

NACA

RESEARCH MEMORANDUM

A PRELIMINARY INVESTIGATION OF METHODS FOR IMPROVING
THE PRESSURE-RECOVERY CHARACTERISTICS OF VARIABLE-
GEOMETRY SUPERSONIC-SUBSONIC DIFFUSER SYSTEMS

By Lowell E. Hasel and Archibald R. Sinclair

Langley Aeronautical Laboratory
Langley Field, Va.

NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS
WASHINGTON

October 16, 1957
Declassified April 15, 1958

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

A PRELIMINARY INVESTIGATION OF METHODS FOR IMPROVING
THE PRESSURE-RECOVERY CHARACTERISTICS OF VARIABLE-
GEOMETRY SUPERSONIC-SUBSONIC DIFFUSER SYSTEMS

By Lowell E. Hasel and Archibald R. Sinclair

SUMMARY

An investigation has been initiated to study methods for improving the pressure-recovery characteristics of two-dimensional, variable-geometry, supersonic-subsonic diffuser systems. The recovery characteristics of the basic configuration and of a configuration with an injector at the end of the supersonic diffuser have been determined at Mach numbers from 2.5 to 4.75, and at stagnation pressures from 40 to 205 pounds per square inch absolute. The Reynolds number based on the height of the test section and a stagnation pressure of 120 pounds per square inch absolute varied from 12.2×10^6 to 4.0×10^6 over the Mach number range from 2.5 to 4.75.

The pressure recovery of the basic configuration, which had a relatively long supersonic diffuser to minimize shock boundary-layer interaction effects, varied from 0.71 to 0.21 as the test-section Mach number varied from 2.5 to 4.75 for a stagnation pressure of 120 pounds per square inch absolute. These recoveries are significantly higher than the corresponding theoretical normal-shock recoveries which vary from 0.50 to 0.075. For a given contraction ratio of the supersonic diffuser, the effect of Reynolds number on pressure recovery was negligible. The maximum recovery decreased somewhat as the Reynolds number decreased because the amount of contraction of the supersonic diffuser decreased. Use of a Mach number 2.19 injector with a relatively high mass flow (injector to mainstream mass-flow ratio of 2.5 at a test-section Mach number of 4.0) at the end of the supersonic injector resulted in recoveries which varied from 0.65 to 0.56 as the test-section Mach number varied from 2.95 to 4.70.

The results indicate that the maximum test Mach number of existing supersonic wind tunnels with limited compression ratio and conventional normal-shock diffusers can be significantly increased by modifying the diffuser to incorporate a relatively long variable-geometry supersonic diffuser followed by an injector. For example, a compression ratio of 2.5 instead of about 20 is sufficient for a test Mach number of about 5. The power consumption of high Mach number tunnels with conventional normal-shock diffusers can be significantly reduced by modifying the tunnel to incorporate a relatively long variable-geometry supersonic diffuser.

INTRODUCTION

The rapid increase in the supersonic performance of airplanes and missiles has generated major technical problems in the design of the companion wind-tunnel research facilities. In less than a decade the operational speeds of airplanes have increased from near sonic to a Mach number of 3, while some missiles are already operating at a Mach number of 5 and above. This upsurge in flight speeds has made the Mach number range of existing wind-tunnel facilities inadequate.

At Mach numbers to about 5 the maximum test Mach number is generally determined by the compression ratio of the drive equipment. For conventional wind tunnels where the normal shock occurs essentially at test-section Mach number, a compression ratio of 2 is required for a Mach number of 2. A Mach number of 5, however, requires a compression ratio of 20. A continuous-operation facility for this Mach number range requires an elaborate compressor system with complex staging. There is a great need for research on methods for decreasing the compression-ratio requirements of supersonic wind tunnels, not only from the standpoint of simplifying the design of new facilities but also for improving the supersonic performance of those now in operation.

Numerous studies have been made of methods for improving the recovery characteristics of wind tunnels (refs. 1 to 8, for example). Most of these results are summarized in reference 1. The best results were obtained by the use of a variable-geometry supersonic diffuser downstream of the test section to reduce the Mach number at which the normal shock occurred. Further improvements might be obtained by the use of some form of boundary-layer control such as that used with inlets (refs. 9 to 11) since the measured recoveries of the wind-tunnel configurations are well below calculated values.

Existing low Mach number tunnels have relatively constant-volume compressors and constant-area test sections. If these tunnels are to be operated at higher Mach numbers a considerable amount of air must bypass the test section in order to satisfy the compressor flow requirements. Consideration may therefore be given to methods of increasing wind-tunnel recoveries by reintroducing into the tunnel the bypassed air by means of injectors. The effects of air injection immediately downstream of the test section have been considered in references 7 and 8. The resultant recoveries were not as good, however, as those obtained from the variable-geometry diffuser tunnels.

The combination of a variable-geometry supersonic diffuser and a supersonic injector has not been studied previously. Location of the injector at the end of the supersonic diffuser might improve the pressure recovery by several favorable effects. First, the injector air

may have a higher total pressure at the injection point than the main airstream since the latter has encountered viscous losses and the oblique-shock-wave losses in the supersonic diffuser. The injector air may therefore mix with main airstream and raise the overall total pressure. Second, the injector air may be introduced into the tunnel at a static pressure which is higher than the static pressure of the main airstream and thereby further decelerate the latter. Third, the injection air may serve as a means of boundary-layer control by reenergizing the boundary layer at the end of the supersonic diffuser and thereby reduce the losses due to the normal-shock—boundary-layer interaction.

It thus appears that several methods are available whereby the recoveries of supersonic wind-tunnel diffusers may be further improved. The effectiveness of these methods can only be determined by experimental studies. A two-dimensional, variable-geometry, supersonic-diffuser apparatus has therefore been built to study various methods of improving the pressure-recovery characteristics of supersonic wind tunnels. This apparatus has been designed so that the effects of supersonic-diffuser configuration and contraction ratio, boundary-layer control, air injection, Mach number, Reynolds number, and a typical research model configuration can be evaluated. Inasmuch as the problems of pressure recovery in supersonic inlets are similar to those of wind tunnels, the results of this investigation may assist in the design of high Mach number inlets.

This report presents the results obtained from the basic supersonic-diffuser configuration and some preliminary results obtained by air injection at the end of the supersonic diffuser. The Mach number was varied from 2.5 to 4.75. The Reynolds numbers corresponded to the values for a tunnel with a 4.5-foot, square test section and with stagnation pressures of 0.3 to 1.7 atmospheres.

SYMBOLS

A	area, sq ft
w	air flow, lb/sec
M	Mach number
p	static pressure, lb/sq ft
p _t	total pressure, lb/sq ft
δ	wall angle or flow turning angle, deg

Subscripts:

∞	free stream or test section
a	atmospheric
j	injector
ts	test section
av	mass-flow weighted average pressure
b,c,d,e,f,	diffuser-apparatus station (see figs. 1 and 2) and so forth
bc, ef	wall section of diffuser apparatus

DESIGN AND DESCRIPTION OF APPARATUS

General

The diffuser apparatus is shown schematically and pictorially in figures 1 to 4. The apparatus consisted of fixed side walls which were 6.75 inches apart and movable top and bottom walls. Each movable wall (figs. 1 and 2) was made up of 7 hinged sections. The names assigned to the various sections in figure 1 describe in general terms their respective functions when the injector was located at station d. In figure 1 all wall sections except the supersonic nozzle and subsonic diffuser have been drawn parallel to the center line to show their actual lengths. Typical wall settings which were used during the investigation are shown in figure 2 for configurations with and without the main injector. The assembled apparatus is shown in figure 3. The photographs of figure 4 were taken with one side wall removed to illustrate details of the various parts of the apparatus. The various wall sections were in random positions when the photographs were taken and these positions do not represent experimental wall settings.

Each section of the apparatus is described in detail in the following paragraphs of the text. Included in the description is a discussion of the basic considerations involved in the design of each section. This discussion has been included to describe the aerodynamic function of each section of the apparatus and to describe the overall design philosophy.

Supersonic Nozzle

The variable Mach number flow was generated by the nozzle shown in figures 1, 2, and 4(a). The nozzle was formed by fixed contour blocks which pivoted about fixed hinges at station 24. The nozzle was designed to produce a uniform flow at station 24 at a Mach number of 3.3. Other test Mach numbers from 2.5 to 4.75 were obtained by pivoting the blocks to change the height of the nozzle throat. It was recognized that this method of varying the Mach number would produce a nonuniform flow at station 24 at off-design Mach numbers. This method was chosen, nevertheless, because of its mechanical simplicity and because it was thought that some flow variation at station 24 would not seriously affect the accuracy of the results of the investigation.

Main Supersonic Diffuser

The supersonic diffuser (figs. 1, 2, and 4(a)) is a very important part of any diffuser apparatus designed for high pressure recovery. The purpose of the supersonic diffuser is to efficiently decelerate the air from the initial Mach number to a minimum supersonic Mach number at the normal shock. Theoretically the deceleration could be accomplished to a Mach number of 1 with no loss of pressure recovery in either a reverse supersonic nozzle or a very long, straight-walled, decreasing-area duct. Practically, of course, this ideal deceleration cannot be attained.

For wind-tunnel installations a straight-walled, variable-area, supersonic diffuser of minimum length is probably most suitable because of its mechanical simplicity. Several factors must be considered when selecting the diffuser length. In a diffuser of this type the deceleration of the air is accomplished by a series of oblique shock waves which reflect several times from the diffuser walls. The turning angle of these waves equals the diffuser wall angle. As the diffuser becomes shorter the strength of each shock wave must become stronger to accomplish the same overall Mach number reduction. Theoretical calculations indicate that as the shock waves become stronger the associated total-pressure losses increase. These losses become relatively large when the turning angle of each shock wave exceeds 5° or 6° .

As the diffuser becomes shorter and the wall angle increases to maintain the same area reduction, the possibility of creating boundary-layer separation increases because the static-pressure rise at the points of shock-wave reflection becomes larger. The pressure rise required to separate a turbulent boundary layer under these conditions has been studied experimentally (ref. 12, for example). The tests of reference 12 were conducted on a flat plate with a fully turbulent boundary layer and with a Reynolds number, based on momentum thickness, which exceeded several thousand. The results of this investigation are presented in

figure 5, and indicate that the boundary layer on a flat plate may separate when the turning angle of the reflected oblique shock wave exceeds about 8° . Since the boundary layer must traverse several shock reflections in the straight-wall supersonic diffuser, it would seem advisable to limit the shock-wave turning angle to 5° or 6° and thus minimize the possibility of boundary-layer separation.

With the preceding considerations in mind, a diffuser was designed with a length—test-section-height ratio of 4.37. This diffuser when set with a wall angle of 5° would decelerate the flow from a Mach number of 4.0 to a Mach number of 2.45 at station c. The top and bottom walls of the main supersonic diffuser pivoted about the fixed hinge at station 24 and incorporated a movable hinge and a sliding joint at the downstream end (figs. 1 and 4(a)). The relationship between the diffuser wall angle δ_{bc} and the diffuser-contraction ratio A_c/A_b is presented in figure 6. The Mach number at the end of the diffuser M_c has been computed for various values of initial Mach number M_∞ and contraction ratio A_c/A_b by the use of the oblique-shock-wave equations, neglecting viscous effects. These results are presented in figure 7.

Injector

The use of injectors after the test section as a means of increasing the diffuser pressure recovery has been discussed in references 7 and 8. No mention was made, however, of the advantages which may result from the combined use of a variable-geometry diffuser and an injector. Location of the injector at the end of the supersonic diffuser permits the injector air to enter the diffuser at a relatively low Mach number when compared to the test-section Mach number and at a total pressure which is higher than that of the main airstream since the latter has encountered viscous losses and oblique-shock-wave losses in the supersonic diffuser. The injector air may therefore mix with the main airstream and raise the average total pressure ahead of the normal shock. Location of the injector at the end of the supersonic diffuser may also be a method of boundary-layer control. The injected air may reenergize the boundary layer which has formed in the supersonic diffuser and thereby reduce the normal-shock boundary-layer interaction losses. Finally, as was pointed out in references 7 and 8, the injector air may be introduced into the tunnel at a static pressure which is higher than that of the main airstream and thereby further decelerate the latter. In this process the injector air is necessarily accelerated to some extent.

The injector installation is shown in figures 1, 2(a), and 4(b). The main injector section was designed to move in a vertical direction only. As a result the diffuser wall from stations 53.5 to 61.83 was

always parallel to the tunnel center line. The air was injected at station 61.83 through a nozzle whose axis was inclined 10° to provide more strength for the injector nozzle. The nozzle contours were designed for a Mach number of 2.0. The nozzle width diverged to some extent downstream of the throat for structural reasons and as a result, the injector Mach number based on the nozzle exit to throat areas was 2.19. The injector Mach number should be variable to obtain maximum pressure recovery over the test-section Mach number range, but for this apparatus such an arrangement was not practical. The injector Mach number was chosen so that over the anticipated range of test conditions the static-pressure differences between the merging streams would be as small as possible. Thus, the strengths of the resultant shock waves would also be kept to a minimum and the possibility of initiating boundary-layer separation lessened.

If the injector were to serve as a scheme for matching the air flow of the compressor and test section, the injector mass flow should vary with test-section Mach number. Such a design was not practical for the present apparatus. Therefore the constant mass-flow injector was designed to provide the necessary matching characteristics at a relatively low test-section Mach number. The ratio of the injector to the test-section air w_j/w_{ts} for the diffuser apparatus is presented as a function of test-section Mach number in figure 8. This ratio varied from 1 to 4.8 as the Mach number varied from 3.0 to 4.75. For comparison purposes the injector mass-flow ratio which would be available at various Mach numbers for a typical low-compression tunnel is also presented in figure 8. The tunnel was assumed to have a fixed test-section area and a constant volume compressor of the correct capacity for a Mach number of 2. For this tunnel, sufficient bypass air would be available for the injector used in this investigation at Mach numbers above 2.6.

Mixing Section

The mixing section (figs. 1, 2, 4(b), and 4(c)) was incorporated in the diffuser apparatus to promote mixing of the injector and main air-stream before encountering the normal shock. The length of the section was chosen in a somewhat arbitrary manner and was thought to be too short for adequate mixing at large values of supersonic-diffuser contraction ratio A_c/A_b . This wall section was parallel to the horizontal center line for all of the injector tests.

During the no-injector tests the upstream end of the mixing section was flush with the main injector surface as shown in figures 2(b) and 4(c). For these tests the wall was set at an angle of 1° to produce a slightly divergent section.

Second Supersonic Diffuser

The second supersonic-diffuser section was added to provide additional compression after the mixing section. The boundary layer which developed on the injector nozzle and mixing section surfaces was thought to be able to withstand some adverse-pressure gradient without separation. Use of the diffuser section should therefore increase the overall pressure recovery of the apparatus. Overall length considerations of the apparatus required that this section must be relatively short. The contraction ratio A_f/A_e of this section is a function of the contraction ratio A_c/A_b and the wall angle δ_{ef} . This relationship is given in figure 9. During the no-injector tests the wall angle of this section was set at 1° (fig. 2) to continue the diverging area passage of the mixing section.

Normal-Shock Section

The normal-shock section was designed to reduce the total pressure losses across the normal shock. Several investigations (refs. 13 to 15, for example) have shown that the deceleration from supersonic to subsonic speeds usually occurs through a series of shocks instead of a single shock when the pressure rise across the shock is sufficient to separate the boundary layer. These results also show that the complete shock-wave pattern should occur in a passage of nearly constant area if the normal-shock boundary-layer interaction losses are to be minimized. On the basis of this information the normal-shock section of the diffuser apparatus (figs. 1 and 2) was made 19 inches long. This length was somewhat arbitrary and was thought to be too short for large values of diffuser wall height at station f. As shown in figure 1 the normal-shock section included a sliding joint. The portion of the wall upstream of the joint had a wall angle of 2° ; the downstream section of the wall was designed to be parallel to the tunnel center line.

Subsonic Diffuser

The subsonic-diffuser design (figs. 1, 2, and 4(d)) was dictated by dimensional requirements of the overall apparatus instead of the conventional requirements for efficient subsonic deceleration of the air. As a result the diffuser wall angles for running conditions were relatively high (fig. 2). A splitter plate was placed on the tunnel center line for the entire length of the subsonic diffuser in an attempt to improve the diffusion. The actual effectiveness of such a fix is not known. It must be remembered that for this apparatus and for all high Mach number wind-tunnel diffusers designed for high recoveries the subsonic-diffuser performance would have only a minor effect on the

overall pressure recovery. If the overall length is limited, design compromises should be made in the subsonic-diffuser section rather than in the supersonic-diffuser section.

Downstream End of Apparatus

The downstream end of the apparatus extended from the end of the subsonic diffuser, station 116.25, to station 159 (figs. 1, 2, and 4(d)). This portion of the apparatus was designed to facilitate the experimental tests and would not be included in an actual wind-tunnel diffuser. The section from stations h to k was designed to provide a more nearly uniform distribution of total pressure at the rake station k. The exit total pressure was measured by a 15-tube rake located on the vertical center line of the apparatus at station k. Ten of the tubes, equally spaced from top to bottom, were manifolded to a small pressure-averaging chamber. The exit total pressure was assumed to be equal to the pressure in this chamber. The other 5 tubes, also equally spaced, were used to check the total-pressure distribution at station k. The butterfly valve was used to control the exit back pressure.

Typical Research Model

The research model, including its support system, which is usually located in the test section of a wind tunnel may have an adverse effect on diffuser pressure recovery. These effects were studied by use of a typical research model (fig. 10) which consisted of a body of revolution and sting at an angle of attack of 15° , and the associated support strut. All of the components were approximately scaled copies of the equipment used in the Langley 4- by 4-foot supersonic pressure tunnel. The leading edge of the strut was located at station 24.0.

TESTS

General

The diffuser apparatus was installed in a test cell of the Langley gas dynamics laboratory and was operated with dry air from the high-pressure air-storage facilities. Most of the tests were run at stagnation pressures of 40 to 205 pounds per square inch absolute and at an average stagnation temperature of about 130° F. The Reynolds number, based on a test-section height of 6.75 inches and a stagnation pressure of 120 pounds per square inch absolute, varied from 12.2×10^6 at

$M_\infty = 2.5$ to 4.0×10^6 at $M_\infty = 4.75$. These Reynolds numbers correspond to the values for a tunnel with a test section which is about 4.5 feet square and with a stagnation pressure of 1 atmosphere.

The pressure recovery was based on the exit total pressure at station k. This exit pressure was assumed to be the average pressure obtained by manifolding to a common chamber the pressures of 10 equally spaced total-pressure tubes located on the vertical center line of the apparatus.

The pressure data were measured with Bourdon-tube pressure gages or mercury manometer boards. These data and the wall positions, which were indicated by the counters visible in figure 3, were photographically recorded.

The diffuser airflow characteristics were continuously observed during the tests by means of a shadowgraph system. These shadowgraphs were photographed simultaneously with the pressure data.

Procedure

The pressure-recovery data were obtained in the following manner. After supersonic flow was established in the diffuser the movable walls were set in the desired positions. Then the back pressure was gradually raised by closing the butterfly valve until the flow in the main supersonic diffuser became subsonic. The average exit total pressure at station k was read visually on a pressure gage at the instant of flow breakdown and this pressure was used in calculating the pressure recovery.

During the tests, wall sections cd, de, ef (no-injector configuration only), and fg were maintained in the angular positions shown in figure 2. Preliminary tests with no injector indicated that the pressure recovery was not improved by increasing the angle of subsonic-diffuser wall sections de and ef to 3° .

All the starting data and minimum running-contraction-ratio data were obtained with the butterfly valve completely open and with the top and bottom wall angles beyond the minimum area station approximately as shown in figure 2(b).

Accuracy

The accuracy of the pressure-recovery data cannot be evaluated in specific numbers, but it is thought to be good on the basis of the facts discussed below, and on the basis of data repeatability which was about

± 0.01 to ± 0.02 , depending on the stagnation pressure. The accuracy is primarily dependent upon the uniformity of the total-pressure distribution at the rake station k. As previously mentioned in this report, the apparatus from stations h to k was designed to promote a uniform distribution at the rake. The uniformity in the vertical plane of the rake was checked by 5 total-pressure tubes equally spaced from top to bottom and was found to be about ± 0.01 of the stagnation pressure. The numerically averaged pressure of these 5 tubes agreed with the pressure obtained from the 10 manifolded total-pressure tubes. No checks were made of the horizontal distribution. However, it would seem reasonable that this distribution should be as good as the vertical distribution.

The Mach number distribution at the end of the supersonic nozzle was not uniform (fig. 11) at the higher test-section Mach numbers because of the nozzle design. These distributions were obtained by measuring the static pressures on one side wall at station 23.7 and computing the Mach number from the static to stagnation pressure ratio. The test-section Mach number M_∞ was obtained by averaging these distributions. Since the survey did not extend to the top and bottom walls the resultant accuracy of M_∞ is probably no better than ± 0.1 at the higher Mach numbers. An error of this magnitude was considered acceptable for the type of investigation being made.

Corrections

The total pressure of the injector air was about 10 percent less than the corresponding pressure $p_{t,\infty}$ of the test-section air because of piping losses. These losses have been taken into account by use of an average total pressure $p_{t,av}$ as the reference pressure for all data obtained with the injector configuration. This pressure is the mass-flow-weighted average total pressure of the injector and test-section airstreams. The maximum correction occurred at a Mach number of 4.70 where the average total pressure was about 0.92 of the test-section total pressure. Use of $p_{t,av}$ instead of $p_{t,\infty}$ as the reference pressure raised the maximum pressure recovery at a Mach number of 4.70 from a measured value of 0.52 to a corrected value of 0.56.

