) and record the skin temperatures with time, thus obtaining heat-flow coefficients. Now, before showing you this technique, if you will follow your group leader across the shop, we'll show you, very briefly, some of the elements of the laboratory air distribution, heating and pressure regulating equipment.

Heater Room Description

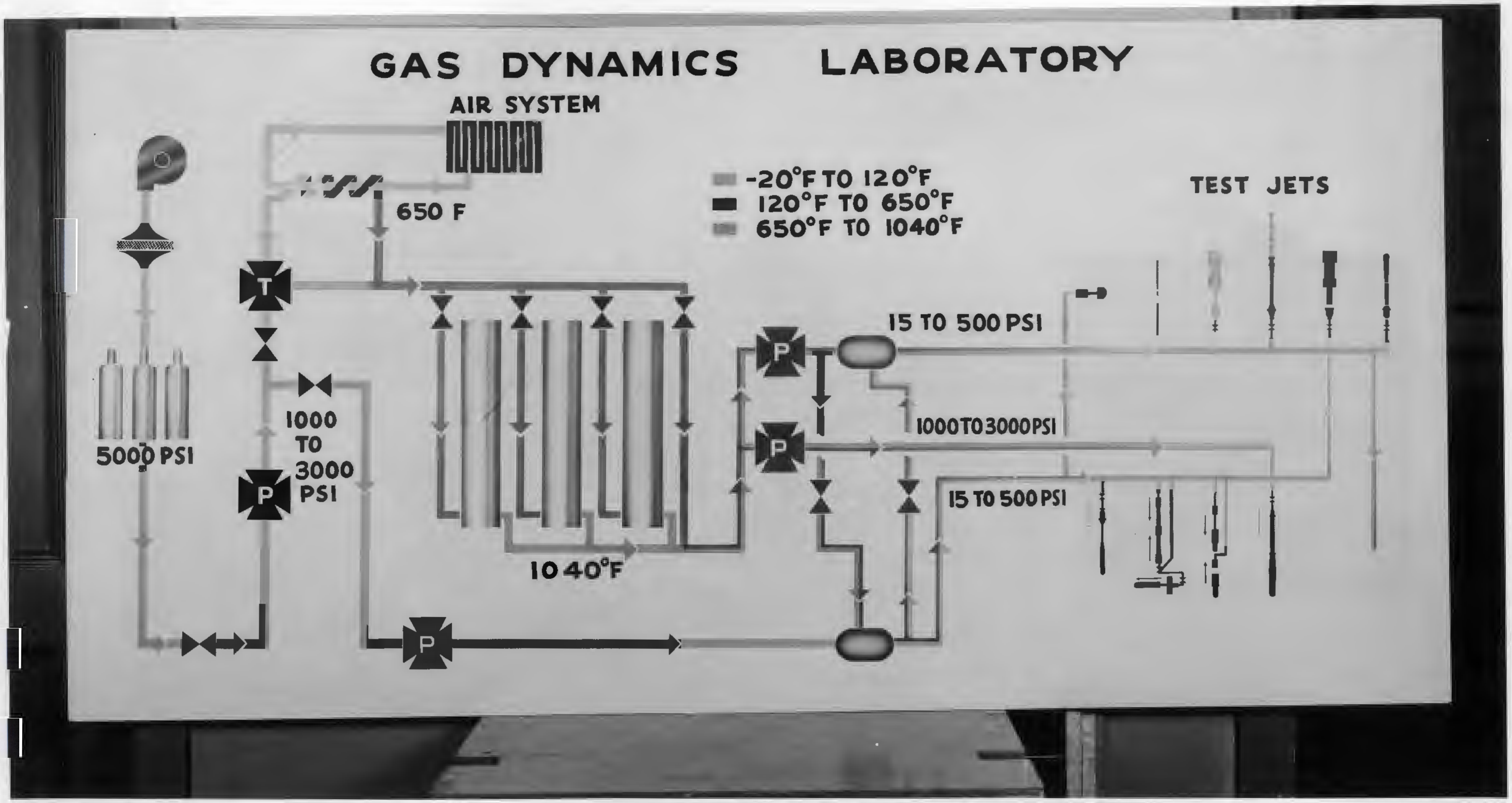
Most of the equipment is localized in this area, and although most of the piping is beneath the floor, you can see some of the more important pieces of equipment. Seen against the wall are the pressure regulators, and, above, two of the larger pressure regulating valves. At the back of the room is the heat source for the first heating stage, i.e., an oil-fired boiler which vaporizes "Dowtherm", a commercial heat-exchange medium. "Dowtherm" is a liquid having a vapor pressure of about 85 psig at 700° F. "Dowtherm" vapor at about 700° F is fed into the heat exchanger which you see on the overhead shelf, raising the air temperature to 650° F. The second-stage electric heaters are seen to your left and these raise the temperature to 1040° F. Now, if you will follow your group leader back across the shop, we will show you our heat-transfer test technique.

