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AN EXPERIMENTAL INVESTIGATION AT SUPERSONIC SPEEDS OF ANNULAR DUCT INLETS SITUATED IN A REGION OF APPRECIABLE BOUNDARY LAYER

By

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AN EXPERIMENTAL INVESTIGATION AT SUPERSONIC SPEEDS OF ANNULAR DUCT INLETS SITUATED IN A REGION OF APPRECIABLE BOUNDARY LAYER

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SUMMARY

Annular air inlets situated several diameters behind the apex of various bodies of revolution were tested over the range of Mach numbers between 1.36 and 2.01 to determine the effects of relatively thick boundary layers upon the characteristics of duct entrances in supersonic flight. With all the models tested, the recovery of total pressure after diffusion to a low subsonic velocity was found to be approximately two-thirds of that through a normal shock wave occurring at the same free-stream Mach number. Schlieren photographs show that the cause of this low-pressure recovery is the interaction between the boundary layer and the back pressure in the diffuser; when the back pressure reaches only a moderate value, the boundary layer thickens and separates upstream of the duct entrance. Once separation has occurred, the flow through the inlet fluctuates violently.

A comparison of an inlet situated several diameters behind the apex of a body with an inlet having only a short, 50° cone ahead of it shows that, even though the thickness of the laminar boundary layer is apparently about the same in each case, the total-pressure recovery attainable with the 50° cone model is more than 30 percent greater at a Mach number of 1.70. This large difference in pressure recovery is caused by the greater local Mach number at the duct entrance of the longer model and the more severe interaction between the boundary layer and the back pressure in the diffuser.

It is concluded that compression at a local Mach number comparable to that of the supersonic stream will result in large losses in total pressure if the compression occurs in the presence of an appreciable boundary layer.

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INTRODUCTION

Since air must enter the combustion chamber of a ram-jet engine or the compressor of a turbo-jet engine at a relatively low velocity and with the maximum total pressure possible, the problem of reducing the velocity of a supersonic stream to a low subsonic speed with the minimum loss in total pressure is of importance in the design of supersonic aircraft. A considerable amount of work has been done upon the problem, and, in general, two methods for attaining a high pressure recovery have been suggested. In one, the stream is first decelerated in a converging channel to a low supersonic velocity; it then enters a throat, or section of minimum area, where compression to a high subsonic velocity occurs through a normal shock wave. Finally, the speed is further reduced in a subsonic diffuser. (See references 1, 2, 3, 4.) The other method employs oblique shock waves and the compression that occurs along the surface of a cone to produce a low supersonic Mach number prior to the normal shock wave in the entrance throat. (See references 3, 4, 5, 6, 7.) The principle of both schemes is to reduce the Mach number at which the normal shock wave occurs in order to maintain a more nearly isentropic flow. In the first method, the compression takes place entirely within the duct system; whereas, in the second, some of the compression is external.

The investigations at supersonic speeds that have been performed in the past have been concerned with duct inlets in a position where they receive only the air of the free stream or, at least, air that has flowed but a short distance over a solid boundary. In a practical application, such a position is not always feasible, for other design considerations may interfere. For example, an airplane, in which the jet engine is in the rear of the fuselage, can attain a very high total pressure at the engine intake with a duct entrance at the nose of the fuselage. However, this arrangement is often not practicable because internal space requirements, such as a pilot's cockpit, cargo space, or structural members, will obstruct the passage between the duct inlet and the engine. In this case, it is desirable to place the entrance on the side of the fuselage close to the engine where the subsequent ducts will not cause design complications. An inlet in such a position will be in a region where the boundary layer resulting from the flow over the fuselage is relatively thick. Both the total pressure at the engine intake and the drag force of the duct entrance may be seriously affected by the presence of this boundary layer.

Air inlets situated in regions of relatively thick boundary layer on supersonic aircraft are being investigated at the Ames

Laboratory of the National Advisory Committee for Aeronautics. The present report contains the results of the first series of tests. Inlets which received all of the boundary-layer air from the flow over comparatively long forebodies were tested in order to evaluate the importance of the problem and to study the nature of the flow.

SYMBOLS

P pressure coefficient $\left(\frac{p-p_o}{q_o}\right)$

m rate of mass flow

H total pressure

p static pressure

q dynamic pressure

γ ratio of specific heat at constant pressure to the specific heat at constant volume, 1.400

M Mach number

R Reynolds number based upon the length of body ahead of the entrance

A area

x distance from the apex of a forebody to a station ahead of the duct entrance

l length of the ogive of the forebodies of models A and B

d distance from the duct entrance to a station ahead of the settling chamber

L distance between the entrance throat and the settling chamber

The subscripts indicate the station of the measured quantity.

o free stream

1 duct entrance

2 entrance throat

3 settling chamber

4 exit throat

APPARATUS

Wind Tunnel

The investigation of duct inlets at supersonic speeds is being performed in the Ames 8- by 8-inch supersonic wind tunnel. This is a tunnel of the closed-throat, nonreturn type. Three centrifugal compressors, driven by motors of 4500 horsepower total rated capacity, furnish a continuous supply of air to the wind tunnel. Silica-gel dryers maintain an absolute humidity of less than 1 pound of water per 10,000 pounds of dry air.

The Mach number in the test section can be varied continuously while the wind tunnel is in operation between the limits of 1.20 and 2.13 if no model is present. This variation is produced by changing the area of the nozzle throat. The total pressure in the wind tunnel can also be varied continuously, but the pressure range that is available for changing the Reynolds number decreases as the Mach number increases. The Reynolds number per foot of length may be set between 6 and 8 million at the lowest Mach number and at 11 million for the highest.

Models

Figure 1 shows a typical installation of a model in the test section of the wind tunnel; figure 2 is a photograph of the bodies tested; and figure 3 shows the dimensions of the models. The principles used in designing these models are discussed in the section entitled "Design Considerations."

The duct inlets of all the models are annuli of equal diameters and of areas equal to 34.8 percent of the frontal area enclosed by the lip of the entrance. The forebody of model A consists of a 10-caliber ogival nose followed by a cylindrical section that is approximately 60 percent of the length of the ogive. The length of the body ahead of the duct entrance is five forebody diameters. The internal duct consists of a short, constant-area section immediately behind the inlet which is followed by a curved throat of adjustable area. This adjustment of area can be accomplished by moving the central body fore and aft relative to the outer shell

while the wind tunnel is in operation by the mechanism shown in figures 1 and 3. The throat is inclined at an angle of about 10° to the axis of the model, and its length is approximately six times the thickness of the entrance annulus. A subsonic diffuser connects the throat to a settling chamber. The surfaces of this diffuser diverge at an included angle of about 6.3° to form an equivalent cone angle of 12.6°. The exit of the passage through the model consists of a sonic throat of variable area. The purpose of this variable throat is to permit control of the pressure in the settling chamber.

Model B was designed for tests to provide data for a comparison between a fixed inlet having no contraction with the variable type of inlet represented by model A. Except for the shape of the entrance, the two models are identical. The area ratio between the inlet and the settling chamber of model B is 4.8, which is sufficient to maintain a Mach number of less than 0.25 in the settling chamber. With the exception of the shape of the forebodies, models C and D are the same as model B. The body of the former is of the same length as model B, but it has no cylindrical section; the shape is ogival between the apex and the entrance. The forebody of the latter consists of a shortened ogive the length of which is 2.50 forebody diameters.

In order to compare the results of tests in the 8- by 8-inch wind tunnel with those obtained in other supersonic wind tunnels, an inlet similar to one described in reference 7 was tested. This inlet consists of an annular entrance located about one forebody diameter behind the apex of a cone having a 50° vertical angle. The subsonic diffuser is the same as those of models B, C, and D.

The models are supported in the wind tunnel by vertical and horizontal struts as shown in figure 1. The vertical strut serves as the main support and as a fairing for the pressure-measurement tubes. The horizontal strut prevents lateral movement of the model and also houses the shafts and bevel gears that drive the movable parts. Though it is possible to change the angle of attack of the model by altering the support, no tests were made at angles other than 0° because of the preliminary nature of the first series of tests.

Instrumentation

Because of the difficulties involved in constructing equipment with which both pressure and drag forces can be measured simultaneously, the preliminary tests upon duct inlets are being performed

with models in which only pressure measurements are made.

The static pressure distribution along the diffusers is obtained with flush orifices, situated as shown in figure 3, that are connected to a multiple-tube mercury manometer. The total pressure in the settling chamber is measured by two pitot tubes which are located in the upper and lower halves of the settling chamber in order to indicate nonuniformity in the flow. The dynamic pressure in the settling chamber is measured by the difference between readings from a static pressure orifice in the chamber wall and the total pressure tubes. An orifice at the exit throat indicates whether sonic velocity exists through the outlet.

A qualitative picture of the flow about the models is furnished by a schlieren apparatus. Photographs of about eight microseconds exposure time are taken to record the flow patterns. The knife edge of the schlieren apparatus is placed parallel to the direction of the flow to emphasize gradients normal to the stream; it is in such a position that a decreasing density in a downward direction appears black in the upper half of the pictures. The photographs do not show images that are of uniform sensitivity because vibration of the floor which supports the schlieren apparatus causes a slight movement of the knife edge with respect to the light rays. Although each of the components of the apparatus is mounted upon a beam the purpose of which is to prevent any difference in the motion of each part, and even though this beam is spring-supported from the floor, there is still sufficient relative motion to affect the sensitivity. The vibration is especially detrimental in this case because the knife edge is perpendicular to the plane of the vibration.

METHODS

In preparation for the tests of duct inlets, the 8- by 8-inch supersonic wind tunnel was calibrated to determine the Mach number, pressure gradient, and stream angle throughout the test section as functions of the total pressure and the area of the nozzle throat. The Mach number was determined by schlieren photographs of the oblique shock waves originating from the apex of a cone and also by measurements of the static pressure. The stream angle was determined by tests with a wedge in which the static pressure difference upon the upper and lower surfaces was measured and compared with a calibration curve.

With a model installed in the test section of the wind tunnel, the available testing range is reduced. In the present tests, the

minimum Mach number at which supersonic flow can be maintained is 1.36. The maximum Mach number attainable is reduced to 2.01 because of excessive vibration of the models under certain conditions of flow into the duct. The majority of the tests were performed at a Mach number of 1.70 and at the maximum and minimum Reynolds numbers. Other tests were made at the maximum Reynolds number obtainable at Mach numbers of 1.36, 1.50, 1.90, and 2.01.

The following procedure is used in performing a test:

1. The throat areas of the wind tunnel, the duct entrance, and the duct exit are all set at their maximum values.

2. Air is released through the tunnel at a total pressure that will maintain supersonic flow at the minimum supersonic Mach number. Then, the throat of the wind tunnel is contracted. After supersonic flow has been established, the throat area of the tunnel is set to produce the Mach number of the test and the total pressure is increased to the value that will give the desired Reynolds number.