RESULTS AND DISCUSSION

Without Injection

Pressure recovery.- The basic pressure-recovery results which were obtained at Mach numbers of 2.50 to 4.75 from the diffuser apparatus

with no injector at station 61.8 are presented in figure 12. The measured recovery is plotted as a function of the contraction ratio of the main supersonic diffuser A_c/A_b . The computed recovery accounts for the oblique shock losses in the main supersonic diffuser but neglects the effects of viscosity. The normal shock was assumed to occur at a Mach number equal to M_c . The measured pressure recoveries (fig. 12) increase at a given Mach number with decreasing values of contraction ratio of the supersonic diffuser A_c/A_b as would be expected. The rate of increase is less than predicted by the computed recoveries, probably because the viscous losses due to normal-shock—boundary-layer interaction increase as the adverse pressure gradient in the supersonic diffuser becomes larger. The pressure recoveries at a given contraction ratio were not significantly affected by Reynolds number (varying stagnation pressure) and were reduced slightly when the model was placed in the airstream.

The maximum measured pressure recoveries and the corresponding minimum running-contraction ratios of the supersonic diffuser A_c/A_b are summarized in figures 13 and 14. The computed recovery in figure 13 is based on the contraction ratio at which the maximum recovery was measured at 120 pounds per square inch absolute.

The maximum measured pressure recovery (fig. 13) at a stagnation pressure of 120 pounds per square inch absolute varied from 0.71 at $M_\infty = 2.50$ to 0.24 at $M_\infty = 4.50$. (This stagnation pressure corresponds to the Reynolds number for a tunnel with a 4.5-foot square test section and with atmospheric stagnation pressure). These maximum recoveries are significantly higher than the corresponding test-section normal-shock recoveries which vary from 0.50 to 0.075, but are from 0.15 to 0.24 lower than the computed recoveries. Most of the computed loss of total pressure is due to the normal-shock loss. The losses due to the oblique shock system in the supersonic diffuser are a maximum at $M_\infty = 4.50$ and are equal to about 0.07 of the stagnation pressure. At $M_\infty = 2.5$ the oblique shock losses are negligible.

The maximum recoveries decreased as the stagnation pressure decreased because the minimum running-contraction ratios (fig. 14) became larger. This is probably a Reynolds number effect. The addition of a model reduced the maximum recoveries at a stagnation pressure of 120 pounds per square inch absolute by 0.05 to 0.10. This loss in pressure recovery was also primarily caused by an increase in the minimum running-contraction ratios (fig. 14).

The variation of the minimum running-contraction ratio with Mach number (fig. 14) is similar to that required to isentropically decelerate the Mach number to 1.0 at station c. The minimum values at a

stagnation pressure of 120 pounds per square inch absolute are about 0.1 larger than the theoretical minimum contraction ratios. This difference increased when the Reynolds number was decreased and when a model was placed in the airstream. The factors which may determine the minimum running-contraction ratio are discussed in a subsequent section of this report.

In principle the diffuser apparatus is similar to an internal-contraction supersonic inlet, and the comparison of the respective recoveries is presented in figure 15. Two of the inlets (refs. 16 and 17) were two-dimensional designs with one movable straight wall. The supersonic diffuser of the third two-dimensional inlet (ref. 18) was shaped like a reverse supersonic nozzle. The other inlet (ref. 19) was axisymmetric with a movable, conical centerbody. The maximum recoveries of the diffuser apparatus are 0.06 to 0.08 less than the best recoveries obtained from the internal compression inlets. Two factors should be considered in the comparison of these recoveries. First, the recoveries of the diffuser apparatus would probably have been improved to some extent if the boundary layer generated by the supersonic nozzle had been removed at station b. Second, the supersonic diffusers of the inlets were at least 50 percent shorter than that of the diffuser apparatus. This factor probably was favorable for the recoveries of the diffuser apparatus. Significant increases of pressure recovery were obtained in reference 18 when a portion of the side-wall boundary layer was removed near the inlet throat. Similar increases in recovery could probably be obtained from the diffuser apparatus. For wind-tunnel applications, removal of this air would require more pumping facilities and might not be practical.

Flow characteristics in the supersonic diffuser.- The static-pressure distributions measured on the top wall of the supersonic diffuser at maximum recovery conditions and the corresponding shock-wave patterns are shown in figures 16 and 17. The pressures which were measured with no model in the airstream (fig. 16) are about the same as those computed by the oblique shock theory. These pressures are indicative of the maximum static-pressure rise in the supersonic diffuser since they were obtained at minimum values of A_c/A_p . The pressure rise (ratio of final to initial static pressure) varies from 3.6 at $M_\infty = 2.50$ to 12 at $M_\infty = 4.50$. It seems probable at $M_\infty = 4.50$ that the orifice at station 51.3 was too far upstream to measure the entire rise due to the last shock reflection and that the actual pressure ratio may have been about 20. These measured rises are significantly greater than the rise which could have been tolerated without separation at one shock reflection (fig. 5), and indicate the importance of the long supersonic diffuser for reducing the Mach number at the normal shock.

The computed shock-reflection points are indicated by the calculated pressure distributions. The photographs (fig. 16) indicate that the

actual reflection points are upstream of the predicted reflections as might be expected since the boundary-layer growth tends to increase the wall angle and oblique shock angle, and decrease the Mach number at any given point. Some indication of the thickness of the boundary layer may be gained by observing the shock pattern near the wall reflection points.

The distributions and shadowgraphs of figure 16 do not show conclusively the factors which govern the minimum running-contraction ratios of the supersonic diffuser. It was observed throughout the entire Mach number range that with no model the minimum values of contraction ratio were reached when the fourth oblique shock reflection moved upstream past station c into the supersonic diffuser. The significance of this observation, if any, is not known at present. It was not observed to occur when the model was in the airstream (fig. 17).

Although conclusive proof is not available it would seem probable that the minimum contraction ratios are probably determined by some choking phenomena at station c. This choking may be caused by boundary-layer separation or by too much geometric contraction of the supersonic diffuser. The latter phenomena would be more likely to occur at low test-section Mach numbers. Measurement of the shock-wave angles at the end of the supersonic diffuser indicates, however, that at low test-section Mach numbers the flow outside the boundary layer is still appreciably above a Mach number of 1 and should not choke at station c because of too much geometric contraction. At the lower test-section Mach numbers it is thought that boundary-layer separation on the moving walls would not be likely to occur since the local and overall static pressure rises are less relative to the pressure-rise data for separation (fig. 5) than at the higher test-section Mach numbers. At the higher Mach numbers it is more likely that the controlling factor could be separation on the moving walls. It does not appear logical, however, that the separation can be attributed directly to the fourth reflection moving into the supersonic diffuser since this occurred throughout the Mach number range. Perhaps the controlling factor throughout the Mach number range is the separation of the relatively thick side-wall boundary layer. Further studies would be required to establish this fact.

With the model in the airstream the maximum static pressure rise at each Mach number (fig. 17) was somewhat lower than without the model because the minimum contraction ratios were larger. No calculated pressure distributions are presented because of the complex shock pattern in the diffuser. The minimum contraction ratios now occur before the fourth shock reflection moves upstream of station c.

Some details of the side-wall boundary-layer flow may be deduced from figure 17(a) from the oil-flow pattern on the window. The effect of the static-pressure distribution in deflecting the wall boundary layer toward

the diffuser center line is evident in the upstream portion of the picture. Further downstream a small area of separation apparently exists near the wall center line where the shock waves from the top and bottom wall intersect. The existence of this local area of separation at a relatively low static-pressure rise is probably a result of the thick boundary layer which exists on the side-wall center line because of the reduction of side-wall area and the static-pressure effects previously noted. The piling up of the boundary layer on the side-wall center line has been noted in reference 18.

Starting characteristics.- The starting contraction- and compression-ratio characteristics of the basic diffuser apparatus are shown in figures 18 and 19. These data were obtained with atmospheric back pressure.

The starting compression ratios are a function both of the model in the airstream and the contraction ratio of the supersonic diffuser. The latter effect has been previously noted (ref. 20, for example). With no model in the airstream the minimum starting-compression ratios at $M_\infty = 4.0$ and above were less than the theoretical values which were based on a normal-shock recovery at test-section Mach number. As mentioned in reference 20 the separation in the supersonic nozzle apparently reduces the test-section Mach number during the starting process and thereby reduces the starting-compression ratio.

The starting-contraction ratios (fig. 19) are independent of Reynolds number, but are affected by the model in the airstream. These ratios are less than the theoretical values because of the reduced test-section Mach number which occurs during the starting process. This effect was also noted in reference 20.

With Injection

Pressure recovery.- The basic data which were obtained with the injector at station 61.8 are presented in figures 20 and 21. In the top portions of figure 20 the data obtained with no model in the airstream are plotted as a function of the contraction ratio A_f/A_e of the second supersonic diffuser for various values of A_c/A_b . These data have been cross-plotted on the bottom of figure 20 as a function of A_c/A_b . The computed pressure recoveries were obtained by the use of the injector equations presented in reference 7. For these computations, Mach number and total pressure at station c were determined by oblique shock theory, and the normal shock was assumed to occur at station f. In figure 21 the pressure-recovery data which were obtained with the model in the airstream are presented.

The pressure recoveries obtained with no model in the airstream (fig. 20) vary with contraction ratio in a manner which is generally similar to that of the computed recoveries. The actual recoveries are about 0.05 to 0.17 lower than the computed results. The minimum running contraction ratios of the main supersonic diffuser A_c/A_b (fig. 14) were not significantly affected by the injector. Only a limited amount of data were obtained with the model in the airstream. These data, however, were obtained at near minimum running-contraction ratios and indicate the maximum recoveries to be expected under these conditions.

The injector data are summarized in figure 22. Maximum recoveries which were obtained with no model in the airstream and with minimum values of A_f/A_e varied from 0.64 at $M_\infty = 2.95$ to 0.56 at $M_\infty = 4.70$. These recoveries were reduced about 0.08 when the contraction of the second supersonic diffuser was eliminated ($A_f/A_e = 1.0$), or when the model was placed in the airstream. With the model in the airstream and with $A_f/A_e = 1.0$, the maximum recoveries varied from 0.55 at $M_\infty = 2.95$ to 0.45 at $M_\infty = 4.45$.

The recoveries of the injector configurations are appreciably higher than the recoveries of the basic diffuser apparatus at the higher Mach numbers (fig. 22). At $M_\infty = 4.7$ the maximum recoveries with and without injector (with no model) were 0.56 and 0.21, respectively. The large increase in recovery may result from several factors. First, the total pressure of the injector air may be greater than the total pressure of the main airstream because of the viscous and oblique shock losses which occur in the main airstream. Hence, when the relatively large quantity of injector air (fig. 8) mixes with the main airstream, the total pressure of the merged streams may be greater than that of the main airstream with no injector. No surveys have been made to date to determine the total pressure of the main airstream at the injection station and no estimates of this mixing effect can therefore be made. Second, some aerodynamic compression of the main airstream may occur in the mixing section if the static pressure of injector air is greater than the static pressure of the main airstream. This effect, however, is thought to be small at maximum recovery conditions on the basis of the static-pressure measurements of figure 16. Third, the injector air may be acting as a boundary-layer control device by reenergizing in the mixing section the boundary layer of the top and bottom diffuser walls, and reducing the normal-shock boundary-layer interaction losses. It should be mentioned that with the injector configuration, the wall boundary layer at the normal shock is generally relatively thin since most of the wall surface originates at the injector. At the higher Mach numbers a large portion of the air enters the tunnel through the injectors and one might expect the recovery to be primarily dependent on the characteristics of the injected air. Hence, the measured recovery might be expected to be somewhat less than 0.63, the theoretical normal-shock recovery at the

injector Mach number of 2.19 and to be relatively independent of test-section Mach number. With no model in the airstream and $A_f/A_e = 1.0$ the measured recoveries (fig. 22) varied from 0.53 to 0.48 as the test-section Mach number varied from 3.9 to 4.7. As A_f/A_e becomes less than 1.0 both the theoretical and experimental recoveries would be expected to increase.

At the lower test-section Mach numbers the recoveries could undoubtedly have been improved by the use of lower injector Mach numbers. As will be mentioned later the injector Mach number could not have been significantly reduced at the higher test-section Mach numbers.

Flow details.— The flow details at near maximum pressure-recovery conditions are shown in figures 23(a) to 23(d). The wake which leaves the top surface of the main supersonic diffuser at station d gives an indication of the static pressure of the diffuser and injector airstreams at station d. Inasmuch as the axis of the injector was inclined 10° inward with respect to the tunnel center line, equal pressures in the two streams would be indicated (to a first approximation) by a wake inclined at 5° toward the center line. At $M_\infty = 2.95$ the wake is inclined away from the tunnel axis indicating that the diffuser air expands at station d and the injector air is compressed by a significant amount. At $M_\infty = 3.90$ and 4.30 the wake is inclined by slightly less or more, respectively, than 5° , indicating that at these Mach numbers the static pressures of the two streams are about equal. At $M_\infty = 4.70$, the pressure of the diffuser air appears to be less than that of the injector. The small curved shock (figs. 23(b) to 23(d)) which exists near the tunnel center line between station e and f is thought to indicate that the side-wall boundary has been separated by the shock waves generated by the second supersonic diffuser at station e.

It was previously mentioned that at the higher Mach numbers the injector Mach number was at a near-minimum value. This statement is based on the flow phenomena shown in figures 23(d) to 23(g). The photographs indicate that as the static-pressure difference between the injector and main airstreams increased (A_c/A_b increased for a given M_∞), a normal shock formed in the main airstream (fig. 23(f)). In figure 23(g) the static-pressure difference was large enough to force the normal shock to move upstream of station d. Further increases of the pressure difference (A_c/A_b increased to values greater than 0.270) forced the normal shock upstream of the diffuser into the supersonic nozzle. A similar effect would be expected if the contraction ratio of the main supersonic diffuser was maintained at a near-minimum value and the static pressure of the injector air was increased (injector Mach number decreased).

Effect of Diffuser Configuration on Wind-Tunnel Performance

Pressure recovery.— The pressure recoveries of supersonic wind tunnels in which the normal shock occurs at approximately test-section Mach number (no supersonic diffuser) are about 70 to 85 percent of the normal-shock recoveries (fig. 24). Data presented in references 7 and 8 indicate that these basic recoveries may be significantly improved by the use of

air injection $\left(0.9 = \frac{w_j}{w_{ts}} = 2.2\right)$ just downstream of the test section to reduce, mainly by aerodynamic compression, the normal-shock Mach number. Still larger increases of the basic recoveries have been obtained in references 2, 3, 4, and 6 by the use of a variable-geometry supersonic diffuser to reduce the normal-shock Mach number.

The recoveries obtained with the injector at the end of the supersonic diffuser are much higher at the higher Mach numbers than any previously reported recoveries for wind-tunnel diffusers. Conversely, the compression ratios are lower. At $M_\infty = 5.0$, for example, the compression ratios required by a conventional normal-shock diffuser, a design with variable-geometry supersonic diffuser, and a design with both a supersonic diffuser and injector are about 20, 10, and 1.9, respectively. For an existing tunnel with a conventional, fixed-diffuser design, and a compression ratio of about 2.5, the advantages of modifying the diffuser design to incorporate a variable-geometry supersonic diffuser and injector are obvious. Taking into account the recoveries which may be expected with a model in the airstream the available compression ratio of the existing drive system should be sufficient to increase the test-section Mach number from about 2.5 to 5.0. This assumes that the tunnel may be started at low test-section Mach numbers.

This method of starting and operating a tunnel was demonstrated with the diffuser apparatus. Supersonic flow in the test section was established at $M_\infty = 2.5$, and the stagnation pressure increased to 37 pounds per square inch absolute to equal a compression ratio of 2.5 with atmospheric back pressure. Then the Mach number was continuously increased to 4.5 without increasing the stagnation pressure. As the Mach number increased, the values of the contraction ratios A_c/A_b and A_f/A_e were decreased according to a preset schedule. It was later determined by measuring the pressure recoveries that the required compression ratio for this demonstration had not exceeded 2.0, although the Reynolds numbers were relatively low. During this demonstration the model support strut was in the airstream.

The recoveries of the subject diffuser with no injector were about 0.05 to 0.15 higher than the recoveries of other configurations with

variable-geometry supersonic diffusers. Analysis of the data of reference 6 (which had a configuration very similar to subject diffuser) indicates that at equal contraction ratios the recoveries of the subject diffuser and the diffuser of reference 6 were about equal. The difference in maximum recoveries is apparently primarily due to a difference in the minimum running contraction ratios. This difference may be due to the lower Reynolds numbers of the tests of reference 6. For this to be true, however, the Reynolds number effects must become proportionately larger as the numbers approach those of reference 6.

Power.- The power requirements of a wind tunnel are a function not only of the pressure recovery but also of the weight flow of air in the tunnel circuit. Since the high recoveries of the injector configuration were obtained with large air-flow quantities this configuration may not be an optimum as far as power consumption is concerned. This fact is illustrated in figure 25 by the computed power characteristics of several hypothetical wind tunnels having the pressure recovery and weight-flow characteristics shown in the figure.

Three hypothetical tunnels are considered. One tunnel has the characteristics which might be obtained from an existing relatively low Mach number tunnel which has been modified to incorporate a variable-geometry supersonic diffuser and injector (tunnel 1, fig. 25). The total mass flow was assumed to be constant over the Mach number range (curve n, fig. 25) and the recovery (curve a) equal to that obtained during the present investigation with a model in the airstream. A second tunnel (tunnel 2) was assumed to have the same recovery as the first tunnel but to have a compressor capable of matching the air requirements of the constant-area test section and the injector of the present investigation (curves o + p). The third tunnel (tunnel 3) incorporates a variable-geometry supersonic diffuser and has a compressor system which can match the air requirements of a constant-area test section (curve p). The recoveries of this configuration were assumed equal to those of the present investigation (curve b). The reference tunnel is a conventional design with a recovery equal to 85 percent of the normal-shock recovery and with compressors matched to supply only the air required by the test section.

If the reference tunnel were modified to incorporate a variable-geometry diffuser without an injector, the power consumption would be approximately one-half because of the higher pressure recoveries. For large facilities this would amount to a very significant reduction of operating costs. Further modification of the reference tunnel and drive system to include the injector studied in the present investigation results in less power savings than the initial modification. This conclusion might be altered if significant increases in recovery could be obtained with low injector mass flows. As previously mentioned, the use of the variable-geometry diffuser and injector permit conventional

tunnels designed for a Mach number of 2 to operate at Mach numbers up to about 5. The relative power consumption, however, becomes very large at the higher Mach numbers because of the constant airflow throughout the Mach number range.

CONCLUSIONS

An investigation has been initiated to study methods for improving the pressure-recovery characteristics of two-dimensional, variable-geometry, supersonic-subsonic diffusers. The recovery characteristics of the basic supersonic diffuser configuration and of a configuration with an injector at the end of the supersonic diffuser have been determined at Mach numbers from 2.5 to 4.75, and at stagnation pressures from 40 to 205 pounds per square inch absolute. The test section was 6.75 inches square. The Reynolds numbers based on the height of the test section and a stagnation pressure of 120 pounds per square inch absolute varied from 12.2×10^6 to 4.0×10^6 for a Mach number range from 2.5 to 4.75. The following conclusions have been obtained.

1. The pressure recovery of the basic configuration, which had a relatively long supersonic diffuser to minimize shock boundary-layer interaction effects, varied from 0.71 at a Mach number of 2.5 to 0.21 at a Mach number of 4.75 for a stagnation pressure of 120 pounds per square inch absolute. These recoveries are significantly higher than the corresponding normal-shock theoretical recoveries which vary from 0.50 to 0.075.

2. Analysis of the results from the basic configuration and of other results from a similar configuration indicate that the effect of Reynolds number on pressure recovery is negligible at a given contraction ratio of the supersonic diffuser. As the Reynolds number decreases, the amount of contraction which can be obtained in the supersonic diffuser decreases and the pressure recovery therefore decreases. This effect was relatively small for the Reynolds number range of the present investigation, but appears to become larger at lower Reynolds numbers.

3. Use of a relatively high mass-flow injector (injector to main stream mass flow ratio of 2.5 at a Mach number of 4.0) at the end of the supersonic diffuser resulted in recoveries which varied from 0.65 at a Mach number of 2.95 to 0.56 at a Mach number of 4.70.

4. The pressure recoveries of both configurations were reduced 0.05 to 0.10 by a typical research model and support system located at the end

of the supersonic nozzle. These recoveries were lower primarily because the minimum running-contraction ratios of the supersonic diffuser were larger.

5. The maximum test Mach number of existing supersonic wind tunnels with limited compression ratio and conventional normal-shock diffusers can be significantly increased by modifying the diffuser to incorporate a relatively long variable-geometry supersonic diffuser followed by an injector. For example, a compression ratio of 2.5 instead of about 20 is sufficient for a test-section Mach number of about 5.

6. The power consumption of high Mach number tunnels with conventional normal-shock diffusers can be significantly reduced by modifying the tunnel to incorporate a relatively long variable-geometry supersonic diffuser.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., July 29, 1957.