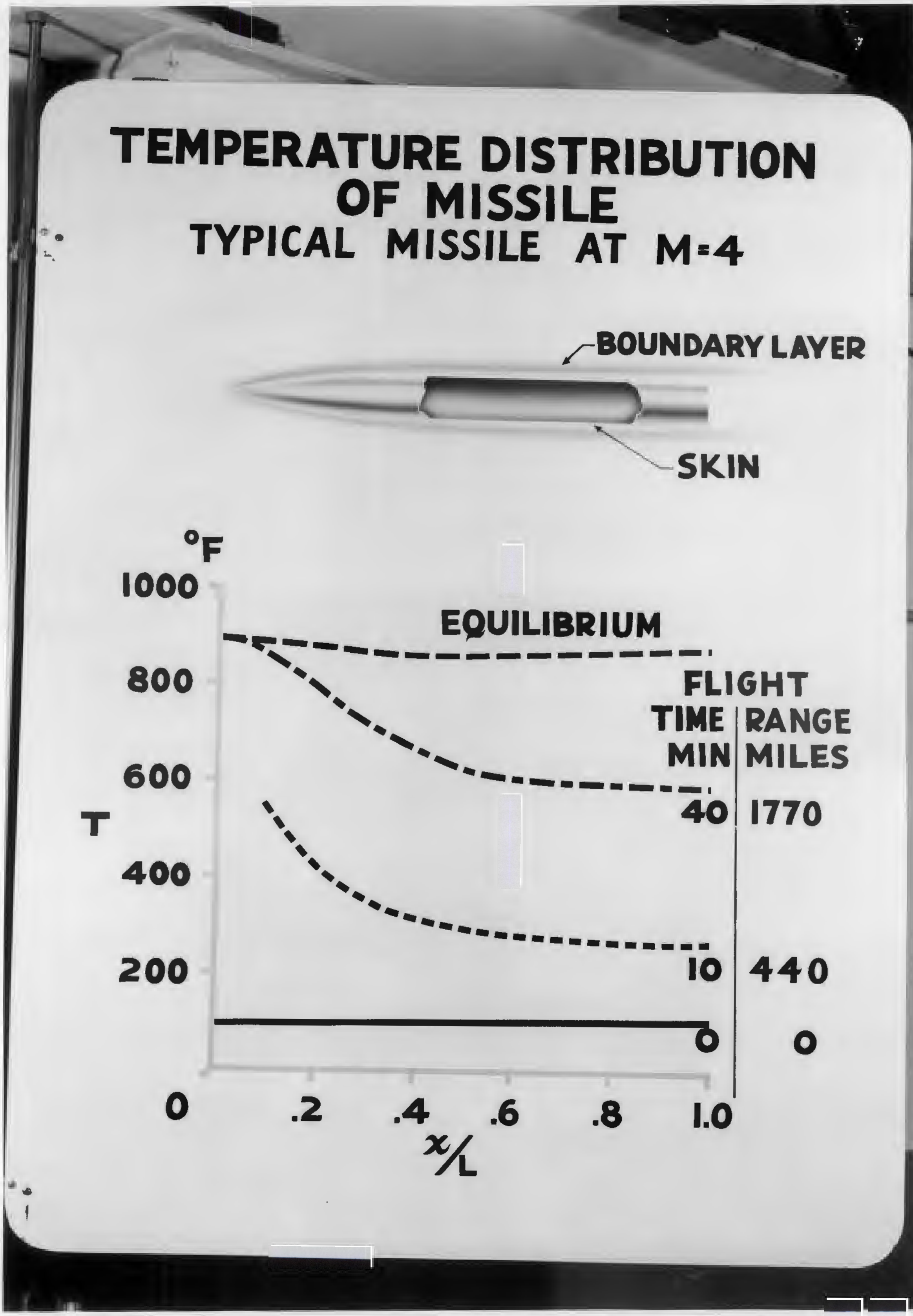
LAL 80140



Display in First Station in Gas Dynamics

NACA
LAL 79877




TEMPERATURE DISTRIBUTION
OF MISSILE
TYPICAL MISSILE AT M=4
BOUNDARY LAYER
SKIN
°F
1000
800
600
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1770
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440
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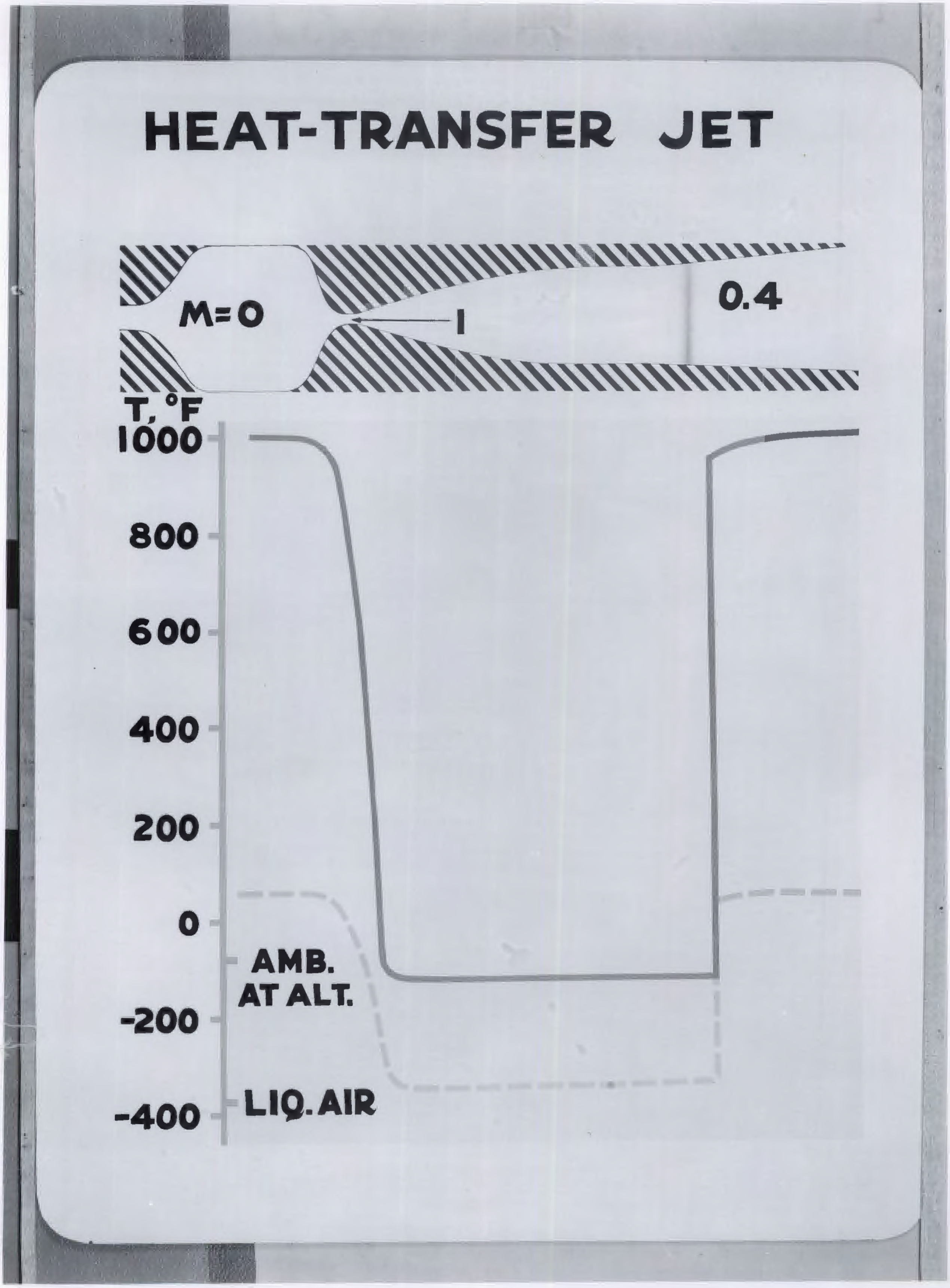
HEAT-TRANSFER JET
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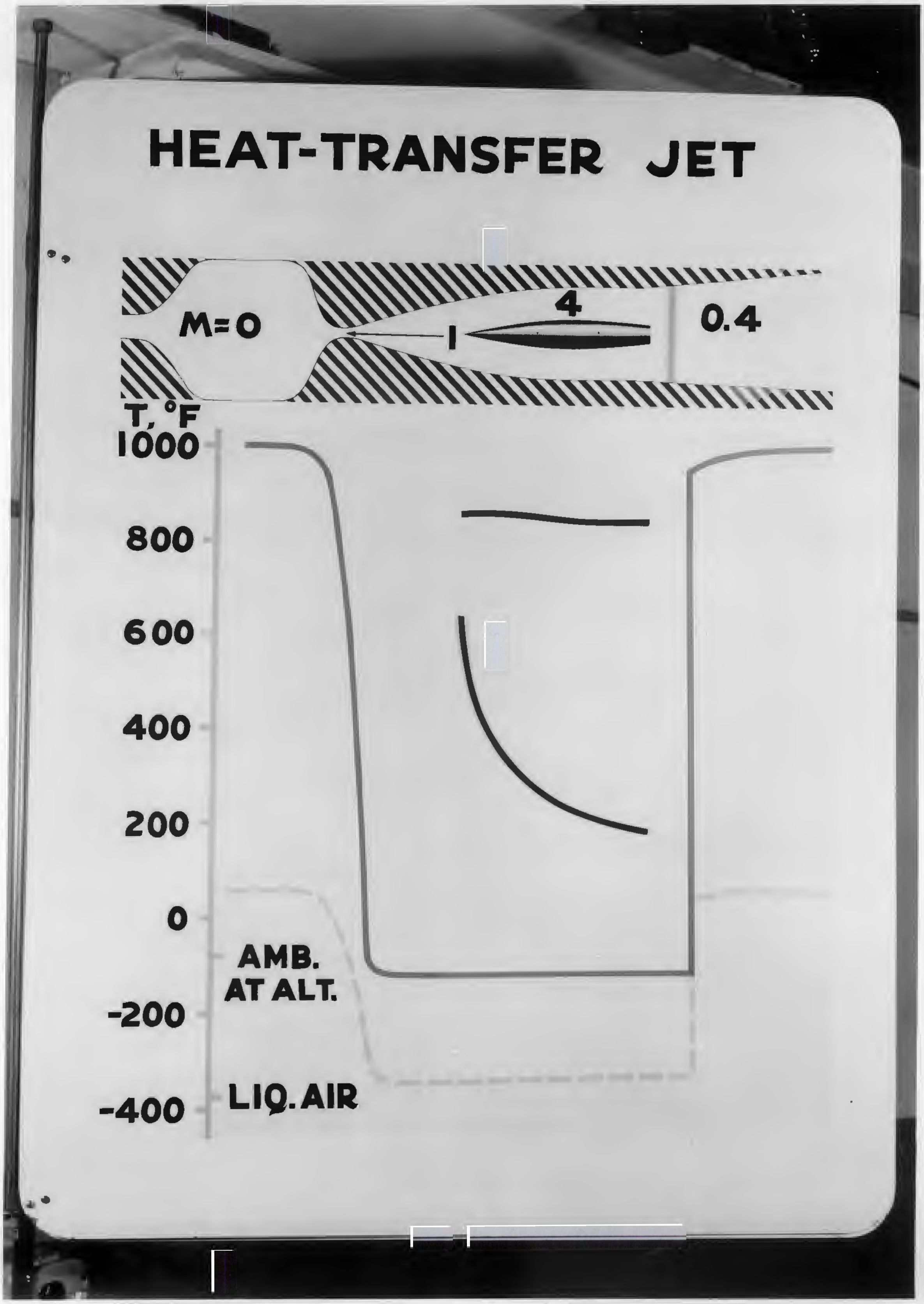
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T, °F
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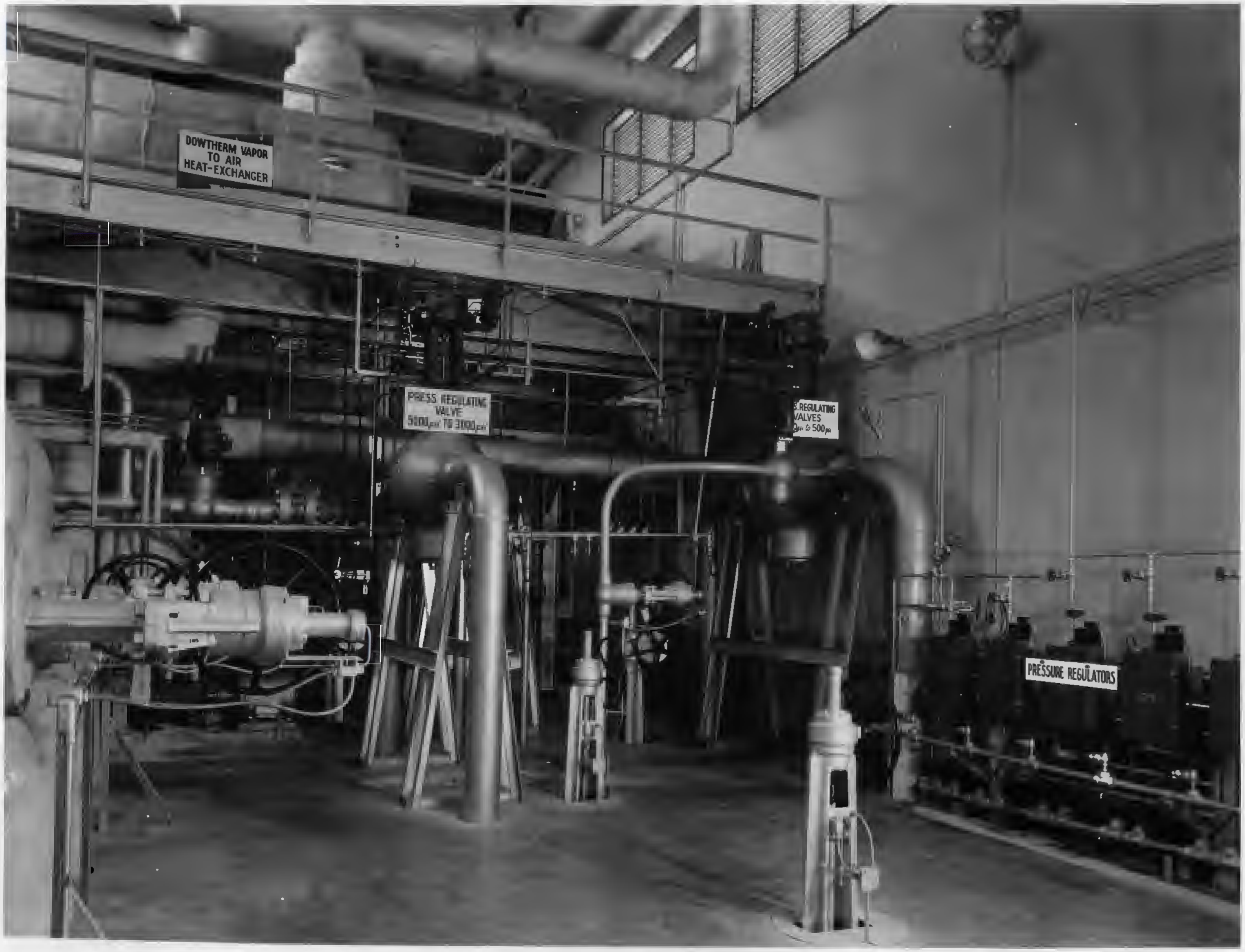
0
AMB.
AT ALT.
-200
LIQ. AIR
-400

