RM No. A8A13

NACA RM No. A8A13

NACA

RESEARCH MEMORANDUM

PRESSURE RECOVERY AT SUPERSONIC SPEEDS THROUGH ANNULAR DUCT INLETS

SITUATED IN A REGION OF APPRECIABLE BOUNDARY LAYER.

I - ADDITION OF ENERGY TO THE BOUNDARY LAYER

By Wallace F. Davis and George B. Brajnikoff

Ames Aeronautical Laboratory,
Moffett Field, Calif.

NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

WASHINGTON

April 1, 1948

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

PRESSURE RECOVERY AT SUPERSONIC SPEEDS THROUGH ANNULAR DUCT INLETS SITUATED IN A REGION OF APPRECIABLE BOUNDARY LAYER.

I – ADDITION OF ENERGY TO THE BOUNDARY LAYER

By Wallace F. Davis and George B. Brajnikoff

SUMMARY

A model having a nozzle upstream of an annular duct inlet for the purpose of ejecting high-velocity air into the boundary layer of the flow along the forebody was tested at Mach numbers between 1.36 and 2.01. The size and location of the nozzle and the total pressure of the air in the jet were varied to determine their effects upon the pressure recovery attainable after diffusion through the duct inlet. The results of the tests showed that the maximum total-pressure recovery was greater than that of a model having no air ejected. The causes of the greater recovery were the delay in the separation of the boundary layer that resulted from mixing between the boundary layer and the jet and also the reduction in the intake Mach number caused by an oblique shock wave that originated from the nozzle outlet when the free boundary of the jet was divergent.

If the high-pressure air ejected through the nozzle were supplied by the compressor of a turbo-jet engine contained in the model, the improvement in the pressure recovery measured during the tests would not be entirely useful in increasing the thrust force of the engine, because some of the available energy would have to be used to recirculate the air. Calculations showed that recirculating air through the intake system of an assumed engine caused an 8-percent improvement in the pressure recovery effective in producing thrust. This improvement would probably be larger if the tests were performed under full-scale flight conditions.

INTRODUCTION

The results of a preliminary investigation at supersonic speeds of annular duct inlets situated in a region of appreciable boundary layer show that the recovery of total pressure after diffusion was about two-thirds that of a normal shock wave occurring at the free-stream Mach number. (See reference 1.) The total-pressure recovery was relatively low because the severe adverse pressure gradient produced by a rapid deceleration of the flow inside the diffusor caused the boundary layer to thicken and to separate upstream of the duct inlet. In order to improve the recovery, the intensity of the compression inside the duct must be decreased, the amount of low-energy air of the boundary layer that enters the inlet must be diminished, or both factors must be reduced simultaneously.

These principles were used in the design of the models described in reference 2. The rate of compression of the supersonic stream was decreased by creating an oblique shock wave upstream of the duct inlet to reduce the Mach number at which the major portion of the compression occurred. The amount of boundary layer relative to the total mass of air flowing through the intake was diminished by using an inlet of the same area as the annular entrances but which enclosed only a portion of the forebody. The maximum recovery of total pressure that was attained in the tests of these models, was about four-fifths that of a normal shock wave.

Another method for diminishing the amount of retarded air that flows through the inlet is to accelerate the boundary layer of the flow over the forebody by ejecting a stream of high-velocity air along the surface upstream of the duct inlet. Such a system might be practical for aircraft propelled by turbo-jet engines because high-pressure air is available from the compressor of the engine. If a small portion of this air were recirculated and ejected through a nozzle into the boundary layer, separation of the flow might be deferred with a resultant increase in the recovery of total pressure at the face of the compressor. The thrust force of the engine would be increased if the gain in the available energy due to the improved pressure recovery were greater than the amount of energy expended in recirculation. It is the purpose of the present report to investigate the feasibility of such a system.

SYMBOLS

H	total pressure
M	Mach number
A	area
m	rate of mass flow
x	distance between the nozzle outlet and the duct entrance

The subscripts indicate the station of the measured quantity.

o free stream

1 duct entrance

2 reservoir of the nozzle

3 settling chamber

4 exit throat

APPARATUS AND TESTS

The tests were performed in the Ames 8- by 8-inch supersonic wind tunnel at Mach numbers between 1.36 and 2.01 and Reynolds numbers, based upon the length of the body ahead of the entrance, of 2.21 to 3.10 million. The apparatus and methods used during the investigation are described in reference 1.

The model and the equipment for supplying high-pressure air are shown in figure 1, and the dimensions of the model are given in figure 2. Except for the presence of the annular nozzle, the model is similar to model B of reference 1. The forebody, which consists of a 10-caliber ogival nose followed by a cylindrical section, is five forebody diameters in length. The pressure distribution over this shape produces a local Mach number at the duct inlet that is very nearly equal to that of the free stream. The projected frontal area of the annular inlet is 34.8 percent of the area enclosed by the outside lip of the entrance. The subsonic diffusor diverges at an equivalent cone angle of 10° with an area ratio of 4.45 between the inlet and the settling chamber. The exit of the passage through the model consists of a variable throat to permit control of the pressure in the settling chamber.

During the tests of each model configuration, measurements of the total-pressure recovery for incremental changes in the total pressure of the jet or in the area of the exit throat were made with the model at an angle of attack of 0°. Annular nozzles of two sizes were tested at three positions relative to the duct inlet. The nozzle throats were 0.0045 inch and 0.0090 inch in width, and the expansion ratio of both was 2.0, the area ratio that theoretically produces an outlet Mach number of 2.20. The outlets of the nozzles were moved from the station of the duct inlet by screwing the central body forward on its supporting tube. As a result, the area of the annular duct entrance was increased, but since the projected frontal area (0.2209 sq in.) was the same in each case, it has been used as the inlet area A_1.

High-pressure air was supplied to the nozzle through the piping arrangement shown in figure 1 by an air bottle located outside the wind tunnel. Since the pressure losses through the pipe were negligible, the total pressure of the air flowing through the nozzle was equal to that measured by the Bourdon gage of the pressure-regulating valve. The stagnation temperature of the ejected air was measured by a thermometer placed in the settling chamber of the supply line and was found to be nearly constant during a test. The rate of air flow through the nozzle was measured by weighing the air bottle before and after a test and timing the period during which the quick-acting release valve was open.

