11-INCH HYPERSONIC TUNNEL BIENNIAL INSPECTION TALK 1951

A great deal of effort is being expended in providing research data for missile designs in the Mach number range up to about 3. There are, however, certain applications where much higher maximum Mach numbers are contemplated - Mach numbers in the range of 5 to 10 and even higher. In order to explore the aerodynamic problems at these extremely high speeds, new equipment and techniques have been recently developed by the NACA. One of these facilities is this 11-inch hypersonic tunnel which was put into operation in 1947. The principal elements of this tunnel are shown schematically on this chart (chart 1). Air is stored in a high-pressure tank at 730 psi which is 50 times atmospheric pressure. From the high-pressure tank the air passes through a heater, settling chamber, nozzle, test section, cooler, and vacuum tank. Only this portion of the tunnel can be seen here in this room and the test section is located here. Mach numbers in the range of 5 to 10 are obtainable. If the tunnel were a lower speed tunnel, say for a Mach number of 2, the pressure in the high-pressure tank would only need to be twice the pressure in the vacuum tank. However, at a Mach number of 7, the pressure in the high-pressure tank must be at least 100 times the vacuum tank pressure in order to establish the flow in the nozzle.

The nozzle, in which the air is accelerated to the high Mach number, is shown schematically at the top of the next chart (chart 2). For a Mach number of 7, the air must pass through this very small slit which is only 1/10 inch high. The large increase in area which follows is required to develop a Mach number of 7 in the test section which is 11 inches high. In passing through this nozzle the large changes in area are accompanied by large changes in pressure. Assuming a pressure of 35 times atmospheric pressure in the settling chamber at this point, the pressure in the test section would drop to only 1/100 of atmospheric pressure. This very low pressure creates problems in flow measurement and visualization. The large pressure change is accompanied by a very large temperature change through the nozzle. If the air is initially at room temperature, the temperature will drop along this dashed curve to about 410° F below zero. This is well into the shaded region in which the air will liquefy. This liquefaction is a problem peculiar to wind-tunnel operation. A heater which could provide air temperatures up to 800° F was therefore included so that the formation of liquid air particles could be either studied or entirely avoided. With the temperature in the settling chamber of 700° F the temperature will drop along the solid curve and will remain above the liquid air zone. Very shortly after the tunnel was put into operation it was determined that liquid air particles actually were present when the air was not heated. It was also found that liquefaction could be avoided by use of the heater. All aerodynamic tests, of course, are made with the heater in operation.

We are now preparing to make a run without heat in which you will be able to observe the presence of the liquid air particles in the tunnel. The sound you have just heard is the opening of the valve to the vacuum tank. One of the methods that we have developed to show the presence of liquid air particles utilizes a strong beam of light passed through the test section (beam on and lights off). Without the flow, reflections are seen where the light beam passes through the test section windows. The light beam cannot be seen in the test section. This is exactly the same appearance as obtained in the heated condition. Without heat, the liquid air particles in the stream scatter light making it visible, giving the familiar appearance of a strong light beam passing through an ordinary fog. It must be emphasized that the air is dry and that the fog that will be seen is a fog of liquid air particles and not a fog of water particles.

The tunnel is still being prepared for the demonstration run. The starting sequence employed consists of first opening the valve to the vacuum tank exposing the nozzle and test section to the low pressure. Next, the valve just downstream of the heater will be opened exposing the heater to the low pressure. It should be pointed out that the heater will be cold during this run in order to maintain the air temperature entering the nozzle at about room temperature. The run will finally be started by opening the valve just upstream of the heater. The pressure in the settling chamber will be maintained at approximately 30 times atmospheric pressure.

The run will now begin. Be sure to direct your attention to the test section.

<u>After demonstration</u>

Valid aerodynamic tests cannot be made with the liquid air particles in the stream which you have just observed. Since they are not present in flight, all aerodynamic tests are made in the heated condition in which liquefaction is entirely avoided. The next speaker, Mr.   , will summarize some of the results obtained in this facility.

