[redacted]

Copy
RM E53A12

[redacted]

RESEARCH MEMORANDUM

INVESTIGATION OF PRESSURE RECOVERY OF A SINGLE-CONICAL-SHOCK NOSE INLET AT MACH NUMBER 5.4

By Harry Bernstein and Rudolph C. Haefeli

Lewis Flight Propulsion Laboratory
Cleveland, Ohio

[redacted]

[redacted]

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

April 6, 1953 [redacted]

[redacted]

[redacted]

[redacted]

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

INVESTIGATION OF PRESSURE RECOVERY OF A SINGLE-CONICAL-SHOCK NOSE INLET AT MACH NUMBER 5.4

By Harry Bernstein and Rudolph C. Haefeli

SUMMARY

An experimental investigation of the performance of a single-conical-shock diffuser was conducted at a Mach number of 5.4 and a Reynolds number based on model diameter of 375,000. Total-pressure recoveries of 13.7 and 13.1 percent were obtained at angles of attack of 0° and 3°, respectively. The corresponding kinetic energy efficiencies were 86.4 percent at an angle of attack of 0° and 86.0 percent at an angle of attack of 3°.

INTRODUCTION

In recent years, consideration has been given to the possibility of flight at speeds many times that of sound, that is, at hypersonic Mach numbers. A number of theoretical and experimental investigations concerned with the aerodynamics of wings and bodies at these flight Mach numbers have been reported. Some thought has also been given to the power-plant requirements for hypersonic missiles. Among the engines capable of providing sufficient thrust in the lower range of hypersonic Mach numbers (M_0 = 4 to 10) is the ram-jet engine. However, both analytical and experimental information regarding the operating characteristics of the ram jet at these Mach numbers are meager.

An analytical investigation of ram-jet engine performance in the Mach number range 3 to 7 is reported in reference 1. For this analysis a high-efficiency diffuser (92-percent kinetic energy efficiency) was assumed to be available in order to estimate the maximum performance that might be expected at these moderate hypersonic velocities. The validity of any assumptions upon which engine performance is based must be determined by experimental investigation.

To obtain an indication of the merits of a nose inlet for application to ram-jet engines or auxiliary air supplies, an experimental investigation of the performance of a single-conical-shock diffuser designed for a Mach number of 5.4 was undertaken at the NACA Lewis laboratory. Inasmuch as no effort was made to optimize the performance of the diffuser by modifying the original design, the results herein are preliminary.

SYMBOLS

The following symbols are used in this report: 2813

A stream cross-sectional area

M Mach number

m mass-flow rate

P total pressure

α angle of attack

γ ratio of specific heats

η_{KE} kinetic energy efficiency,

$$\frac{\text{kinetic energy of air expanded isentropically from diffuser exit to free-stream static pressure}}{\text{free-stream kinetic energy}}$$

Subscripts:

0 free-stream tube entering inlet

1 combustion-chamber conditions

2 station of minimum area at diffuser exit

APPARATUS

The tests were conducted in the Lewis 6- by 6-inch continuous-flow hypersonic tunnel at a Mach number of 5.4. The test section total pressure was between 78 and 86 pounds per square inch absolute, with a variation of ± 0.5 pound per square inch during any one run. The stagnation temperature was $225 \pm 6^{\circ}$ F. These inlet conditions were sufficient to avoid condensation of the air components, as evidenced by use of the light scattering technique described in reference 2. The test section Reynolds number, based on an average total pressure of 83 pounds per square inch absolute and on the model diameter, was 375,000.

Figure 1 is a photograph of the tunnel showing the model in its test position. The model was placed far up into the first test rhombus to avoid the effects of the large sidewall boundary layer due to secondary flow in the nozzle (ref. 3).

The model, shown in figures 2 and 3, was a single-conical-shock nose inlet with a design Mach number of 5.4. The theoretically optimum cone half-angle (27°) was determined by extrapolation from Mach number
2813 5.0 of the calculations presented in reference 4. The internal angle of the cowl lip was designed to cause an oblique shock in the diffuser entrance. The inlet had an internal contraction ratio equal to the Kantrowitz ratio for the average Mach number behind this shock. To maintain a high mass flow, the cowl-lip external angle (31.9°) was kept less than the limiting angle for shock attachment at Mach number 5.4 (41.5°).

The instrumentation for measuring combustion-chamber pressures is shown in figures 2(b) and 3. The eight pitot tubes were made from 0.050-inch outside diameter steel tubing with the openings flattened to inside dimensions of 0.002 by 0.040 inch. The three static orifices had diameters of 0.021 inch. The pressures were read on a mercury manometer.

The pitot and static probes described in reference 3 were used to determine the free-stream conditions. The corresponding pressures were measured with mercury and butyl-phthalate manometers, respectively.

REDUCTION OF DATA

The results of a Mach number survey at three axial stations in the test section of the Lewis 6- by 6-inch tunnel are presented in figure 4. These stations were $9\frac{1}{8}$, 16, and 18 inches downstream of the tunnel throat; the cone tip of the model was located $15\frac{3}{8}$ inches from the tunnel throat. The Mach numbers, determined by use of the Rayleigh equation from pitot and static pressure measurements, were reproducible within 2 percent. Inasmuch as the variations from Mach number 5.4, indicated in figure 4, are generally within the reproducibility, a nominal Mach number of 5.4 was chosen for computations of diffuser performance.

The test section pitot pressure was measured at locations approximately 1 inch ahead of the nose of the model before each run. A value of the free-stream total pressure was computed from these readings and from the normal-shock relation for Mach number 5.4.

The pressure recoveries of the model were based on an arithmetic average of the eight pitot-pressure readings in the combustion chamber. This method of averaging was believed to be sufficiently accurate, as differences between the eight readings were, in most cases, less than 1/2 inch of mercury. Because of the unsymmetrical location of the pitot tubes with the model at angle of attack, the pressures were measured at both positive and negative values of the same α, and the 16 pitot readings were averaged in the computation of the pressure recovery. For this method, the probable error in the maximum recovery is estimated to be about 1 percent of its value. 2813

The evaluation of diffuser mass-flow ratio was based on the average of the three combustion-chamber static readings (six readings at angle of attack) and on a Mach number computed from the ratio of the minimum exit area to the combustion-chamber area A_2/A_1. The sharp turning angle and subsequent flow separation or vena contracta at the exit necessitated the application of a correction factor to the geometric outlet areas. This factor was calculated to yield a mass-flow ratio of unity when schlieren observations indicated that the inlet was capturing the entire free-stream tube. For supercritical operation at angles of attack of 0° and 3°, the correction factor increased monotonically from 0.450 to 0.478 as the outlet area was increased. In the subcritical range the correction factor was assumed to have the value 0.450. As a check on this method of mass-flow-ratio computation, effective combustion-chamber total pressures, based upon the measured static pressures and the computed Mach numbers, were computed. These pressures showed good agreement with the measured total pressures.

DISCUSSION OF RESULTS

Schlieren photographs of the flow configuration for the diffuser at zero angle of attack are presented in figure 5. The boundary layer on the cone was observed to separate and the mass flow through the inlet was therefore reduced. Subcritical operation of the diffuser, for this configuration and all others to be discussed, was unstable (buzz).