REFERENCES

1. Lukasiewicz, J.: Diffusers for Supersonic Wind Tunnels. Jour. Aero. Sci., vol. 20, no. 9, Sept. 1953, pp. 617-626.
2. Neumann, E. P., and Lustwerk, F.: High-Efficiency Supersonic Diffusers. Meteor Rep. No. 56 (Contract NOrd 9661), M.I.T., June 1950.
3. Wegener, Peter P., and Lobb, R. Kenneth: NOL Hypersonic Tunnel No. 4 Results II: Diffuser Investigation. NAVORD Rep. 2376, U. S. Naval Ord. Lab. (White Oak, Md.), May 1952.
4. Bertram, Mitchel H.: Investigation of the Pressure-Ratio Requirements of the Langley 11-Inch Hypersonic Tunnel With a Variable-Geometry Diffuser. NACA RM L50I13, 1950.
5. Eggink, H.: The Improvement in Pressure Recovery in Supersonic Wind Tunnels. Rep. No. Aero.2326, British R.A.E., May 1949.
6. Diggins, J. L., and Lange, R. H.: A Systematic Study of a Variable Area Diffuser for Supersonic Wind Tunnels. NAVORD Rep. 2421, U. S. Naval Ord. Lab. (White Oak, Md.), Dec. 1952.
7. Hunczak, Henry R., and Rousso, Morris D.: Starting and Operating Limits of Two Supersonic Wind Tunnels Utilizing Auxiliary Air Injection Downstream of the Test Section. NACA TN 3262, 1954.
8. Spiegel, Joseph M., Hofstetter, Robert U., and Kuehn, Donald M.: Applications of Auxiliary Air Injectors to Supersonic Wind Tunnels. NACA RM A53I01, 1953.
9. Obery, Leonard J., and Cubbison, Robert W.: Effectiveness of Boundary-Layer Removal Near Throat of Ramp-Type Side Inlet at Free-Stream Mach Number of 2.0. NACA RM E54I14, 1954.
10. Mossman, Emmet A., Pfyl, Frank A., and Lazzeroni, Frank A.: Fuselage Side Inlets - A Study of Some Factors Affecting Their Performance and a Comparison With Nose Inlets. NACA RM A55F29, 1956.
11. Hasel, Lowell E.: Investigation at Mach Numbers of 1.41, 1.61, and 1.82 of Two Variable-Geometry Inlets Having Two-Dimensional Compression Surfaces. NACA RM L54K04, 1956.
12. Bogdonoff, Seymour M.: Some Experimental Studies of the Separation of Supersonic Turbulent Boundary Layers. Rep. No. 336 (Contract No. AF 18(600)-498), Princeton Univ., Dept. Aero. Eng., June 1955.

13. Lustwerk, Ferdinand: The Influence of Boundary Layer on the "Normal" Shock Configuration. Meteor Rep. No. 61 (Contract NOrd 9661), M.I.T., Sept. 1950.
14. Douglass, W. M.: Effects of Boundary Layer on Supersonic Diffuser Design. Preprint No. 290, S.M.F. Fund Preprint. Inst. Aero. Sci., July 1950.
15. McLafferty, G. H., Krasnoff, E. L., Ranard, E. D., Rose, W. G., and Vergara, R. D.: Investigation of Turbojet Inlet Design Parameters. Rep. R-0790-13, United Aircraft Corp. Res. Dept., Dec. 1955.
16. Scherrer, Richard, and Gowen, Forrest E.: Preliminary Experimental Investigation of a Variable-Area, Variable-Internal-Contraction Air Inlet at Mach Numbers Between 1.42 and 2.44. NACA RM A55F23, 1955.
17. Ranard, E. D.: Exploratory Tests of a Variable-Geometry Convergent-Divergent Inlet at Subsonic and Supersonic Mach Numbers. Rep. R-13534-6, United Aircraft Corp. Res. Dept., Mar. 22, 1954.
18. Gunther, Fred: Development of a Two-Dimensional Adjustable Supersonic Inlet. Progress Rep. No. 20-247 (Contract No. DA-04-495-Ord 18, Ord. Dept.), C.I.T., Jet Propulsion Lab., Nov. 20, 1954.
19. Mossman, Emmet A., and Pfyl, Frank A.: An Experimental Investigation at Mach Numbers From 2.1 to 3.0 of Circular-Internal-Contraction Inlets With Translating Centerbodies. NACA RM A56G06, 1956.
20. Moore, John A.: Investigation of the Effect of Short Fixed Diffusers on Starting Blowdown Jets in the Mach Number Range from 2.7 to 4.5. NACA TN 3545, 1956.

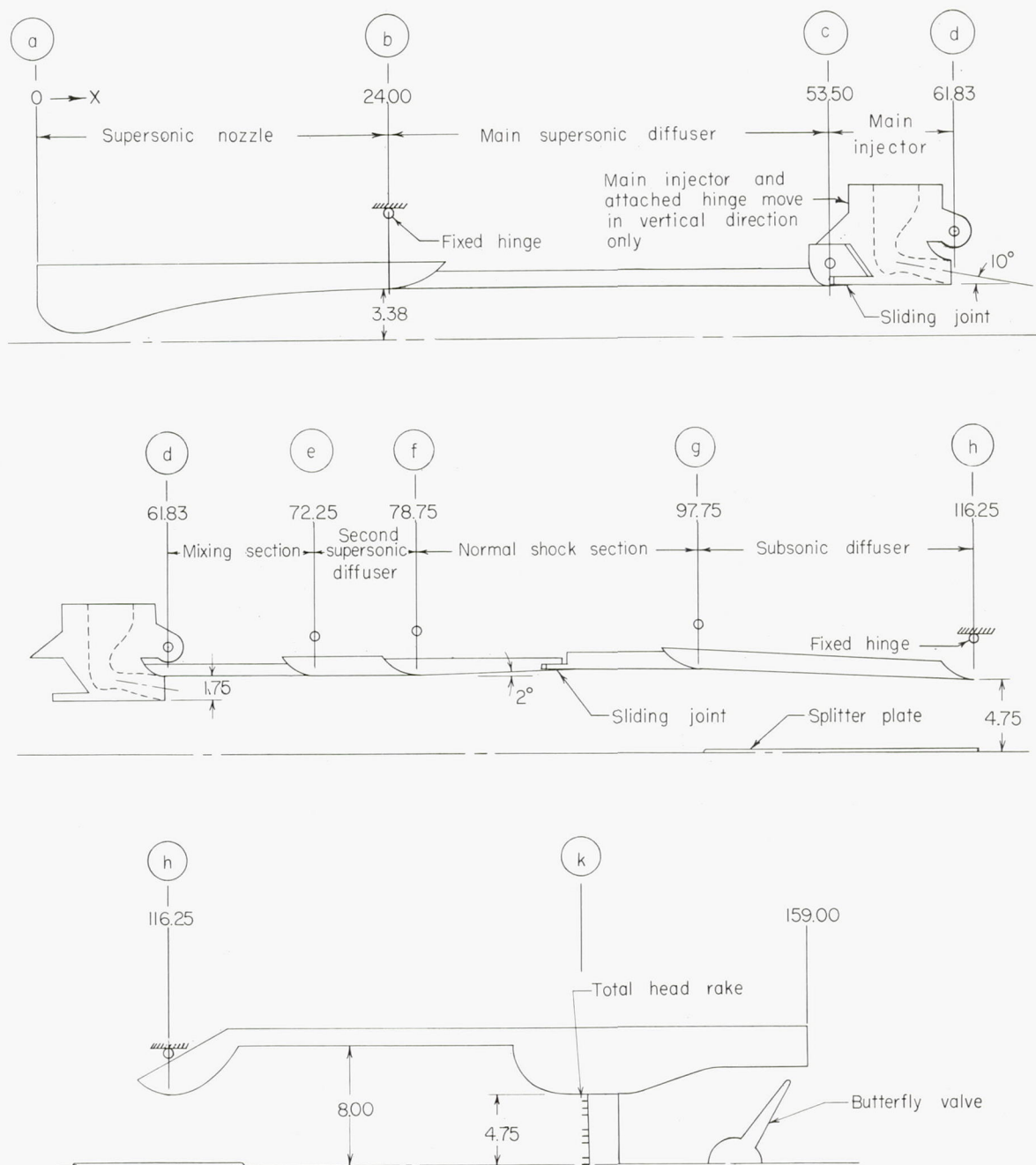
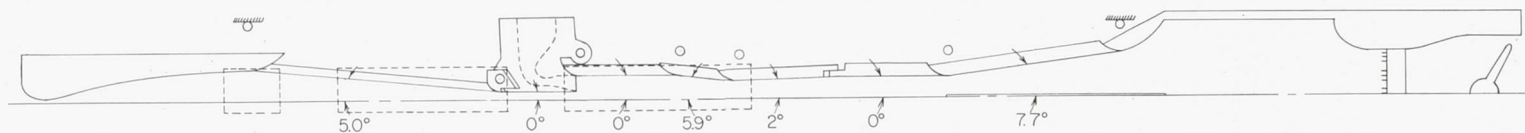
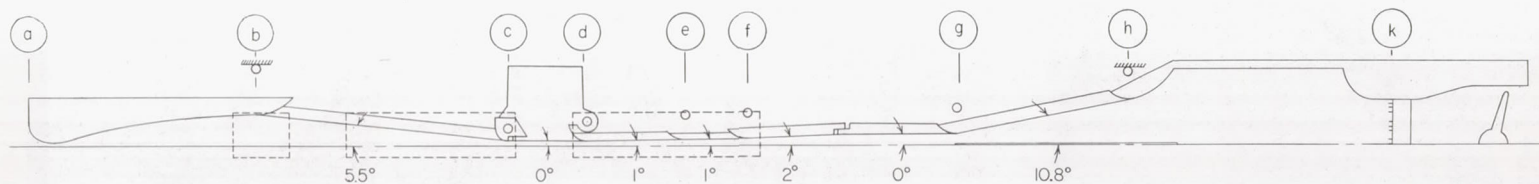


Figure 1.- Schematic drawing of variable-geometry diffuser apparatus.
All dimensions are in inches.

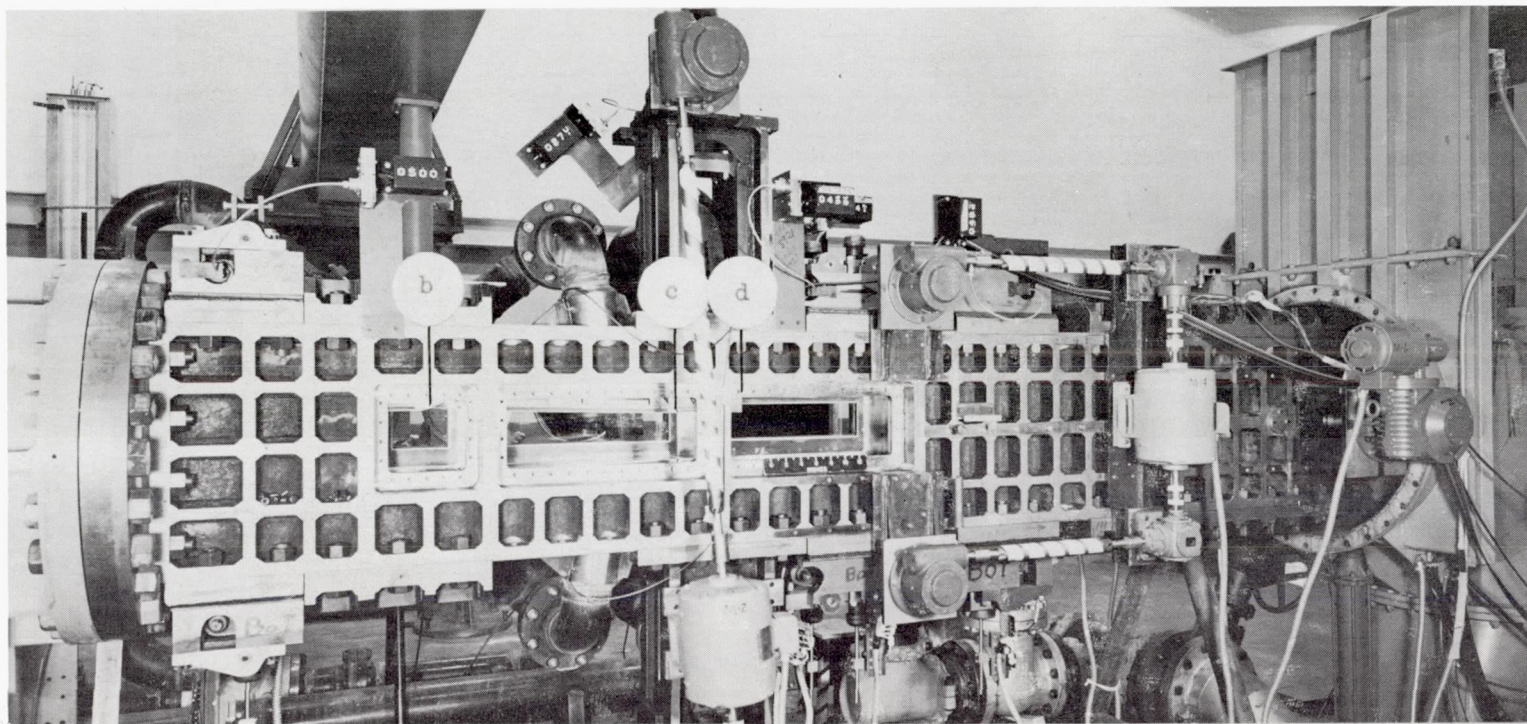


(a) $A_c/A_b = 0.240$; with main injector; $A_f/A_e = 0.738$.



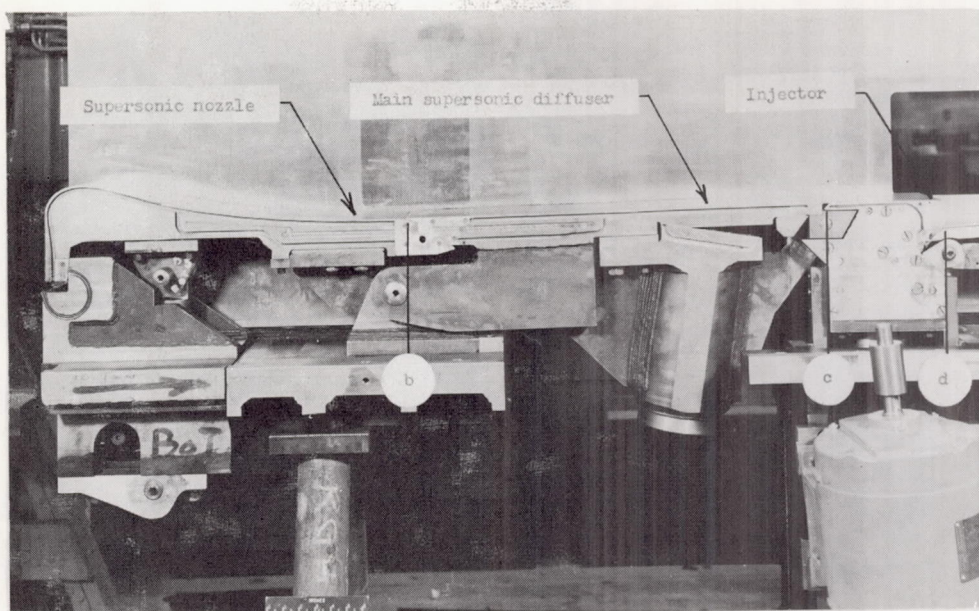
(b) $A_c/A_b = 0.166$; without main injector.

Figure 2.- Typical wall settings of variable-geometry diffuser apparatus.

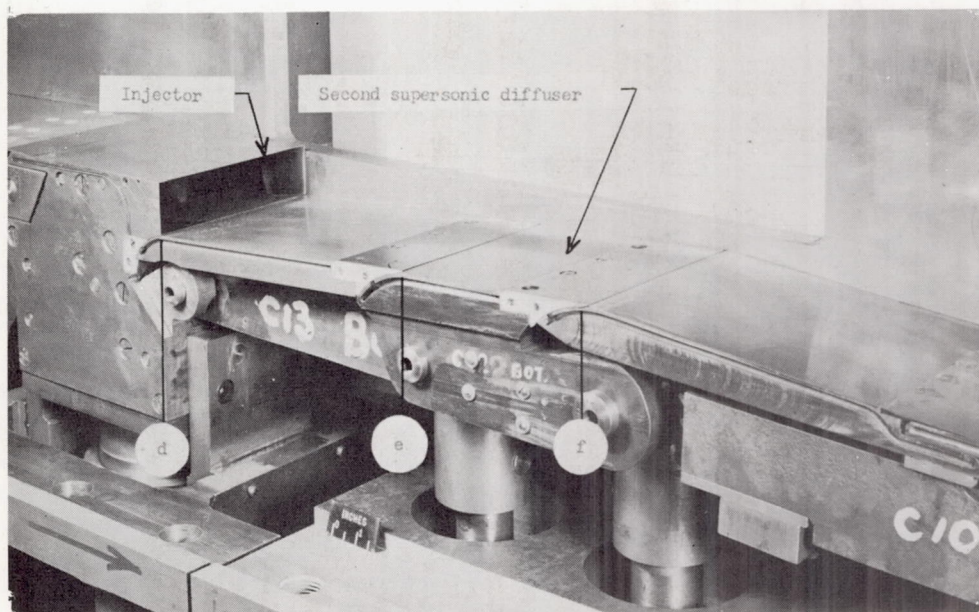


L-93182

Figure 3.- External view of variable-geometry diffuser apparatus.



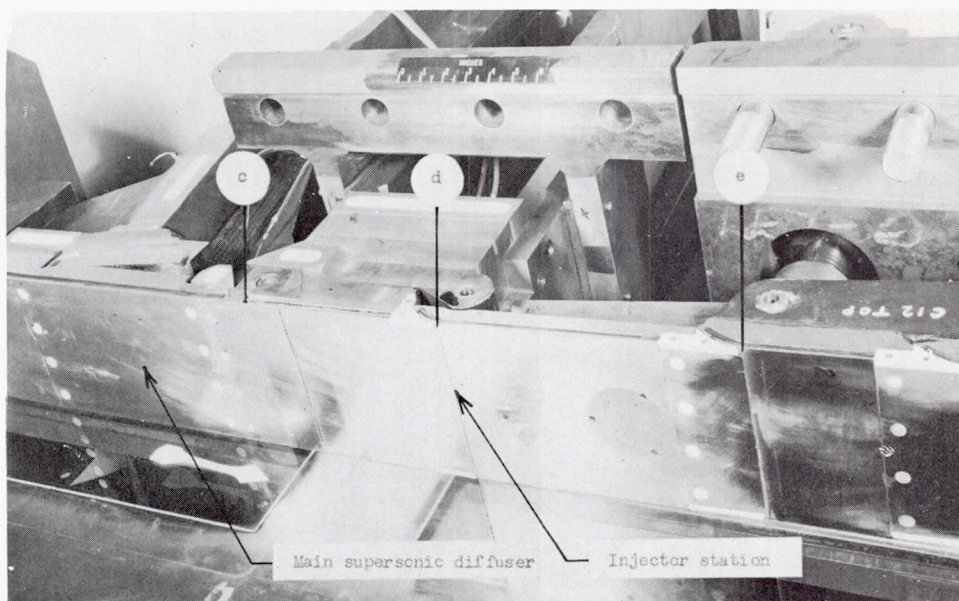
(a) Supersonic nozzle, main supersonic diffuser and injector sections of bottom wall.



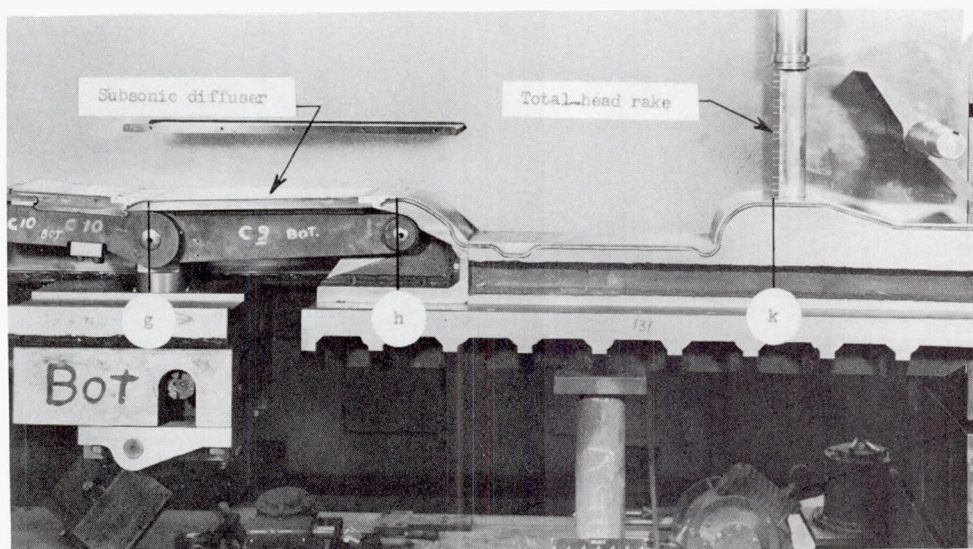
(b) Injector, mixing, and second supersonic diffuser sections of bottom wall.

L-57-2727

Figure 4.- Photographs of apparatus with one sidewall removed.



(c) Main supersonic diffuser and downstream sections of top wall of configuration with no injector.



(d) Subsonic diffuser and downstream end of apparatus. L-57-2728

Figure 4.- Concluded.

