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# RESEARCH MEMORANDUM

FLOW STUDIES IN THE ASYMMETRIC ADJUSTABLE NOZZLE OF  
THE AMES 6- BY 6-FOOT SUPERSONIC WIND TUNNEL

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## NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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## SUMMARY

Surveys of the flow in the test section of the asymmetric adjustable nozzle of the Ames 6- by 6-foot supersonic wind tunnel have been made to determine the uniformity of the air stream. The results of the surveys show only small variations of stream pressure and direction at a nominal Mach number of 1.4. As the Mach number is increased or decreased from a value of 1.4, however, vertical pressure gradients of significant magnitude are found. Smaller axial gradients also exist. There are no transverse gradients of appreciable magnitude.

Test techniques for minimizing the effects of stream irregularities are discussed and the results of force and pressure-distribution tests of a swept-wing model are presented to illustrate the effectiveness of these techniques.

## INTRODUCTION

In reference 1, H. J. Allen has shown that the continuous adjustment of the Mach number of the flow in the test section of a supersonic wind tunnel may be accomplished by the use of an asymmetric adjustable nozzle. The advantages of this scheme, in contrast with nozzles with fixed dimensions or those with flexible walls, in increasing the utility of a supersonic wind tunnel are apparent. One disadvantage of such a nozzle, however, is of special interest to an experimenter, namely, that, with the present technique of nozzle design, it has not been found possible to obtain a uniform stream at all Mach numbers. In view of the increase in the usefulness of the wind tunnel, however, some deviations in stream flow may be tolerable if their influence on the test results can be accurately determined or can be shown to be small.

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Speed control by use of the asymmetric adjustable nozzle was incorporated into the design of the Ames 6- by 6-foot supersonic wind tunnel. This facility, which was planned in 1944-1945, was put into operation in the summer of 1948. Extensive surveys of the air stream of this wind tunnel have been made to determine the characteristics of the air stream both for the purpose of providing essential information needed for the interpretation of test results and for use in the further development of nozzle design techniques. The results of these surveys are given herein together with the results of some model tests which show how the effects of stream irregularities may be minimized.

#### GENERAL CHARACTERISTICS OF THE WIND TUNNEL

The Ames 6- by 6-foot supersonic wind tunnel (fig. 1) is a closed-return, variable-pressure, supersonic wind tunnel with a 6-foot-square test section. The asymmetric adjustable nozzle used in this wind tunnel permits the Mach number of the flow in the test section to be varied continuously from  $M = 1.1$  to  $M = 2.0$ . The ordinates of the nozzle blocks are given in table I. Separate origins are given for both the upper and lower nozzle blocks. The results of pressure and angle surveys presented later will be given for certain values of the axial displacement of the lower block from the upper which is designated as  $D$ , as noted in the sketch on table I.

The tunnel is powered with two 25,000-horsepower wound-rotor-induction motors solid-coupled in tandem to an extension shaft which drives an eight-stage axial-flow compressor. Synchronous speed for the motors is 900 rpm. A slip regulator permits the speed to be lowered to 775 rpm. The drive power is sufficient to permit the attainment of a maximum stagnation pressure of about 17 pounds per square inch absolute at the lower Mach numbers. A stagnation pressure of 2 pounds per square inch is the minimum value attainable.

The wind tunnel is a sealed pressure vessel and is equipped with air drying equipment of sufficient capacity to permit the absolute humidity of the air to be maintained at a value of less than 0.0003 pound of water per pound of air. The deleterious effects of moisture condensation on the uniformity of the flow in the test section are thereby avoided.

The temperature of the air stream may be held at a maximum stagnation temperature of  $560^{\circ}$  Rankine. Temperature control is obtained by use of finned cooling coils which are located in the tunnel just downstream of the compressor.

## SYMBOLS

- x horizontal distance in the axial direction from the vertical plane perpendicular to the test section axis and passing through the center line of the schlieren windows in the test section, positive downstream, inches
- y horizontal distance in the transverse direction from the vertical plane passing through the test-section axis, positive to the right as viewed from a downstream position, inches
- z vertical distance from the horizontal plane passing through the axis of the test section, positive above the test-section axis, inches
- $\delta$  distance from the nozzle wall, inches
- $\epsilon$  the angle that the tangent to any streamline makes with a horizontal plane, positive for upflow, degrees
- D axial displacement of the origin for the lower block of the nozzle from the origin of the upper block of the nozzle, inches
- H total pressure of the stream at any point within the boundary layer, pounds per square inch
- $H_0$  total pressure of the free stream, pounds per square inch
- S model wing area, square feet
- V stream velocity, feet per second
- $\rho$  stream mass density, slugs per cubic foot
- q dynamic pressure  $\left(\frac{1}{2}\rho V^2\right)$  pounds per square foot
- $\bar{c}$  mean aerodynamic chord of the model, feet
- $C_L$  lift coefficient  $\left(\frac{\text{lift}}{qS}\right)$
- $C_m$  moment coefficient  $\left(\frac{\text{moment}}{qSc}\right)$
- $C_D$  drag coefficient  $\left(\frac{\text{drag}}{qS}\right)$

- M Mach number  $\left(\frac{V}{a}\right)$ , where  $a$  is the local speed of sound
- $P_a$  loading coefficient, the change in the difference in pressure, in terms of the dynamic pressure, across the wing surface at any point in terms of the change of angle of attack, per degree
- $\alpha$  angle of attack of the model wing, degrees
- $\Delta\alpha$  the change in model angle of attack used in the calculation of the loading coefficient, degrees
- $\frac{\Delta P}{q}$  stream pressure coefficient, the difference between the pressure measured by the survey needle and the pressure measured by the arbitrary reference orifice in terms of the dynamic pressure

#### SURVEY APPARATUS

Pressure surveys of the test-section air stream were made with a power-driven pressure-survey apparatus (fig. 2) on which were mounted three static-pressure survey needles. These needles consisted of a 100-caliber ogival nose followed by a cylindrical afterbody 5/8 inch in diameter. Static pressure orifices 0.0135 inch in diameter were located in the needle at the axial position for which linear theory indicates that the pressure on the surface is essentially equal to that of the stream over the test range of Mach numbers. These needles were not calibrated, since no known standard exists, but it is believed that the needles read the true static pressure within  $\pm 1/2$  percent of the dynamic pressure.

The pressure-survey apparatus was so designed that surveys at any desired axial position in the test section could be made in concentric circles varying in radius by 3-inch increments up to a radius of 24 inches.

Stream-angle surveys were made with a small cone of  $30^\circ$  included angle. Pressure orifices of 0.0135-inch diameter were drilled into opposite sides of the cone in the plane in which the measurement of stream angle was desired. The cone was mounted on a bar of wedge-shaped section which in turn was mounted as a cantilever beam from a fitting on the wall of the tunnel (fig. 3). The end of the beam passed through the wall of the tunnel and was machined to fit a protractor.

The cone was calibrated by pitching the angle survey apparatus through a wide range of angles of attack at a point in the stream

both in an upright and an inverted position. A comparison of the curves of the difference in pressure across the pressure orifices in terms of the dynamic pressure as a function of the angle for the upright and inverted positions gives the angle deviation of the true zero of the cone from that of the arbitrary reference axis read by the protractor. Stream angles are then obtained by reading the inclination of the arbitrary reference for a null reading of the orifices and correcting by the calibration. The accuracy of measurement is estimated to be  $\pm 0.1^\circ$ . Calibration curves for the cone are shown in figure 4.<sup>1</sup>

Boundary-layer surveys on the curved walls of the tunnel were made at several points with rakes of total-head tubes located as shown in figure 5. These rakes were constructed so as to be very slender in the stream direction and were attached in such a way as to minimize any possible disturbance.

## RESULTS

### Pressure Survey

The results of the pressure surveys<sup>2</sup> of the test section of the wind tunnel are given in figures 6 to 8. The data obtained are presented as a difference between the pressure measured at an arbitrary reference orifice located in the side wall of the tunnel near the upstream end of the test section (at  $x = -28$ ) and the pressure measured with the survey needle. This pressure difference is given in terms of the stream dynamic pressure as calculated from the total head of the stream and the static pressure of the arbitrary reference orifice.

In order to reduce the mass of data accumulated to a size consistent with publication in a report without eliminating essential information, the results of axial surveys are presented for three vertical survey positions, at the center line and 18 inches above and below the center line, and vertical pressure variations are shown for three axial positions. Only a few cross-stream plots of

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<sup>1</sup>The Mach numbers given in figure 4 and subsequent figures are the average Mach numbers along the center line of the tunnel within the test section.

<sup>2</sup>No surveys were made for Mach numbers greater than 1.7, a limitation imposed at the present time by certain deficiencies of the model support system.

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the data are given to show that the variation in pressure across the stream is negligible. Data are given for only one stagnation pressure since it was found that the characteristics of the stream were not appreciably affected by the stagnation pressure in the permissible test range.

In all tests made for the purpose of obtaining these data, the normal shock wave in the wind tunnel was kept downstream of the model support in the diffuser of the wind tunnel. A constant check on the position of the normal shock wave was made by observing the distribution of pressure along the wall of the tunnel as measured by a number of orifices distributed along the horizontal center line from the test section into the diffuser section. Control over the position of the normal shock is maintained by controlling the compression ratio of the compressor by varying the motor speed between 775 and 860 rpm.

Mach number variations in the test section of the wind tunnel are shown in figures 9 and 10.

#### Angle Survey

The vertical pressure gradients observed in the air stream at some Mach numbers are not in themselves as significant as the streamwise variation in the stream angle which they imply. If the flow is two-dimensional, and the stream pressure variations are small and not discontinuous, the rate of change of stream angle with axial position can be related to the vertical pressure gradient as

$$\frac{d\epsilon}{dx} = -28.65 \frac{d(\Delta p/q)}{dz} \quad (1)$$

This equation permits a calculation only of the rate of change of the stream angle; the absolute magnitude must be determined by experiment. Calculated stream-angle variations agree quite well with the results of stream-angle surveys as is shown by the data of figure 11. In view of the agreement shown, the amount of stream-angle survey work done was reduced to those tests needed to establish at any one point the magnitude of the stream angle at any Mach number, as, for instance, in figure 11(a) for a Mach number of 1.23.

#### Boundary-Layer Survey

Surveys of the boundary layer were made at three positions on the nozzle walls as shown by the sketch in figure 5. The results

of these surveys are presented in figure 12 for use in any nozzle calculations which the reader may wish to make. The local stream pressure coefficient is given in the figure.

#### DISCUSSION

Examination of the pressure-survey data shows that the flow in the test section of the Ames 6- by 6-foot supersonic wind-tunnel nozzle is essentially uniform at a nominal Mach number of 1.4. As the Mach number is increased or decreased from a value of 1.4, however, vertical pressure gradients of considerable magnitude are found. Axial pressure gradients also exist but are of smaller magnitude. There are no appreciable transverse pressure gradients in the tunnel which indicates that the flow in the nozzle is two-dimensional, that is, there is no cross-stream flow within the test section. The surveys of stream angle also show that the flow is satisfactory at  $M = 1.4$ , and that appreciable variations in stream angle occur as the Mach number varies from that value.

The deviations of stream pressure and direction from a uniform stream revealed by the results of the surveys are significant only in the error they produce. The tolerable magnitude of such deviations must be established by investigating what errors are entailed in the results of tests of a model in such a stream. Large deviations in stream angle are permissible if their influence is known as, for instance, the influence of the stream curvature induced in the test section of a subsonic wind tunnel by the reflection of the model vortex sheet in the tunnel wall.

Figure 13 presents results of force<sup>3</sup> tests of the model of reference 2 in the air stream of the Ames 6- by 6-foot supersonic wind tunnel for various test conditions<sup>4</sup>. It may be noted that a

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<sup>3</sup>The model was equipped with a 4-inch-diameter, four-component, strain-gage balance which, of course, measures normal force and chord force instead of lift and drag. Lift and drag are obtained by resolution of the chord and normal forces. It should be noted that the model wing was twisted and cambered so that zero lift does not occur at zero angle of attack.

<sup>4</sup>The angle of attack is referred to the horizontal plane or to the vertical plane through the axis of the test section.

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comparison of the results obtained with the model upright and inverted gives the same values of the lift-curve slope and the moment-curve slope. Such a result is to be expected if first-order supersonic wing theory is applied to the estimation of the effects of stream-angle variations which give sections of the wing effective camber and twist. The angle of zero lift, the trim lift coefficient, and the magnitude of the drag, however, are affected to an undesirable degree by the nonuniformity of the stream. The first two of these parameters are necessary in the analysis of the stability of the aircraft. Methods of correcting these results for the effects of stream curvature, at least to a first-order approximation, exist in the literature on supersonic wing theory. However, the application of the theory is tedious and laborious. Further, it is not possible to account for the second-order effects of thickness and induced camber and twist which in conjunction with viscosity effects may, under special conditions, play an important part in determining the characteristics of the model.

The most direct approach toward eliminating the effects of stream irregularities is to orient the model so as to minimize their influence. Since there is no cross-stream flow in the nozzle of the 6- by 6-foot tunnel, it appears that, if the plane of the wing is placed vertically, essentially the same results will be obtained as for a uniform stream except for the influence of the small yaw angles, discussed later, resulting from the stream-angle variation in the vertical plane. Results obtained with the model so placed are also shown in figure 13. It may be noted that the points shown fall midway between the test data for the upright and inverted positions, which indicates that the effects of stream deviations were negligible when the model was mounted with the plane of the wing vertical.

The influence of the stream-angle variations on the drag characteristics are especially large when the wing is oriented in the horizontal plane, since the stream angle influences the inclination of the lift vector. It is essential, therefore, that drag tests be made for conditions where the stream angle is zero or, at least, where it is accurately known. The latter condition cannot be satisfied if the stream angle varies over the span of the test model as is the case for the present swept-wing model mounted with the wing horizontal. It is necessary, therefore, to place the span of the wing vertical.

If the model is mounted with the plane of the wing vertical, certain inaccuracies are still possible because of the following:

1. The Mach number variation across the span of the model. For the present swept-wing model, this effect is relatively small since the variation of model characteristics with Mach number

is small. The effect on the characteristics of other models needs to be investigated.

2. The effects of small yaw angles on the characteristics. These are, of course, small if the characteristics studied are not greatly influenced by the angle of yaw. In this regard, if the model is tested with the plane of the wing vertical, stream angularity influences only those characteristics which are functions of both pitch and yaw which is the unusual case, but which might be noted as applying to the rolling moment and yawing moment due to sideslip for swept wings and to certain characteristics of cruciform wings. Since the lift, drag, and pitching moment do not vary appreciably with small angles of yaw, these characteristics are not affected.

3. The effects of axial pressure gradients in providing a buoyant force and the possibility of the pressure gradients altering the true pressure gradients over wing and body to such an extent that the viscosity effects are changed. This latter effect is remote, however (unless the stream contains discrete shock waves not revealed by the surveys). A correction may be applied for the former. (See reference 3.)

It should be noted that the results of the force tests indicate that experimental investigations of the loading due to angle of attack through measurement of the pressure difference between the upper and lower surface of a wing may as well be done with the model mounted with wing horizontal if more convenient. This may be deduced from the fact that the lift-curve and moment-curve slopes are not influenced by the orientation of the model. Tests were made with the model of reference 2 to demonstrate the validity of this conclusion. The results are shown in figure 14 for the wing horizontal and wing vertical.<sup>5</sup> The agreement is seen to be generally satisfactory except in regions where viscosity effects are large near the trailing edge and tip. In these regions the pressures vary to some extent from test to test with the same model orientation.

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<sup>5</sup>The data given are for 3.70° and 5.74° change in angle of attack with the model horizontal. Data for a nominal change in angle of 5° with wing vertical were obtained by rotating the 5° bent sting used in the tests of reference 2 through 90°. The change in angle of attack  $\Delta\alpha$  has been corrected for the deflection of the model support sting under load.

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If the pressure distribution over either the upper or lower surface of the wing is required independently, however, the influences of stream curvature in producing effective camber and twist must be minimized by mounting the model with the span vertical. There is also the question of correcting for the pressure variation in the stream so that the pressures over the model may be referred to the average ambient pressure of the stream. This may be done by a simple superposition process as in reference 4. The correction can be reasoned as a valid first-order approximation if the pressure disturbances (expansion or compression waves of infinitesimal strength) are small and are not reflected by the wing surface. If the flow in the nozzle is two-dimensional, reflection of the pressure disturbances will not occur if the plane of the model wing is placed so as to intersect at right angles the planes along which the weak pressure waves are propagated. In the Ames 6- by 6-foot supersonic wind-tunnel nozzle, the plane of the model wing must be placed vertically to insure the validity of the correction.

#### CONCLUDING REMARKS

The surveys of the air stream in the test section of the asymmetric adjustable nozzle of the Ames 6- by 6-foot supersonic wind tunnel show that the flow is nearly uniform at a nominal Mach number of 1.4. As the Mach number is increased or decreased from a value of 1.4, however, vertical pressure gradients of significant magnitude are found. Smaller axial gradients also exist. The transverse gradients are of negligible magnitudes which indicates that the flow is essentially two-dimensional.

The existence of large vertical pressure gradients implies an appreciable variation in stream angle with axial position. Stream-angle measurements confirm this.

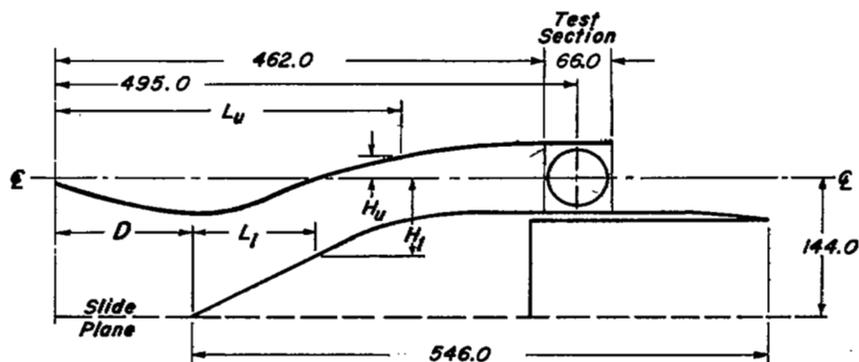
The results of tests of one swept-wing model indicated that for this model, at least, the effects of the nonuniformity of the stream on certain model characteristics may be minimized by testing with the plane of the model wing parallel to the two-dimensional-flow planes. For other model characteristics which are combined functions of the angle of pitch and the angle of yaw, this method will not be effective. In such cases, appropriate corrections need to be applied. At some Mach numbers, the magnitude and uncertainty of these corrections may be such as to preclude certain tests. Research devoted to the refinement of nozzle design techniques is now proceeding with a view toward improving the flow in the wind tunnel at these Mach numbers.

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

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1. Allen, H. Julian: The Asymmetric Adjustable Supersonic Nozzle for Wind-Tunnel Application. NACA RM A8E17, 1948.
2. Stevens, Victor I., and Boyd, John W.: A Comparison of Theoretical and Experimental Loading on a  $63^\circ$  Swept-Back Wing at Supersonic Speeds. NACA RM A9C16, 1949.
3. Chapman, Dean R., and Perkins, Edward W.: Experimental Investigation of the Effects of Viscosity on the Drag of Bodies of Revolution at a Mach Number of 1.5. NACA RM A7A31a, 1947.
4. Frick, Charles W., and Boyd, John W.: Investigation at Supersonic Speed ( $M = 1.53$ ) of the Pressure Distribution over a  $63^\circ$  Swept Airfoil of Biconvex Section at Zero Lift. NACA RM A8C22, 1948.

TABLE I  
Coordinates of Nozzle Blocks



Dimensions in inches.