LAL 80147

LAL 80138



Second Station - Heater Room of Gas Dynamics Laboratory

NACA
LAL 80142

DEMONSTRATION OF AERODYNAMIC HEAT-TRANSFER EQUIPMENT

The previous speaker has discussed the problem of aerodynamic heating of a missile or aircraft flying at supersonic speeds. I would like to show you the research equipment that will be used to obtain heat-transfer data. The installation of this equipment is not completed; however, we can demonstrate how it will work.

This is a Mach number 4 supersonic blowdown jet. The large silver "bell" is the settling chamber. The nozzle blocks are mounted within these sidewalls, and the model is mounted at the rear in the test section. We will look at the model in a moment. These black hoses carry cooling water to the sidewalls and nozzle blocks.

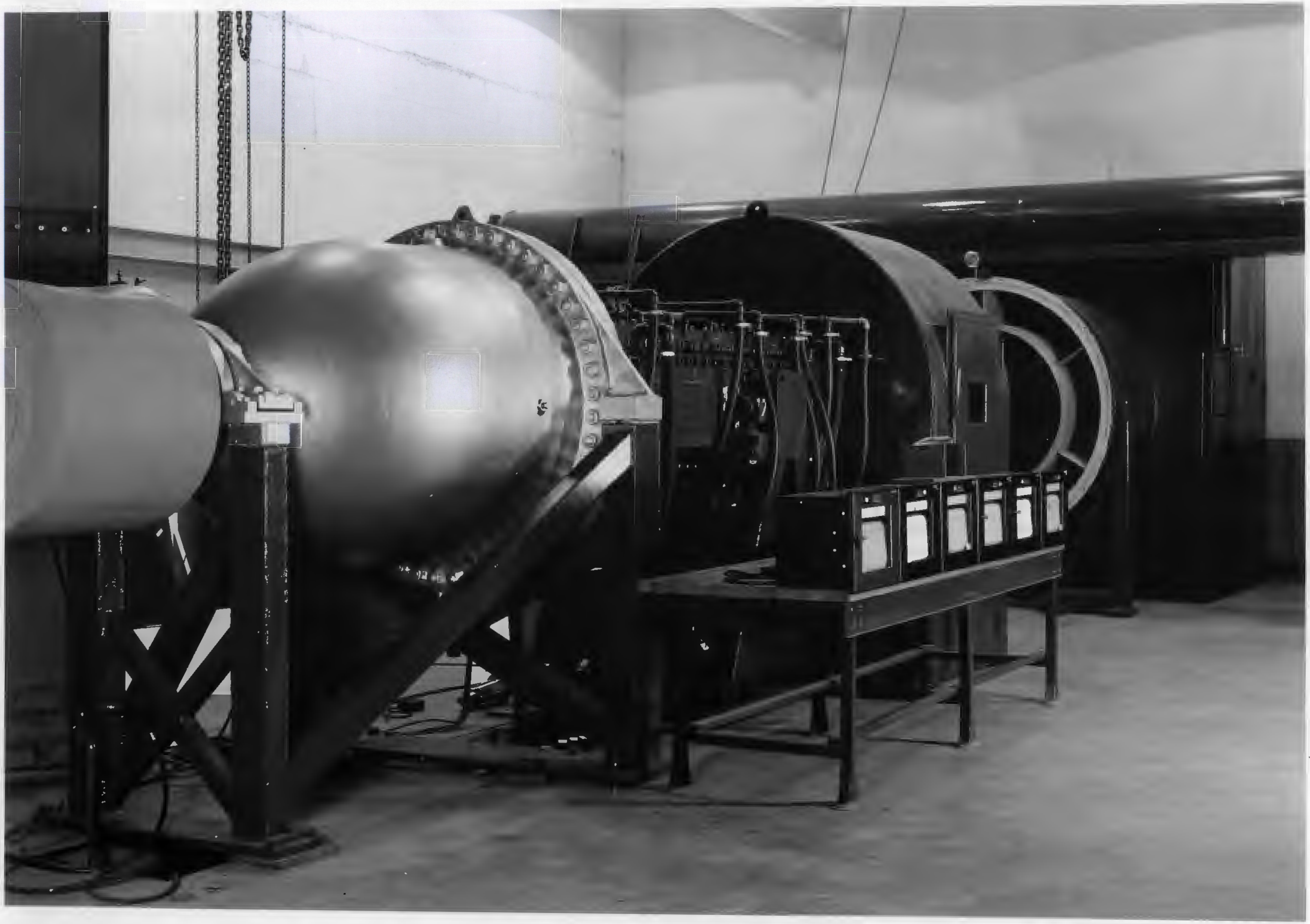
The model we are testing consists of a thin stainless steel shell and has thermocouples mounted along its surface to measure the skin temperature. The positions along the model where the skin temperature will be measured are marked. This is a model before installation of the thermocouples. The instruments you see on this table record the time history of the skin temperature. Now, if you will step around to the rear of the wind tunnel we will show you a model mounted in the test region.