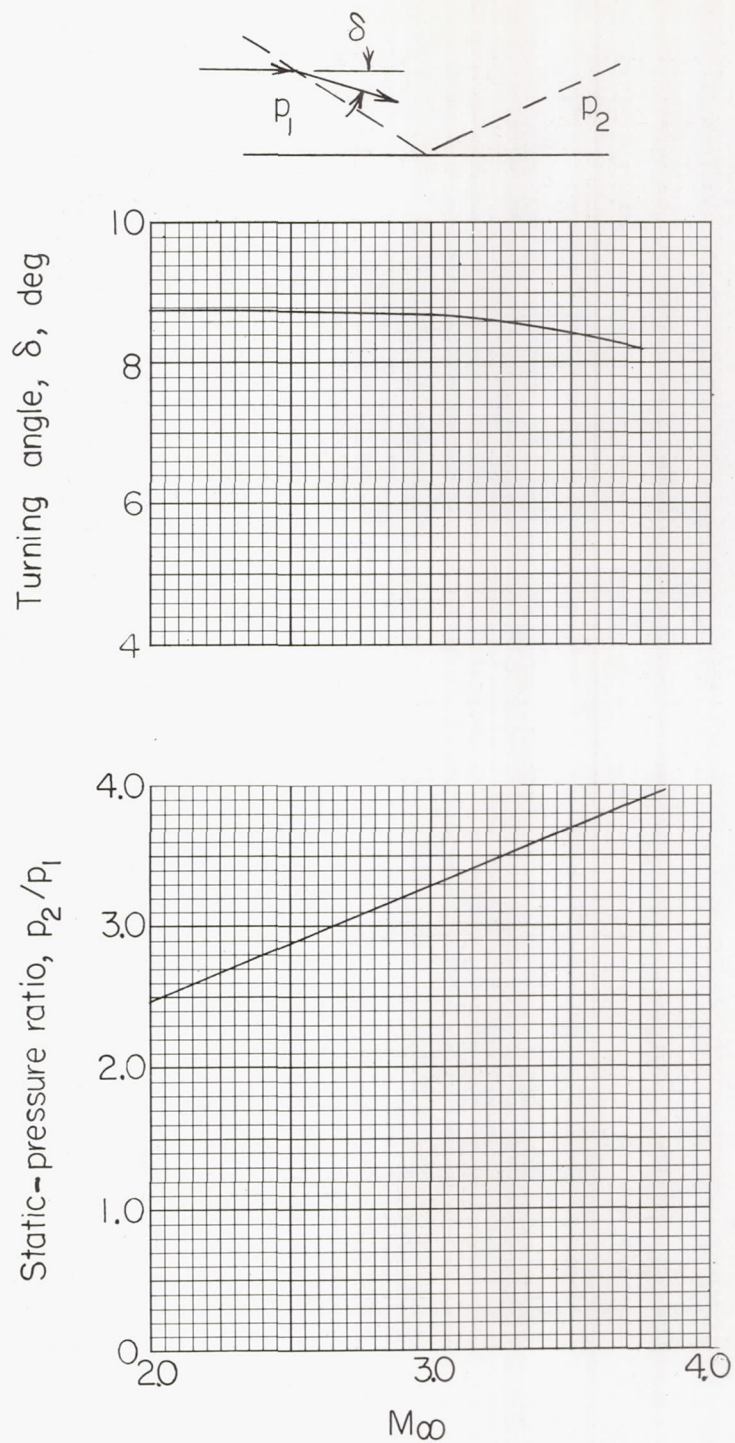


Figure 5.- Shock-wave characteristics which produce separation of a turbulent boundary layer on a flat plate (ref. 12).

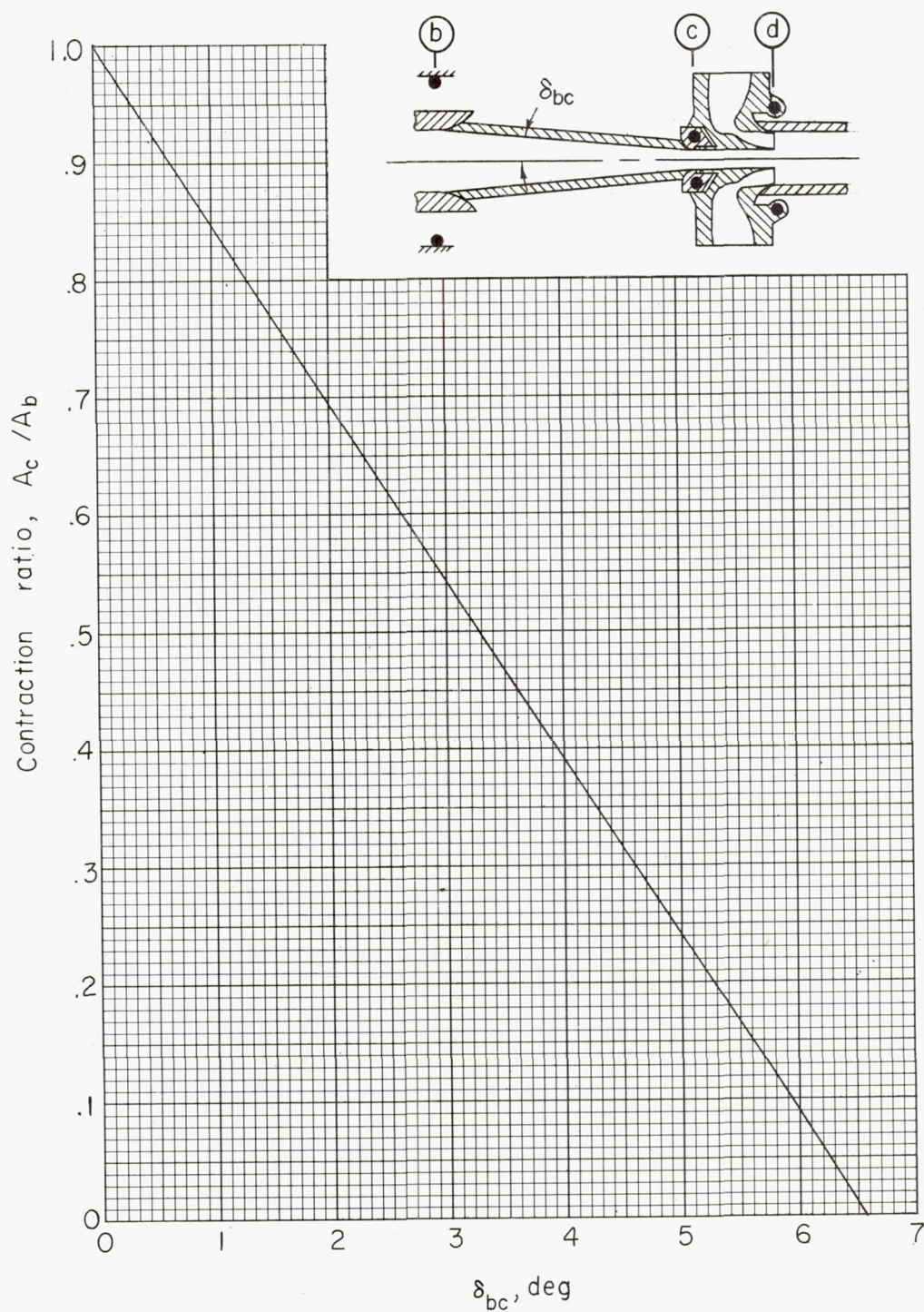


Figure 6.- Relationship between wall angle and contraction ratio of main supersonic diffuser.

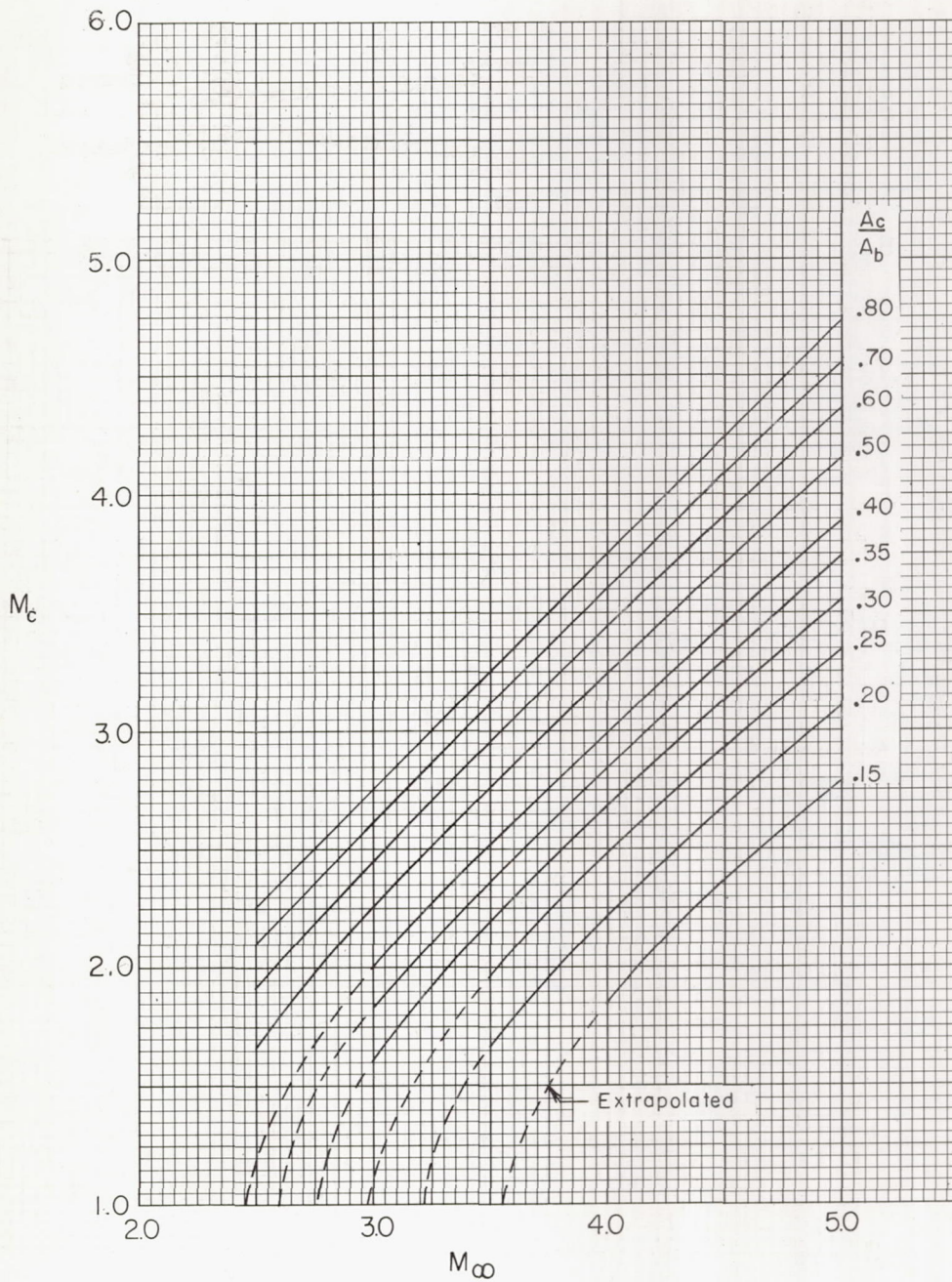


Figure 7.- Variation of computed Mach number at end of main supersonic diffuser with contraction ratio and test-section Mach number.

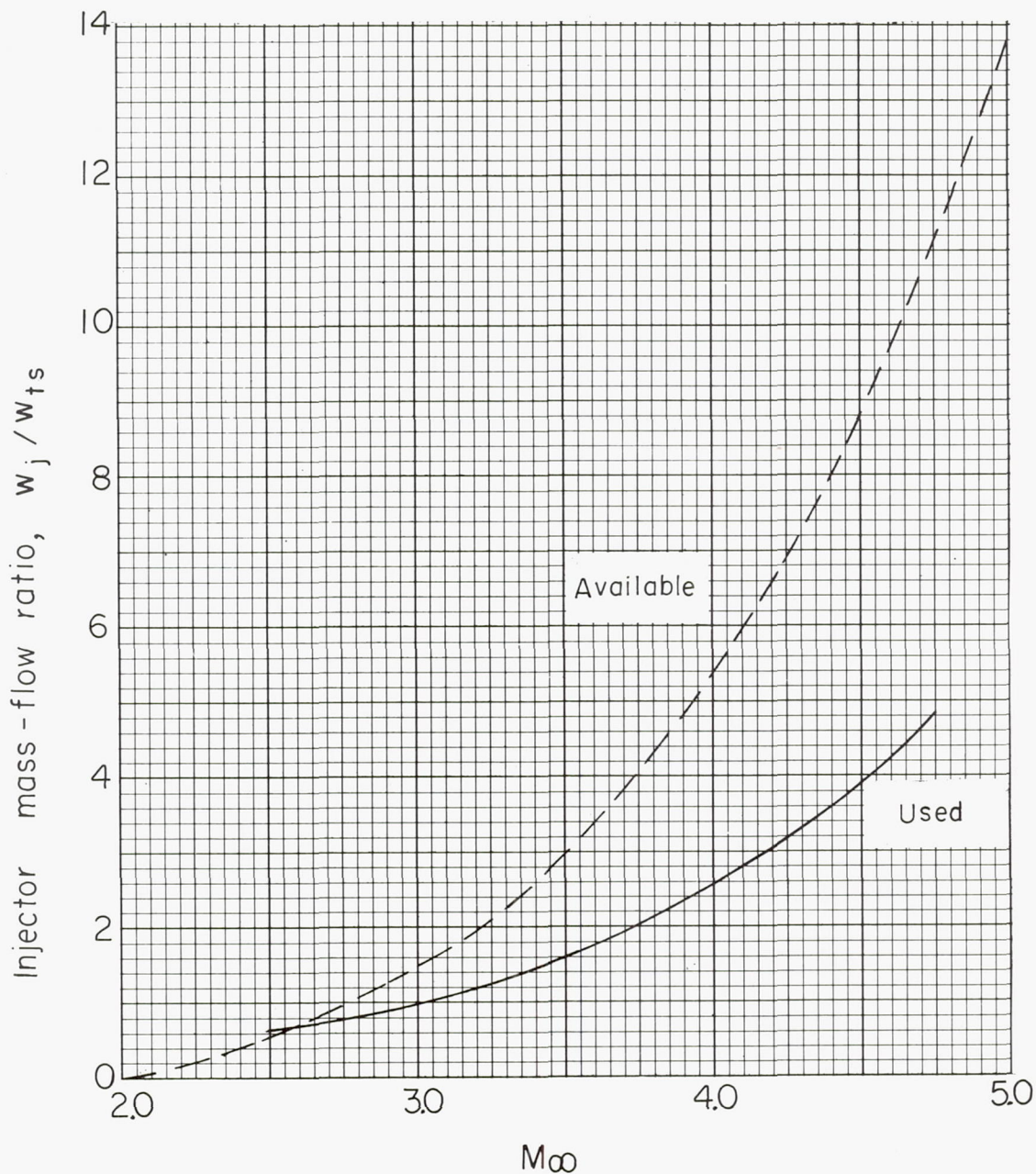


Figure 8.- Comparison of injector mass-flow ratio of diffuser apparatus and mass-flow ratio available for a tunnel assumed to have constant volume compressor and fixed-area test section above a Mach number of 2.0.

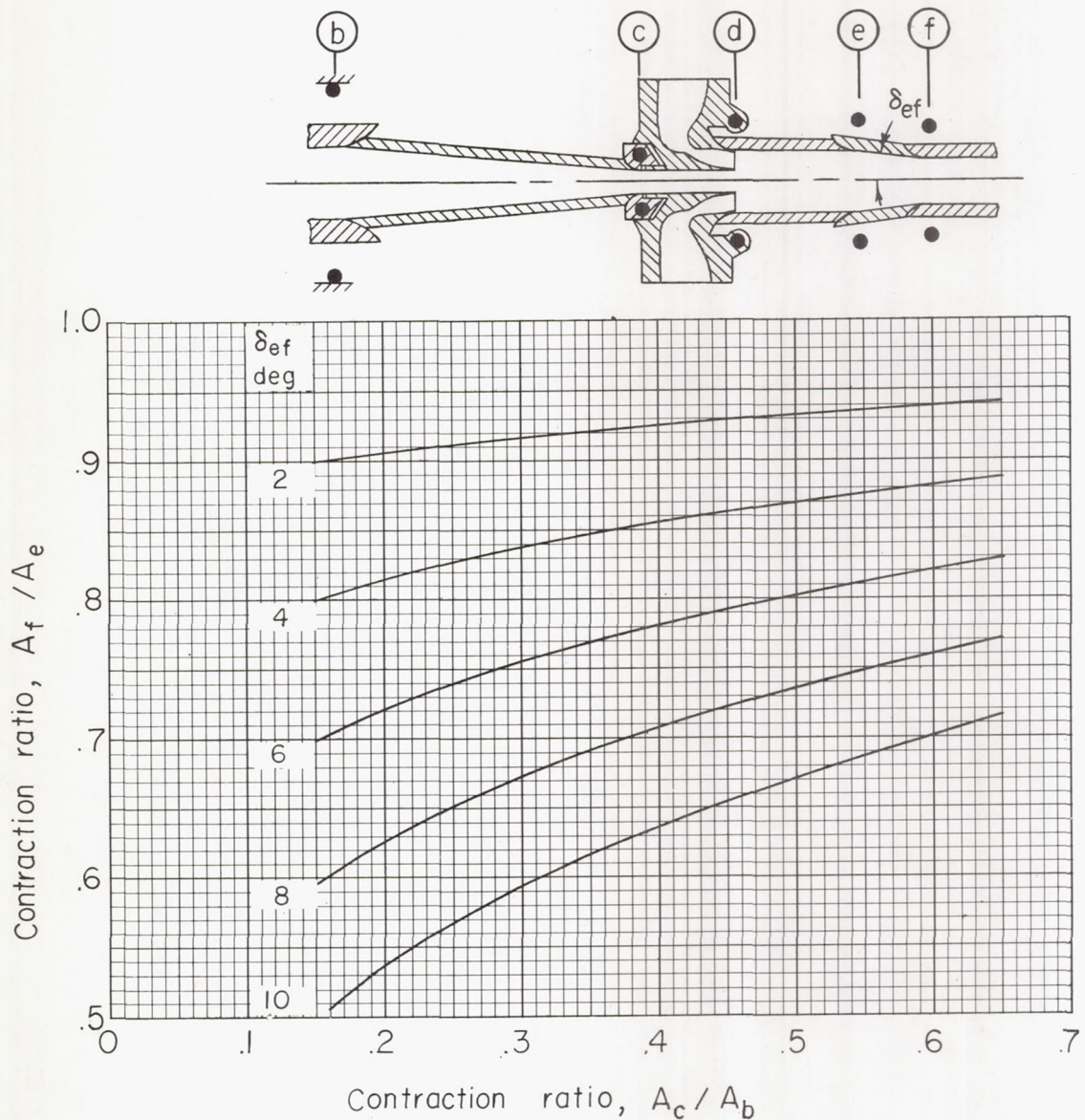


Figure 9.- Relationship between wall angle of second supersonic diffuser and contraction ratios of main and second supersonic diffusers.

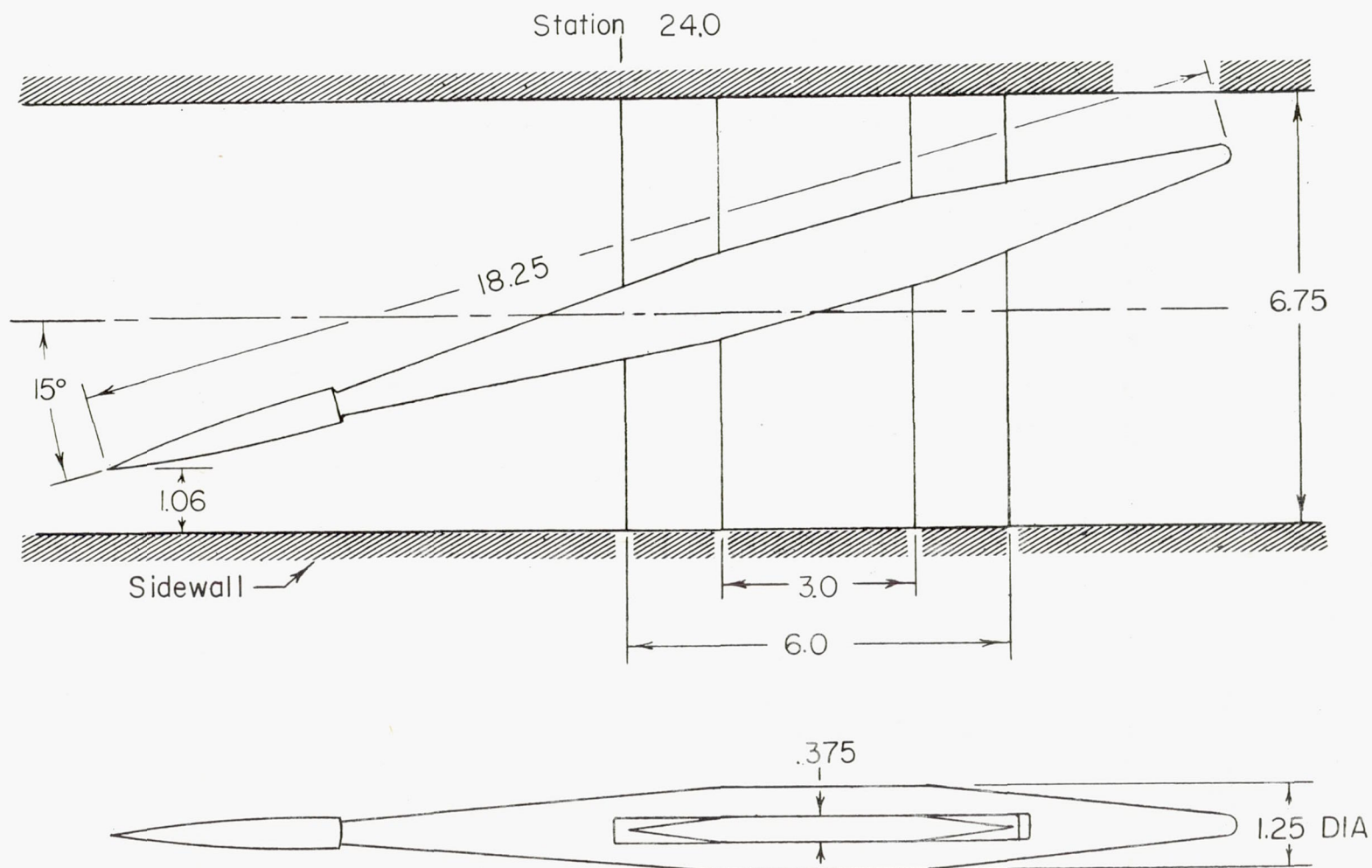


Figure 10.- Sketch of typical research model with its associated support system. All dimensions are in inches.