To avoid the boundary-layer separation, number 80 silicon carbide grain was fixed to the tip of the cone to promote artificial transition from a laminar to a turbulent boundary layer. Schlieren photographs of the diffuser with this artificial transition are shown in figure 6. The roughening of the cone tip was a sufficient measure to avoid separation during supercritical diffuser operation. A slight amount of mass flow, however, was still observed to be spilling over the cowl. This spillage was probably due to the effect of the greater displacement thickness of the turbulent boundary layer on the cone surface, as compared with that of the laminar boundary layer assumed in the design of the diffuser.

So that all the mass flow would be captured, the cone was retracted into the inlet a distance of 0.01 inch. This distance was determined from the schlieren photographs of figure 6 and represents the retraction necessary to make the cone shock intersect the cowl leading edge.

Schlieren photographs of the diffuser with the cone retracted are presented in figure 7. From these and similar photographs it was determined that the mass-flow ratio of the diffuser when operating supercritically was unity for all values of exit area. The exit area correction factors, discussed in the section REDUCTION OF DATA, were therefore based upon the data for the inlet with the cone retracted and with artificial transition.

Figures 8 and 9 present diffuser characteristics without and with cone retraction, respectively. The mass-flow ratio before the cone was retracted was 96 percent and the maximum total-pressure recovery was 14.4 percent. This pressure recovery represents a kinetic energy efficiency of 87.1 percent at the operating Mach number of 5.4, as determined from the equation

$$\eta_{KE} = 1 - \frac{\left(\frac{P_0}{P_1}\right)^{\frac{\gamma-1}{\gamma}} - 1}{\frac{\gamma-1}{2} M_0^2}$$

For comparison, the theoretical values of total-pressure recovery and kinetic energy efficiency calculated for this diffuser, with the assumption of an internal contraction ratio equal to the Kantrowitz ratio for the entrance Mach number, were 19.2 percent and 89.4 percent, respectively. Retracting the cone resulted in a decrease of the maximum recovery to 13.7 percent, which represents a kinetic energy efficiency of 86.4 percent. The maximum recovery decreased because of the reduction in the internal contraction ratio when the cone was retracted. All the flagged data in figures 8 and 9 represent subcritical (unstable) operation. These data, although unreliable quantitatively because of the instability of the flow, indicate the magnitude of the reduction in pressure recovery in the subcritical region.

Schlieren photographs of the diffuser at angles of attack between 2° and 4° are presented in figure 10. With the cone in the unretracted position, the inlet spilled a significant amount of the flow at an angle of attack of 2° even at large outlet area ratios A_2/A_1, as seen in figure 10(a). With the cone retracted, the inlet operated with a high mass-flow ratio at angles of attack of 3° and 4° throughout the supercritical range (figs. 10(b) and 10(d)).

For supercritical operation near maximum recovery, separation of the boundary layer on the low pressure side of the cone (within circle, fig. 10(c)) caused increased flow deflection upstream of the diffuser entrance. This resulted in the formation of a bow wave in front of the cowl in the region of the separation. A slight reduction in the mass-flow ratio therefore occurred.

The schlieren photograph presented in figure 10(e) for an angle of attack of 4° illustrates the shock configuration typical for subcritical operation at all angles of attack ($\alpha = 2^{\circ}$ to 4°). The flow was completely separated from the low-pressure side of the cone, while on the high-pressure surface the shock oscillated rapidly (buzz).

2813

Diffuser characteristics at an angle of attack of 3° are presented in figure 11. A peak recovery of 13.1 percent ($\eta_{KE} = 86.0$ percent) was obtained. Both pressure recovery and mass-flow ratio were observed to decrease rapidly as the outlet area was decreased beyond that for critical operation (maximum recovery).

A performance comparison with similar inlets tested at lower Mach numbers (ref. 5) is presented in figure 12. The kinetic energy efficiencies of the diffusers decrease from 96 to 86 percent as the flight Mach numbers increase from 1.85 to 5.4. Because the thrust coefficient of a ram-jet engine is proportional to the square root of kinetic energy efficiency, the assumption of a 92 percent η_{KE}, as made in the analysis of reference 1, results in no great error in the computation of thrust for Mach numbers up to 5.4.

The combustion-chamber cross-sectional area of an engine is dependent upon the total-pressure recovery of the diffuser (continuity equation). For large Mach numbers and corresponding experimentally probable values of pressure recovery, small changes in kinetic energy efficiency result in large changes in total-pressure recovery. This is illustrated by the fact that a 92-percent efficiency represents a recovery of 27.2 percent, whereas an 86-percent efficiency represents a recovery of only 13.1 percent at a Mach number of 5.4. The combustion-chamber areas reported in reference 1 for $\eta_{KE} = 0.92$ therefore differ considerably at the larger flight Mach numbers from those based upon experimental values of kinetic energy efficiency. This difference is shown in the following table for two values of combustion-chamber Mach number:

2813

Flight (design) Mach number	A_1/A_0 for $M_1 = 0.15$		A_1/A_0 for $M_1 = 0.20$	
	Based on η_{KE} = 92 percent	Based on experimental values of η_{KE}	Based on η_{KE} = 92 percent	Based on experimental values of η_{KE}
3.0	1.492	1.460	1.121	1.107
3.5	1.087	1.152	.816	.873
4.5	.595	.851	.456	.645
5.0	.463	.822	.380	.623
5.4	.384	.834	.296	.632

For the larger flight Mach numbers, the required combustion-chamber areas are still less than the diffuser-inlet area, thereby providing space for auxiliary equipment. Greater amounts of usable volume may be obtained only if larger combustion-chamber Mach numbers can be tolerated.

CONCLUSIONS

An investigation of the performance of a single-conical-shock nose inlet was performed in the Lewis 6- by 6-inch hypersonic tunnel at a Mach number of 5.4 and a Reynolds number based on model diameter of 375,000. At the test Reynolds number it was found necessary to induce artificial transition of the boundary layer on the cone to avoid separation of the boundary layer and subsequent reductions in the mass flow.

From this investigation the following results and conclusions were obtained:

1. At zero angle of attack, a total-pressure recovery of 13.7 percent was obtained, whereas at an angle of attack of 3° the recovery was 13.1 percent. The kinetic energy efficiencies corresponding to these recoveries were 86.4 and 86.0 percent, respectively.