Speakers:

(1) Charles H. McLellan
(2) Ralph D. Cooper
(3) Jim A. Penland

Second Half of Talk

One problem which we have investigated in this tunnel is development of the boundary layer at hypersonic speeds. The boundary layer is the layer of air adjacent to a moving surface which is dragged along with the surface due to the action of viscosity. In the next chart (chart 3), the great increase in boundary-layer thickness at hypersonic speeds is shown. Consider a wing flying through the stratosphere where the air temperature is -67° F, at a Mach number of 1.5 and also at a Mach number of 7. The shaded areas above and below the wing indicate the effective boundary layer. We find that at the lower Mach number the temperature rise through the boundary layer is just beginning to become important as the air temperature rises from -67° F at the outer extremity of the boundary layer to 110° F next to the surface of the model. At a Mach number of 7, the boundary-layer temperature increases from -67° F to 3,800° F near the model surface. This temperature is the temperature of the boundary-layer air near the surface and not the wing surface temperature. As discussed elsewhere during this inspection the wing surface temperature may be much lower. Associated with this high temperature there is approximately an eight-fold increase in effective boundary-layer thickness at the high Mach number. This thick boundary layer, in effect, changes the shape of the surface on which it forms so that the pressures on the surface are altered. At the bottom of this chart (chart 3), a thin wing is shown at a small angle of attack. The dashed line represents the pressure distribution that would be obtained on the bottom surface of the wing if no boundary layer were present while the solid line is that which is actually obtained in the presence of the boundary layer, the difference between the two being, of course, the effect of the boundary layer in altering the profile shape. This effect is of considerable importance at a Mach number of 7 on a thin wing and will assume even greater importance as the Mach number is increased still further.

In order to compare the nature of the flow at hypersonic speeds with the more familiar flows at lower speeds, the next chart (chart 4) has been prepared. At the left of the chart are sketches of the flow about wings in three speed ranges. At low subsonic speeds, at Mach numbers of the order of 3/10, no large disturbances are present in the flow. At a Mach number of 1.5 shocks are present at the leading and trailing edge. At a Mach number of 7 the leading-edge shocks are swept back close to the model. In the center of the chart pressure distributions are presented over the airfoils. The pressures are represented by the arrows. It can be seen from these distributions that at low speeds most of the lift is derived from the upper surface of the wing while at moderate supersonic speeds more of the lift is obtained from the lower surface. At hypersonic speeds, most of the lift is derived from the lower surface. If the Mach number were further increased this trend would continue until virtually all the lift was carried on the lower surface. To the right of this chart (chart 4) perspective views of a wing are shown for the three speed ranges. The blue shaded areas indicate the portion of the wings which are influenced by flows about the wing tips. At subsonic speeds, the flow about the wing tips influences the characteristics of all the elements of the wing having its greatest influence near the tips. Thus, the lifting efficiency of all the elements of the wing are adversely affected. In order to minimize this effect, large wing spans are used at subsonic speeds. At moderate supersonic speeds, disturbances from the tips cannot influence the flow ahead of these boundaries. The zone ahead of these boundaries at the center of the wing is undisturbed by the tip flow and has the same high lifting efficiency of a wing of infinite span. At hypersonic speeds, the boundaries of the tip disturbances are swept backward for the same reason that the leading-edge shocks are swept back. Thus, the area influenced by the tip is further decreased, leaving most of the wing area with a high lifting efficiency. Wings of small span in comparison to the chord can therefore be used without significantly reducing the lifting efficiency of the wing.

In the next chart (chart 5), the maximum lift-drag ratio, which is an index of the aerodynamic efficiency, is plotted against Mach number. The top curve is for a wing of infinite span with skin friction. This curve shows that at a Mach number of 1.5 the maximum lift-drag ratio is about 9 which decreases to about 6 at a Mach number of 7. The dashed curve is for a wing having a span equal to its chord. At low supersonic Mach numbers the lift-drag ratio is decreased considerably by the tip effects. As the Mach number is increased, the lift-drag ratio approaches that for the infinite span wing indicating that considerably lower span wings can be effectively used without seriously altering the aerodynamic efficiency. Also included on this chart is the lift-drag ratio variation of a typical body. Unlike the wings, the lift-drag ratio increases with Mach number indicating that at hypersonic speeds more of the lift can be efficiently carried on the body.

