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TECHNICAL MEMORANDUM

X-145

PERFORMANCE OF A MACH 3.0 EXTERNAL-INTERNAL-COMPRESSION AXISYMMETRIC INLET AT MACH NUMBERS FROM 2.0 TO 3.5

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TECHNICAL MEMORANDUM X-145

PERFORMANCE OF A MACH 3.0 EXTERNAL-INTERNAL-COMPRESSION AXISYMMETRIC INLET AT MACH NUMBERS FROM 2.0 TO 3.5*

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SUMMARY

A Mach 3.0 axisymmetric inlet having both external and internal compression was evaluated in the Lewis 10- by 10-foot supersonic wind tunnel at Mach numbers from 2.0 to 3.5.

At Mach 3.0 a peak recovery of 83 percent was obtained with 4.5 percent bleed on the centerbody ahead of the throat and a cowl pressure drag coefficient of 0.012, based on maximum cross-sectional area. At Mach 2.0 the recovery increased to 90.5 percent and the cowl pressure drag coefficient decreased to 0.009. An additional 7.5 percent bleed in the subsonic diffuser increased the recovery from 83 to 87 percent at the design speed.

INTRODUCTION

For Mach 3.0 applications, one type of induction system that appears attractive from a thrust-minus-drag standpoint is an axisymmetric external-internal-compression inlet with a low drag cowl. A total-pressure recovery of 78 percent has been obtained with an inlet of this type at Mach 3.0 with a cowl pressure drag coefficient of 0.01, as reported in reference 1. The external compression of this inlet was achieved by a 20° half-angle cone, and the internal compression was generated by two oblique shocks off the cowl.

Theoretically the overall total-pressure recovery of this inlet could be improved by about 5 percent by replacing the 20° half-angle cone with an isentropic spike having the same amount of turning. In order to determine whether these gains could be achieved, an experimental program has been conducted on an inlet identical to that of reference 1 except for the improved supersonic compression components. In addition, bleed

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on both centerbody and cowl was provided in the subsonic diffuser. This investigation was made in the Lewis 10- by 10-foot supersonic wind tunnel at Mach numbers from 2.0 to 3.5 and over a range of angle of attack, spike position, and inlet mass-flow ratio. The effect of Reynolds number on inlet performance was also obtained at the design Mach number of 3.0.

SYMBOLS

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A	area
A_{max}	maximum frontal area, 283.5 sq in.
A_i	projected inlet area, 257.5 sq in.
A_4	diffuser-exit area, 138 sq in.
D	diameter
L	length
M	Mach number
m	mass flow
P	total pressure
p	static pressure
x	linear distance
δ	spike translation from design position, positive when spike is extended

Subscripts:

av	average
d	subsonic diffuser
i	inlet lip station
max	maximum
min	minimum
x	conditions at x-distance

[redacted]

0 free-stream station

1 throat station, $x/L_d = 0.157$

2 survey station, $x/L_d = 0.32$

3 survey station, $x/L_d = 0.495$

4 diffuser-exit station, $x/L_d = 1.00$

APPARATUS AND PROCEDURE

At a Mach number of 3.0 the inlet (fig. 1) was designed for 99-percent total-pressure recovery through the initial shock followed by isentropic focused compression to an average Mach number of 2.37 at the cowl lip station (fig. 2). The local flow conditions at the cowl lip station and the spike coordinates were obtained directly from the results presented in table I of reference 2. The inner surface of the cowl turned the flow toward the model axis in two steps of approximately 12° each and focused the resulting oblique shocks on the sharp shoulder of the centerbody. Two-dimensional flow was assumed to trace the internal shocks back to the focal point. A flush slot was located on the spike surface just ahead of the focal point to bleed off the boundary-layer air prior to the high pressure rise associated with the internal compression. The minimum area of this bleed slot was about 5 percent of the inlet capture area. The centerbody and cowl surfaces were alined with the local flow direction behind the oblique shocks. The supersonic diffusion reduced the Mach number to an average of 1.45 ahead of the terminal shock. Two hydraulic diameters of essentially constant area were provided in the initial part of the subsonic diffuser, followed by an equivalent 12° conical area expansion back to the simulated engine face, which was located approximately 1.5 inlet diameters from the cowl lip. Flush bleed slots were located in the diffuser on both cowl and centerbody at the end of the constant-area section (fig. 2). The total open area of the bleed slots was about 7 percent of the inlet capture area.

The external cowl lip angle of 7.5° was held constant back to the maximum body diameter, which resulted in a cowl projected area of 9 percent of the maximum cross-sectional area. Theoretical cowl pressure drag coefficient based on A_{max} at the design Mach number of 3.0 was calculated to be 0.011.

The contraction ratio A_1/A_i at Mach 3.0 was 0.28 with an average throat recovery of 88 percent, based on two-dimensional internal-flow considerations. The spike could be translated forward to reduce the contraction ratio. The diffuser area distributions for Mach numbers of 3.0, 2.5, and 2.0 are presented in figure 3 at spike position parameters δ/D_i of 0, 0.064, and 0.138, respectively.

The model was instrumented with static-pressure orifices on the centerbody and on the cowl inner and outer surfaces. The location of the four sets of total-pressure rakes along the diffuser is shown in figure 3. Inlet mass flow was computed from a measured static pressure just ahead of a remotely actuated control plug with the assumption of one-dimensional isentropic flow to the calibrated sonic discharge area.

This investigation was conducted in the Lewis 10- by 10-foot supersonic wind tunnel at Mach numbers from 2.0 to 3.5 and angles of attack from 0° to 10°. At the design Mach number of 3.0 the effect of Reynolds number was obtained by varying the tunnel Reynolds number from 0.50×10^{6} to 2.5×10^{6} per foot.

RESULTS AND DISCUSSION

The overall performance of the inlet at zero angle of attack and various free-stream Mach numbers is presented in figure 4. At the design Mach number of 3.0, the basic configuration (bleed only ahead of the centerbody shoulder) had a peak total-pressure recovery of 83 percent at a mass-flow ratio of 0.955. When the inlet unstarted, the performance fell off along a line of constant corrected weight flow (dashed line) to a recovery level comparable with that of the 20° isentropic turning ahead of the lip. In addition to the poor performance with the shock expelled, the inlet pulsed in this region. The spike required a forward translation of 7 percent of the inlet diameter to restart the flow. The internal area variation at this spike position would be about the same as that shown in figure 3 for $\delta/D_i = 0.064$. The performance of the inlet without any bleed is presented at Mach 3.0 to show the importance of the small amount of bleed for inlets of this type. With no bleed, the spike could not be retracted to the design point, and both mass-flow ratio and peak recovery were significantly lower than with bleed. At Mach numbers less than 3.0, the total-pressure recovery of the inlet with shoulder bleed increased steadily with decreasing Mach number, reaching a value of 90.5 percent at Mach 2.0. At a Mach number of 2.5 the inlet again pulsed when the shock was expelled. At Mach 2.0, however, the inlet could be operated stably as an external-compression inlet over a small range of mass-flow ratio.

The total-pressure recovery could be increased with additional bleed behind the constant-area section; however, at the design Mach number an additional 7.5 percent bleed was required to increase the recovery from 83 to 87 percent. At the lower Mach numbers, similar recovery increases could be attained with bleed, as shown by the data of figure 4; for example, at Mach 2.0, the total-pressure recovery was increased from 90.5 to 93.5 percent with an additional bleed of 3 percent.

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The measured cowl pressure drag coefficient had a value of 0.012 at Mach 3.0, referenced to maximum body area, and decreased steadily to a value of 0.009 at Mach 2.0 (fig. 4). The additive drag was not measured on this model; however, based on reference 3, an additive drag coefficient of 0.12 is estimated for spilling 50 percent of the flow at Mach 2.0.

