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RESEARCH MEMORANDUM

AN 8-FOOT AXISYMMETRICAL FIXED NOZZLE FOR SUBSONIC MACH NUMBERS UP TO 0.99 AND FOR A SUPERSONIC MACH NUMBER OF 1.2

By Virgil S. Ritchie, Ray H. Wright,
and Marshall P. Tulin

Langley Aeronautical Laboratory
Langley Air Force Base, Va.

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SUMMARY

The design and operation of a fixed nozzle of axial symmetry for high-subsonic Mach numbers and for a supersonic Mach number of 1.2 was investigated in connection with the conversion of a large high-speed subsonic wind tunnel to transonic operation. The nozzle was carefully designed by potential-flow theory and was adjusted for boundary-layer development. The results of flow surveys in the nozzle indicated that the uniformity of the flow in the supersonic test section was sufficient for model testing. In the subsonic test section provided in the region of the nozzle throat, the flow was of remarkable uniformity and permitted the testing of small models at Mach numbers as great as 0.99.

Small flow irregularities, which were propagated along Mach lines, tended to become concentrated near the nozzle axis. Reduction of some waviness in the surface of the nozzle by amounts of the order of 0.006 inch resulted in improvement of the flow, and deviations from the design Mach number at the nozzle axis did not then exceed 0.02. Small surface irregularities of the nature of roughness, cracks in the surface of the plaster liner, and small discontinuities in slope produced no noticeable effect on the flow and were presumed masked by the thick boundary layer.

Humidity effects were controlled by heating of the flow mixture. No condensation shocks could be detected even with considerable cooling. The slow formation of fog, which occurred with supercooling, produced a distributed effect on the Mach number distribution.

The power required for supersonic operation was in substantial agreement with the smallest of several estimates obtained from previously published information.

INTRODUCTION

Because of the special problems encountered in flight near sonic speeds, an urgent need exists for additional aerodynamic testing facilities for use at Mach numbers near unity. As a means of providing the needed facilities, large subsonic wind tunnels already operating at Mach numbers slightly less than unity may be provided with supersonic nozzles to permit testing at Mach numbers slightly greater than unity. The conversion of the Langley 8-foot high-speed tunnel to transonic operation has been considered and one phase of the problem has been the design and operation of a fixed nozzle of axial symmetry to permit testing at high-subsonic Mach numbers and at a supersonic Mach number of 1.2. The tunnel is of the single-return, closed-throat type, of circular cross section throughout, and the low-speed part of the return passage is open to atmospheric pressure. Installation of the nozzle as a plaster liner in the throat region of the tunnel was undertaken late in 1947 and the greater portion of the flow surveys reported herein were obtained during the early part of 1948. This paper, which should be of interest to those concerned with transonic wind tunnels, covers the design and operating characteristics of the nozzle.

NOZZLE DESIGN

Preliminary Considerations

The design of the nozzle was decisively influenced by the original tunnel geometry and by the power available. Experimental data from preliminary surveys with various flow-expansion liners in the throat of the tunnel indicated that the power required at Mach numbers as great as 1.3 should not exceed the 16,000 horsepower available. In order to assure adequate operating power under all test conditions, the nozzle was therefore conservatively chosen to produce a Mach number of 1.2.

A circular nozzle was selected rather than the simpler two-dimensional rectangular nozzle mainly because its installation involved less modification to the existing tunnel, but several other advantages result from the circular cross section. The boundary-layer displacement thickness, knowledge of which is required in the nozzle design, is more reliably estimated for the circular nozzle, because the boundary layer is in general uniformly distributed over the surface, whereas for the rectangular nozzle the boundary layer tends to thicken more rapidly in the corners. Moreover, because of this boundary-layer behavior and also because of the lesser perimeter, with given cross section, the circular nozzle is expected to require less power than a rectangular nozzle with the same mass flow. The circular section may also be advantageous in facilitating the testing of propellers and bodies of revolution.

Several known disadvantages are associated with the circular cross section. Perhaps the most serious of these is the tendency of disturbances arising from inaccuracies in design or from axisymmetrical irregularities at the wall to become concentrated near the center of the nozzle. Because of this tendency, special care is required to assure accuracy of design and installation. It is believed essential, for instance, to allow for variations in displacement thickness of the boundary layer. The circular cross section is also disadvantageous with respect to changing from one supersonic Mach number to another (by means of removable or adjustable nozzle blocks) and with respect to providing windows for the use of flow-visualization equipment.

The shape of the plaster liner was largely dictated by the original tunnel lines. (See fig. 1.) A long, slowly converging entrance liner was required in order to produce a steady uniform flow at the throat (Mach number 1.0), since such a uniform flow was to be assumed in the theoretical nozzle design. Inasmuch as neither the required precision of this flow nor the precision attainable with a given geometrical arrangement was known, the entrance liner was made as long and as slowly convergent as the original tunnel shape in combination with other requirements, such as necessary length of supersonic portion and required thickness of plaster, would allow. The length of the divergent portion of the nozzle was chosen as 90 inches, which is about 1.5 times the minimum possible length for a nozzle producing a Mach number of 1.2. This choice of length was believed to be conservative with respect to the production of a satisfactorily uniform flow in the test region. The thickness of the liner, which affects its fairing into the original tunnel walls at its upstream and downstream ends, was dictated by the requirement of at least 3/4 inch of plaster to permit proper anchoring to the original steel wall. In meeting these various requirements, the liner extended 140 inches upstream and faired into the original tunnel 125 inches downstream from the throat. Space limitations restricted the test section length to only about one-half the tunnel diameter unless sufficient power were available to draw the shock into the diffuser. Lack of space would in any case have prevented the installation of a second throat, but with Mach numbers not much greater than 1.2 the efficiency of diffusion through the normal shock is believed to be little if any less than that in a diffuser operating between the same two Mach numbers. The requirement of a long, slowly converging entrance liner lends itself to the production of a short subsonic test section near the throat, and such a test section was provided. (See fig. 1.) A narrow window in the nozzle wall extending from upstream of the throat to downstream of the supersonic test section was provided for visual observation of flow phenomena.

Potential-Flow Design

Divergent portion of nozzle.- The steady supersonic potential flow in the axially symmetric divergent portion of the nozzle was calculated by the method of characteristics. (See reference 1.) For the application of this method, a Mach number distribution along the axial center line was arbitrarily selected. (See fig. 2.) The Mach number increased from 1.0 at the throat to 1.2 at point A 60 inches downstream of the throat. The calculation then proceeded from the assumed values of Mach number along the center line and from the constant value 1.2 along the characteristic through A, marking the upstream boundary of the uniform-flow region.

The axial Mach number distribution was chosen with zero axial Mach number gradient at the throat in order to be consistent with the assumption of uniform flow at that point. The accuracy of the step-by-step flow calculations decreased near the throat and the Mach number intervals between points of the Mach net were reduced. The fact that the chosen Mach number gradients were small in the region near the throat was considered favorable to the accuracy of the calculation.

The characteristic network for the flow in the divergent portion of the nozzle is shown in figure 3. The numerical calculations were started at point A and were extended step by step into the remainder of the field. Note that the nozzle boundary was not immediately defined but that the network was extended beyond the probable position of the walls. This method of calculating from the center outward and from larger to smaller values of the Mach number is believed to be more accurate than the opposite method of calculation. In particular, attempts to calculate from a given wall shape to the flow near the center led to difficulties due to magnification of the inaccuracies inherent in a step-by-step computing process.

Once the flow field had been calculated, the Mach numbers and the flow angles throughout the field were known. From the flow angles the streamlines could be obtained by a method suggested by Mr. Morton Cooper of the Compressibility Division of the Langley Aeronautical Laboratory. The streamline forming the effective boundary of the nozzle was assumed to pass through the throat at a radius equal to the geometrical radius of the tunnel decreased by the estimated displacement thickness (about 0.25 in.) of the boundary layer. The increment in radius of this streamline at downstream stations was obtained by integrating the tangent of the flow angle (or the flow angle itself since it did not exceed 0.0175 radian) along a line CD (fig. 3) parallel to the axis OA. Special care is required to assure the accuracy of the integration. The angles used in this integration should strictly have been those lying along the streamline itself. A second approximation to the true streamline was therefore obtained by taking the line integral with respect to distance parallel to OA of the flow angles along the approximate streamline CE obtained by the first integration. This process is rapidly convergent and a third approximation CB yielded no further change in radii. The flow angles used in these integrations were carefully determined by interpolation from the points of the Mach net. The effective radii (that is, without adjustment for boundary-layer displacement) determined for the divergent portion of the nozzle and for the supersonic test section are given in table I.

One-dimensional theory was used to check the over-all expansion of the effective cross-sectional area of the nozzle from the minimum section OC (fig. 3) where the flow is one-dimensional to the supersonic test section where the flow is again one-dimensional. The over-all increase in nozzle radius yielded by the one-dimensional theory differed from that obtained by the characteristic method by less than 0.001 inch.

Convergent portion of nozzle.- The convergent portion of the nozzle was designed to meet the requirement of increasingly gradual convergence toward the throat, where the Mach number unity is reached, and to produce a short subsonic test section in the vicinity of the throat. With increase in Mach number toward unity, the flow becomes increasingly sensitive both to changes in cross section of the stream tube and to the curvature of the surface. The entrance liner was therefore designed with the curvature in the direction of flow decreasing to zero at the throat.

The sensitivity of the flow is such that considerable care was required to assure that sonic speed would actually be attained at the assumed throat. Adjustment to allow for development of the boundary layer was therefore required. With uniform flow (zero pressure gradient) the slope of the boundary layer was estimated at 0.0018. If, however, the velocity is increasing through the throat, the boundary-layer thickness will increase less rapidly. In fact, it seems quite possible that at Mach number unity on the supersonic side a zero-gradient flow would be unstable. Any accidental thinning of the boundary layer would then produce an increment in Mach number, and the accompanying pressure gradient would produce the boundary-layer thinning required to maintain the Mach number gradient. Moreover, this effect might be progressive so that the effective throat would move upstream. In order to insure stability of the effective throat position the slope of the entrance liner at the throat was taken as 0.0015 instead of the value 0.0018 estimated from the boundary-layer growth.

A mathematical expression which meets the requirements as to slope, curvature, and slowness of convergence and also connects reasonably smoothly with the original tunnel lines gives the geometric radius in inches as

$$r = 46.25 - 0.0015x + \left(1.3 \times 10^{-8}\right)x^4$$

where 46.25 is the geometric radius at the effective minimum section in inches (fig. 1) and x is the distance in inches upstream of the effective minimum section. This expression was used for calculating the radii given in table II. The geometric minimum occurs 30 inches upstream of the effective minimum section.

Boundary-Layer Compensation

The necessity for taking account of the boundary-layer development has already been pointed out in connection with the design of the entrance liner. With supersonic Mach numbers as low as that for which this nozzle is designed it is also imperative to allow for the boundary-layer development in the divergent part of the nozzle.

In the absence of experimental data on the behavior of the turbulent boundary layer in transonic flow, certain reasonable assumptions were made. The boundary-layer behavior was assumed to be essentially the same, except for direct effects of density changes, as for incompressible flow; and, just as with incompressible flow, the flow outside the boundary layer was assumed to behave as if the streamlines near the boundary were displaced away from the wall by an amount equal to the displacement thickness of the boundary layer.

The calculation of displacement-thickness distributions is made through use of the boundary-layer momentum equation. Through this equation the influence of the outside flow upon the development of the boundary layer is determined. Approximate formulas for the computation of turbulent-boundary-layer momentum thicknesses in compressible flows are given in reference 2. The calculations require a knowledge of the Mach and Reynolds number distributions along the tunnel, the initial boundary-layer condition, and the variation of an important boundary-layer parameter H_c, the ratio of the displacement thickness to the momentum thickness. The displacement thickness is obtained from the momentum thickness by multiplication with H_c. The variation of H_c was determined experimentally from data obtained in the Langley 8-foot high-speed tunnel during a preliminary nozzle investigation in which rough wooden nozzles were tested in the tunnel. For the conditions existing in the 8-foot tunnel, H_c was found to vary mainly with Mach number and to follow approximately the variation suggested in reference 2. A more extended discussion of this variation is given in reference 3.

The nozzle design was facilitated by taking boundary-layer measurements along the wall of the original tunnel. A preliminary check of the applicability of the calculation methods of reference 2 to the prediction of the boundary-layer development axially along the wall of the 8-foot tunnel was made by calculating the displacement-thickness growth in a subsonic liner operating at a test-section Mach number of 0.85 and comparing the calculated values with measured values of the boundary-layer displacement thickness at the tunnel wall (fig. 4). In this preliminary calculation the variation of H_c with Mach number was neglected and a constant value of 1.28 was used. This procedure involved only small error since the variation of H_c with Mach number up to a Mach number of 0.85 was not very great. The agreement between theory and experiment (fig. 4) was considered satisfactory.

The calculation of the distribution of boundary-layer displacement thickness for use in the nozzle design proceeded from a measured value far upstream in the contraction cone. (See fig. 4.) In the upstream portion of the nozzle, the Mach number distribution needed in the calculation was determined from the cross-sectional area at each point in connection with the assumption of uniform axial (one-dimensional) flow at each section and Mach number unity at the throat. For such a slowly converging entrance, the assumption of one-dimensional flow involves negligible error. In estimating the Mach numbers in the sensitive region near the throat, the previously estimated slope of the displacement thickness was taken into account. In the divergent portion of the nozzle, the Mach number distribution at the wall was taken from the potential-flow design. The calculated Mach number distribution is shown in figure 5. The variation with Mach number of the ratio H_c was taken from the results of the preliminary investigations.

The calculated course of the displacement thickness is shown in figure 6. Note that a decrease in thickness is predicted in the region of rapid acceleration just downstream from the throat. The experimental data presented in figure 6 are discussed in a subsequent section in connection with the results of flow surveys in the nozzle. The geometric radius at any section of the divergent part of the nozzle is obtained by adding the boundary-layer displacement thickness at that section to the radius determined in the portential-flow design. The geometric radii so obtained are presented in table I.

NOZZLE INSTALLATION

Installation of the nozzle as a plaster liner inside the original walls of the tunnel involved the use of special construction materials and methods. A high-strength, quick-setting plaster, Hydrocal B-11, was selected for the construction material after practical tests indicated that it was satisfactory for the purpose of the temporary installation. The plaster was very hard and resistant to chipping and on setting increased only slightly in volume.

The method used for installing the plaster liner consisted essentially of fastening metal lath to the steel walls of the tunnel (by welding) in the region selected for the liner installation and applying plaster coats to build up the liner to the proper geometric dimensions. (See tables I and II.) All leaks in the tunnel wall in the region designated for the liner installation were stopped (by welding) before the plaster was applied. The final plaster coat was shaped to the desired axial profile by means of a template rig which was designed to ride on metal rings fastened securely to the tunnel wall at stations 2 feet apart axially along the nozzle length. The reference rings were ground as nearly as possible to true circular shape by means of a grinding attachment rotating off a rigid tube alined along the tunnel axis of symmetry. The average radii of the various reference rings were determined very accurately, using the central tube as a reference, and no deviations of more than 0.004 inch from true circular shape were obtained. This technique of building up the plaster liner left at the ring locations narrow unfilled channels which required filling and sanding after the rest of the nozzle was installed. Filling was also required at the edges of a long narrow window installed in the top of the tunnel for observation of flow phenomena.

A detail view showing stages of the plaster-application technique is given in figure 7. In the foreground of this view, metal lath is shown attached to the tunnel wall; in the center of the view, a basic coat of plaster is shown applied to the metal lath; and in the background, a strip of the final smooth plaster coat is visible. In figure 8, the template rig used for shaping the final plaster coat to the desired profile between adjoining reference rings is shown resting on the tunnel floor. A photograph of the completed nozzle, as viewed from downstream, is given as figure 9.

The accuracy of the installation was of importance primarily in the throat and divergent portion of the nozzle. Physical measurement of the over-all accuracy of the nozzle installation was not feasible because of the large amount of time and careful work required for such an undertaking. The axial profile of the nozzle wall was carefully checked, however, by means of templates of the design geometric shape extending from 24 inches upstream of the effective minimum to 96 inches downstream of the effective minimum section. These checks of the axial profile by means of templates were made at 2-inch intervals around the entire circumference of the nozzle, and maximum deviations of the actual wall shape from the design shape did not exceed about 0.011 inch. These maximum deviations in the original installation were subsequently reduced (by sanding and filling) to deviations of the order of 0.005 inch. The finished wall surface, although glazed in appearance and smooth to the touch (except for slight discontinuities in slope at some reference-ring locations) was not so fine as that commonly used for small supersonic nozzles. Relative to the tunnel size, however, the surface was very smooth.

SURVEYS OF NOZZLE FLOW

Apparatus and Measurements

The arrangement shown in figure 10 was used for the measurement of static pressures at the wall and near the axial center line of the nozzle. The static-pressure orifices in the surfaces of both the wall and the 2-inch-diameter cylindrical axial-survey tube were of 0.031-inch diameter and were installed normal to the surface. Wall orifices were located axially 2 inches apart in the throat and supersonic-flow region and 6 inches apart in other regions of the nozzle. Those in the cylindrical tube were located 2 inches apart in the throat and supersonic test section and 6 inches apart elsewhere. Wall orifices were installed 90° apart around the circumference of the channel at several axial stations to permit checks for symmetry of the flow. Wall and cylindrical-tube surfaces near static-pressure orifices were kept free of irregularities. Local static-pressure measurements by means of orifices in the cylindrical-tube and nozzle-wall surfaces were assumed to be equal to those outside the boundary layer except in the region of shock where pressure changes would occur over an axial distance greater at the surface than outside the boundary layer.