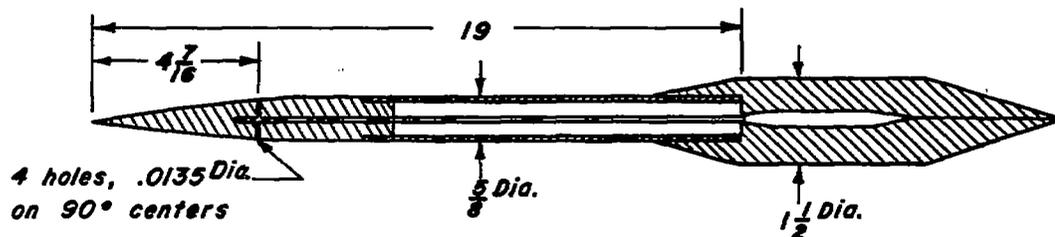
$L_1$	$H_1$	$L_2$	$H_2$	$L_1$	$H_1$
0	144.000	0	-6.000	269.840	6.281
146.724	65.952	*48.000	-21.360	271.591	8.174
149.724	64.354	*138.000	-35.976	277.379	9.951
152.724	62.788	144.000	-35.842	283.203	11.734
155.724	61.243	150.000	-35.428	289.062	13.404
158.724	59.726	156.000	-34.724	294.958	15.002
161.724	58.227	162.000	-33.726	300.887	16.528
164.724	56.743	168.000	-32.440	306.852	17.984
167.724	55.275	174.000	-30.876	312.849	19.370
170.724	53.829	180.000	-29.052	318.160	20.534
173.304	52.620	186.000	-26.991	324.035	21.762
176.043	51.395	192.000	-24.713	329.524	22.929
178.806	50.223	198.000	-22.243	335.825	24.036
181.593	49.103	200.000	-21.381	341.798	25.086
184.403	48.034	202.000	-20.501	347.665	26.080
187.237	47.017	204.000	-19.605	353.601	27.022
190.094	46.051	206.000	-18.693	359.548	27.916
192.975	45.135	208.000	-17.765	365.506	28.764
195.877	44.267	210.000	-16.821	371.472	29.567
198.803	43.449	212.000	-15.865	377.449	30.326
201.750	42.677	213.000	-15.385	383.436	31.044
204.718	41.953	213.899	-14.953	389.433	31.723
207.708	41.276	214.800	-14.522	395.439	32.363
210.719	40.643	215.702	-14.092	401.453	32.960
213.750	40.056	216.605	-13.663	407.478	33.512
217.600	39.378	217.509	-13.235	413.514	34.016
220.570	38.956	218.414	-12.808	419.560	34.471
223.548	38.582	221.133	-11.539	425.617	34.874
226.528	38.231	226.596	-9.056	431.683	35.221
229.513	37.909	232.056	-6.642	437.762	35.504
232.501	37.610	237.630	-4.303	443.852	35.716
235.494	37.336	243.198	-2.037	449.954	35.860
238.489	37.086	248.804	-1.156	456.070	35.948
241.487	36.860	254.446	2.273	462.000	36.000
244.490	36.656	260.124	4.314	468.000	36.139
247.496	36.476				
250.504	36.321				
253.517	36.193				
256.534	36.094				
259.556	36.027				
262.583	35.998				
264.000	36.000				
274.000	37.119				
300.366	38.712				
346.000	44.000				

Circular arc of radius 284.382 inches between (\*) stations

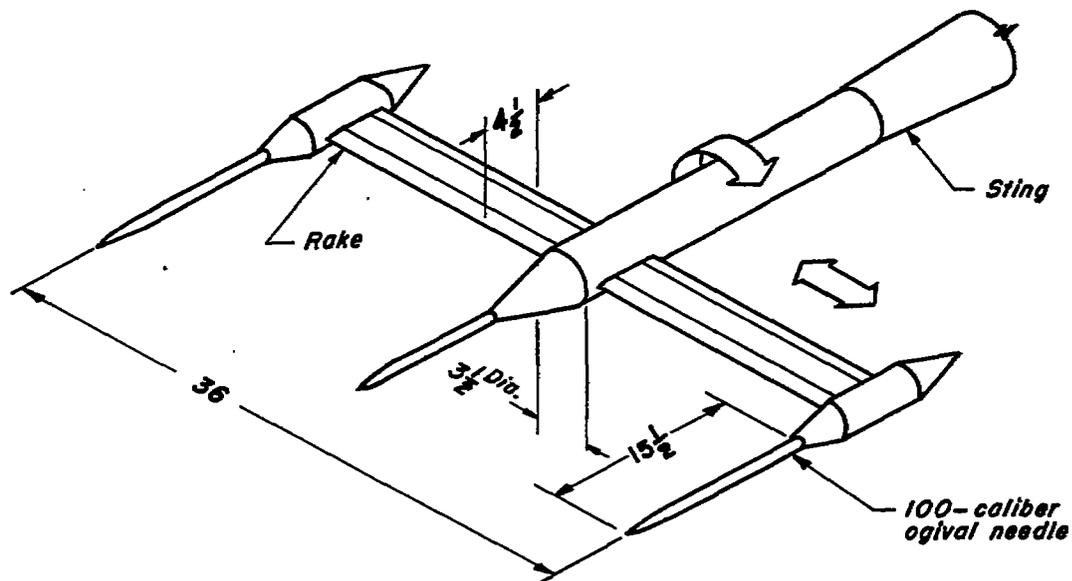








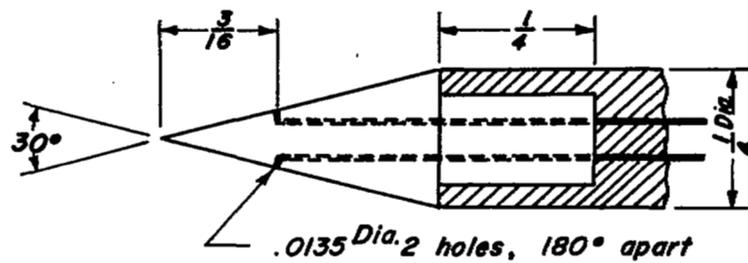
Needle detail



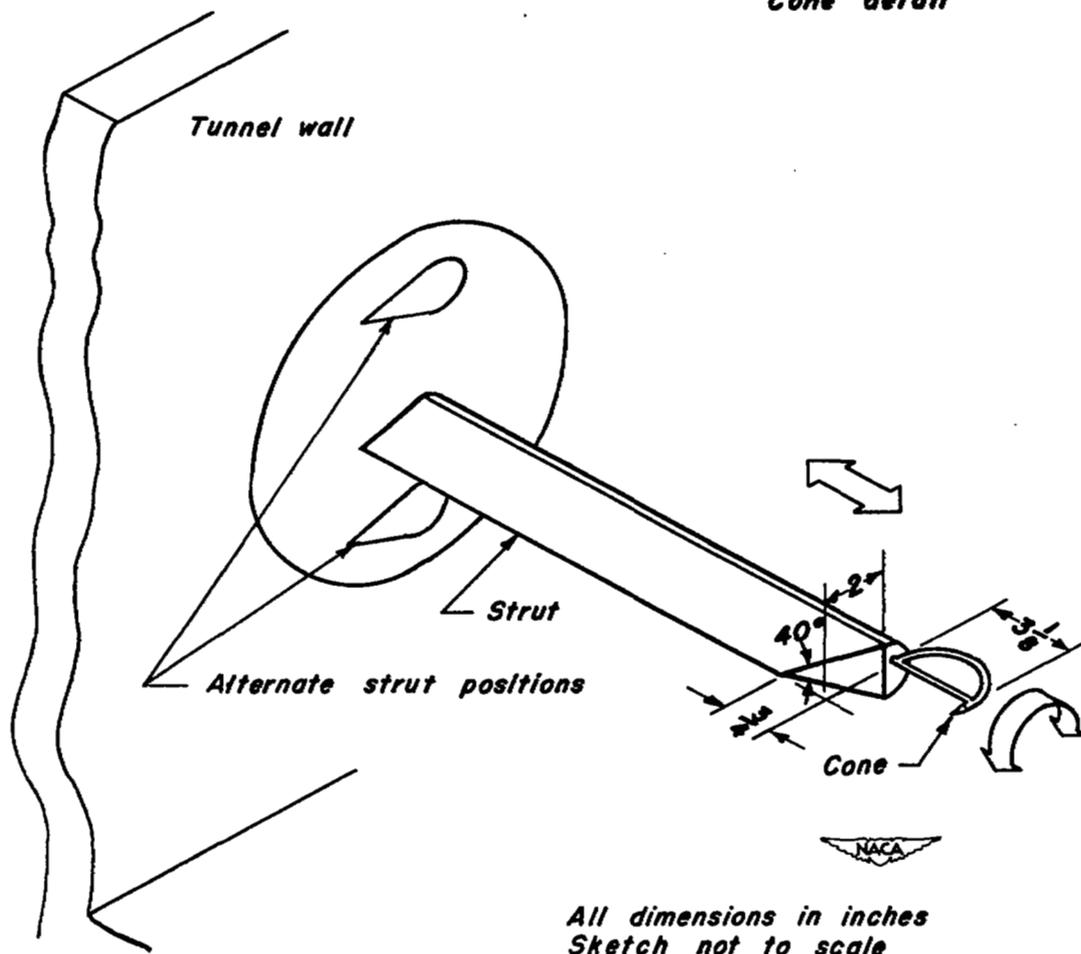
All dimensions in inches  
Sketch not to scale.

Figure 2 .- Static-pressure survey apparatus.





Cone detail



All dimensions in inches  
Sketch not to scale

Figure 3 - Stream-angle survey apparatus.

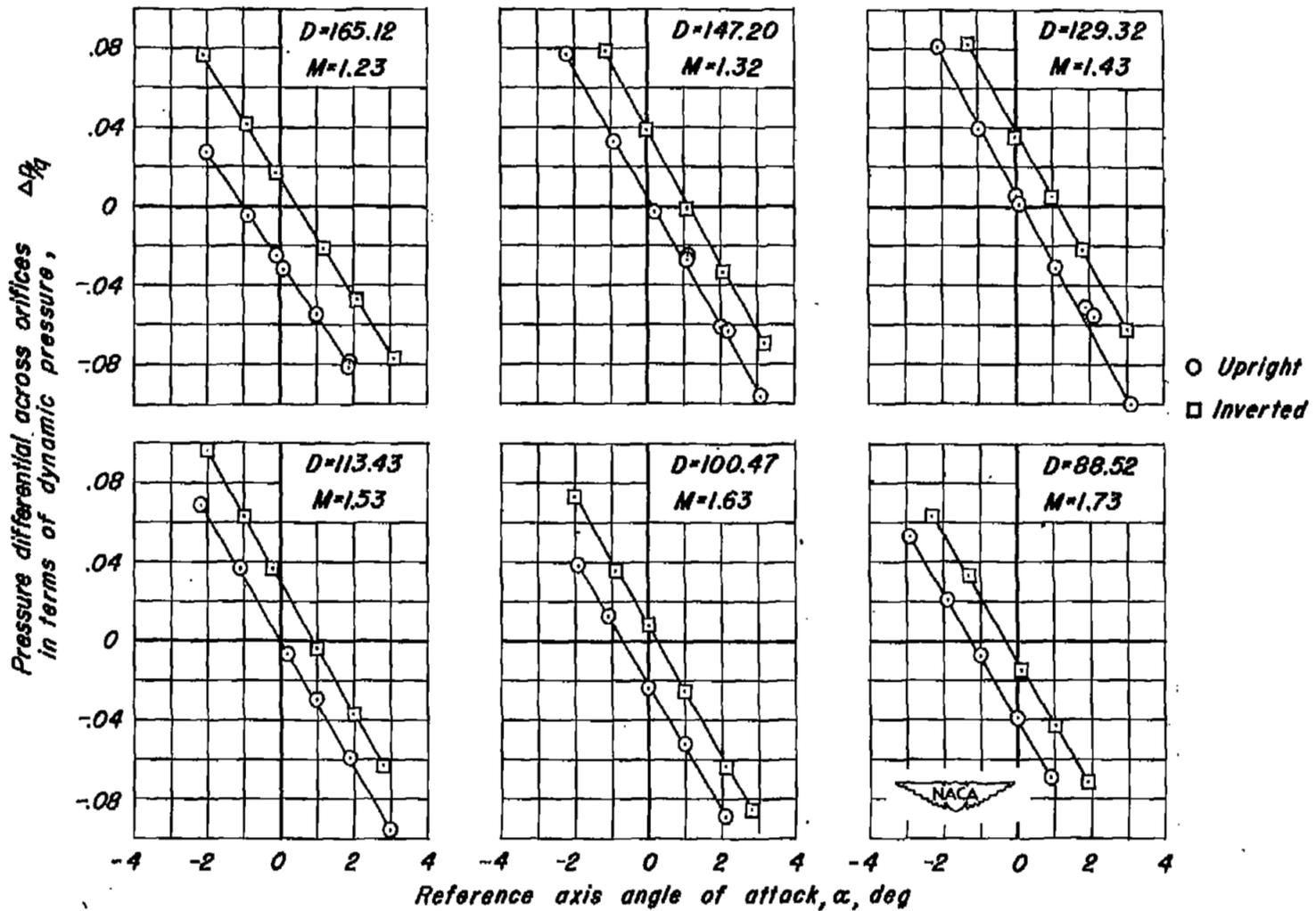
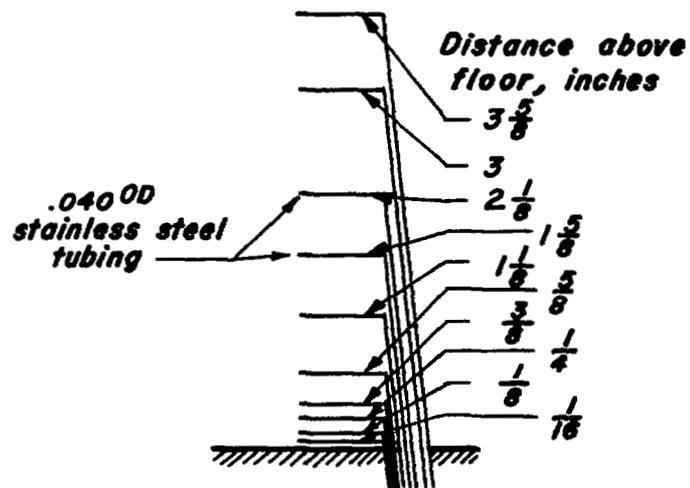


Figure 4.-Pressure differential across orifices of angle-survey cone in upright and inverted positions.



Dimensions in inches.

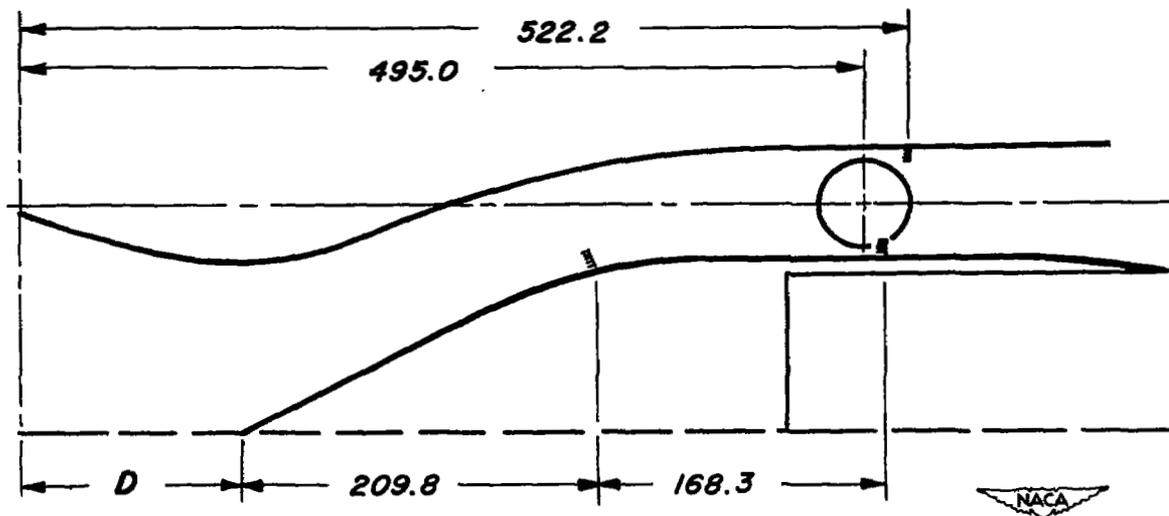


Figure 5.—The apparatus for boundary-layer surveys in the Ames 6-by-6-foot supersonic wind tunnel.