The development of the total-pressure profiles through the subsonic diffuser from the throat to the engine face is presented in figure 5. A comparison of the measured throat recovery (station 1) with the theoretical value indicates some overcompression in the center of the passage. An inspection of static- and total-pressure measurements in the region between the cowl lip and the inlet throat indicated that the flow around the second step was essentially isentropic. Evidently the bounday layer on the inner cowl surface bridged across the step and set up an isentropic shock fan around the corner. The relatively thick boundary layer on the spike and cowl surfaces at station 2 suggests the possibility of further increasing the overall recovery by bleeding in this area. A comparison of the total-pressure profiles between stations 2 and 4 indicates that the effectiveness of bleed just ahead of station 2 showed up increasingly as the flow diffused.

The effects of overspeed on inlet performance were investigated briefly at a Mach number of 3.45 with the aft-bleed configuration. A peak total-pressure recovery of 59 percent was obtained with the spike retracted slightly from the Mach 3.0 position. Schlieren photographs (fig. 6(a)) indicated that the oblique shocks from the external spike could be placed inside the cowl lip without unstarting the inlet. The external shock structure at and below design Mach number is presented in figures 6(b) to (d). At Mach 3.0 the initial shock and the isentropic fan focused very nearly at the cowl lip, which indicated that most of the reduction in mass flow at this speed was due to the internal bleed flow. At Mach 2.0 the spike has been extended to its maximum throat size. At Mach numbers below 2.0 the internal flow would detach and the inlet would operate as an external-compression inlet with a 20° isentropic spike, the mass flow being regulated by the choked throat.

The effects of angle of attack on total-pressure recovery and mass-flow ratio are presented in figure 7 at Mach numbers of 3.0, 2.5, and 2.0 for the shoulder-bleed inlet. The points shown represent the maximum pressure recovery obtained at each Mach number and angle of attack by translating the spike. In all cases, recovery and mass flow decrease with angle of attack, the rate of decrease being about the same as that of the inlet of reference 1 at Mach 3.0.

The effects of spike translation on inlet recovery and mass-flow ratio at zero angle of attack are presented in figure 8 for a range of free-stream Mach numbers. Both pressure recovery and mass-flow ratio decreased almost linearly with forward translation of the spike at the design speed, and the rate of decrease became smaller as the Mach number was reduced. An inlet of this type, with mixed compression, has a very short translation over the Mach number range; for example, operation from the design Mach number to Mach 2.0 and below (maximum throat area) requires a total movement of the spike of about 13 percent of the inlet diameter. The spike also had to be extended for angle-of-attack operation (fig. 9), and again the amount of movement decreased with decreasing Mach number.

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The basic configuration (with shoulder bleed only) had a distortion of about 22 percent at Mach 3.0 with the inlet operating at critical conditions (fig. 10). This distortion did not vary greatly with Mach number, but increased markedly with angle of attack (data not presented), becoming 60 percent at Mach 3.0 and a 10° angle of attack. Additional bleed in the subsonic diffuser reduced the distortion at all Mach numbers; for example, at Mach 3.0 the distortion was reduced from 21 to 12 percent. The high distortion level of this inlet is due to its short length, high diffusion rate (equivalent to 12° conical expansion), and high discharge Mach number ($M_4 \approx 0.40$), all of which have been shown to affect distortion adversely (refs. 4 and 5).

The effects of Reynolds number, based on inlet diameter, on total-pressure recovery and mass-flow ratio at Mach 3.0 and zero angle of attack are shown in figure 11. At Reynolds numbers less than 2.0×10^{6} the spike could not be retracted to its design position; consequently, both pressure recovery and mass-flow ratio decreased, as shown in figure 11. This effect resulted from the thickening of the boundary layer on the spike and cowl and the subsequent reduction in effective throat area.

Some representative static-pressure distributions on both the internal cowl and the centerbody, from the cowl lip station to the engine station, are presented in figure 12, primarily to aid in the design of control systems for this type inlet. Figure 12(a) presents the critical and supercritical operation of the shoulder bleed inlet at Mach 3.0, while figure 12(b) presents the effects of free-stream Mach number at the peak recovery points. At Mach 3.0 the normal shock train at peak recovery occurred in the constant-area section and was positioned very close to the inlet throat (station 1).

SUMMARY OF RESULTS

A Mach 3.0 axisymmetric external-internal-compression inlet was investigated at Mach numbers from 2.0 to 3.5. The following results were obtained:

1. A peak total-pressure recovery of 83 percent was obtained at Mach 3.0 with 4.5 percent bleed ahead of the centerbody shoulder. A cowl

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pressure drag coefficient of 0.012, based on maximum body area, was obtained with a 9-percent cowl projected area. At Mach 2.0 a total-pressure recovery of 90.5 percent was obtained with a cowl drag coefficient of 0.009.

2. The peak recovery was increased at Mach 3.0 from 83 to 87 percent with an additional 7.5 percent bleed in the subsonic diffuser. At Mach 2.0 an additional 3 percent bleed in the diffuser increased the recovery from 90.5 to 93.5 percent.

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Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio, September 16, 1959

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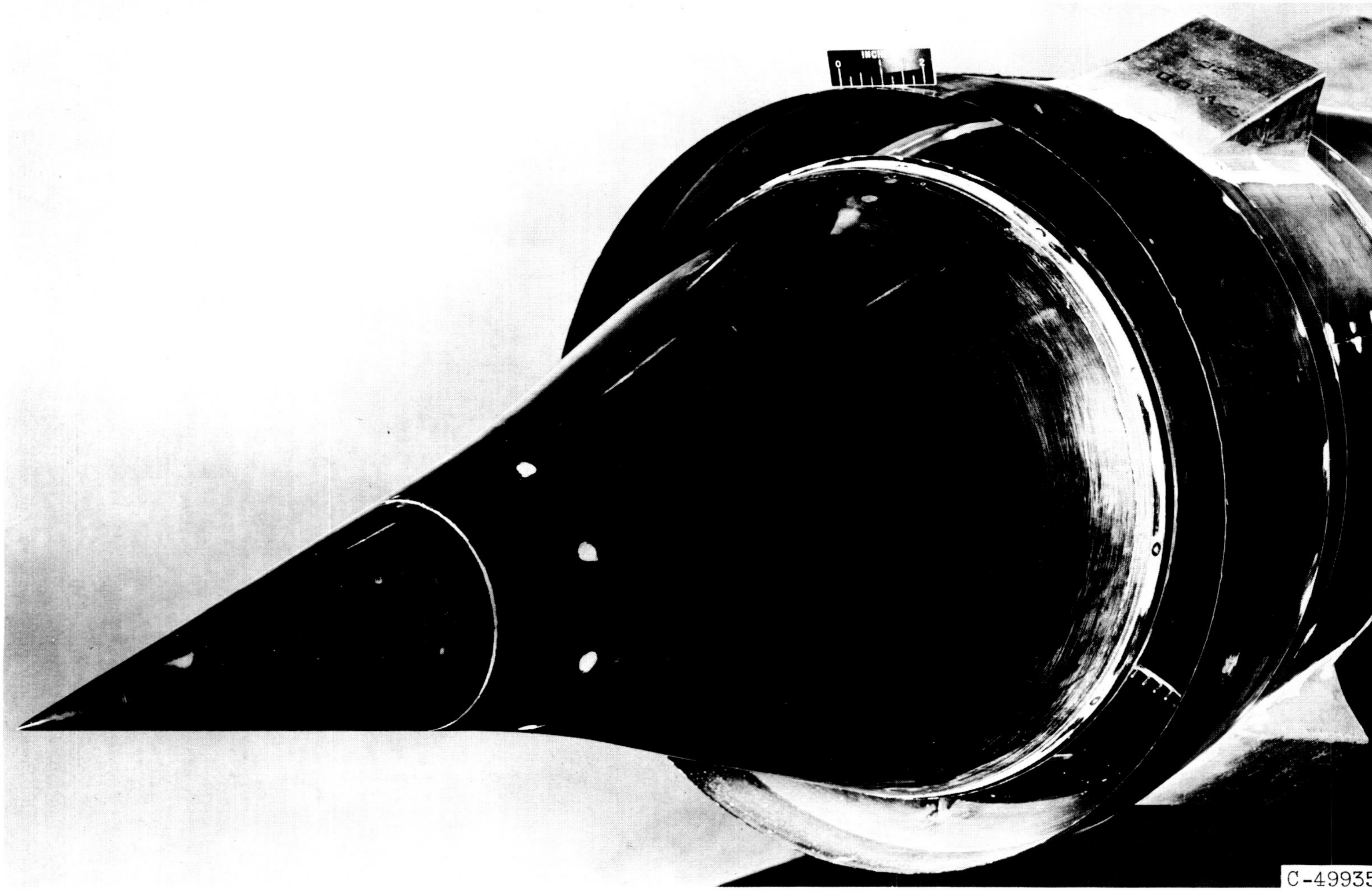

Figure 1. - Photograph of 19-inch isentropic two-step inlet.