Total-pressure measurements in the subsonic flow upstream of the nozzle throat were obtained by means of a total-pressure tube in the ellipsoidal nose at the upstream end of the cylindrical tube. (See fig. 10.) The total pressures were assumed to be correct as measured and to apply downstream of the station of measurement as long as the flow remained essentially shock free.

The cylindrical tube was positioned along the axial center line of the nozzle so that the upstream end of the tube was located sufficiently far upstream of the throat to introduce no appreciable disturbances in the flow near the effective minimum section. The tube was maintained in position by a rigid support system located in the tunnel diffuser (fig. 10) and by sweptback stay wires running from the wall of the contraction cone to the nose of the tube. The tube was capable of axial adjustment to permit static-pressure measurements at intervals as close as desired.

Cones (of 3° included angle) equipped with static-pressure orifices and total-pressure tubes were used for more detailed surveys of the flow in the supersonic test section than could be obtained by use of the cylindrical tube. Static-pressure orifices were of 0.010-inch diameter and were installed normal to the surface at 2-inch intervals along the length of the cone. Orifices were also installed at angular locations 90° apart in the cone surfaces to permit approximate checks for symmetry and inclination of the flow. Total-pressure tubes of 0.010-inch inside diameter were installed in cone tips which were interchangeable with sharp tips normally used with the cones. The cones were capable of axial adjustment to permit measurements at axial intervals as close as desired. Static pressures measured by means of the cones required correction for induced velocity at the surfaces of the cones; but this correction was very small, corresponding to a theoretically estimated Mach number increment of the order of 0.001 and was near the accuracy of the pressure measurements. A large cone 66 inches long (fig. 11) was designed for flow surveys very near the axial center line of the supersonic test section, and an arrangement of two small cones located 180° apart angularly (fig. 12) was used for surveys at distances of 3, 7, and 13 inches off the center line.

Satisfactory precision in pressure measurements was more difficult to attain by means of orifices in cones than by means of orifices in the cylindrical tube, because the thinner boundary layer on the cones rendered the cone surface conditions more critical. The thin boundary layers on the cones were considered advantageous, however, for pressure measurements in the vicinity of shocks.

Pressures were measured with multiple-tube manometers containing tetrabromoethane. These manometers were photographed simultaneously. The random error in the pressure measurements was estimated to be no greater than 3 pounds per square foot, which is equivalent to an error of less than 0.0033 in the flow Mach number throughout a Mach number range extending from 0.4 to 1.3.

A shadow system was provided to supplement the pressure apparatus in examination of the supersonic flow for the presence of strong disturbances. The equipment used for producing shadow images due to changes in density gradients in the supersonic flow consisted of an intense point-source light and a condensing lens which were used to project a beam of approximately parallel light rays across the nozzle diameter. The shadow equipment was made portable for greater versatility in examination of flow disturbances. Disturbance images were observed on the nozzle wall opposite the observation window. The sensitivity of the system was sufficient to permit the observation of normal shocks at Mach numbers as low as 1.06.

The stagnation temperature of the flow mixture in the tunnel was measured by means of electrical-resistance thermometers located at several points between the tunnel wall and center line in the low-speed region upstream of the entrance cone. The static temperature, equivalent to the stagnation temperature in the low-speed region, was assumed to decrease isentropically with flow expansion.

Apparatus used for measurement of pressures and temperature in the boundary layer adjacent to the nozzle wall and methods used for reduction of the measurements are described in reference 3.

The measurements reported in this paper were, with the exception of those made during several runs in which condensation effects were deliberately introduced, obtained with sufficiently high tunnel temperatures to preclude the existence of significant condensation effects in the flow. For typical tunnel-operation temperatures, the Reynolds numbers attained in the Mach number 1.2 test section were approximately 3.8×10^6 per foot. Mach numbers were obtained from the ratio of static to total pressures.

Results and Discussion

Mach number distributions.- Mach number distributions obtained from pressure measurements at the wall and at the surface of the cylindrical tube along the center line of the nozzle for both subsonic and supersonic operation are presented in figures 13 and 14. These results indicated that with subsonic operation the flow in the throat region of the nozzle was very uniform and suitable for subsonic testing purposes. An essentially zero-gradient region existed in the flow over a considerable length in the vicinity of the throat for Mach numbers up to 0.97, but above 0.97 as the Mach number approached unity the gradients in the downstream portion of this region gradually increased. (See fig. 13.) The flow in the upstream portion of this subsonic test section was uniform up to Mach numbers as high as 0.99; and at this Mach number a model 8 inches long and $1\frac{1}{3}$ inches in diameter was successfully tested with negligible interference from the tunnel walls. The length of the uniform test region was such at any Mach number that with fineness ratios about 6 the size of the model that could be tested was limited by choking rather than by the length of the test section.

With supersonic operation the experimental Mach number distribution is seen to be in excellent agreement with the design distribution (figs. 13 and 14). The fact that the experimental position of the throat (Mach number unity) is within an inch of the design position is regarded as particularly significant. A comparison of the actual distribution obtained in the contraction cone or convergent portion of the nozzle with one-dimensional theory, which includes calculated boundary-layer-displacement effects, is presented in figure 15. Near the throat region of the nozzle the agreement between the theoretical

Mach number distribution and that measured along the contraction-cone center line and wall was especially good. This agreement indicates that the flow attained near the minimum section was practically one-dimensional and free of appreciable induced velocities. The agreement became increasingly poor with distance upstream of the throat region largely because of induced velocities brought about by increasingly rapid changes of the contraction-cone radii. The generally favorable agreement between theory and experiment in the contraction cone appeared to validate the use of one-dimensional flow relations in the design of gradually converging channels for subsonic flow.

In order to obtain a more accurate indication of the character of the flow, pressure measurements were taken at $\frac{1}{4}$-inch axial increments near the center line of the divergent part of the nozzle. The corresponding Mach number distribution is presented in figure 16. These measurements were obtained by moving the cylindrical survey tube (fig. 10) axially at $\frac{1}{4}$-inch increments between runs for eight successive runs. The flow near the axis in the divergent portion of the nozzle and in the supersonic test section is seen to be free of large disturbances. The maximum flow disturbances in the supersonic test section were equivalent to deviations of 0.02 from the design Mach number of 1.2.

The Mach number variation with distance off the center line, and particularly the tendency of the flow disturbances to become concentrated at the center, was investigated by means of the survey cones (figs. 11 and 12). The larger cone (fig. 11) could be moved parallel to the nozzle axis to place pressure orifices from 0.1 inch to 1.5 inches off the axis at any station. Mach numbers obtained with this cone in two positions are compared in figure 17 with those obtained from the cylindrical tube (orifices 1 in. off center line). The comparison is affected by the lack of precision of the cone measurements and by a change, with time between tests, in the disturbance at the 77-inch station; but within the accuracy of the data no consistent variation in the Mach number within 1.5 inches of the center line can be detected.

With the cone mounting arrangement of figure 12, the Mach number distribution could be obtained at 3, 7, and 13 inches off the center line. This arrangement was used to trace the extension from the center line outward of disturbances near the 75-inch station. In the two cases investigated (two intensities of the disturbance as shown in figs. 18 and 19) the disturbance is seen to decrease, as expected, from the center outward. Considerable scatter present in the first of these cone measurements was reduced (downstream of the 73.5-in. station at 7 in. off the center line, fig. 19) by improvement of the cone surface. These disturbances followed the Mach lines of the flow and became broader outward from the center line, and because of this reason as well as because of the tendency to become concentrated at the center, they are believed to be due not to shock waves but to gradual convergence of Mach lines of compression. The Mach number distributions at 7 and 13 inches off the center line may be compared in figure 18 with those calculated by the characteristics method starting from the experimental distribution at 1 inch off the center line. From these investigations, the flow appears to be more uniform elsewhere than at the center.

From these surveys and from shadowgraph observations, the flow in the test section appears to be fairly uniform and free of shocks, a conclusion which is further supported by the agreement between the experimental and design Mach number distributions. If greater uniformity of the Mach number distribution than that existing near the center line is required for any particular test, the model can be placed in an off-center position. By the procedures hereinbefore described, a circular nozzle of size comparable to the 8-foot high-speed tunnel supersonic nozzle for Mach number 1.2 can evidently be designed to produce a satisfactorily uniform supersonic flow.

Effects of wall irregularities on flow uniformity.- The supersonic flow in the nozzle immediately after installation was not so smooth as desired (circular symbols, fig. 20) and a limited amount of time was spent in attempting to improve the flow. It was found that the flow deviations measured near the nozzle center line could be traced to irregularities in the wall shape by use of the calculated characteristics network (fig. 3). The use of this network for locating the points of origin of flow disturbances was made possible by the close agreement of the actual and design flows. Careful checks of the wall shape by means of axial-profile templates in regions where flow disturbances were suspected to originate indicated small deviations from the design shape. These deviations usually consisted of shallow bumps and depressions, sometimes axisymmetrical and sometimes localized, in the nozzle wall. In each of several instances where wall irregularities were reduced by sanding bumps and filling depressions, the associated flow disturbances were observed to have decreased. Deviations of the actual nozzle profile from the design profile, as determined by means of axial-profile templates, were reduced from ±0.011 inch to ±0.005 inch in a region extending from 24 inches upstream of the effective minimum to 96 inches downstream of the effective minimum section. Figure 20 shows a comparison of the measured Mach number distributions axially along the wall and near the center line of the nozzle before and after reduction of the wall-profile deviations; a noticeable improvement in the flow is evident, especially near the center line at a station 75 inches downstream of the effective minimum section where the deviation from the design Mach number (1.2) was reduced to about one-half of the original deviation. Further improvement of the flow was considered possible by additional work on the installation accuracy, although exact agreement of actual and design flow distributions could not be expected because of possible error in the boundary-layer prediction and because of possible periodic flow disturbances not due to wall irregularities.

An indication of the relatively small bump dimensions required to produce severe flow disturbances was obtained by fastening a 0.030-inch-diameter string around the nozzle wall at an axial station 30 inches downstream of the effective minimum section and observing its effect on the supersonic flow. A strong compression disturbance followed by a strong expansion (fig. 21), with deviations of as much as 0.13 from the disturbance-free flow of Mach number 1.2, was produced near the nozzle axis. The disturbance, which was strong near the nozzle axis, was observed as a weak compression and accompanying expansion at its intersection with the wall in the downstream part of the supersonic test section. (See fig. 21.) Small irregularities of the nature of roughness, cracks in the plaster surface, and small local discontinuities in slope produced no noticeable effect on the flow. This behavior is believed to be partially due to the masking effect of the thick boundary layer and partially due to the random nature of these disturbances, as contrasted with the axially symmetrical disturbance produced by the string.

Boundary-layer development.- An attempt was made by means of boundary-layer surveys to check the calculated displacement thicknesses at various positions along the nozzle wall (fig. 6). Although in some cases the experimentally determined values were in good agreement with the calculated values, in others considerable divergence occurred. Moreover, values of the displacement thickness obtained at the same time and at the same axial position but at different angular positions on the wall of the nozzle failed to agree among themselves. Several possible reasons for these divergences include local accumulations of boundary-layer air due to lack of perfect symmetry in the pressure distributions, localized leaks into the tunnel upstream of the measuring position, and variation of stagnation temperature (assumed constant) through the boundary layer. An analysis of the boundary-layer velocity profiles obtained in this investigation is contained in reference 3.

The success of the over-all design in producing the design Mach number distributions, particularly in the vicinity of the throat, indicates that the assumptions made in the design as to the behavior of the turbulent boundary layer in transonic flow and as to its effect on the outside flow are at least approximately correct. If the changes in boundary-layer displacement thickness had been neglected in the design, the nozzle would have been expected to produce a Mach number of about 1.18 instead of 1.20.

OPERATIONAL CHARACTERISTICS

Effects of Humidity and Heat Transfer

Because the low-speed part of the tunnel was open to the atmosphere, care was required to prevent flow disturbances due to condensation in the high-speed, low-temperature part of the nozzle. Adverse condensation effects were prevented by allowing the tunnel to become heated, a method that was quite effective in winter when stagnation temperatures not above 180° F were required but was somewhat inconvenient in summer when the required stagnation temperatures might reach 230° F. The stagnation temperature was controlled by adjusting the amount of air exchange through an annulus around the tunnel in the low-speed section.

Condensation shocks, which have been the source of some difficulty in other supersonic tunnels, could be prevented according to reference 4 by limiting supercooling to less than 54° F. In the supersonic nozzle herein described, however, no condensation shock could be detected either from pressure distributions or from shadowgraph observations, even though at times the nominal supercooling was allowed to exceed 54° F. On the other hand, fog could usually be observed in the nozzle. The fog was most evident in the cooler region near the wall where, because of the incomplete mixing of the cooling air with the remainder of the air in the tunnel, the temperature might be as much as 50° F less than that at the center. The reason for this behavior is believed to be that in a continuous-circuit tunnel such as the Langley 8-foot high-speed tunnel the solid or liquid particles, and possibly ions, carried around in the air stream are sufficiently numerous to provide nuclei for condensation. Moreover, in such a large nozzle, the rates of temperature change are sufficiently small to permit slow condensation (fog formation) on these nuclei, a possibility which is suggested in reference 5. The 54° F supercooling is therefore never even approached, so that the cataclysmic process of formation and growth of molecular nuclei which is responsible for the condensation shock cannot occur.

The slow condensation process produces only a small effect on the Mach number distributions, and this effect is distributed rather than concentrated as in the case of the condensation shock. In order to investigate this effect, the nozzle was operated with varying humidity conditions. The corresponding Mach number distributions at the wall and near the center of the nozzle are shown in figure 22, where the relative humidity in the low-speed section of the tunnel is increasing from runs 1 to 4. The Mach number at which the estimated saturation temperature is reached is indicated for each case. The Mach number is seen to decrease with increasing condensation throughout the nozzle.

This result was at first surprising because heat addition, due to the condensation, increases the Mach number in subsonic flow but has the opposite effect in supersonic flow. It must be remembered, however, that the amount of condensation is not uniform throughout the nozzle but is increasing with increasing Mach number (decreasing temperature). The amount of condensation is therefore greater at the throat than in any subsonic part of the nozzle; but the Mach number at the throat cannot exceed unity. The mass flow is therefore reduced throughout the subsonic part of the nozzle. The amount of the reduction in Mach number at any station in the subsonic part due to the reduction in mass flow exceeds the increase due to condensation at that station because the condensation at the throat is always greater. In the supersonic region, the Mach numbers are reduced because the amount of condensation is everywhere greater than at the throat.

Another effect between runs 1 and 4 (fig. 22) is the movement of the effective throat approximately 7 inches downstream. Such an effect might be due to a time lag in condensation, but it is believed due to the influence of heat transfer on the boundary-layer development. Because of the decreasing stagnation temperature between runs 1 and 4, the relatively increasing heat transfer into the tunnel increases the rate of growth of the boundary-layer displacement thickness. In the very sensitive region near the throat, this effect may be sufficient to shift the effective throat the observed distance downstream. Such a shift requires less than 0.00014 increase in the slope of the boundary-layer displacement thickness.

The flow near the center of the nozzle is influenced by condensation in the cooler region near the wall. This effect is evident in figure 23, which shows the dependence of the Mach number in the test section on the temperature relative to saturation temperature. Note that the supercooling at the wall is much greater than that at the center. In spite of the fact that the design Mach number on the center line is the same at the 90-inch station (B" in fig. 3) as at the 60-inch station (A in fig. 3) the decrement due to condensation is much greater at the 90-inch station (fig. 23). This effect is due to the progressive condensation in the region near the walls, for the flow in this region affects that downstream along the Mach lines.

Wall Interference for a Typical Transonic-Airplane Model

In order to obtain an estimate of the maximum fuselage length, for a typical transonic-airplane model, which could be tested free of wall interference effects in the Mach number 1.2 nozzle, calculations were made of the flow pattern about a simple body of revolution located at the axis of symmetry in the supersonic test section. This body consisted of a cone of revolution, whose angle (35° included angle) was identical with the nose angle of the model fuselage, followed by a cylindrical afterbody whose diameter (approx. 3.75 in.) was equal to the maximum diameter of the model. The flow calculations depended, for the most part, on the method of characteristics but the use of some approximate theory was also necessary because of the existence of subsonic flow near the cone. The results of the calculations are shown in figure 24. Tests of the transonic-airplane model yielded the nozzle-wall Mach number distribution shown in figure 24, which indicated a measured shock slightly upstream of the calculated nose-shock location at the wall. The measured shock appeared as a gradual compression because of the thick boundary layer at the nozzle wall. The intersection of the reflected shock and the body could not be calculated because of the occurrence of subsonic flow behind the shock in the region of intersection, but the maximum fuselage length was estimated to be about one tunnel radius. Consideration of the interference problem by making use of the free-stream Mach lines yields nonconservative results as is shown in figure 24. No measurements were available to determine the axial location of the reflected shock at the center of the tunnel downstream of the model fuselage.