Since we are using air heated to 1000° F, the piping system must be preheated in order to avoid excessive stresses caused by large temperature differences in the pipes. Preheating is done by blowing a small amount of hot air through the piping system and wind tunnel for several hours prior to making a test. The model must be mounted in the wind tunnel during this preheating period; consequently, a new technique had to be developed to keep the model cool. The method decided upon consists of a water-cooled shroud that encloses the model as you see here. This shroud remains in position around the model during the preheating period just as it is now. When a test is to be made, supersonic flow is established in the wind tunnel then the shroud is retracted, leaving the model exposed to the air stream. The model skin temperature immediately starts rising due to the aerodynamic heating of the boundary layer and this temperature rise is recorded on the instrumentation you have seen.

By measuring the time history of the skin temperature we are able to determine the magnitude of the rate of heat transfer. The data we obtain in this manner can be used to evaluate theoretical methods, or to calculate the heat transfer of a free-flight missile.

The technique we are using enables us to simulate the free flight case in that we have a cool model whose skin temperature increases with time exactly like the free-flight missile does. These tests being in a wind tunnel are made with more closely controlled test conditions than can be achieved in flight tests.

LAL 80139



Third Station - High-Temperature Jet Room

Model Cooling Apparatus for Transient Heat-Transfer Studies

LAL 80143

Before discussing the work done in this particular jet room, I would like to describe an investigation which is being conducted in one of the NACA's large, continuous operation, supersonic wind tunnels.

SURFACE ROUGHNESS AT SUPERSONIC SPEEDS

The previous speakers have discussed some of the fundamental supersonic flow problems which can be investigated effectively through use of the jet equipment of this new gas dynamics laboratory. There are certain types of basic research, however, for which the small supersonic jet equipment is not suited. For example, in the study of surface roughness effects on boundary-layer development and surface friction, large test models must be used in order to permit proper simulation and control of the minute roughness elements. This type of basic research thus requires the use of a large supersonic wind tunnel facility such as the Langley 4- by 4-foot supersonic pressure tunnel. One of the current projects in this large tunnel is an investigation of surface roughness effects at Mach numbers ranging from 1.4 to 2.2.

This supersonic roughness investigation is aimed at providing answers to the following broad questions:

First: What roughness tolerances must be maintained to permit the existence of low-drag, low-heat-transfer laminar boundary layers in supersonic flight?

Second: For operating conditions where laminar flow is not attainable and the boundary layer is largely turbulent, what are the allowable roughness limits?

We will confine the present discussion largely to results pertaining to the second question - that is: results of roughness studies for turbulent boundary layers.

Early low-speed research showed that skin-friction coefficients for a given surface condition varied, depending on air density, air speed, body length, and viscosity. It was found, however, that the different values of skin friction could be correlated or brought into agreement if plotted against a single term containing these four quantities - the product of density, velocity, and body length, divided by viscosity. This single term, called the Reynolds number, is used on this first chart to provide a correlation for the values of skin-friction drag obtained on surfaces of different roughness at subsonic speeds. For any given size of roughness, a critical Reynolds number exists below which the roughness causes no appreciable drag increment above the skin friction of a smooth body. Beyond this critical Reynolds number, a drag increment is found which becomes larger as the Reynolds number increases. (Point out two different roughness curves.) This drag increment at the higher Reynolds numbers arises from the direct drag of the roughness particles as they protrude into the boundary layer.

Up to the present time there has been little or no information available concerning the effect of roughness at supersonic speeds. Some of the first results from the current roughness investigation in the 4-foot supersonic pressure tunnel are presented in this next chart. The test body is shown here with one of the larger sizes of surface roughness, 5 ten-thousandths-inch ridges made by lathe tool marks. These new supersonic results, obtained at M = 1.6, indicate the same general trend of roughness effects as shown on the first chart for subsonic conditions. A critical Reynolds number was found beyond which roughness began to cause an appreciable drag increment over the turbulent drag of a comparable smooth body. It is noteworthy that the value of the critical Reynolds number in this supersonic test appears to be about the same as for the older subsonic results. This suggests that the considerable volume of existing subsonic roughness data can be used as a guide in estimating allowable roughness limits for supersonic applications.

Let us now consider the drag penalties that would be incurred if this particular body with the same 5 ten-thousandths-inch roughness were flown at a fixed speed, M = 1.6, at different altitudes. In this chart the drag increment due to roughness is expressed as a percentage of the total drag of the corresponding smooth body with a turbulent boundary layer. The total drag is made up of forebody pressure drag, base drag, and skin friction. The bar graph shows the proportion of the component drags to the total drag at 6,500 feet altitude. At low altitudes, where the Reynolds numbers are very high, the total drag is seen to be increased by about 23 percent. As the altitude increases, the Reynolds number decreases due to the decreasing density of the air, and the roughness penalty diminishes. Thus, we are simply coming back along this curve to a lower Reynolds number. (Chart Number 2.) Beyond 50 thousand feet, roughness of this magnitude would cause no drag penalty for this 50-inch body. If this regime were of primary interest operationally, there would be no need for refinement of production processes to make the surface any smoother.

Roughness effects are probably of greater concern for a larger body more representative of a long-range missile such as the 40-foot body shown on the last chart. If this configuration were operated under equilibrium temperature conditions at M = 2.5 and 50,000 feet altitude, the boundary layer would be largely turbulent even if the surface were perfectly smooth. It has been estimated from the supersonic data previously shown that the surface of this missile could have a roughness of 0.0003 inches without an increase in drag due to roughness. A sample section of the surface of such a missile is shown here. A surface of this quality can be obtained without special finish grinding or buffing operations. The data which have been presented today are a very small part of the total research needed to evaluate roughness effects on laminar and turbulent boundary layers at supersonic speeds. Further studies must be made at higher Mach numbers; the effects of roughness distribution have to be evaluated, and single roughness elements such as ridges and bumps must be tested.

NACA
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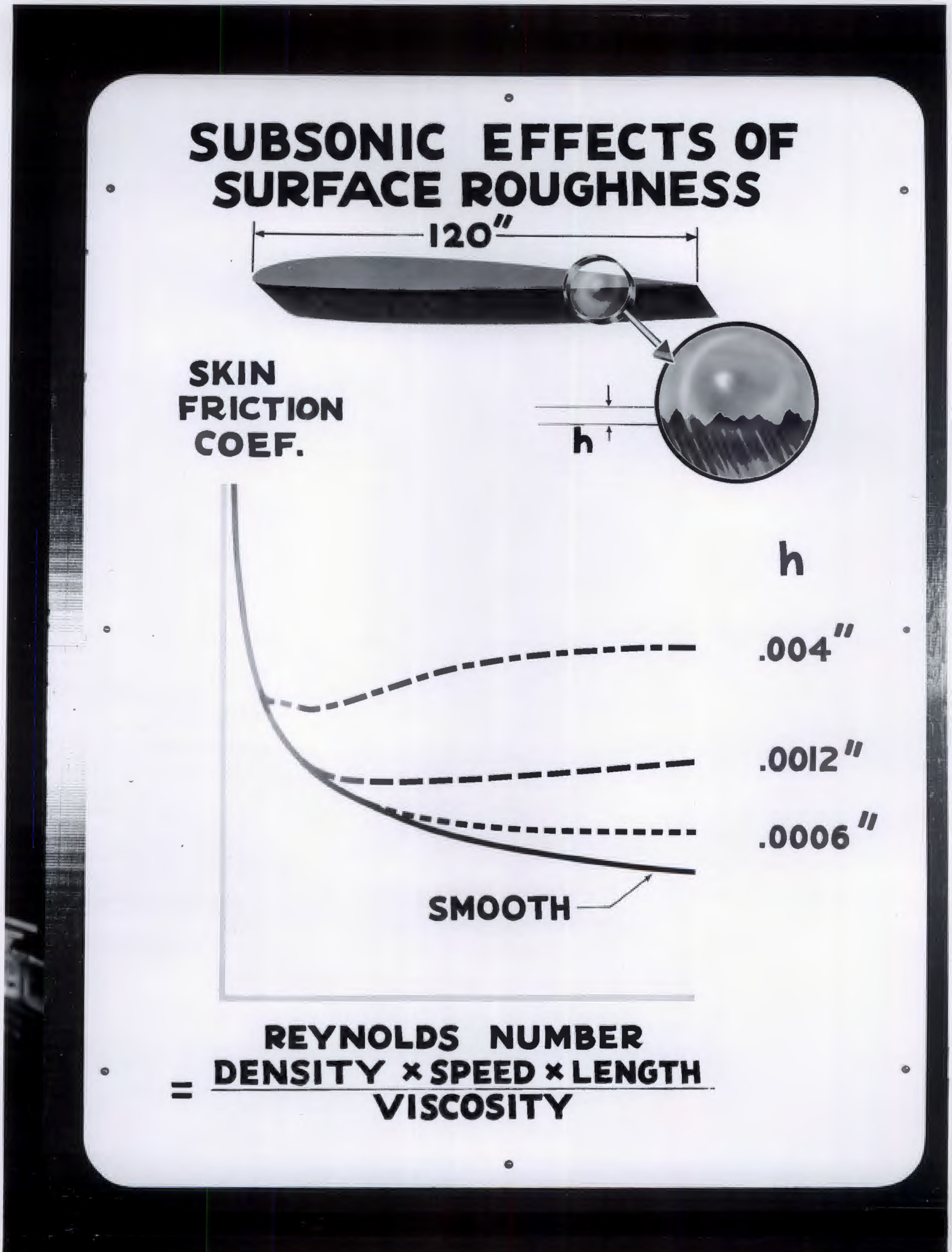
Fourth Station - Display for Talks on Surface Roughness and Shock-Boundary-Layer Separation