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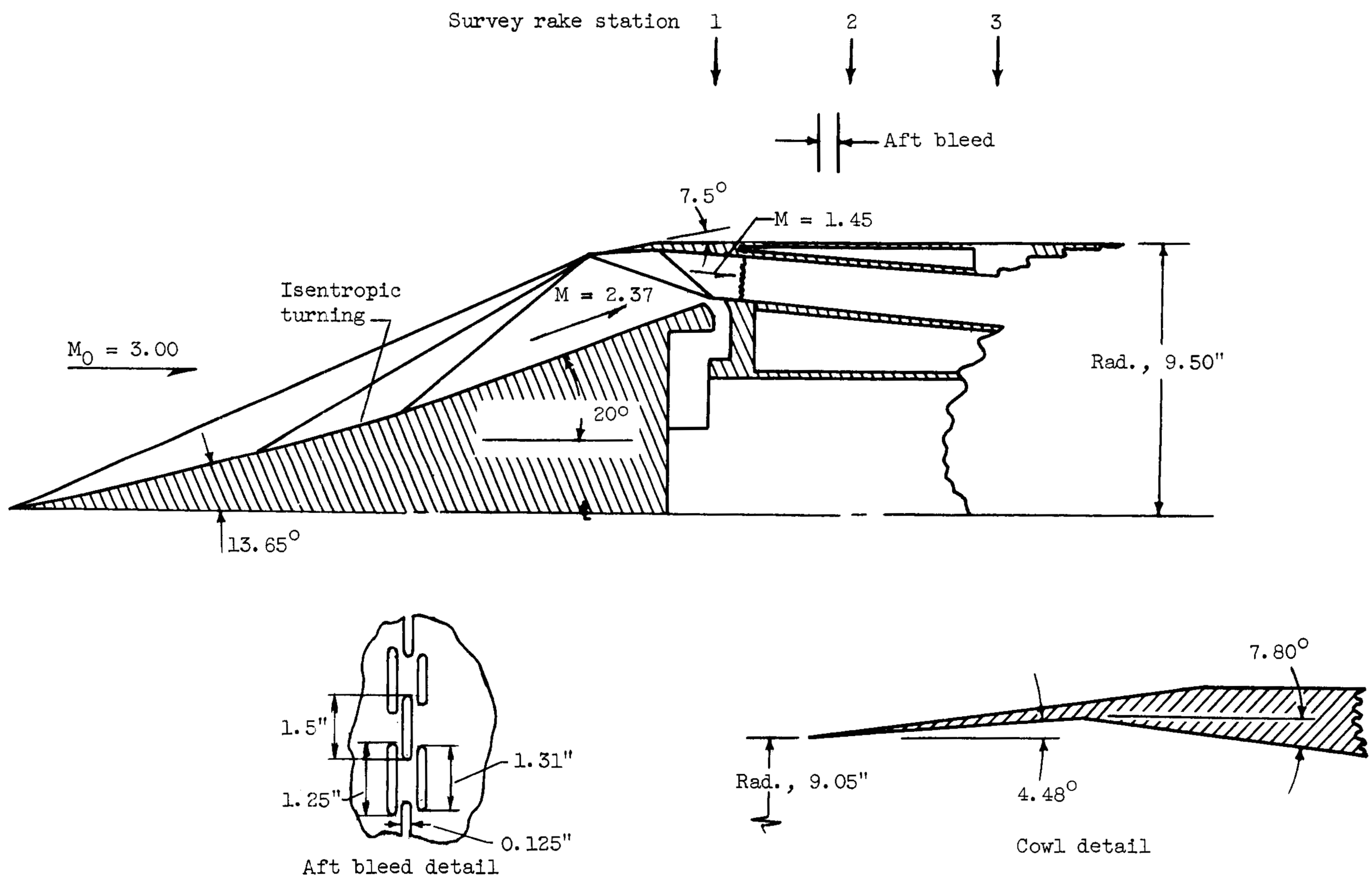

Figure 2. - Design of axisymmetric external-internal-compression Mach 3.0 inlet.

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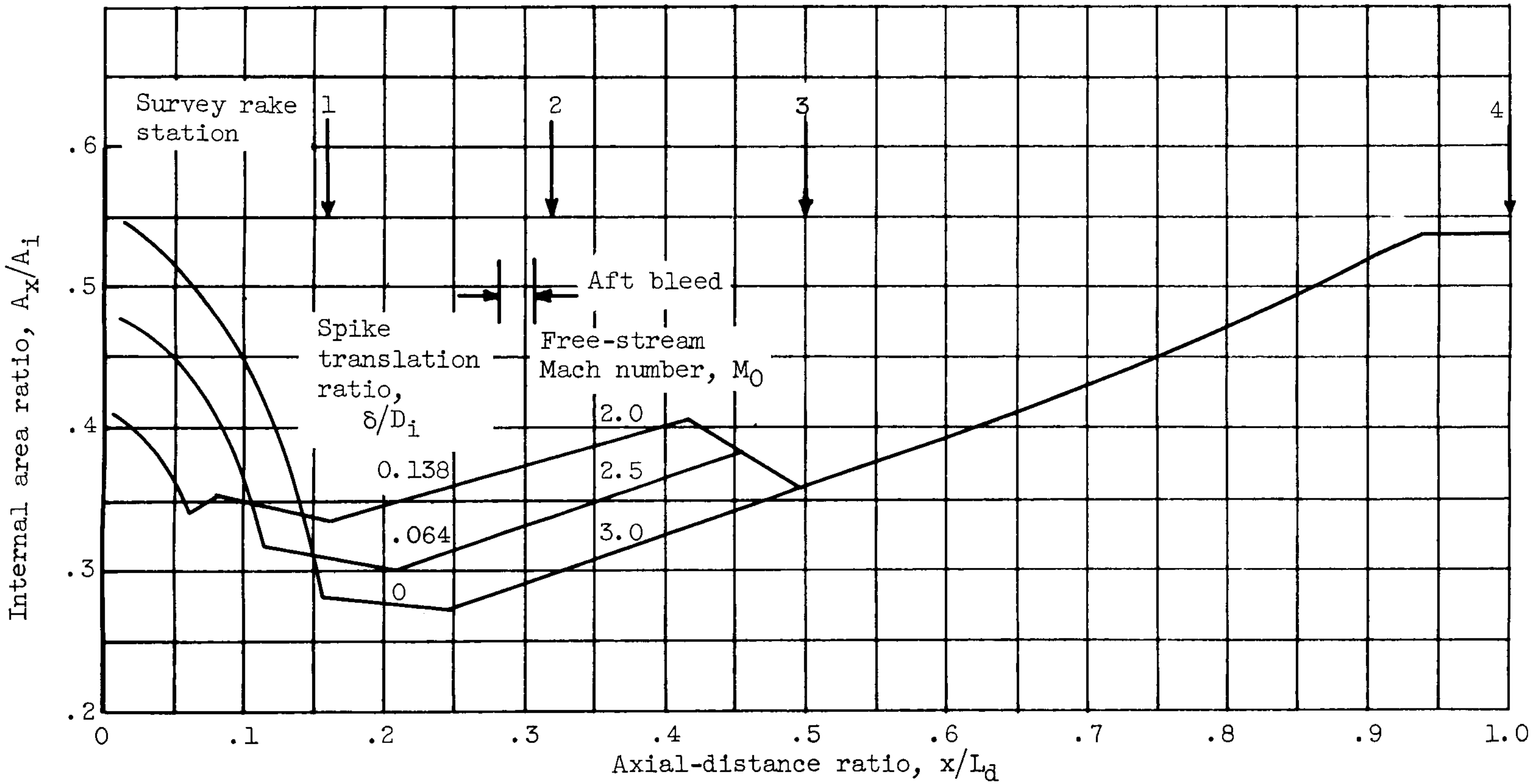

Figure 3. - Variation of diffuser internal area ratio for various spike positions. Subsonic diffuser length, L_d, 29.0 inches; $x/L_d = 0$ corresponds to cowl lip station.