Up to this point we have considered Mach numbers of the order of 7. Here the air temperatures about the body are high - they are not high enough, however, to cause any significant change in the molecular structure of the air. At higher flight Mach numbers, that is, in the range above 10, the temperatures can become enormous becoming 10,000° F or greater, and can cause marked changes in the structure of the air passing about a body. Such high temperatures are not obtainable in wind tunnels but they can be realized by using ballistic techniques. In order to get some idea of the characteristics of these high temperature flows, small-scale exploratory ballistic tests are being made at this laboratory. By firing from a modified gun, a small model similar to this through a stationary gas, a velocity of about 7,000 ft. per sec. can be obtained. In air at room temperature this corresponds to a Mach number of 6.5; however, if some other gas which has a low speed of sound is used, a Mach number of about 11 can be reached. By cooling the gas in order to still further reduce the speed of sound a Mach number of 17 has been obtained using a small sphere similar to the one mounted on this cartridge case.

This (chart 6) is a photograph of the flow about a conical-nosed body traveling through Xenon gas at a Mach number of 11. The gas temperature here behind the shock wave is calculated to be 19,000° F. This is high enough to cause significant changes in the molecular structure of the gas which can be seen visually because of the emission of considerable light as shown by the light area at the nose of the model.

Preliminary analysis of data obtained in exploratory tests of this kind indicates that the drag is not greatly affected as a result of these changes in the structure of the gas; however, there is an indication that the rate of heat transfer to the body is considerably increased. That is, the body temperature in flows of this type will increase more rapidly than expected from experience with ordinary air flow.

Some of the models used in the ballistic tests are shown here and a few of the models used in the 11-inch hypersonic tunnel such as this highly swept wing are shown here.

Speakers:

(1) Mitchel H. Bertram
(2) John A. Moore
(3) Alexander Sabol

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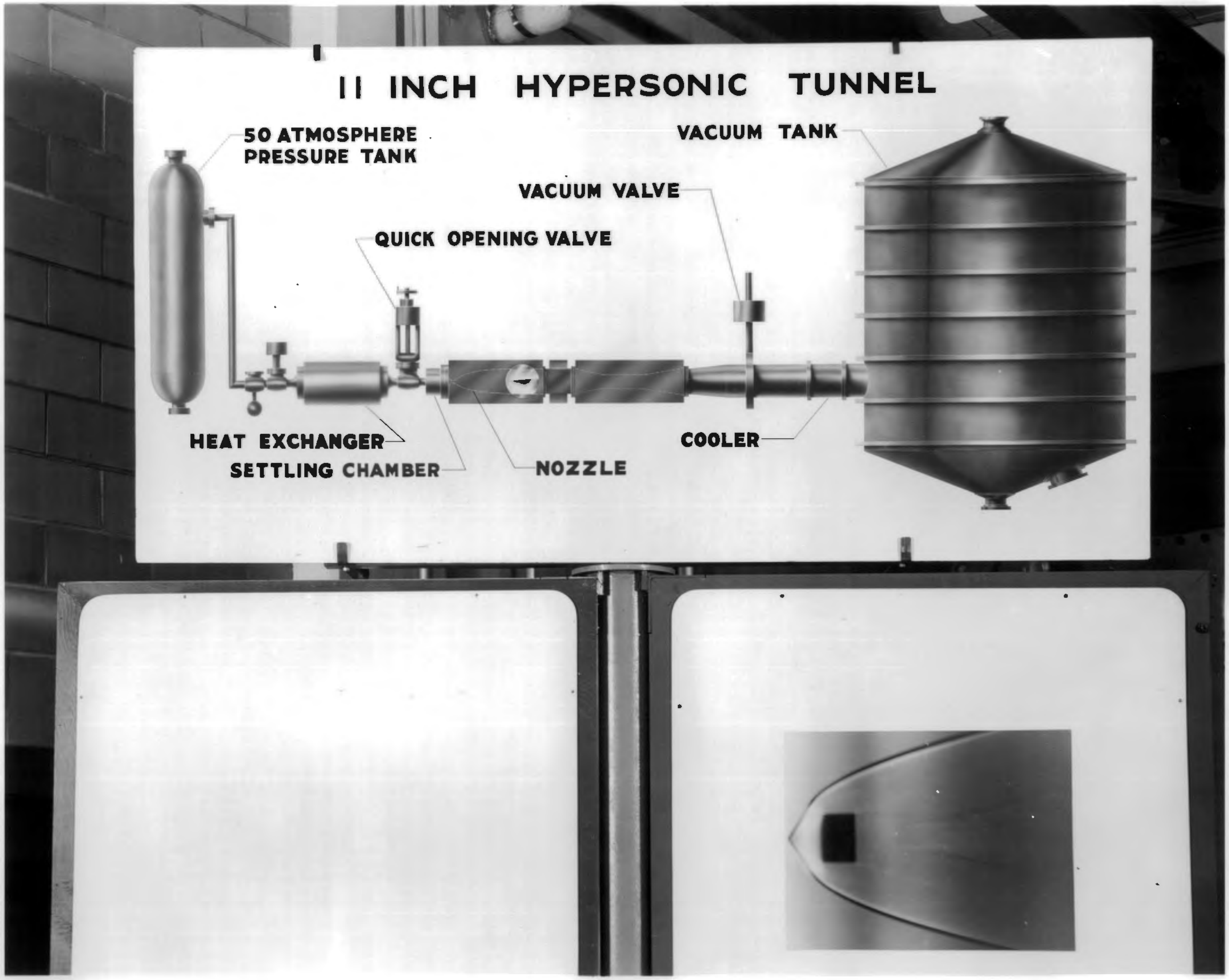