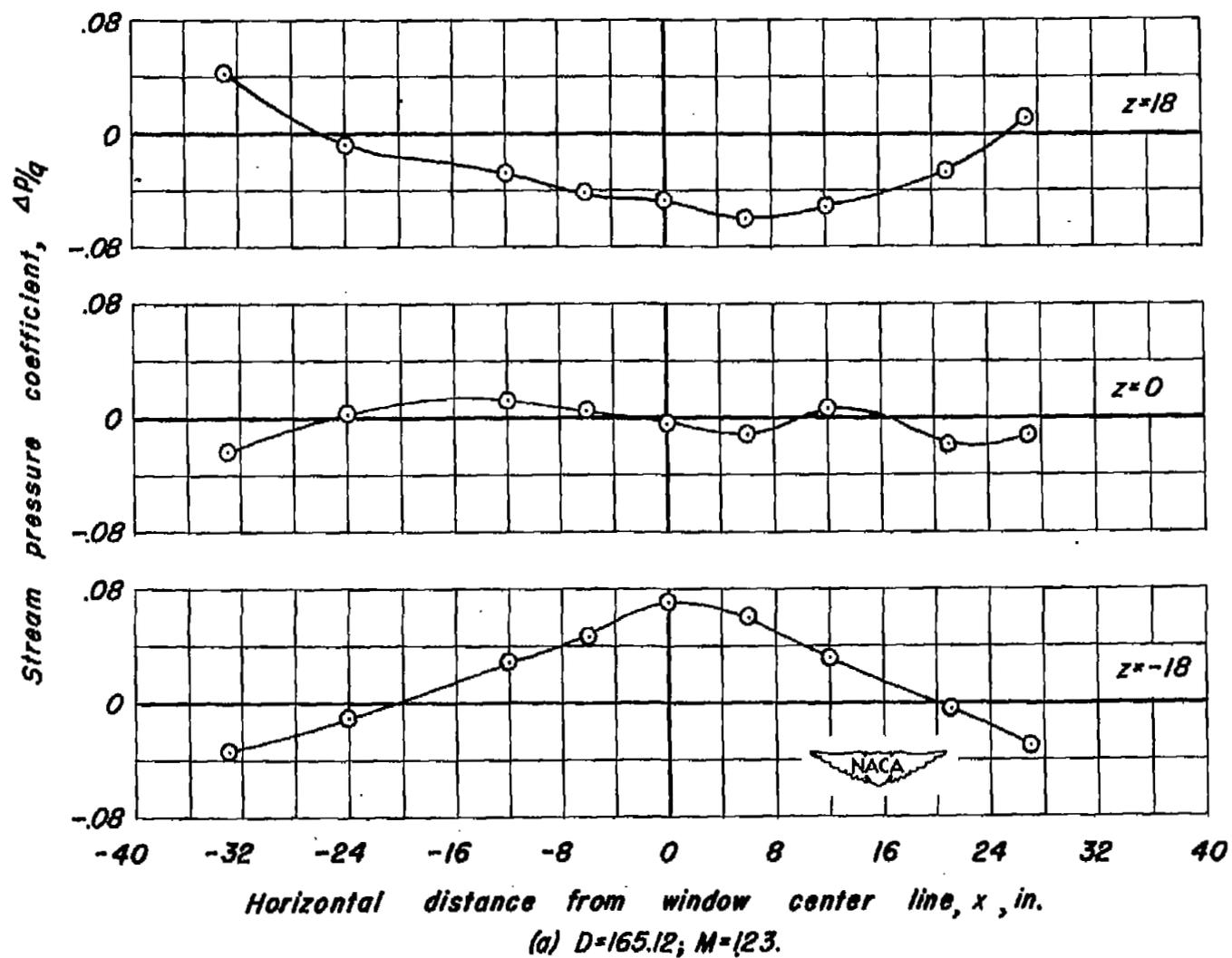
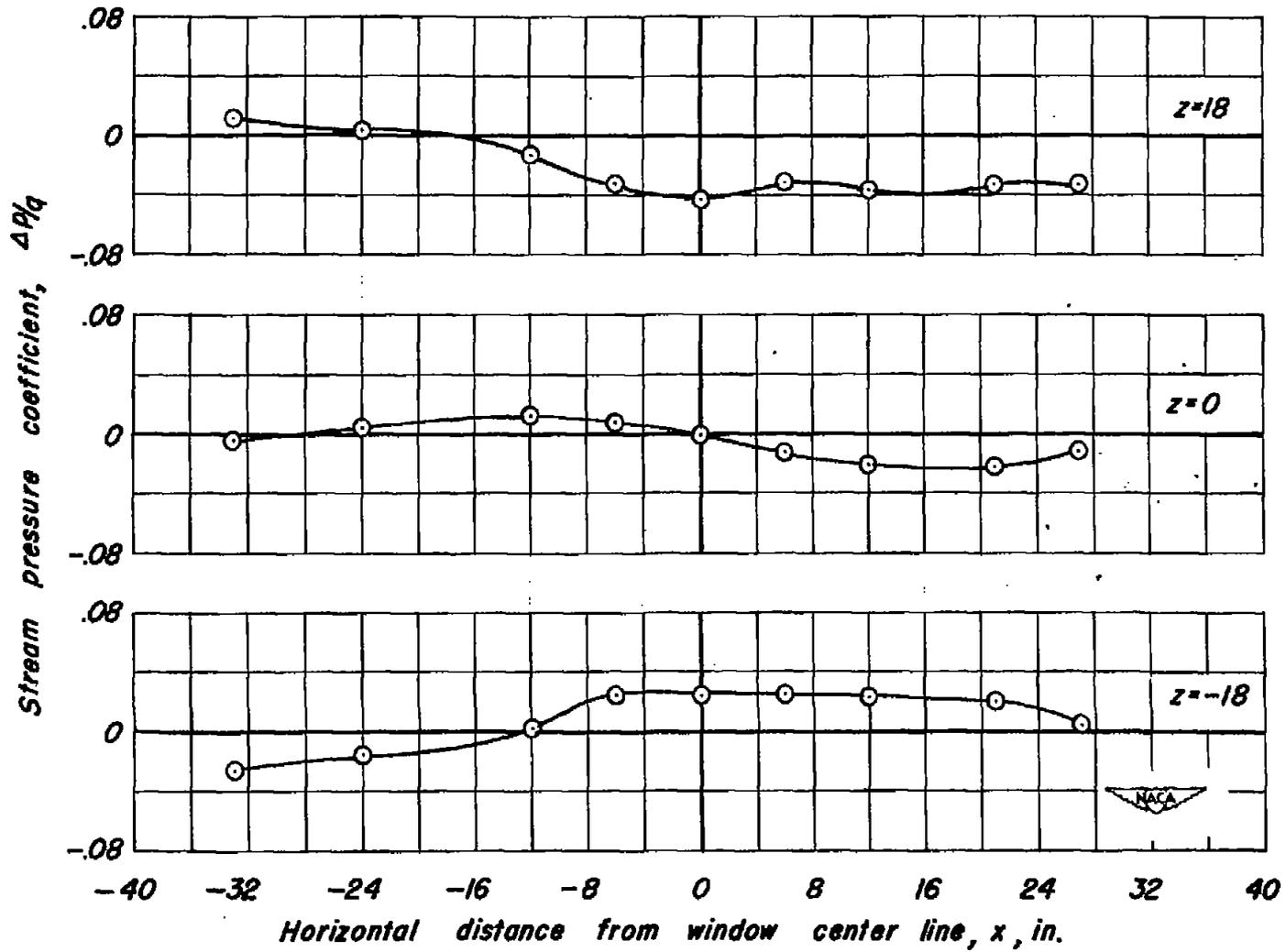
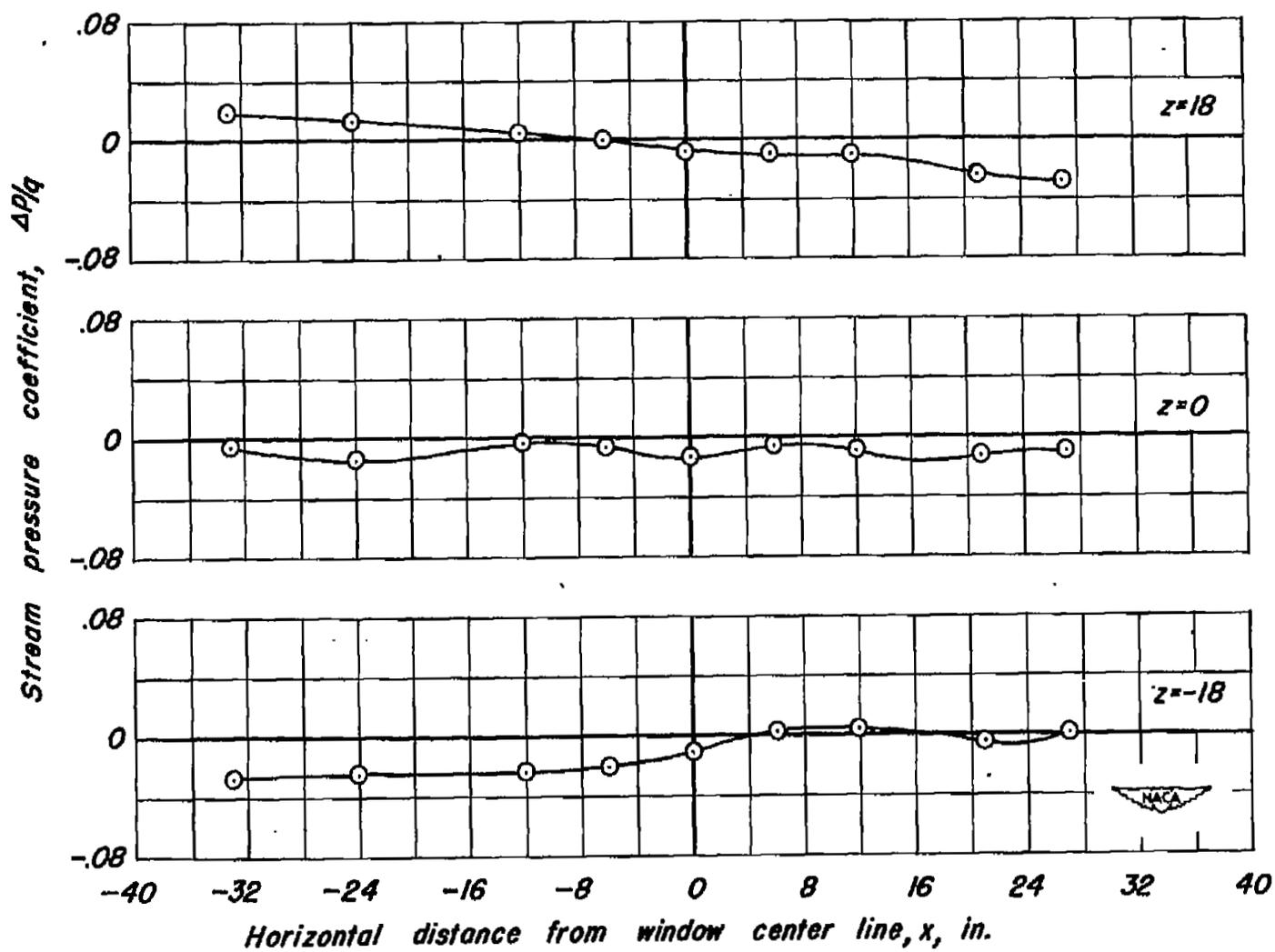


Figure 6.- The variation of static pressure axially in the Ames 6-by 6-foot supersonic wind tunnel.  $y=0$ ; stagnation pressure = 9 lb/sq in. abs.



(b)  $D=147.20$ ;  $M=132$ .

Figure 6.-Continued.



(c)  $D=129.32$ ;  $M=1.43$ .

Figure 6, -Continued.

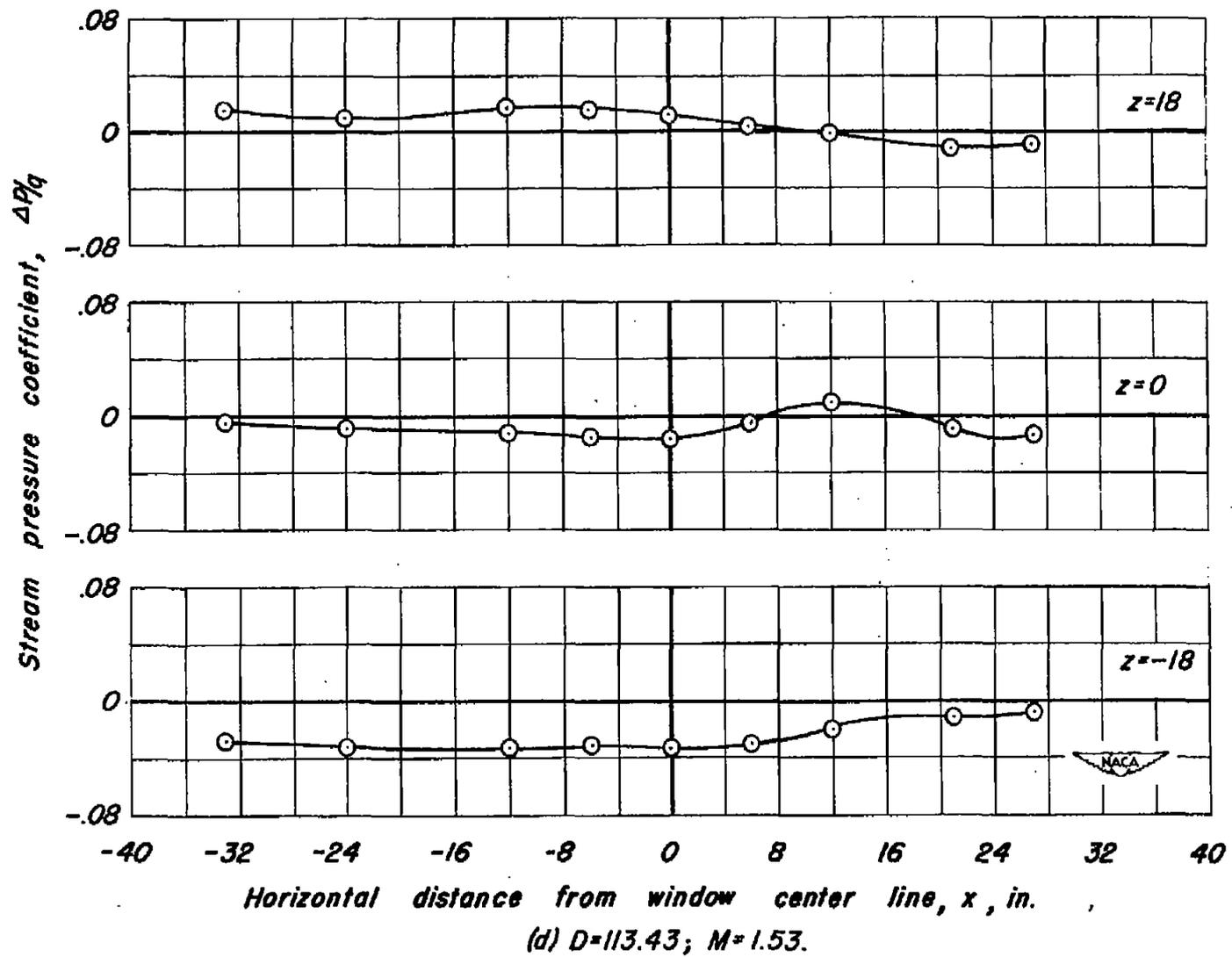


Figure 6.-Continued.

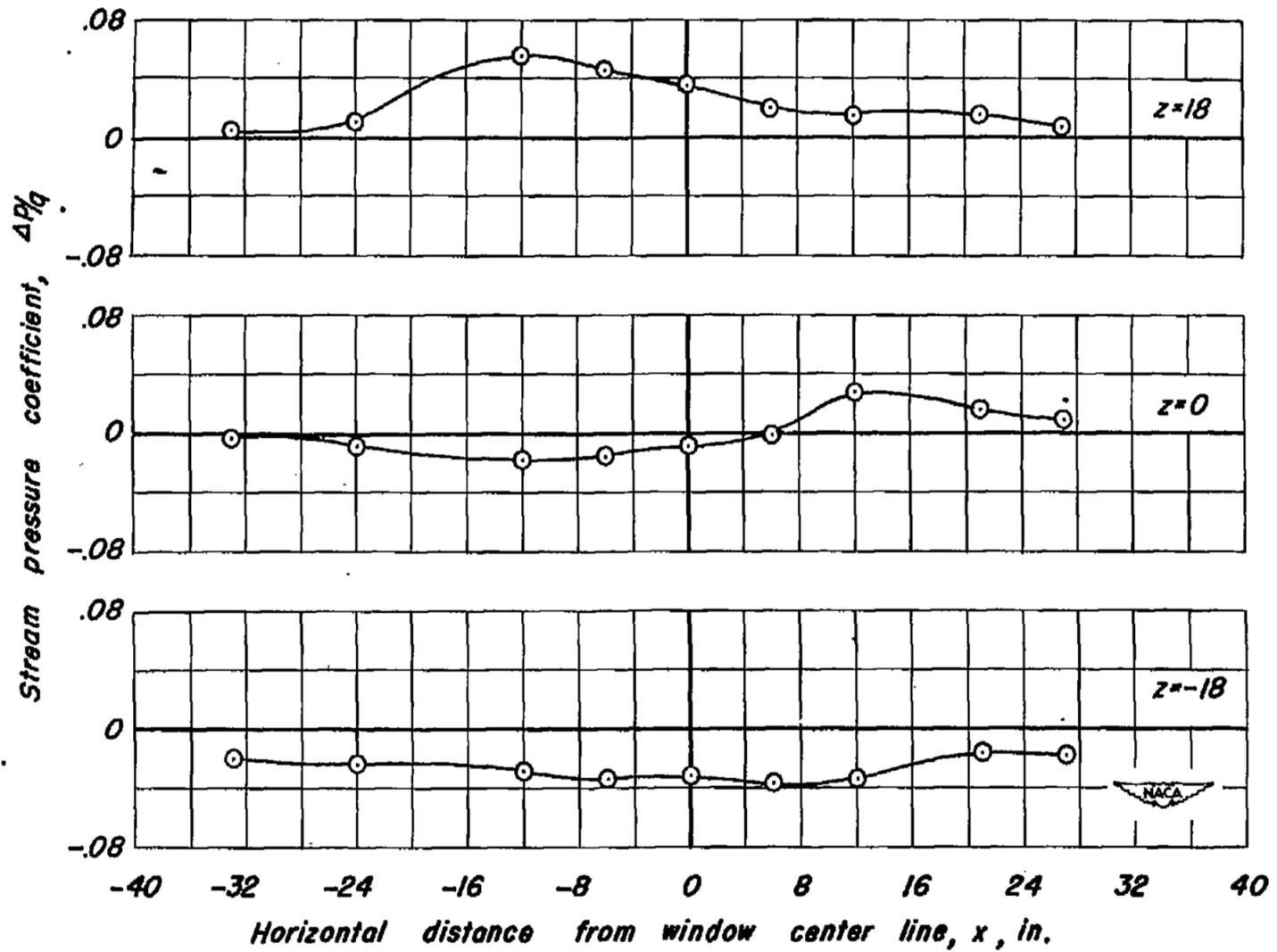


Figure 6.-Continued.

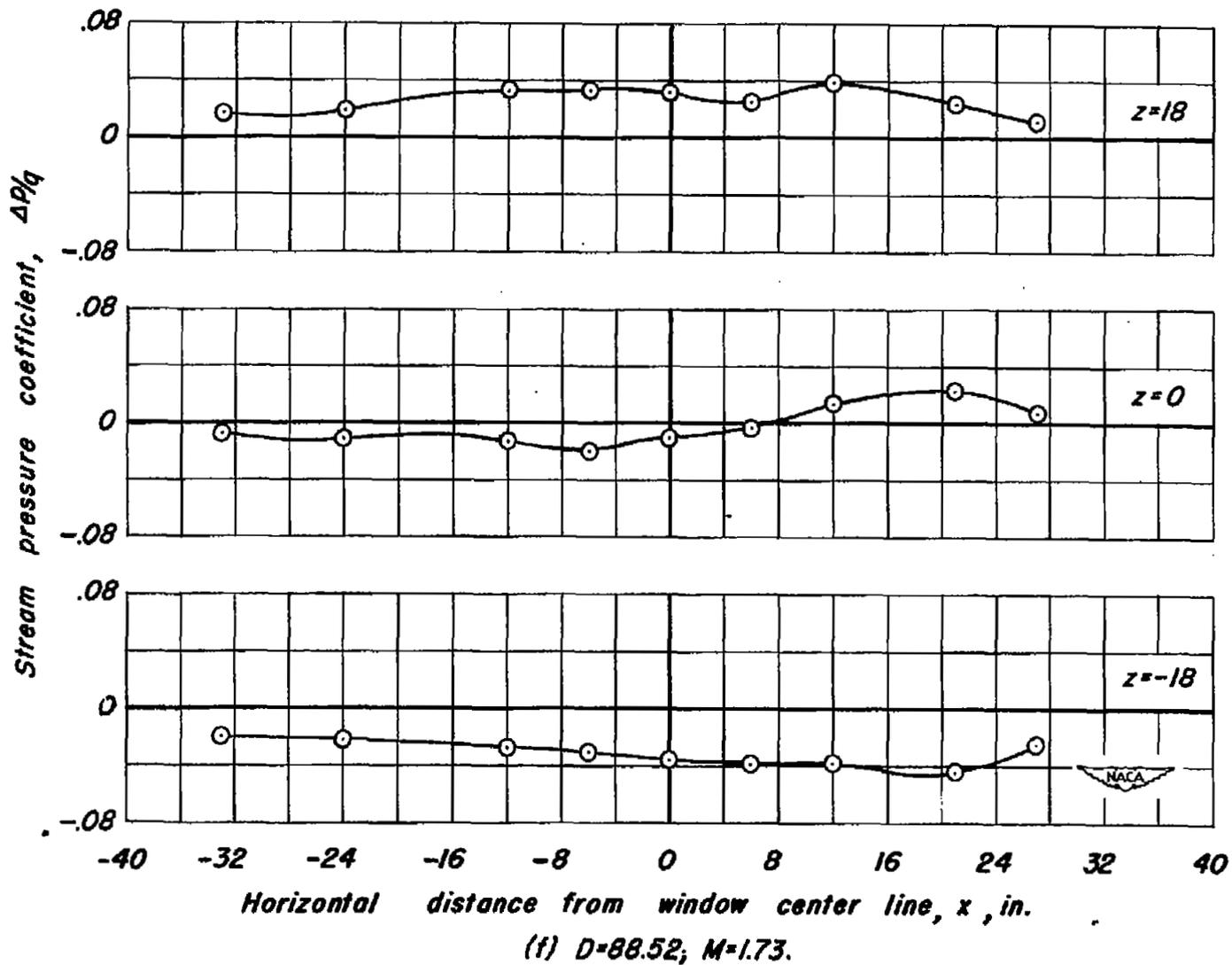
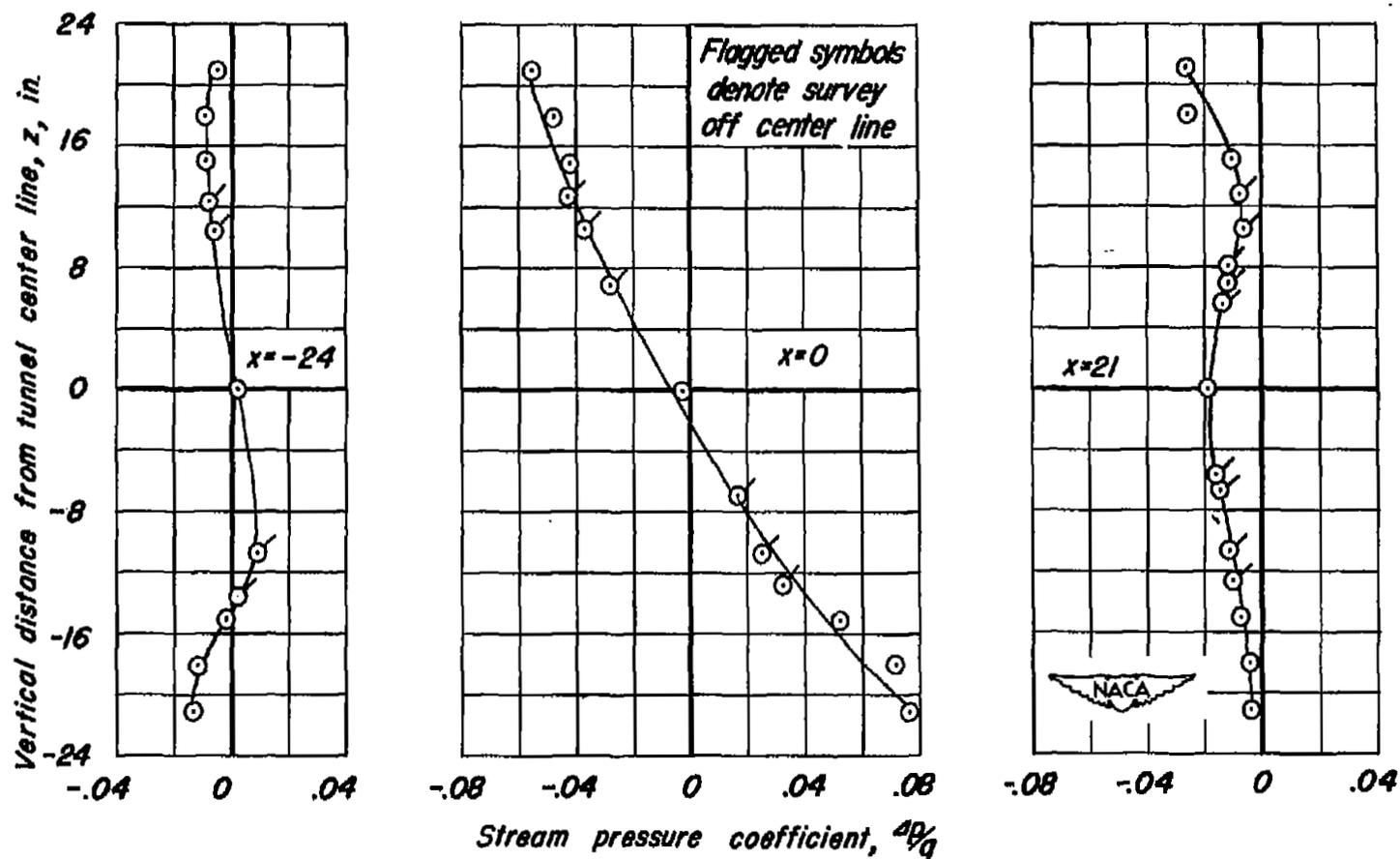
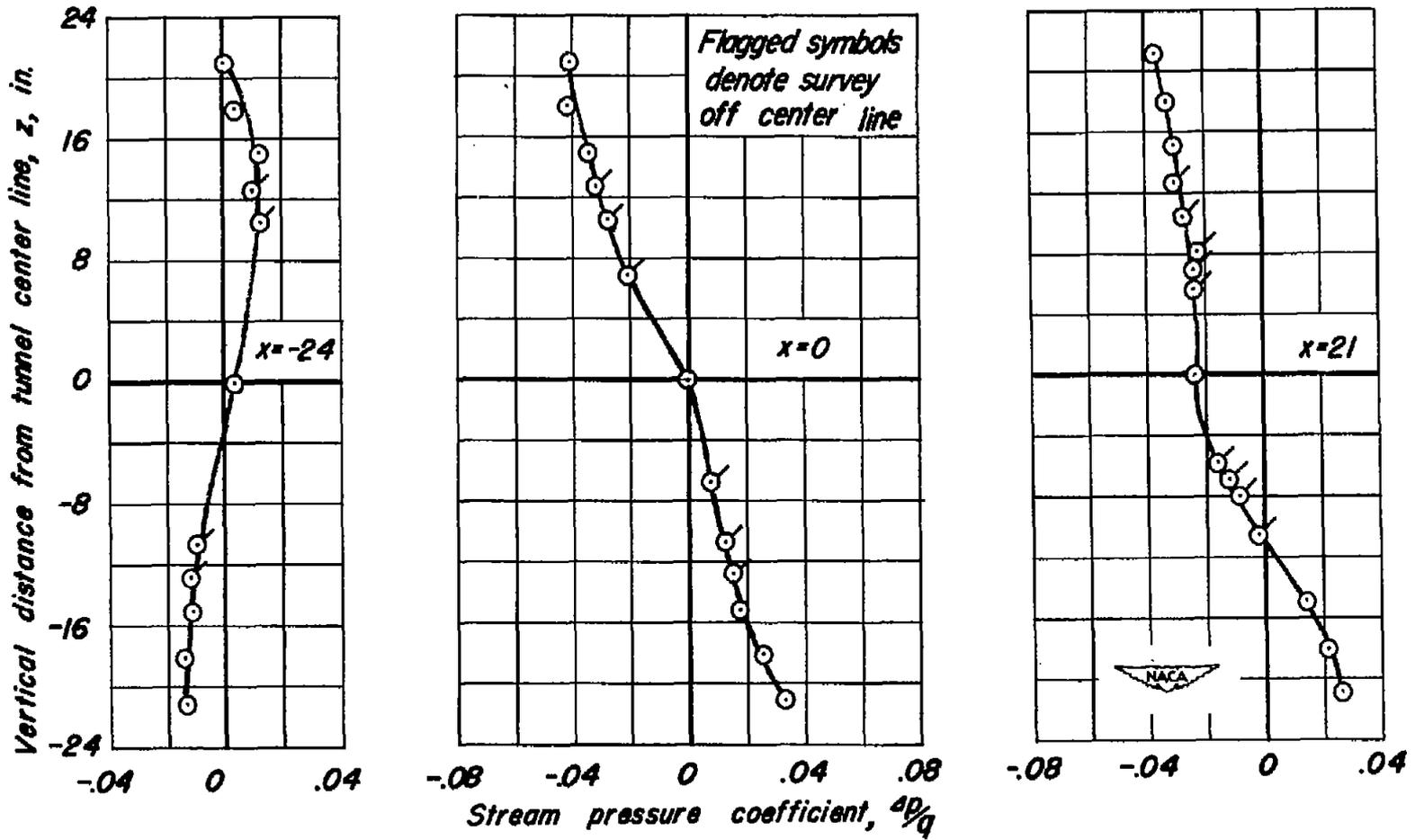