Effects of Flow Nonuniformities on Model Forces

In order to obtain practical information concerning the effects of flow disturbances on model force measurements, a complete model of a transonic airplane was investigated in the supersonic test section when the flow disturbance at the 75-inch station on the axial center line produced a deviation of as much as 0.05 from the test-section Mach number of 1.2 (fig. 19). The axial location of this moderately strong flow disturbance was varied with respect to the model by changing the axial location of the model along the center line of the test section; and model forces were measured with the flow disturbance located in several regions near the wing and tail surfaces. The results of this investigation, reported separately in reference 6, indicated that at Mach number 1.2 no significant changes of lift, drag, and pitching-moment coefficients for the given model were produced by flow nonuniformities equivalent to Mach number decrements of the order of 0.05 or less at the axis of symmetry and extending over an axial distance of 2 or 3 inches or 6 to 10 percent of the model length. Reference 6 also contains experimental data which indicate that the fluctuating normal shock in the stream at the downstream end of the supersonic test section has no significant effect on model forces until it approaches the region of the model base and tail surfaces.

Power Requirements

The power required to operate the nozzle is shown in figure 25. Additional power data are included from investigations in the diffusing portion of a liner originally installed in the tunnel to facilitate testing at high subsonic speeds and from the preliminary rough wooden nozzle tests. The apparent scatter is due to the wide variation in conditions under which the data were obtained. For the plaster nozzle operating at Mach number 1.2 the power given, about 12,500 horsepower, is that required for maintenance of the supersonic flow in the test section. This value was obtained in winter; with the higher operating temperature required for avoiding condensation effects in summer, the power required for the same Mach number would be somewhat greater. In general, the Mach number values above 1.2 were maintained over only short distances, and the power values may therefore be somewhat less than would be required for the production of a test region at the same Mach numbers. In the case of the original tunnel with subsonic liner and also for the rough wooden nozzles, shock waves existing in the flow upstream of the terminal normal shock may have affected the power required. Because the Mach number is in most cases not constant over the cross section of the tunnel, the Mach number corresponding to any given power value may also depend on the position at which the Mach number is obtained. The power absorbed by the strut-and-survey tube was estimated not to exceed 850 horsepower.

The estimated power required because of the normal shock terminating the supersonic flow is shown in figure 25 for comparison. The shock power at Mach number 1.2 is still small and is only a minor part of the total power. With increasing Mach number, the shock-power curve and the course of the experimental power values tend toward convergence. This behavior is to be expected if the diffuser flow is not spoiled by the shock, because with increasing Mach number the diffusion through the normal shock exceeds by an increasing amount the preceding expansion in the nozzle, so that decreasing diffusion is required of the diffuser. As pointed out in reference 3, the flow did not separate behind the normal shock terminating the supersonic flow with Mach number 1.2.

The experimental power data may be compared with nozzle-empty estimates made without consideration of differences in tunnel geometry and Reynolds number from information presented in references 7 to 9. The power required to operate the 8-foot high-speed tunnel was considerably less than estimated from the first two of these references (fig. 25). The upper curve in figure 25 was computed for adiabatic compression from blower pressure ratios given in reference 7; a tunnel-fan efficiency of 80 percent was assumed. The curve from reference 8 was obtained for test-section pressures representative of those in the 8-foot high-speed tunnel during operation. Reference 9, which became available after completion of this investigation, leads to a power estimate in good agreement with the present experimental data (fig. 25). The power estimates contained in reference 9 were based on diffuser-efficiency test results from a number of wind tunnels including a high-speed subsonic tunnel larger than the 8-foot high-speed tunnel; in reference 9 certain reasonable assumptions were made regarding the variation of total-pressure losses with Mach number.

CONCLUSIONS

1. A large, approximately 8-foot-diameter, supersonic nozzle of circular cross section for Mach number 1.2 was made to produce a test region of satisfactorily uniform flow.

2. Considerable care, including compensation for boundary-layer development, was required in the design and construction, particularly in the region of the throat.

3. Significant improvement of the nozzle flow was achieved by reducing irregular surface waviness by amounts of the order of 0.006 inch in height.

4. Experimental Mach number distributions in the nozzle were in excellent agreement with the design distributions. The flow in the supersonic test section was free of shocks and of a degree of uniformity satisfactory for aerodynamic testing; deviations from the design Mach number at the center line did not exceed 0.02.

5. The design features leading to the supersonic test section were consistent with the production in the throat region of a subsonic test section of satisfactorily uniform flow for Mach numbers as great as 0.99. The length of the uniform test region was such at any Mach number that with fineness ratio about 6 the size of the model that could be tested was limited by choking rather than by the length of the test section.

6. Flow irregularities, which were propagated along Mach lines, tended to become concentrated near the axis of symmetry.

7. Small surface irregularities of the nature of roughness, cracks in the plaster surfaces, and small localized discontinuities in slope produced no noticeable effect on the flow and were therefore concluded to be masked by the thick boundary layer. An axisymmetrical 0.030-inch-diameter bump around the wall in the divergent portion of the nozzle produced a large disturbance corresponding to a Mach number deviation of 0.13 at the axis.

8. In this large nozzle, no evidence was found of the existence of a condensation shock. Slow condensation occurred and produced a distributed effect on the Mach numbers.

9. The power required for supersonic operation was in substantial agreement with the smallest of several estimates obtained from previously published information.

10. Except for the direct effects of density changes, the turbulent boundary layer in transonic flow behaved in essentially the same manner as for incompressible flow; its effect on the outside flow was equivalent to a displacement of the streamlines near the wall by an amount equal to the boundary-layer displacement thickness.

Langley Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Air Force Base, Va.

REFERENCES

1. Ferri, Antonio: Elements of Aerodynamics of Supersonic Flows. Ch. XIII. The Macmillan Co., 1949.

2. Tetervin, Neal: Approximate Formulas for the Computation of Turbulent Boundary-Layer Momentum Thicknesses in Compressible Flows. NACA ACR L6A22, 1946.

3. Tulin, Marshall P., and Wright, Ray H.: Investigation of Some Turbulent-Boundary-Layer Velocity Profiles at a Tunnel Wall with Mach Numbers up to 1.2. NACA RM L9H29a, 1949.

4. Oswatitsch, Kl.: Formation of Fog in Wind Tunnels and Its Influence on the Investigation of Models. Reps. and Translations No. 459, British M.A.P. Völkenrode, June 1946.

5. Oswatitsch, Kl.: The Formation of Fog in High-Speed and Supersonic Tunnels. Reps. and Translations No. 948, British M.A.P. Völkenrode, Sept. 1, 1947.

6. Ritchie, Virgil S.: Effects of Certain Flow Nonuniformities on Lift, Drag, and Pitching Moment for a Transonic-Airplane Model Investigated at a Mach Number of 1.2 in a Nozzle of Circular Cross Section. NACA RM L9E20a, 1949.

7. Ackeret, J.: High-Speed Wind Tunnels. NACA TM 808, 1936.

8. Crocco, Luigi: High-Speed Wind Tunnels. Translation No. 366, Materiel Command, U. S. Army Air Corps, July 30, 1943.

9. Goethert, Bernhard: Power Requirements and Maximum Reynolds Number in Transonic Wind Tunnels Having a Maximum Mach Number of 2.0. AAF No. 5564, Air Materiel Command, Army Air Forces, March 25, 1947.

TABLE I

COORDINATES FOR DIVERGENT PORTION OF THE MACH NUMBER 1.2 NOZZLE

Distance downstream of effective minimum section, x (in.)	Geometric radius, r (in.)	Effective radius, r' (in.)	Distance downstream of effective minimum section, x (in.)	Geometric radius, r (in.)	Effective radius, r' (in.)
0	46.2500	46.00270	56	46.8771	46.5820
2	46.2543	46.00275	58	46.8973	46.6001
4	46.2574	46.0028	60	46.9153	46.6157
6	46.2602	46.0029	62	46.9313	46.6295
8	46.2620	46.0035	64	46.9460	46.6422
10	46.2631	46.0051	66	46.9595	46.6532
12	46.2669	46.0101	68	46.9717	46.6621
14	46.2741	46.0199	70	46.9828	46.6700
16	46.2870	46.0342	72	46.9928	46.6764
18	46.3054	46.0520	74	47.0023	46.6819
20	46.3280	46.0543	76	47.0103	46.6865
22	46.3540	46.0988	78	47.0174	46.6898
24	46.3827	46.1270	80	47.0238	46.6922
26	46.4142	46.1558	82	47.0292	46.6940
28	46.4470	46.1870	84	47.0339	46.6951
30	46.4810	46.2187	86	47.0386	46.6960
32	46.5157	46.2513	88	47.0426	46.6964
34	46.5507	46.2856	90	47.0467	46.6966
36	46.5855	46.3199	92	47.0502	46.6967
38	46.6203	46.3512	94	47.0535	46.6967
40	46.6532	46.3833	96	47.0575	46.6967
42	46.6860	46.4131	98	47.0615	46.6967
44	46.7169	46.4415	100	47.0652	46.6967
46	46.7470	46.4678	105	47.27	-------
48	46.7758	46.4930	110	47.53	-------
50	46.8043	46.5200	115	47.80	-------
52	46.8302	46.5450	120	48.06	-------
54	46.8550	46.5645	125	[a]48.32	-------

[a]Wall of tunnel diffuser.

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'TABLE II

COORDINATES FOR CONTRACTION CONE OR CONVERGENT PORTION OF THE

MACH NUMBER 1.2 NOZZLE

Distance upstream of effective minimum section, x (in.)	Geometric radius, r (in.)
0	46.2500
2	46.2470
4	46.2440
6	46.2410
8	46.2381
10	46.2351
12	46.2323
14	46.2295
16	46.2269
18	46.2244
20	46.2221
22	46.2200
24	46.2183
26	46.2169
28	46.2160
30	46.2155
32	46.2156
34	46.2164
36	46.2178
38	46.2201
40	46.2233
42	46.2275
44	46.2327
46	46.2392
48	46.2470
50	46.2563
55	46.2865
60	46.3285
65	46.3846
70	46.4571
80	46.6625
90	46.9679
100	47.4000
110	47.9883
120	48.7657
130	49.50
140	50.27

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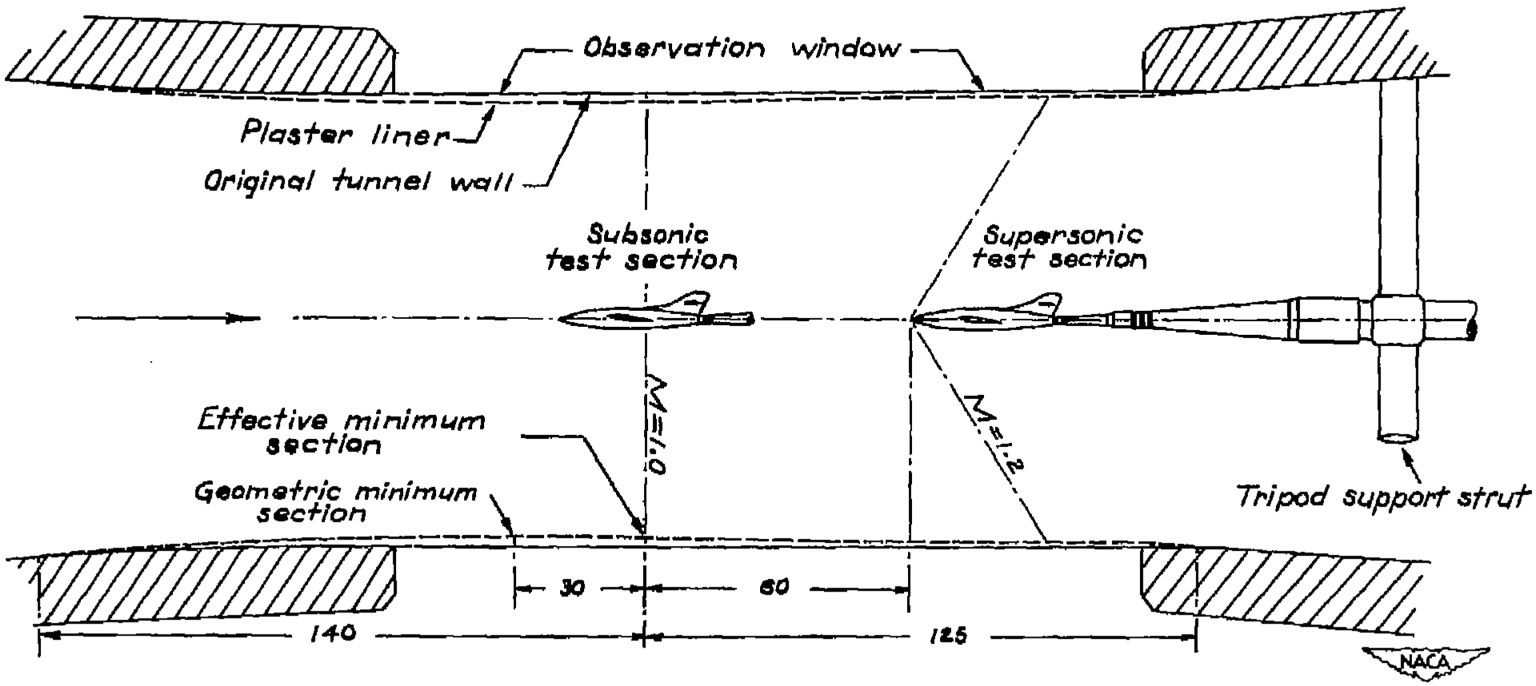


Figure 1.- Approximate shape and dimensions of and tentative test sections in the Mach number 1.2 temporary nozzle installed as a liner in the Langley 8-foot high-speed tunnel. All dimensions are given in inches.

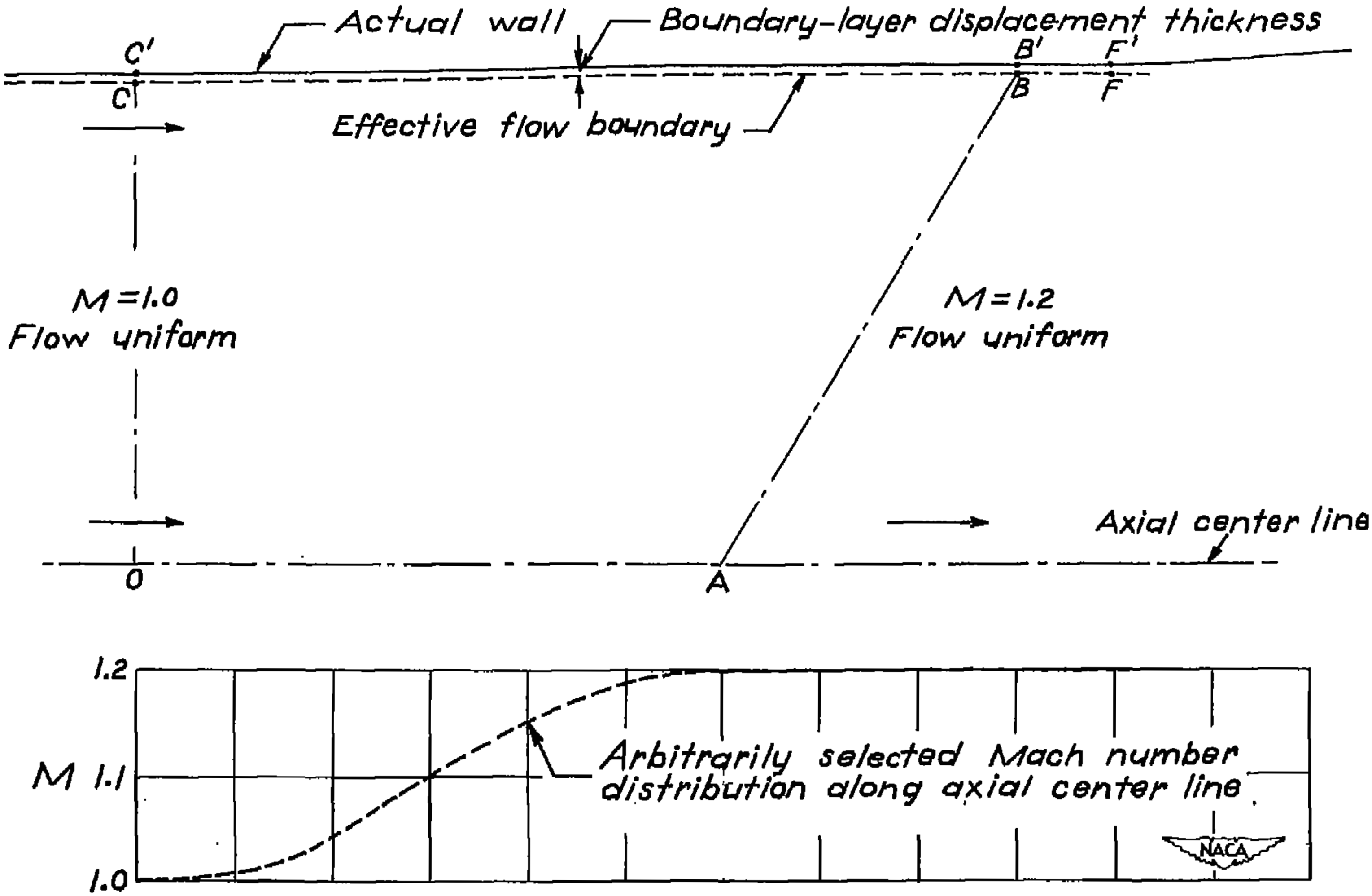


Figure 2.- Diagrammatic sketch illustrating flow conditions assumed for calculations of velocity field and wall shape of divergent portion of nozzle.