SUBSONIC EFFECTS OF SURFACE ROUGHNESS
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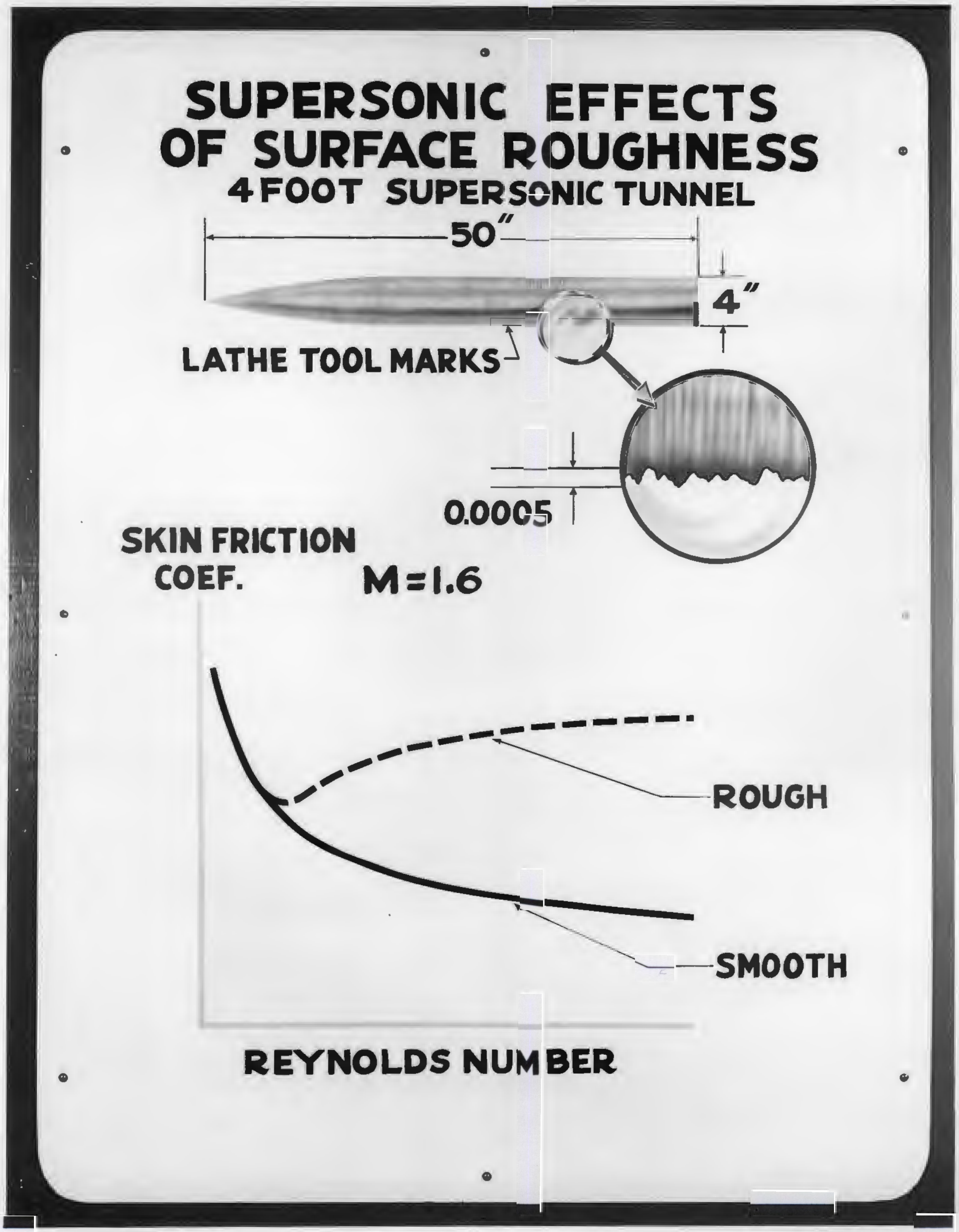
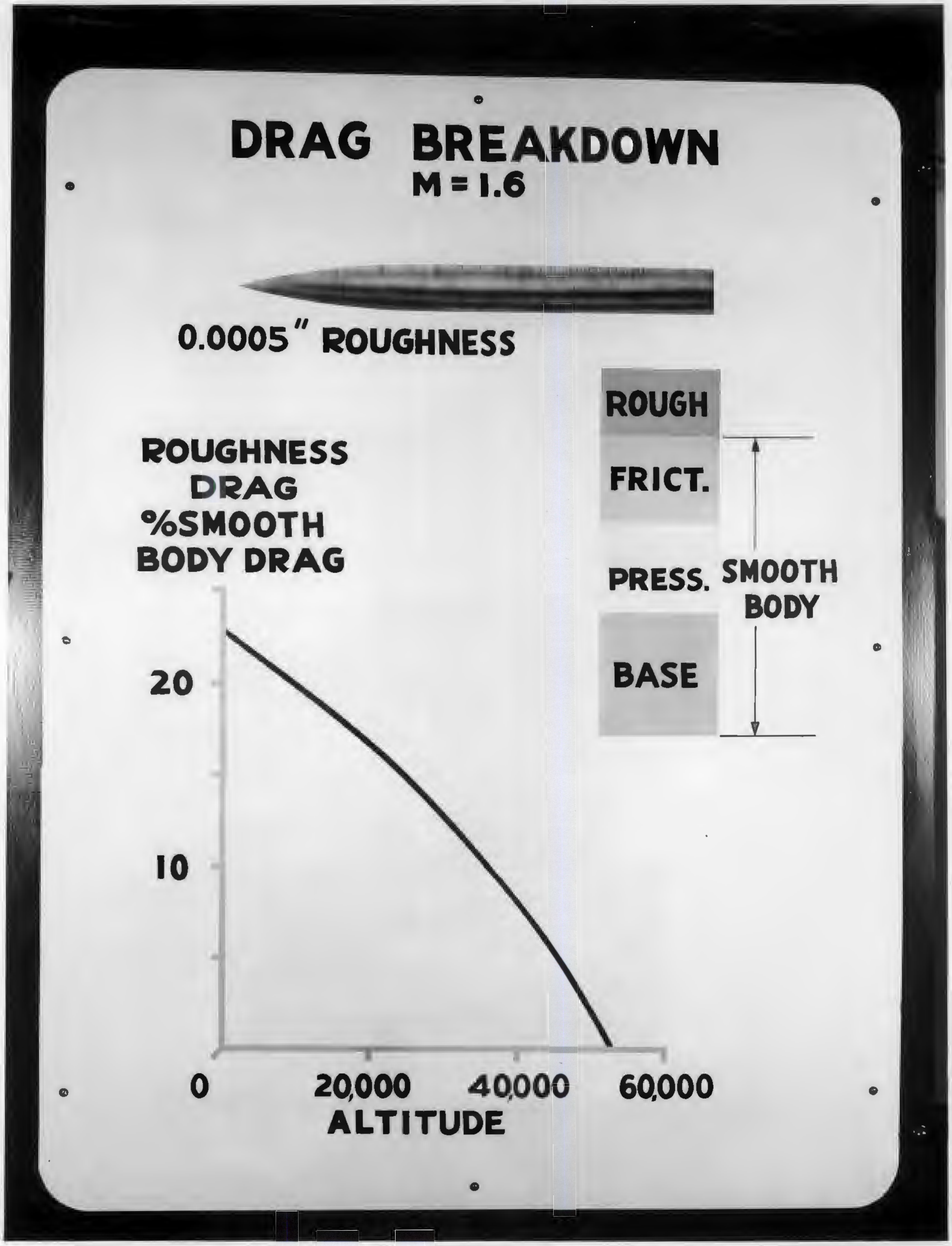
.0012"
.0006"
SMOOTH
REYNOLDS NUMBER
= DENSITY × SPEED × LENGTH / VISCOSITY



NACA
LAL 80144


DRAG BREAKDOWN
M = 1.6
0.0005" ROUGHNESS
ROUGH
FRICT.
PRESS.
BASE
SMOOTH BODY
ROUGHNESS DRAG %SMOOTH BODY DRAG
20
10
0
20,000
40,000
60,000
ALTITUDE

NACA LAL 79874

INTERACTION OF A BOUNDARY LAYER AND A SHOCK WAVE

The facility in this room is a low-density jet. Whereas most of the tunnels in the Gas Dynamics laboratory are concerned with high pressure-high Reynolds number flow, this jet is designed for investigations in the low density, low pressure regime.