RESULTS AND DISCUSSION

Since an investigation in the 8- by 8-inch supersonic wind tunnel must be performed at a relatively small scale, any phenomena in which viscous effects are pronounced must include a consideration of the Reynolds number of the test and of the intended application. In the following discussion, the results of the tests are described, the data are evaluated in terms of a hypothetical propulsive system, and predictions are made to estimate the performance under full-scale flight conditions.

Wind-Tunnel Data

As shown in figure 3, a large improvement in the recovery of total pressure can be produced by ejecting high-velocity air into the boundary layer upstream of an annular duct inlet. The total pressure of the air ejected through the nozzle during the tests was limited by the strength of the model. When this pressure was the maximum, the total-pressure ratio H_2/H_3 was approximately 5.5. This ratio represents the compression ratio that would be required of the compressor of a turbo-jet engine to produce the conditions of the tests, assuming negligible pressure losses between the compressor and the nozzle. For this condition, the maximum total-pressure recovery $(H_3/H_0)_{max}$ at a free-stream Mach number of 1.36 is 94 percent, and at a Mach number of 2.01, it is 58 percent. The improvement is greatest at low supersonic speeds, and it decreases as the Mach number of the free stream approaches that of the jet. The ratio $(H_3/H_0)_{max}$ is that attained at the optimum setting of the variable exit throat of the model. Figures showing the variation of total-pressure ratio with mass-flow ratio[1] m_1/m_0

[1]The mass-flow ratio is defined as the total mass of air that enters the inlet minus the mass of air ejected from the nozzle divided by the mass that flows through a tube of the same area as the inlet in the free stream.

from which the maximum values were obtained are not shown in the present report. Except for larger total-pressure ratios, the curves are similar to those for model B of reference 1.

The position of the nozzle outlet with respect to the duct inlet has a small effect upon the maximum total-pressure ratio. As shown in figure 3, the pressure recovery is a few percent greater when the outlet is 0.250 inch ahead of the duct entrance than when it is flush with the inlet or at a position 0.340 inch upstream.

The increase in the maximum total-pressure recovery resulting from an increase in the ratio of the total pressure in the jet to that in the free stream is shown in figure 4. These data were obtained with the jet in the most favorable position tested. The ratio of the total pressure in the jet to that after diffusion through the inlet (H_2/H_3) is also shown. As the pressure of the air in the jet is increased, the maximum total-pressure ratio $(H_3/H_0)_{max}$ increases slowly, then rapidly, and finally more slowly again as it approaches the maximum value. As suggested by schlieren photographs that are presented later and as indicated by theoretical considerations, the cause of this variation is the relationship between the shape of the jet and the mixing of the jet and the boundary layer. When the pressure ratio H_2/H_0 is small, the static pressure of the ejected air is less than the static pressure of the adjoining stream. As a result, the boundary of the jet is convergent near the outlet, and the Mach number of the air flowing into the duct is relatively high because of the expansion caused by this convergent boundary. For such conditions, the jet produces very little improvement in the pressure recovery. As H_2/H_0 is increased, the static pressure in the jet and in the adjoining stream become equal. Then, the boundary of the jet is parallel to the stream direction, the average inlet Mach number is reduced, and the pressure recovery is improved. The total-pressure ratios H_2/H_0 at which this condition occurs have been estimated from theoretical considerations, and are shown in figure 4. When the total-pressure ratio across the nozzle is increased further, the jet boundary is divergent, and an oblique shock wave originates from the outlet. Consequently, the inlet Mach number is further reduced, and the pressure recovery is again improved. It is also possible that the mixing between the jet and the boundary layer increases as the shape of the jet changes from convergent to divergent and thus augments the variation of pressure recovery with nozzle-pressure ratio. Eventually, the oblique shock wave would become a detached normal wave, but before such a change can occur, the pressure rise through the oblique wave becomes sufficiently large to cause the boundary layer to thicken upstream of the nozzle outlet. As this occurs, the oblique shock wave moves forward and the boundary layer behind it thickens still more because of the rise in pressure. Then, more retarded air flows through the inlet, and the rate of increase in the maximum total-pressure recovery decreases.

In order to show the effects of nozzle size, the variation of maximum total-pressure ratio with the ratio of the mass of air flowing through the jet to that of the free stream flowing through the duct inlet is shown in figure 5 for the two nozzles tested. Doubling the mass of air in the jet by doubling the size of the nozzle has very little effect upon the total-pressure recovery. This fact suggests that nozzles smaller than those of the tests with correspondingly smaller mass flows might produce the same pressure recovery. However, no tests could be made of such models because of the difficulties involved in accurately constructing a smaller nozzle.

Schlieren photographs of the flow at a Mach number of 1.70 with the jet operating at a nozzle-pressure ratio H_2/H_0 of 3.05 are shown in figure 6 for various outlet-inlet area ratios A_4/A_1. The general nature of the flow is the same as that described in reference 1. At outlet-inlet area ratios above 1.15, the flow through the inlet is supersonic and the mass-flow ratio is nearly constant. An oblique shock wave occurs at the lip of the jet outlet because of the deflection of the stream resulting from the divergent boundary of the jet. Reducing the outlet-inlet area ratio increases the total-pressure recovery to the maximum value, and a further reduction causes the flow to fluctuate as shown in the photographs taken consecutively at an area ratio of 0.94. These pictures show that the jet prevents the violent separation of the boundary layer that occurs at the corresponding area ratios, but at lower total-pressure ratios, with the models of references 1 and 2. Further reduction of the outlet area causes the boundary layer to separate as with the other models. The boundary layer separates upstream of the jet outlet, a fact which indicates that the influence of the adverse pressure gradient resulting from the deceleration of the flow from supersonic to subsonic speeds extends through a subsonic portion of the flow that has not been mixed with the supersonic jet.

Schlieren photographs showing the effects of the total pressure in the jet are presented in figure 7 in order to substantiate the explanation given for the shape of the curves of figure 4. When the nozzle-pressure ratio H_2/H_0 is low, an expansion is seen to occur at the lip of the nozzle outlet. At the intermediate pressure ratio, only very weak disturbances are visible, and at the maximum nozzle-pressure ratio, an oblique shock wave is seen to originate from the outlet.

Equivalent Pressure Recovery

Although a large increase in the total-pressure recovery is produced by ejecting high-velocity air into the boundary layer upstream of a duct entrance, the thrust force created by a turbojet engine in a supersonic airplane would not be increased correspondingly. If the engine must furnish the air, all of the energy available could not be converted into thrust because some of the power from the turbine would have to be used in the recirculation process.