2. Subcritical operation of this diffuser was unstable (buzz).

3. The required combustion-chamber areas, as computed from experimental total-pressure recoveries, were smaller than the inlet areas of the diffuser for combustion-chamber Mach numbers of 0.15 and 0.20.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio

REFERENCES

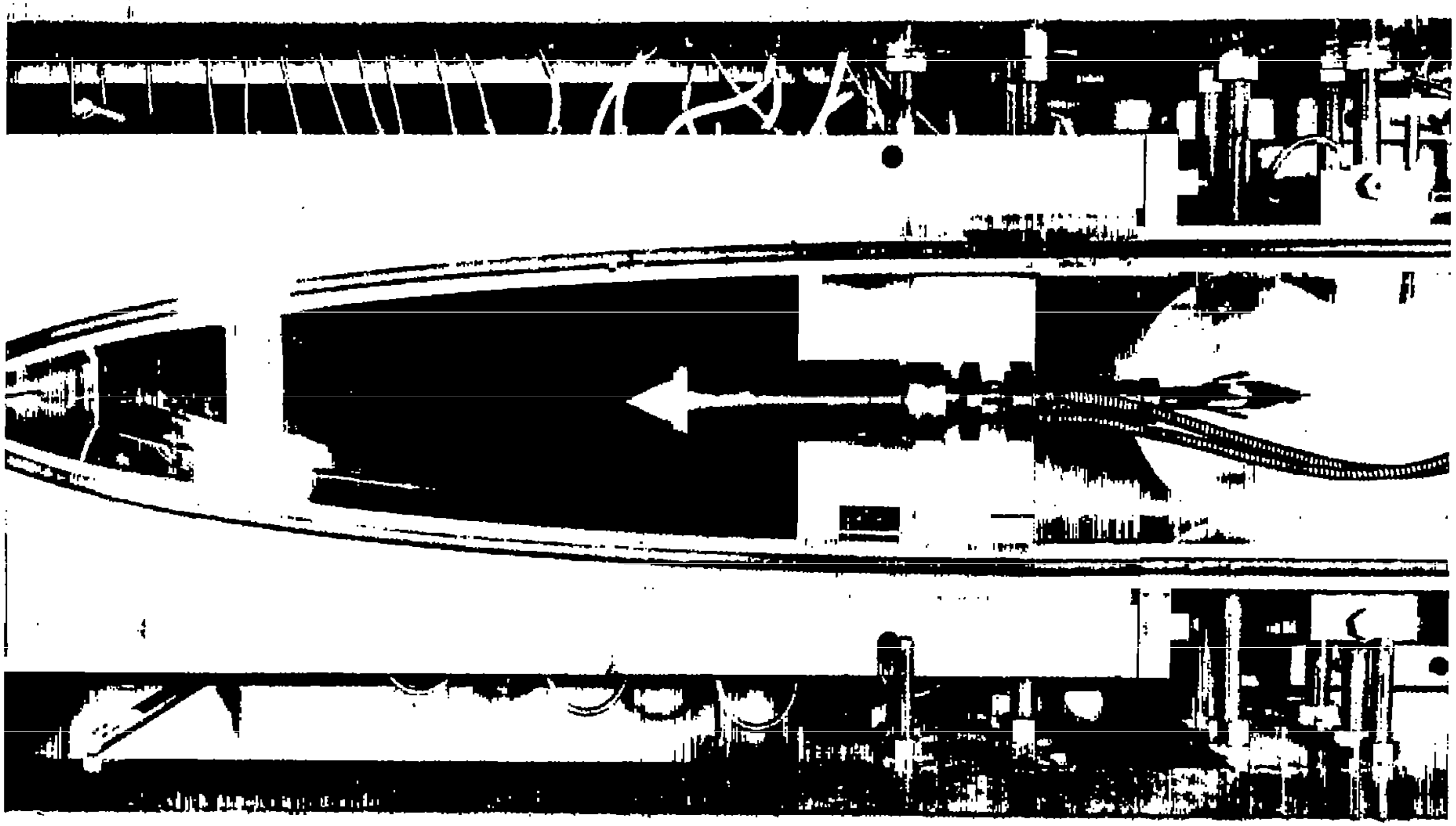
1. Evans, Philip J., Jr.: Analytical Investigation of Ram-Jet-Engine Performance in Flight Mach Number Range from 3 to 7. NACA RM E51H02, 1951.

2. Hansen, C. Frederick, and Nothwang, George J.: Condensation of Air in Supersonic Wind Tunnels and its Effect on Flow about Models. NACA TN 2690, 1952. 2813

3. Haefeli, Rudolph C.: Use of Fences to Increase Uniformity of Boundary Layer on Side Walls of Supersonic Wind Tunnels. NACA RM E52E19, 1952.

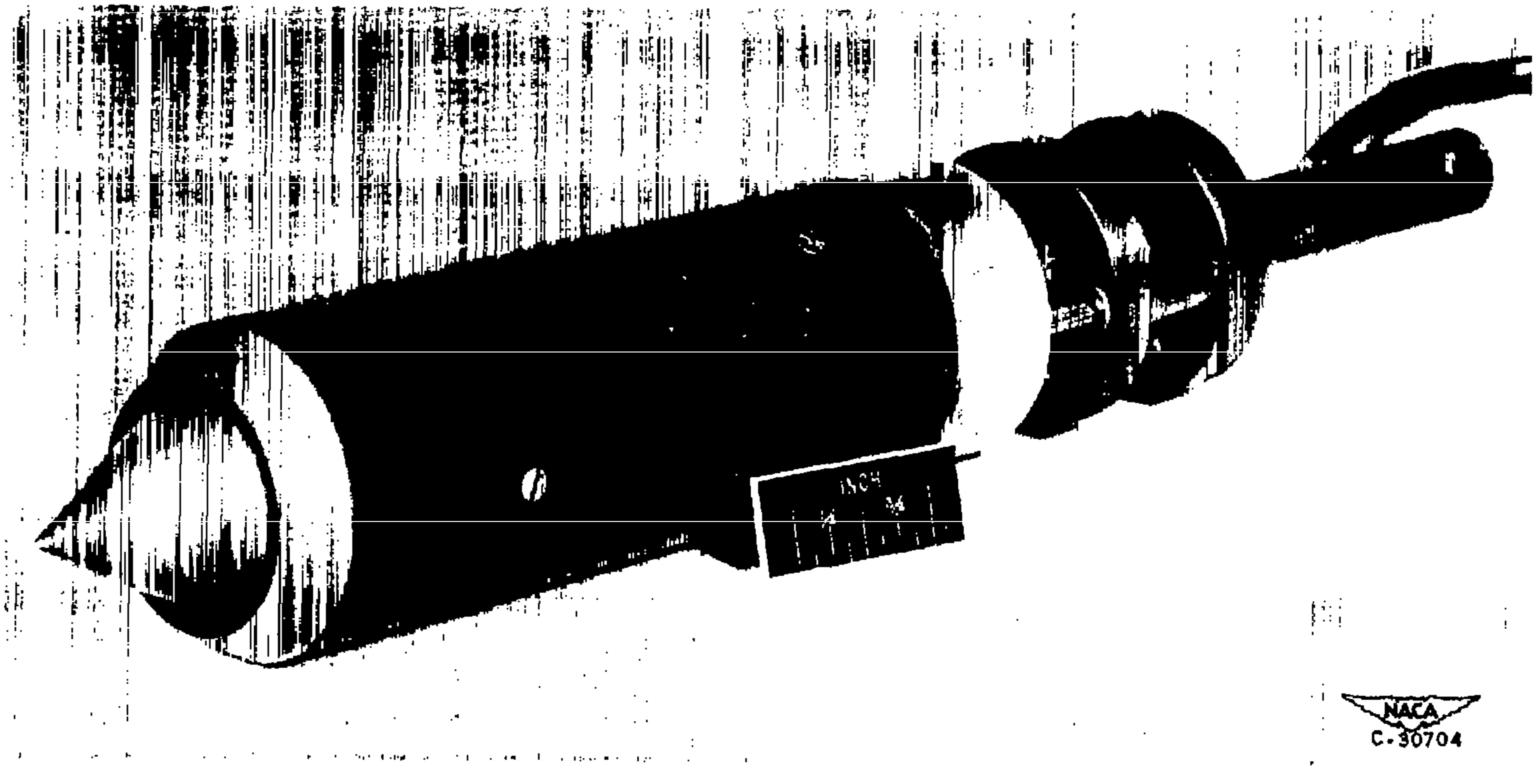
4. Moeckel, W. E., Connors, J. F., and Schroeder, A. H.: Investigation of Shock Diffusers at Mach Number 1.85. I - Projecting Single-Shock Cones. NACA RM E6K27, 1947.