3. The area of the throat at the duct entrance is set to produce the desired contraction ratio.

4. The area of the exit throat is reduced to zero and then opened to the maximum value in predetermined increments. Pressure measurements and schlieren photographs are made at each setting.

The reason for releasing air into the tunnel at a low Mach number and a low total pressure is to reduce the intensity of the normal shock wave that moves through the test section when supersonic flow is established in order that the model and its supports will not suffer from a sudden, fluctuating load. Since a normal shock wave that is caused by the deceleration of the flow through the duct system must be in a diverging channel if it is to be stable, the contraction ratio at the duct entrance is reduced only after supersonic flow has been established through the inlet. Measurements are made for both increasing and decreasing values of the exit-throat area in order to obtain check points and also to detect any hysteretic phenomena.

Several tests were made to determine the effect of a relatively thick turbulent boundary layer entering the duct. This boundary layer was produced by a 3/4-inch band of No. 60 carborundum grit at the nose of the body.

DESIGN CONSIDERATIONS

The aims of a duct inlet design are as follows:

1. To reduce the velocity of the flow through the duct to a low subsonic speed with the least loss in total pressure and the least increase in external drag

2. To maintain a uniform distribution of the flow across the entrance to the settling chamber

3. To avert any discontinuity in the character of the flow that might result from a change in attitude, in speed, or in the pressure conditions within the settling chamber

The forebody and duct inlet of model A are intended to represent the fuselage of a typical supersonic airplane that has a duct entrance located near the stern, in a region of appreciable boundary layer. In order to reduce the number of variables of the tests, the subsonic diffuser was designed to minimize the loss even though it probably will not represent a practical application. No particular care was taken in the design of the external surface of the diffuser shell because only the internal pressure recovery was to be measured in the preliminary tests. The shape of the forebody of model A was determined by the requirement that the Mach number at the duct inlet be low in order to reduce the intensity of a normal shock wave occurring inside the entrance. A cylindrical section was used behind the ogival nose because a compression, or reduction in Mach number, occurs along its surface. The pressure-coefficient distribution, as computed by the linearized theory of reference 8, is shown in figure 4. The pressure coefficient at the inlet is small, -0.020 at a Mach number of 1.70; in other words, the local Mach number, 1.73, is nearly that of the free stream. The variation of the pressure coefficient with Mach number at the position of the inlet is also small, from -0.027 at a Mach number of 1.20 to -0.018 at a Mach number of 2.10; therefore, the velocity at the duct entrance is always nearly that of the free stream. Since the distribution of the pressure coefficient along the cylindrical section approaches zero asymptotically, very little additional compression can be attained by placing the inlet farther aft. The lip at the duct entrance of model A was made as sharp as possible and the internal surface was designed to be parallel to the local stream in order to minimize the internal disturbance caused by the lip. A variable contraction ratio at the entrance was used, because it has been shown that additional

pressure recovery can be attained once supersonic flow has been established by reducing the throat area and thus the Mach number at which the normal shock wave occurs. (See references 1 and 3.) The throat was extended for a short distance with very little divergence in the passage in order to stabilize the position of the normal shock wave as suggested in reference 9.

Although the cylindrical sections of the bodies of models A and B provide some compression ahead of the duct inlet, there is a conflicting effect; namely, an adverse pressure gradient that will thicken the boundary layer and, for a laminar boundary layer, decrease its stability. In order to avoid these consequences, the body of model C was designed to have a favorable gradient over its entire length. (See fig. 4.) The Mach number at the duct entrance is increased slightly as a result, for at a free-stream Mach number of 1.70, the pressure coefficient at the duct entrance is -0.041 which corresponds to a local Mach number of 1.76.

Since the length of surface which the flow must traverse before reaching the entrance affects the thickness of the boundary layer, the forebody of model D was designed to reduce this length substantially. The distance as measured along the surface between the apex of the body and the duct entrance is 3.665 inches for models A and B, 3.650 inches for model C, and 1.887 inches for model D; in other words, the length of run over model D is about 50 percent of that of the other models. The pressure gradient along the forebody is entirely favorable, and the pressure coefficient at the entrance is zero at a free-stream Mach number of 1.70. (See fig. 4.)

RESULTS

Presentation of Data

The data of the tests are presented as curves of total-pressure ratio H_s/H_o plotted against mass-flow ratio m_1/m_o. The latter term is defined as the mass flow that enters the inlet $\rho_1 V_1 A_1$ divided by that which would flow through a tube of the same area as the inlet in the free stream $\rho_o V_o A_1$. Since the two pitot tubes in the settling chamber indicate total pressures that agree within 2 percent and since these measurements repeat whether the area of the exit throat is being decreased or increased, only the measurements that were made with one pitot tube as the exit area was reduced are presented. Figure 5 shows the effect of inlet-contraction ratio upon the pressure recovery of model A; figure 6 compares the

recovery of models A, B, C, and D; and figure 7 shows the effect of Reynolds number and a turbulent boundary layer upon model B. The static pressure distribution along the subsonic diffuser of model B is shown in figure 8. These curves represent the results at a Mach number of 1.70. Figure 9 shows the effect of Mach number on the pressure recovery attainable with model B. A cross plot of the maximum pressure recovery attained by each model as a function of the free-stream Mach number is shown in figure 10 together with curves showing the pressure recovery across a normal shock wave occurring at the same free-stream Mach number and the recovery with the 50° cone model.

Schlieren photographs of the flow about the entrance of model A for entrance contraction ratios of 1.0 and 0.8 are shown in figure 11. Figure 12 shows the flow over model B with a turbulent boundary layer. The turbulent character of the flow can be identified by the diffuse, grey region next to the surface of the body. The laminary boundary layer, shown in the other pictures, is characterized by a sharp, white or black region. The fluctuation that is typical of the flow about the entrance of all the models at outlet-inlet-area ratios below the value which produces the maximum pressure recovery is shown in figure 13. These photographs were made consecutively with no change in any of the externally variable parameters. They are of model C because the effect is most pronounced in this case. Figure 14 shows the flow about the entrance of model D. Since the schlieren photographs show not only the flow disturbances caused by the presence of a model in the wind tunnel but also imperfections in the glass windows and density gradients in the stream that are not caused by the model, photographs of these extraneous effects are shown in figure 15.

Precision

The accuracy of the results can be judged by considering two general classifications of the sources of error. First, are the errors that result from variations in the uniformity of the flow through the test section of the wind tunnel; second, are those that result from inaccuracy in the measuring technique.

The flow in the wind tunnel was studied during the calibration tests. The results show that pressure and, therefore, Mach number gradients exist in the test section but that they are relatively small. For instance, the longitudinal variation of the Mach number through the test section at a nominal Mach number of 1.70 is between the limits of 1.71 and 1.69, less than 1 percent.

The gradient of Mach number over the length of the model between the apex of the body and the duct inlet is only 0.3 percent. The variation in the stream angle throughout the test section is between ±1°, but the variation over the length of the model is ±0.4°. The effect of these deviations upon the tests of duct inlets is thought to be small.

In a supersonic wind tunnel, the presence of moisture in the air can cause an error if the assumption is made, as it was in this case, that the total pressure in the test section is the same as that in the settling chamber. However, if the water content of the air is maintained at less than 0.0001 pound of water per pound of dry air, as was done in the present tests, the effect upon the total pressure is negligible.

Since the areas of the entrance and the exit to the duct determine the nature of the flow, they must be known accurately. The diameters of the entrance and exit were therefore measured precisely. There is a slight variation in the area of the exit throat that results from play between the threads of the lead screw and also between the teeth of the miter gears. Measurements show that this variation is within ±1.3 percent.

The accuracy of the pressure measurements depends upon the flow conditions about the duct entrance. When the mass-flow ratio is below that for maximum pressure recovery, the flow into the inlet is unsteady. Because of the lag in the tubing connecting the orifices and the manometer board and because of the inertia of the mercury in the manometer, the readings made in this mass-flow range represent average values, and they may not be as accurate as they appear. When the flow conditions are steady, the manometer tubes can be read to within ±1 millimeter of mercury, or within ±0.1 percent. Under the most adverse conditions the readings can be made to within ± 5 millimeters, or within 1 percent.

The determination of the mass flow through the model is dependent upon the total pressure and temperature in the settling chamber and the area of the exit throat. The assumption is made that the total temperature is the same as that of the free stream. It is believed that the mass-flow ratio is accurate to within ±1.5 percent.

The total-pressure measurements in the settling chamber of the models indicate not only the losses at the duct inlet but also the losses that occur in the subsonic diffuser. The magnitude of the latter losses can be estimated from the tests of the 50° cone model

which had the same subsonic diffuser as models B, C, and D. The maximum total-pressure recovery attained is 89 percent at a Mach number of 1.85 (fig. 10). This value agrees with those of similar tests described in references 6 and 7 and indicates that the losses resulting from the subsonic diffuser are less than 4 percent of the total pressure available.

DISCUSSION

Since there is no appreciable difference in the total pressure attained with models A, B, C, or D, the general properties of flow into annular duct inlets situated in a region of relatively thick boundary layer are discussed, and then the causes of the small differences in the flow through the models are described. Finally, the flow conditions about model D are compared to those about the 50° cone model in order to explain the large difference in total-pressure recovery attainable with each type of inlet.

General Flow Properties

If it is assumed that the total temperature in the settling chamber of the models is the same as that of the free stream and if sonic velocity is maintained at the exit throat, the relationship between the mass-flow and total-pressure ratios is indicated by the following equation:

$$\frac{m_I}{m_o} = \frac{m_4}{m_o} = \frac{H_3}{H_o} \times \frac{A_4}{A_1} \times \frac{1}{M_o}\left[\frac{2}{\gamma+1} + \frac{\gamma-1}{\gamma+1}M_o^2\right]^{\frac{\gamma+1}{2(\gamma-1)}}$$

The mass-flow ratio at a given Mach number is thus a function of the total-pressure and outlet-inlet-area ratios, and because of the compressibility of the fluid, it can be greater than one. For the 50° cone model, the mass-flow ratio reaches a value of 1.3.

The total-pressure ratio is dependent upon the outlet-inlet-area ratio. If the area ratio is large, the total pressure in the settling chamber is low compared to the maximum attainable. As shown in the schlieren photographs, the flow through the duct inlet is supersonic for such a condition, and inside the subsonic diffuser, the flow velocity increases as shown by the decrease in

the static pressure immediately behind the entrance. (See fig. 8.) Therefore, the deceleration to a subsonic speed occurs abruptly from a relatively high local Mach number, and the resulting shock losses are large. As the outlet-inlet-area ratio is reduced toward one, the pressure in the settling chamber rises rapidly. This increase is the result of the reduction in the intensity of the shock losses. As the back pressure in the settling chamber increases, the position at which the shock losses occur moves toward the duct inlet and into a region of lower local Mach number with a resulting decrease in the entropy rise. This movement of the shock losses as the outlet-inlet-area ratio is reduced is indicated in figure 8 by the position of the abrupt rise in the static pressure in the subsonic diffuser. Since, with the exception of the fluid in the boundary layer, the flow through the duct entrance is supersonic for these large values of the outlet-inlet-area ratio, the mass-flow ratio is very nearly constant.