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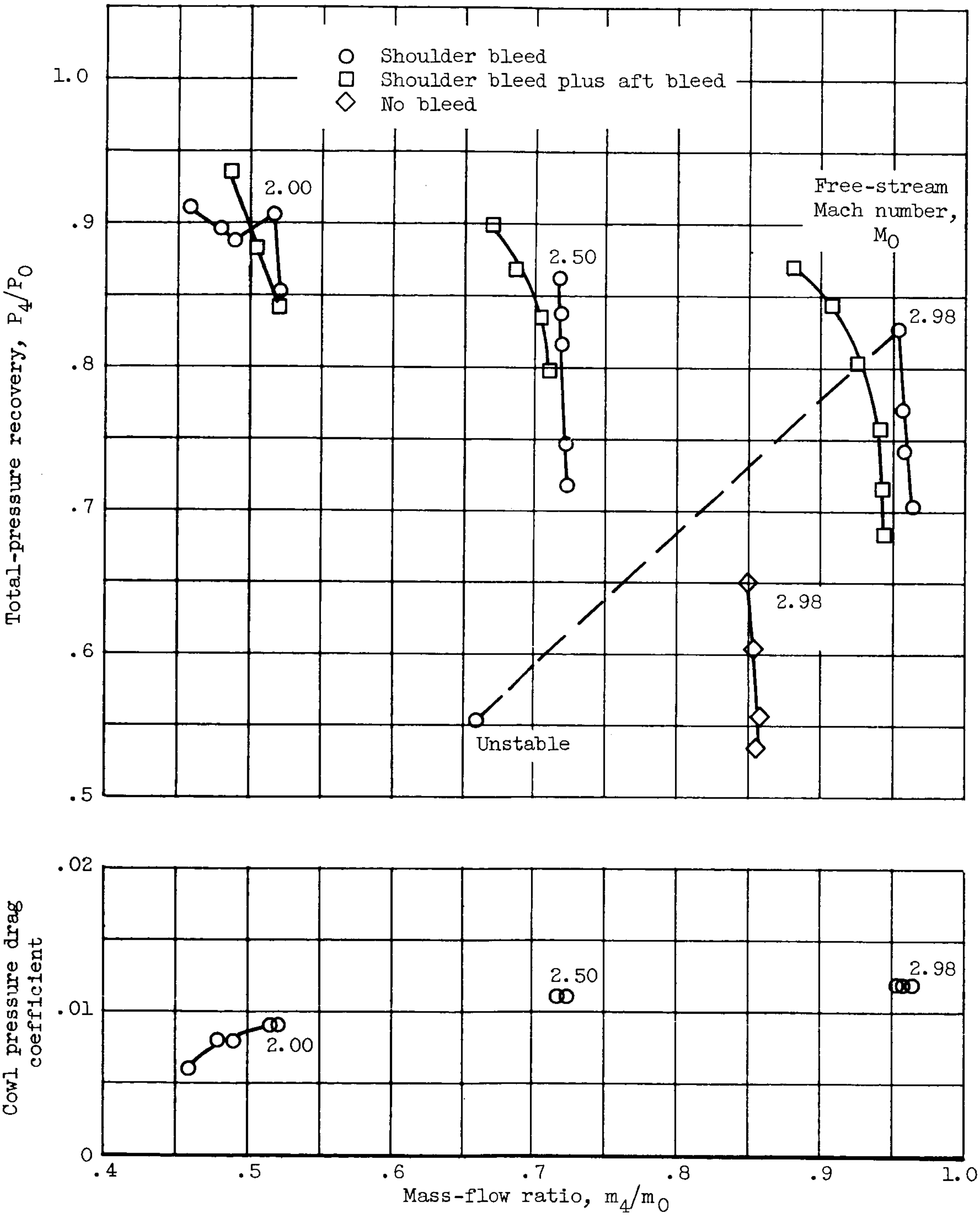

Figure 4. - Inlet performance at various Mach numbers. Angle of attack, 0.

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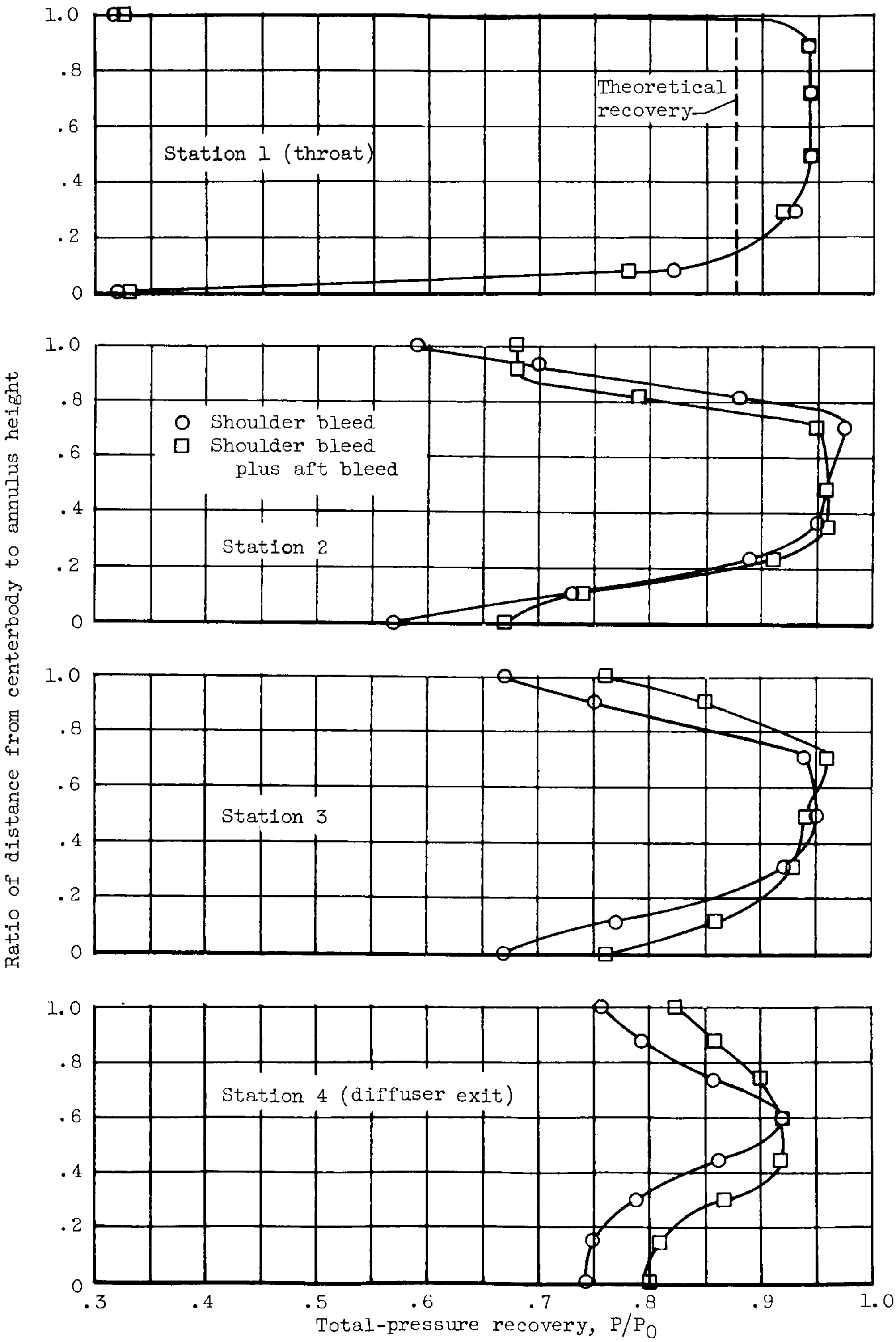

Figure 5. - Total-pressure profiles in subsonic diffuser at Mach number of 2.98. Angle of attack, 0.