In order to compare the performance of an intake which utilizes recirculated air to delay separation of the boundary layer with that of other types of inlets, an equivalent total-pressure ratio of an assumed propulsive system has been calculated. This "equivalent" total-pressure ratio is defined as the pressure recovery required of the intake for an engine having no recirculation that would produce a thrust force equal to that of an engine utilizing recirculation, provided that the same mass of air flows through both engines. The following conditions were assumed in the analysis:

1. The efficiencies of the compressors are equal.

2. The flow through the compressors is isentropic.

3. The compressor of the system having no recirculation produces a compression ratio[2] of 4.0.

4. The recirculated air is compressed to the pressure ratio H_2/H_3 of the test results, and the pressure losses in the recirculating system are negligible.

5. The energy available from the two turbines is the same whether air is recirculated or not, and the thrust of the engines is proportional to the product of the respective total-pressure ratios across the intake systems and the pressure ratios of the compressors.

6. The static temperature of the air in the free stream is -67° F, the temperature in the isothermal atmosphere.

The computations were made in three steps:

---

[2]The compression ratio is defined as the total pressure at the outlet of the compressor divided by the total pressure at the inlet of the compressor.

1. The work required to compress 1 pound of air per second through a compression ratio of 4 was calculated by the method of reference 3. This amount of work is constant for all conditions at a given stagnation temperature, or Mach number, and represents the power required from the turbine.

2. The difference between the power required from the turbine as determined in step 1, and the power used in compressing the recirculated air was used to compute the reduced compression ratio of the compressor of an engine utilizing recirculation.

3. The equivalent total-pressure recovery was then calculated by multiplying the measured pressure recovery of the inlet through which air was recirculated by the reduced compression ratio (step 2) and dividing by the original ratio of 4.

Therefore, the feasibility of a propulsive system in which some of the air is recirculated depends upon whether the improvement in the measured recovery overcompensates for the decrease in the compression ratio of the compressor.

The equivalent total-pressure ratios that correspond to the conditions of figure 5 for the nozzle having a throat width of 0.0045 inch are shown in figure 8. The results for the nozzle having a 0.0090-inch throat are not shown because the mass of air that must be recirculated to produce a given pressure recovery is nearly twice that of the smaller nozzle; therefore, the equivalent total-pressure ratio is much smaller. The maximum equivalent recovery for the Mach number range of the tests occurs when the mass of air expelled is about 14 percent of the mass of free-stream air that flows through the inlet.

The variation with free-stream Mach number of the maximum equivalent total-pressure ratio and the recovery measured at the corresponding recirculated mass-flow ratio are shown in figure 9. The equivalent recovery is about 8 percent greater than the total-pressure ratio of a model having no air ejected into the boundary layer. This improvement is about equal to that attained by a model having an annular entrance with 5° ramp. (See reference 2.) Although this increase in the pressure recovery appears small and not worth the complication of the ducting required to recirculate the air, the equivalent total-pressure ratio attained at a larger scale will probably be somewhat larger.

Scale Effects

If the tests were performed under full-scale flight conditions, it is probable that the equivalent total-pressure recovery would approach the maximum total-pressure ratios measured in the wind tunnel. Assuming that the same type of flow exists in either case, the thickness of the boundary layer on the small model is larger in proportion to the size of the model than it would be on an airplane in flight. The Reynolds number of the flow over an equivalent airplane having a forebody diameter of 6 feet and flying at a Mach number of 1.70 at an altitude of 60,000 feet would be about 13 times that occurring in the wind-tunnel tests. The ratio of the boundary-layer thickness at corresponding longitudinal stations to the forebody diameter would be about a quarter of that of the small-scale model if the boundary layer, as in the model tests, were laminar. Assuming that the same ratio of the mass of air in the boundary layer to the mass of air in the jet would be required, roughly one-fourth of the mass ratio m_2/m_1 indicated in the wind-tunnel tests would be required to produce a given pressure recovery in flight. For such a reduction in the mass of the recirculated air, the equivalent pressure recovery would surpass the measured recovery of a duct system having no recirculation by about 17 percent. As suggested by the data of figure 5, a proportionately smaller nozzle than that of the wind-tunnel tests might be used on a larger installation to produce the same maximum total-pressure ratios. If so, less air would be recirculated and the equivalent recovery would be still larger.

CONCLUSIONS

Tests at Mach numbers between 1.36 and 2.01 of a model having a nozzle upstream of an annular duct inlet for the purpose of ejecting high-velocity air into the boundary layer of the flow along the forebody have shown the following effects:

1. The maximum total-pressure ratio attainable after diffusion is greater than that of a comparable model having no air ejected.

2. The causes of the greater recovery are the delay in the separation of the boundary layer resulting from mixing between the boundary layer and the jet and also the reduction in the intake Mach number caused by an oblique shock wave originating from the nozzle outlet when the free boundary of the jet is divergent.

3. If the high-pressure air ejected through the nozzle were supplied by the compressor of an assumed turbo-jet engine, the pressure recovery effective in producing thrust is about 8 percent greater for an engine that utilizes recirculation than for an engine that does not.

Ames Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Moffett Field, Calif.

REFERENCES

1. Davis, Wallace F., Brajnikoff, George B., Goldstein, David L., and Spiegel, Joseph M.: An Experimental Investigation at Supersonic Speeds of Annular Duct Inlets Situated in a Region of Appreciable Boundary Layer. NACA RM No. A7G15, 1947.

2. Davis, Wallace F., and Goldstein, David L.: Experimental Investigation at Supersonic Speeds of Twin-Scoop Duct Inlets of Equal Area. I – An Inlet Enclosing 61.5 Percent of the Maximum Circumference of the Forebody. NACA RM No. A7J27, 1947.

3. Keenan, Joseph H., and Kaye, Joseph: Thermodynamic Properties of Air Including Polytropic Functions, John Wiley & Sons, Inc., 1945.