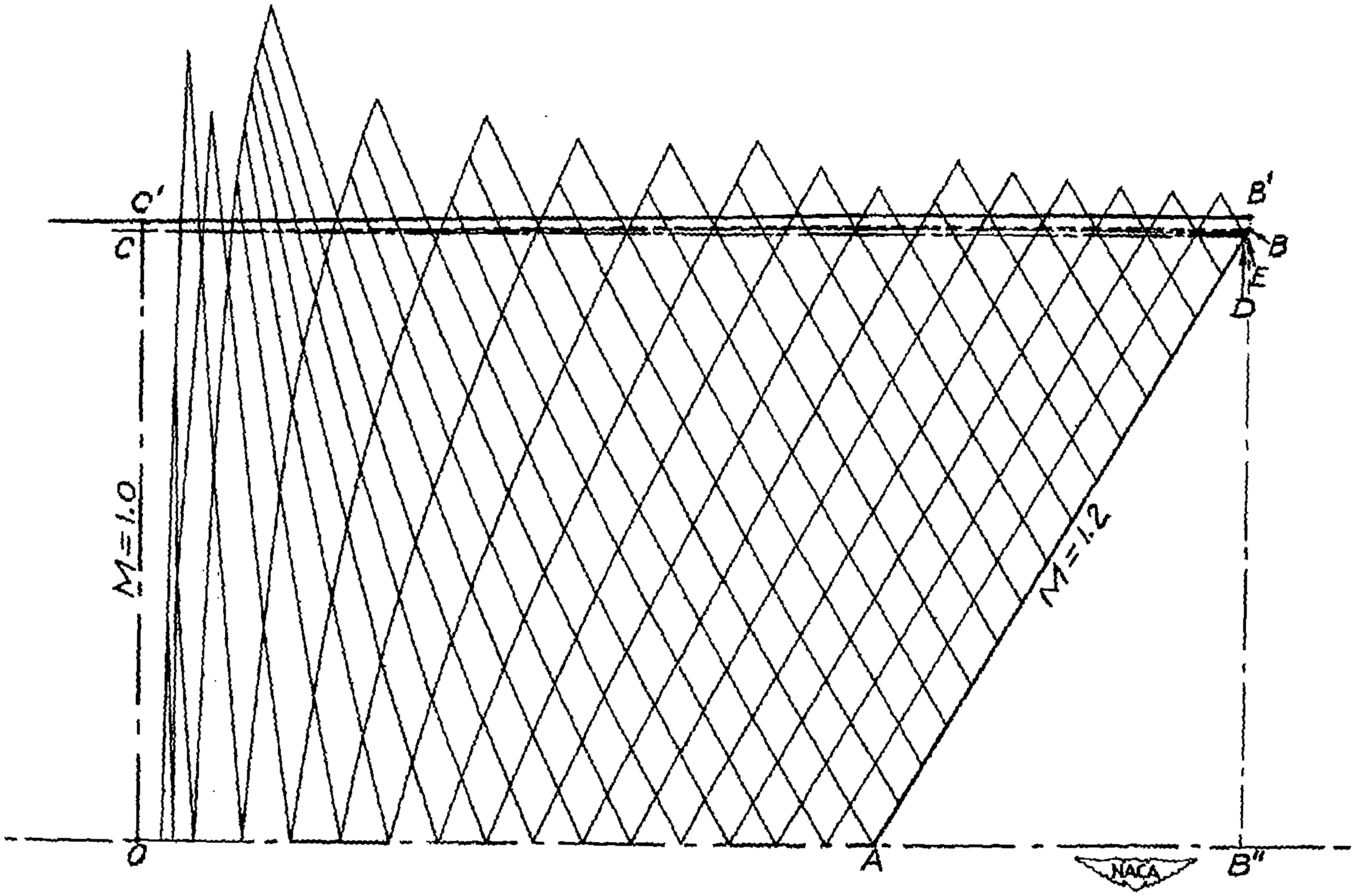


Figure 3.- Characteristic network calculated for the supersonic expansion in the divergent portion of the nozzle.

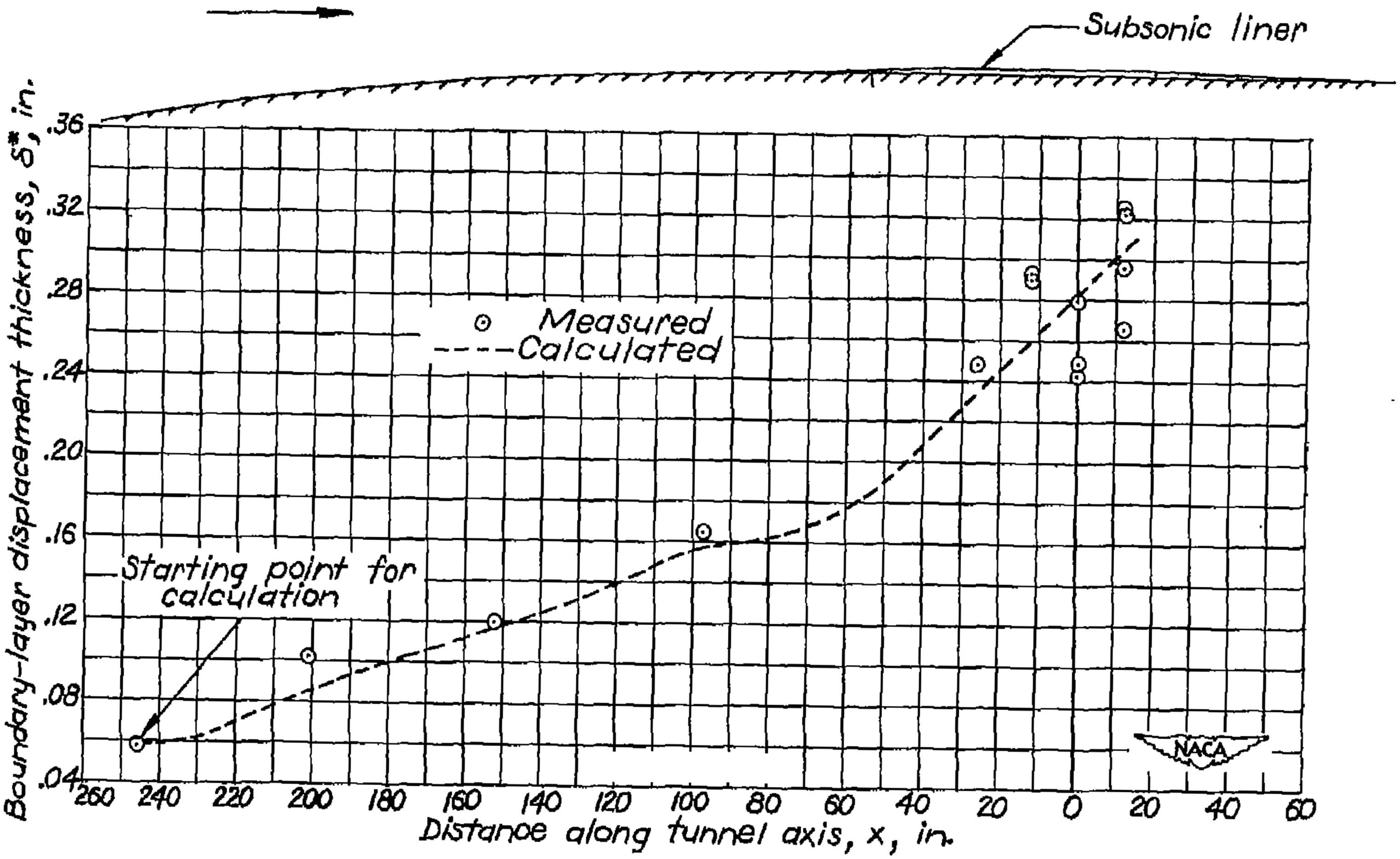
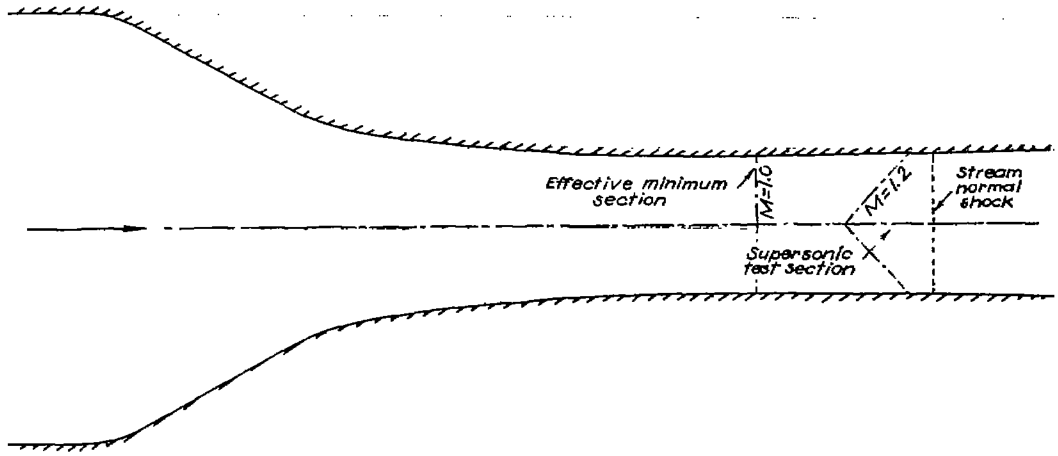


Figure 4.- Comparison of calculated and measured boundary-layer displacement thicknesses at the tunnel wall with subsonic liner installed in the throat region. Mach number in liner test section, 0.85.



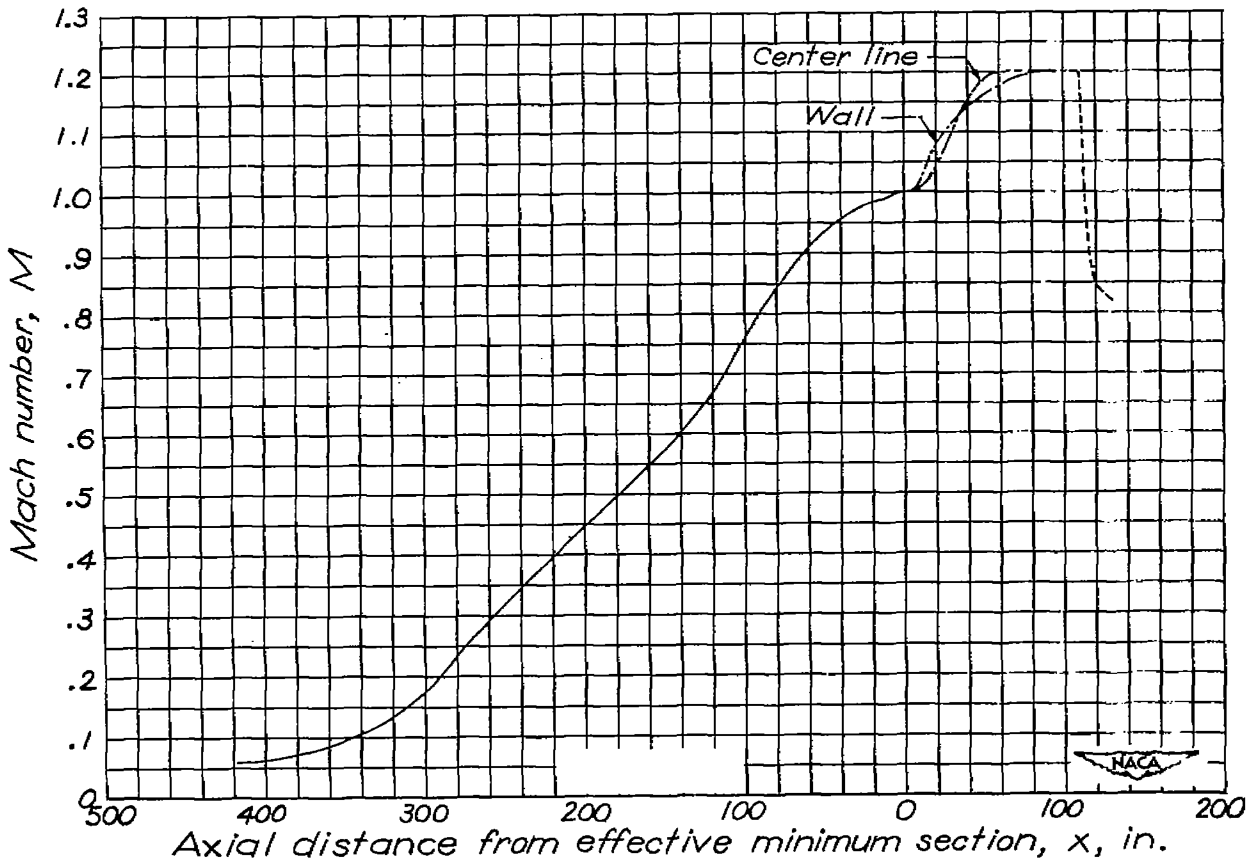


Figure 5.- Calculated distribution of Mach number axially along the entire length of the contraction cone and nozzle.

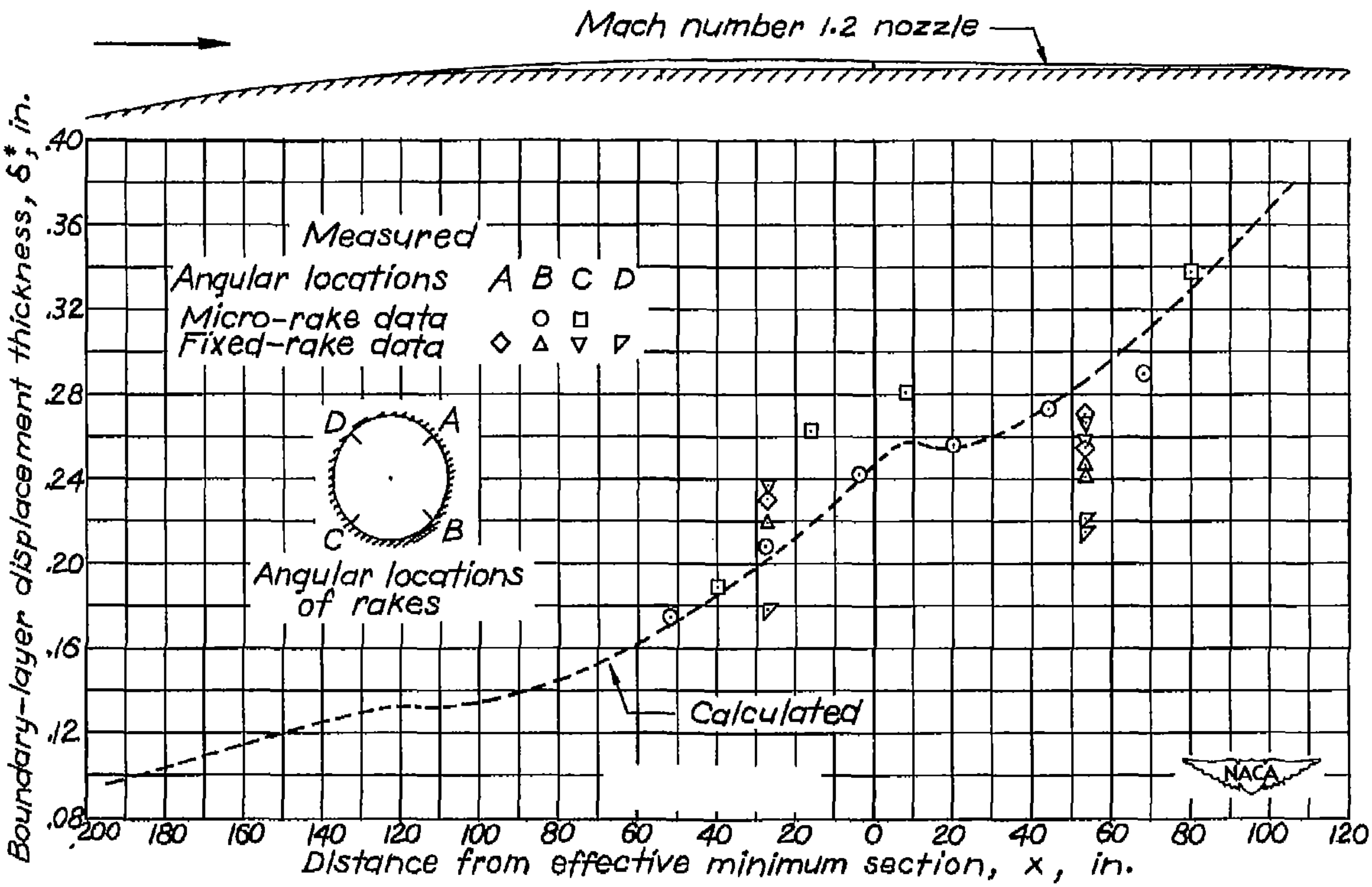


Figure 6.- Comparison of calculated and measured boundary-layer displacement thicknesses axially along the nozzle wall.