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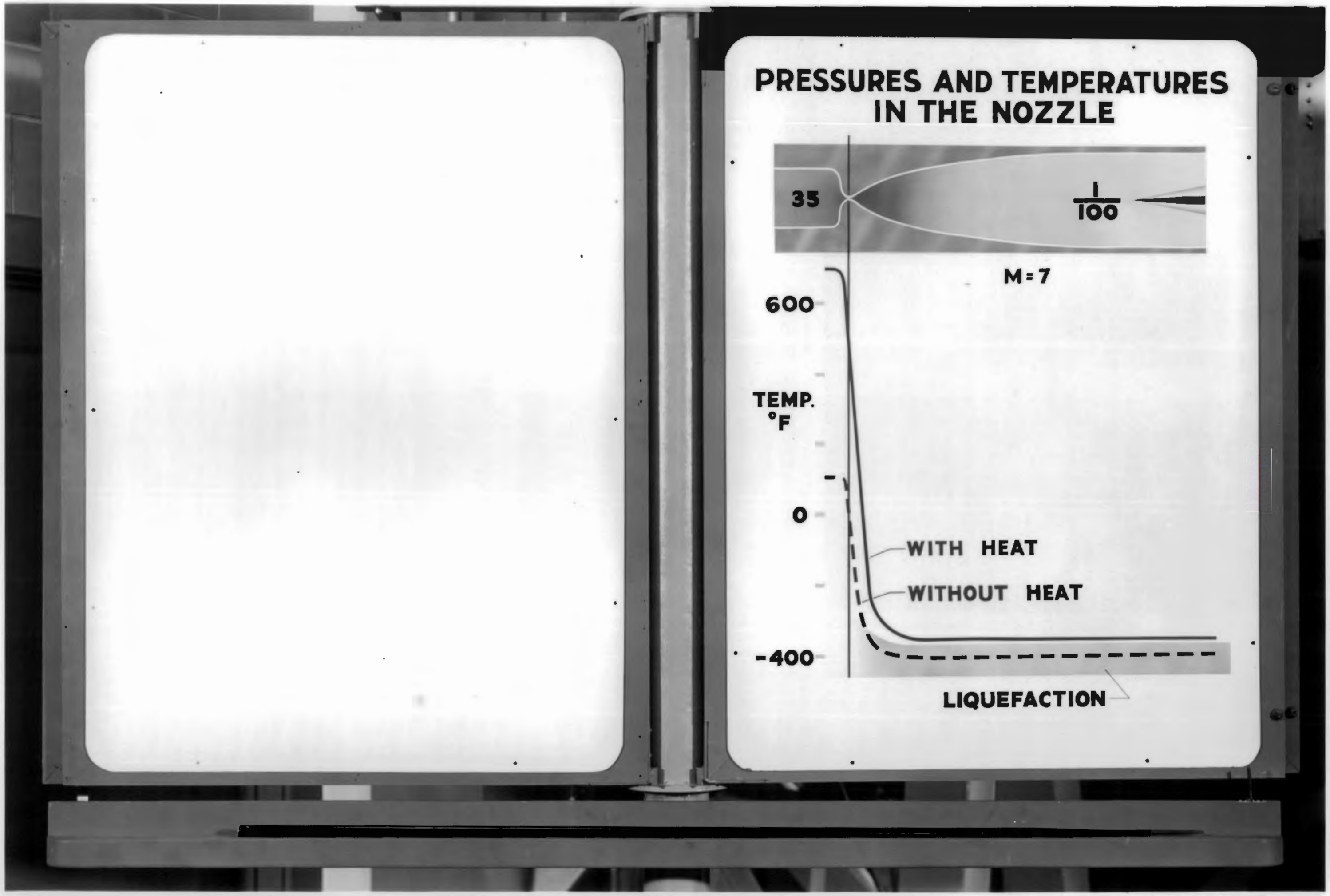