Figure 6.-Concluded.



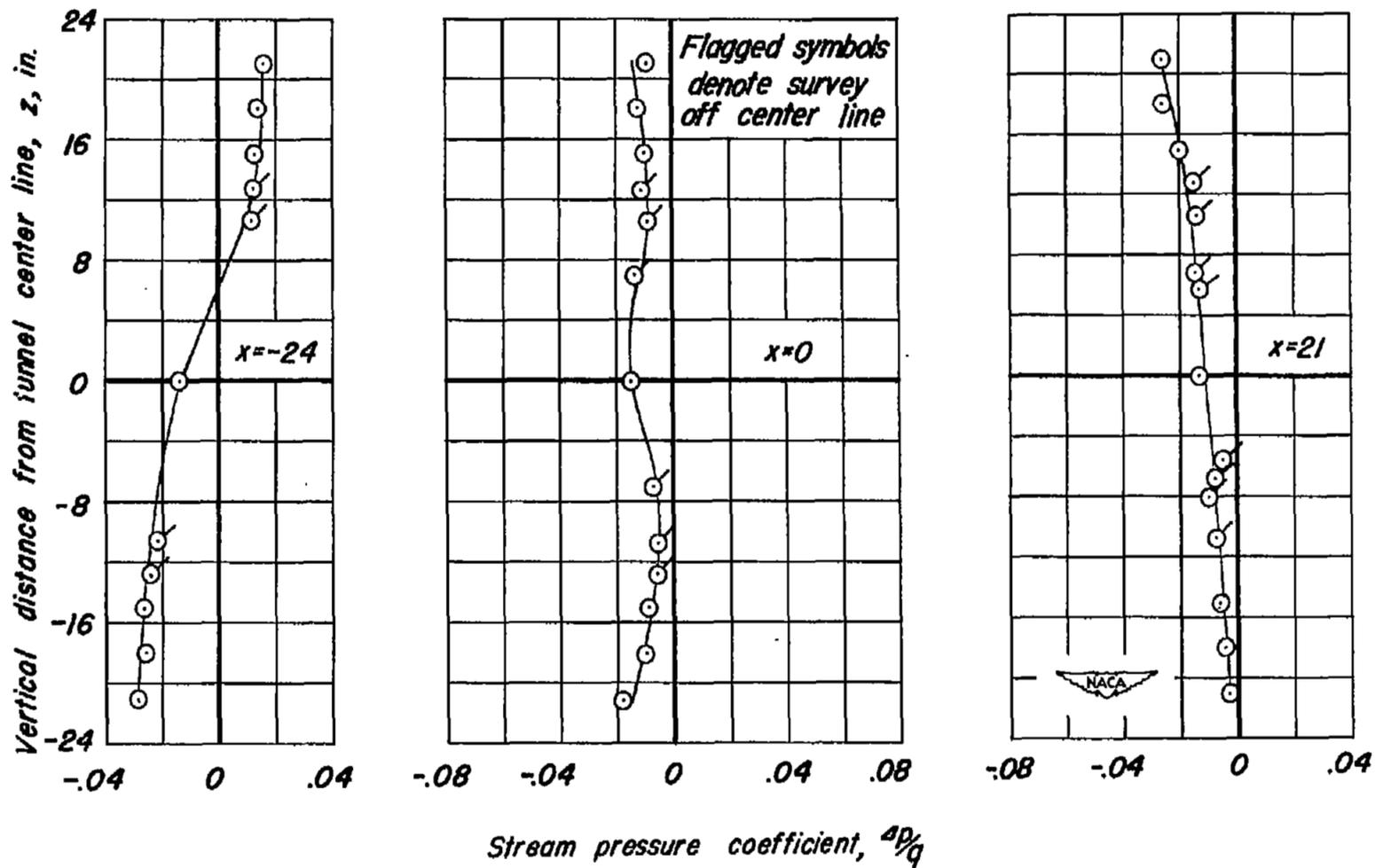
(a)  $D=165.12$ ;  $M=1.23$ .

Figure 7.- The variation of static pressure vertically in the Ames 6- by 6-foot supersonic wind tunnel.  $y=0$ ; stagnation pressure = 9 lb/sq in. abs.



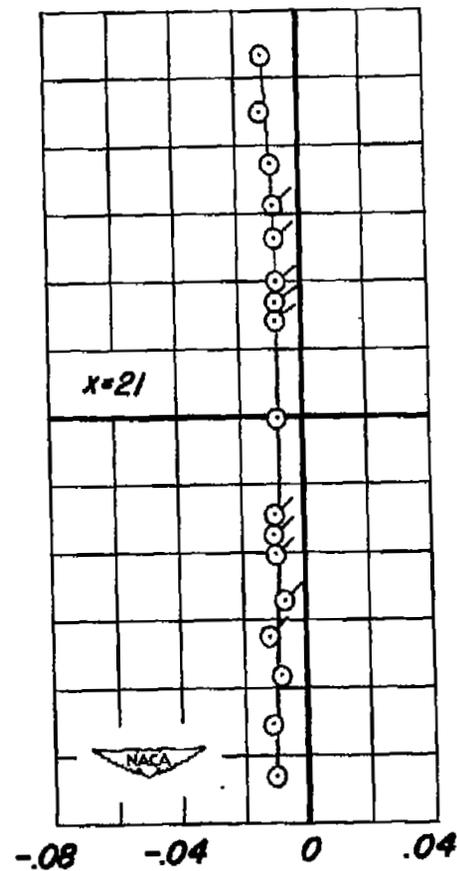
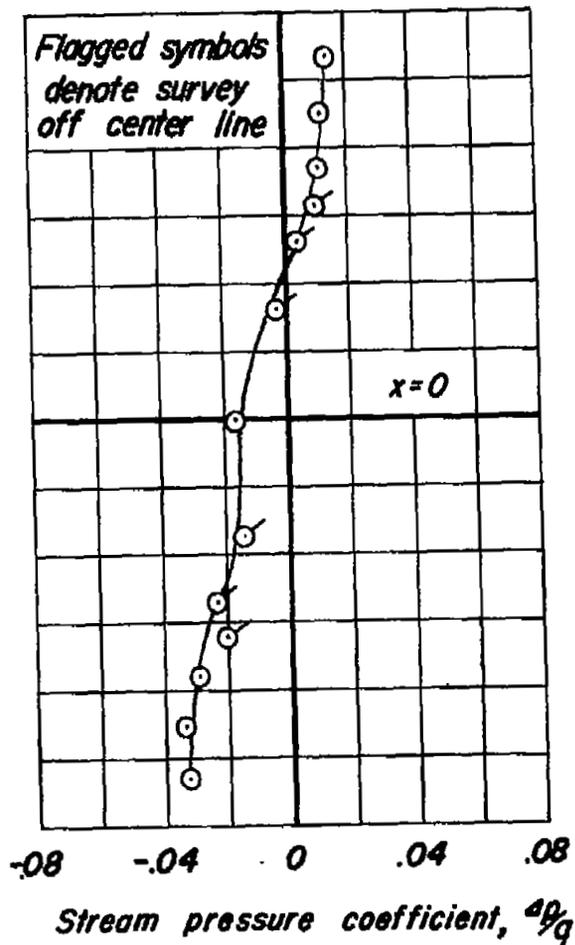
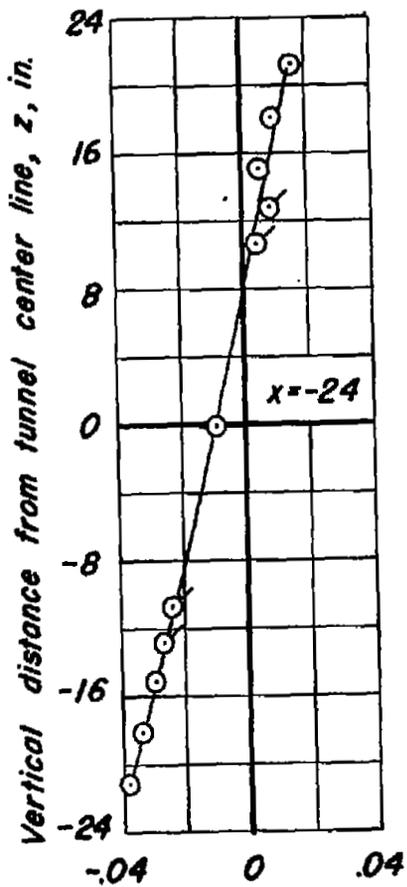
(b)  $D=147.20$ ;  $M=1.32$ .

Figure 7.- Continued.



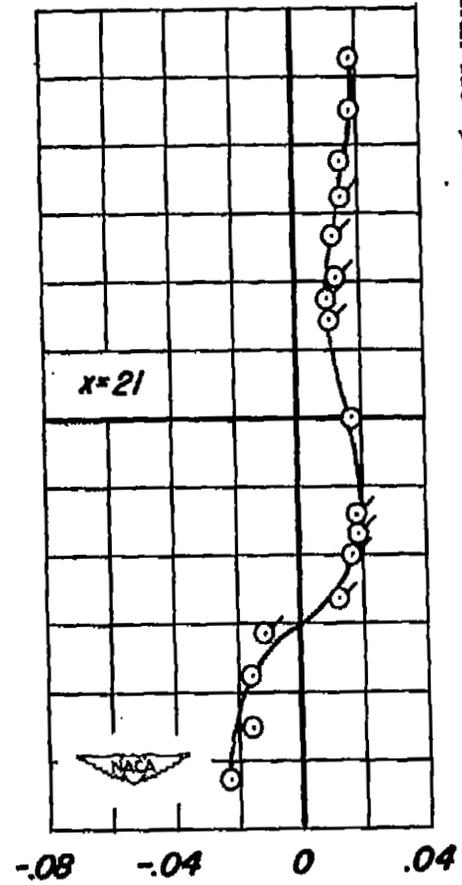
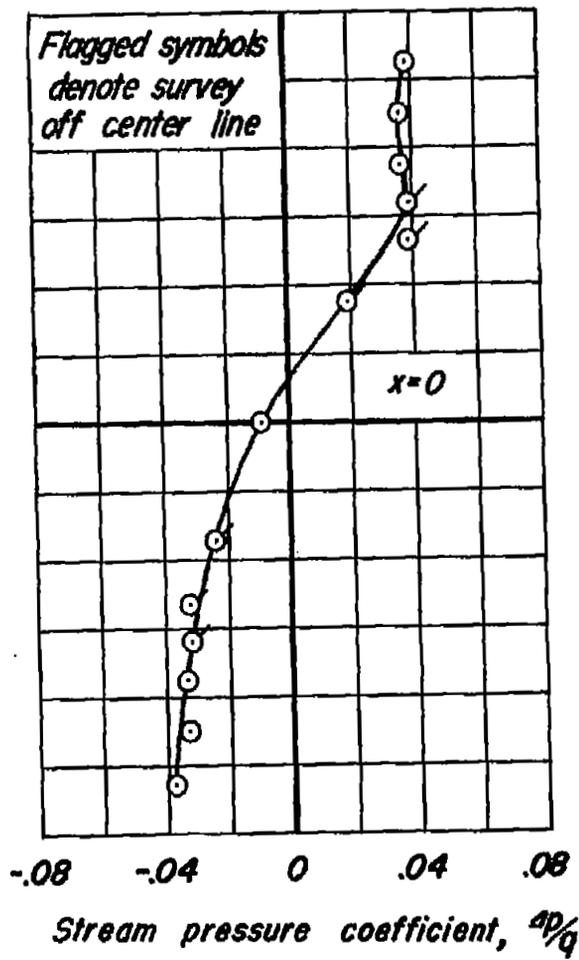
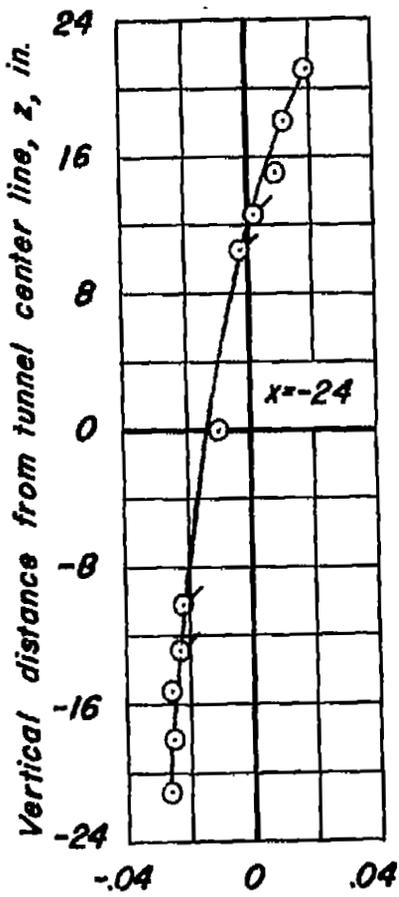
(c)  $D=129.32$ ;  $M=1.43$ .

Figure 7.- Continued.



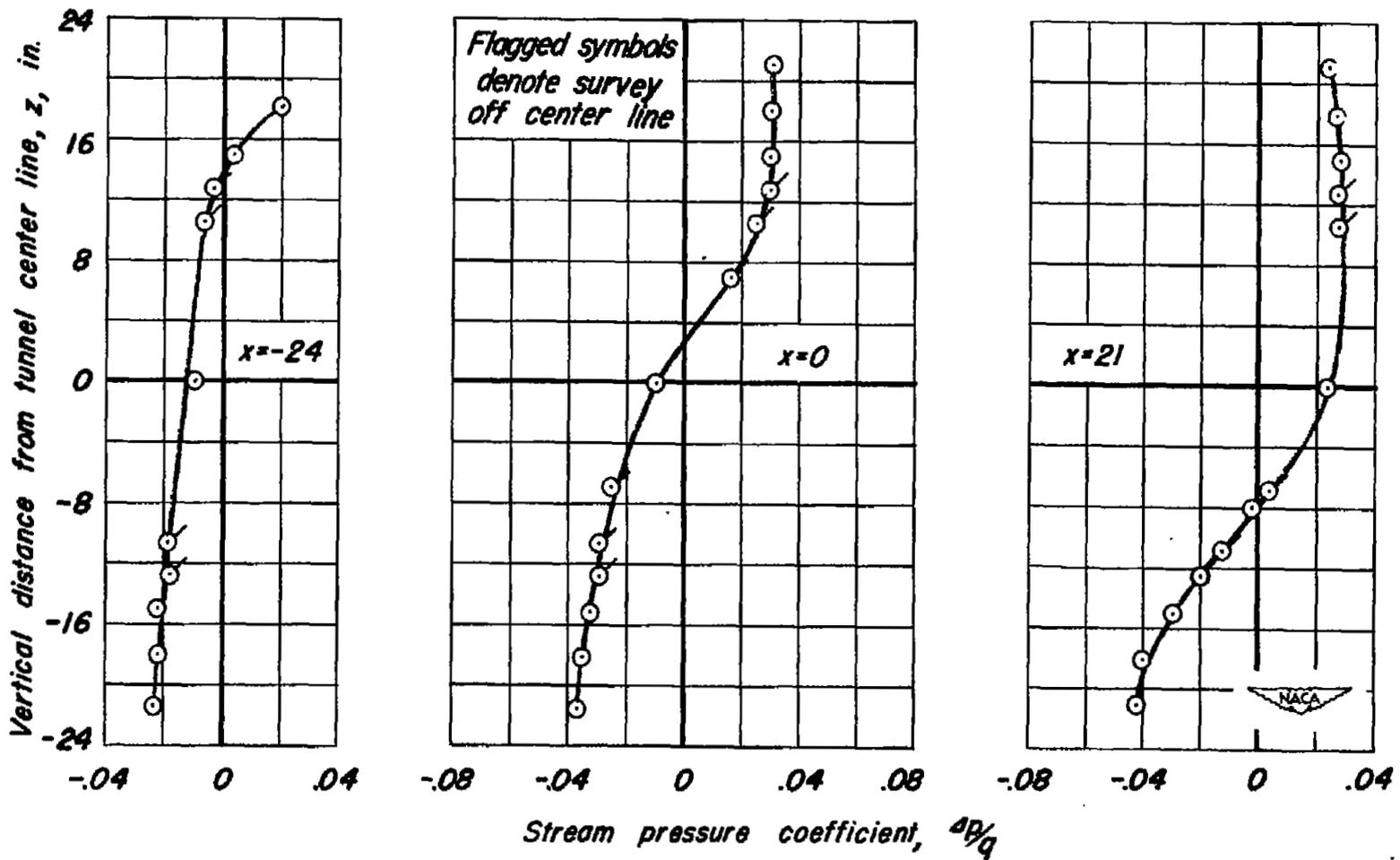
(d)  $D=113.43$  ;  $M=1.53$ .