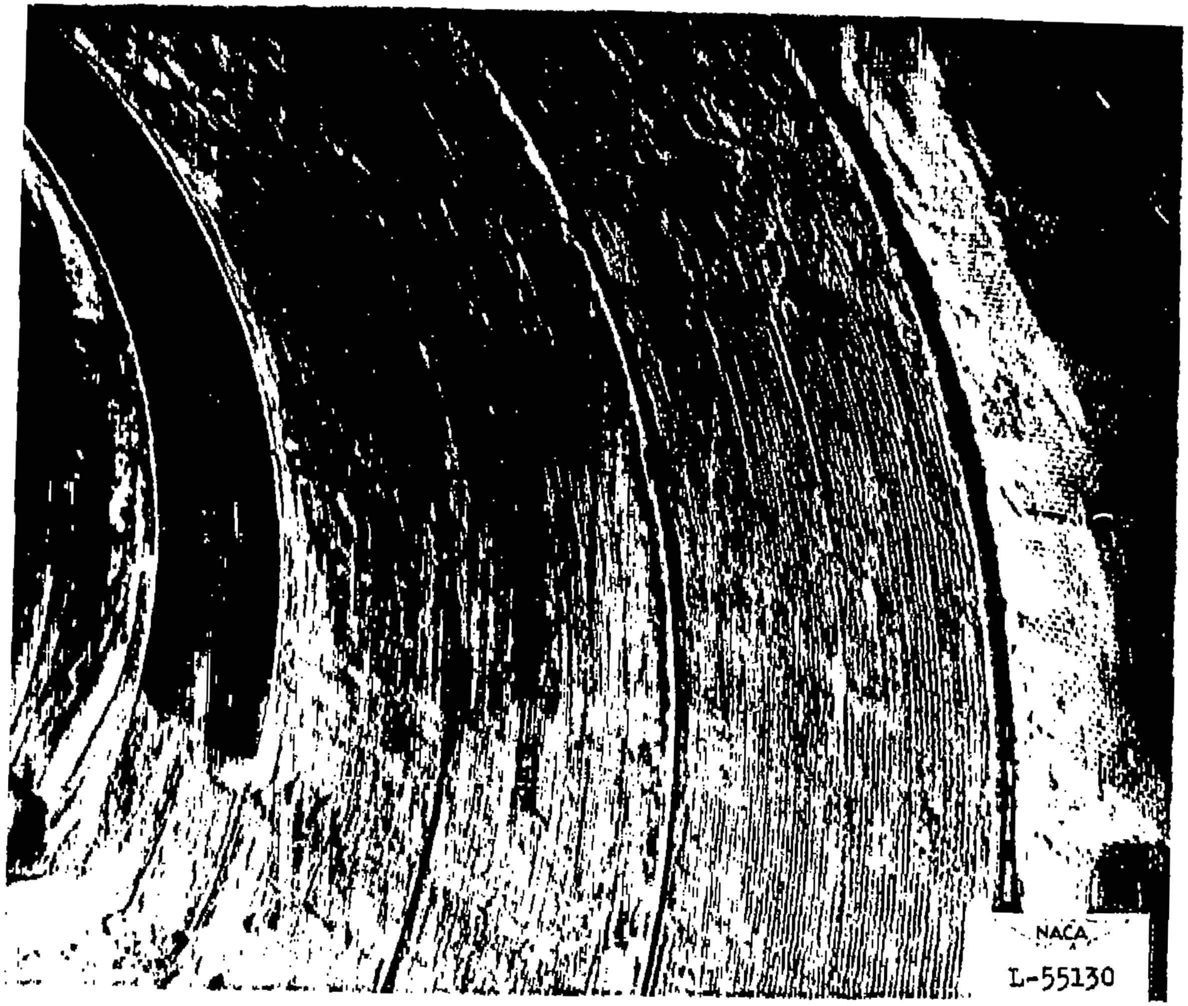


Figure 7.- Detail view showing stages of the plaster application.

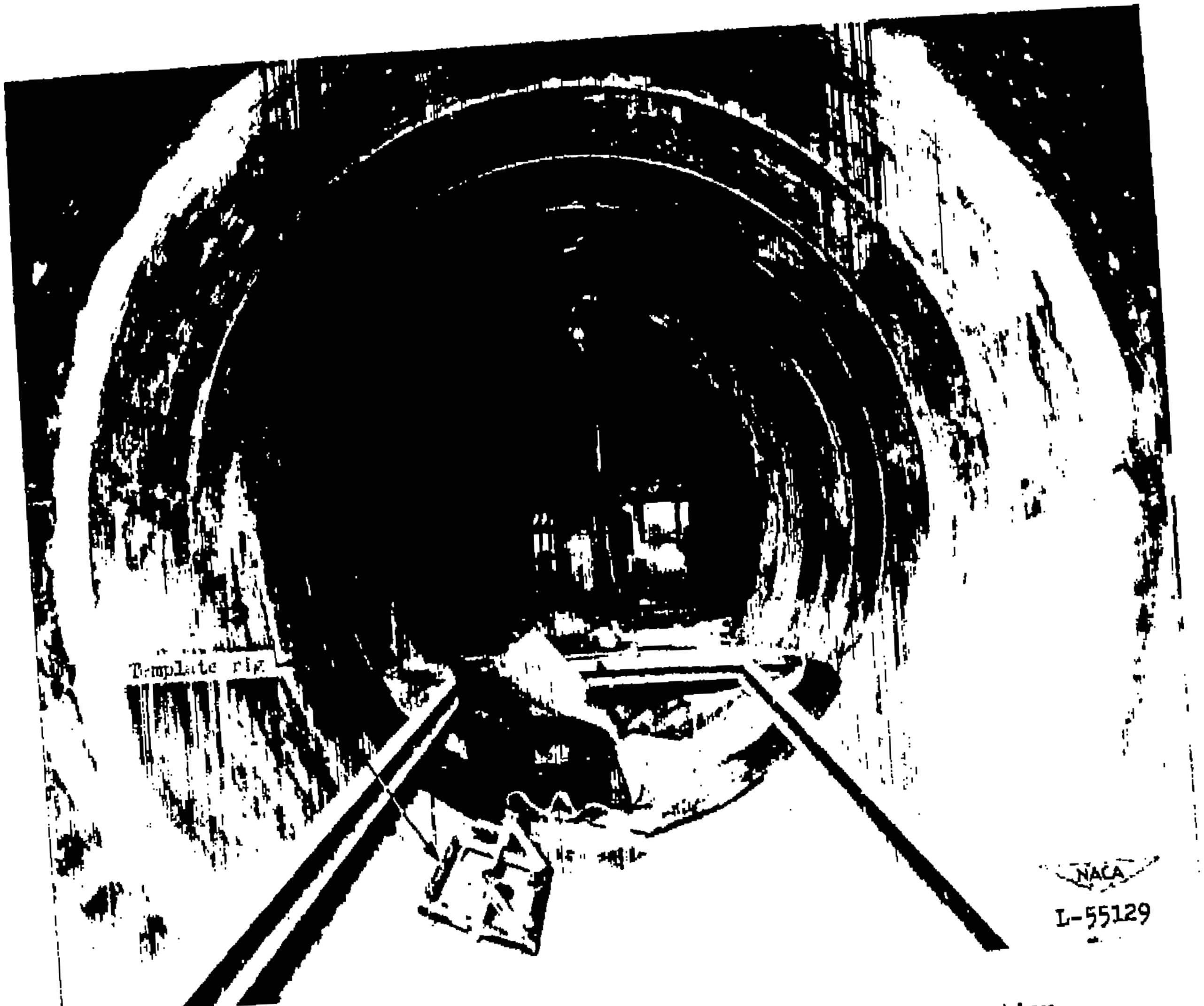


Figure 8.- View of plaster liner in process of construction.

Figure 9.- The completed temporary plaster nozzle for Mach number 1.2, as viewed from downstream.

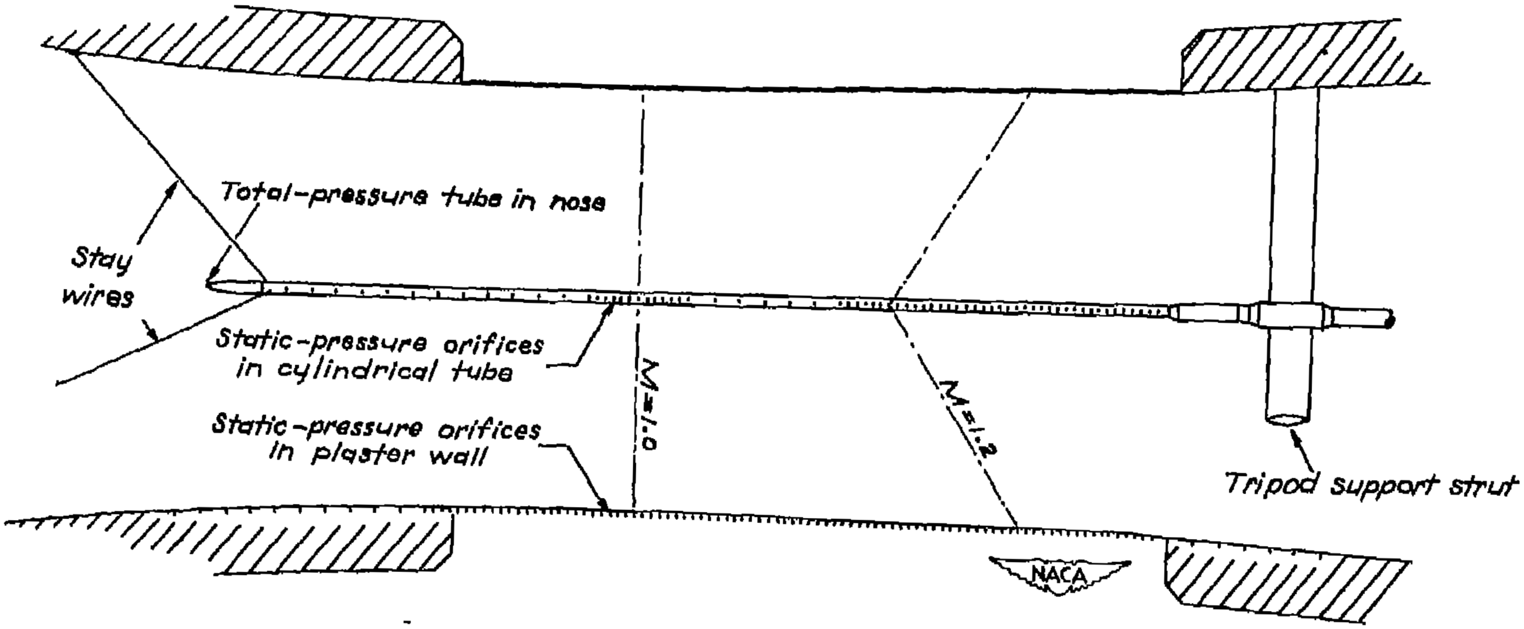


Figure 10.- Arrangement for measuring total and static pressures near the axial center line and static pressures at the wall of the nozzle.

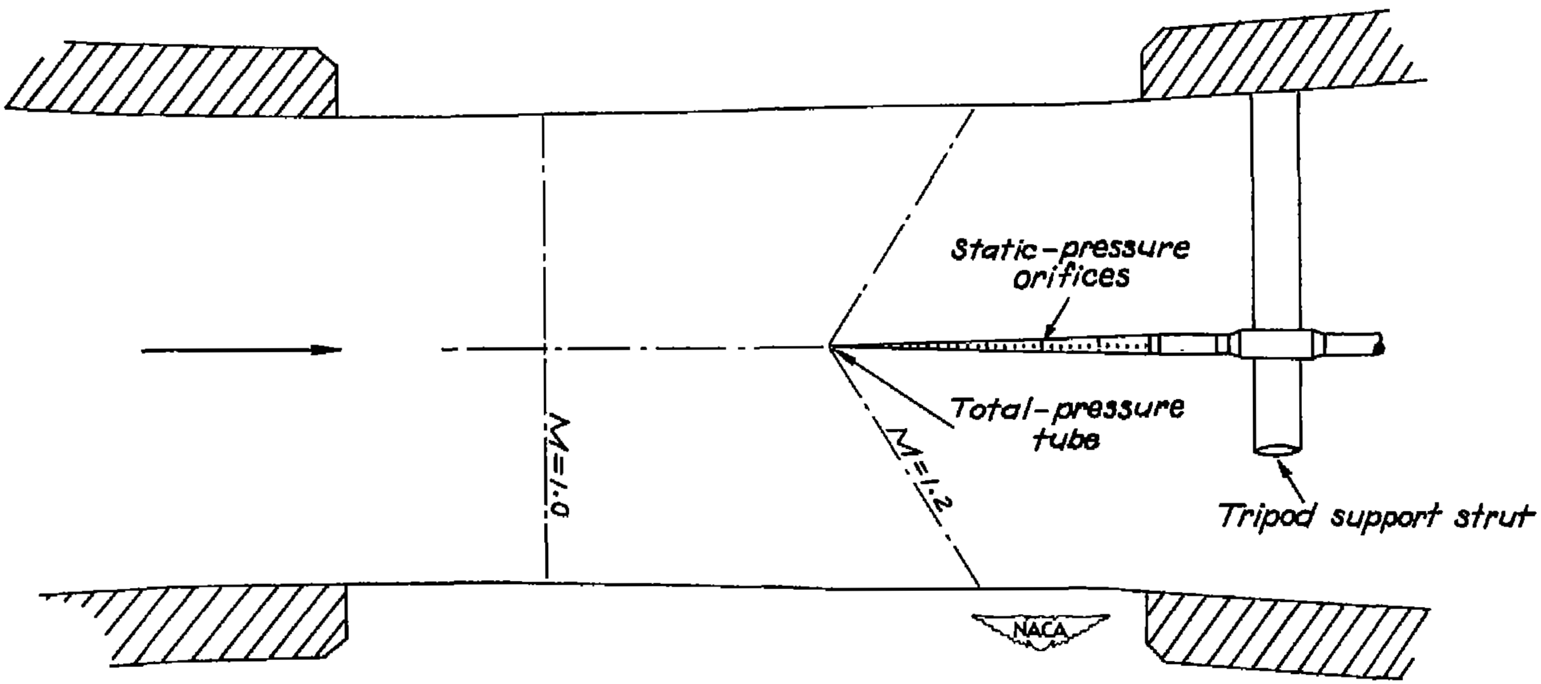


Figure 11.- Cone (3° included angle) for measuring total and static pressures along the axial center line of the supersonic test section.

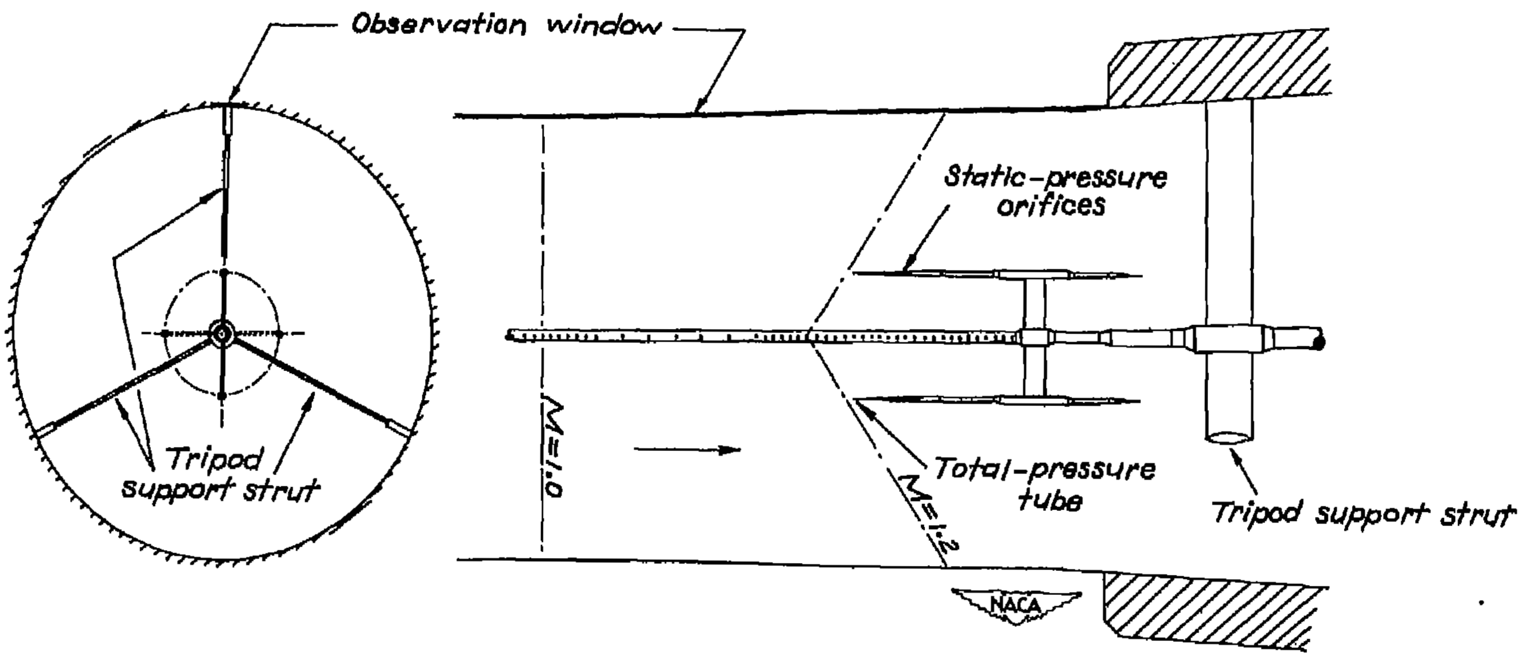


Figure 12.- Arrangement of 3° included-angle cones for measuring total and static pressures off the axial center line of the supersonic test section.