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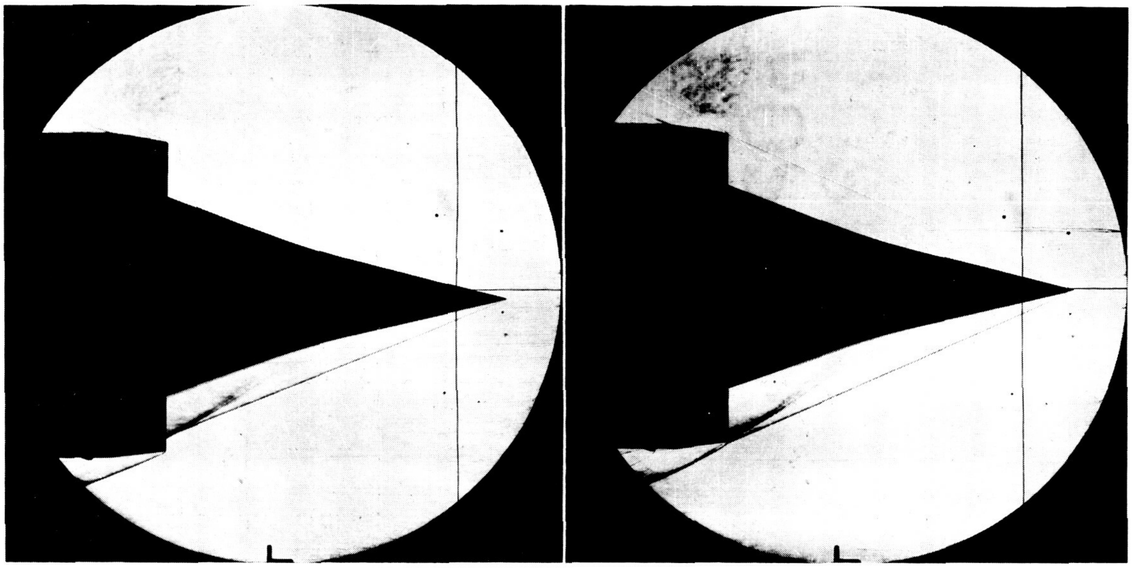
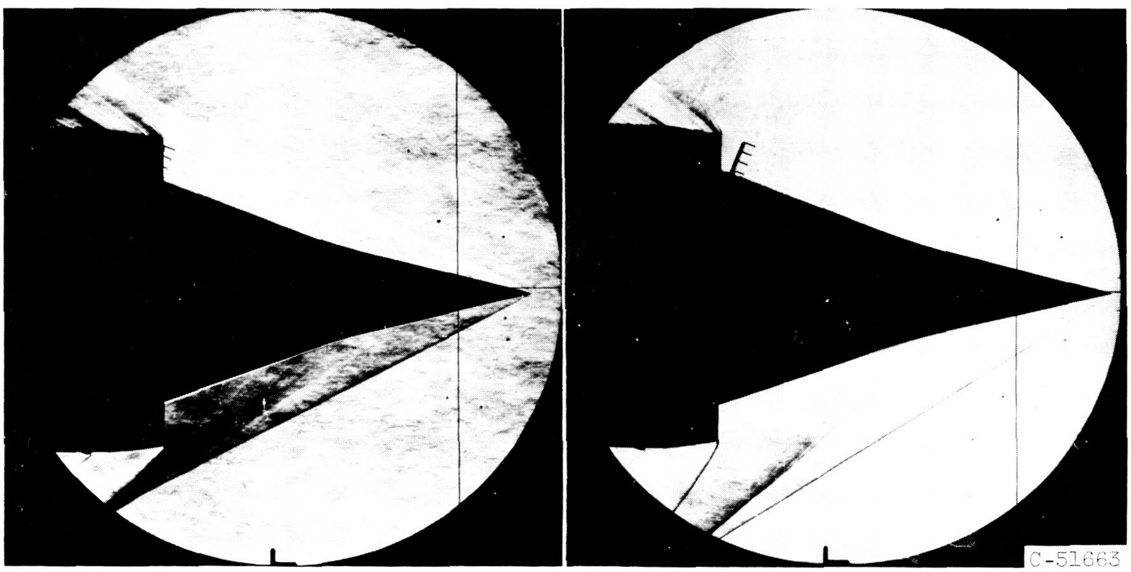


(a) Free-stream Mach number, M_0, 3.45; total-pressure recovery, P_4/P_0, 0.59; mass-flow ratio, m_4/m_0, 0.925; spike translation from design, δ/D_i, -0.033.

(b) Free-stream Mach number, M_0, 2.98; total-pressure recovery, P_4/P_0, 0.827; mass-flow ratio, m_4/m_0, 0.96; spike translation from design, δ/D_i, 0.



(c) Free-stream Mach number, M_0, 2.50; total-pressure recovery, P_4/P_0, 0.863; mass-flow ratio, m_4/m_0, 0.718; spike translation from design, δ/D_i, 0.064.

(d) Free-stream Mach number, M_0, 2.00; total-pressure recovery, P_4/P_0, 0.905; mass-flow ratio, m_4/m_0, 0.517; spike translation from design, δ/D_i, 0.138.

Figure 6. - External shock structure at zero angle of attack over range of Mach number.

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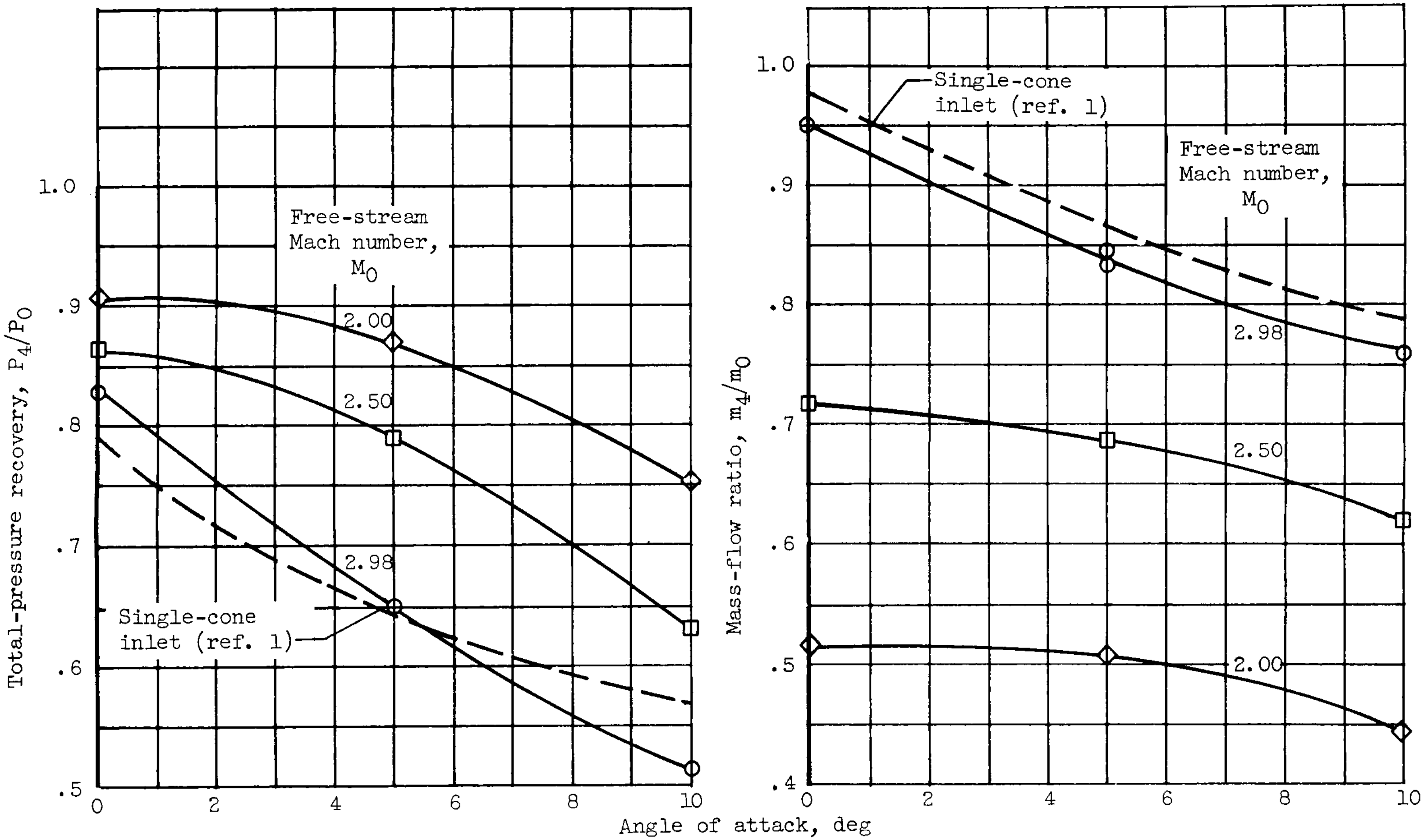