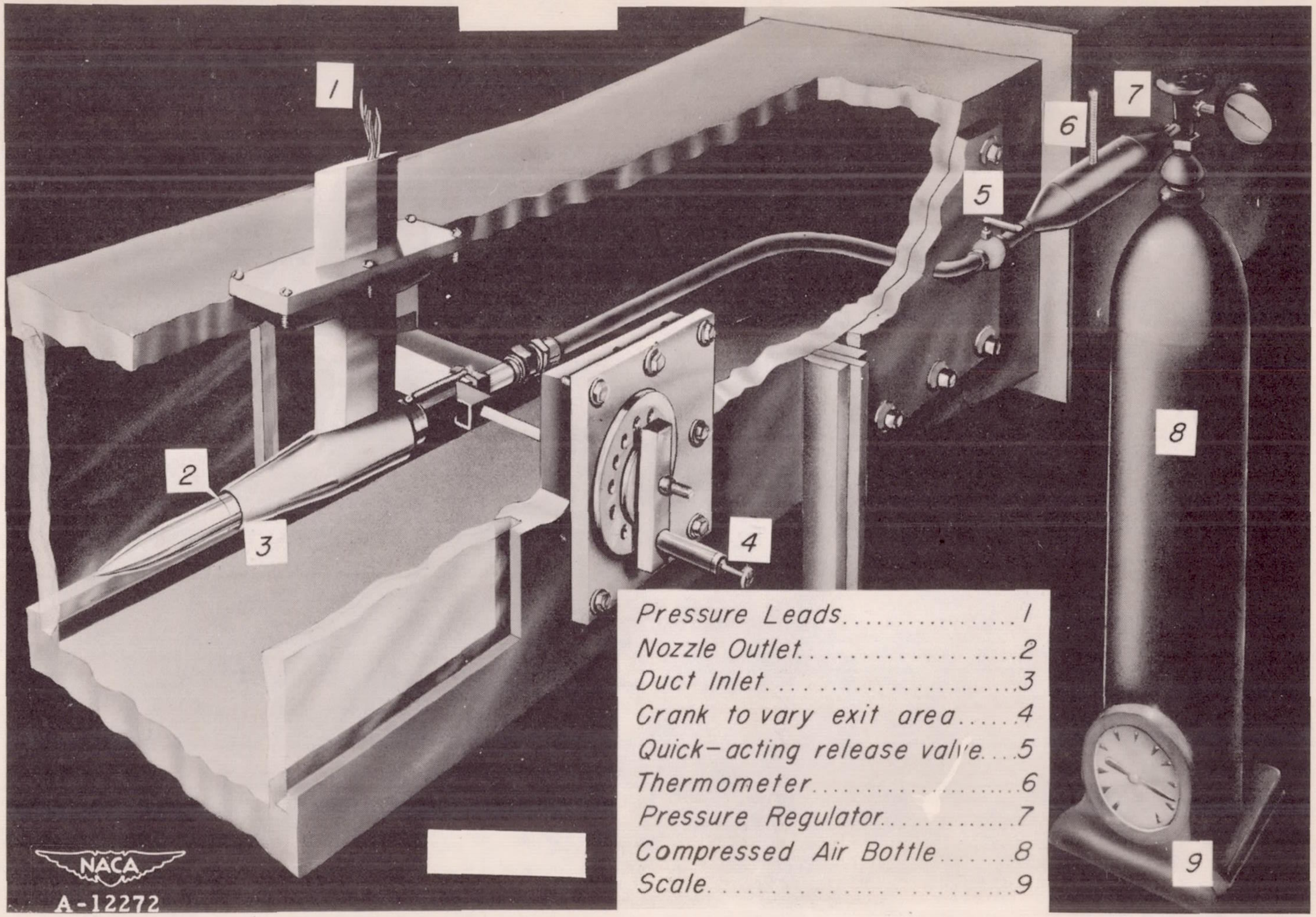
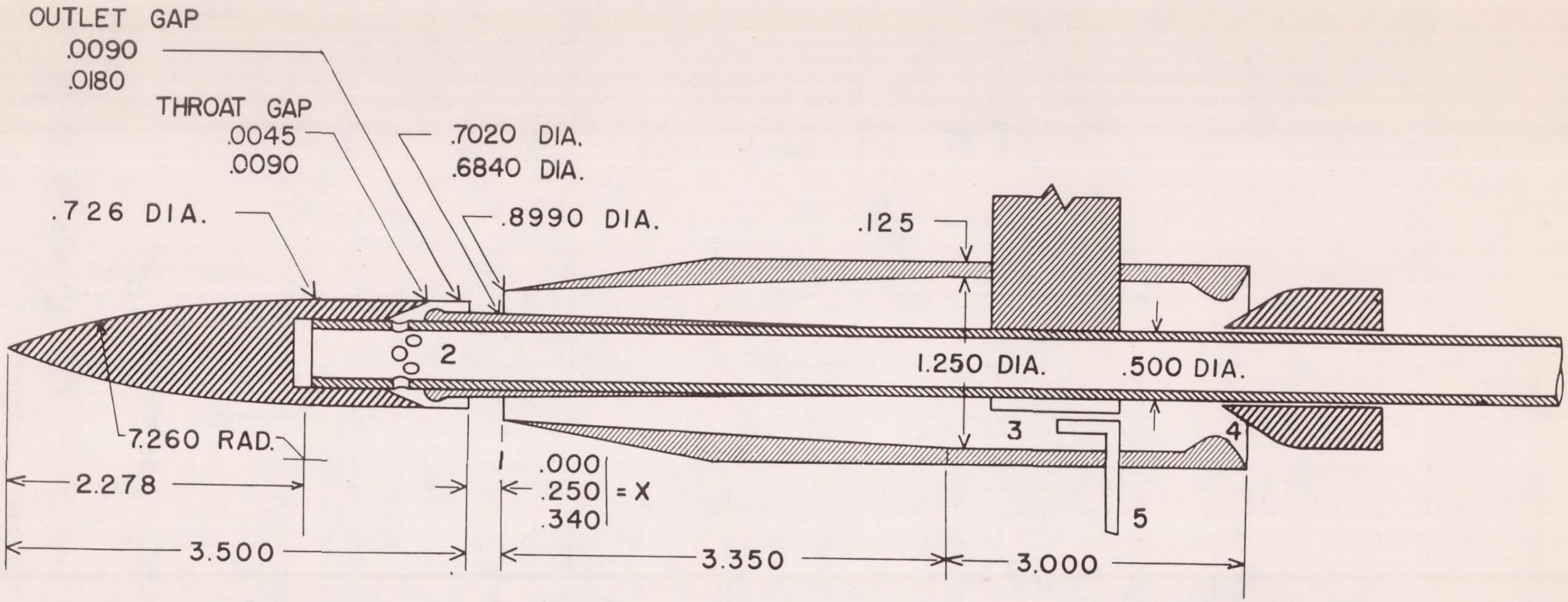


Figure 1.— Model installation in the Ames 8— by 8—inch supersonic wind tunnel.