Figure 7.- Continued.



(e)  $D=100.47$ ;  $M=1.63$ .

Figure 7.— Continued.



(f)  $D=88.52$ ;  $M=1.73$ .

Figure 7.- Concluded.

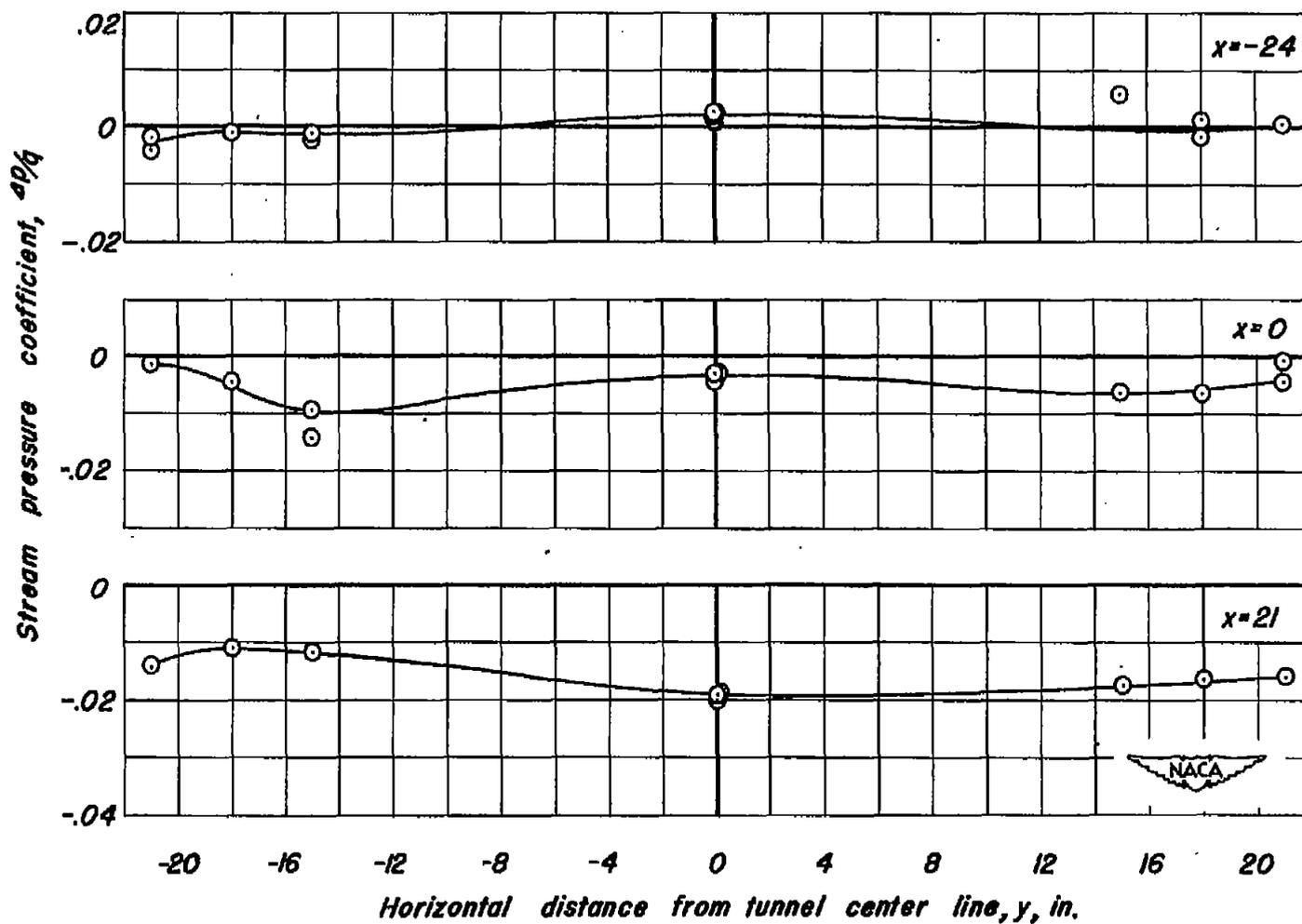
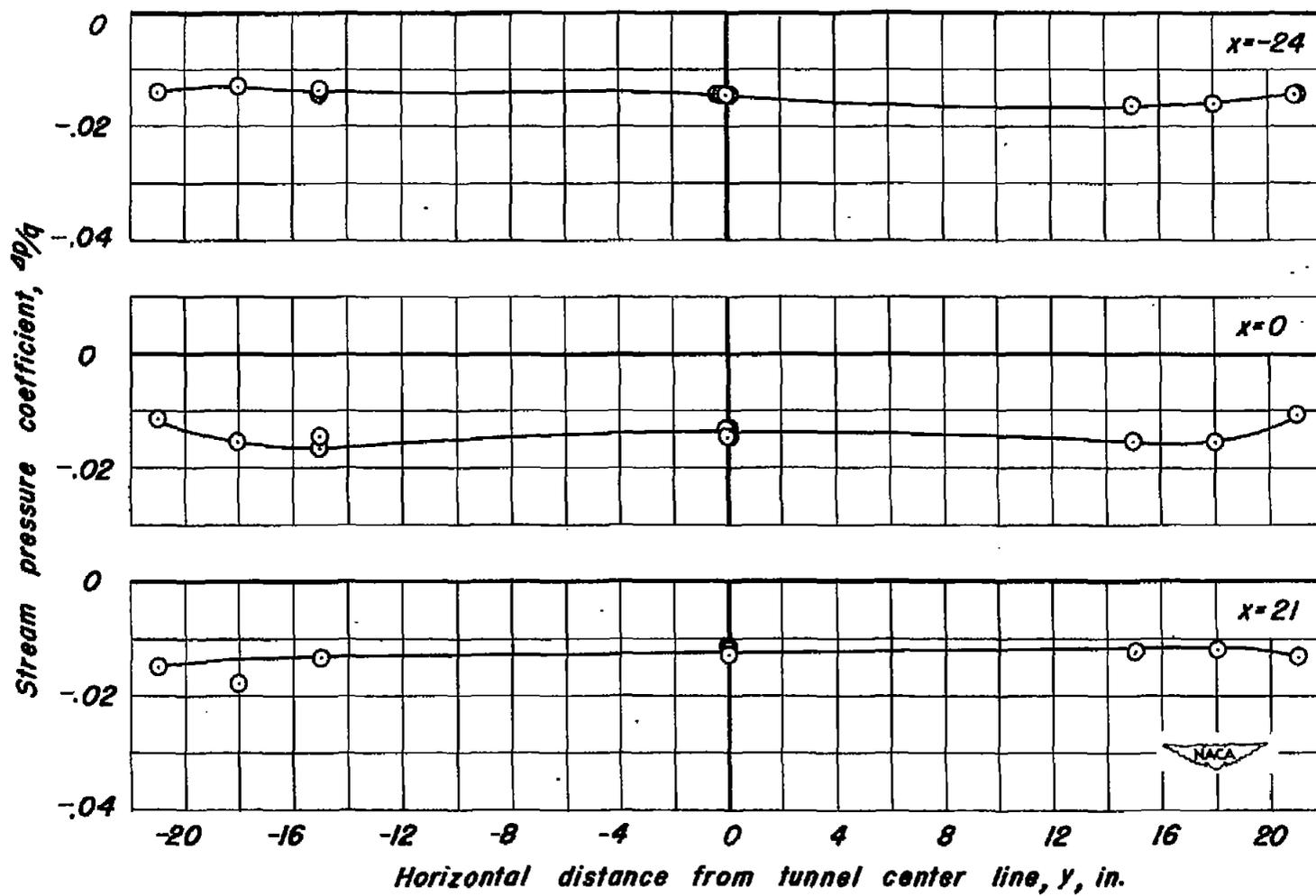
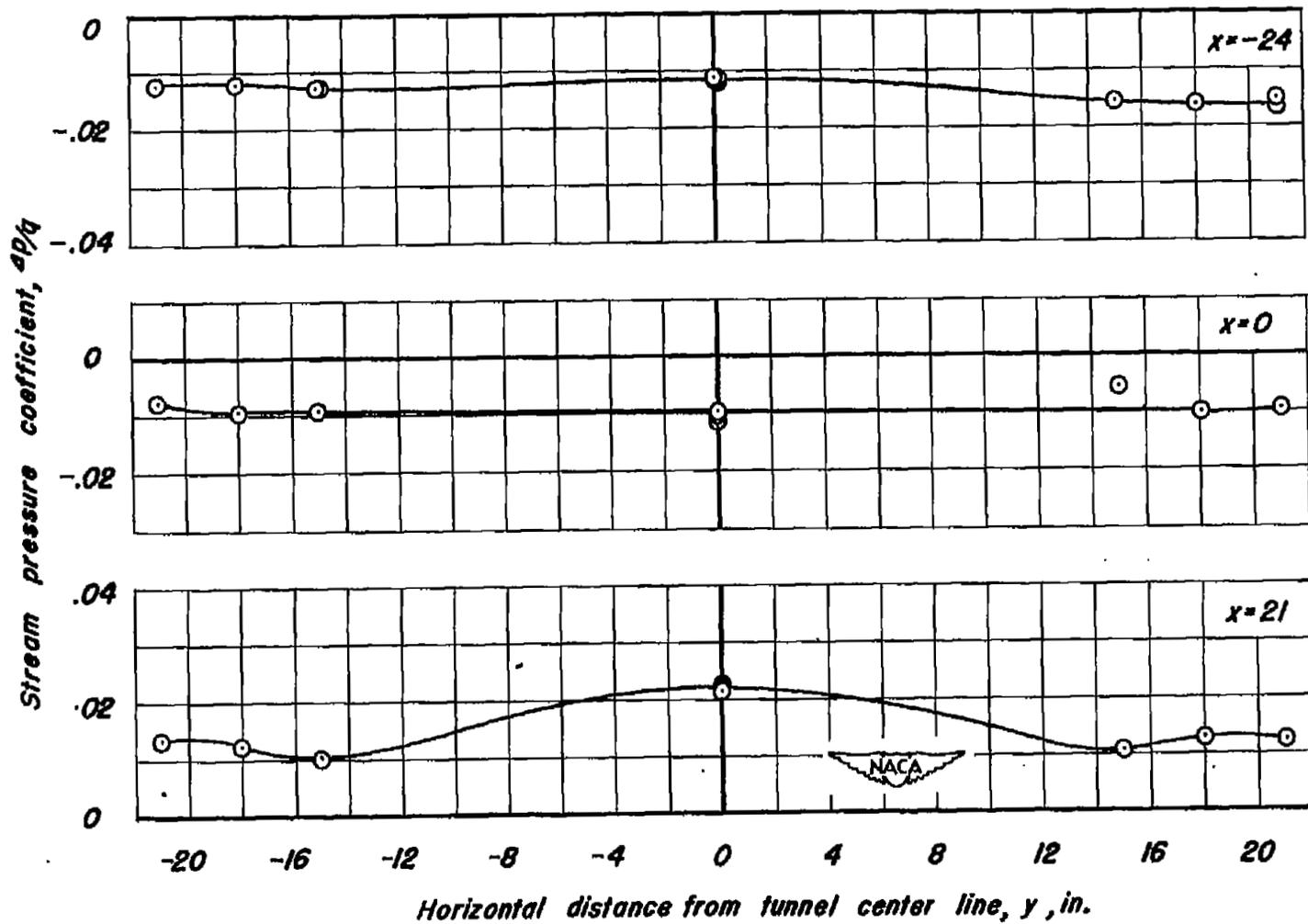
(a)  $D=165.12$ ;  $M=1.23$ .

Figure 8.— The variation of static pressure in the transverse direction in the Ames 6- by 6-foot supersonic wind tunnel.  $z=0$ ; stagnation pressure = 9 lb/sq in. abs.



(b)  $D=129.32$  ;  $M=1.43$ .

Figure 8.-Continued.



(c)  $D=88.52$ ;  $M=1.73$ .

Figure 8.- Concluded.

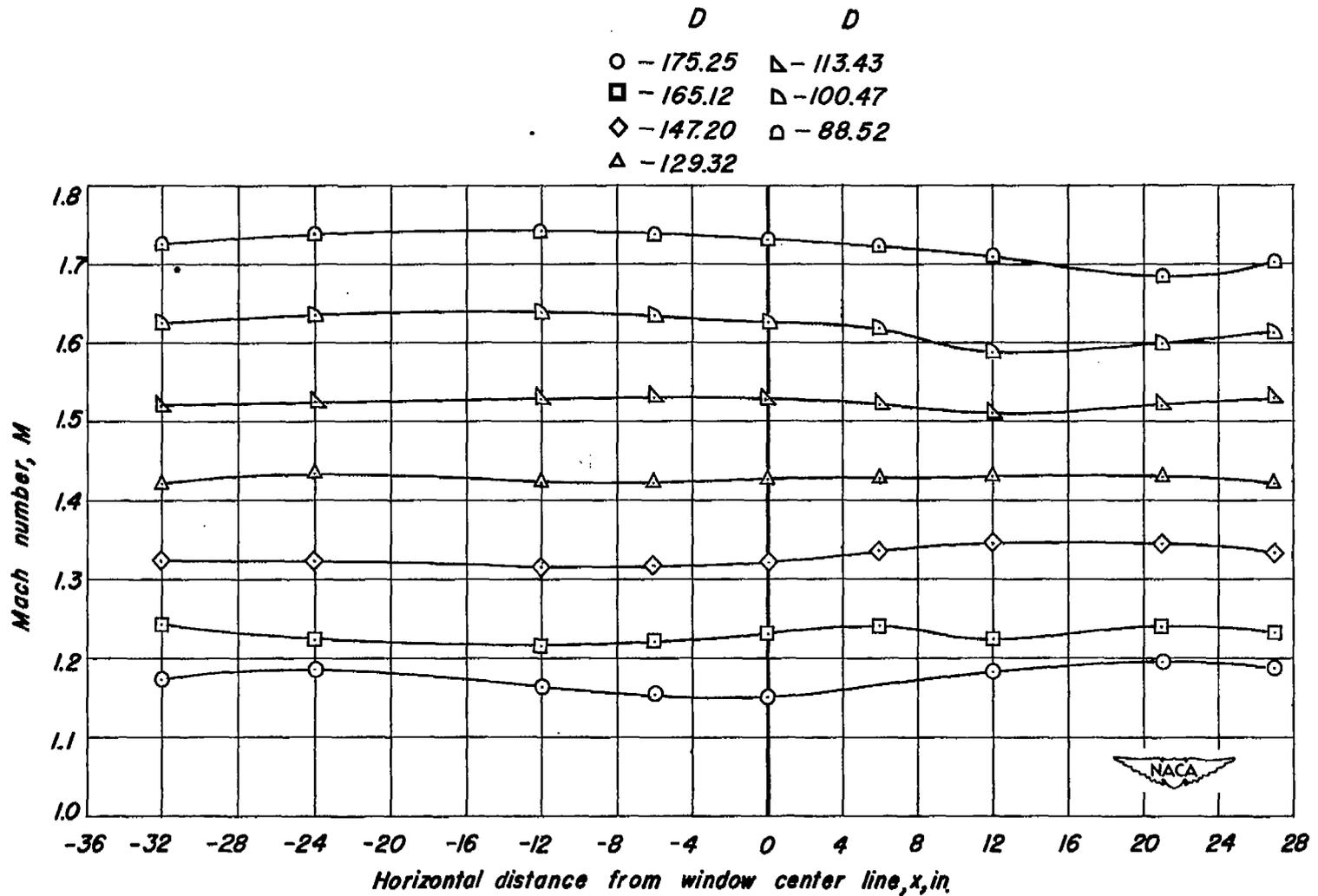


Figure 9.—The variation of Mach number along the horizontal center line of the Ames 6- by 6-foot supersonic wind tunnel. Stagnation pressure = 9 lb/sq in. abs.

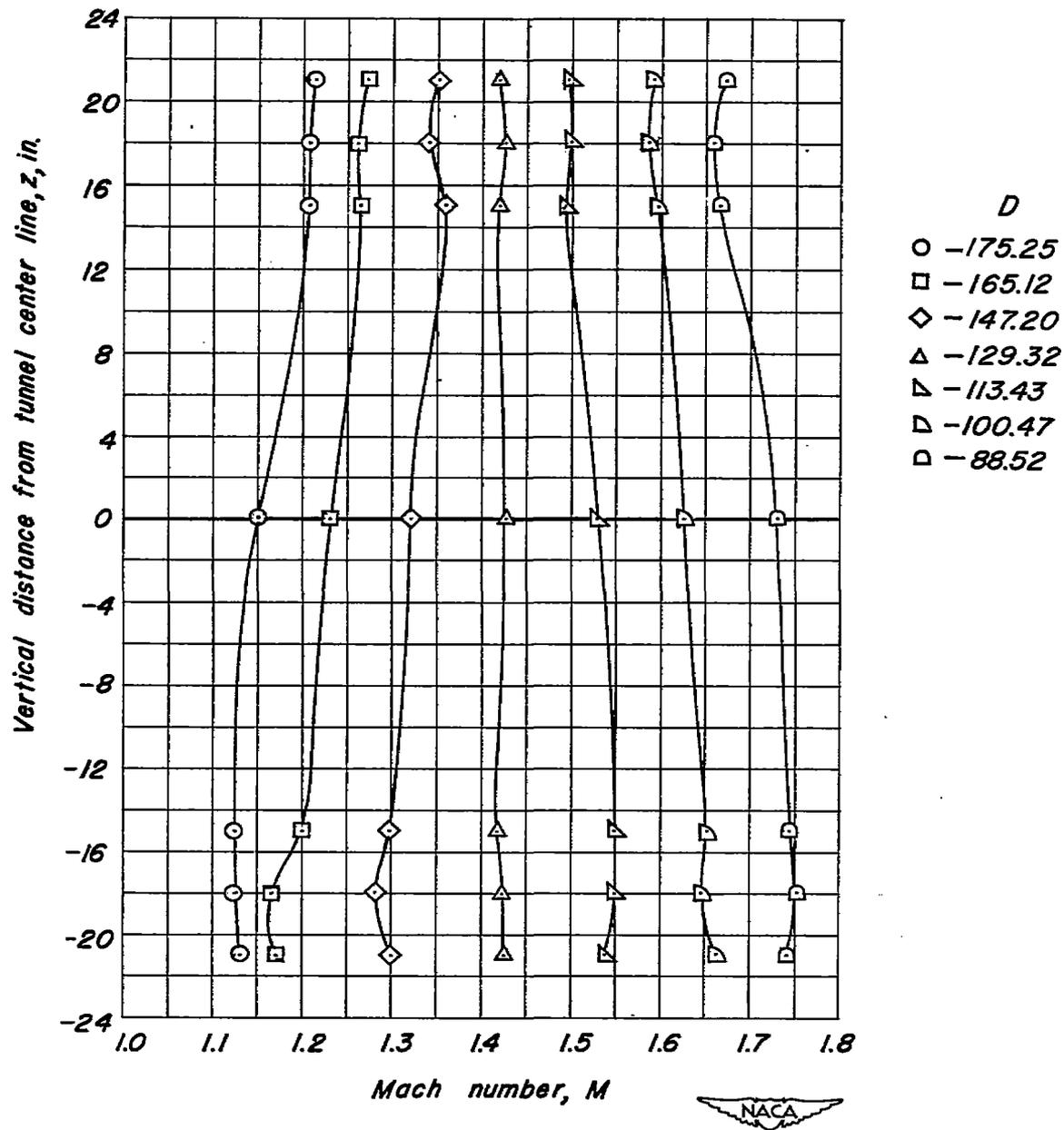
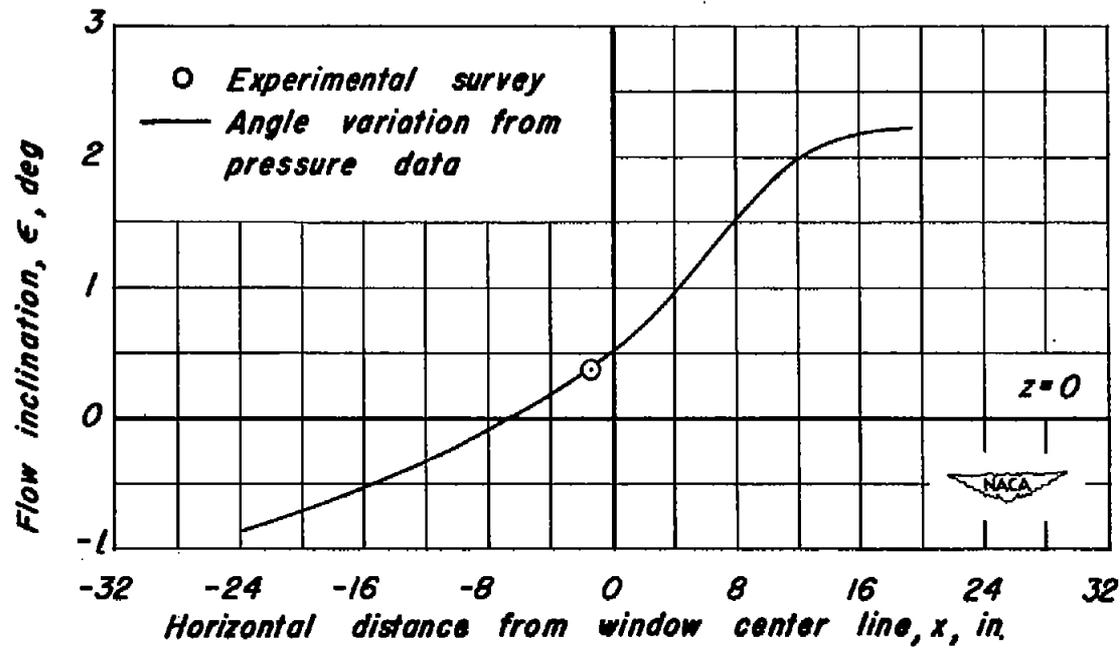


Figure 10.- The variation of Mach number vertically at the center line of the test section of the Ames 6-by 6-foot supersonic wind tunnel. Stagnation pressure = 9 lb/sq in. abs.