5. Cortright, Edgar M., Jr., and Connors, James F.: Survey of Some Preliminary Investigations of Supersonic Diffusers at High Mach Numbers. NACA RM E52E20, 1952.



C-30778

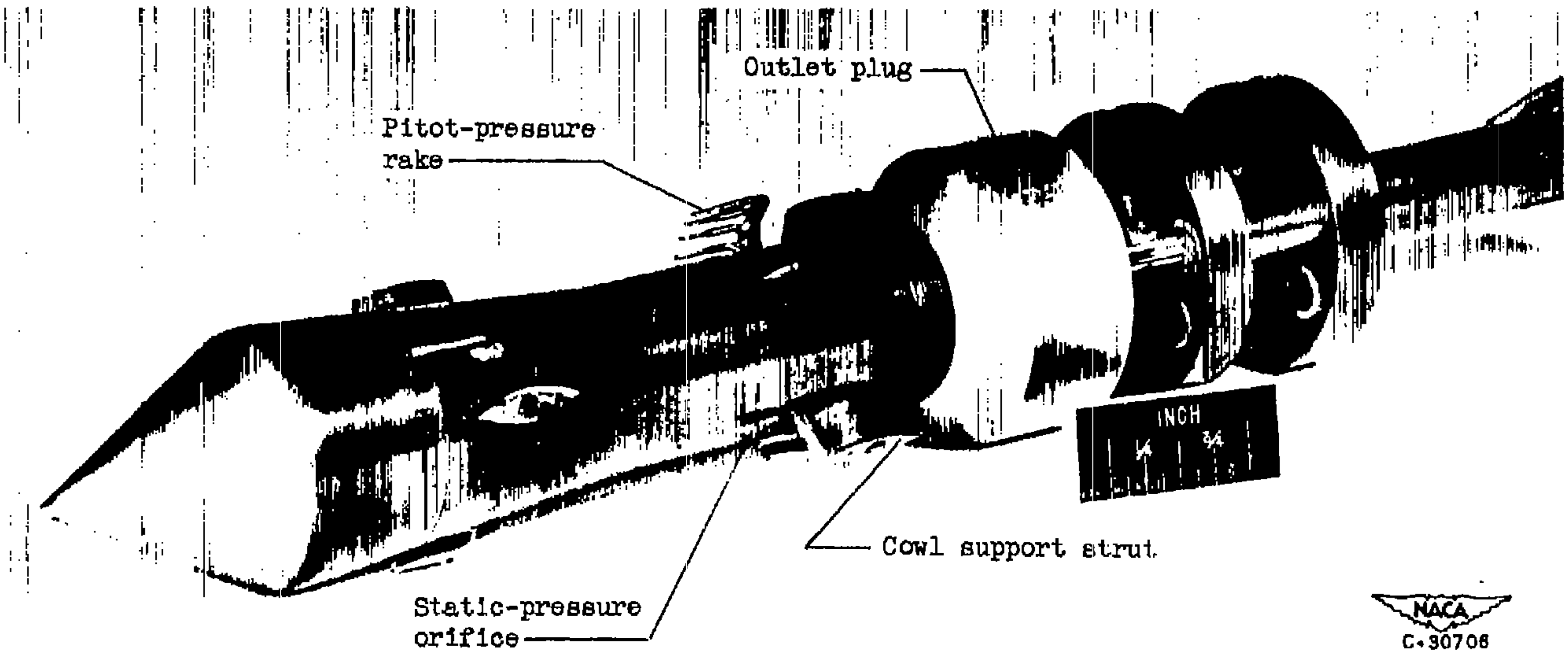
Figure 1. - Single-conical-shock inlet installed in 6- by 6-inch hypersonic tunnel.



(a) Assembled.

Figure 2. - Single-conical-shock inlet.

2813



(b) Cowl removed to show instrumentation.

Figure 2. - Concluded. Single-conical-shock inlet.