DUCT ENTRANCE 1
RESERVOIR OF THE NOZZLE 2
SETTLING CHAMBER 3
EXIT THROAT 4
PITOT TUBE 5

ALL DIMENSIONS IN INCHES.

FIGURE 2. – MODEL DIMENSIONS.

NACA
A-12298

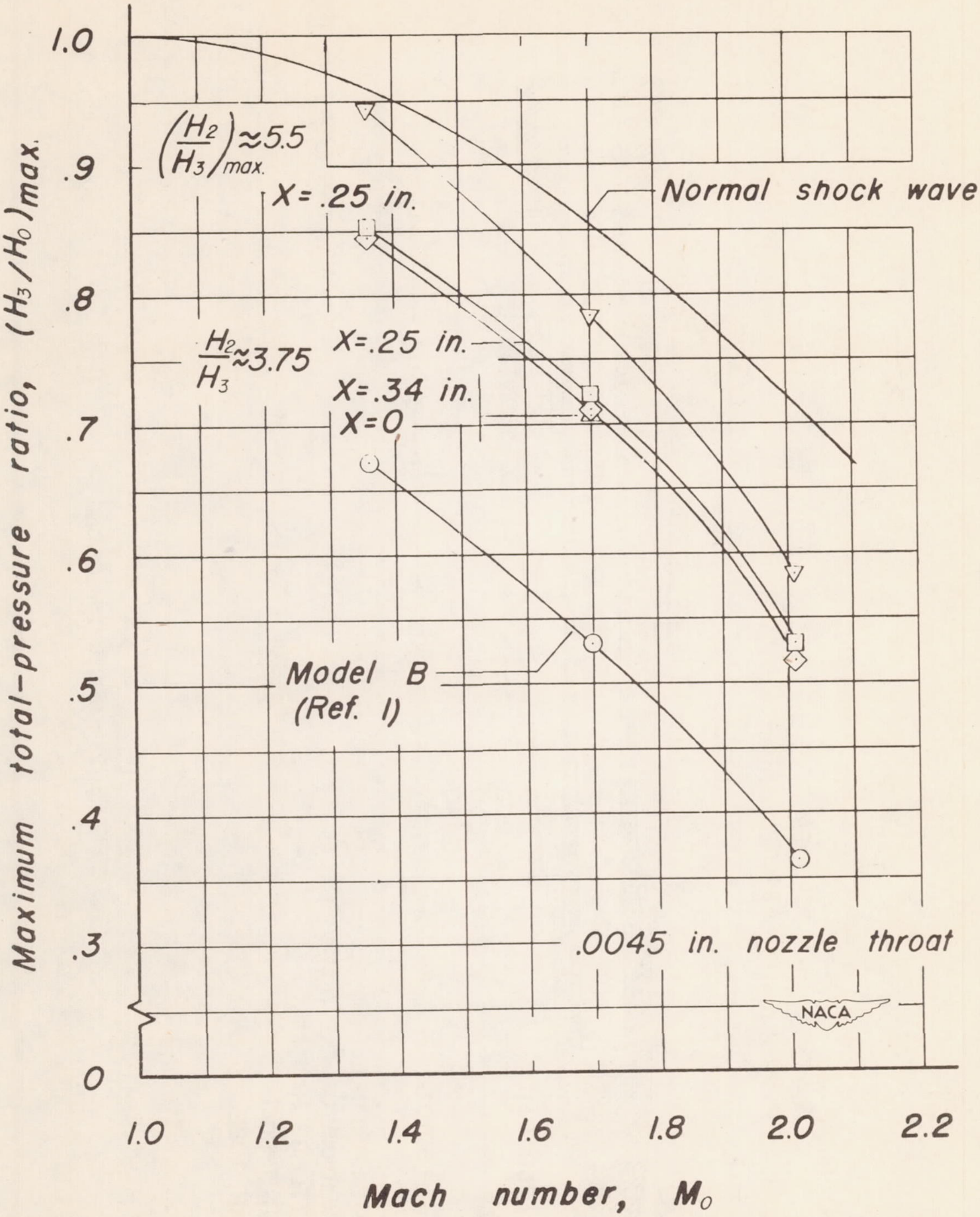


Figure 3.—Variation of maximum total-pressure ratio with Mach number and nozzle position.