The largest total-pressure ratio occurs, of course, when the losses in pressure are the least. This condition exists when the shock losses in the subsonic diffuser occur near the inlet, or at the minimum local Mach number. The flow through the entrance is supersonic, and the mass-flow ratio is very nearly the same as it was for larger values of the outlet-inlet-area ratio.

As the schlieren photographs show, when the outlet-inlet-area ratio is reduced below the value that produces the maximum total-pressure recovery, the boundary layer thickens and separates upstream of the duct entrance. This phenomenon is possible in supersonic flow because the effect of the adverse pressure gradient at the inlet extends upstream through the subsonic boundary layer. The result is that only air of a relatively low dynamic pressure flows through the entrance. Further reduction in the outlet-inlet-area ratio reduces the mass-flow ratio toward zero, but there is little change in the total-pressure ratio.

After separation has occurred upstream of the duct entrance, the flow becomes unsteady. Consecutive schlieren photographs show that the velocity through the inlet may be either supersonic with a relatively thin boundary layer, or it may be subsonic with a completely separated boundary layer. (See fig. 13.) The reason for the fluctuating flow is that, after separation has once occurred, the back pressure in the settling chamber decreases and the cause of the separation disappears. The boundary layer then resumes its normal course along the surface of the body, and the high-energy air of the supersonic stream once again enters the duct. Such a condition is transitory, for the back pressure in

the settling chamber immediately rises, thickens the boundary layer, and causes the cycle to repeat.

A notable fact is that no normal shock wave is evident in the static pressure distribution along the subsonic diffuser or in any of the schlieren photographs. No abrupt rise in static pressure of the magnitude that would be expected with a single normal shock wave occurs. If there were no boundary layer flowing into the duct, a sudden rise in static pressure at least twice that indicated by the tests would result from a normal shock wave inside the subsonic diffuser. (See reference 2.) The effect of the boundary layer is to obscure any pressure discontinuities as measured by static pressure orifices and also to change the effective shape of the channel. As discussed in references 10 and 11, the thickening of the boundary layer that results from an adverse pressure gradient causes weak oblique shock waves that reduce the intensity of the subsequent normal wave and thus distribute the pressure rise over an appreciable length. While the boundary layer is separated upstream of the inlet, the velocity of the air flowing into the duct is subsonic, and a normal shock wave cannot exist.

The effect of increasing the free-stream Mach number is to reduce the total-pressure ratio. (See fig. 9.) A comparison of the maximum total-pressure ratio attainable with models A, B, C, and D with the total-pressure ratio across a normal shock wave occurring at the same free-stream Mach number shows that the recovery with the models is only about two-thirds that of the shock wave. (See fig. 10.)

Specific Models

Model A

If there were no boundary layer at the duct inlet or inside the diffuser, a pressure recovery greater than 93 percent should be theoretically attainable at a free-stream Mach number of 1.70 with model A having an entrance contraction ratio of 0.73. A normal shock wave would exist in the entrance throat at a local Mach number slightly greater than one, and it would be of minimum intensity. The lowest recovery, about 85 percent, would occur if the normal shock wave existed in the relatively high Mach number region immediately ahead of the entrance. The presence of the boundary layer seriously alters these limits, for the best recovery, as attained in tests of model A at a Mach number of 1.70, is only 56 percent. A contraction at the entrance, which in the absence

of a boundary layer improves the recovery, has a detrimental effect. (See fig. 5.)

When the entrance contraction ratio of model A is reduced, the maximum value of the mass-flow ratio decreases. It is apparent from the equation that relates the ratios of mass flow, total pressure, and outlet-inlet area that the reason for this reduction is a loss in total pressure between the supersonic stream and the settling chamber at equal values of the outlet-inlet-area ratio. The schlieren photographs show that constriction of the duct immediately behind the entrance causes an adverse pressure gradient that is sufficient to thicken the boundary layer ahead of the inlet even at large values of the outlet area. (See fig. 11.) The result is a loss in total pressure in the settling chamber that increases as the inlet passage is contracted. The mass-flow ratio is greatly affected by a constriction, while the maximum total-pressure ratio is affected only slightly because the outlet-inlet-area ratio at which the maximum occurs decreases with the contraction ratio.

At a Mach number of 1.70, the mass-flow ratio corresponding to the maximum total-pressure ratio attainable with model A is about 0.92 or less than that of any of the other models. (See fig. 6.) The decreased flow rate is the result of a greater loss in total pressure at equal values of the outlet-inlet-area ratio. This lower recovery of model A is probably the result of the adverse effect of the extended entrance throat upon the boundary layer. The natural growth of the boundary layer effectively produces a converging channel even though the walls of the passage are parallel for a short distance and then only slightly divergent. The resulting pressure gradient further increases the boundary-layer thickness and causes an increase in entropy. Though the maximum mass-flow ratio of model A is less than those of models B and C, the maximum total-pressure ratio is slightly greater. It is possible that this improvement is the result of the stabilizing effect of the extended throat, for the back pressure in the settling chamber of model A can be increased to greater values than with models B and C because the boundary layer will not separate as readily. The extended throat may stabilize the flow at the entrance of the diffuser as explained in reference 9, and it also separates the boundary layer ahead of the entrance from the back pressure in the diverging diffuser by an appreciable distance.

Model B

The maximum total-pressure ratio attainable with model B at a Mach number of 1.70 and a Reynolds number of 2.9 million is

53 percent which occurs at a mass-flow ratio of 0.95. Decreasing the Reynolds number consistently improves the maximum total-pressure ratio a few percent as shown in figure 7. The cause of this improvement is not understood, for a greater loss in total pressure would be expected as a result of the increase in the thickness of the boundary layer.

The effect of increasing the surface roughness with a band of carborundum grit to ensure a turbulent boundary layer over the entire length of the forebody is to decrease both the maximum total-pressure ratio and the mass-flow ratio. (See fig. 7.) At a free-stream Mach number of 1.70, the turbulent boundary layer causes a loss of about 3 percent in the maximum total-pressure ratio and 6 percent in the mass-flow ratio. Since a turbulent boundary layer is more resistant to separation than a laminar one, it would be expected that the flow through an inlet would remain supersonic at greater values of the back pressure if a completely turbulent boundary layer existed over the forebody. However, the results show that separation occurs at nearly the same value of the outlet-inlet-area ratio whether the boundary layer is laminar or turbulent. (See figs. 11 and 12.) It is possible that a thinner turbulent boundary layer than that produced by carborundum grit at the nose of the forebody would result in some improvement.

Model C

At a Mach number of 1.70, the maximum total-pressure ratio of model C is 51 percent at a mass-flow ratio of 0.96. (See fig. 6.) The pressure recovery of model C is less than that of any of the other models because the boundary layer separates ahead of the inlet at a greater value of the outlet-inlet-area ratio. In other words, the back pressure in the diffuser has a greater adverse effect.

Model D

The thickness of the boundary layer can be substantially reduced without altering the character of the flow into this type of duct entrance, for only a slight improvement in pressure recovery is attained with model D. (See fig. 6.) The boundary layer thickens and separates in the same manner that it does with the other models. (See fig. 14.) The thickness of the boundary layer at the duct entrance of all the forebody shapes has been computed, assuming no back pressure in the diffuser, by the method

of reference 12. The thickness of the boundary layer, as defined in these calculations is the distance, normal to the surface, at which the local velocity is equal to 0.707 times the velocity outside the boundary layer. At a free-stream Mach number of 1.70, the thickness for models A and B is 0.0053 inch; for model C, 0.0044 inch; and for model D, 0.0019 inch. Comparison of the velocity profiles and schlieren photographs of the boundary layers described in reference 13 shows that the density gradient indicated by the schlieren apparatus extends to a normal distance at which the local velocity is roughly nine-tenths the velocity outside the boundary layer. If this figure is assumed, the calculated thickness of the boundary layers of the various models agree in order of magnitude with those determined from inspection of the schlieren photographs.

Comparison with the 50° cone model.- The thickness of the boundary layer of the 50° cone model, as computed for the same conditions and by the same method as for the other models, is 0.0016 inch, nearly equal to that of model D. However, the maximum total-pressure recovery is 91 percent at a free-stream Mach number of 1.70 as compared to 58-percent recovery with model D. (See fig. 10.) This large difference in pressure recovery is caused by the greater local Mach number at the duct entrance of the longer model. The local Mach numbers, as determined theoretically, are 1.17 for the 50° cone model and 1.70 for model D. Therefore, the compression that occurs at the entrance of the latter model is greater, the interaction with the boundary layer is more severe, and the resulting losses are much larger. To compare the inlets at the same entrance Mach number of 1.5, the 50° cone must be at a free-stream Mach number of 2.1, at which value the pressure recovery is about 78 percent. With model D, the free-stream Mach number is nearly 1.5 and the recovery is 66 percent. The reason for this difference at the same entrance Mach number is not understood at the present time. An investigation of the effects of local Mach number upon the boundary layer is being performed to determine the causes.

Although the total-pressure recoveries with models A, B, C and D are much less than that of the 50° cone model, this criterion does not fully determine their worth. The drag caused by the forebody and the duct system of each model will differ from that of other models; therefore, final comparisons of inlets must not only include the total-pressure recovery but also the drag forces upon the fuselages that contain them.

It is apparent that large losses in total pressure result from a duct entrance situated in a region of appreciable boundary layer where the local Mach number is comparable to that of the free-stream. Reduction of these losses can be achieved by reducing the interaction between the boundary layer and the back pressure inside the diffuser. This reduction can be accomplished either by decreasing the local Mach number at the duct entrance by a method that will produce external compression with no adverse effect upon the boundary layer or by decreasing the amount of boundary-layer air that reaches the entrance.

CONCLUSIONS

Tests at Mach numbers between 1.36 and 2.01 of annular duct inlets situated several diameters behind the apex of bodies of revolution have shown the following effects:

1. Because of the interaction between the back pressure inside the diffuser and the boundary layer flowing into it, the total-pressure recovery attained is approximately two-thirds of that of a normal shock wave occurring at the same free-stream Mach number.

2. When the mass-flow ratio is less than that which produces the maximum total-pressure ratio, the flow into the duct fluctuates violently. The flow may be either supersonic through the inlet with a relatively thin boundary layer, or it may be subsonic with a completely separated boundary layer.

3. An appreciable change in the thickness of the laminar boundary layer or even a relatively thick turbulent layer has only negligible effects upon the recovery of total pressure.

4. Reducing the local Mach number immediately behind the duct entrance by constricting the channel has a detrimental effect if a relatively thick boundary layer flows through the inlet.