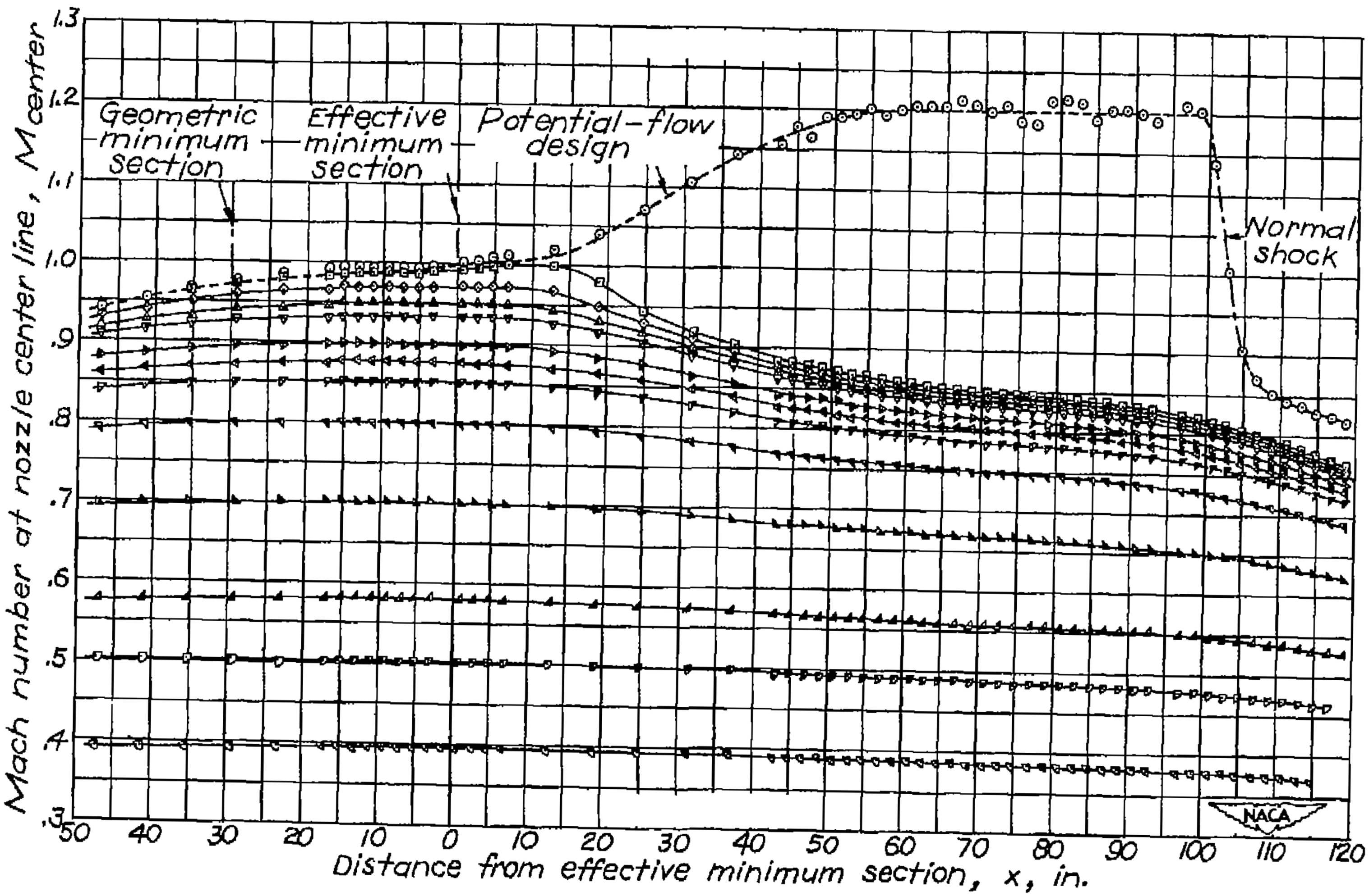


Figure 13.- Mach number distributions obtained from static-pressure measurements (at surface of cylindrical tube) axially along the nozzle center line for subsonic and supersonic flows.

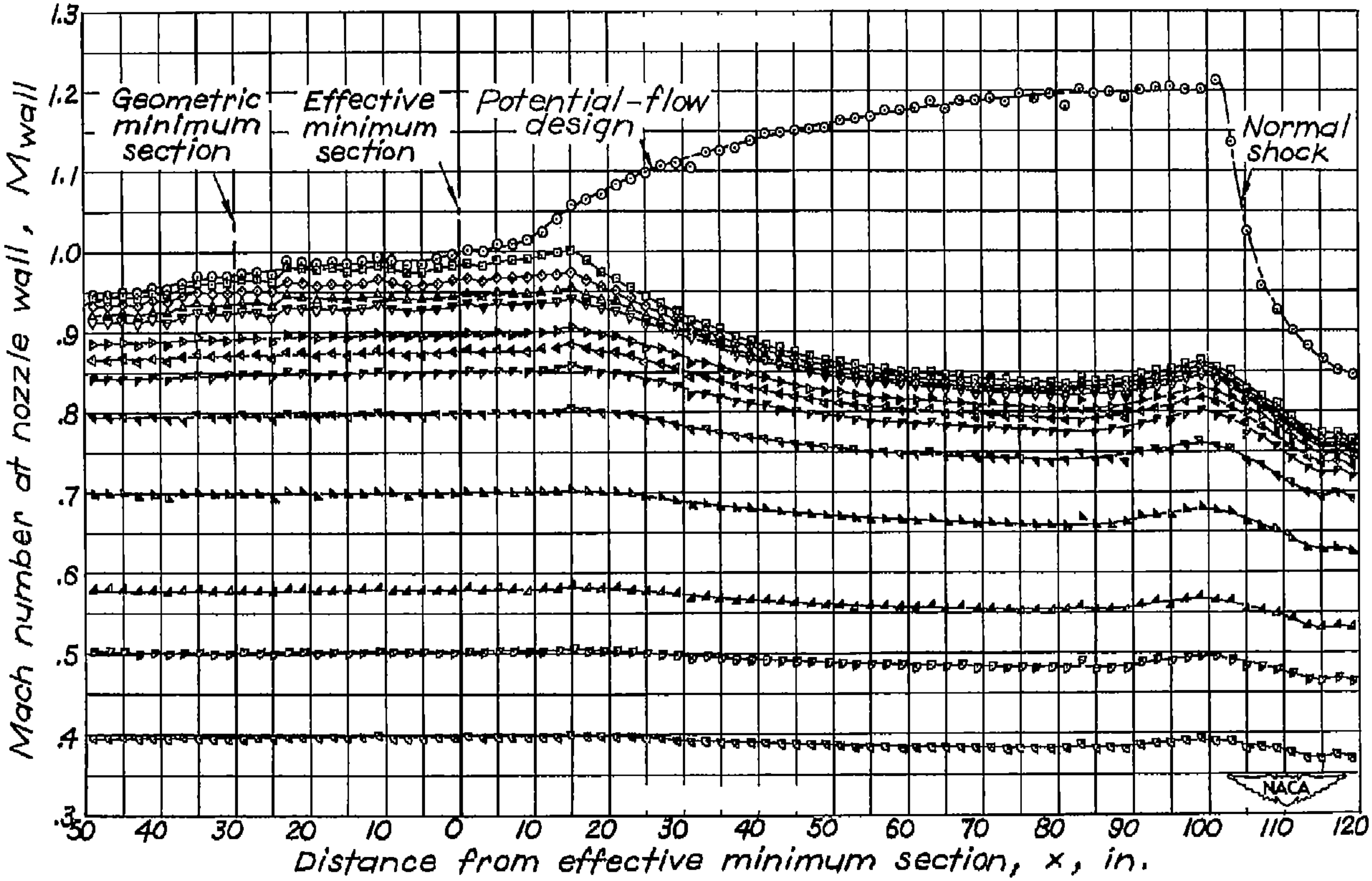


Figure 14.- Mach number distributions obtained from static-pressure measurements axially along the nozzle wall for subsonic and supersonic flows.

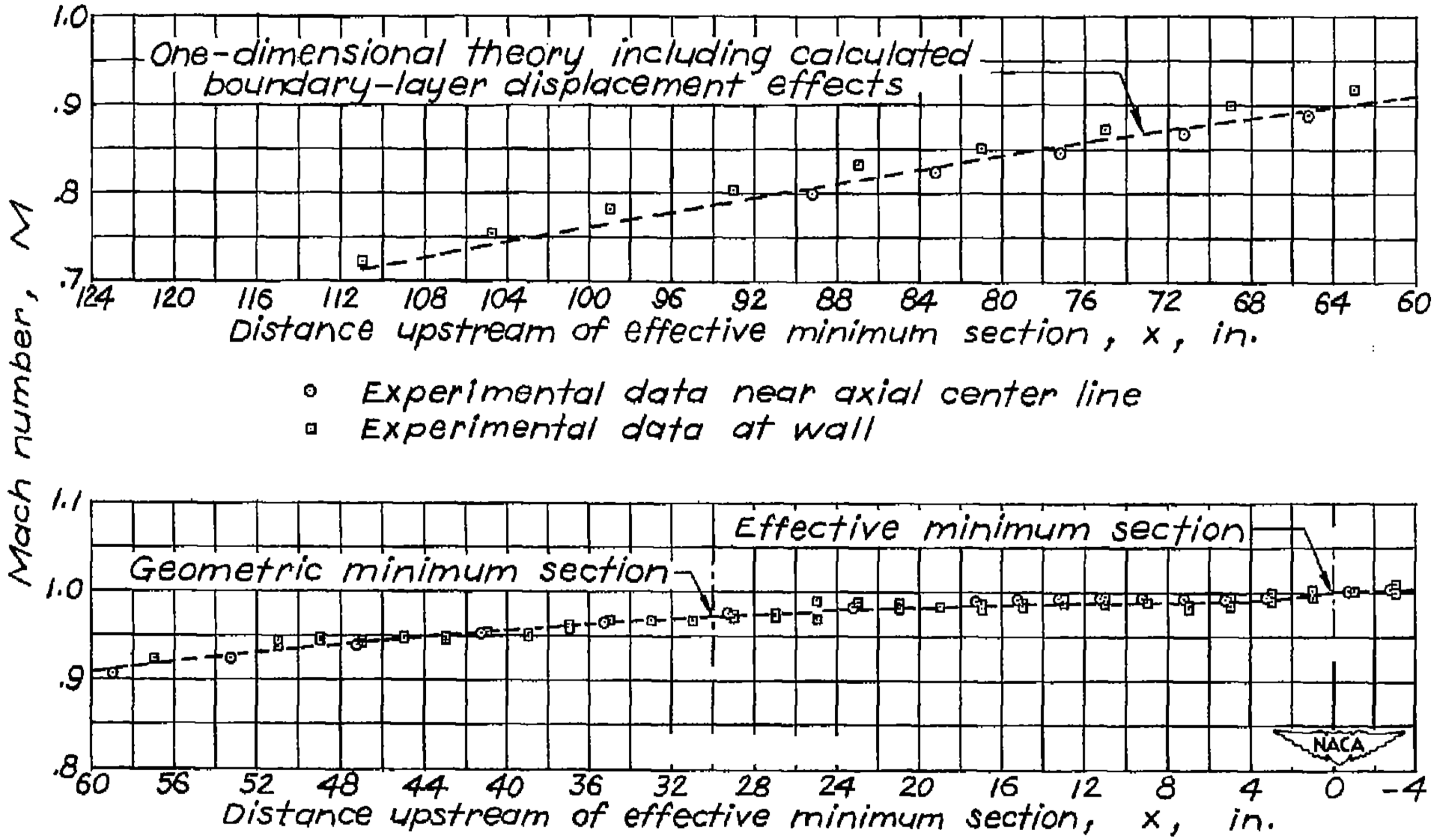


Figure 15.- Comparison of experimental Mach number distributions axially along the wall and near the center line of the contraction cone with the potential-flow theory.

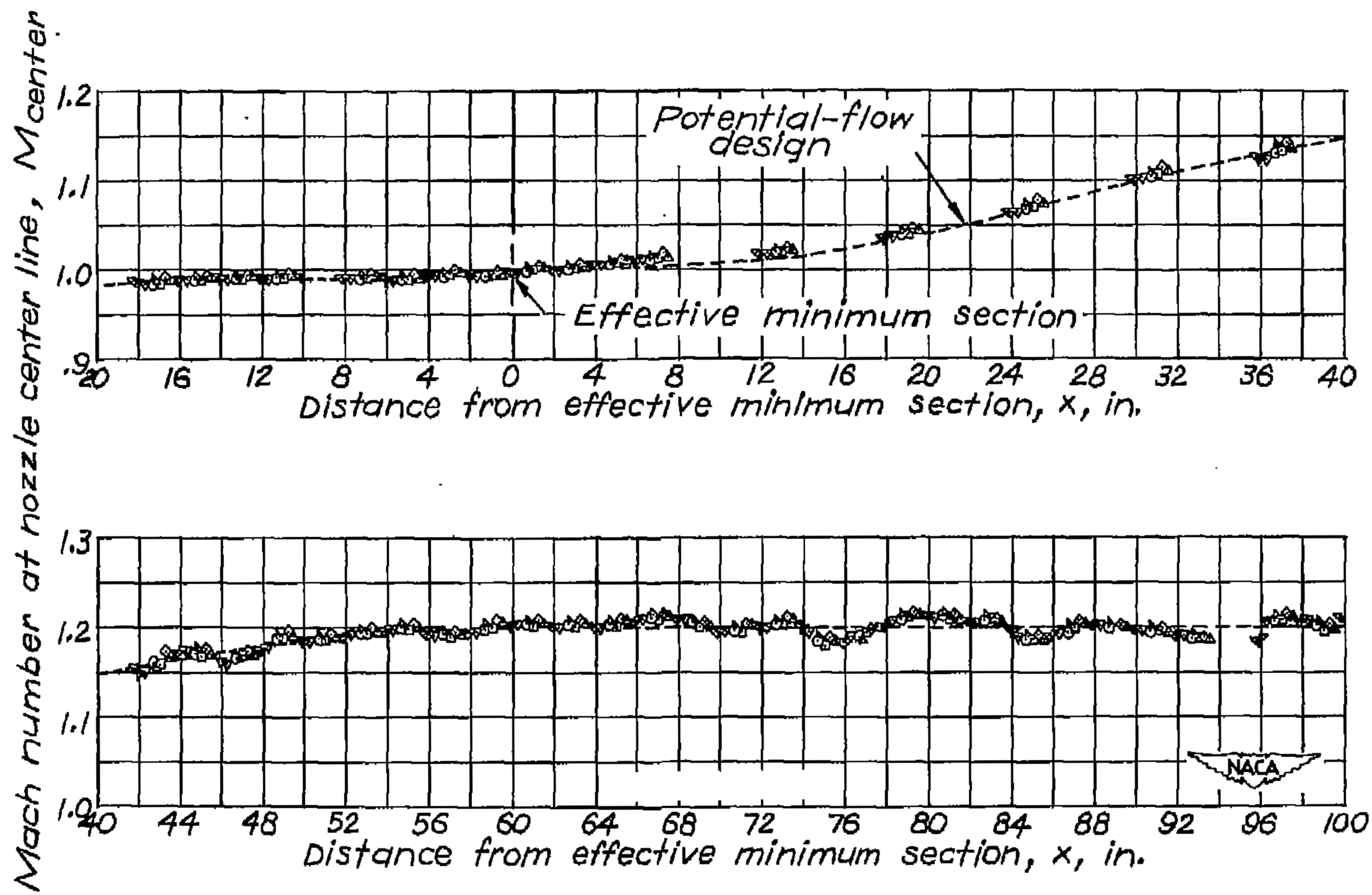


Figure 16.- Comparison of theoretical Mach number distribution axially along the nozzle center line with experimental values obtained at close axial intervals from eight successive runs.