(a)  $D=165.12$ ;  $M=1.23$ .

Figure 11.— The variation of stream angle axially in the Ames 6- by 6-foot supersonic wind tunnel.  $y=0$ ; stagnation pressure = 9 lb/sq in. abs.

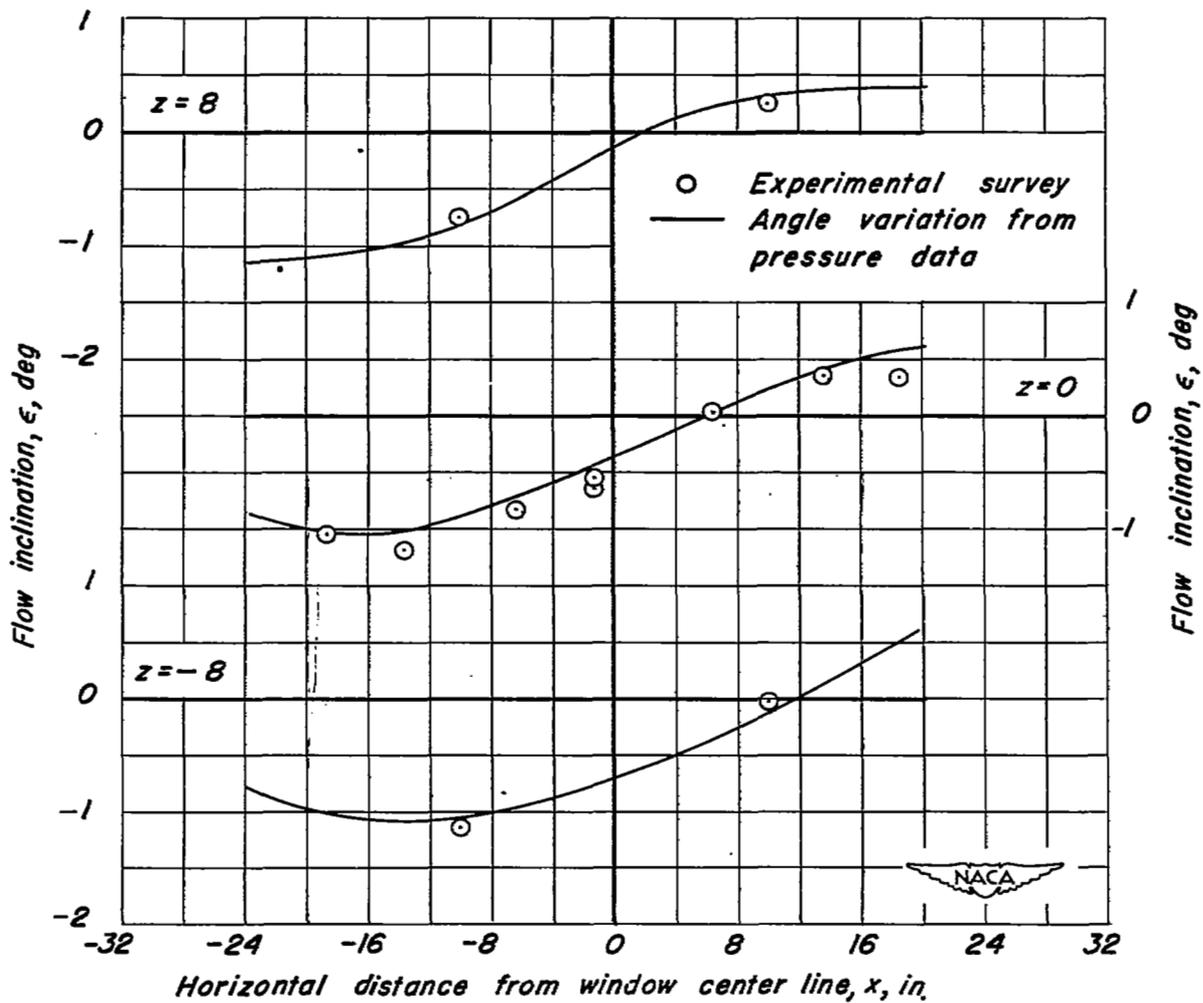


Figure 11. - Continued.

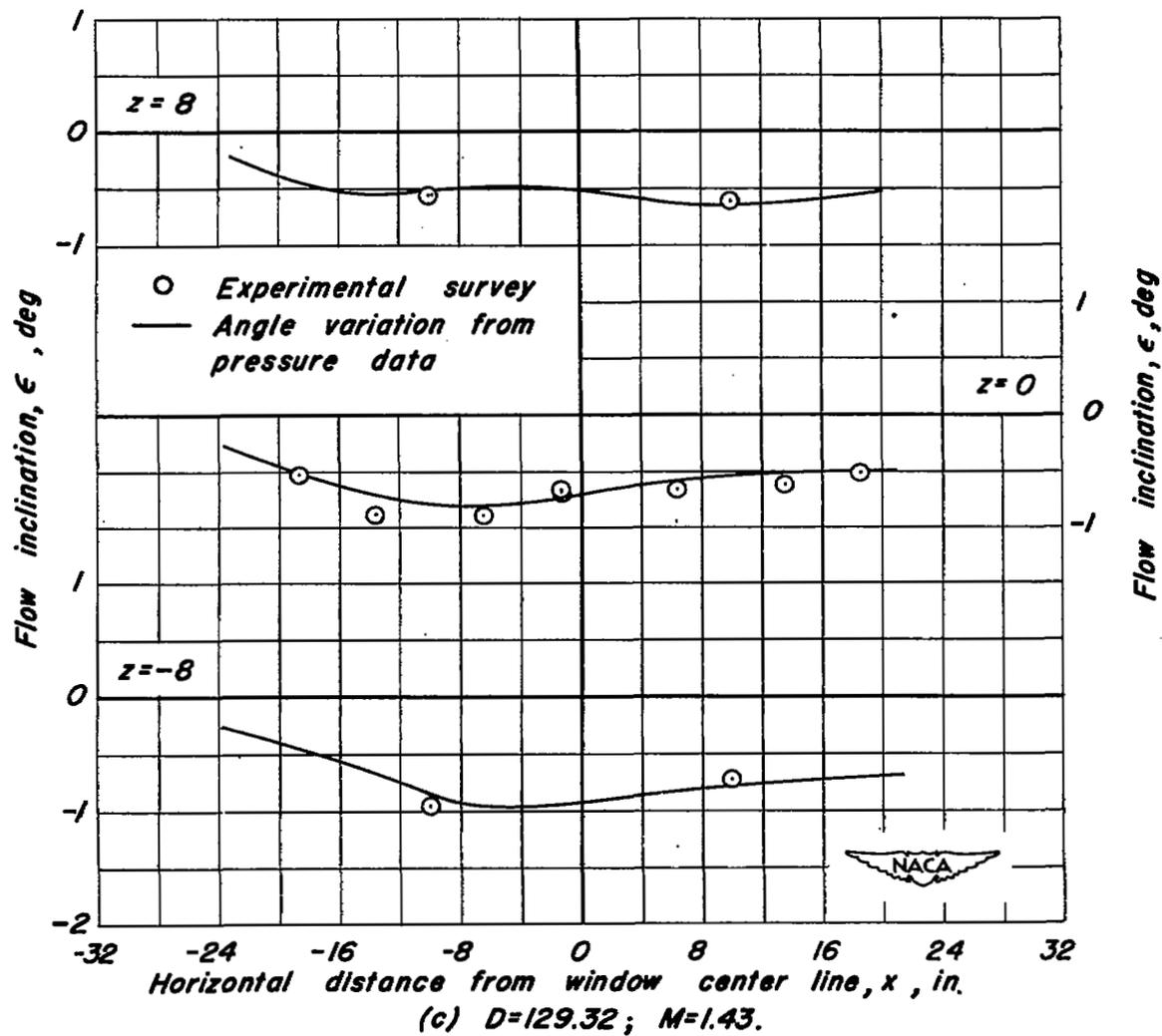
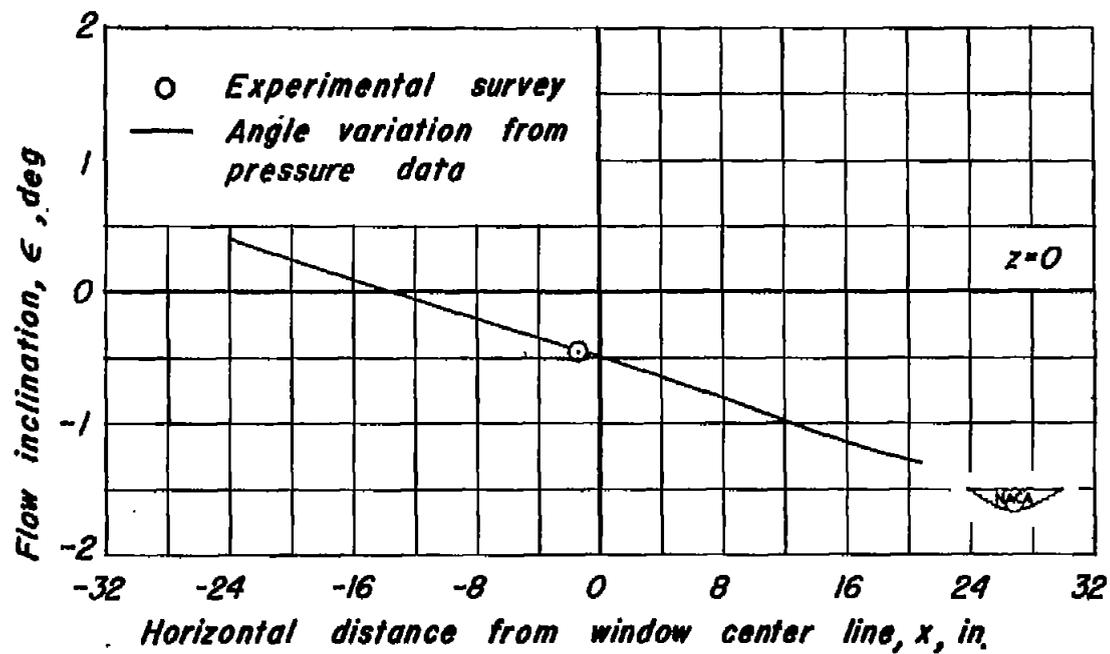


Figure 11.—Continued.



(d)  $D=113.43$ ;  $M=1.53$ .

Figure 11.-Continued.

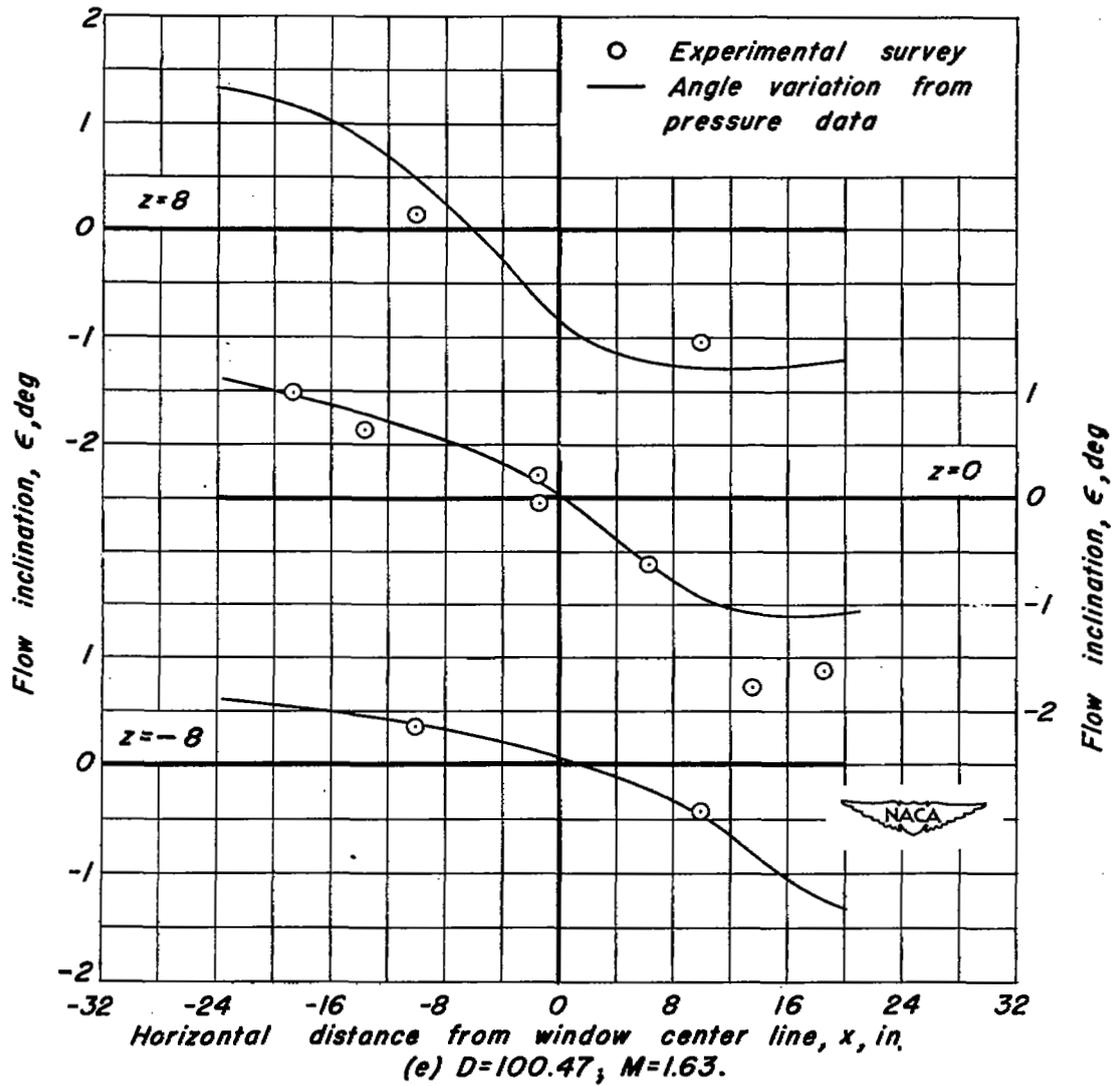


Figure 11.-Continued.