In general, compression at a Mach number comparable to that of the supersonic stream will result in large losses in total pressure if the compression occurs in the presence of an appreciable boundary layer.

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National Advisory Committee for Aeronautics,
Moffett Field, Calif.

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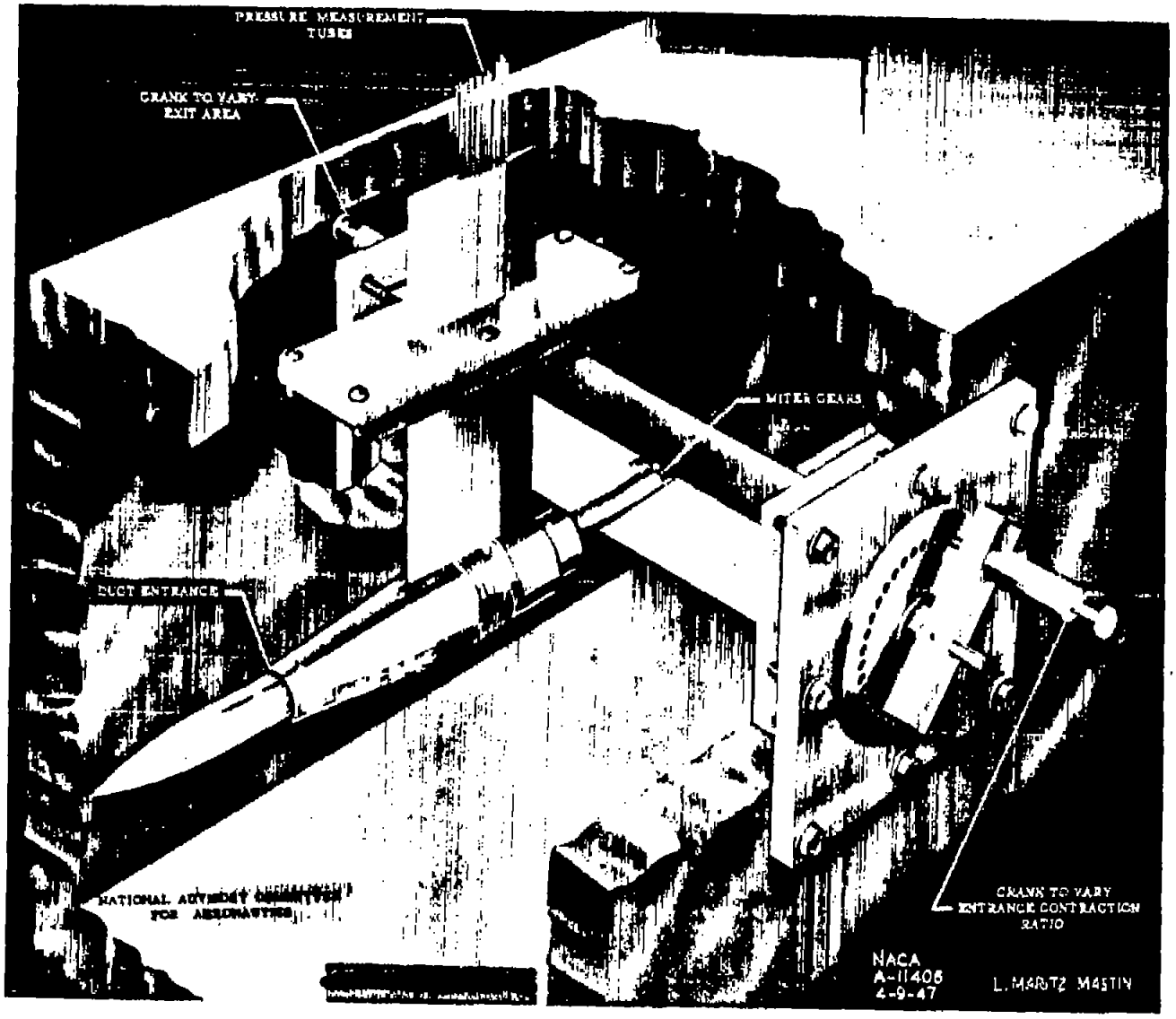

Figure 1.- Model installation in the Ames 8- by 8-inch supersonic wind tunnel

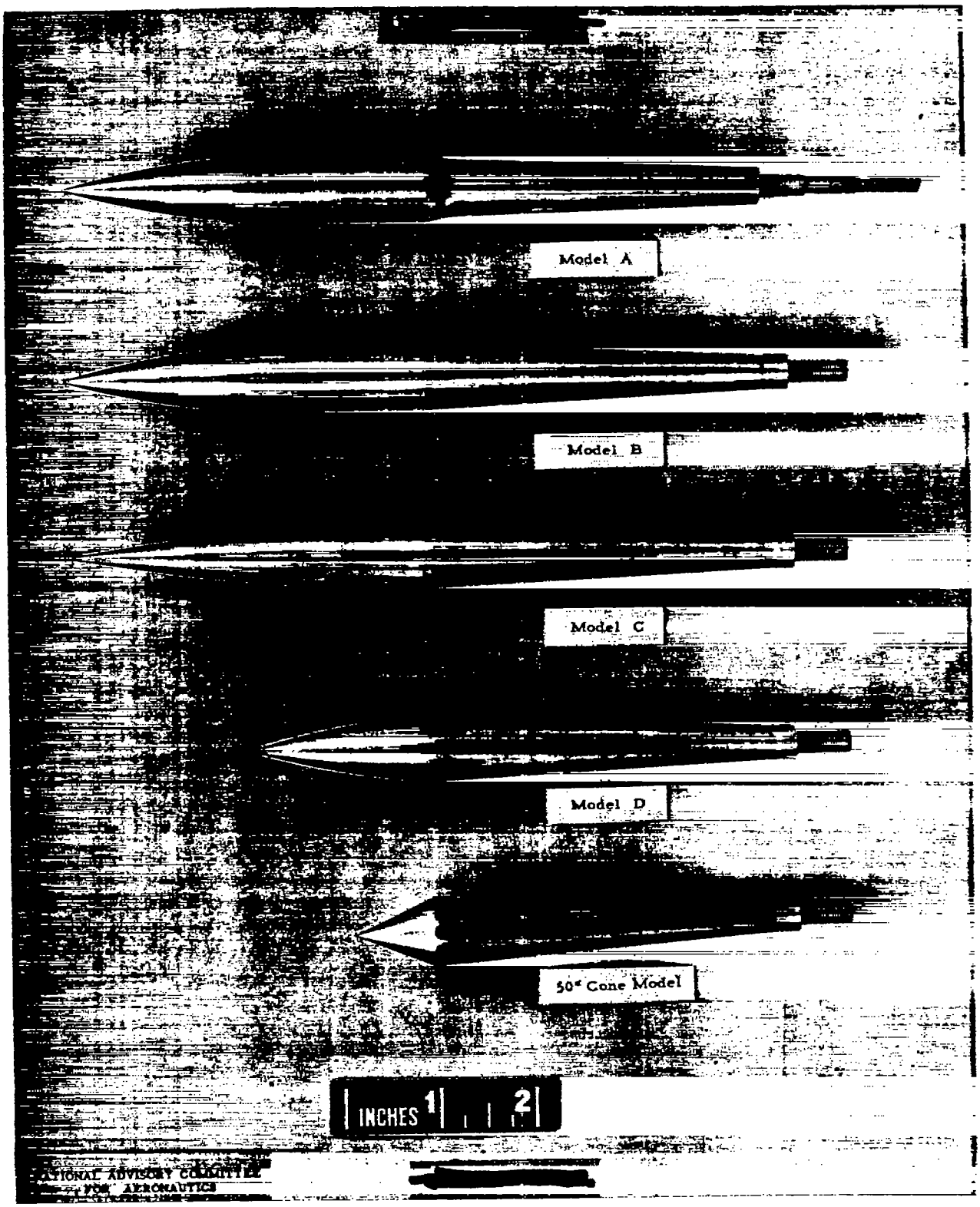

Figure 2.- Bodies of Models.

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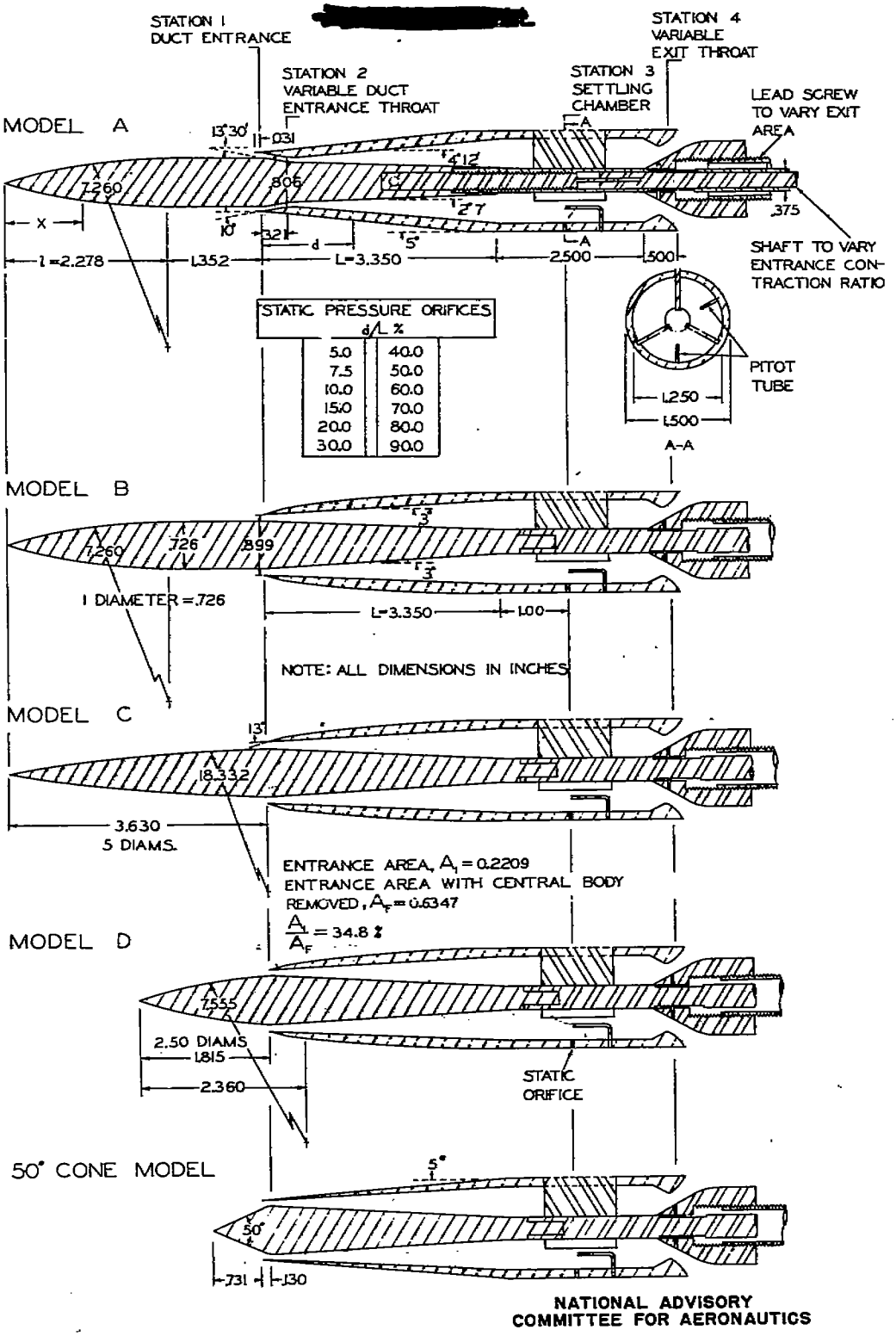