Figure 7. - Effects of angle of attack on inlet performance.

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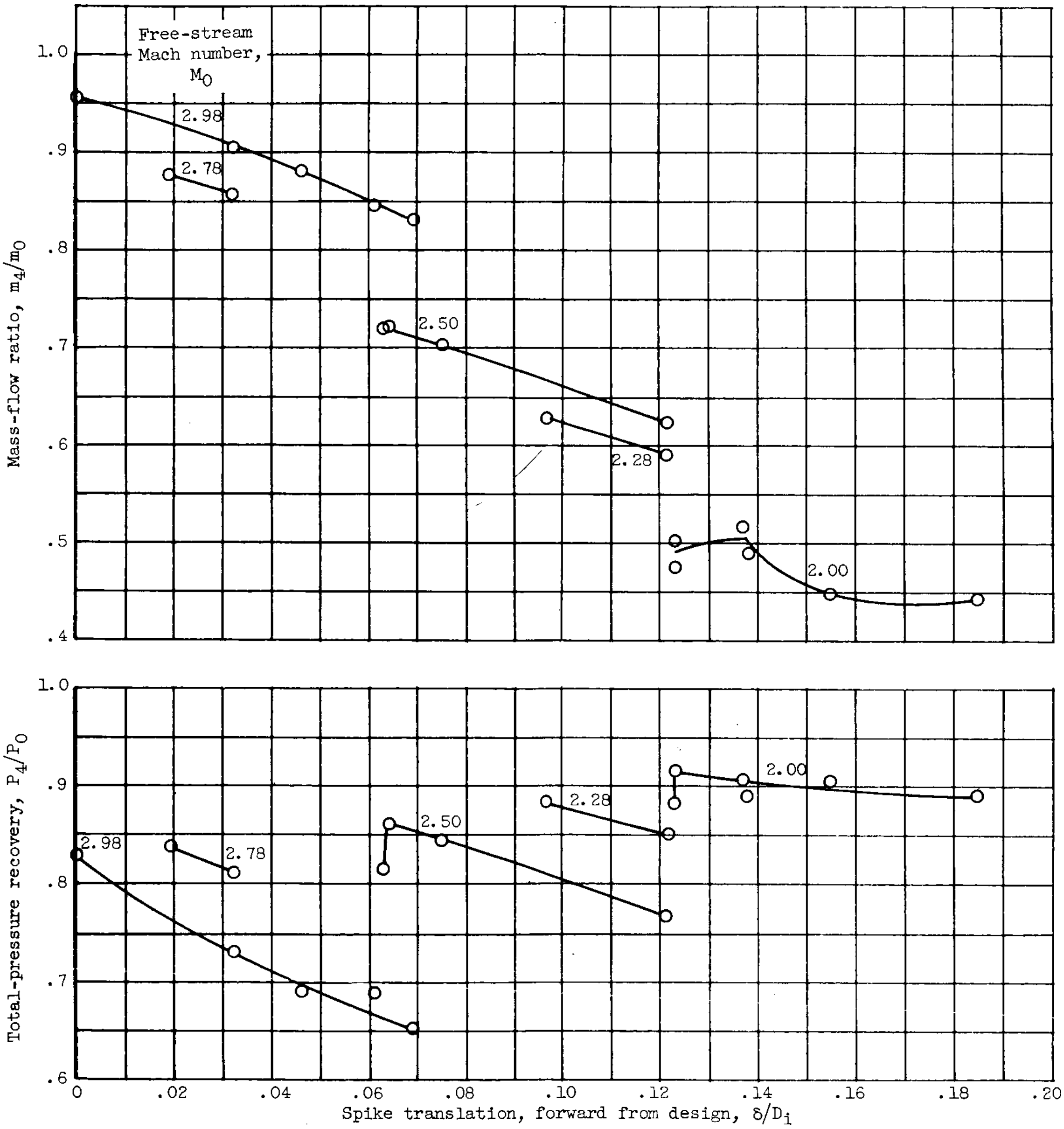

Figure 8. - Effect of spike translation on inlet performance. Angle of attack, 0.

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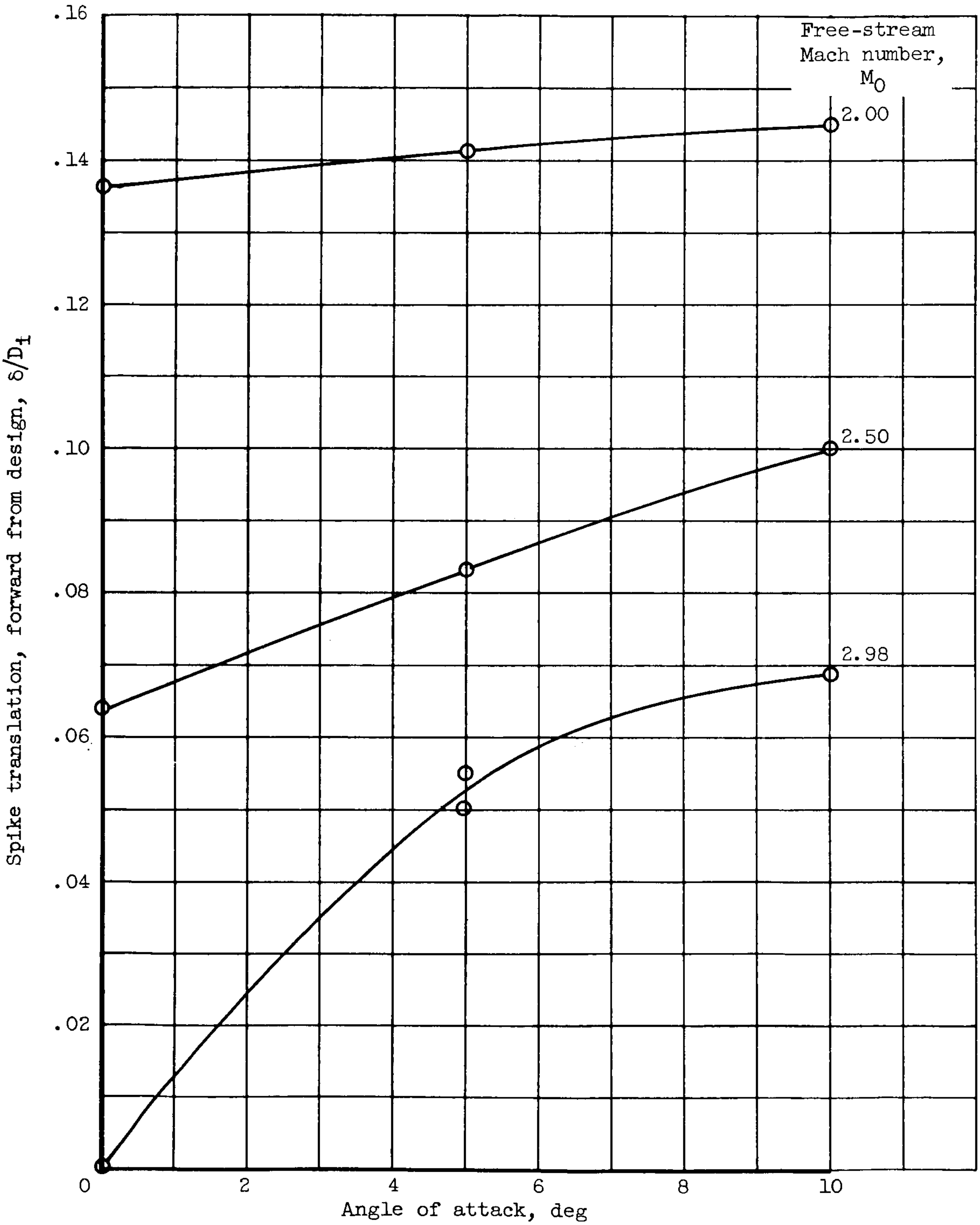