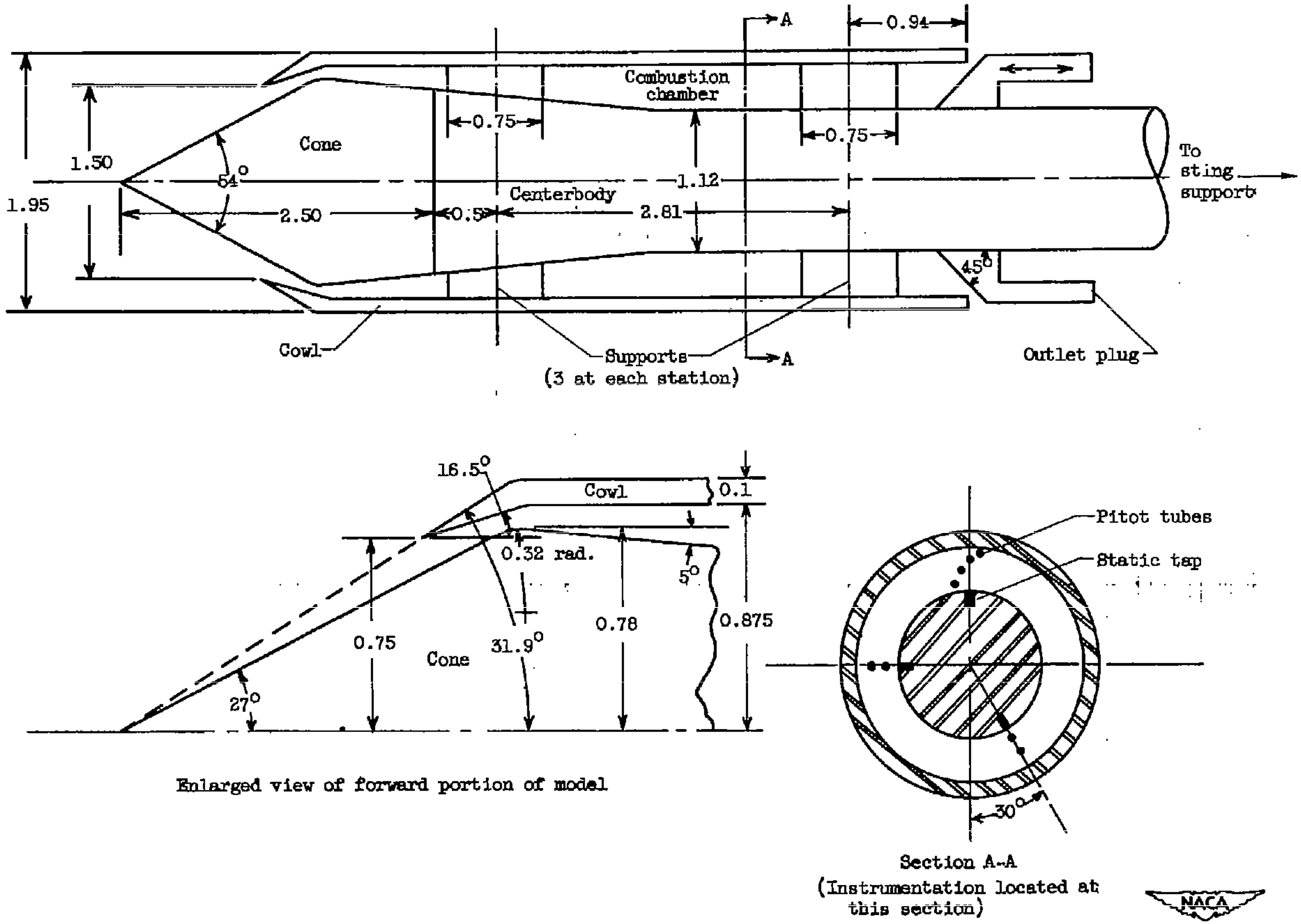
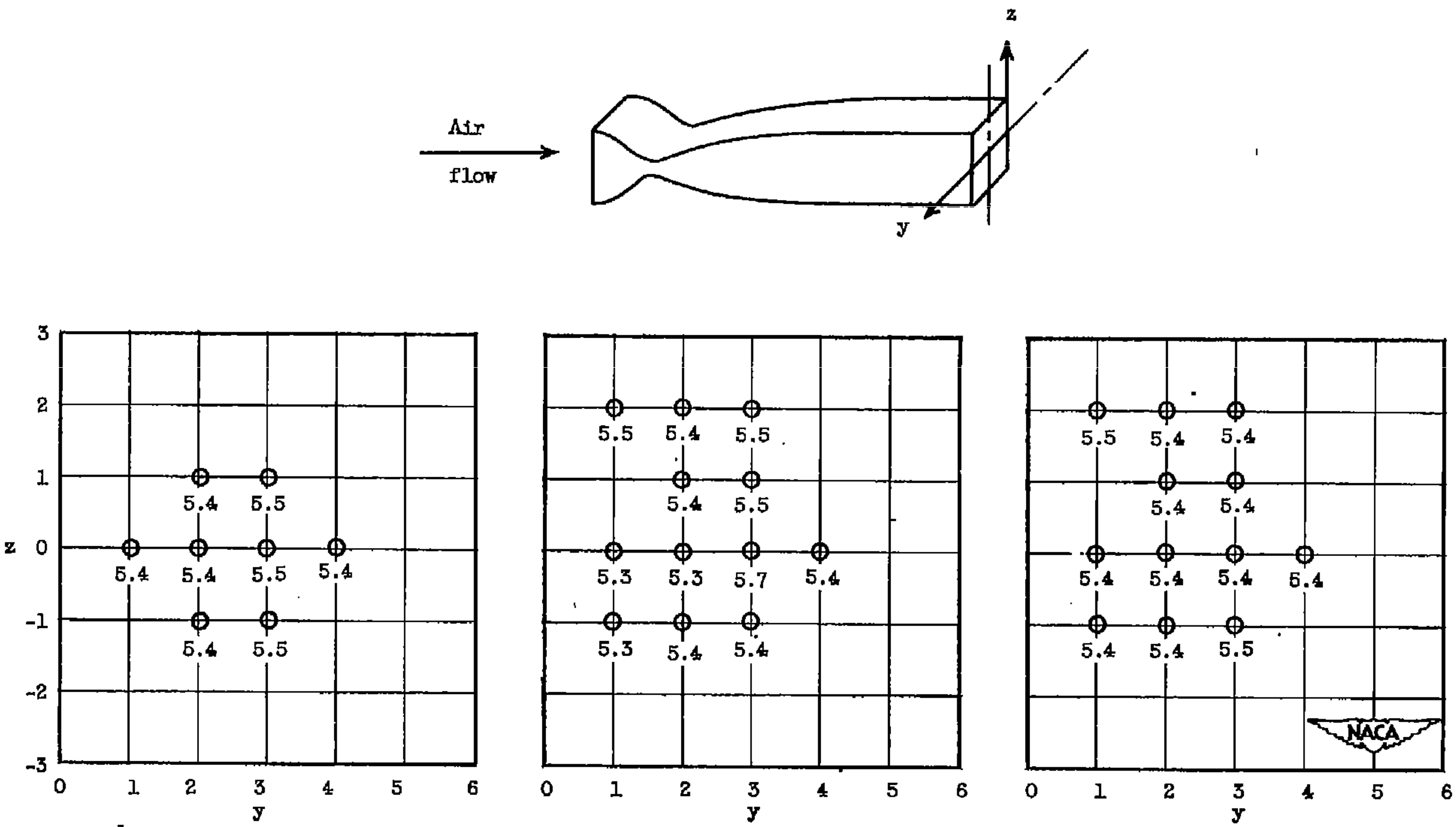


Figure 3. - Single-conical-shock nose inlet showing dimensions and location of instrumentation. (All dimensions are in inches.)

[redacted]



(a) $9\frac{1}{8}$ inches downstream of throat.

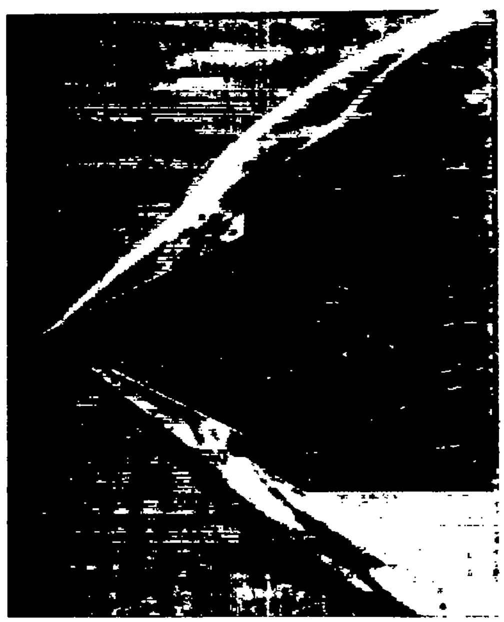
(b) 16 inches downstream of throat.

(c) 18 inches downstream of throat.

Figure 4. - Mach number calibration of Lewis 6- by 6-inch hypersonic tunnel.

[redacted]

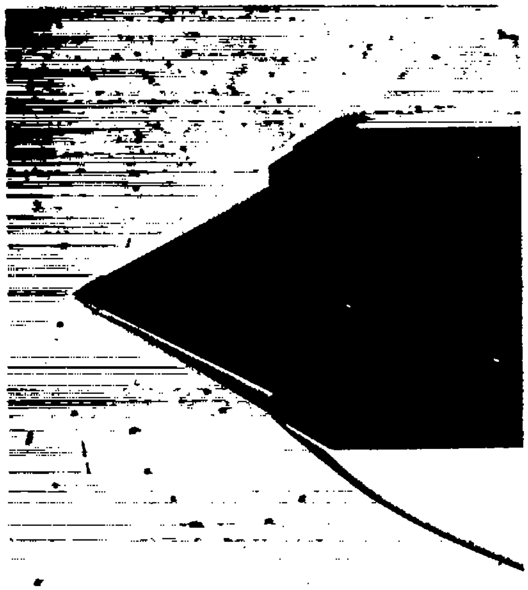
(a) Supercritical; $A_2/A_1 \geq 0.303$.