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FIGURE 3-MODEL DIMENSIONS

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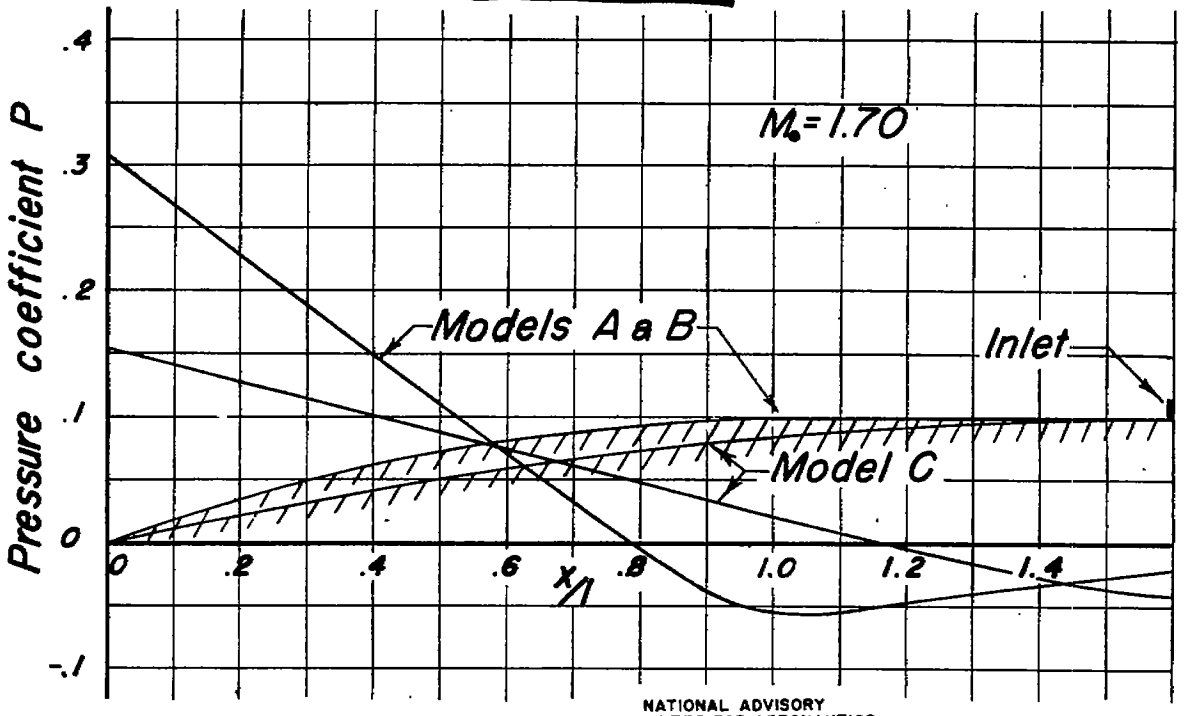

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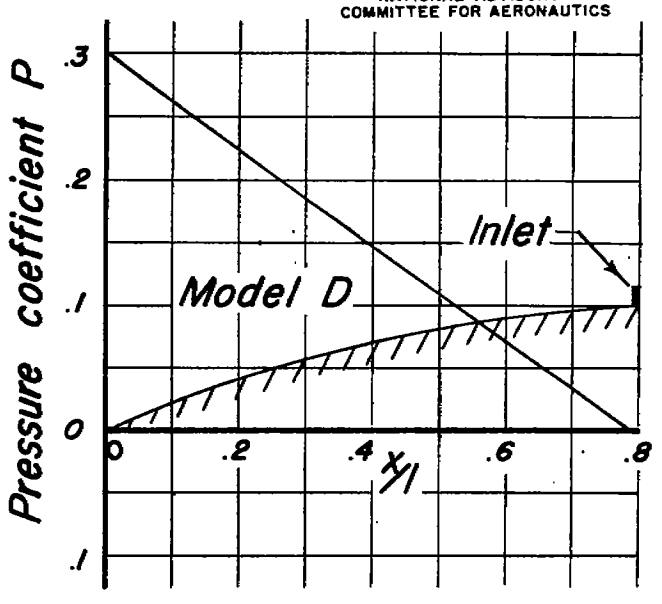

Figure 4.-Pressure-coefficient distribution over the forebodies of models A,B,C,D

[redacted]

[redacted]

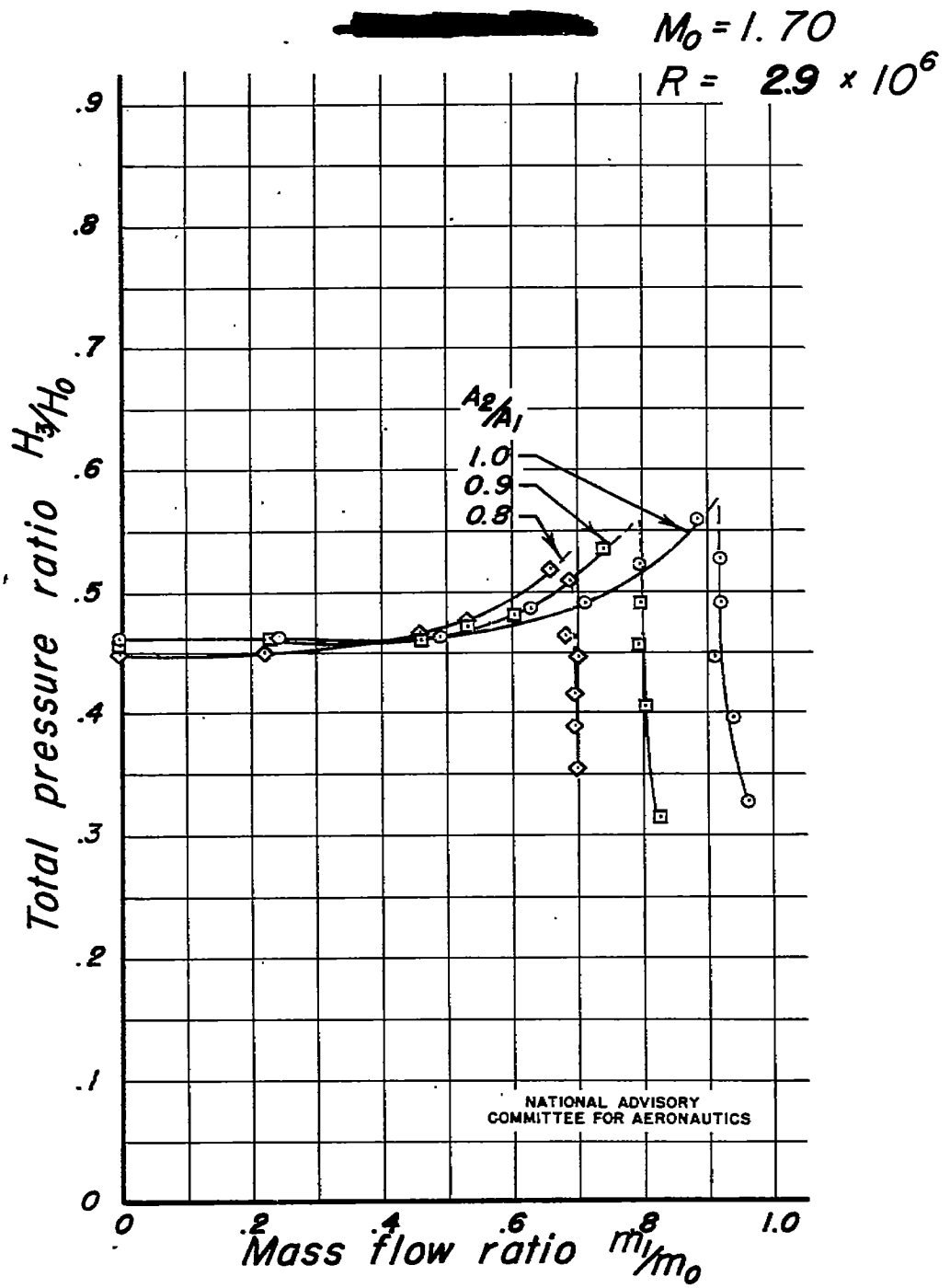

Figure 5.- Variation of total pressure ratio with mass flow ratio for model A with several entrance contraction ratios

[redacted]

[redacted]