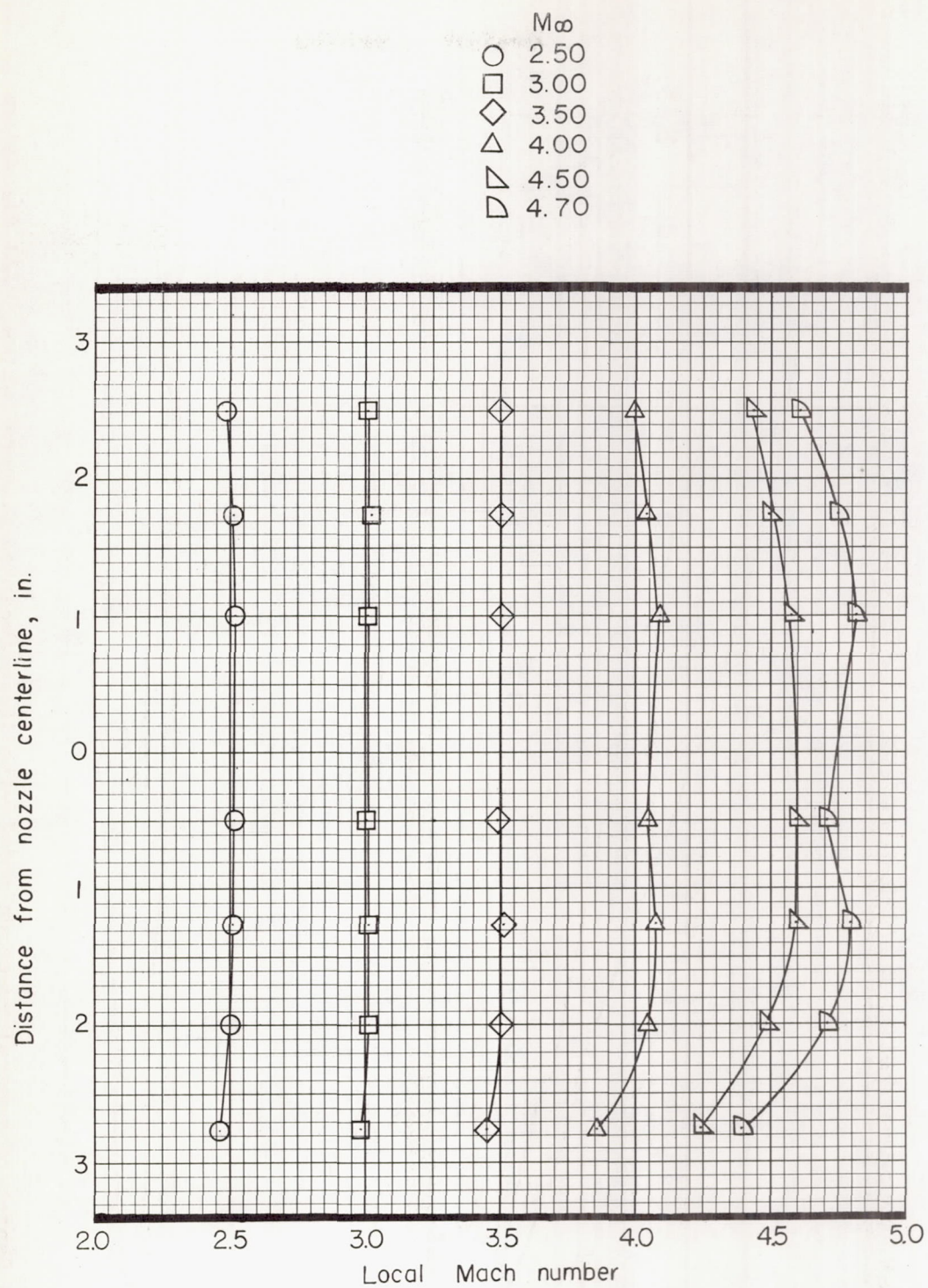
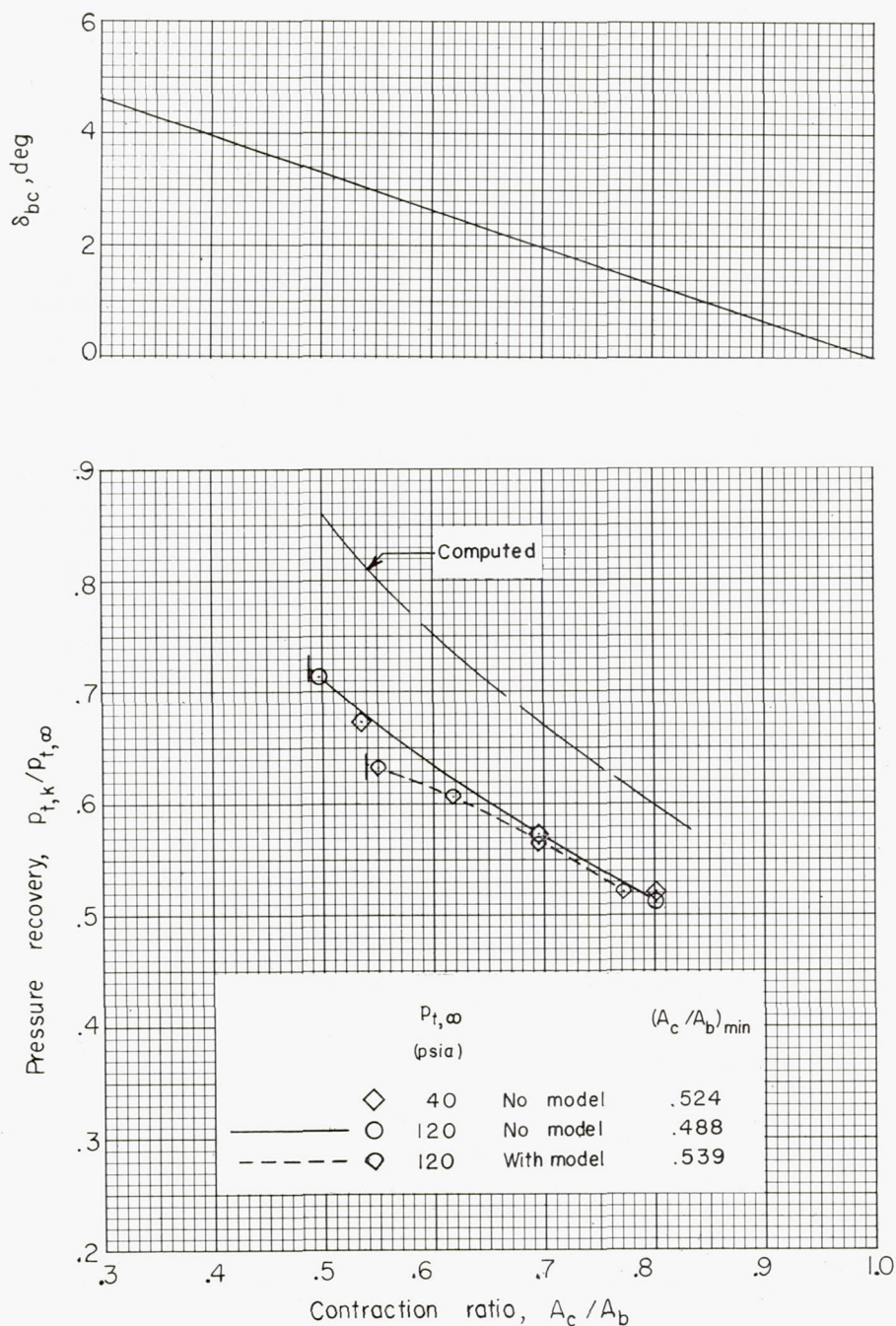
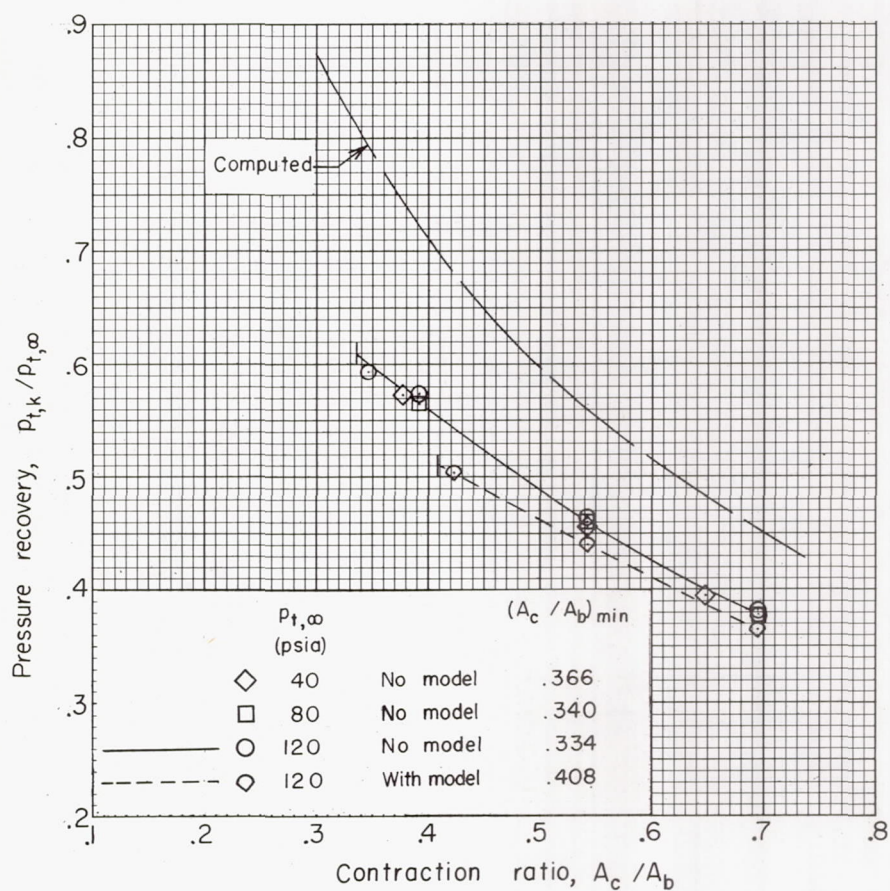
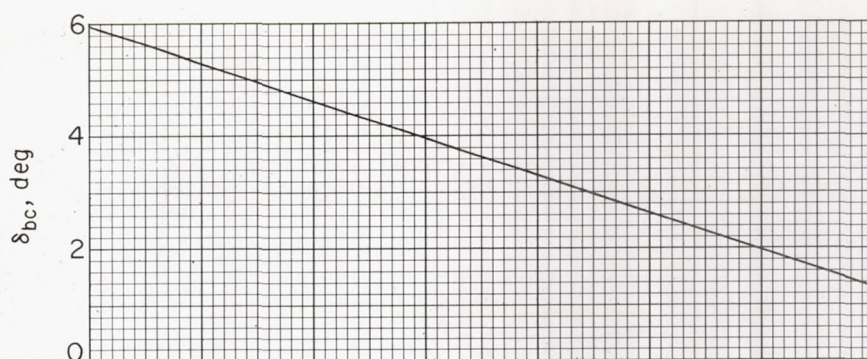


Figure 11.- Typical Mach number distributions at end of supersonic nozzle.



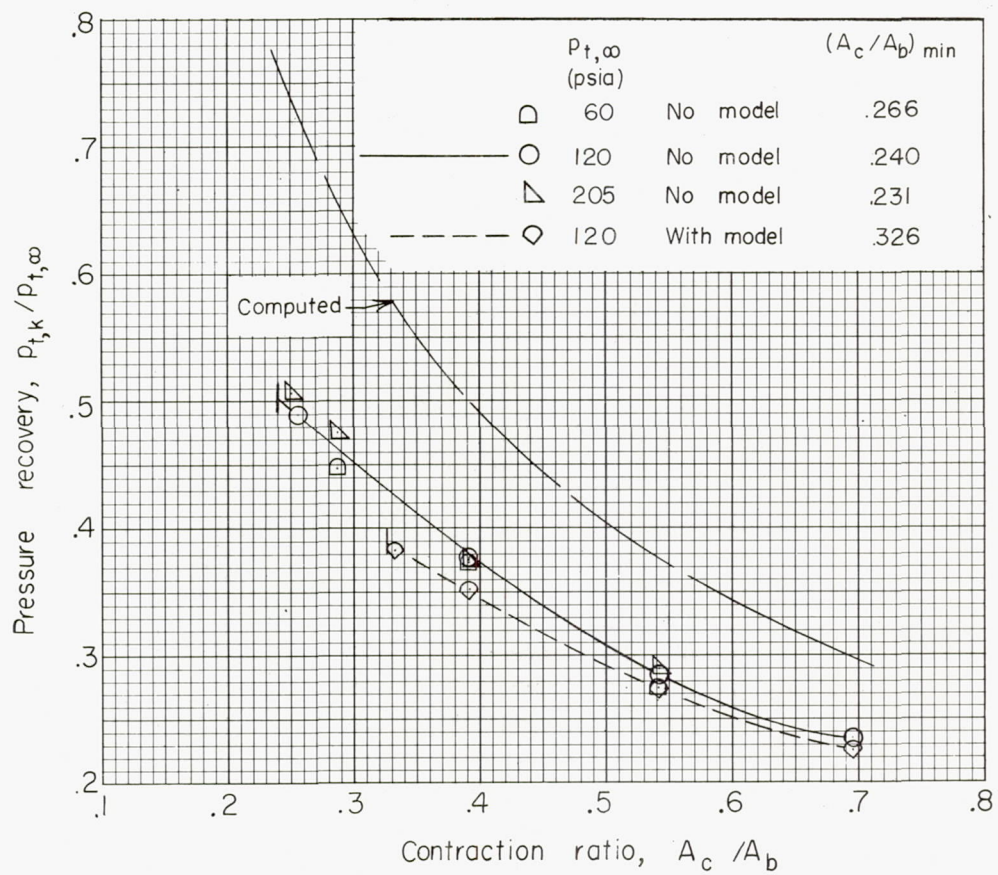
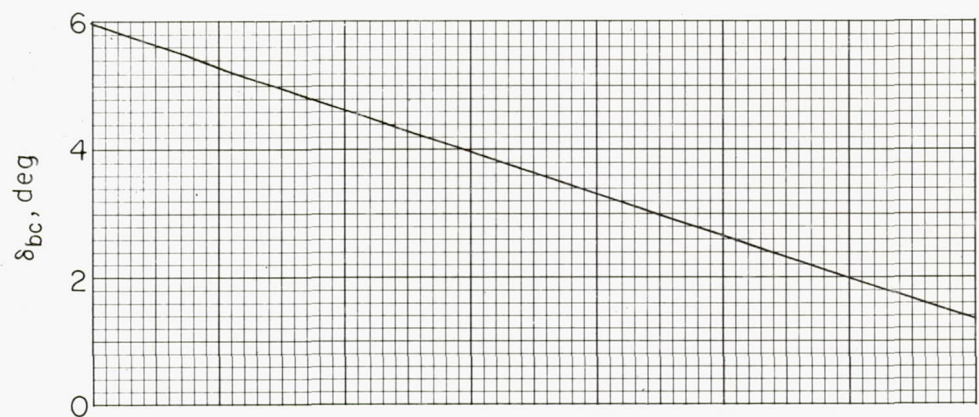
(a) $M_\infty = 2.50$.

Figure 12.- Effect of main-supersonic-diffuser contraction ratio on pressure-recovery characteristics of diffuser apparatus without injector.



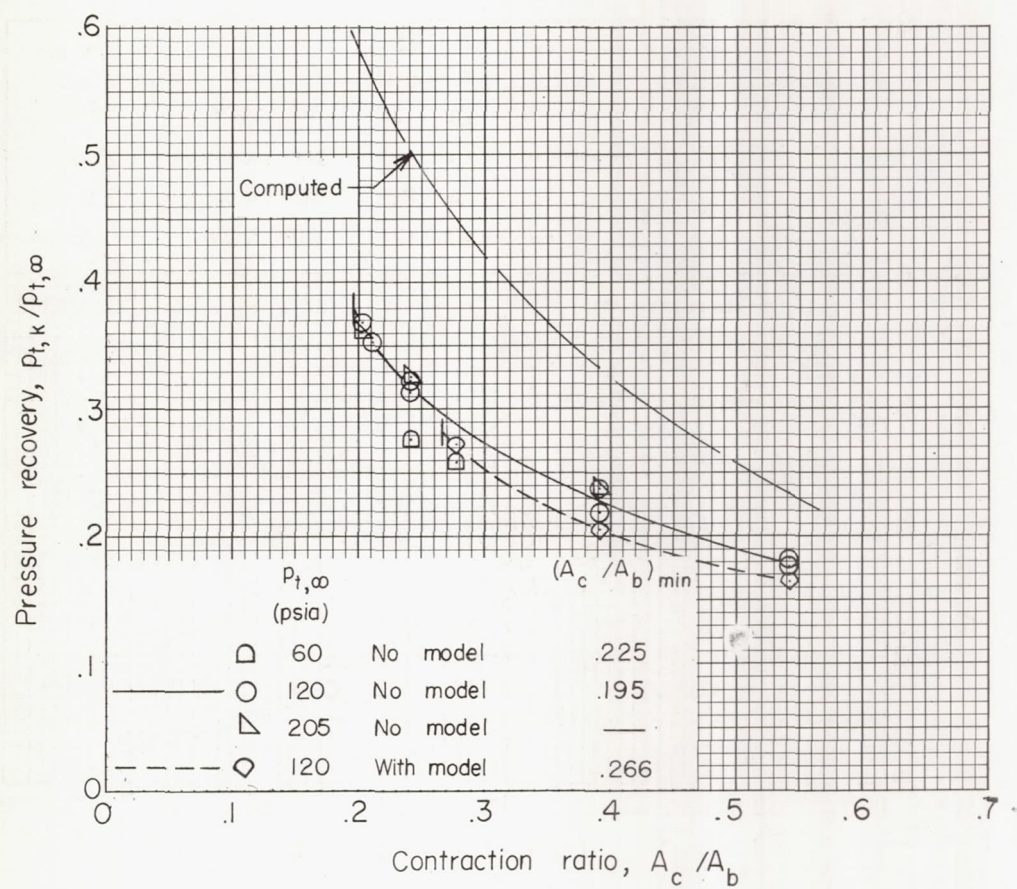
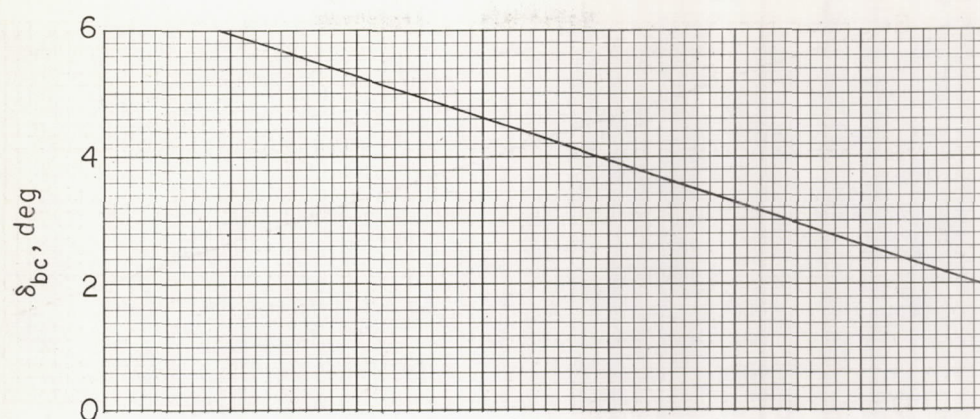
(b) $M_\infty = 3.00$.

Figure 12.- Continued.



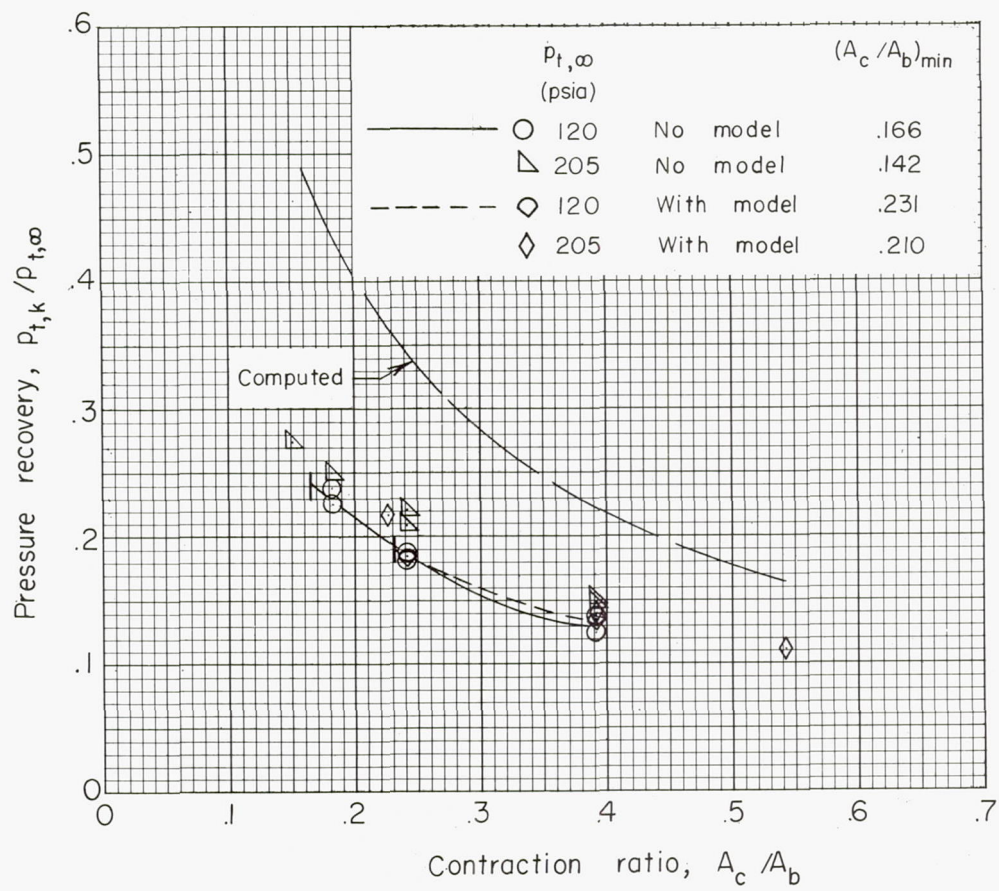
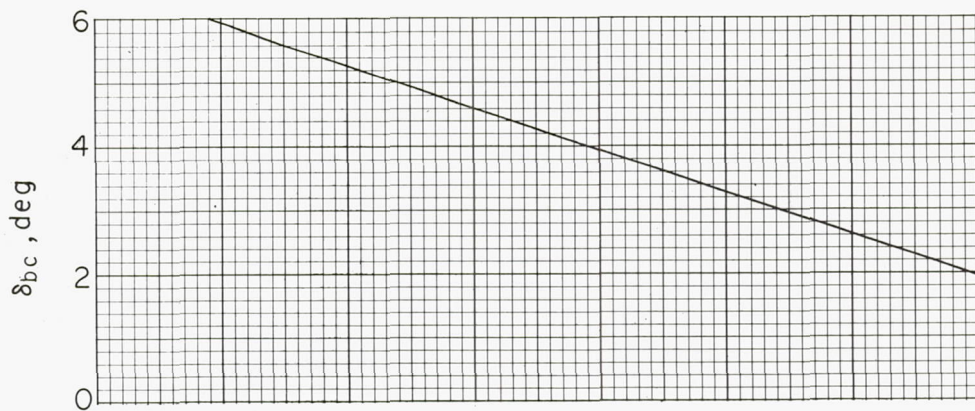
(c) $M_\infty = 3.50$.