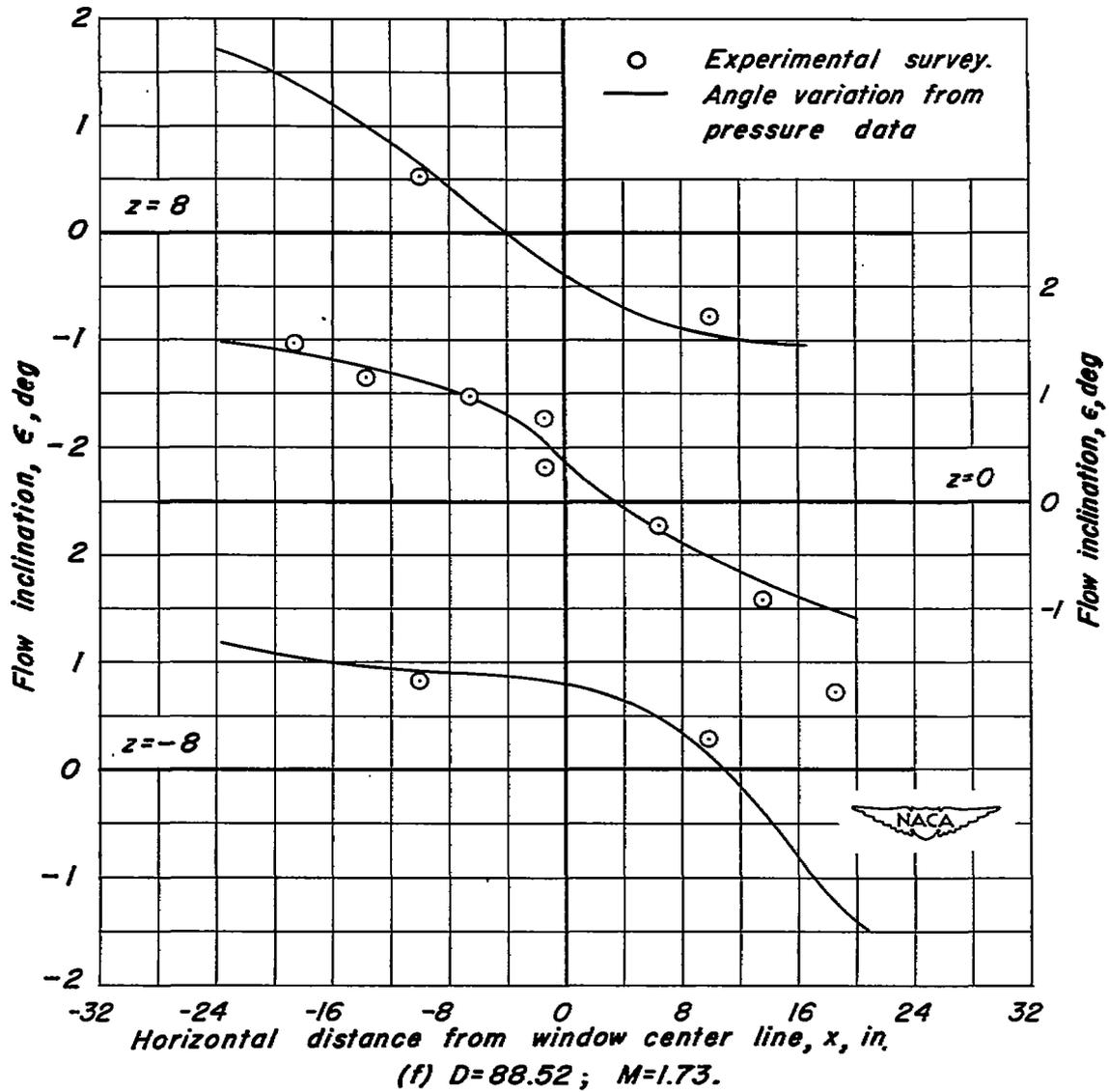
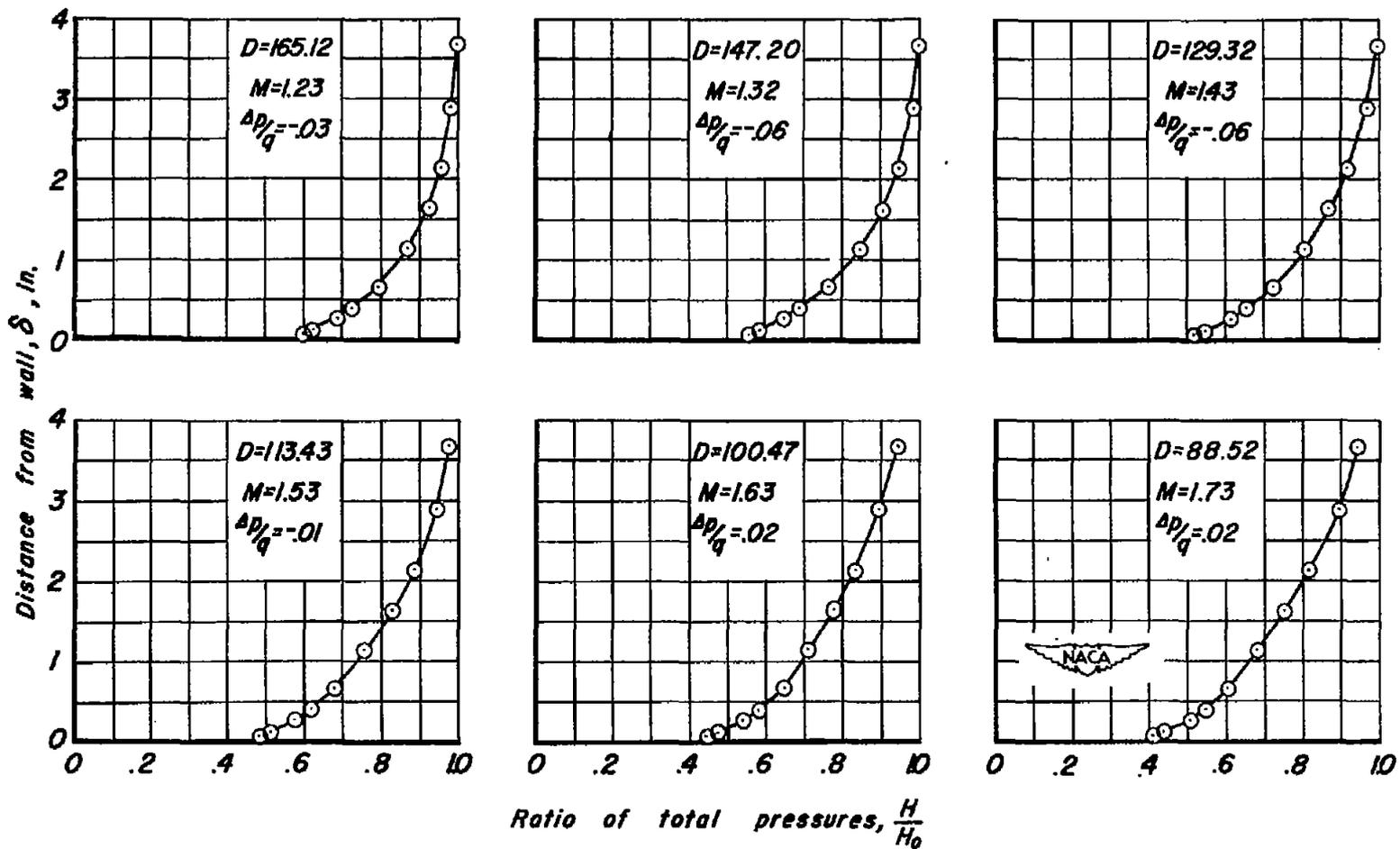
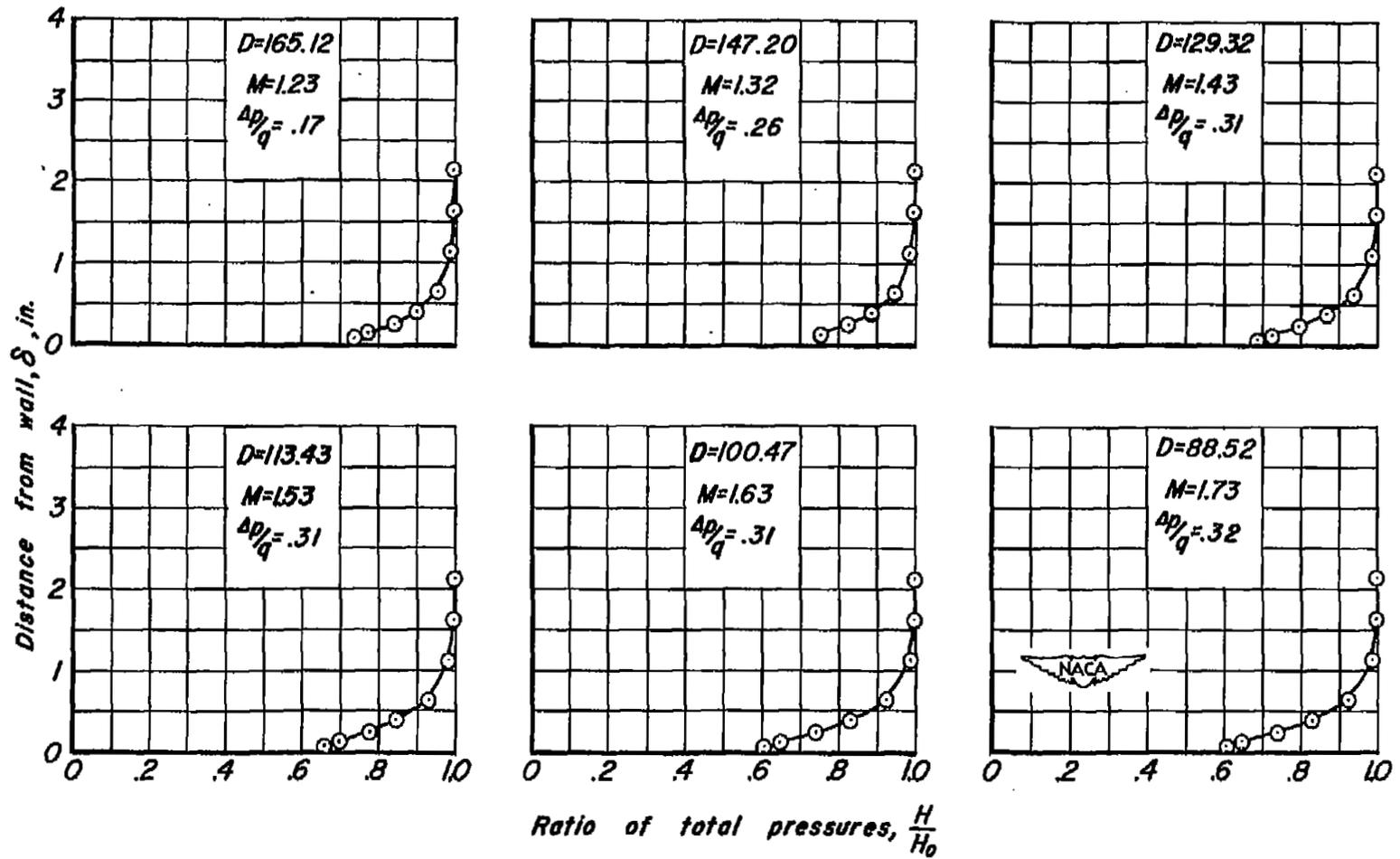


Figure II.—Concluded.



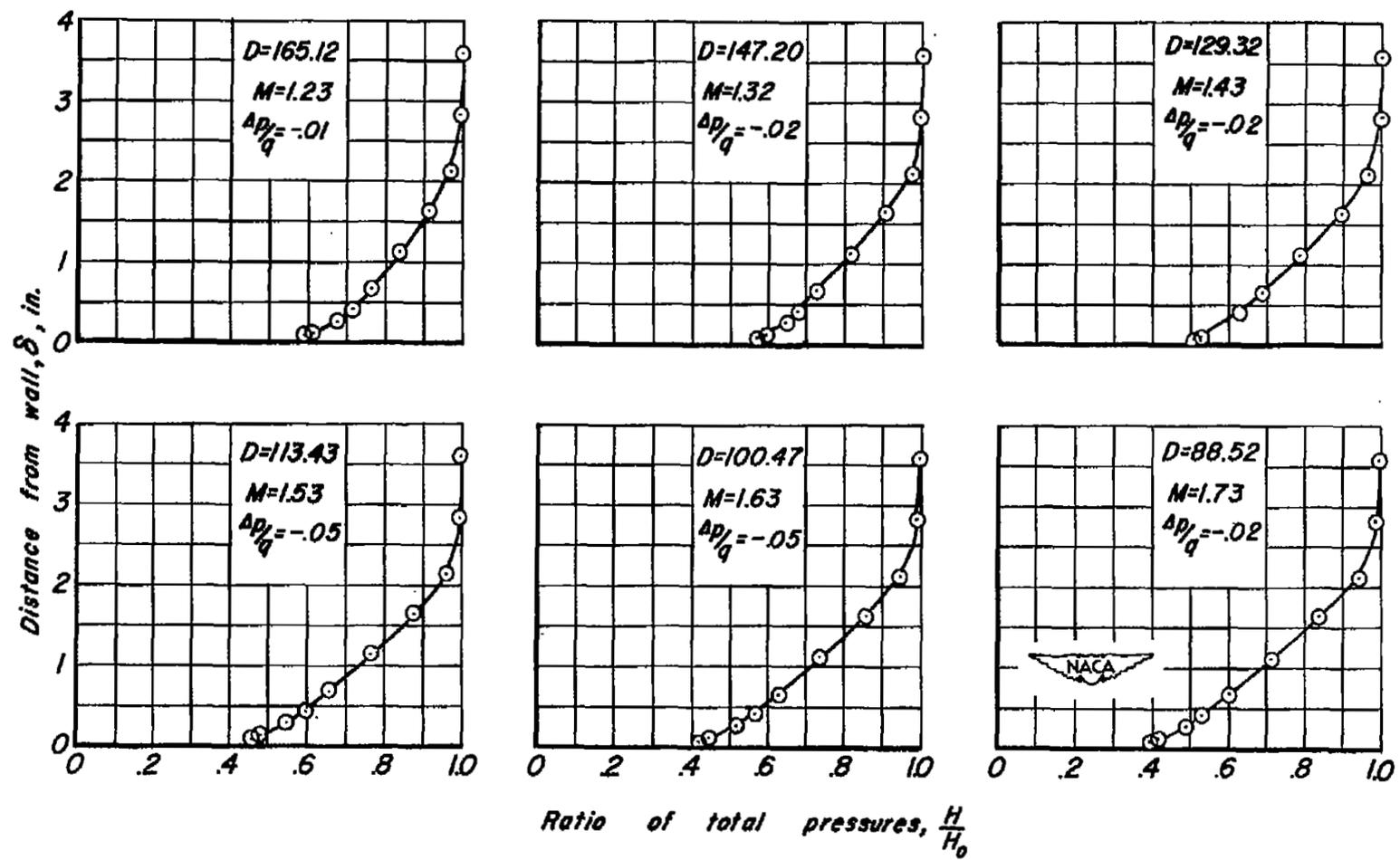
(a) Upper rake.

Figure 12.— The boundary-layer profiles on the nozzle walls of the Ames 6-by 6-foot supersonic wind tunnel. Stagnation pressure = 9 lb/sq in. abs.



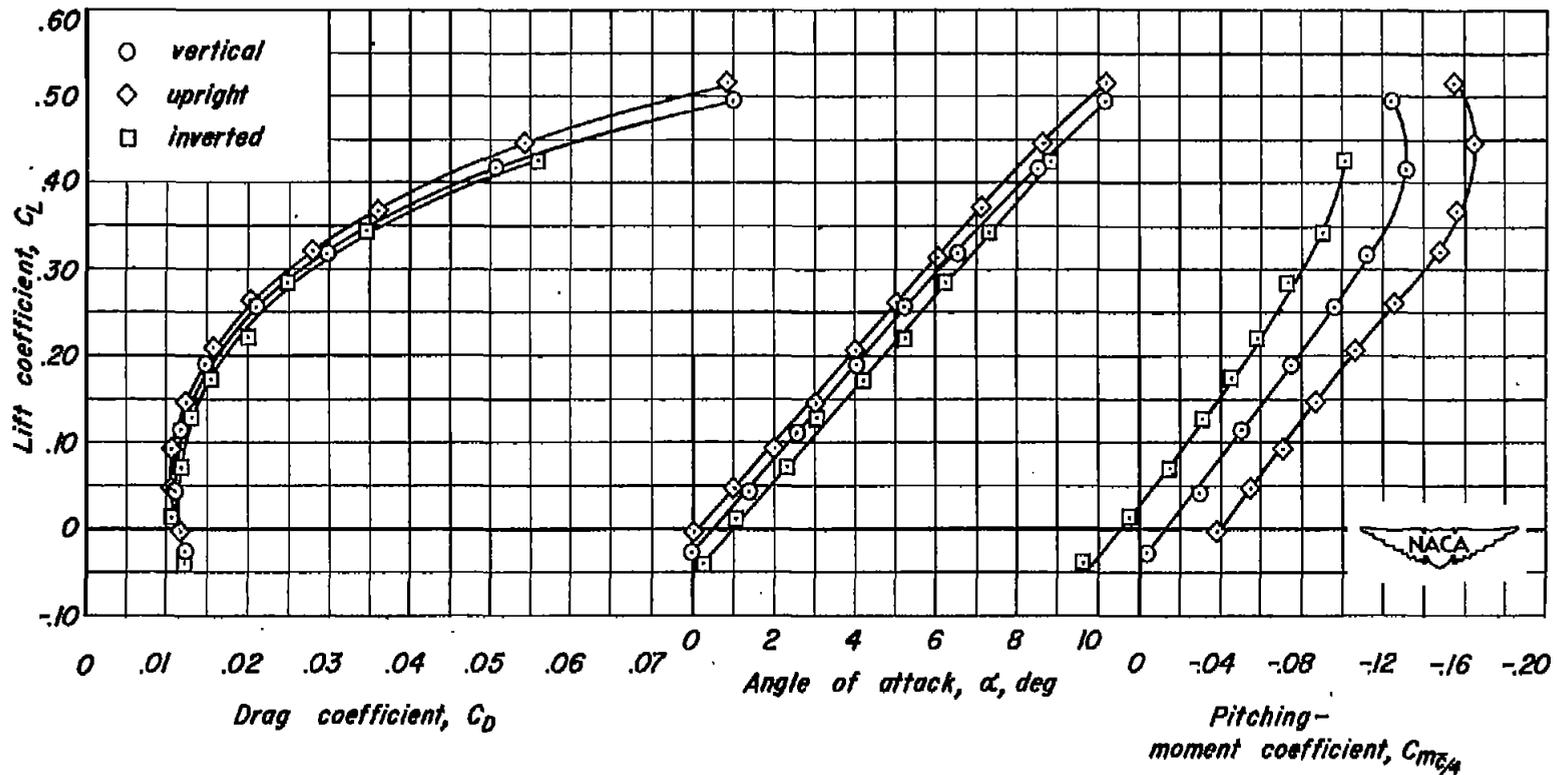
(b) Lower front rake.

Figure 12.-Continued.



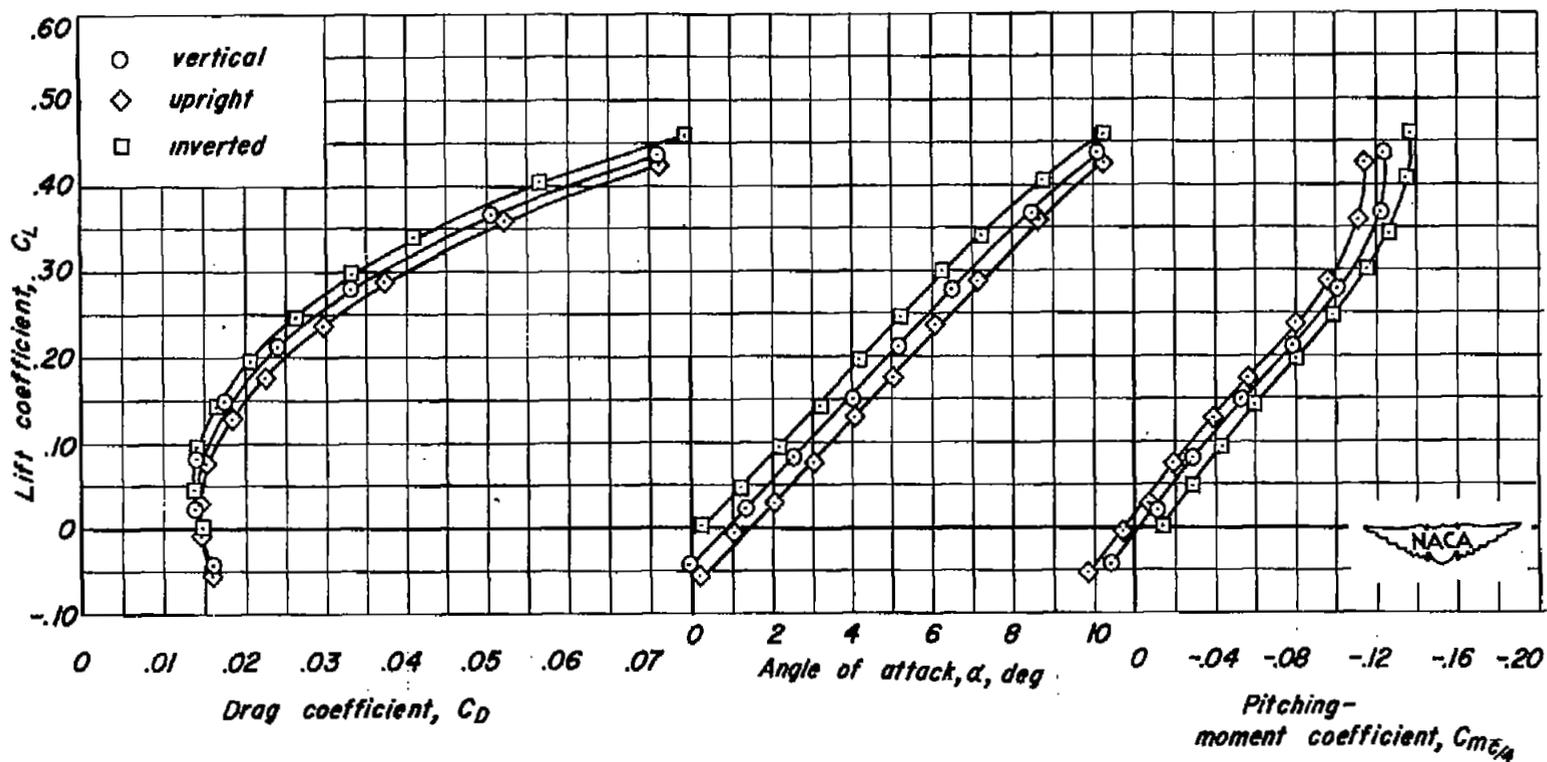
(c) Lower rear rake.

Figure 12.- Concluded.



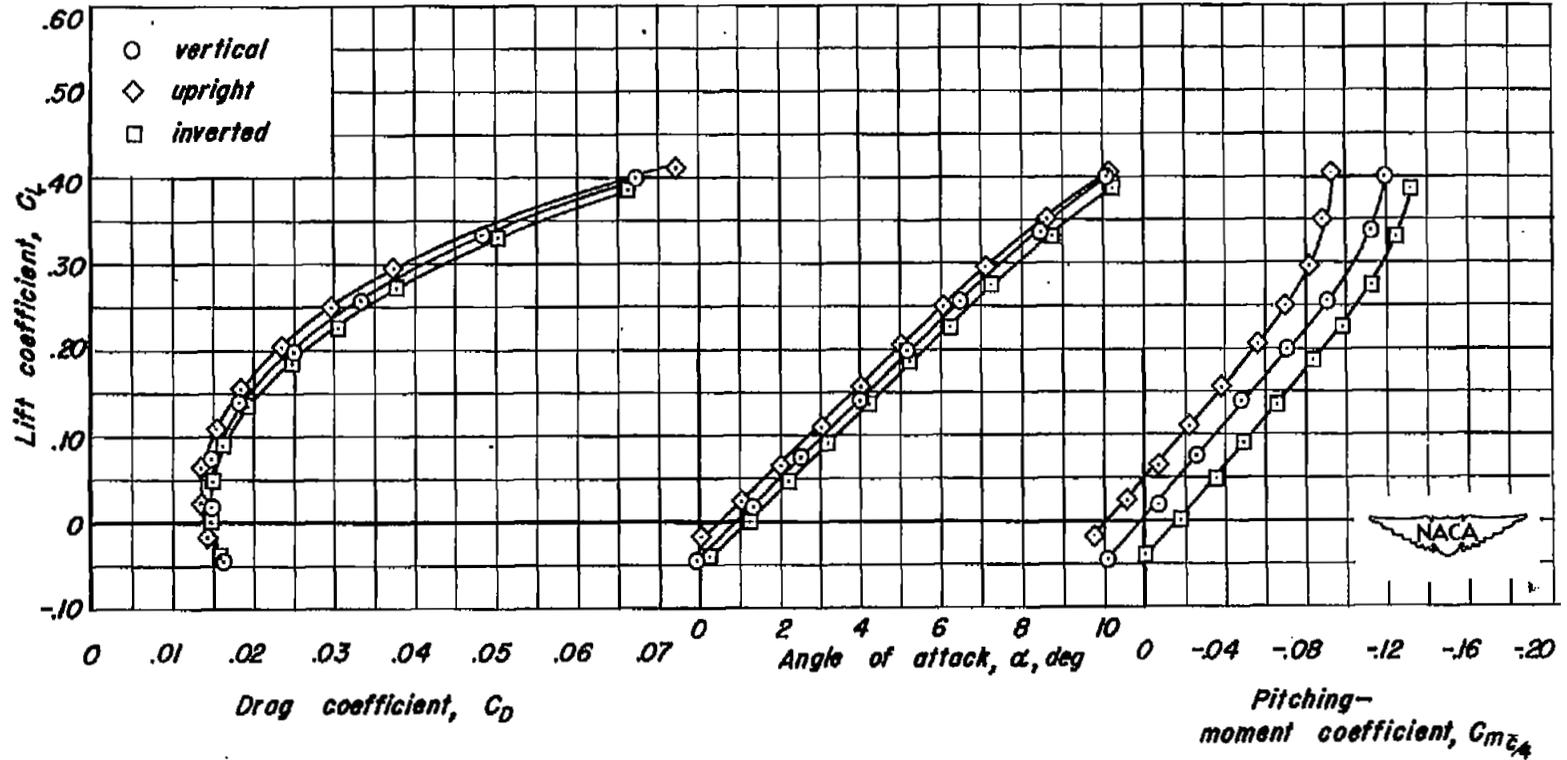
(a)  $D=165.12$ ;  $M=1.23$ .