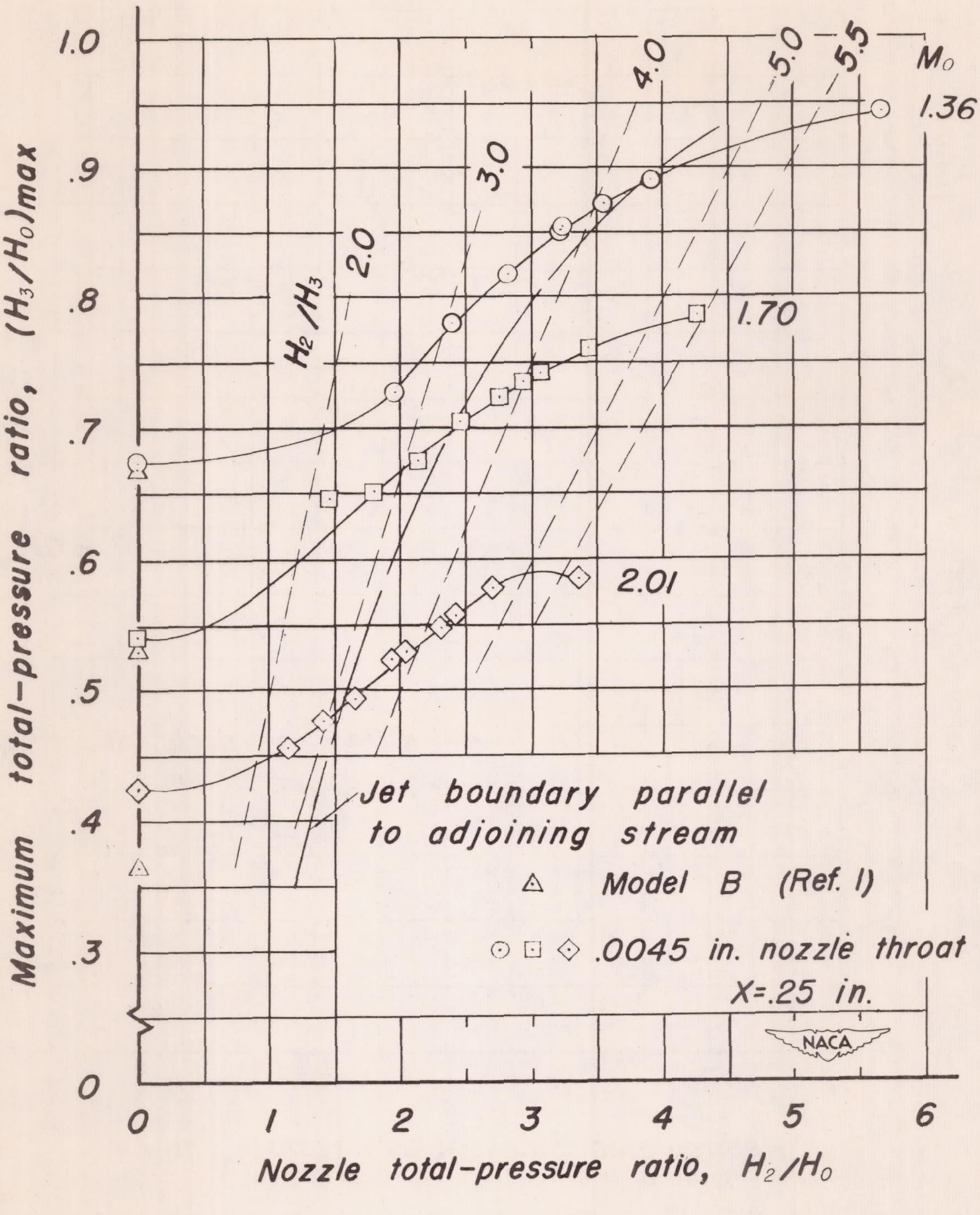


Figure 4. —Variation of maximum total-pressure ratio with nozzle total-pressure ratio for several Mach numbers.

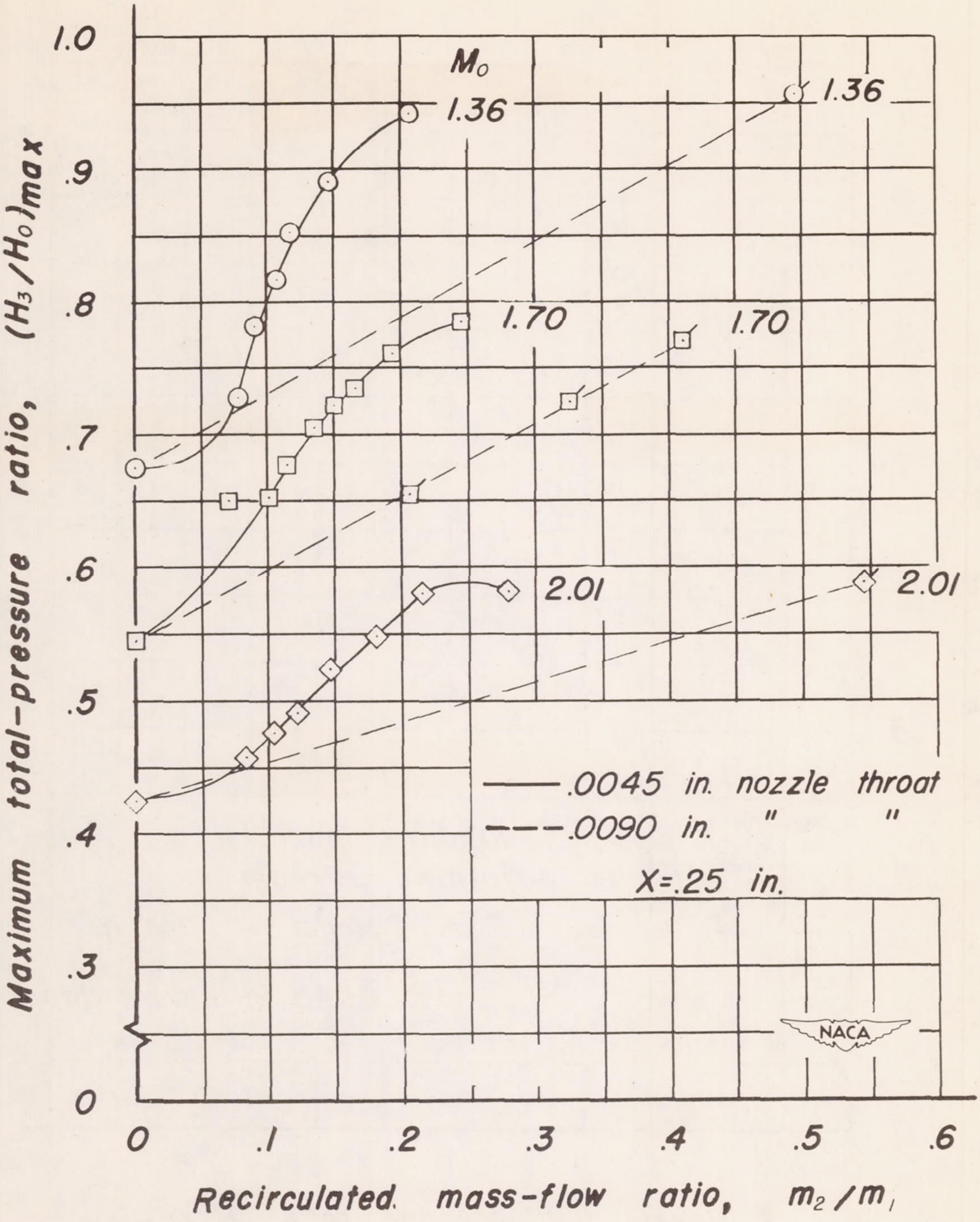
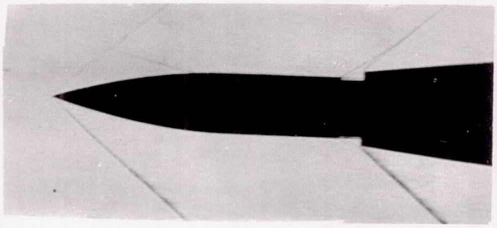
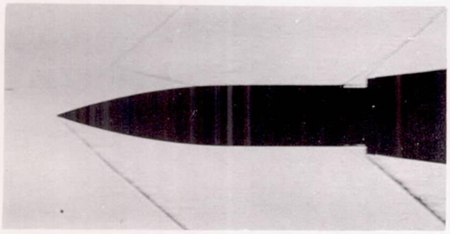
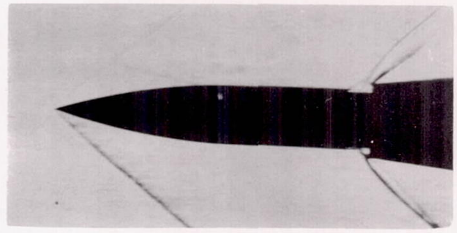


Figure 5.—Variation of maximum total-pressure ratio with recirculated mass-flow ratio for several Mach numbers.