Figure 12.- Continued.



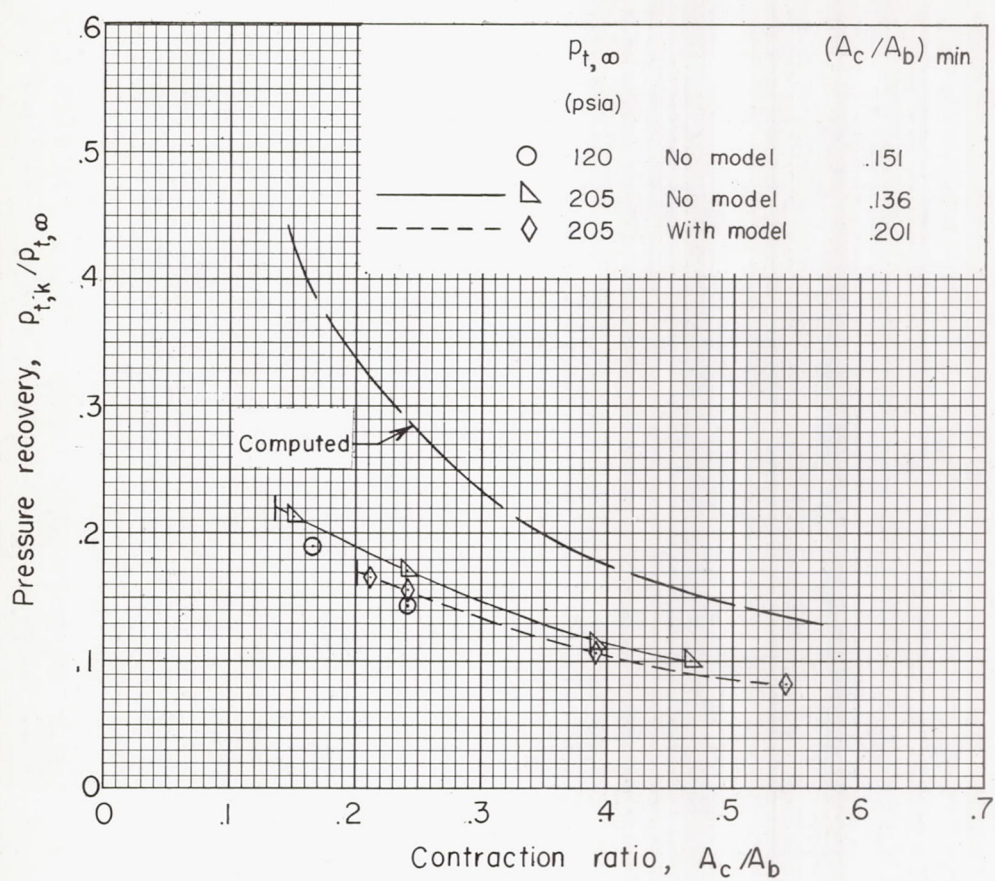
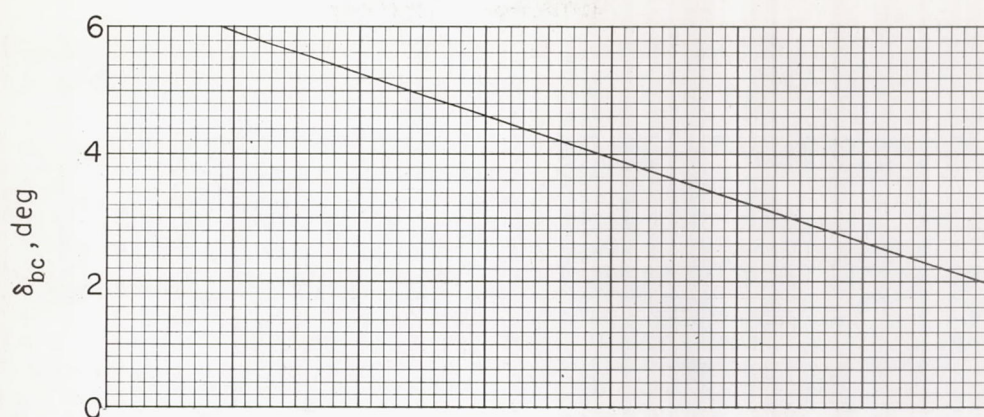
(d) $M_\infty = 4.00$.

Figure 12.- Continued.



(e) $M_\infty = 4.50$.

Figure 12.- Continued.



(f) $M_\infty = 4.75$.

Figure 12.- Concluded.

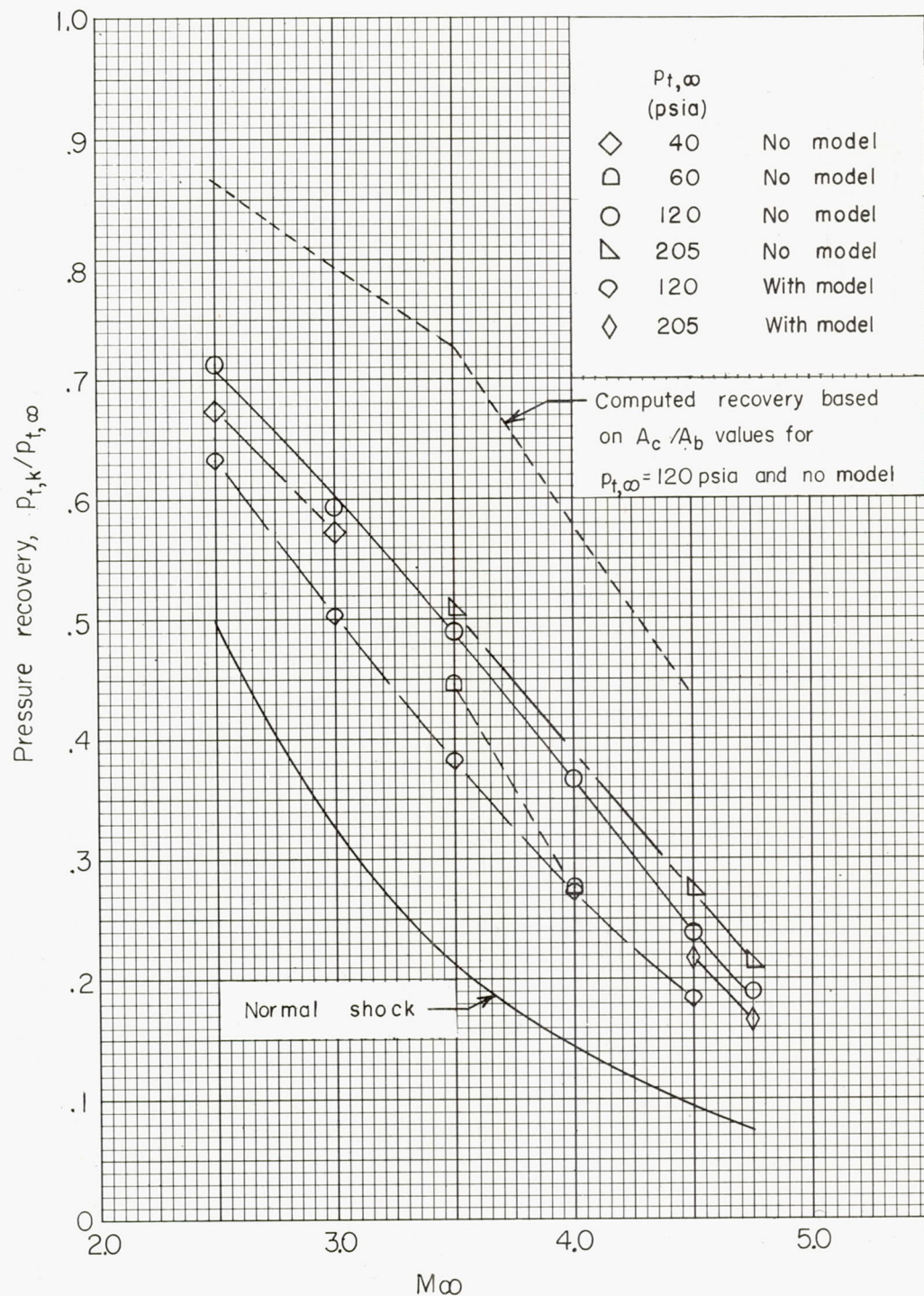


Figure 13.- Variation with Mach number of maximum measured pressure recovery of diffuser apparatus without injector.

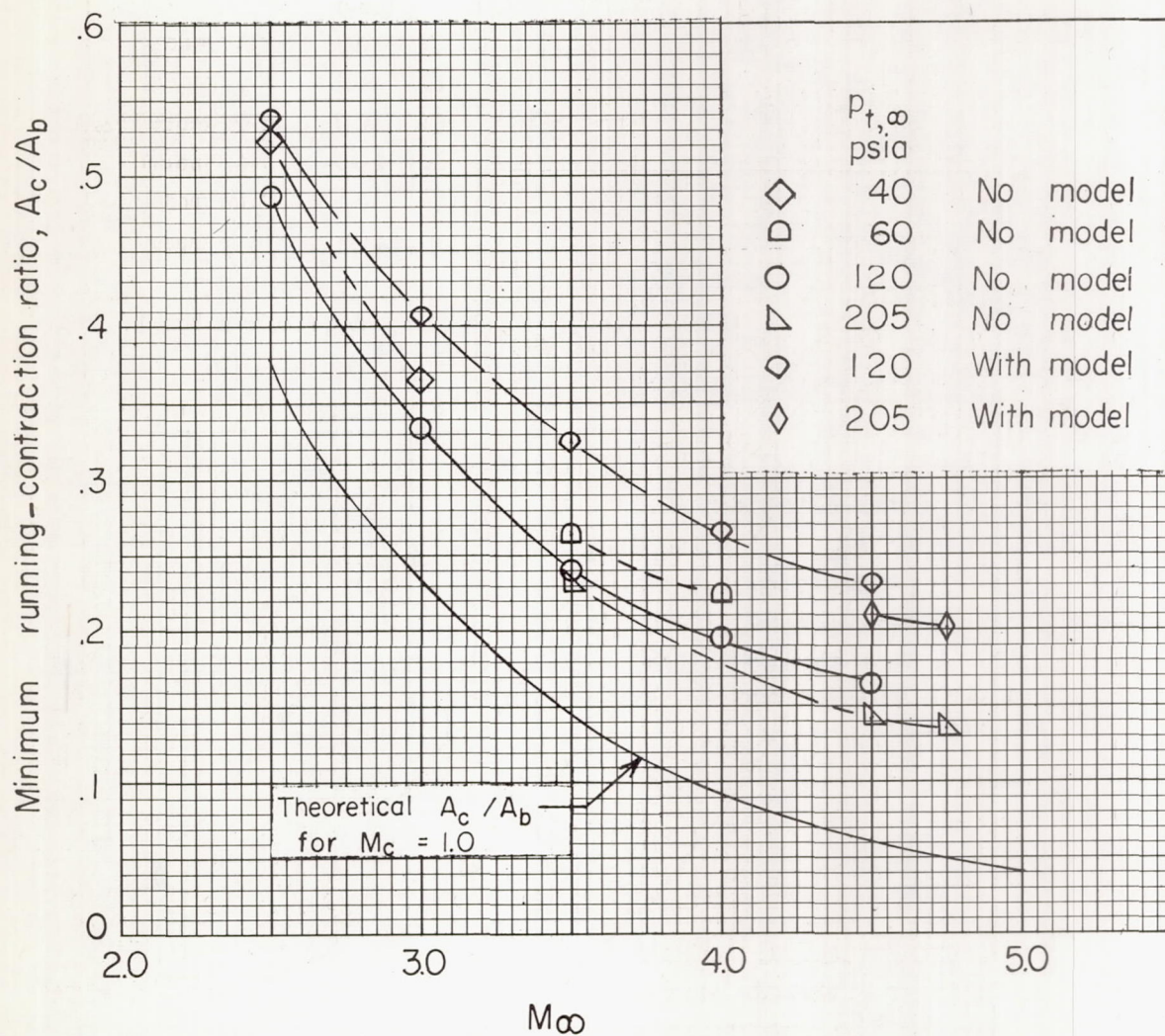


Figure 14.- Variation with Mach number of minimum running contraction ratio of main supersonic diffuser.

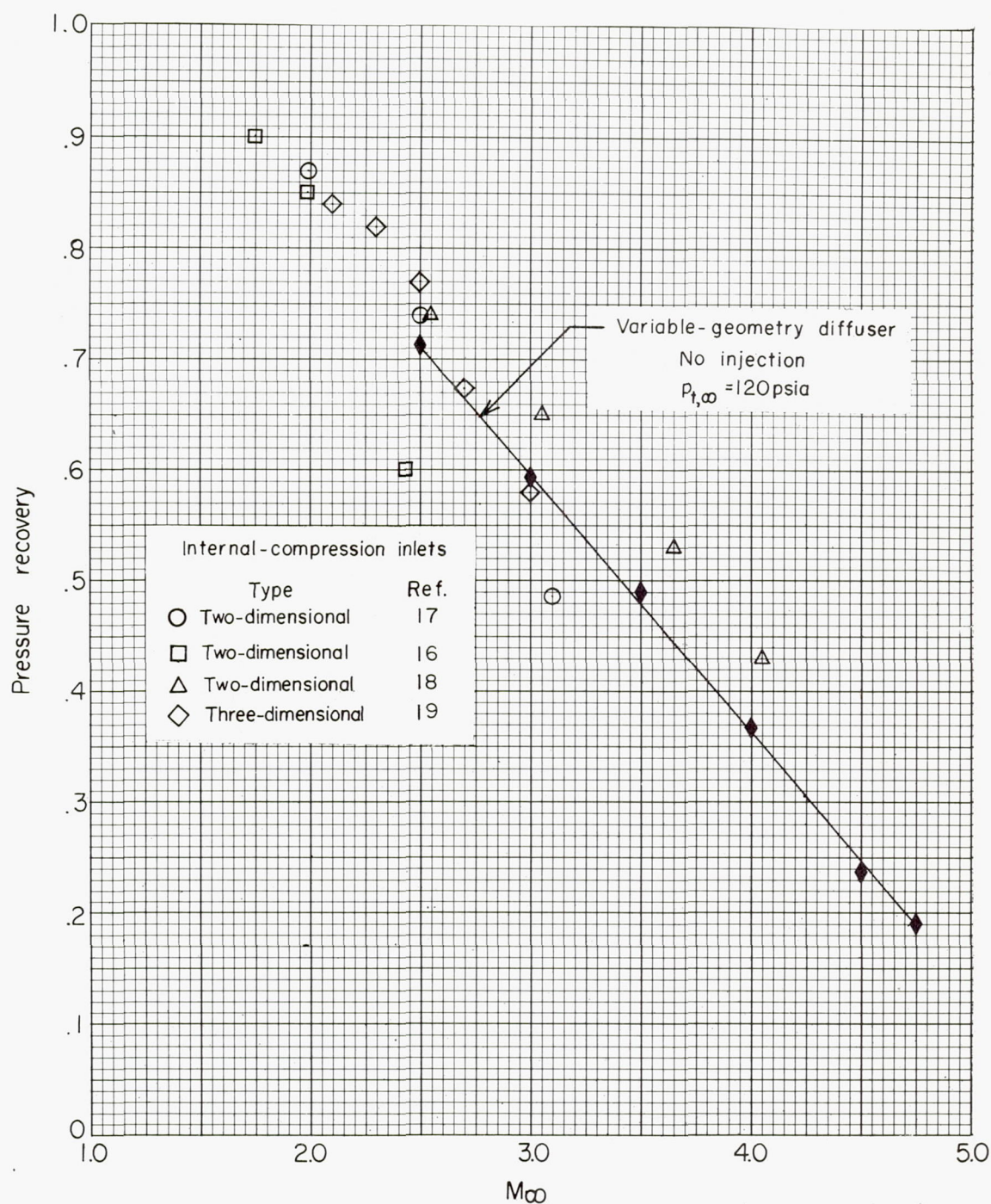


Figure 15.- Comparison of pressure-recovery characteristics of diffuser apparatus and several internal-compression supersonic inlets with no boundary-layer control.

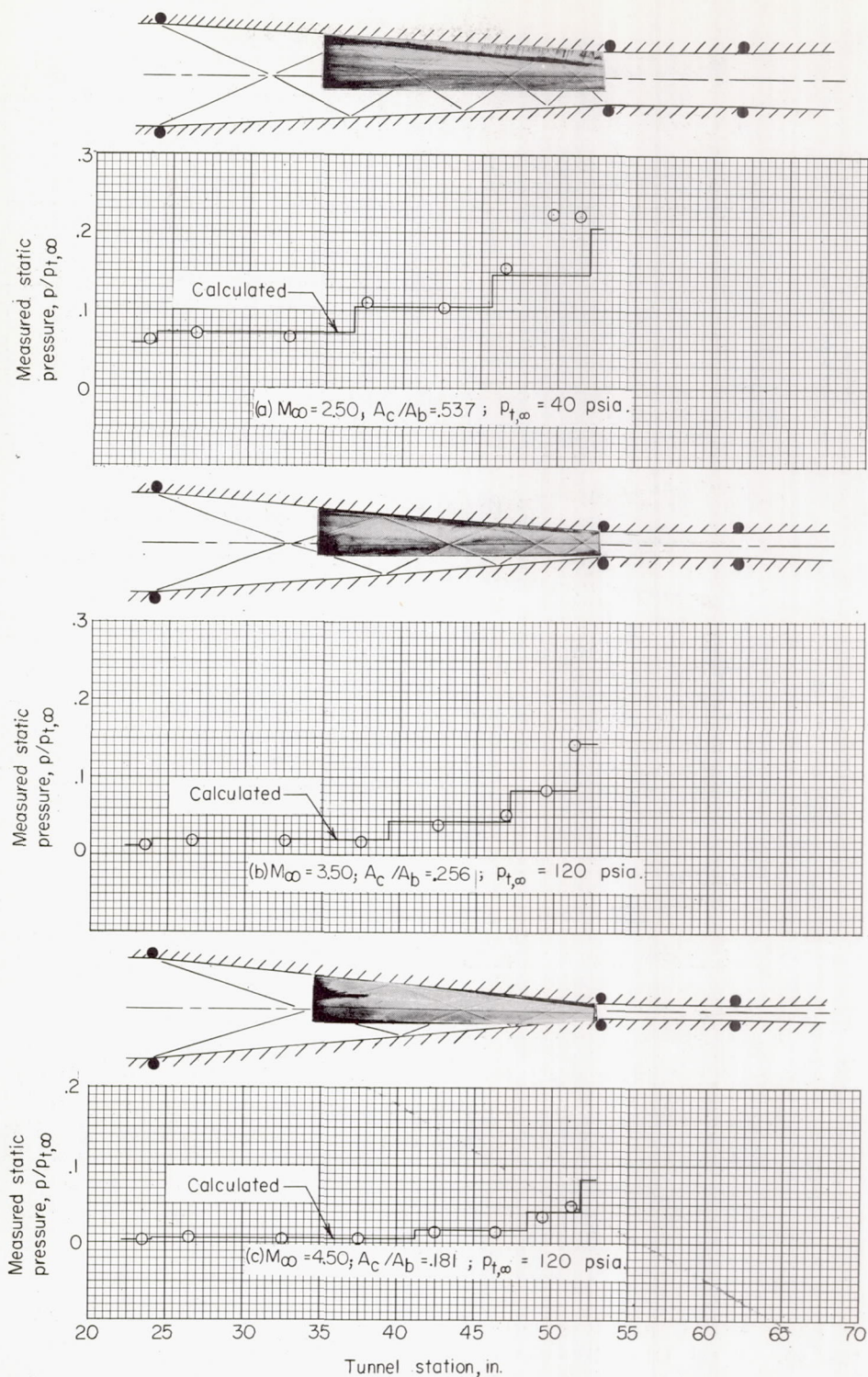


Figure 16.- Typical static pressure distributions on moveable wall and flow patterns in main supersonic diffuser at near minimum values of minimum-running contraction ratio. No model.