The reduced pressure is gained by exhausting the jet into the large vacuum sphere which is located just outside the room. This type of tunnel is known as a "suck-down tunnel", since the air is literally sucked, or drawn, through the tunnel. Since at the present time we are running at the moderate Mach number of 2.9 and at low pressure, we are taking the air from the room for convenience. However, the tunnel is easily connected to the high pressure line which you see. When the high pressure line is connected, pressure ratios as high as 10,000 to 1 can be achieved across the tunnel. These high pressure ratios are necessary to drive the tunnel at the high supersonic speeds. Mach numbers as high as 9 are possible with this tunnel. This first chart shows the wide range of Mach number and pressure which this tunnel can reproduce.

The pressure is shown as equivalent altitude; that is, the altitude at which the pressure is that of the tunnel test section. The low altitude or high pressure limit is determined by the maximum pressure available to us in the high pressure line. The low pressure-high altitude limit is determined by the vacuum pumps' capacity to evacuate the sphere. So much for the facility itself.

DESCRIPTION OF INVESTIGATION

One of the major investigations being done in the field of supersonic aerodynamics is that of the problem of the interaction between a shock wave and a boundary layer. The understanding of this problem presents the key to the design of a great many aerodynamic shapes. In most cases it is desired that the shock wave not be of sufficient strength to separate, or detach the boundary layer from the surface. The particular portion of the overall investigation with which this study is concerned is the experimental evaluation of the strength of the shock wave needed to separate any boundary layer. This being a fundamental investigation the experiments were confined to the simple case of the boundary layer on a flat plate. Any other boundary layer must then be evaluated in terms of its equivalent on a flat plate.

As an example of the application of this shock wave-boundary layer interaction see what happens on a supersonic wing section. You see here on Chart number 2 a wing section with its control surface deflected. The theoretical shock wave pattern is similar to that shown. A shock wave, which represents a jump in pressure, is necessary on this side and an expansion, a pressure fall, on the other side, in order to turn the flow parallel to the control surface. If the boundary layer flows smoothly over the surface this theoretical picture is very close to the actual case.

However, if the control angle is too great, the pressure jump across the accompanying shock wave will be too much for the boundary layer to negotiate, and the boundary layer will separate from the surface. In this case the pattern will be similar to the lower diagram. Here the shock wave has moved forward and is followed by a separated, "dead water" region. The result is that the control characteristics are not quite what the designer would anticipate from his theoretical calculation. Let me say again that this is an example. What we are concerned with is this interaction in this small region and the resultant separation which follows it.

Now the results of this fundamental investigation are shown on Chart number 3. This represents the experimental evaluation of strength of shock wave necessary to separate various boundary layers and these results are gathered from data taken from all over the laboratory. This low density jet provides data in the low Reynolds number range.

This is a logrithmetric plot of the R_x, the primary variable of a boundary layer, vs. $\frac{\Delta p}{q}$ which represents the pressure jump associated with any shock wave. These lines, one for laminar boundary layers and one for turbulent boundary layers, represent the dividing lines between the strength of shock waves which will separate the boundary layer and that which will not. For instance, if we have a turbulent boundary layer with a R_x of 10^6, a shock wave with a pressure jump greater than this value will separate that boundary layer. A weaker one will not.

Notice that the turbulent boundary layer is able to withstand a much greater pressure jump than a laminar one.

DEMONSTRATION

What we wish to show you today is an example of this boundary layer separation due to an interaction with a shock wave. The model in the tunnel is a wing section with a movable control surface. We will start the tunnel with the control surface in the neutral position as it is now. Then I will deflect the control slightly and you should be able to see a shock wave originate here at the hinge line. This is the shock wave needed to turn the flow parallel to the control surface. As the control surface is deflected further the shock wave would continue to have its base at the hinge line were it not for separation. However, this boundary layer does separate and the shock wave is pushed forward along the wing surface. Thus by watching the shock wave move forward along the wing you will be able to observe the growth of the separated region. As we run I will point out to you on the schlieren screen the hinge line, the shock wave in which we are interested and the boundary of the separated region which appears as a dark line on the screen. You will be able to see this with the aid of our schlieren system which is an optical device which allows us to see density variation.

30-SECOND RUN

What you have just seen represents one example of separation with an undesired result. This last chart illustrates some other cases.

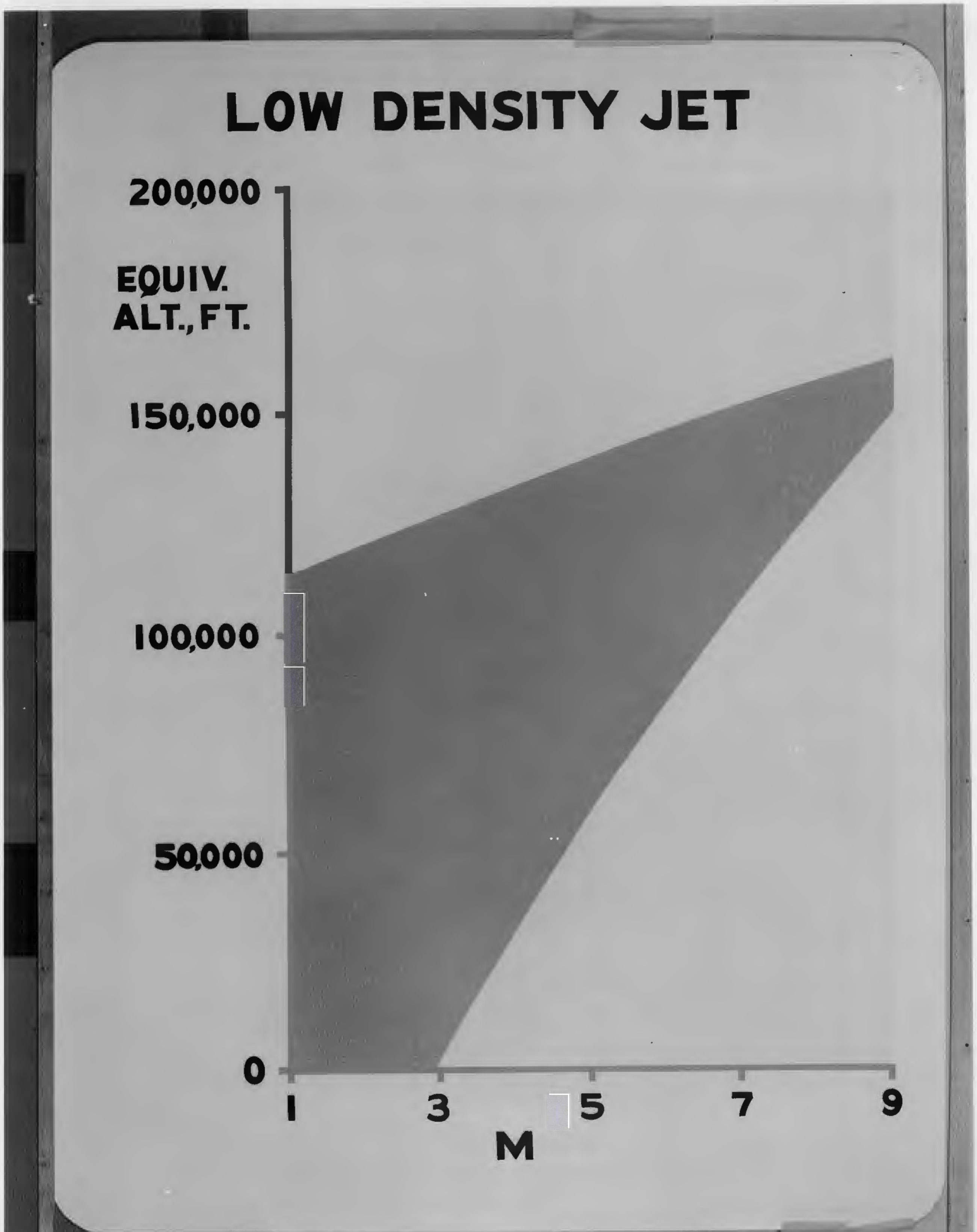
This first example shows a wing section which is travelling at a high subsonic speed. There is a supersonic region on the surface terminated by a shock wave. In this case, the shock wave has separated the boundary layer. It is this separation and not merely the presence of the shock wave alone which is the source of the stability and control problems at transonic speeds. These troubles include loss of lift, loss of control efficiency, pitch up, and buffeting.