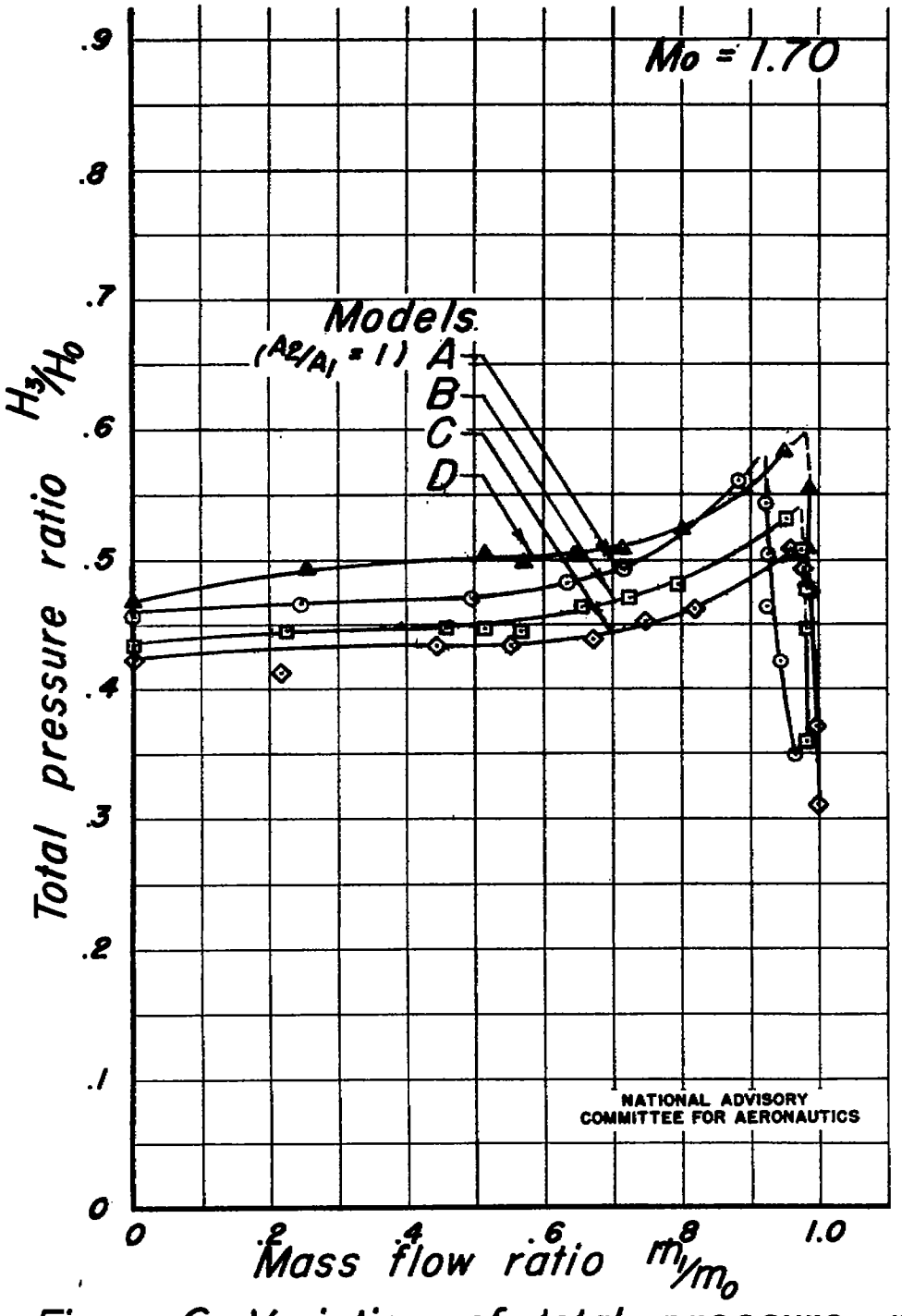

Figure 6.-Variation of total pressure ratio with mass flow ratio for models A,B,C,D

[redacted]

[redacted]

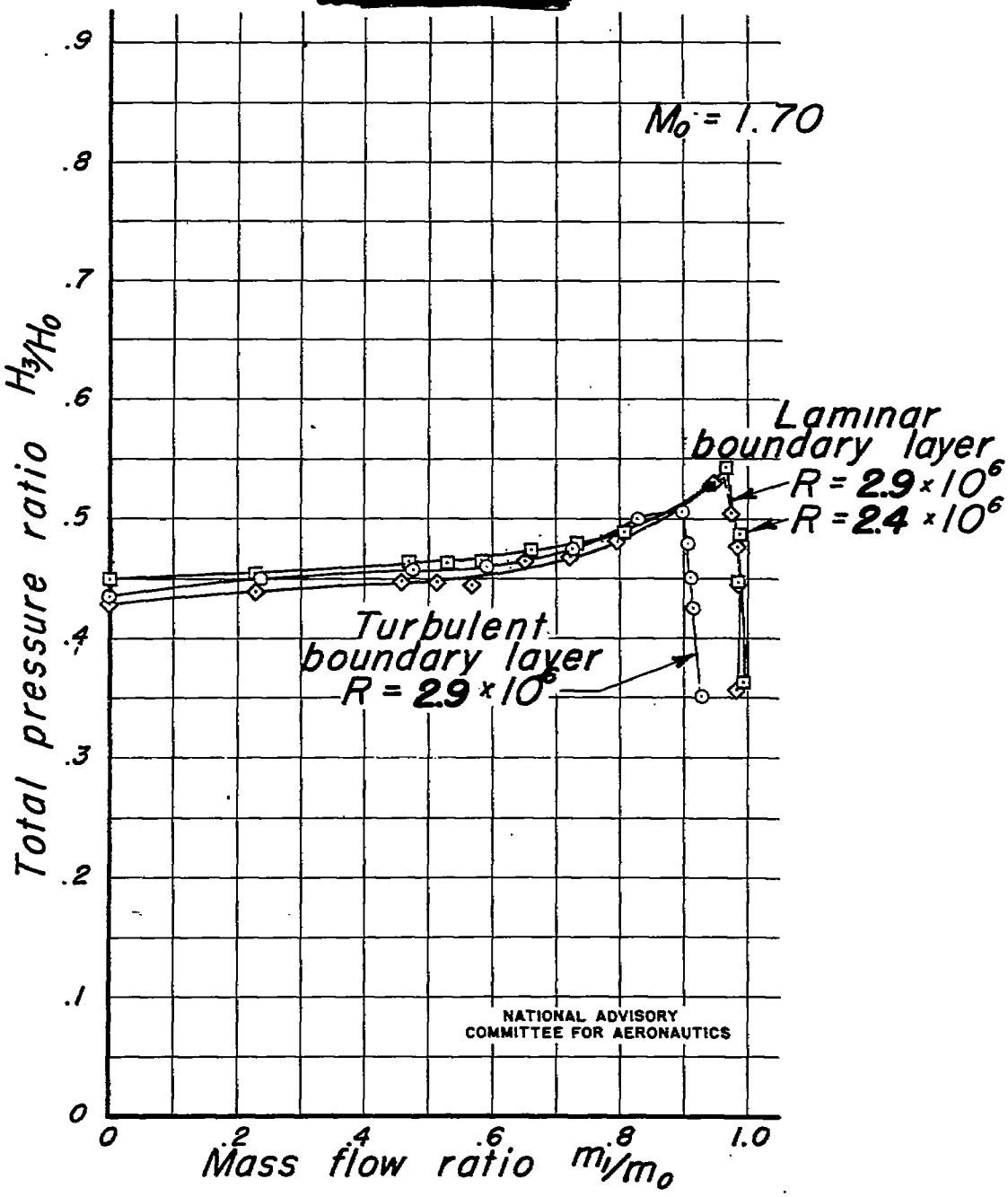

Figure 7.- Effect of Reynolds number and a turbulent boundary layer on the variation of total pressure ratio with mass flow ratio for model B

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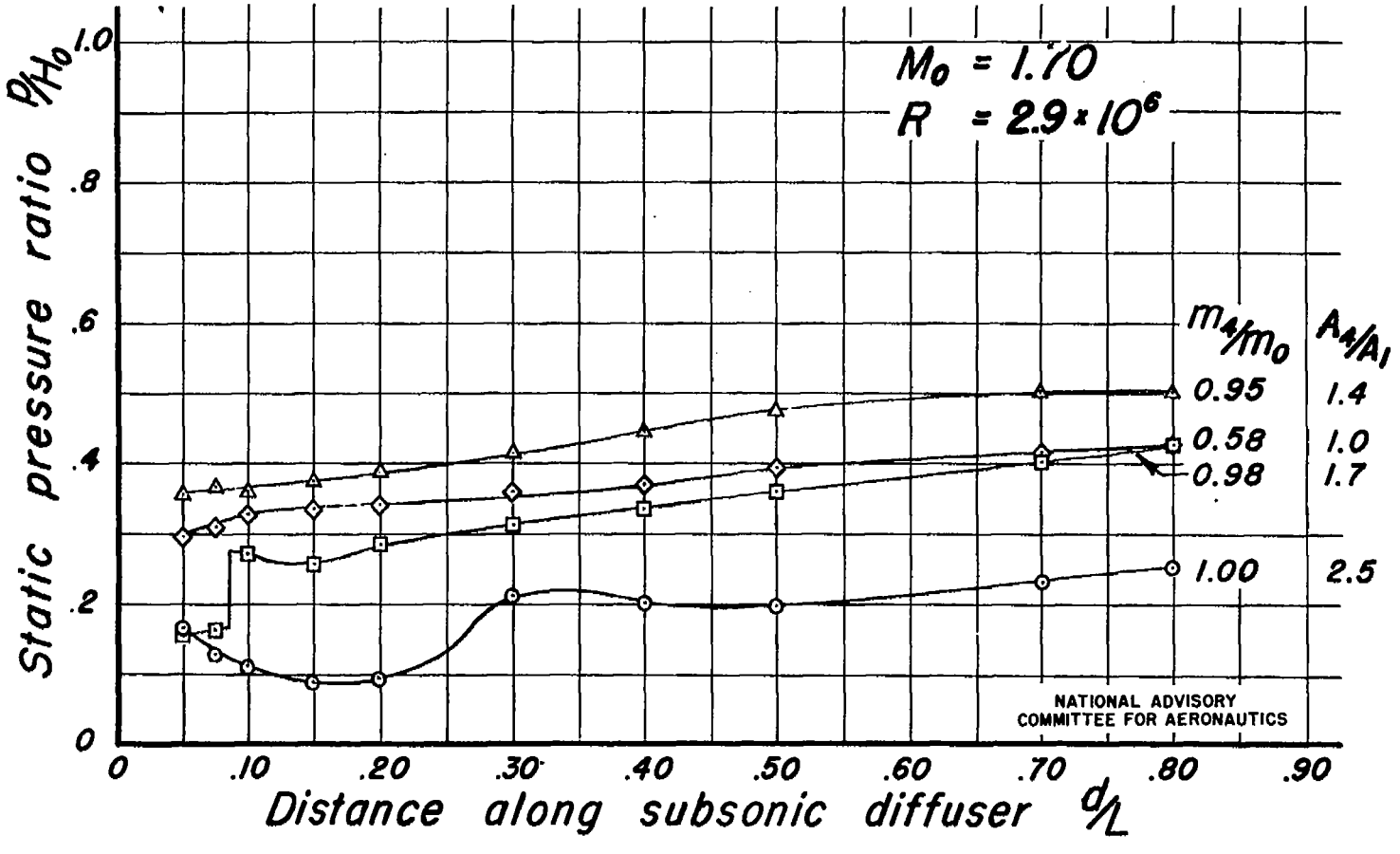