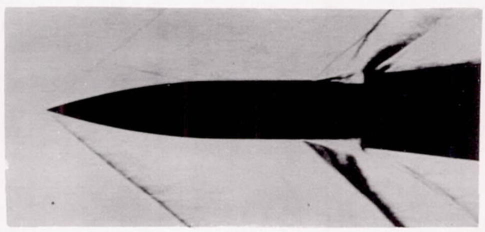


$\frac{A_4}{A_1} = 1.15$ $\frac{m_1}{m_0} = 0.99$ $\frac{H_3}{H_0} = 0.69$





$\frac{A_4}{A_1} = 0.94$ $\frac{m_1}{m_0} = 0.93$ $\frac{H_3}{H_0} = 0.74$

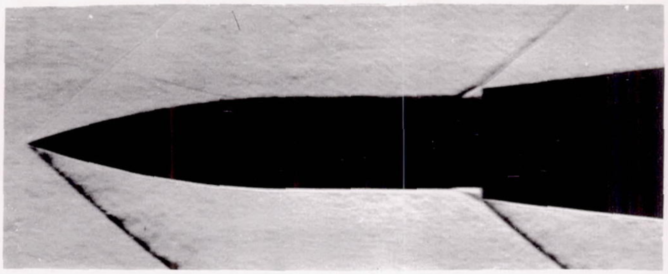


A-12343

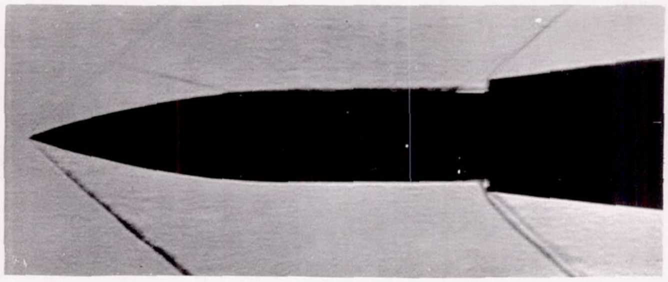
$\frac{A_4}{A_1} = 0.75$ $\frac{m_1}{m_0} = 0.66$ $\frac{H_3}{H_0} = 0.66$

Note: Knife-edge parallel to the stream direction: $\frac{H_2}{H_0} = 3.05$; $x = 0.25$ in.

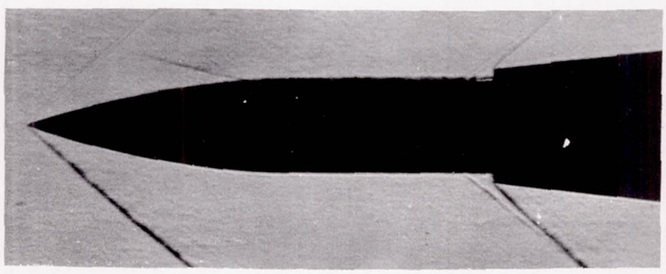
Figure 6.- Schlieren photographs of the flow at a Mach number of 1.70 about the model with various outlet-inlet area ratios.



$\frac{A_4}{A_1} = 1.14$ $\frac{m_1}{m_o} = 0.99$ $\frac{H_2}{H_o} = 1.80$ $\frac{H_3}{H_o} = 0.652$



$\frac{A_4}{A_1} = 1.01$ $\frac{m_1}{m_o} = 0.98$ $\frac{H_2}{H_o} = 2.45$ $\frac{H_3}{H_o} = 0.705$



$\frac{A_4}{A_1} = 0.87$ $\frac{m_1}{m_o} = 0.91$ $\frac{H_2}{H_o} = 4.25$ $\frac{H_3}{H_o}$ 0.786

NACA

A-12344

Note: Knife-edge parallel to the stream direction x = 0.250 in.

Figure 7.- Schlieren photographs of the flow at a Mach number of 1.70 about the model with various nozzle-pressure ratios.