This next example is of a supersonic inlet, which consists of a solid central body, with a cowling around it. The job of the inlet is to convert the velocity energy of the flow to pressure energy through a system of shock waves in the annular passage. Separation here means a loss in efficiency with a resultant loss of engine thrust.

In this picture the inlet is being operated at conditions which result in an external shock wave pattern. Separation of this type can result in severe flow oscillations which must be avoided.

This last example shows a cross section through the blades of a supersonic compressor. Here again separation is frequently encountered and results in loss of pressure rise and efficiency.

What we have gained so far from this basic investigation is a knowledge of the strength of shock waves which will separate a given boundary layer, and the results are useful in all the examples shown and many others. However, a complete understanding of the mechanism of separation and the details of the interaction between a shock wave and a boundary layer will require much more intensive research.


LOW DENSITY JET
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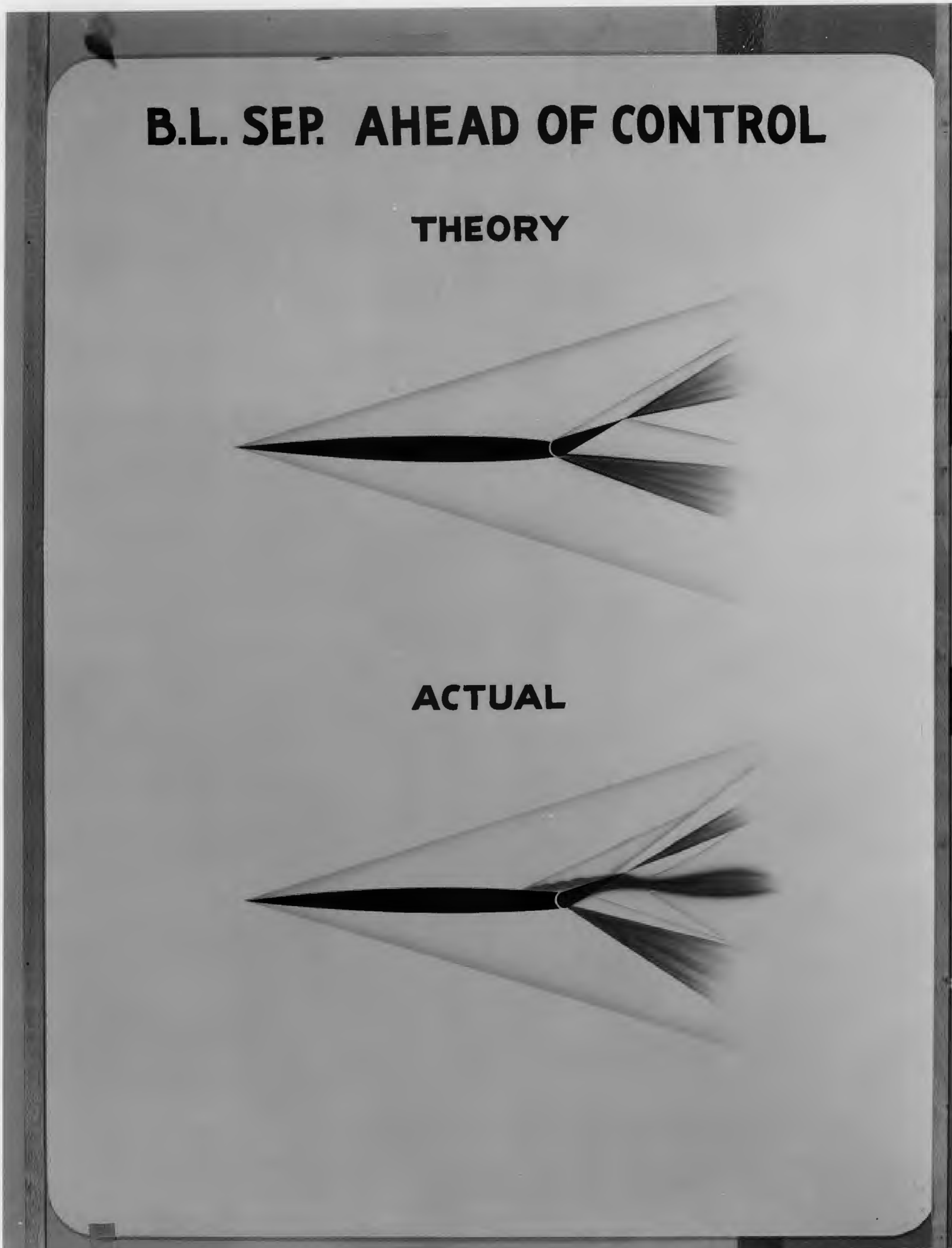
LAL 79875