Figure 8.- Variation of static pressure along the subsonic diffuser of model B

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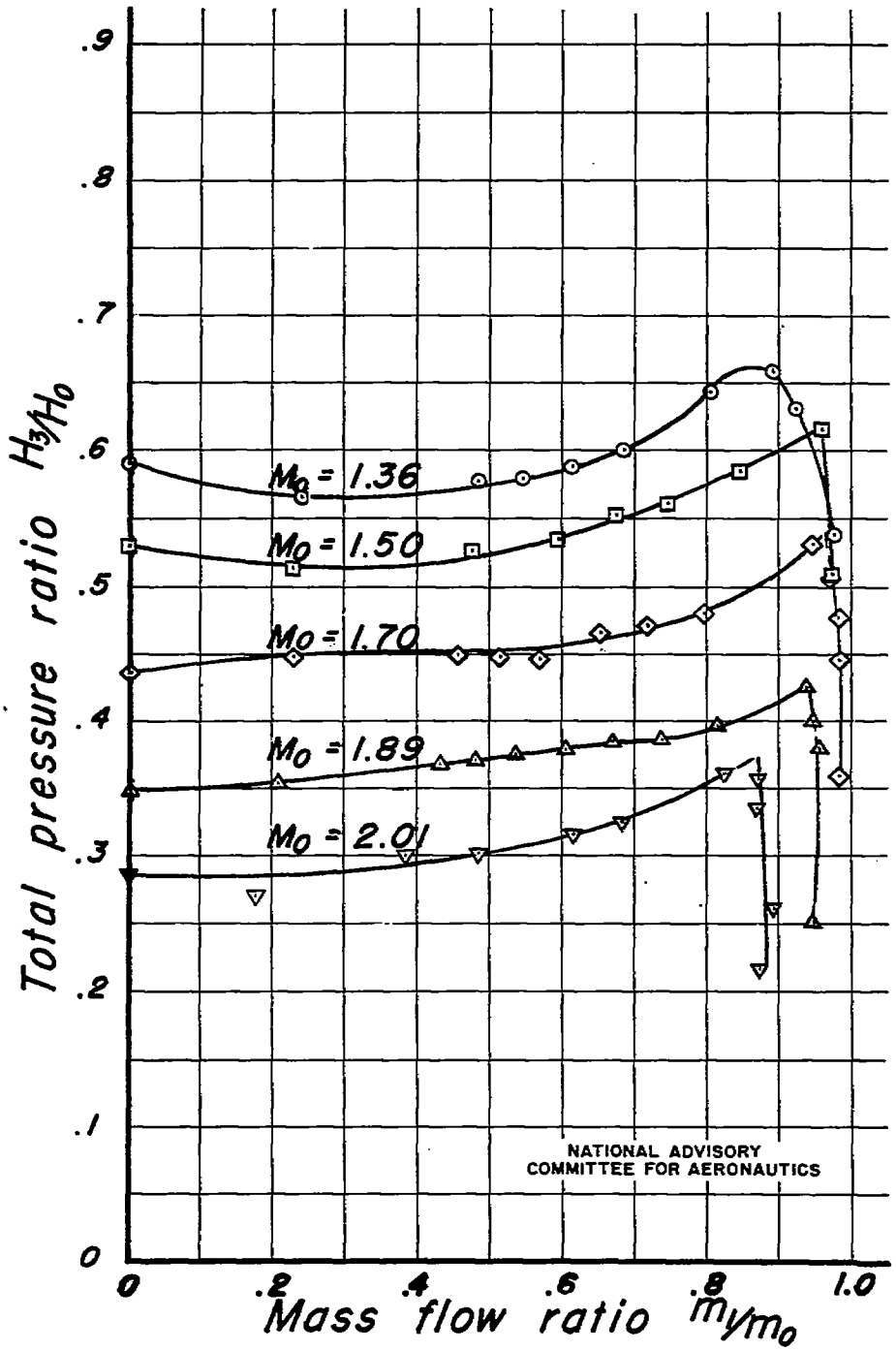

Figure 9.- Variation of total pressure ratio with mass flow ratio for model B at several mach numbers

[redacted]

[redacted]

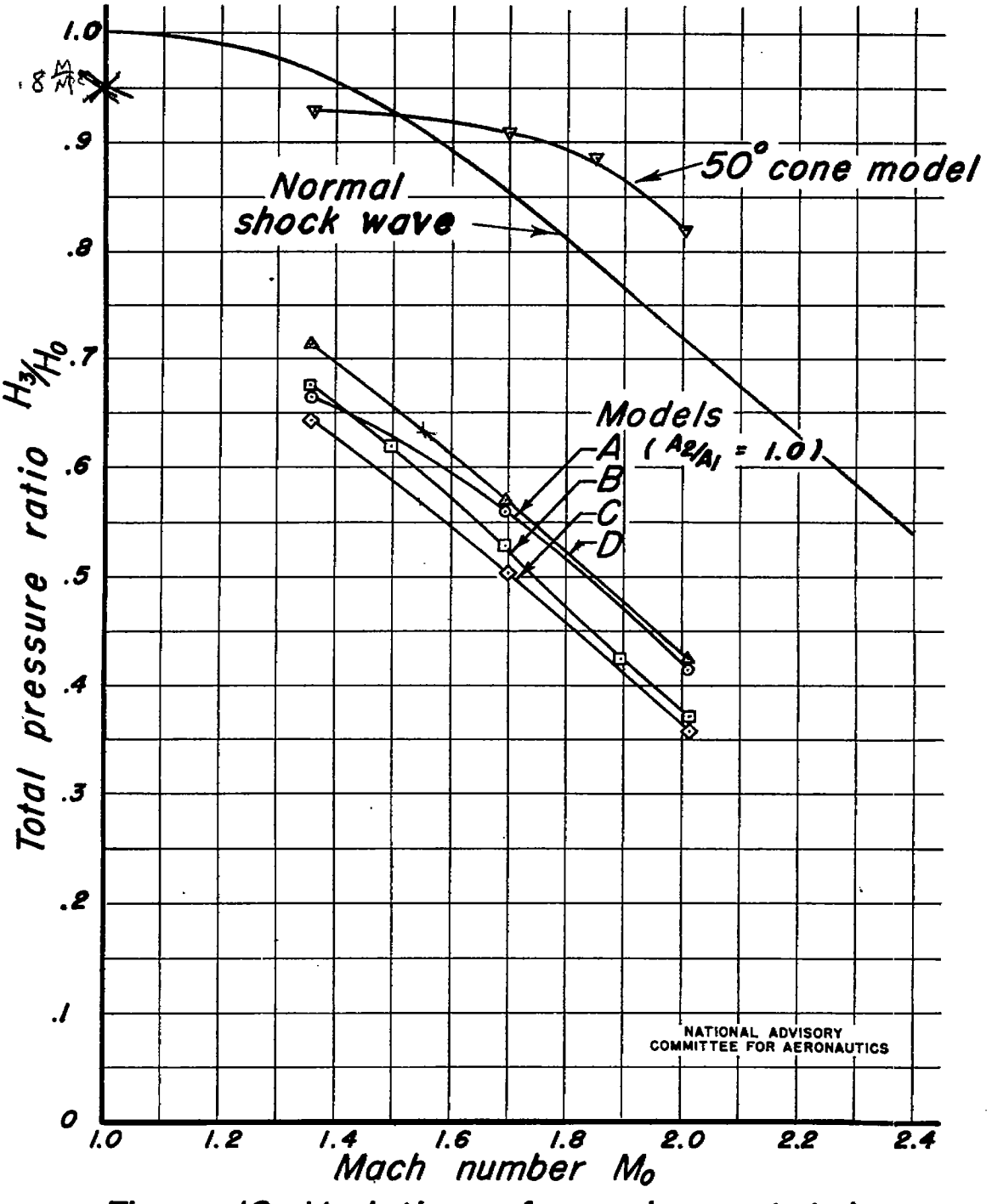

Figure 10.- Variation of maximum total pressure ratio with mach number

[redacted]

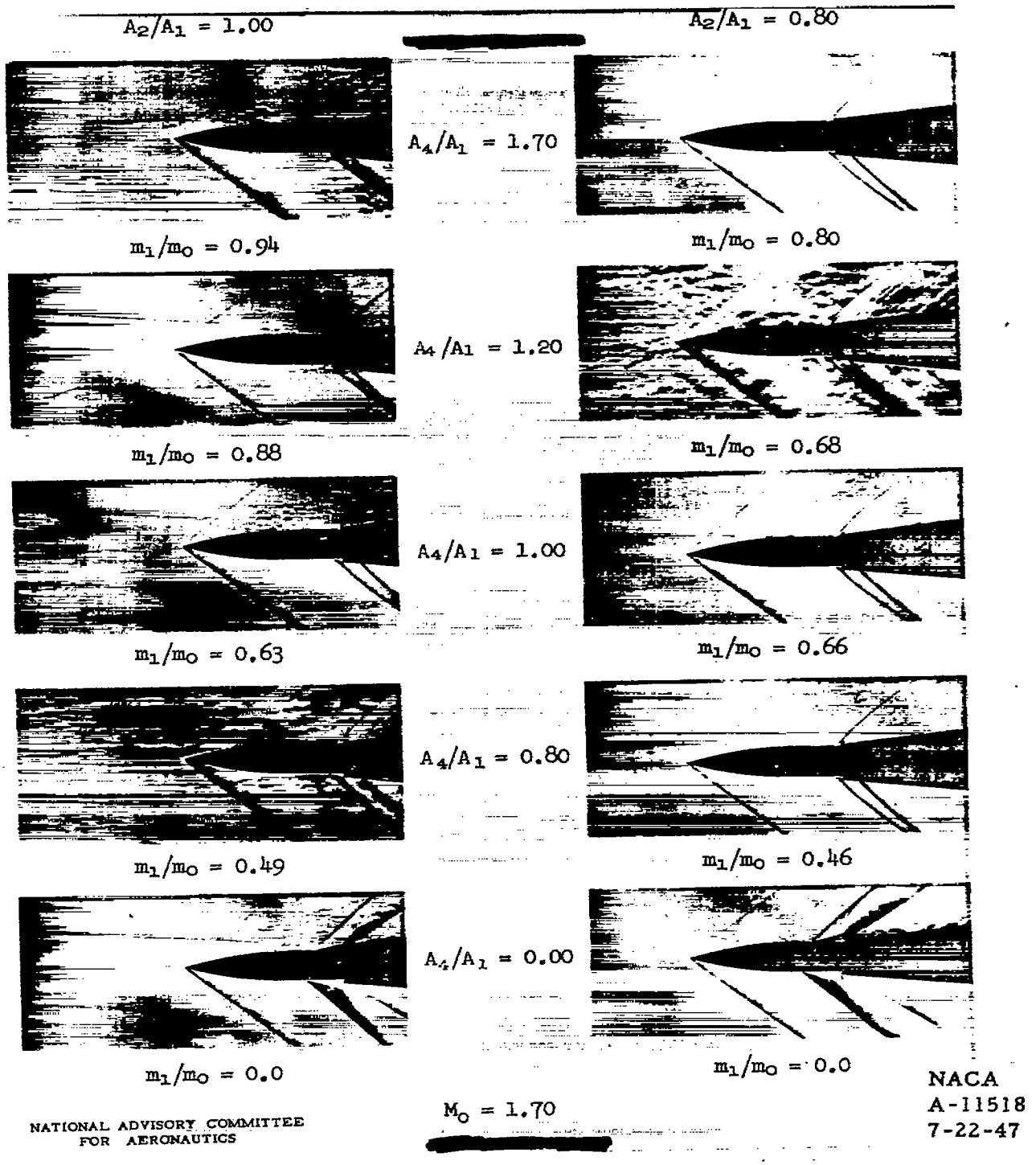

Figure 11.- Schlieren photographs of Model A showing the flow at various outlet-inlet area ratios for entrance contraction ratios of 1.0 and 0.8.