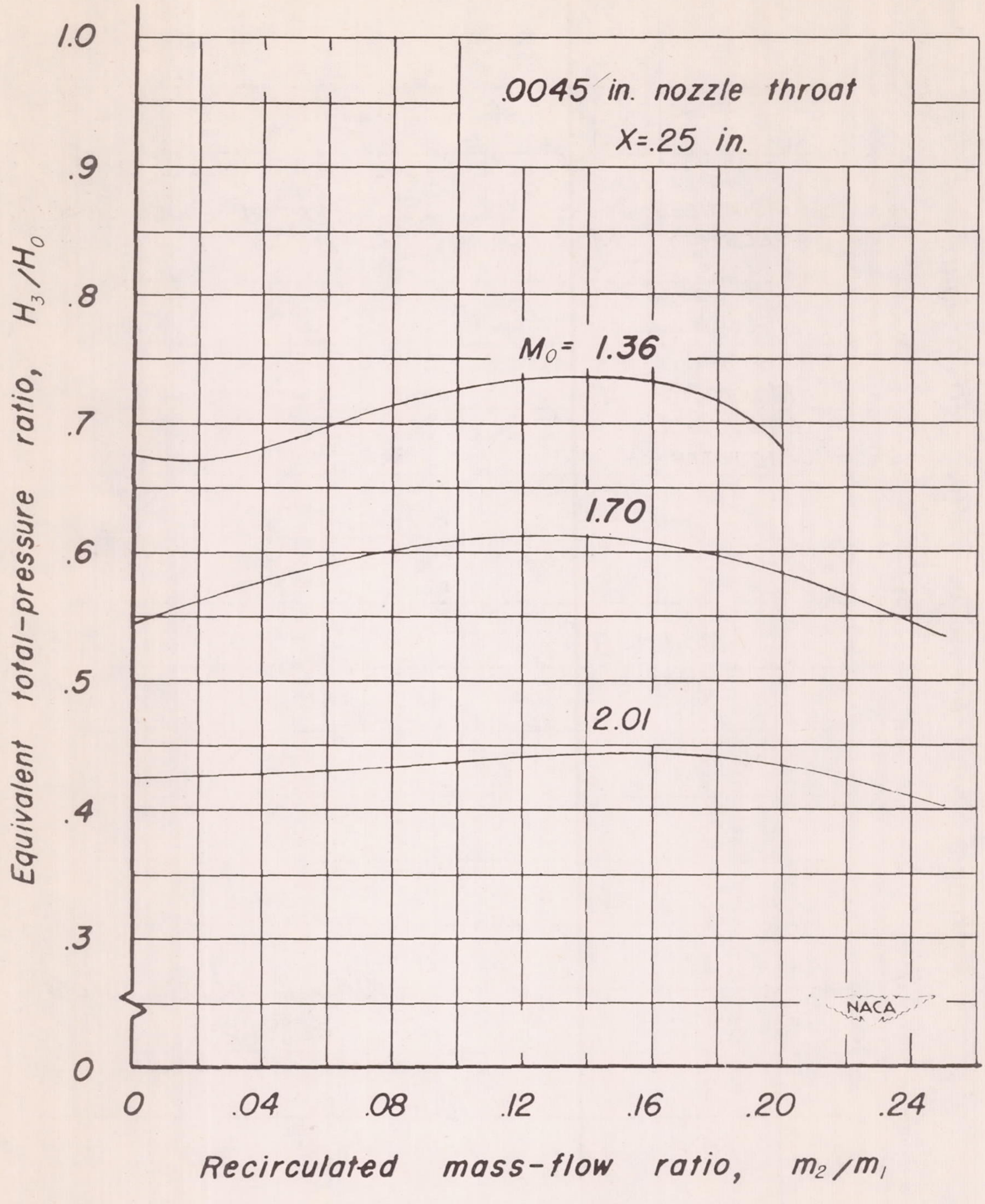


Figure 8.—Variation of equivalent total-pressure ratio with recirculated mass-flow ratio.

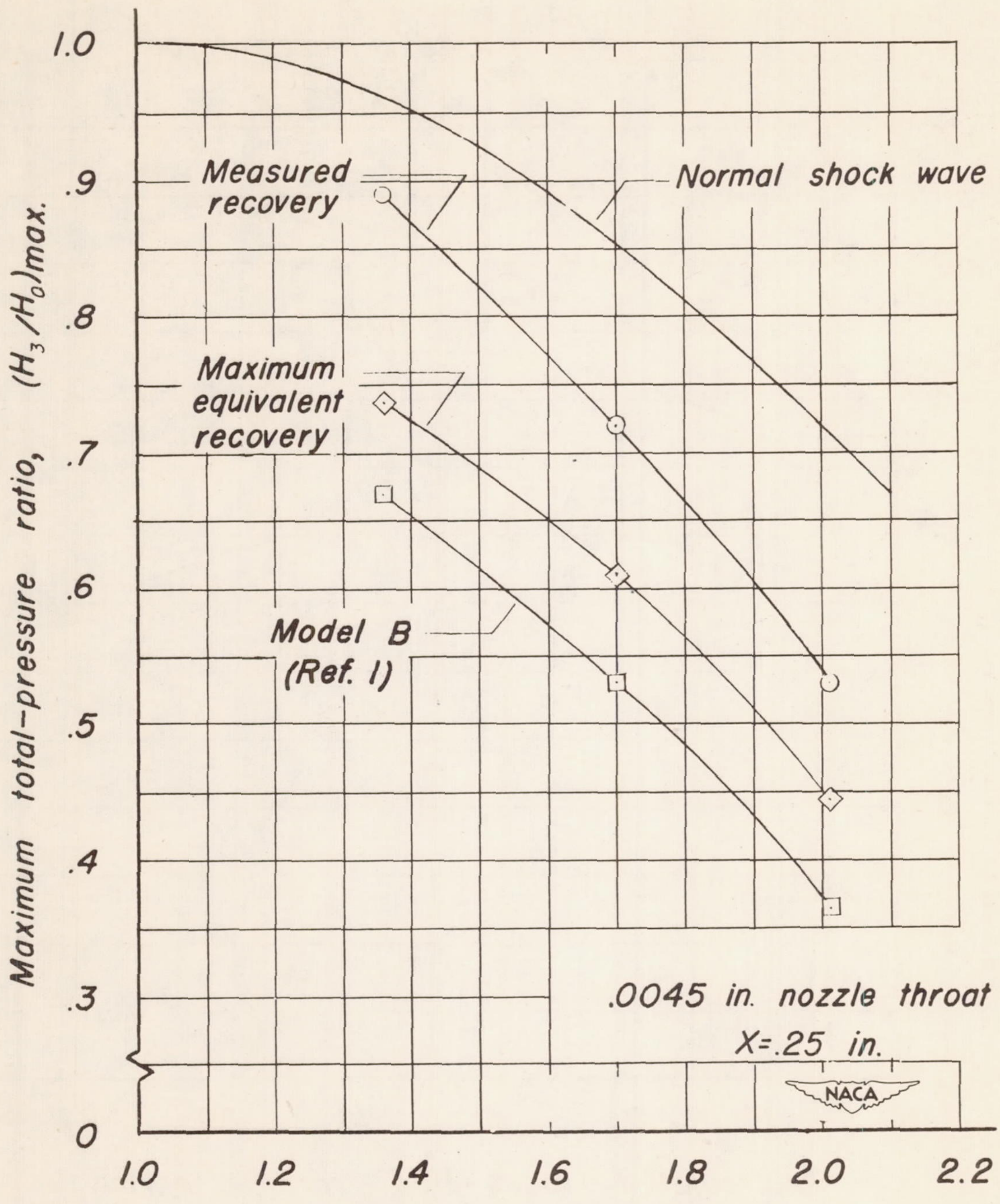


Figure 9. —Variation of maximum equivalent total-pressure ratio and the corresponding measured ratio with Mach number.