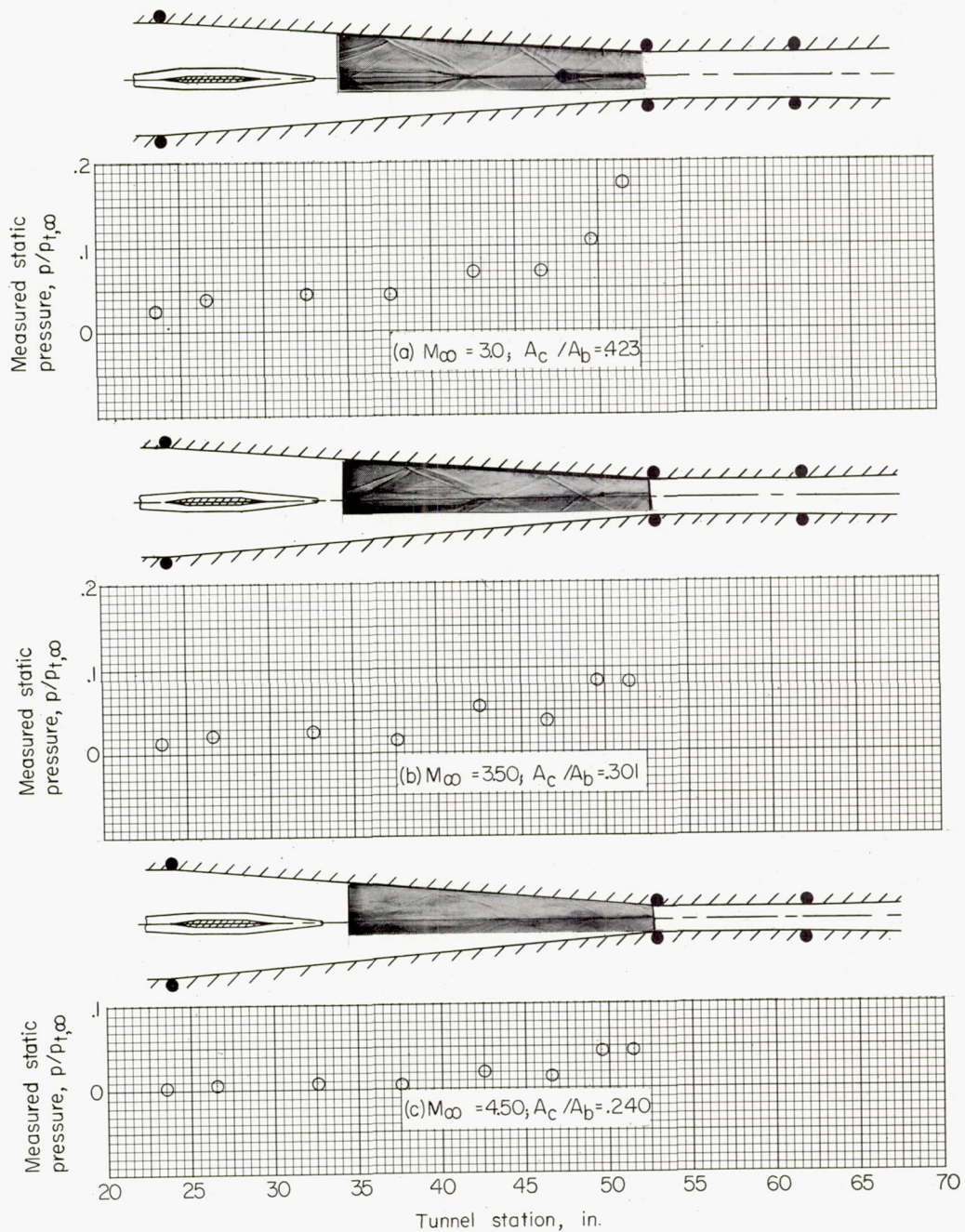


Figure 17.- Typical static pressure distributions on moveable wall and flow patterns in supersonic diffuser at near minimum values of minimum-running contraction ratio. With model; $p_{t,\infty} = 120$ pounds per square inch absolute.

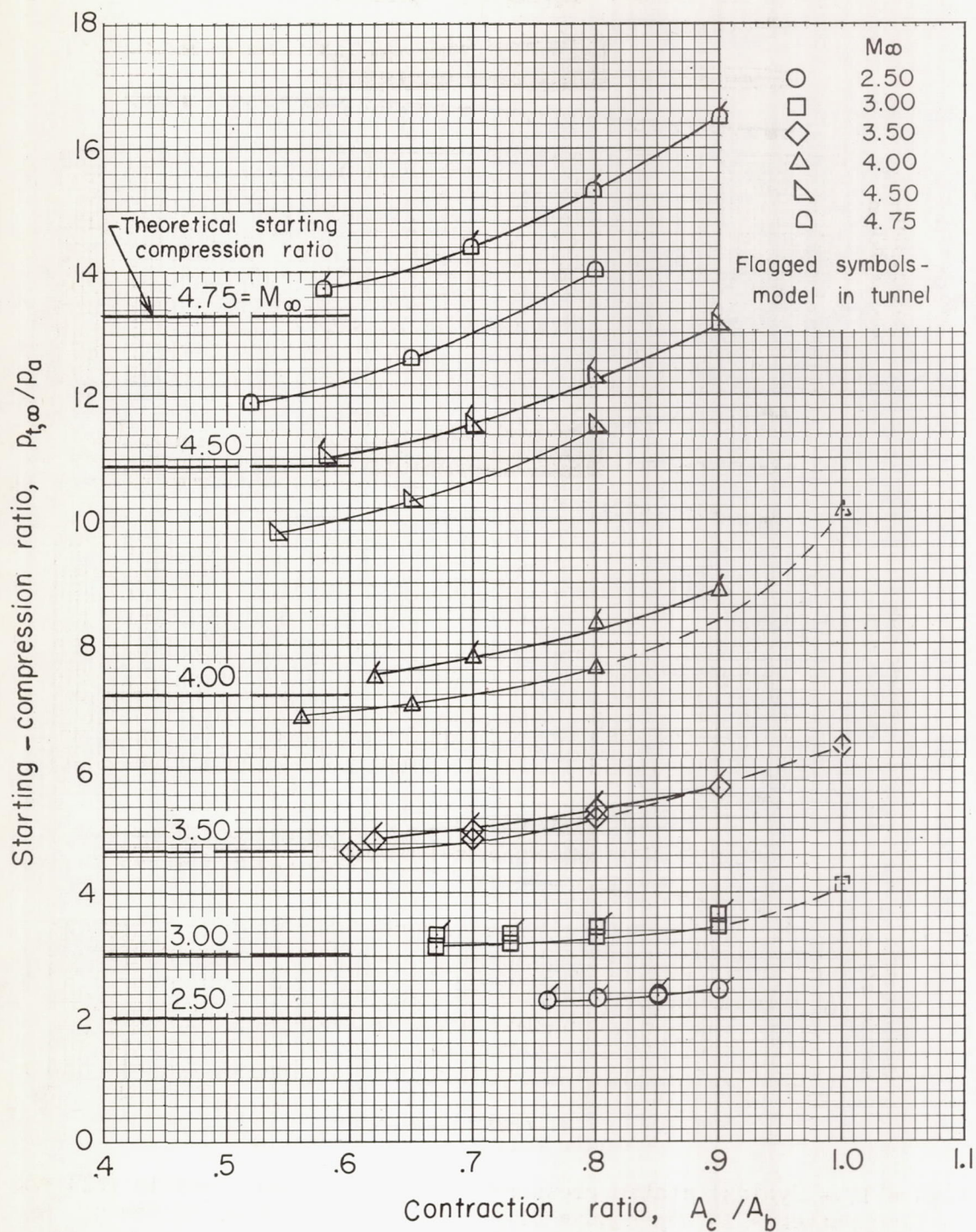


Figure 18.- Starting compression-ratio characteristics of diffuser apparatus without injector.

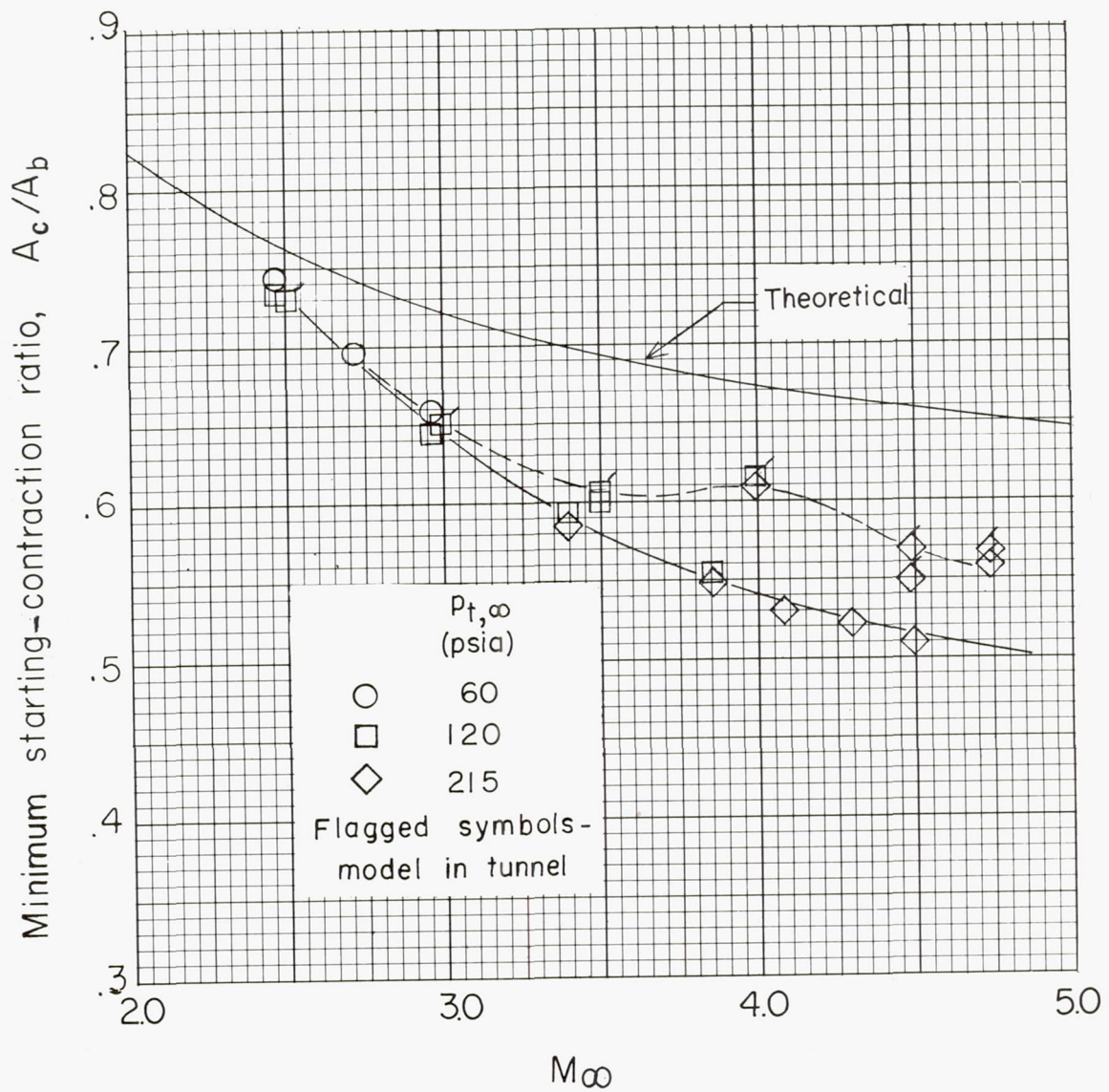


Figure 19.- Minimum starting-contraction ratio characteristics of diffuser apparatus without injector.

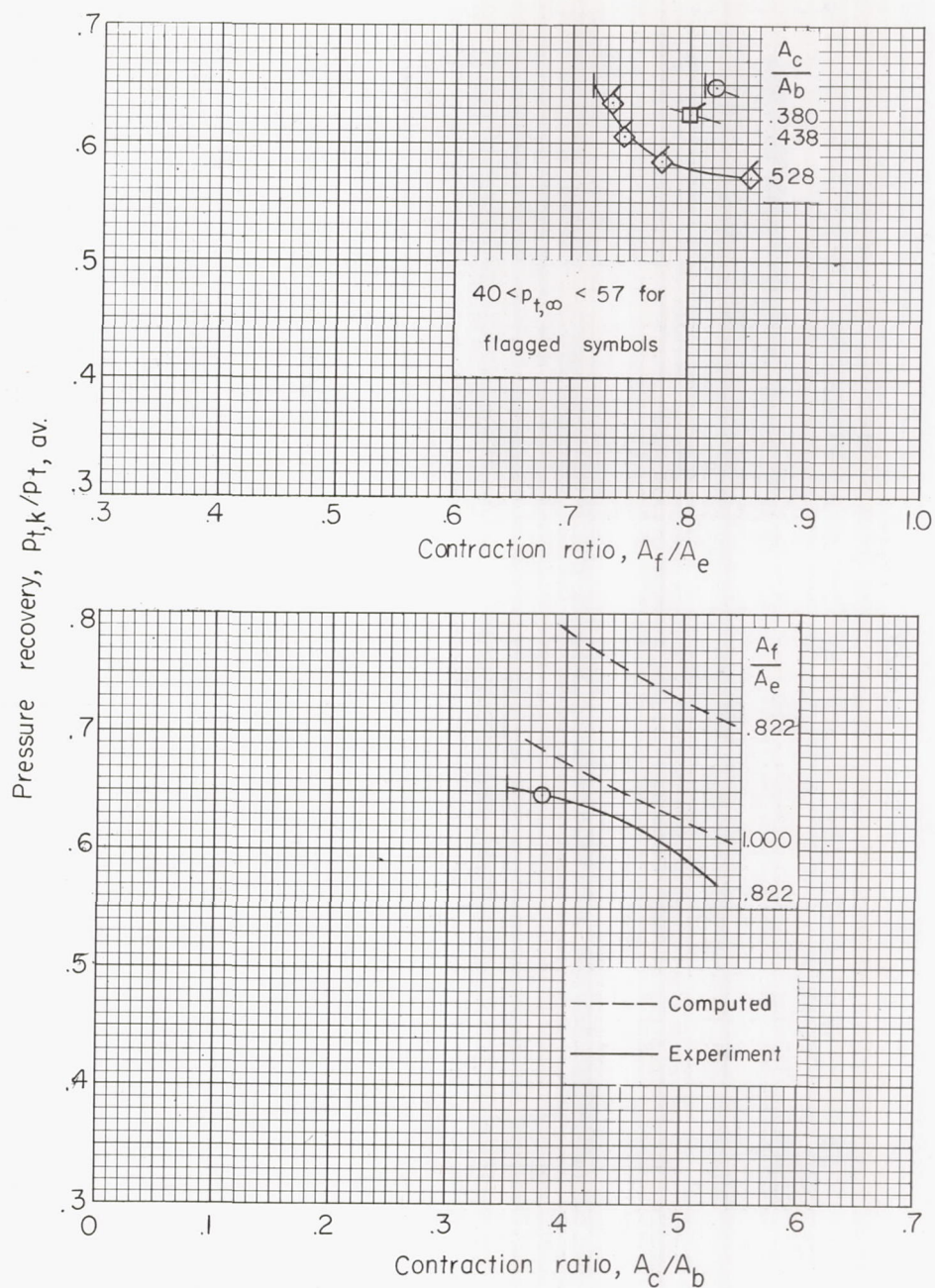
(a) $M_\infty = 2.95$.

Figure 20.- Pressure-recovery characteristics of diffuser apparatus with injector and without model in tunnel. $P_{t,\infty} = 120$ pounds per square inch absolute unless noted.

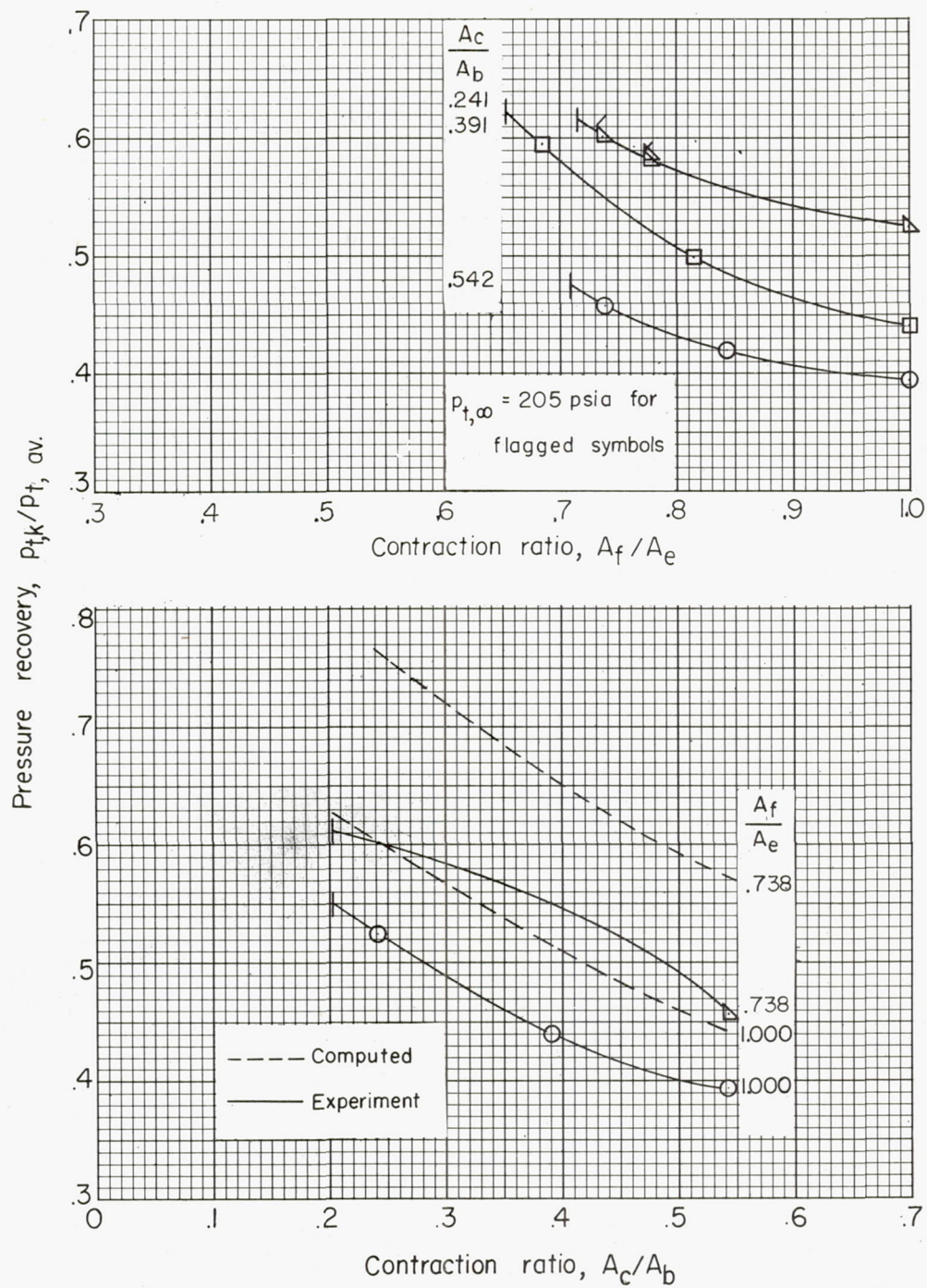
(b) $M_{\infty} = 3.90$.

Figure 20.- Continued.