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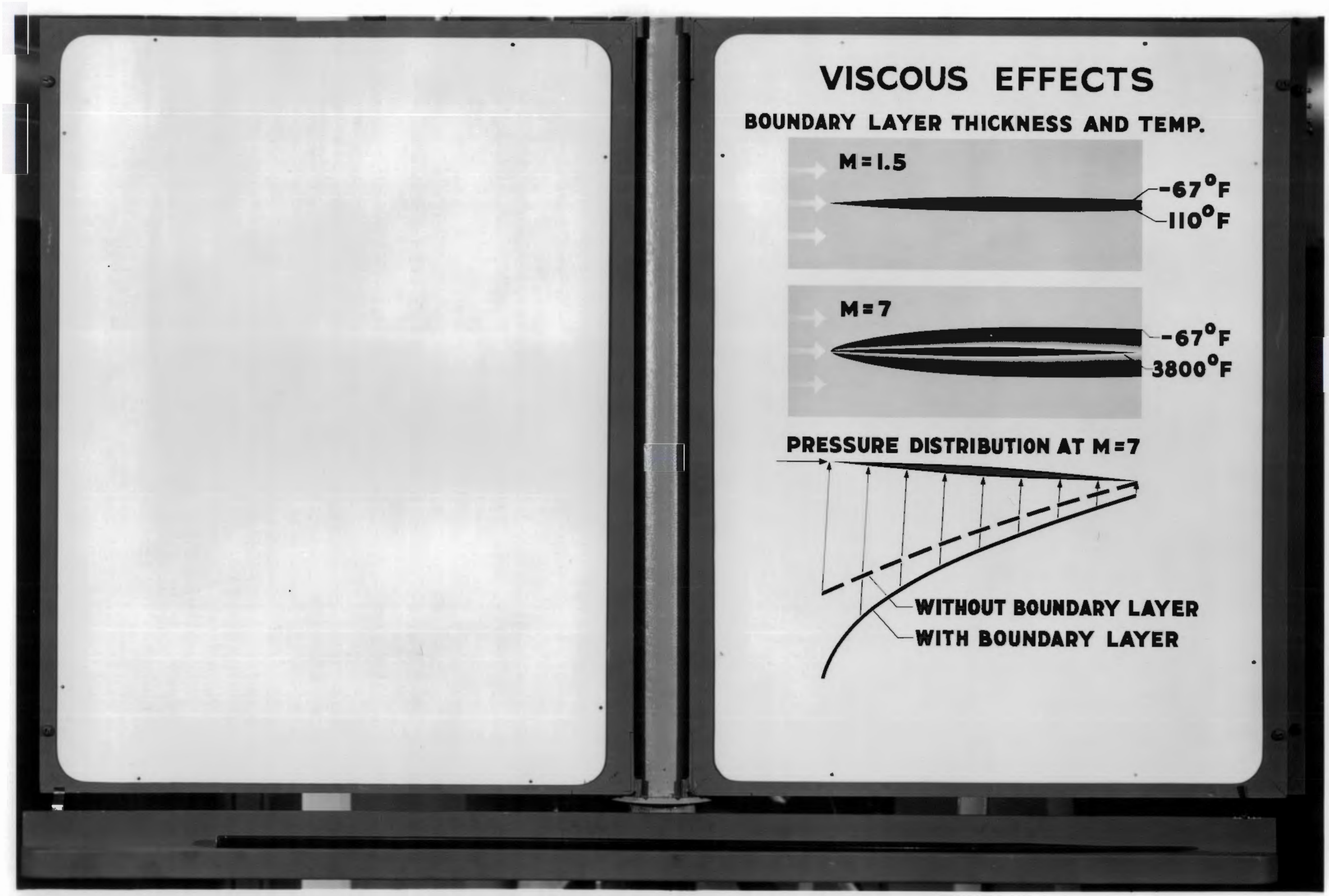