Figure 9. - Spike translation required for angle-of-attack operation.

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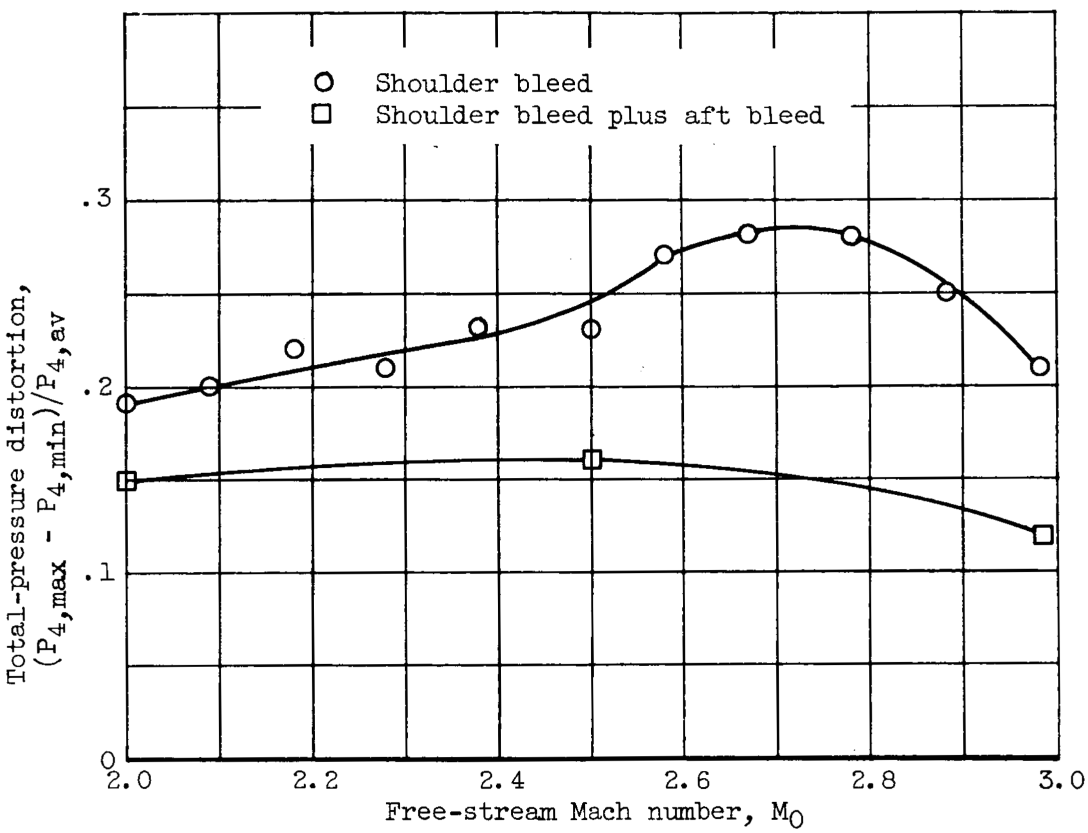

Figure 10. - Effect of diffuser bleed on critical inlet distortion. Angle of attack, 0.

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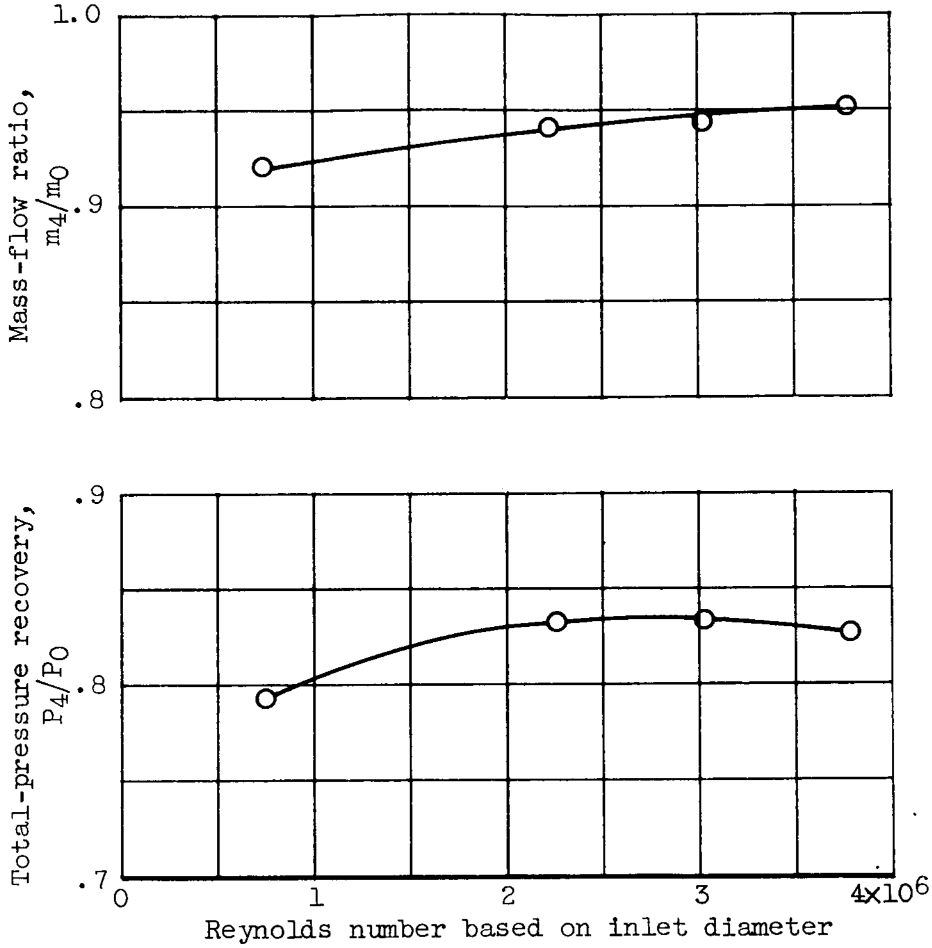

Figure 11. - Effect of Reynolds number on critical performance at Mach 2.98. Angle of attack, 0.