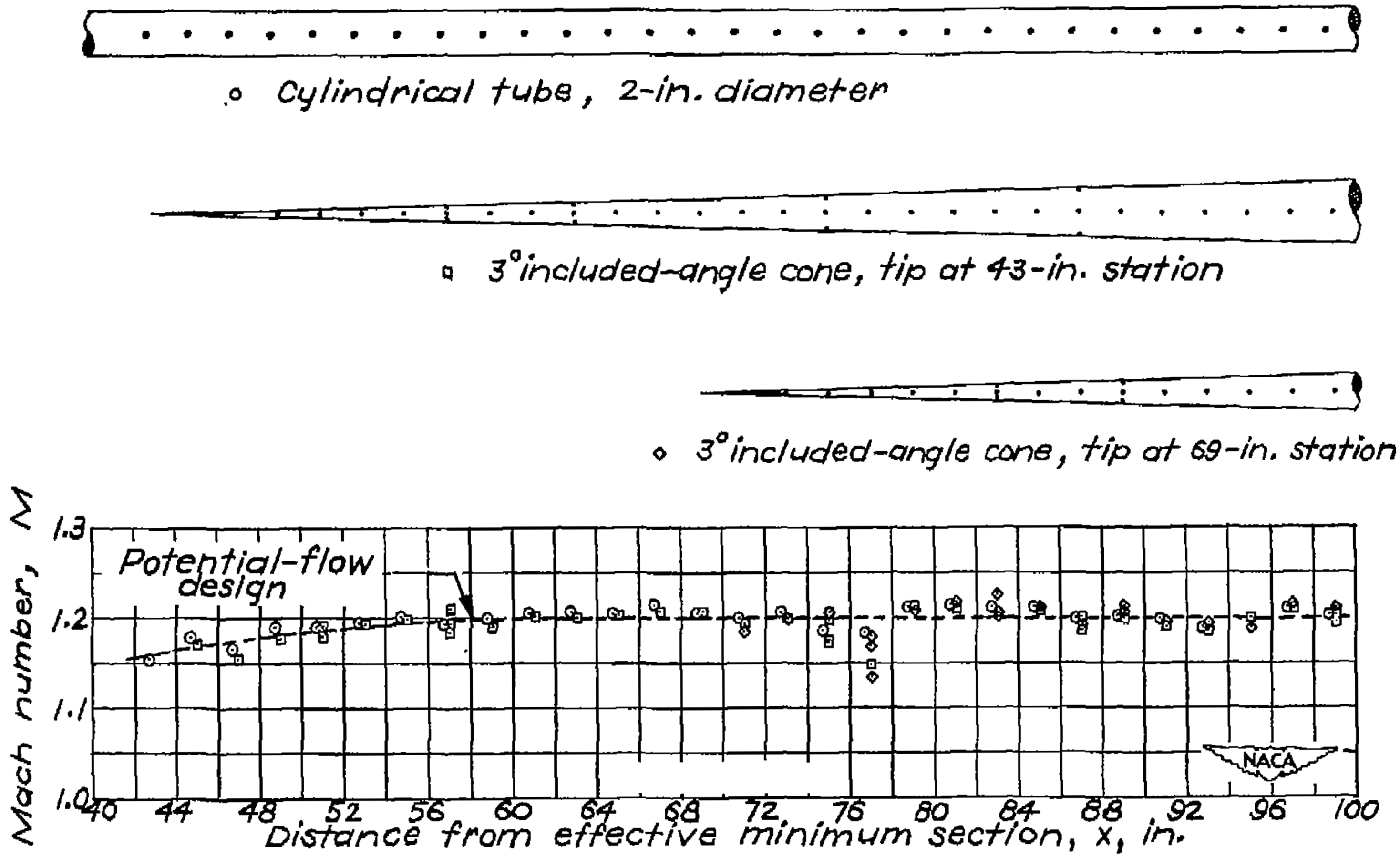


Figure 17.- Comparison of Mach number distributions obtained from cone and cylindrical-tube pressure measurements axially along the center line of the supersonic test section.

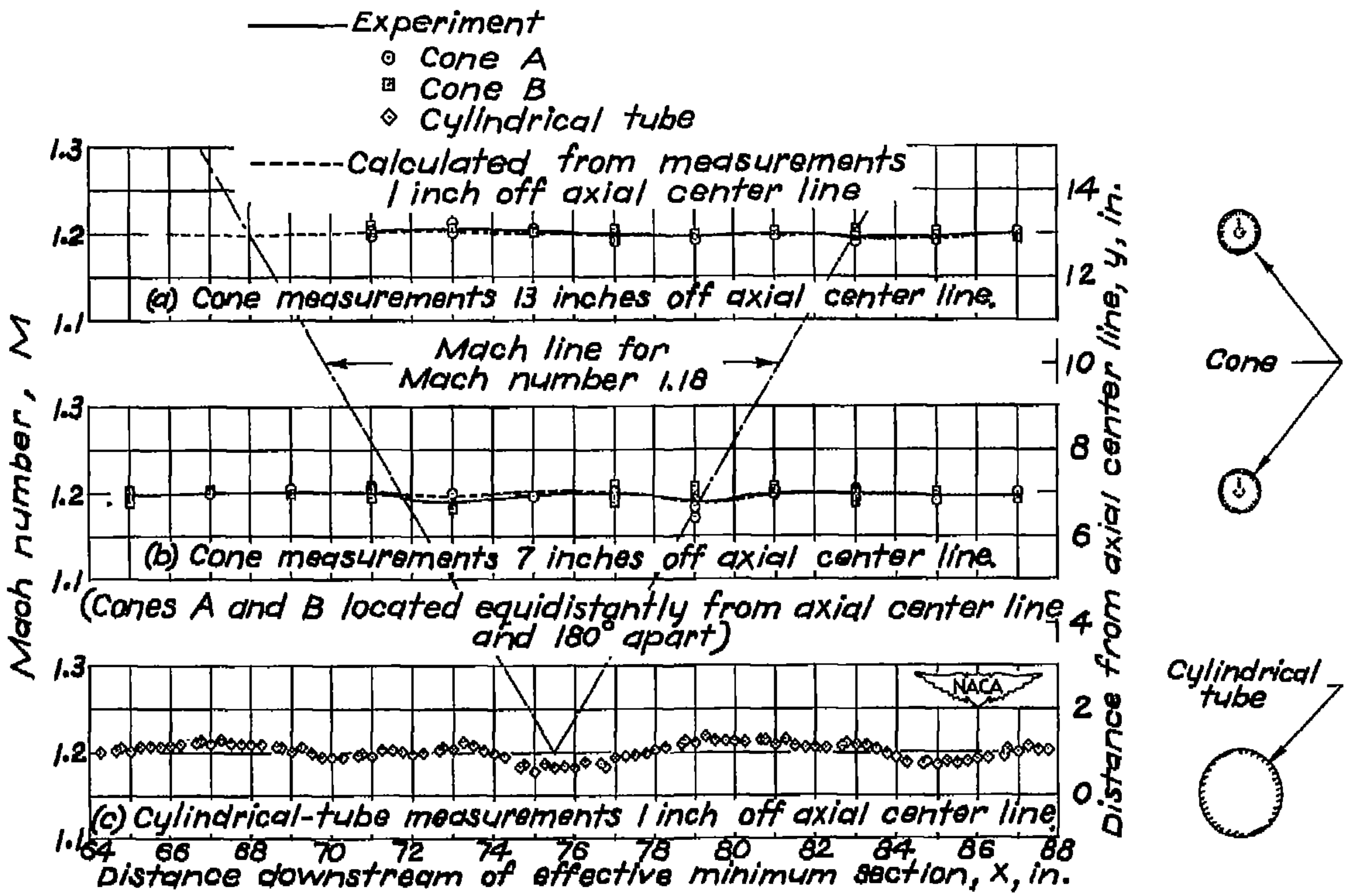


Figure 18.- Mach number distributions obtained from pressure measurements at approximately 1, 7, and 13 inches off the axial center line of the supersonic test section with a slight disturbance in the flow.

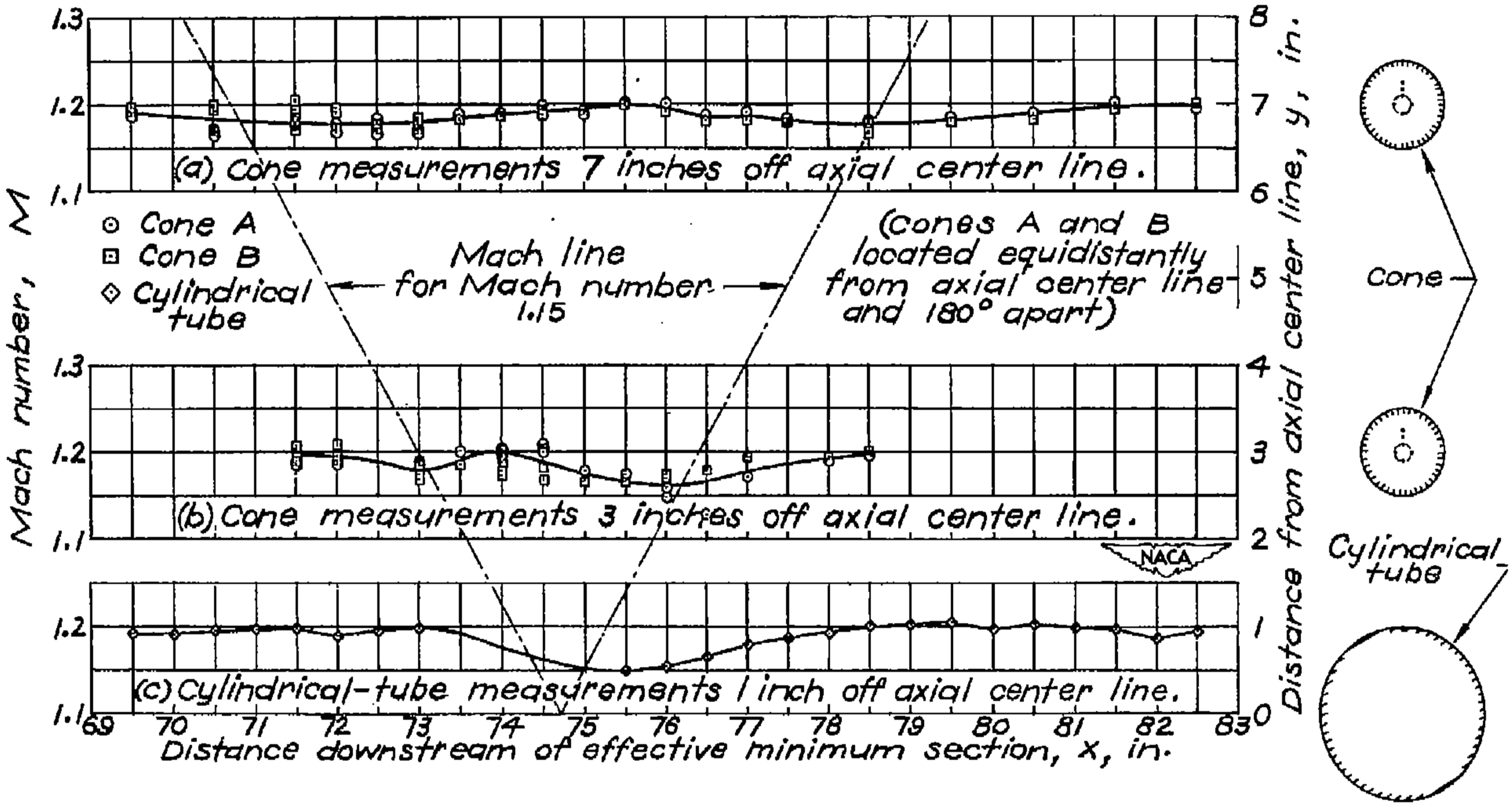


Figure 19.- Mach number distributions obtained from pressure measurements at approximately 1, 3, and 7 inches off the axial center line of the supersonic test section with a moderate disturbance in the flow.

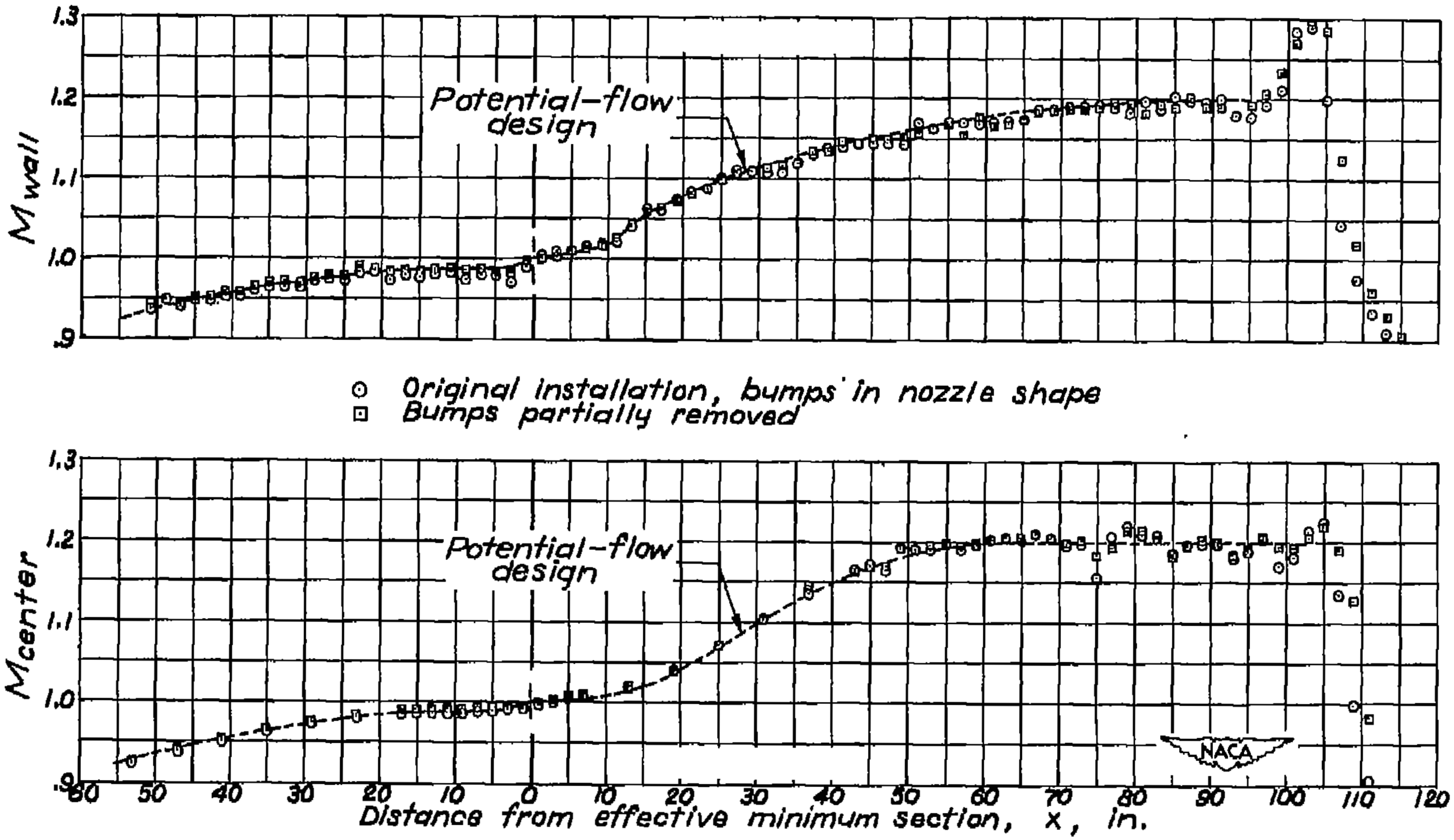


Figure 20.- Improvement of nozzle flow resulting from partial removal of bumps introduced in the nozzle shape during installation.

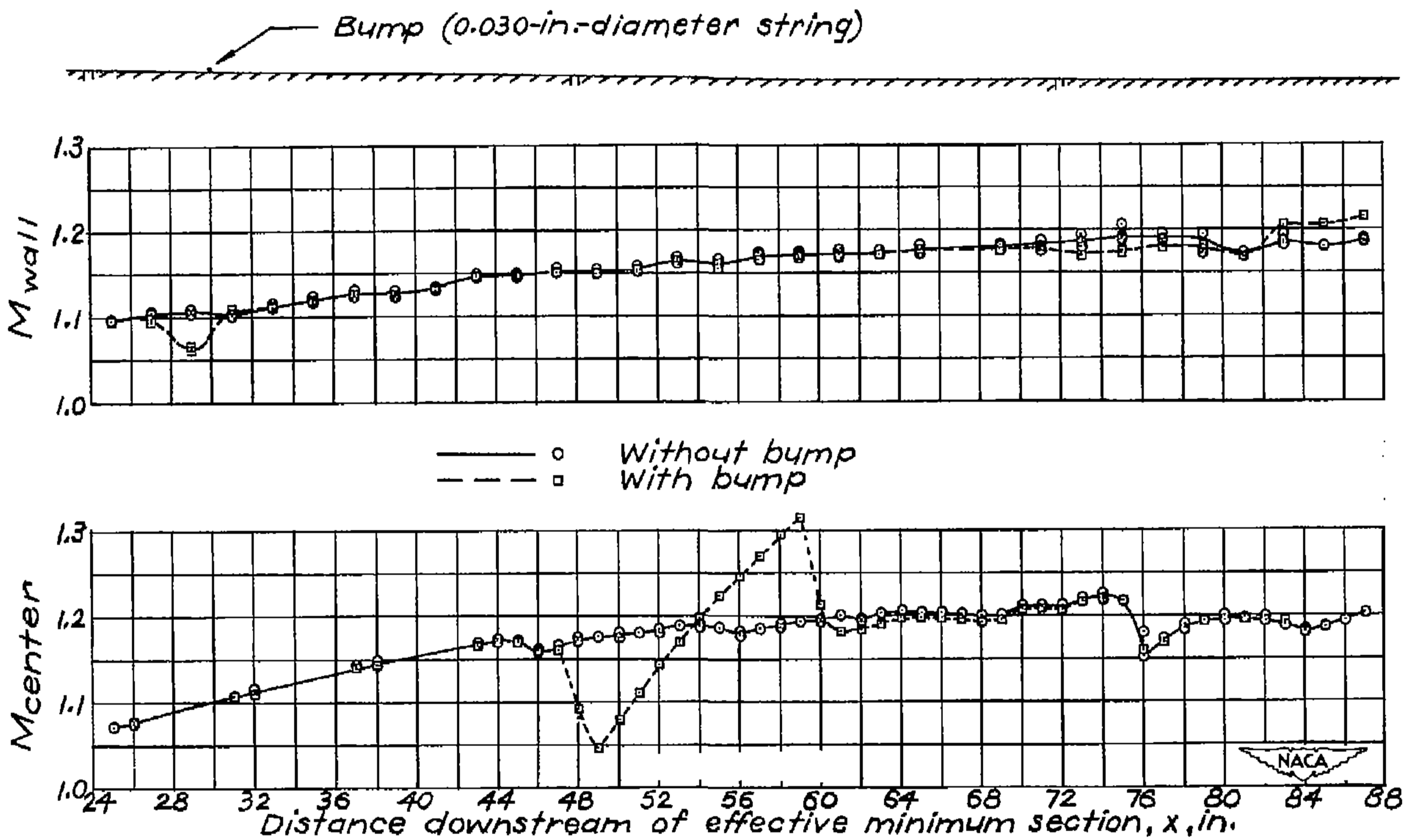


Figure 21.- Flow disturbance produced by an axisymmetrical bump (0.030-inch high) on the nozzle wall.