C-31621

(b) Subcritical (buzz); $A_2/A_1 < 0.303$.

Figure 5. - Schlieren photographs of diffuser. Angle of attack, 0°.



(a) Supercritical; $A_2/A_1 \geqq 0.290$.

C-31622

(b) Subcritical (buzz); $A_2/A_1 < 0.290$.

Figure 6. - Schlieren photographs of diffuser with artificial boundary-layer transition on cone. Angle of attack, 0°.

2813

(a) Supercritical; $A_2/A_1 \geqq 0.263$.

C-31623

(b) Subcritical (buzz); $A_2/A_1 < 0.263$.

Figure 7. - Schlieren photographs of diffuser with artificial boundary-layer transition on cone with cone retracted. Angle of attack, 0°.

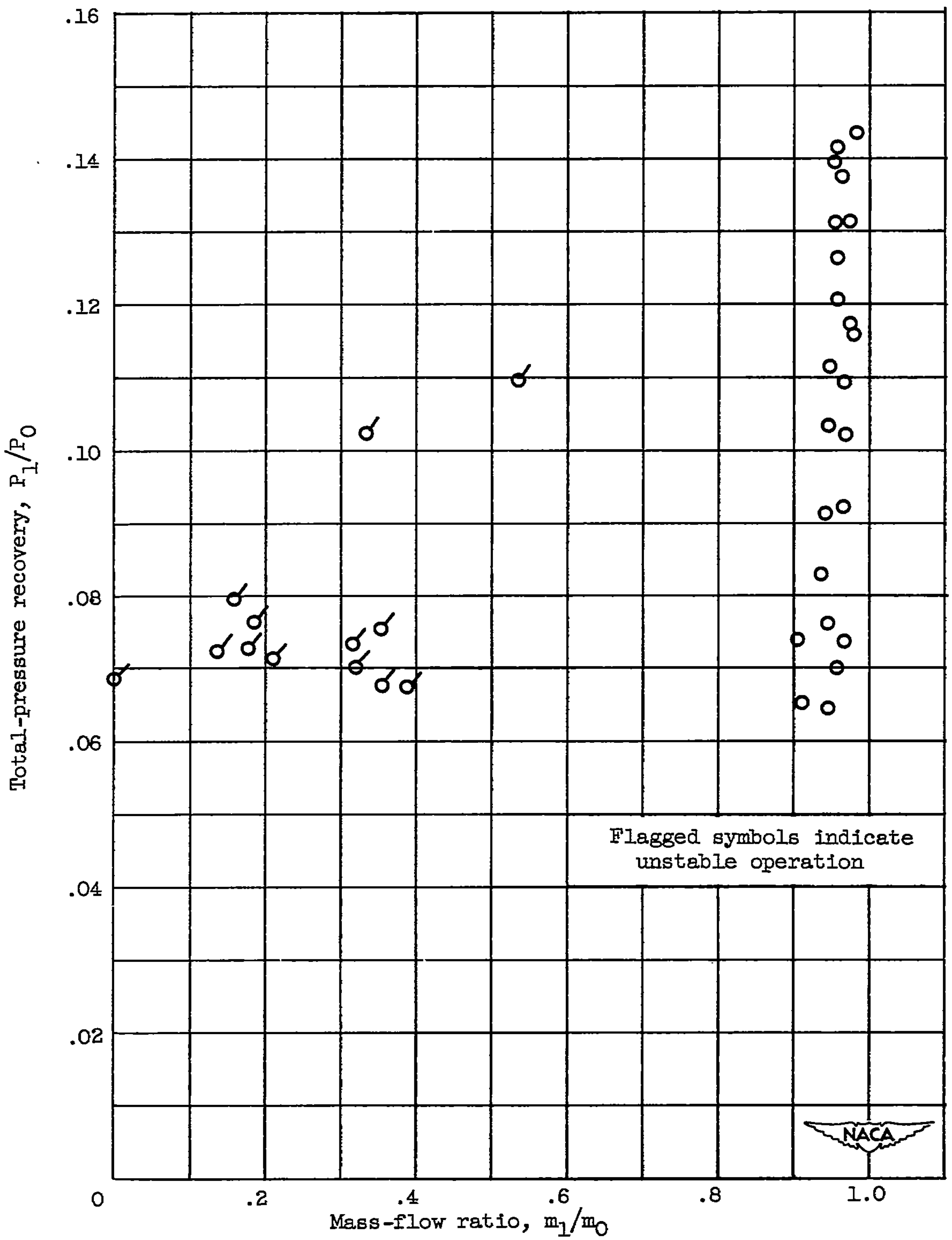


Figure 8. - Diffuser characteristics at zero angle of attack with roughness on cone tip and without cone retraction.