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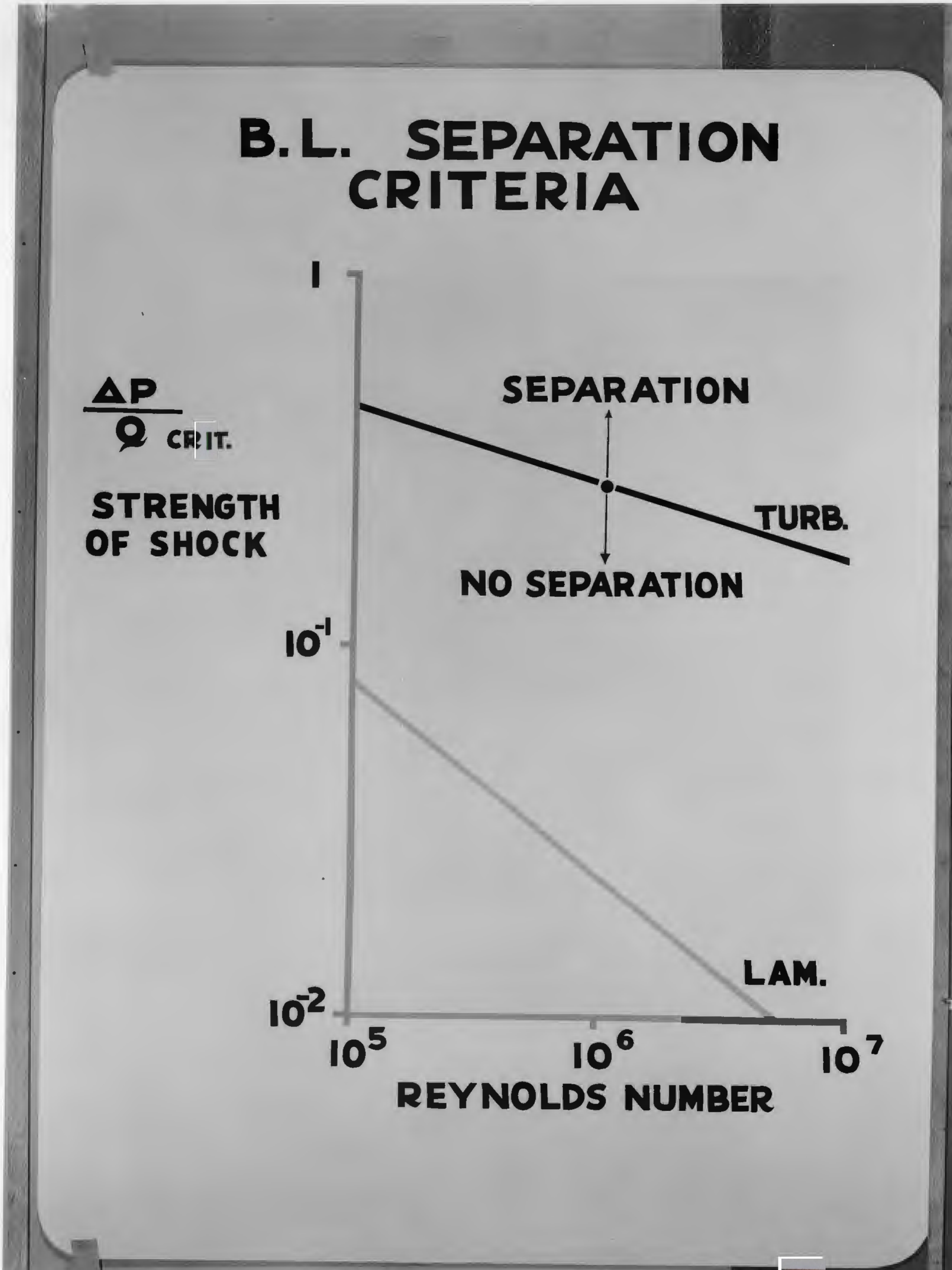
B.L. SEPARATION
CRITERIA
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OF SHOCK
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10⁻²
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TURB.
LAM.
10⁵
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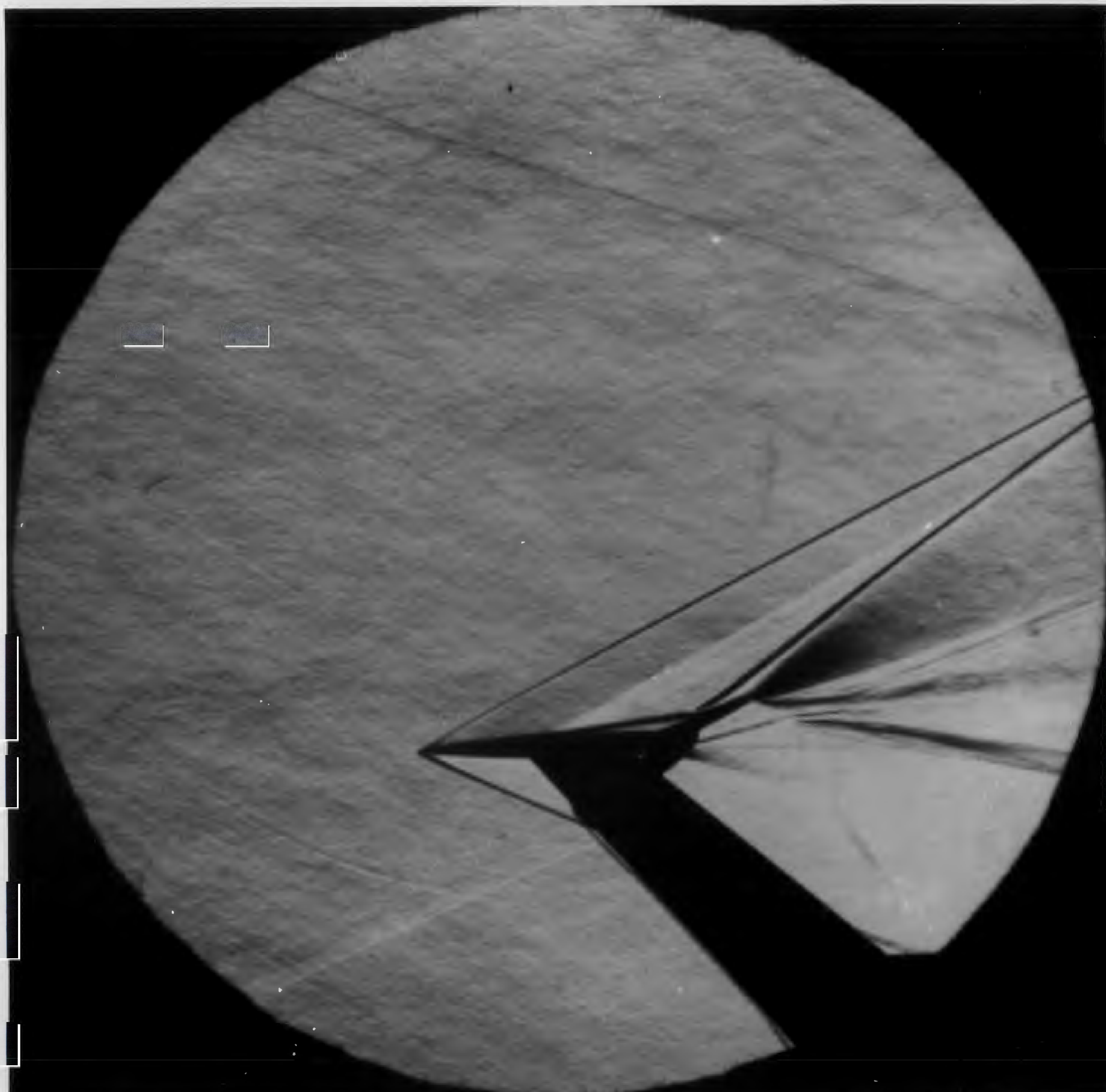
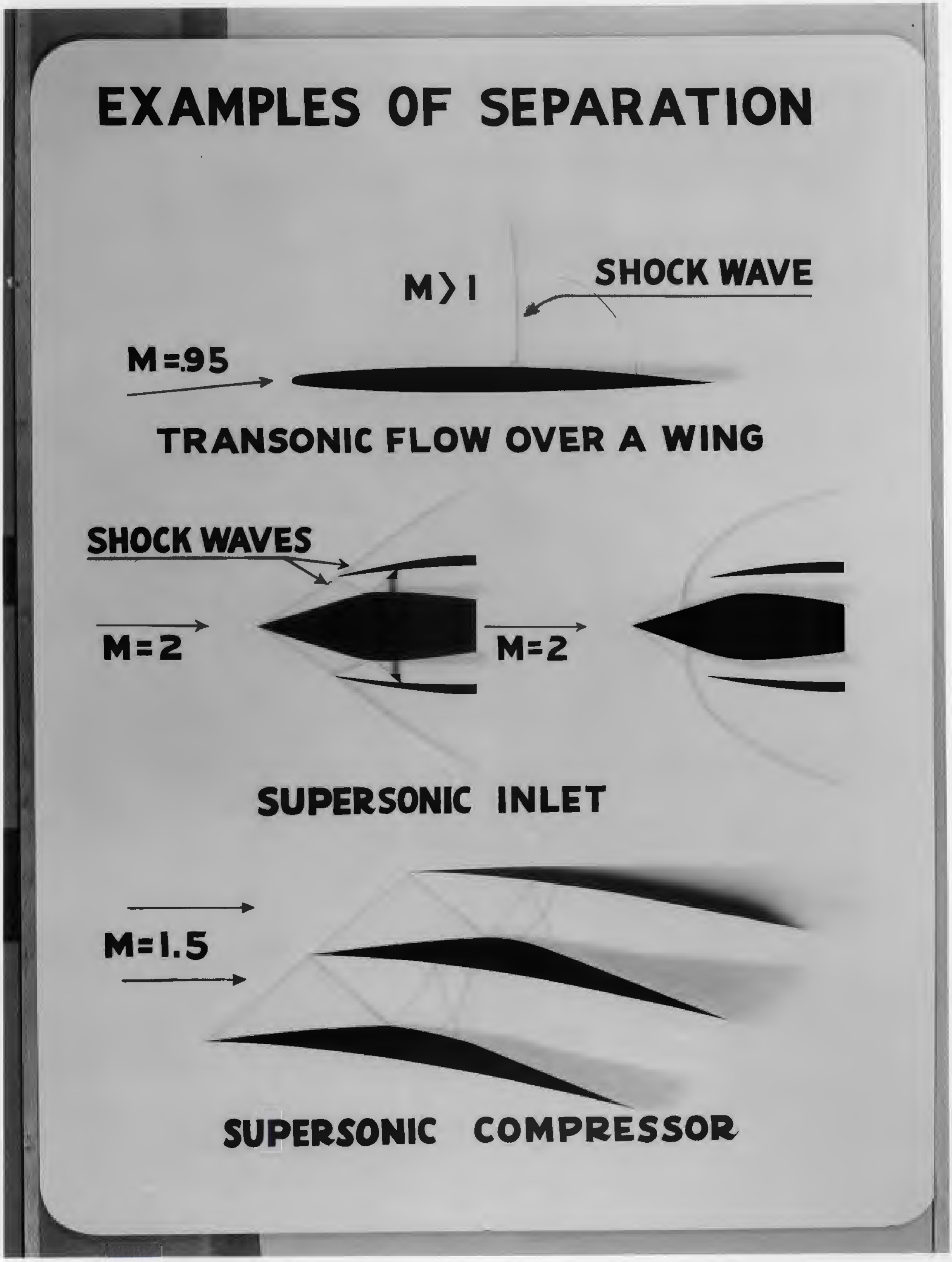
REYNOLDS NUMBER

LAL 79885



Schlieren Photograph Taken During Demonstration of Shock-Boundary-Layer Separation


EXAMPLES OF SEPARATION
M〉1
SHOCK WAVE
M=.95
TRANSONIC FLOW OVER A WING
SHOCK WAVES
M=2
M=2
SUPERSONIC INLET
M=1.5
SUPERSONIC COMPRESSOR