Figure 13.-Aerodynamic characteristics of a wing having the leading edge swept back  $63^\circ$  from tests of the wing in the vertical, upright, and inverted positions in the Ames 6- by 6-foot supersonic wind tunnel.



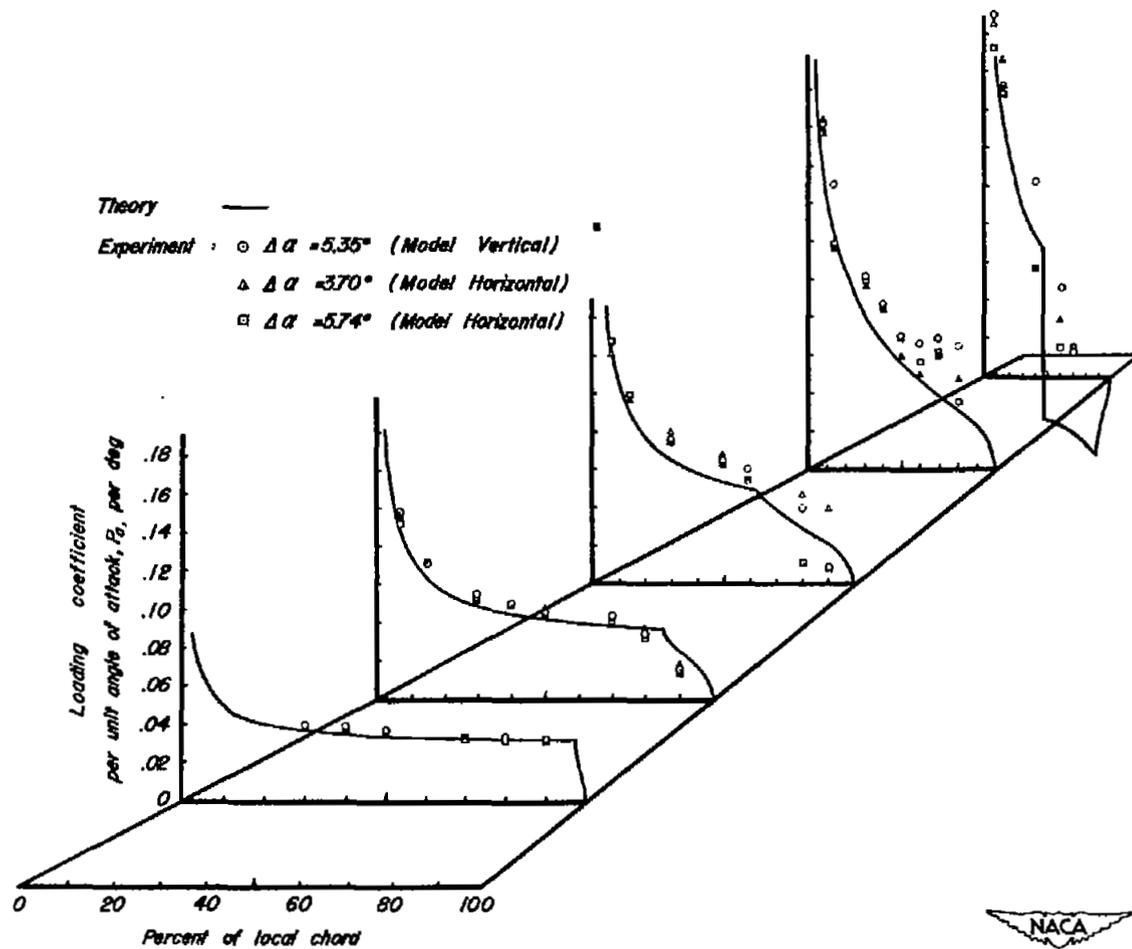
(b)  $D=113.43$ ;  $M=1.53$ .

Figure 13.—Continued.



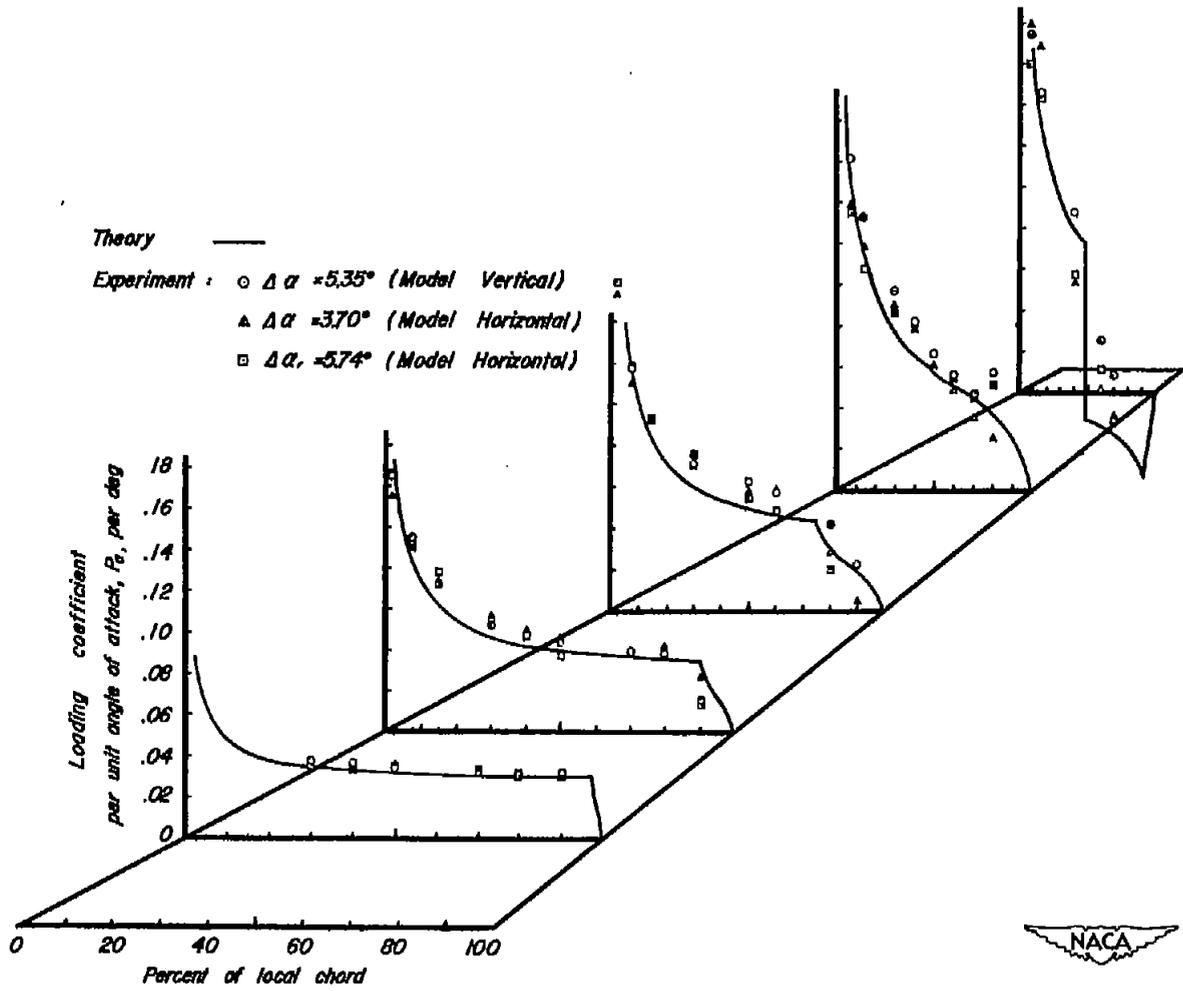
(c)  $D=88.52$ ;  $M=1.73$ .

Figure 13.- Concluded.



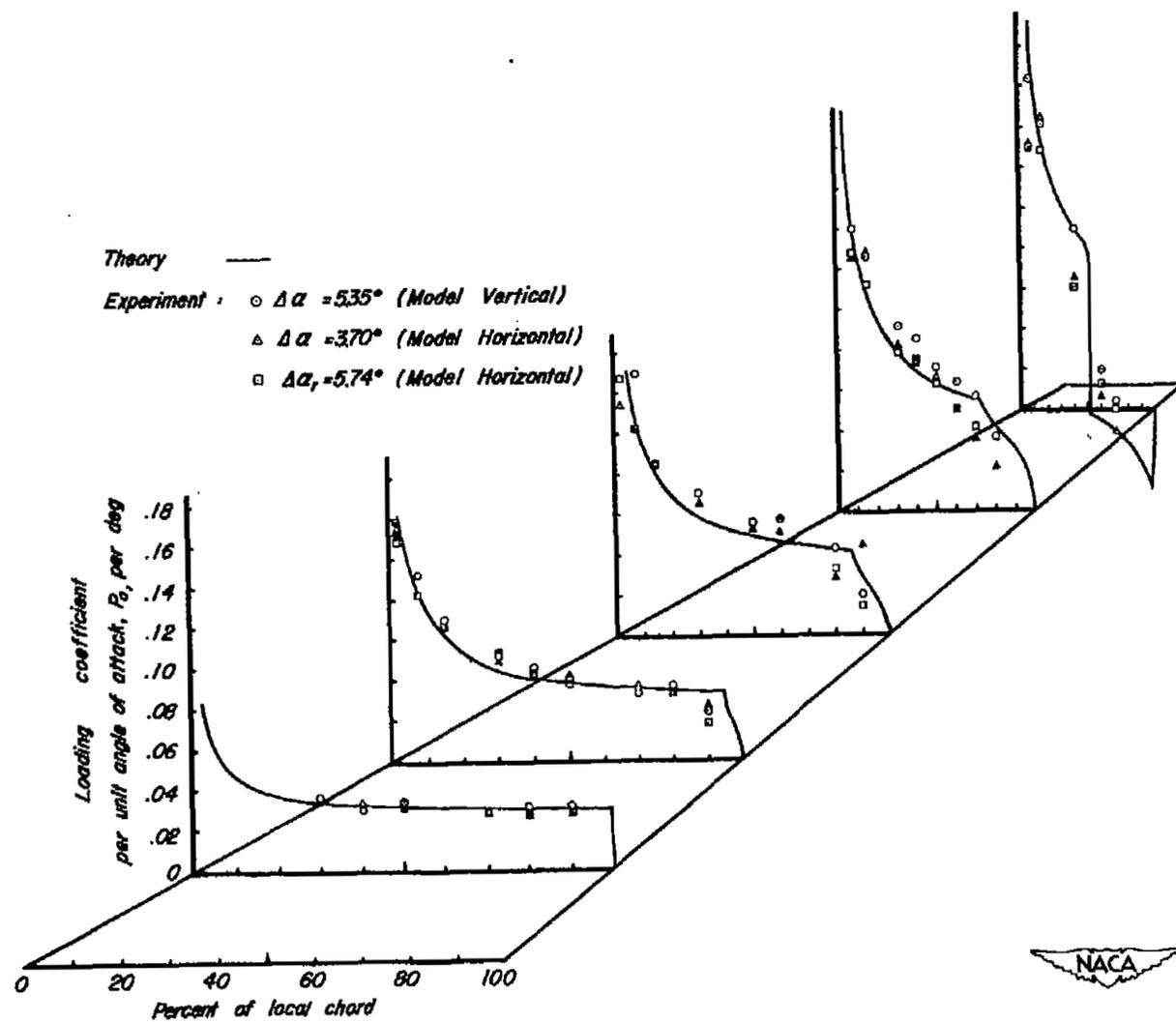
(a)  $M=130$ .

Figure 14.— Load distribution on a wing with the leading edge swept back  $63^\circ$ .



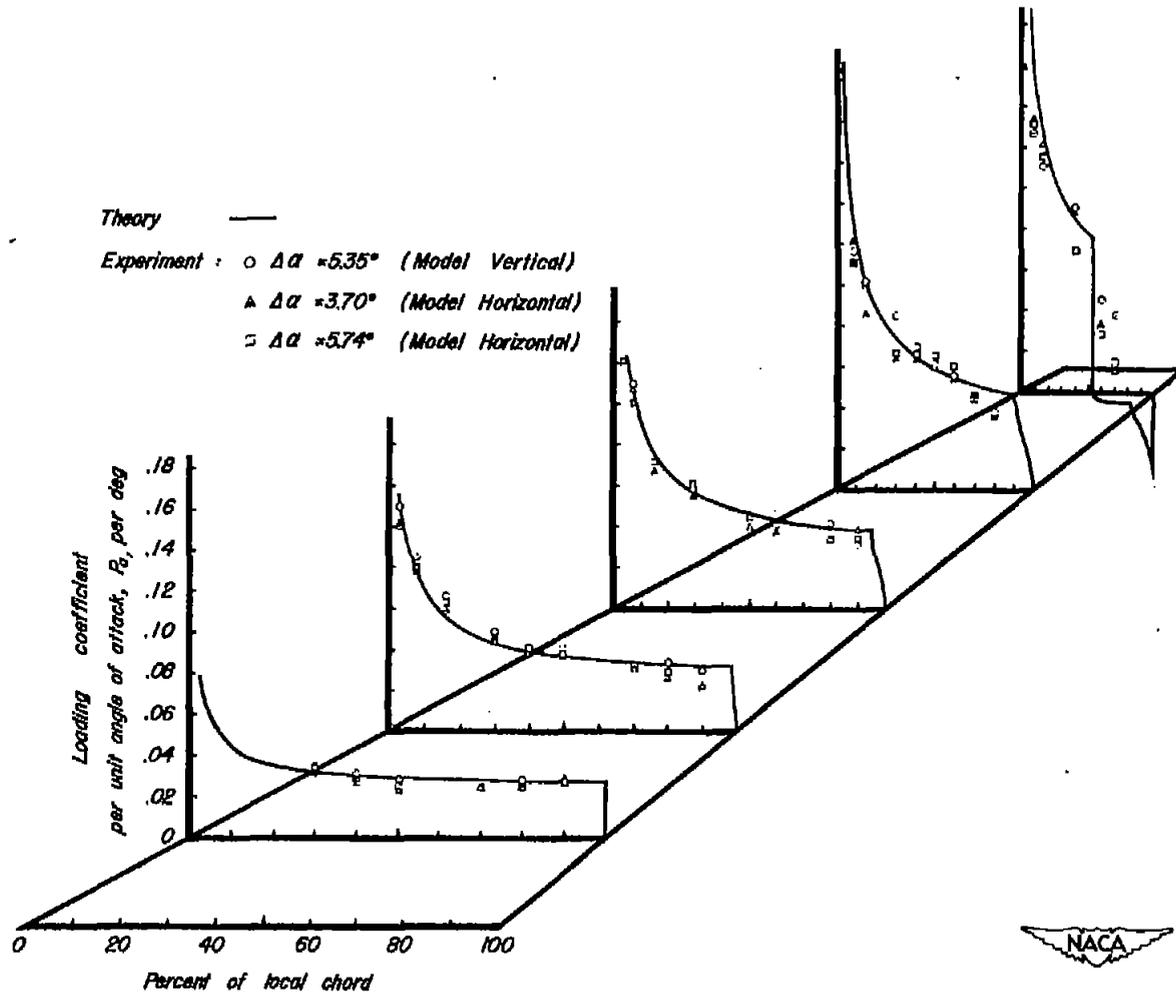
(b)  $M = 1.40$ .

Figure 14.- continued.



(c)  $M=1.50$ .

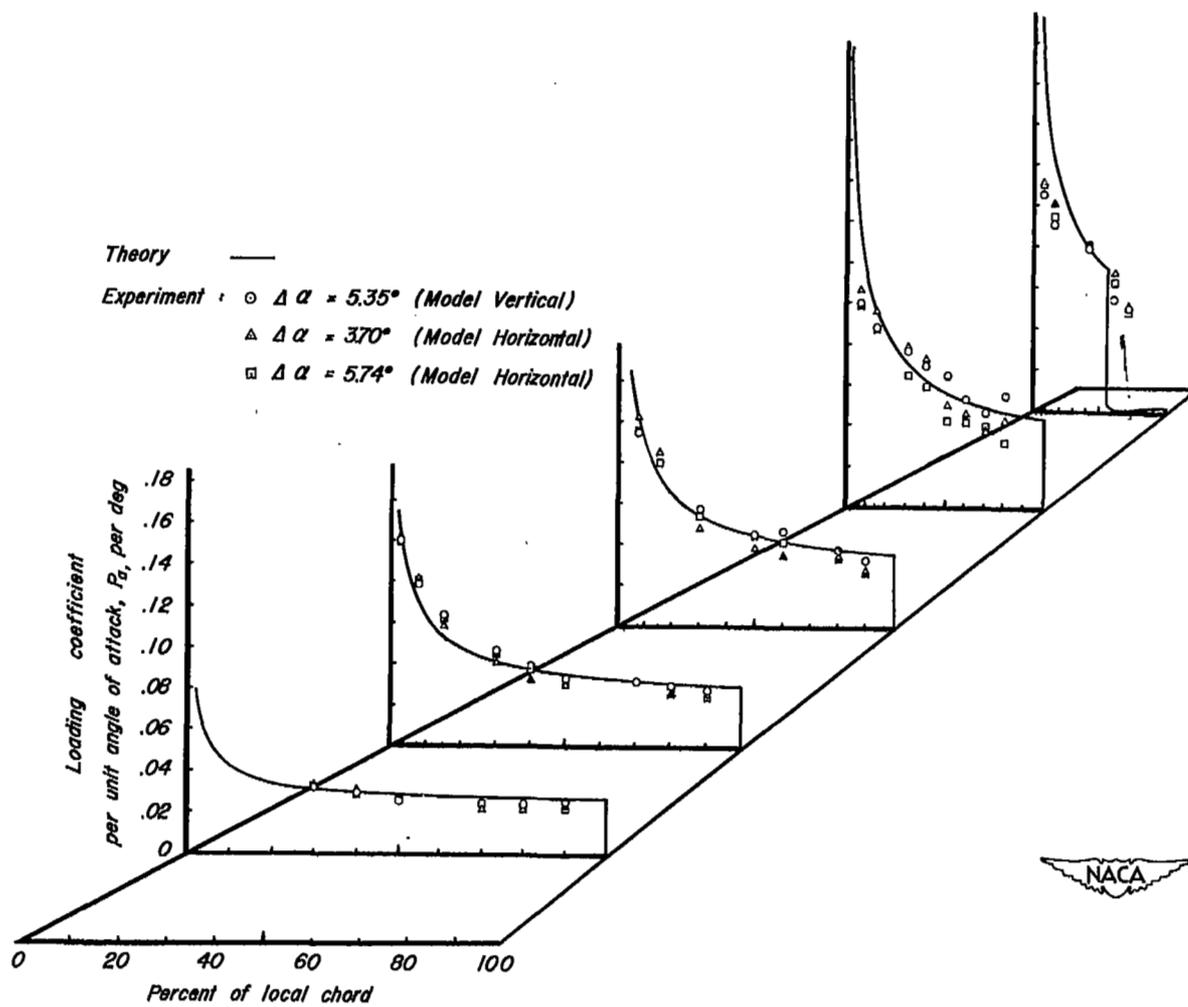
Figure 14.- continued.



(d)  $M=1.60$ .

Figure 14.— continued.





(e)  $M=1.70$ .

Figure 14.- concluded.

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