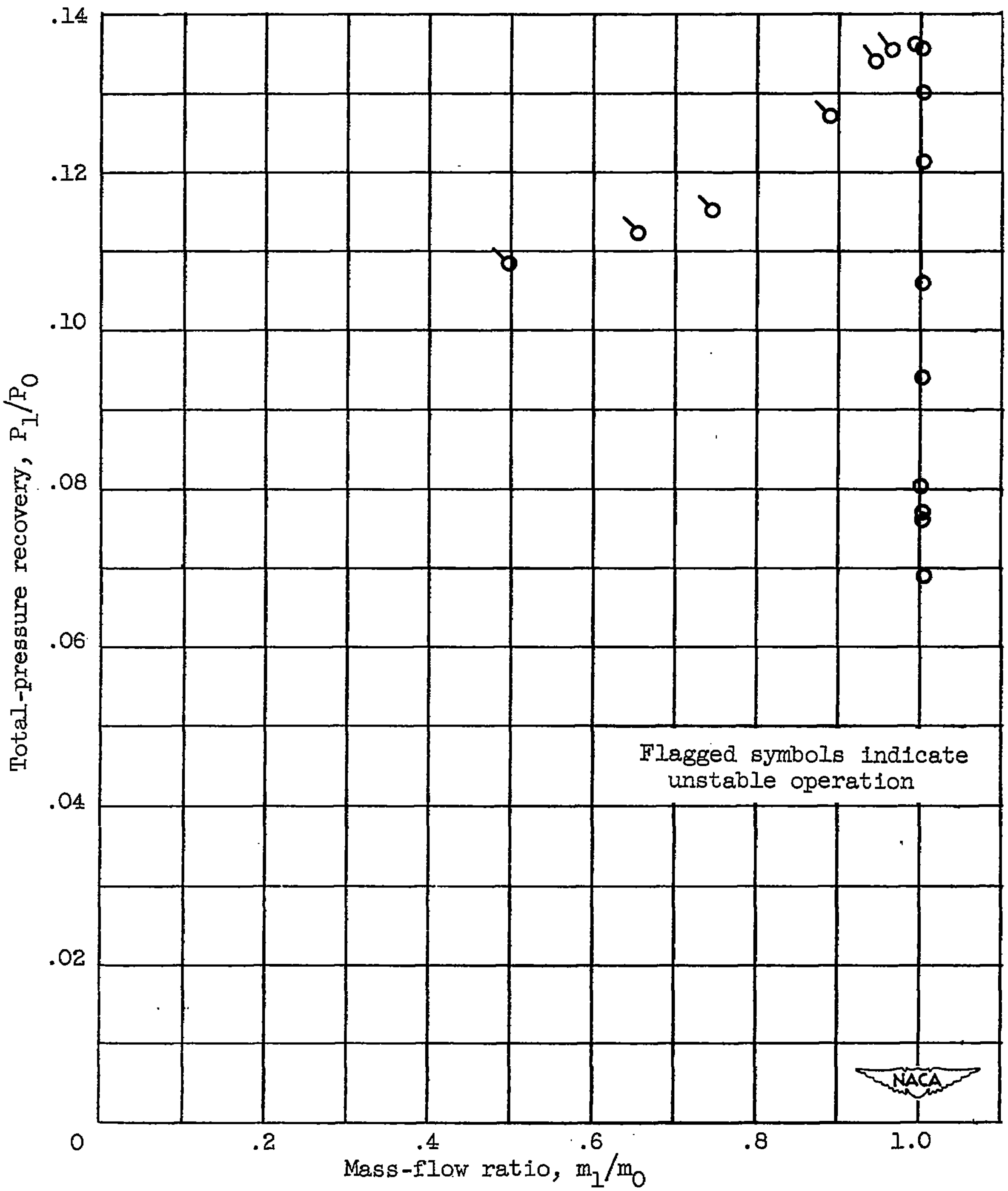
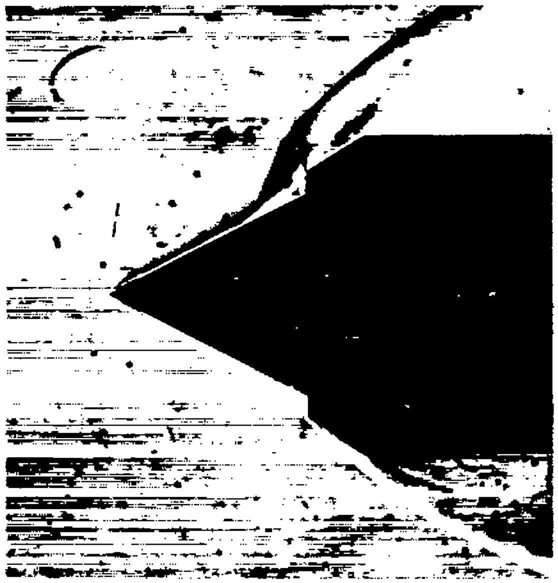
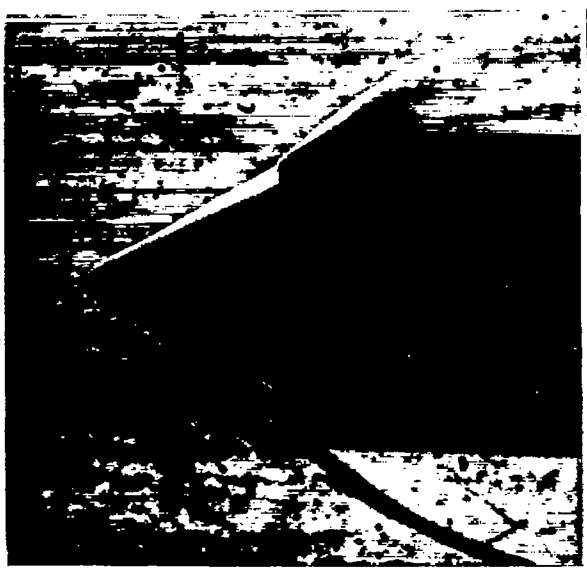
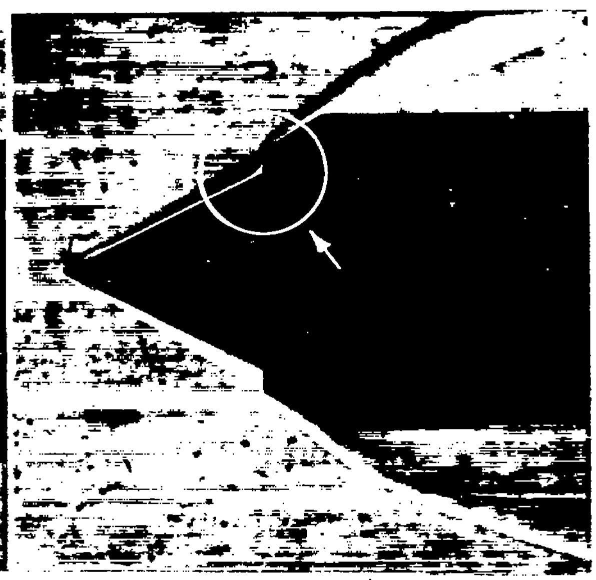


Figure 9. - Diffuser characteristics of zero angle of attack with roughness on cone tip and cone retracted 0.01 inch.



(a) Cone not retracted; $\alpha = 2^{\circ}$; $A_2/A_1 = 0.325$.



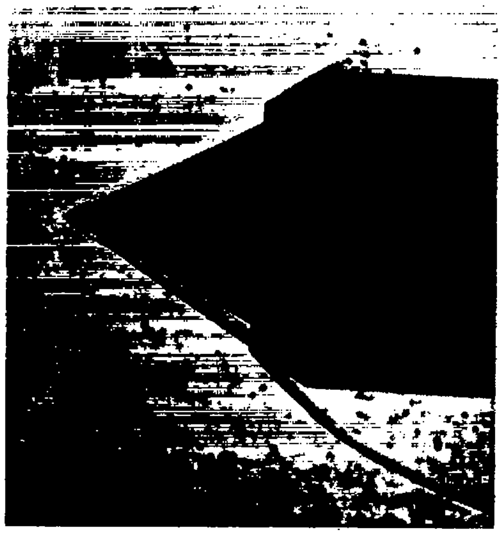


C-31624

(b) Supercritical operation; $\alpha = 3^{\circ}$; $A_2/A_1 \geqq 0.290$.

(c) Critical operation; $\alpha \doteq 3^{\circ}$; $A_2/A_1 = 0.270$.

Figure 10. - Schlieren photographs of diffuser at angle of attack with artificial boundary-layer transition on cone. Cone in retracted position for all photographs except 10(a).



(d) Supercritical operation; $\alpha = 4^{\circ}$; $A_2/A_1 \geqq 0.276$.

NACA
C-31625

(e) Subcritical operation (buzz); $\alpha = 4^{\circ}$; $A_2/A_1 = 0.236$.

Figure 10. - Concluded. Schlieren photographs of diffuser at angle of attack with artificial boundary-layer transition on cone. Cone in retracted position for all photographs except 10(a).