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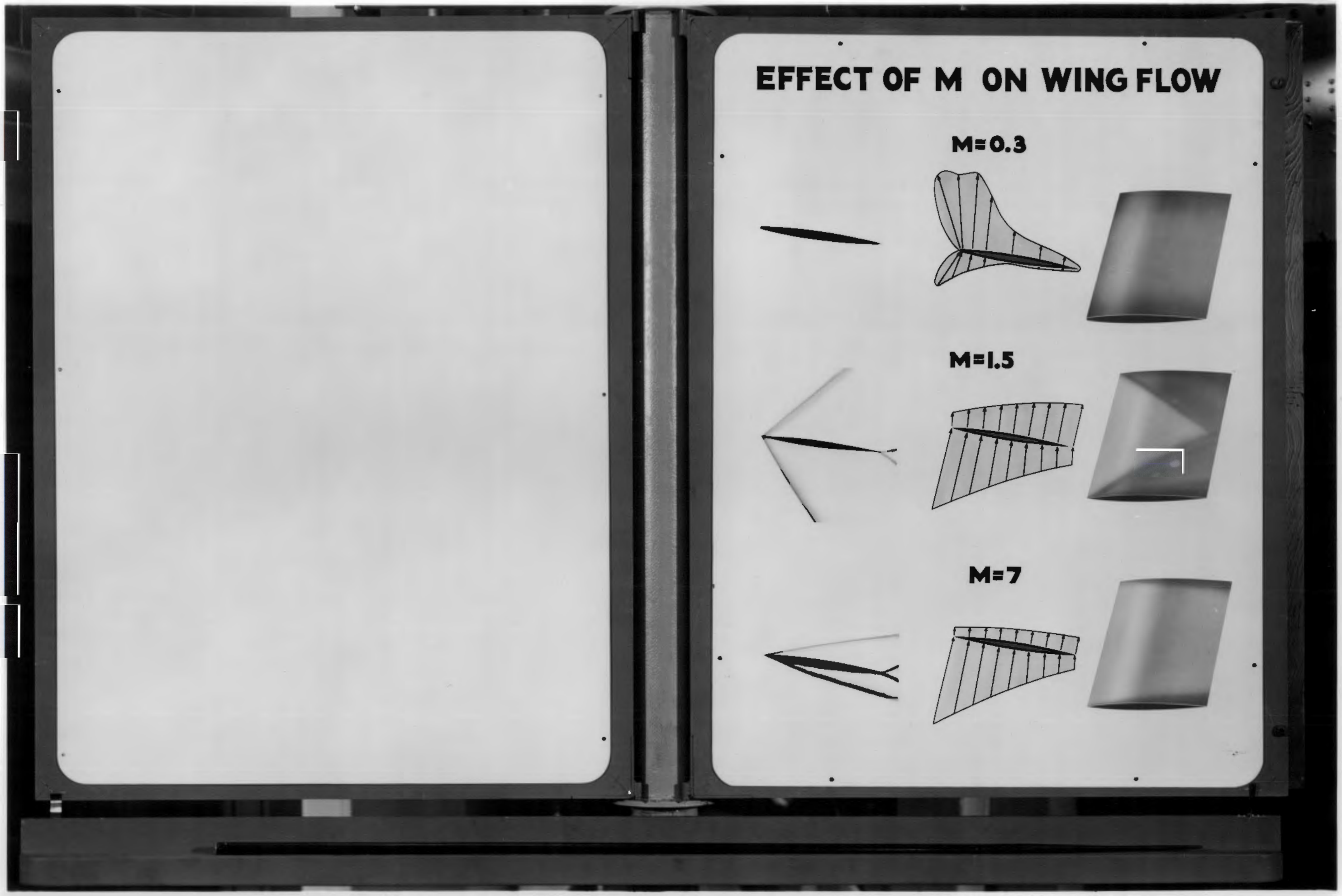