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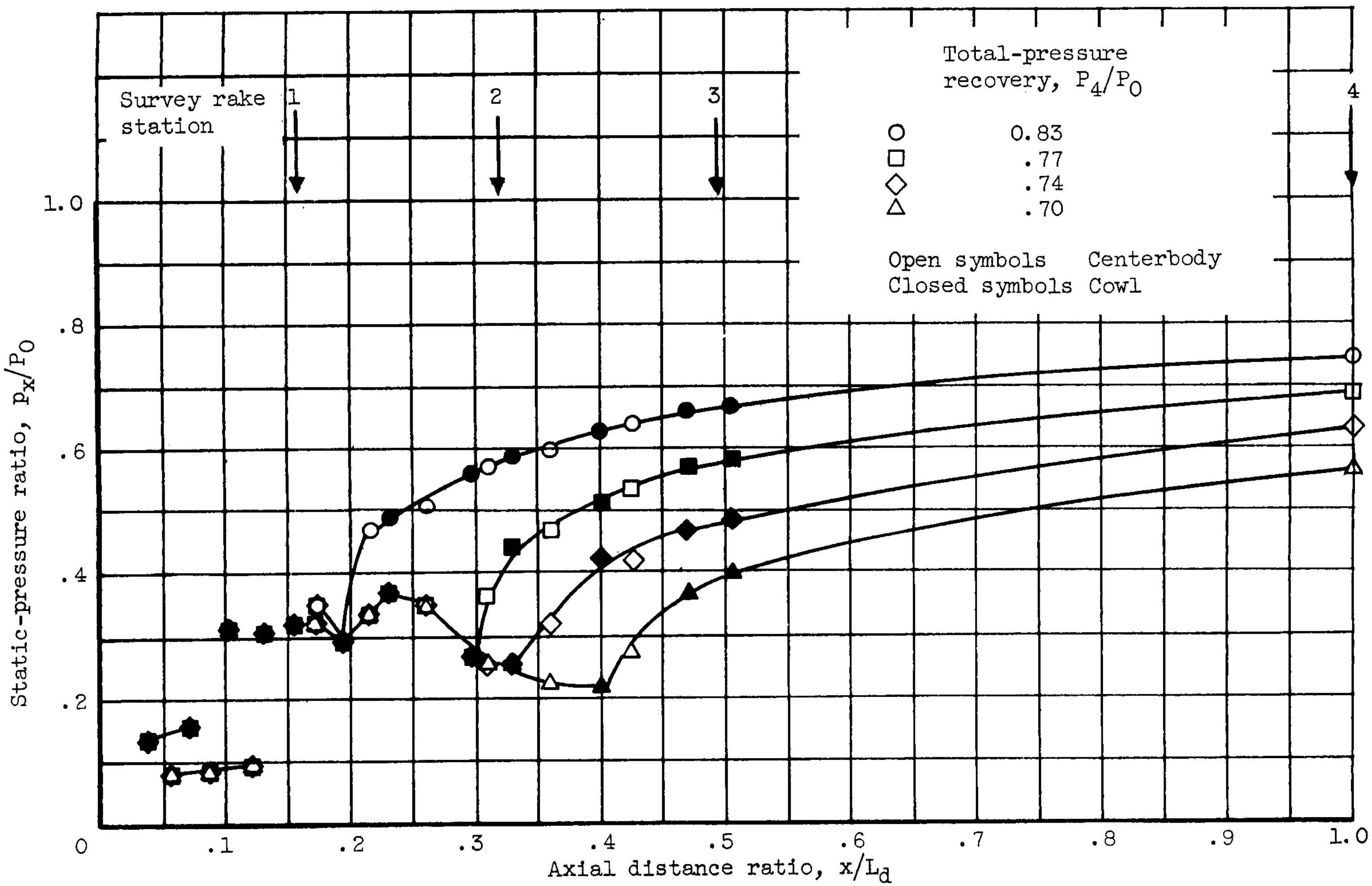

(a) Critical and supercritical operation at Mach 3.0.

Figure 12. - Static-pressure distributions with shoulder bleed only. Subsonic diffuser length, L_d, 29 inches.

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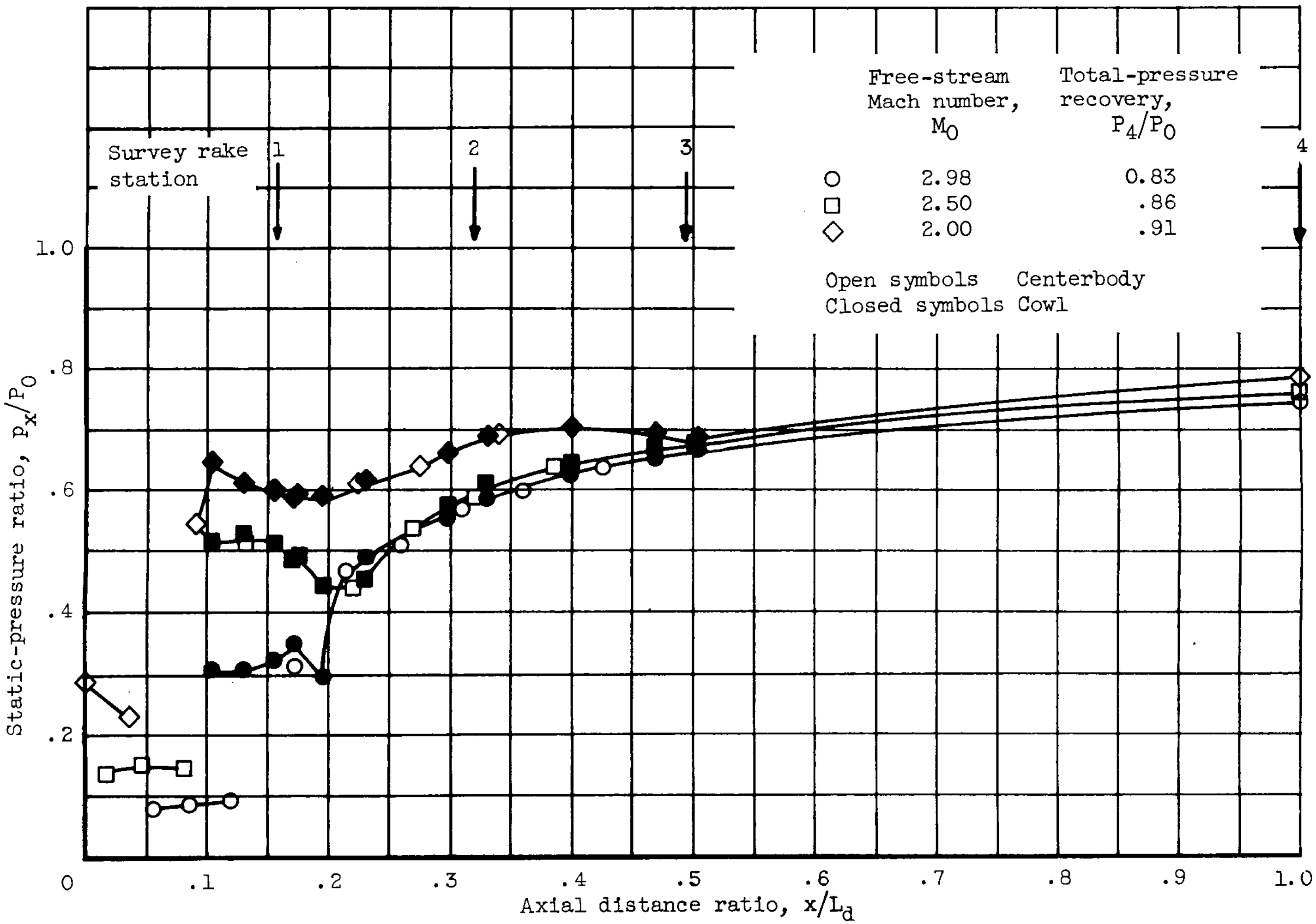

(b) On- and off-design operation at peak recovery.

Figure 12. - Concluded. Static-pressure distributions with shoulder bleed only. Subsonic diffuser length, L_d, 29 inches.

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