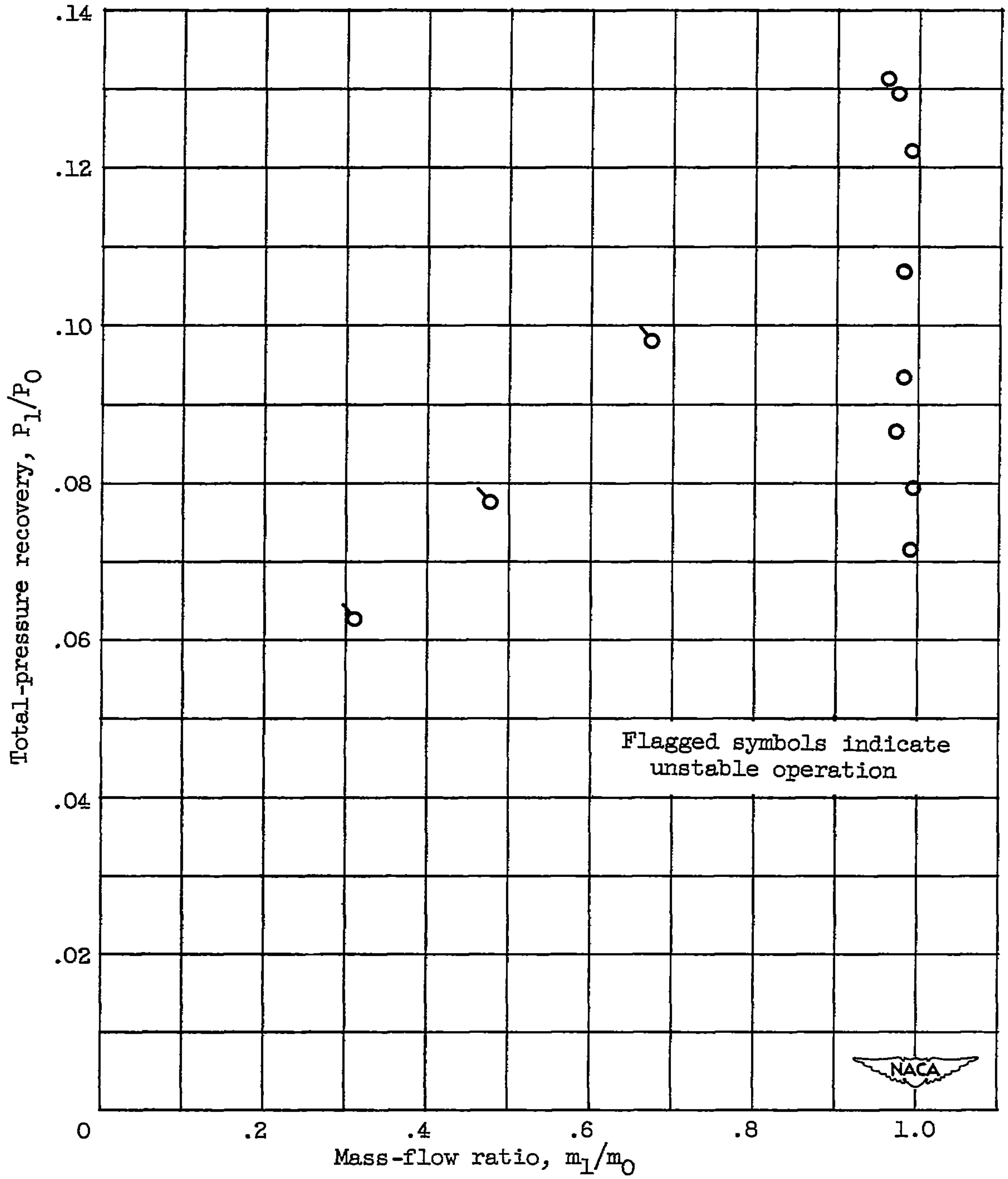


Figure 11. - Diffuser characteristics at angle of attack of 3° with roughness on cone tip and cone retracted 0.01 inch.

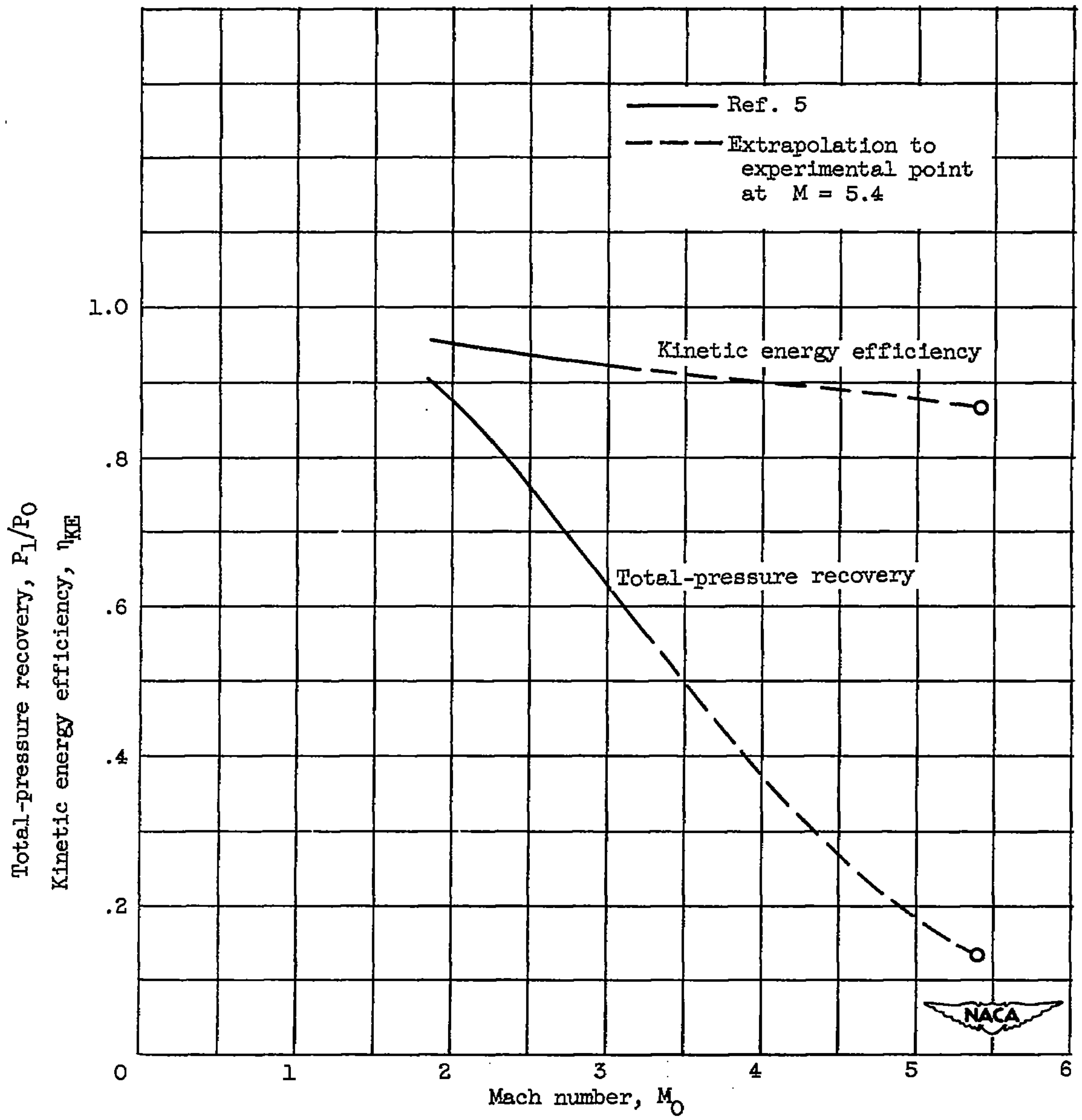


Figure 12. - Performance of single-conical-shock nose inlets with internal contraction.

NACA-Langley - 4-6-53 - 325