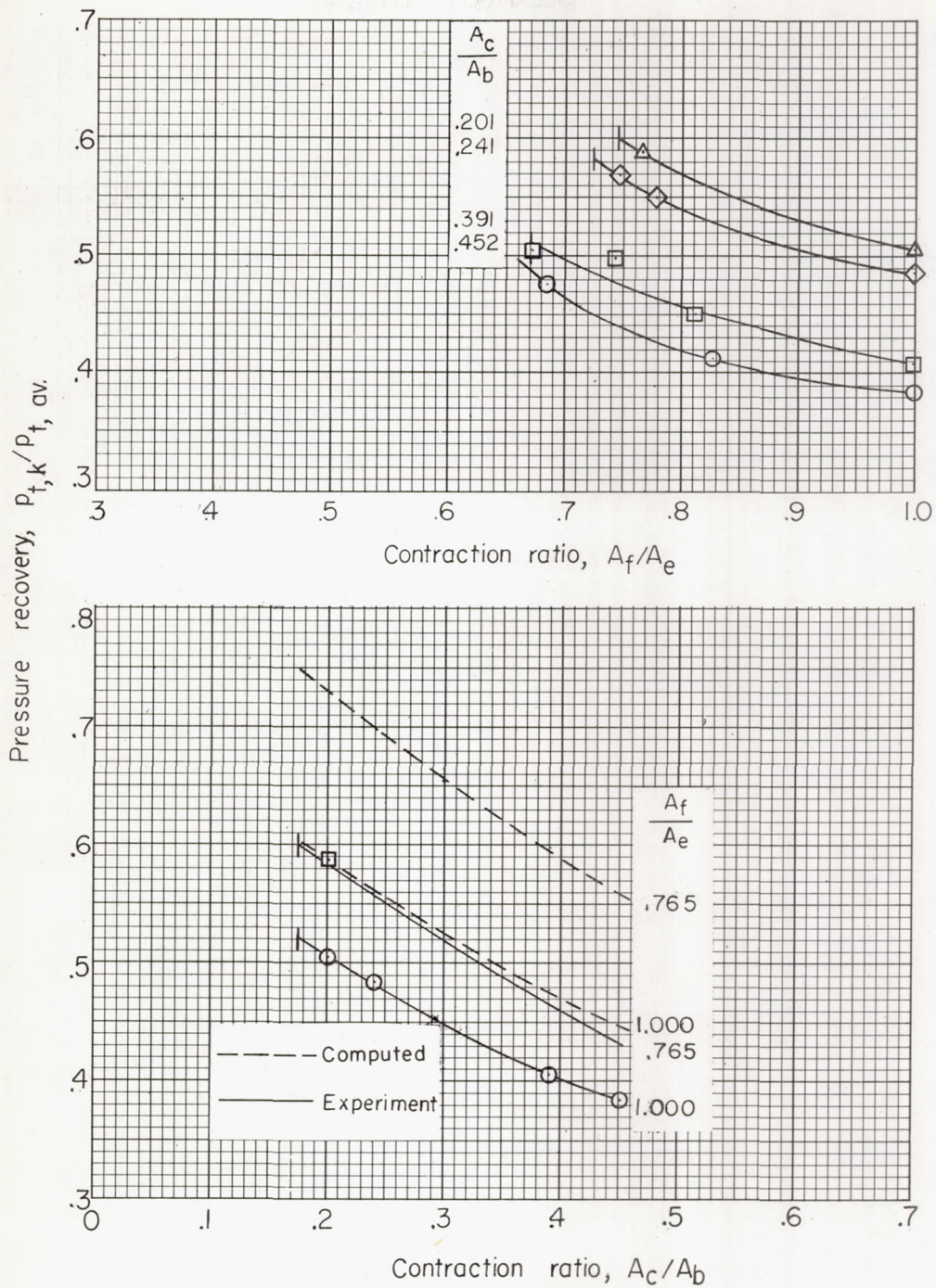
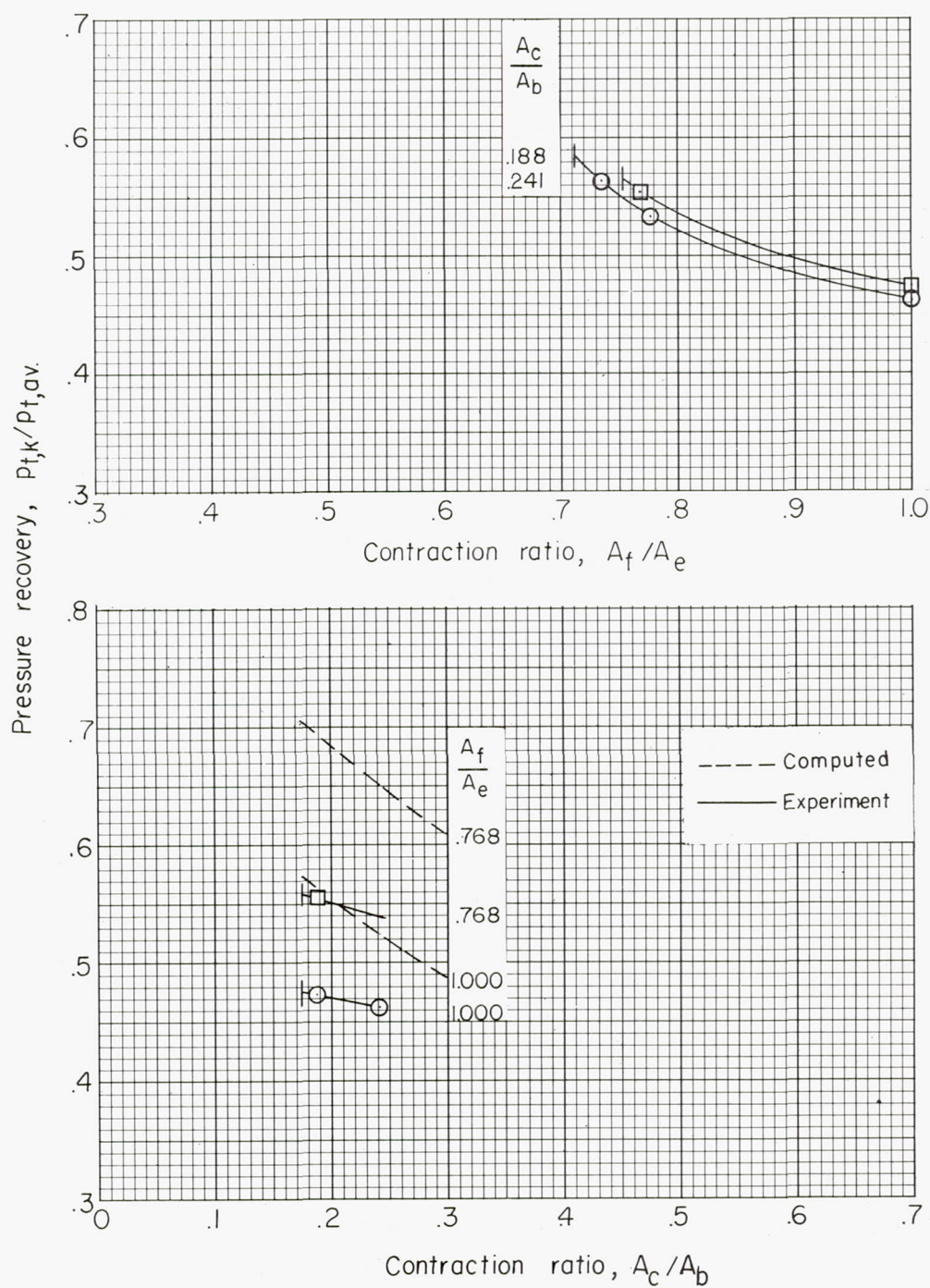
(c) $M_\infty = 4.30$.

Figure 20.- Continued.



(d) $M_\infty = 4.70$.

Figure 20.- Concluded.

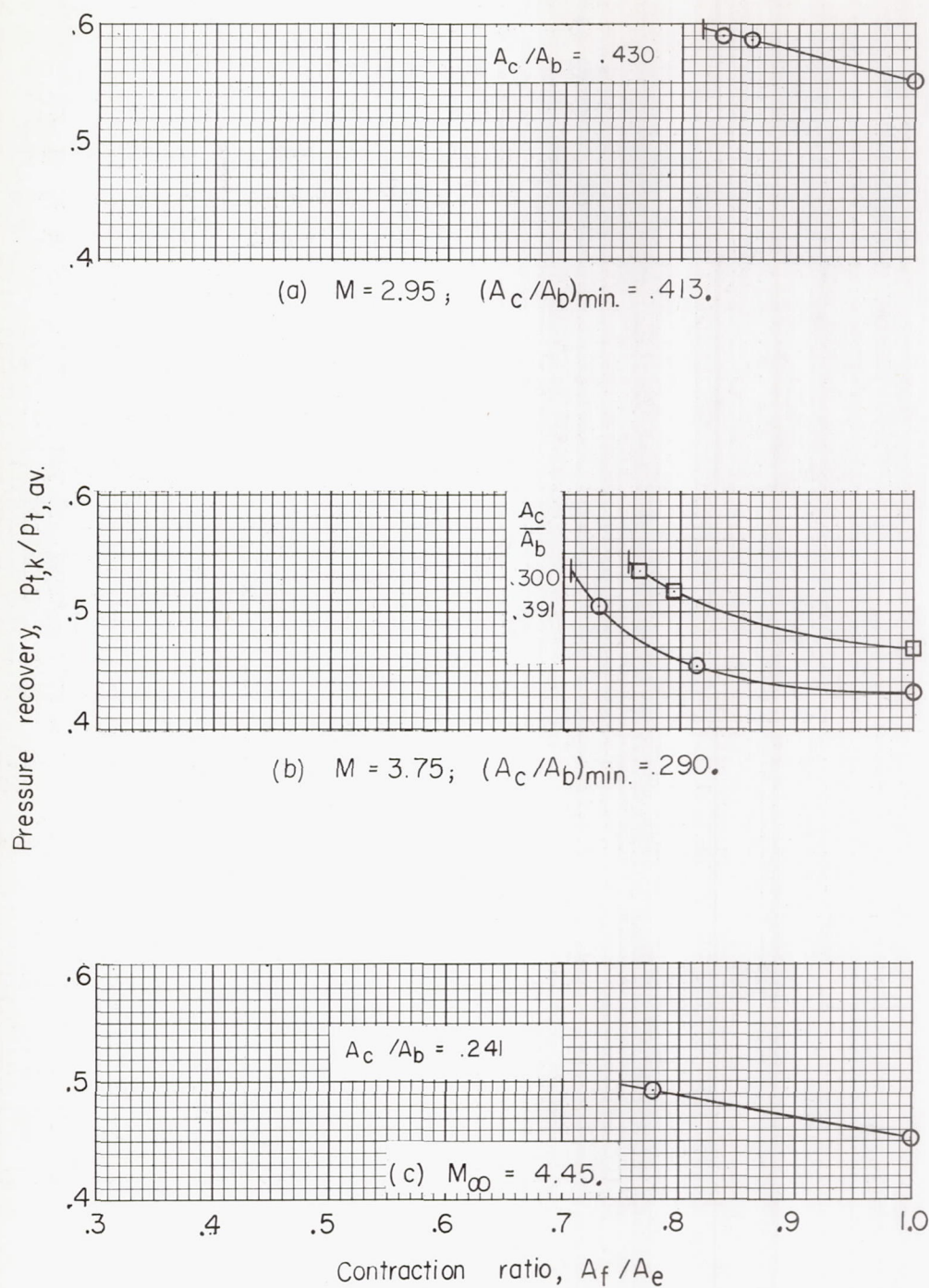


Figure 21.- Pressure-recovery characteristics of diffuser apparatus with injector and with model in tunnel. $p_{t,\infty} = 120$ pounds per square inch absolute.

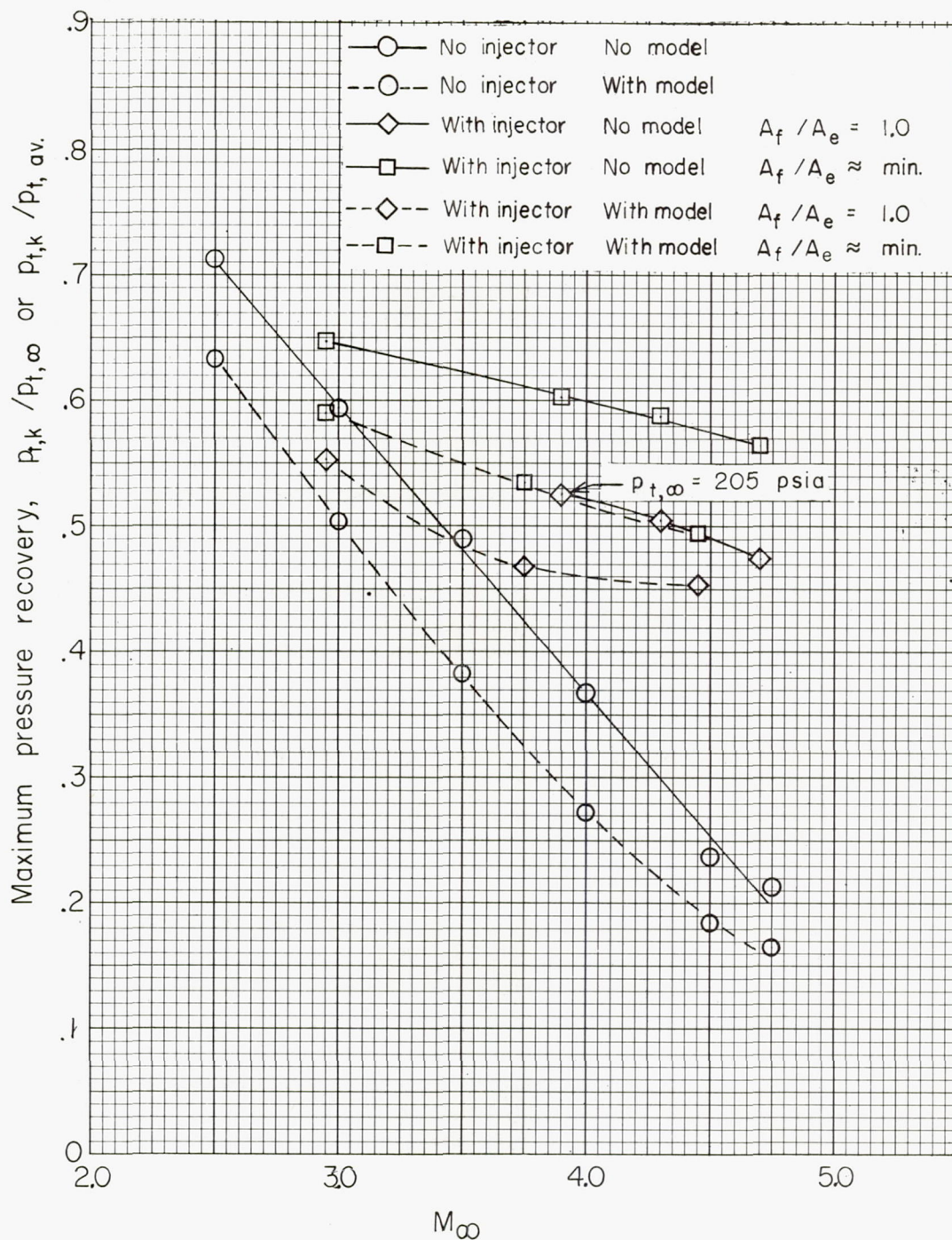
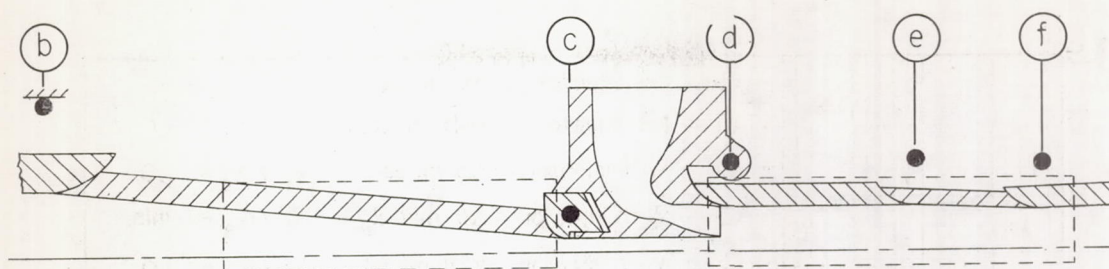


Figure 22.- Comparison of maximum pressure-recovery characteristics of diffuser apparatus with and without injector. $p_{t,\infty} = 120$ pounds per square inch absolute unless noted.



(a) $M_\infty = 2.95$; $A_c/A_b = .379$; $A_f/A_e = .822$.



(b) $M_\infty = 3.90$; $A_c/A_b = .240$; $A_f/A_e = .772$.



(c) $M_\infty = 4.30$; $A_c/A_e = .202$; $A_f/A_e = .765$.



(d) $M_\infty = 4.70$; $A_c/A_b = .187$; $A_f/A_e = .760$.



(e) $M_\infty = 4.70$; $A_c/A_b = .240$; $A_f/A_e = .734$.



(f) $M_\infty = 4.70$; $A_c/A_b = .259$; $A_f/A_e = .970$.



(g) $M_\infty = 4.70$; $A_c/A_b = .270$; $A_f/A_e = .970$.

Figure 23.- Typical shadowgraph of flow in diffuser apparatus with injector installed. No model in airstream.

L-57-2729

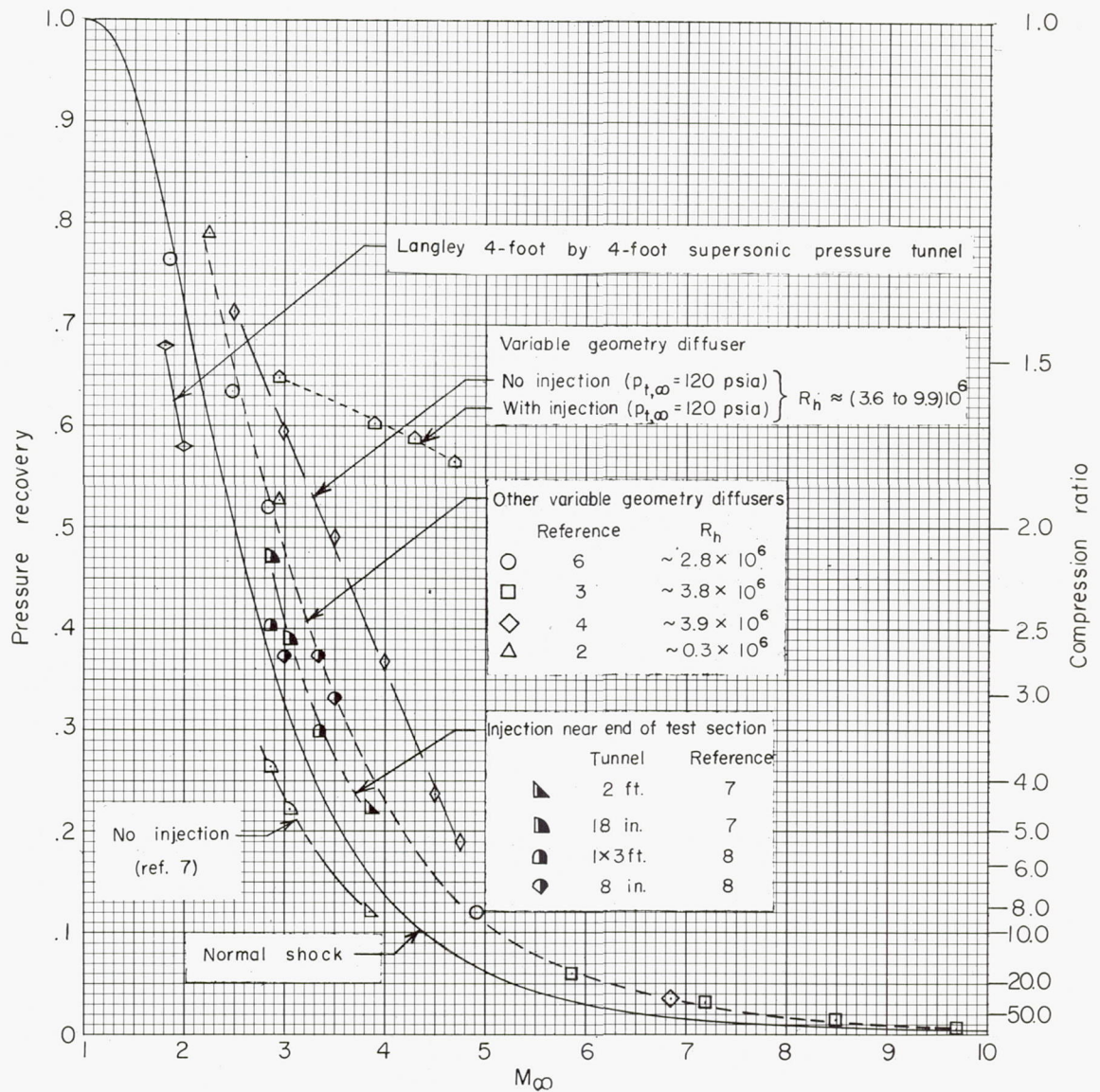


Figure 24.- Pressure-recovery characteristics of various wind-tunnel diffuser configurations.

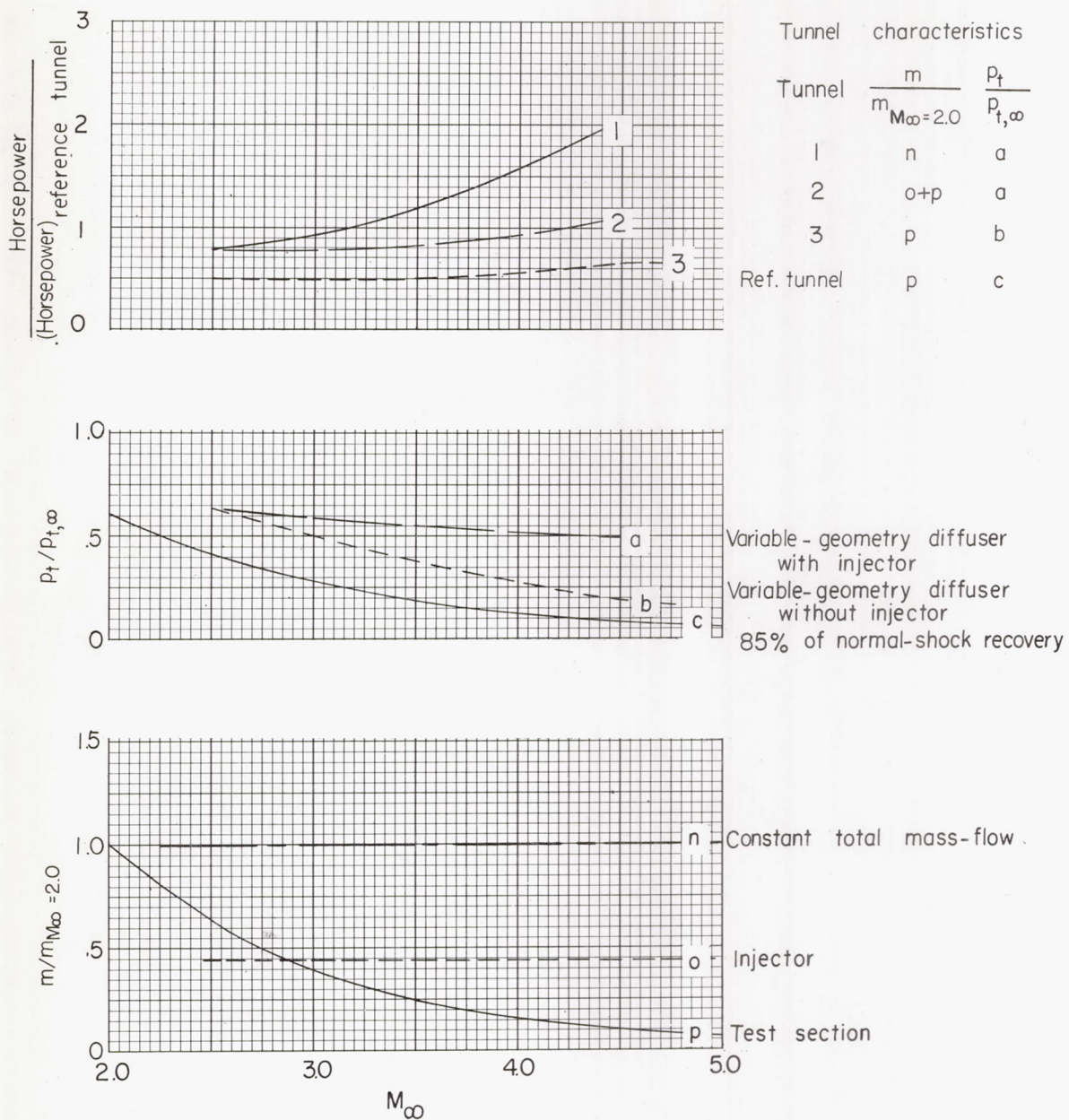


Figure 25.- Horsepower characteristics of several hypothetical wind tunnels, and pressure-recovery and mass-flow characteristics of these tunnels.