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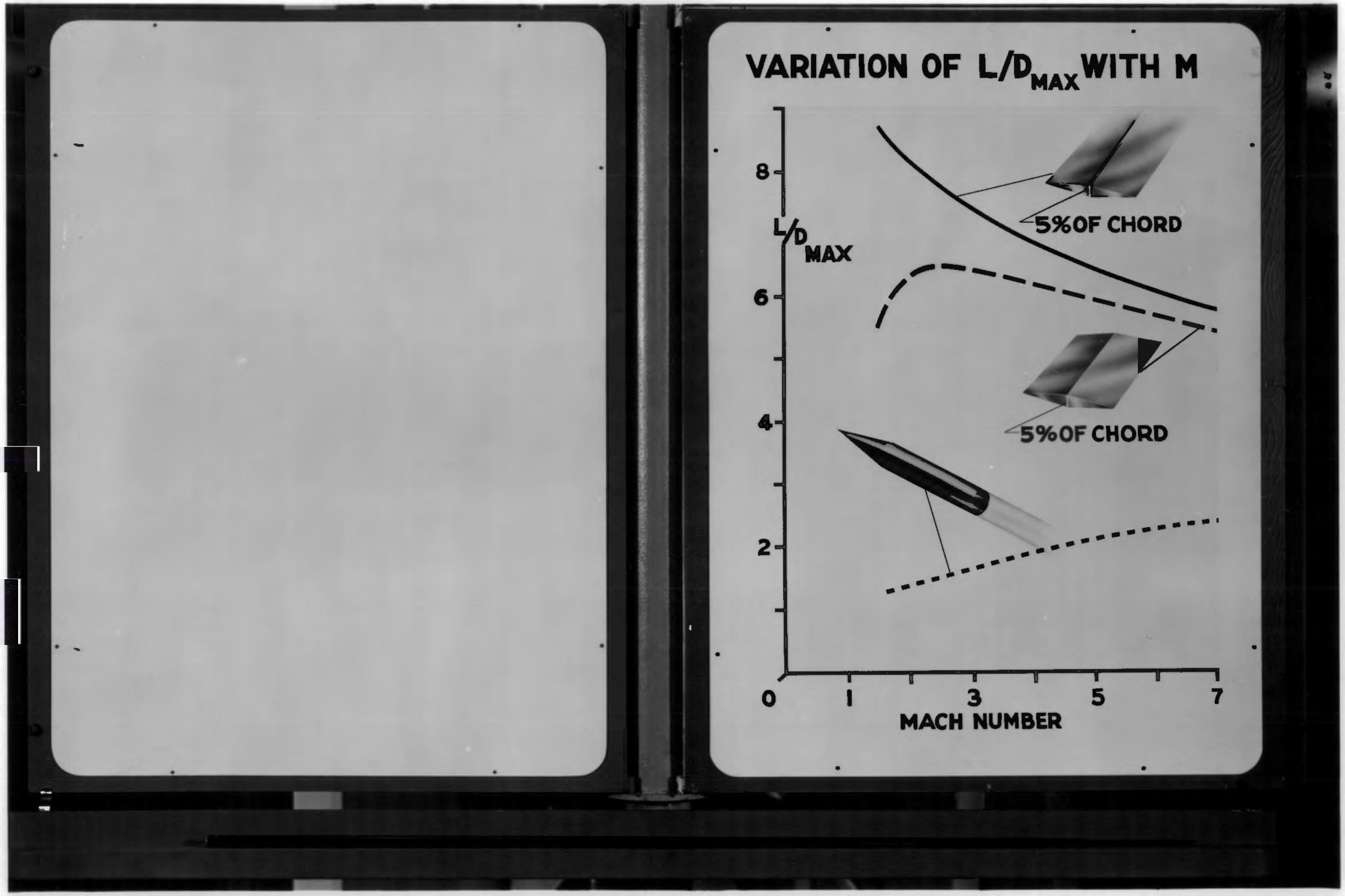
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