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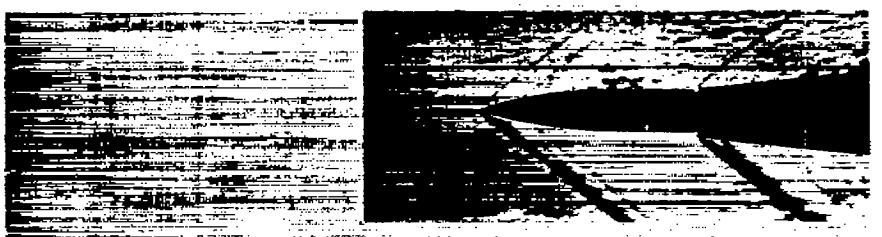
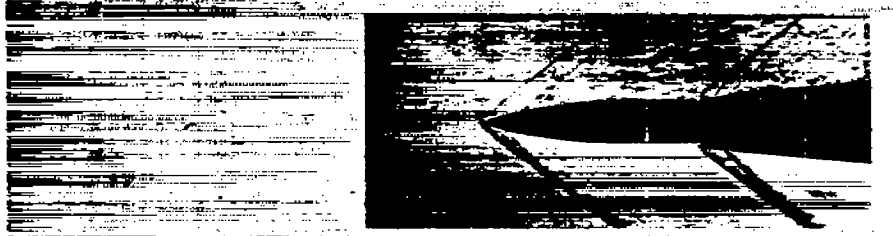


$A_4/A_1 = 1.7$ $m_1/m_o = 0.91$



$A_4/A_1 = 1.2$ $m_1/m_o = 0.83$

$A_4/A_1 = 1.0$ $m_1/m_o = 0.59$

$A_4/A_1 = 0.8$ $m_1/m_o = 0.47$

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$A_4/A_1 = 0.0$ $m_1/m_o = 0.0$

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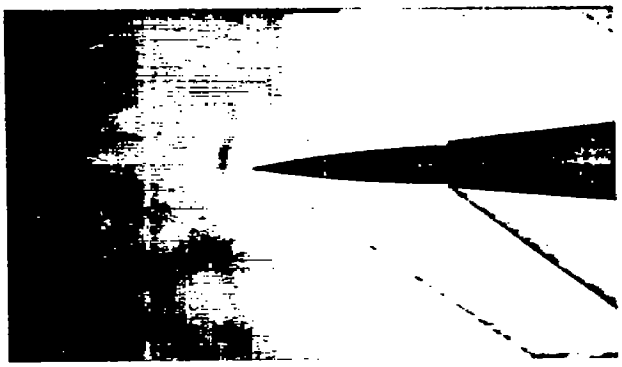
$M_o = 1.70$

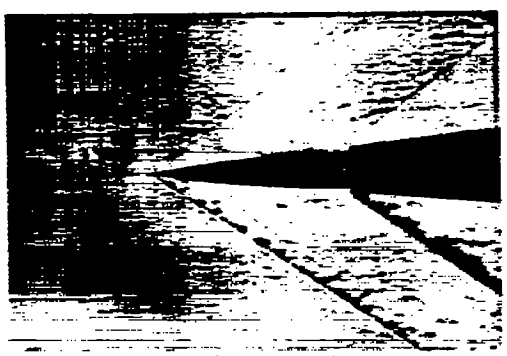
[redacted]

Figure 12.- Schlieren photographs of model B showing the flow with a turbulent boundary layer over the forebody.

[redacted]

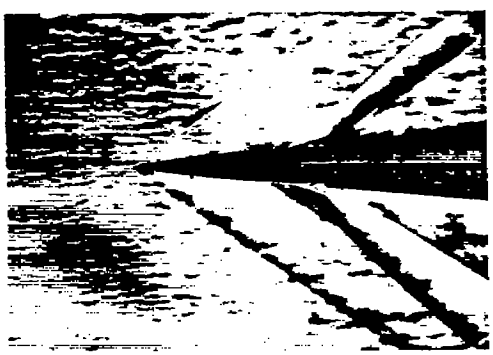


$A_4/A_1 = 1.7$ $m_1/m_o = 0.98$

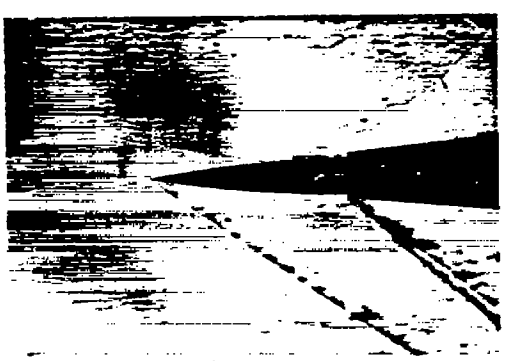


$A_4/A_1 = 1.4$

$\frac{m_1}{m_o} = 0.82$

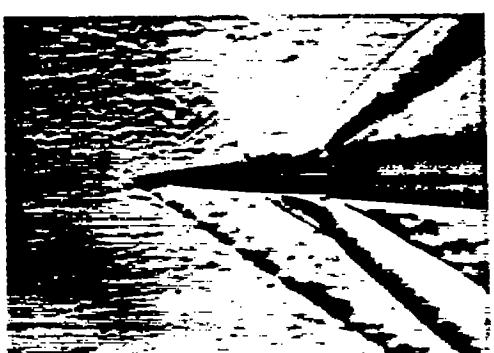


$A_4/A_1 = 1.4$



$A_4/A_1 = 1.2$

$\frac{m_1}{m_o} = 0.67$



$A_4/A_1 = 1.2$

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$M_o = 1.70$

[redacted]

Figure 13.- Schileren photographs of model C showing the fluctuating flow upstream of the entrance.

[redacted]

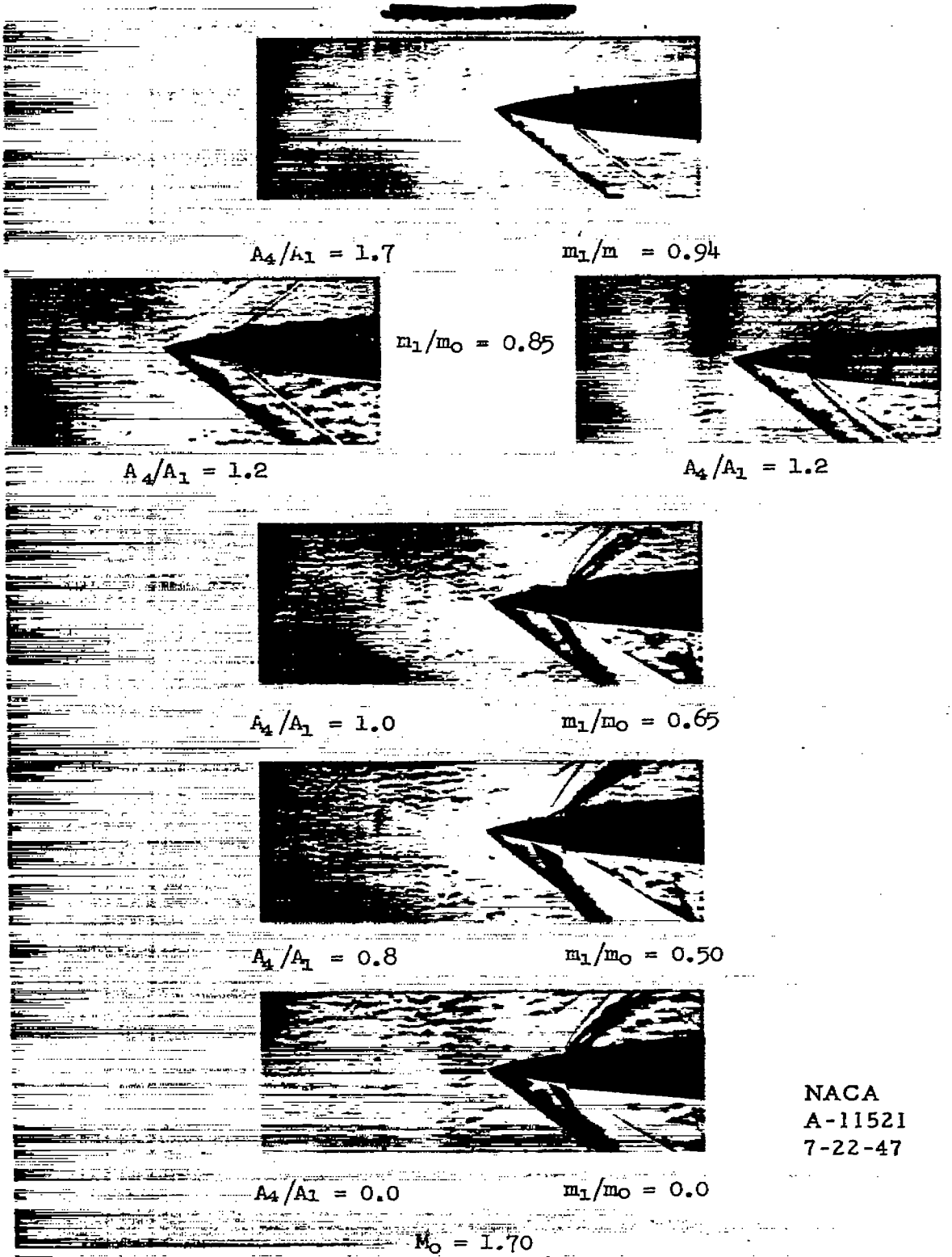

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Figure 14.- Schlieren photographs of model D at various values of the outlet-inlet area ratio.

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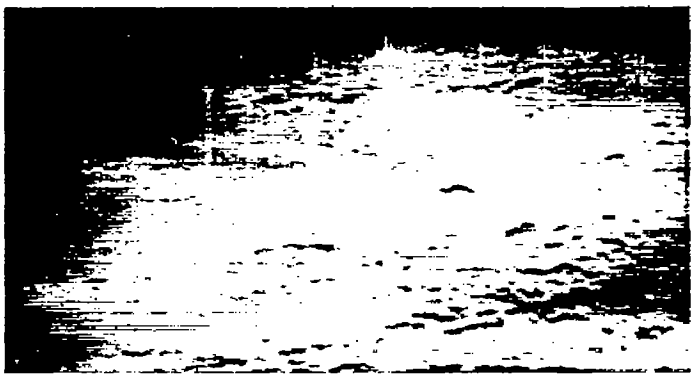
(a) $M_0 = 0$



(b) $M_0 = 1.70$

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Figure 15.- Schlieren photographs of the test section of the 8- by 8-inch wind tunnel with no model installed.

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