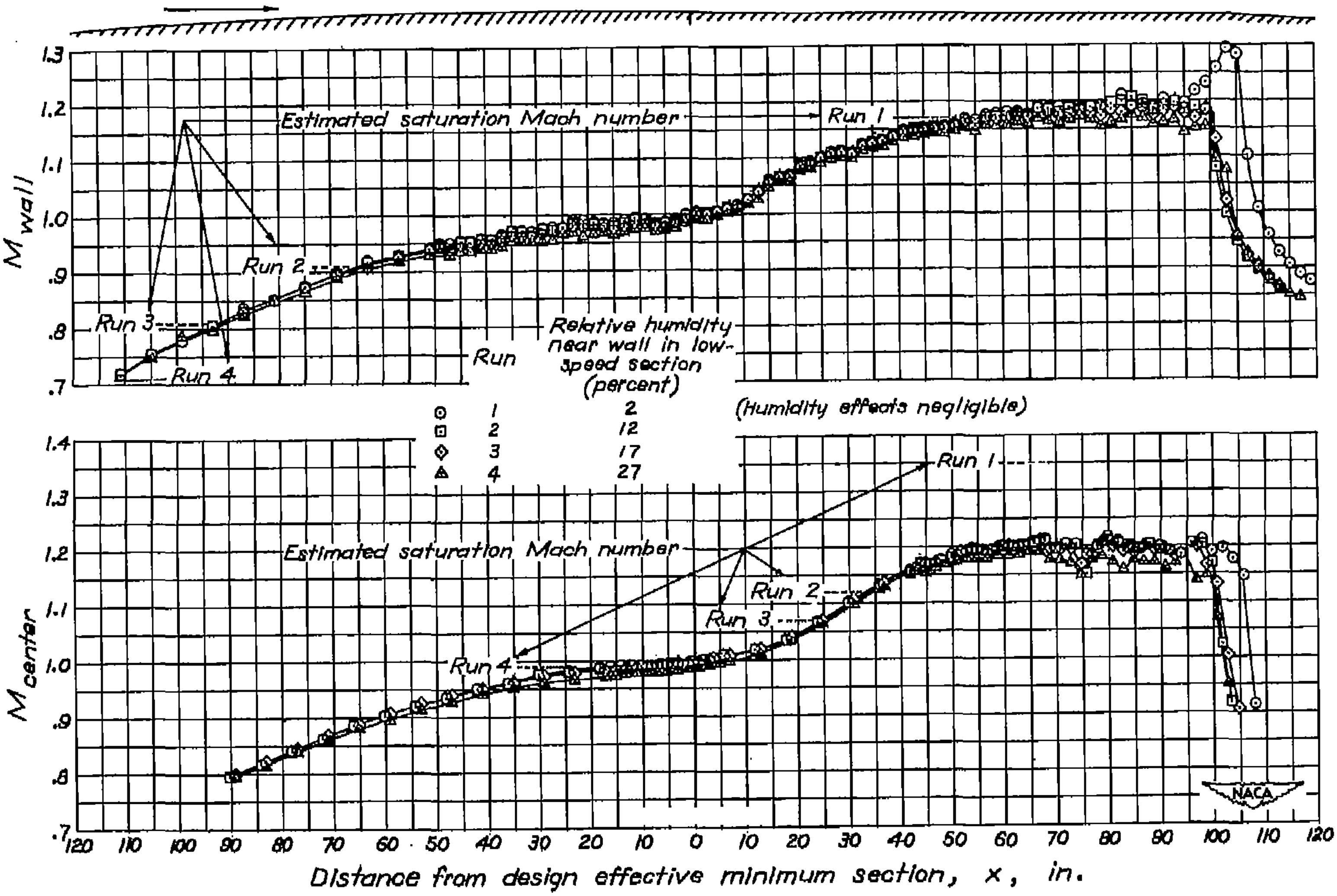


Figure 22.- Introduction of humidity effects in the Mach number distribution by insufficient heating of the flow mixture.

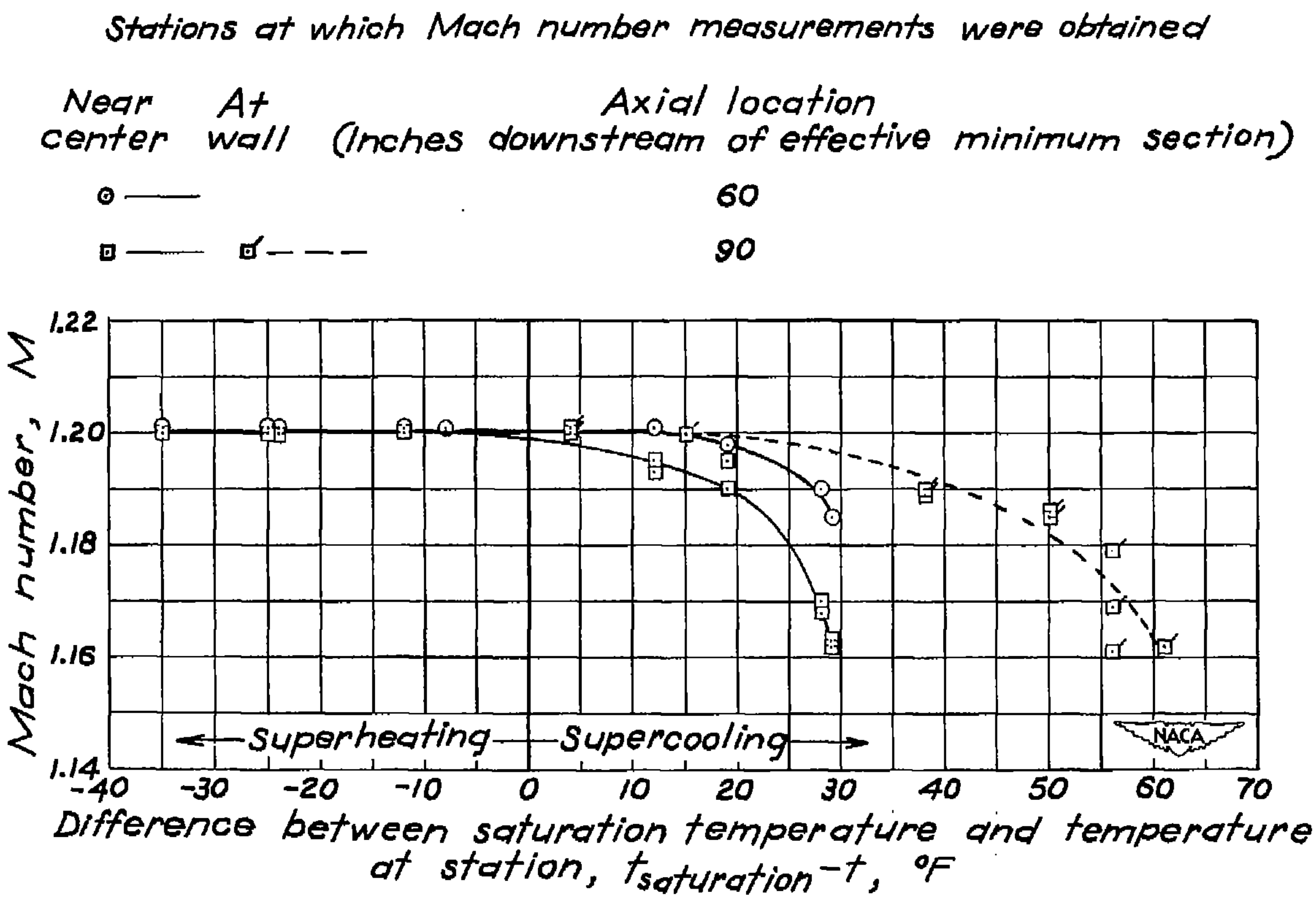


Figure 23.- Effect of supercooling on indicated Mach numbers at the wall and near the center line of the supersonic test section.

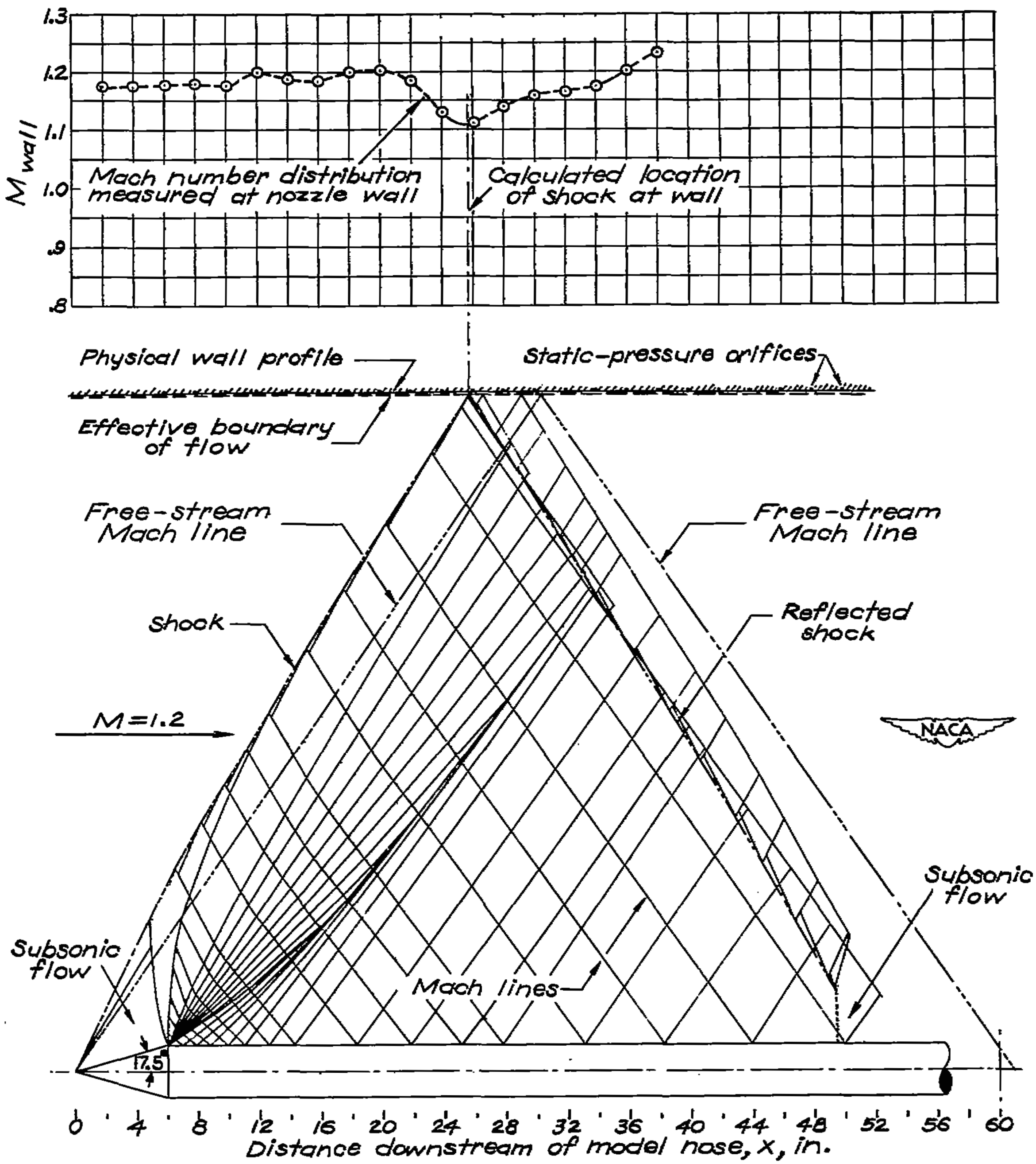


Figure 24.- Calculated flow pattern about a cone of revolution with cylindrical afterbody and comparison of calculated and measured shock locations at the nozzle wall.

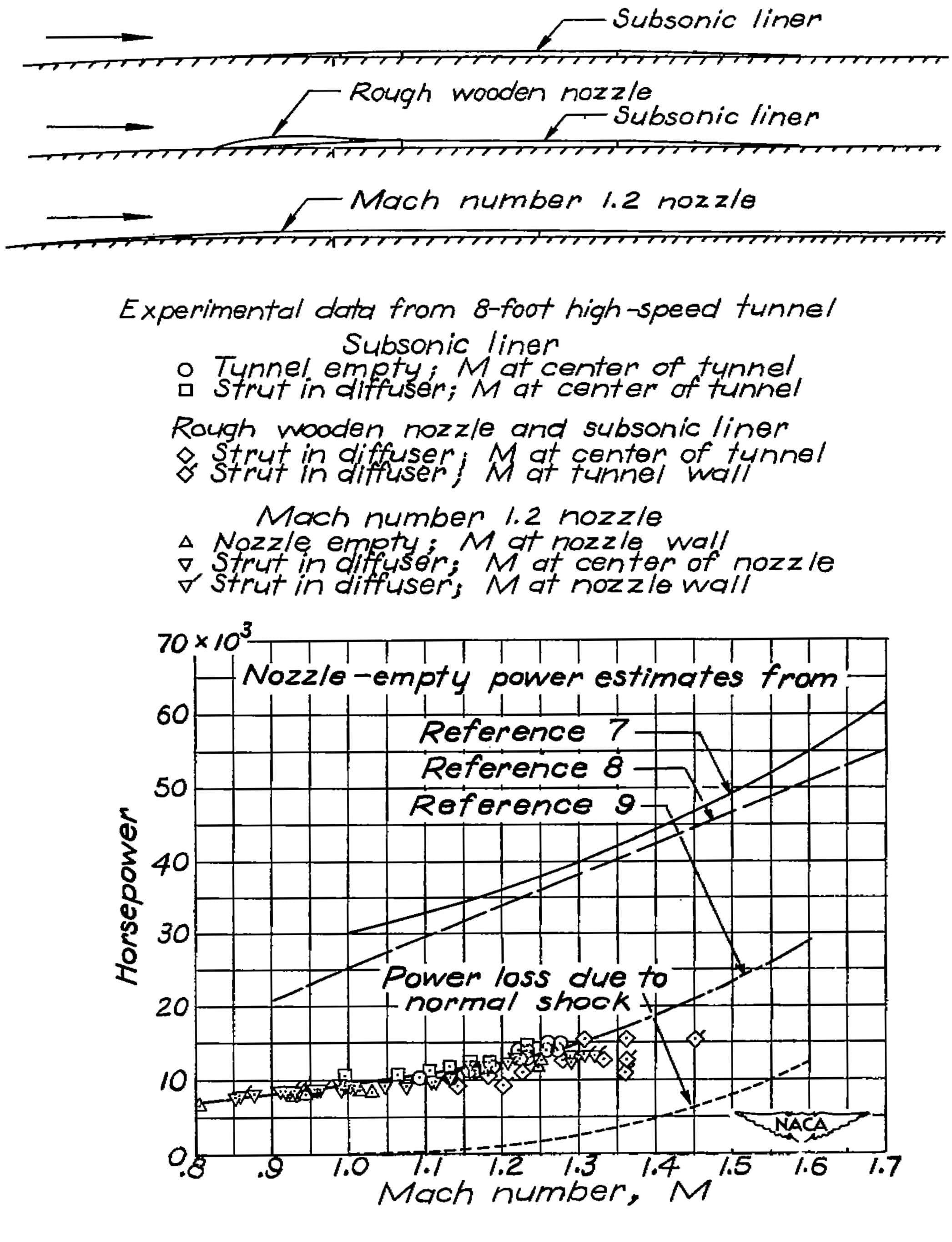


Figure 25.- Power required for transonic operation of the Langley 8-foot high-speed tunnel.