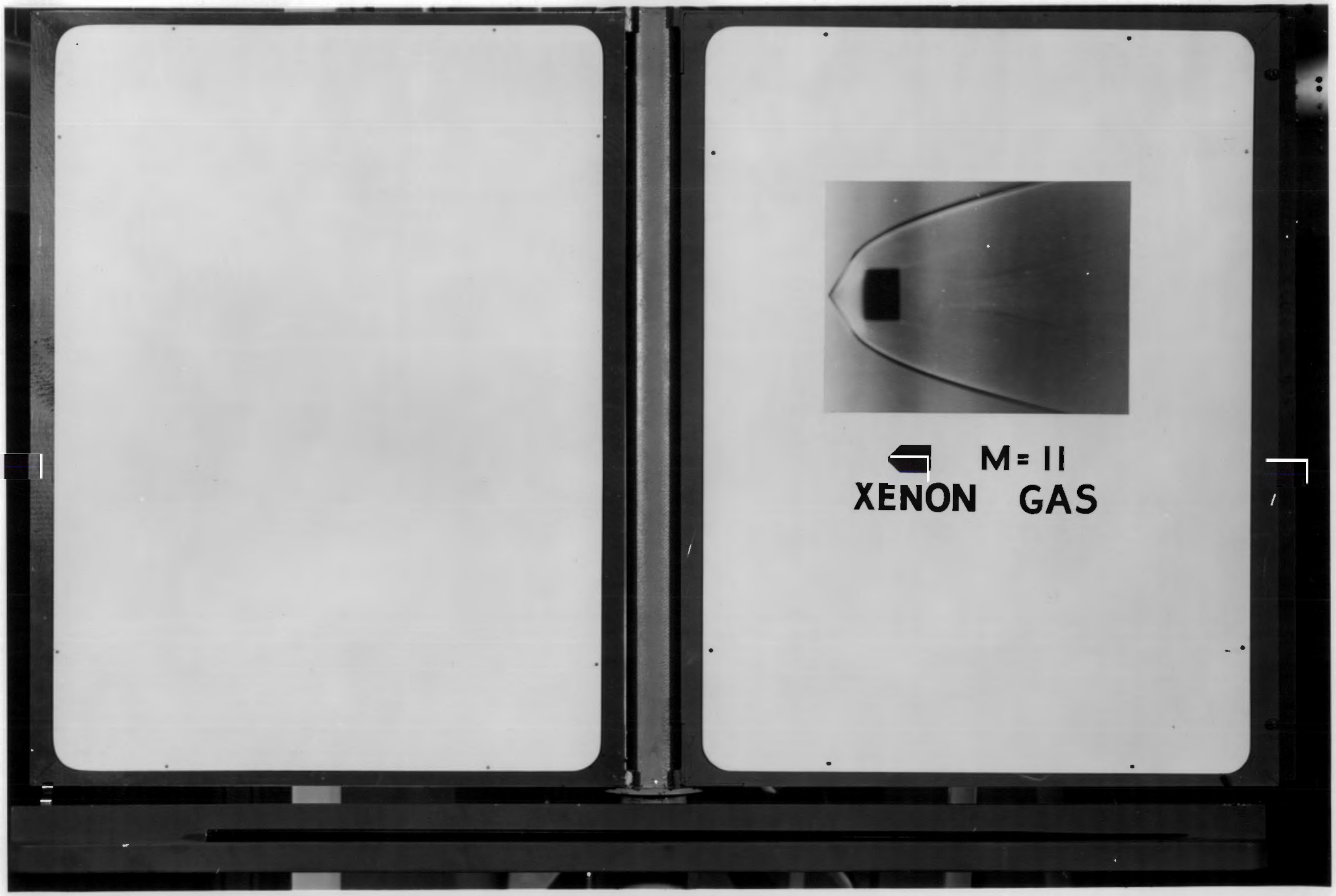

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