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TECHNICAL NOTE

D-1297

AN INVESTIGATION OF THE NORMAL-FORCE AND VORTEX-WAKE
CHARACTERISTICS OF AN OGIVE-CYLINDER

BODY AT SUBSONIC SPEEDS

By Bruce E. Tinling and Clyde Q. Allen

Ames Research Center
Moffett Field, Calif.

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SUMMARY

Measurements of the normal-force and vortex-wake characteristics of a body of revolution were made at subsonic speeds to supplement those reported in NACA Report 1371 for $M = 1.98$. The body tested had an ogival nose 3 diameters long, tangent to a cylindrical afterbody 7.7 diameters long. Results are presented for Mach numbers from 0.3 to 0.95 at the same Reynolds number (0.44×10^6) as the supersonic results. The effects of an increase in the Reynolds number to 3.0×10^6 at a low subsonic Mach number were also studied.

INTRODUCTION

Methods have been developed for estimating the interference effects between the vortices shed from bodies of revolution and the flow over wing and tail surfaces of aircraft. (See, e.g., ref. 1.) The application of these methods requires knowledge of the positions and strengths of the vortices as they develop along the body. Experimental information of this type was obtained in the investigations of references 2 and 3 at supersonic speeds for a body consisting of a circular-arc ogival nose (fineness ratio 3), tangent to a cylindrical afterbody 7 diameters long. It was concluded in reference 3 that the paths of symmetrically shed body vortices can be computed using two-dimensional incompressible flow theory, provided the normal-force distribution and the starting positions and strengths of the vortices are known. Also, it appeared that the strengths of the vortices could be estimated from knowledge of the normal-force distributions and vortex positions. Because there is no theoretical method for determining the origin of the vortices, these starting conditions must be determined experimentally.

The purpose of this report is to extend the information on the ogive-cylinder to subsonic speeds. A body geometrically similar to that described above was tested at Mach numbers from 0.30 to 0.95. The body

dimensions were selected so that the Reynolds number (0.44×10^6) was the same as that for the supersonic results. The data have been reduced in a manner similar to that employed in NACA Report 1371 and are presented without a detailed analysis.

SYMBOLS

A	reference area, πa^2
a	radius of the afterbody
C_m	pitching-moment coefficient, $\frac{\text{pitching moment}}{qAd}$
c_n	local normal-force coefficient, $\frac{r_b}{a} \int_0^{2\pi} C_p \cos \theta \, d\theta$
C_N	normal-force coefficient, $\frac{N}{qA}$
C_p	pressure coefficient, $\frac{p_l - p}{q}$
d	body diameter
l	body length
l_n	length of nose
M	free-stream Mach number
N	normal force
p_l	local static pressure
p	free-stream static pressure
q	free-stream dynamic pressure
R	free-stream Reynolds number based on body diameter
r_b	radius of the body in planes normal to the body axis
r_*	radius of the core of the viscous vortex

V_∞	free-stream velocity
x, r, θ	cylindrical coordinates of a point in space, origin at nose vertex, x axis coincident with body axis, and $\theta = 0^\circ$ in the vertical plane of symmetry on the windward side
α	angle of attack, deg
Γ	circulation strength
x, ξ, η	Cartesian coordinates of vortex filament position (referred to the body axis with the origin at the nose)
ϕ, ψ	flow angles measured from a normal to the crossflow plane at the survey probe axis; ϕ positive for radial flow in the direction of increasing r , ψ positive for tangential flow in the direction of increasing θ

MODEL

The dimensions and details of the model are shown in figure 1. Pressure orifices were installed in two rows on opposite meridian lines. The model was sting supported as shown in figure 2 and could be remotely rotated 180° about its longitudinal axis while the wind tunnel was in operation.

TESTS

The tests were conducted in the Ames 12-Foot Pressure Wind Tunnel. Surface pressures were measured at a Reynolds number of 0.44×10^6 at Mach numbers of 0.30, 0.80, and 0.95, and at a Reynolds number of 3.0×10^6 at a Mach number of 0.30. The model was tested at angles of attack from 5° to 24° . At each angle at attack, circumferential pressure distributions were obtained by rotating the model through 180° in increments of 10° .

Surveys of the flow above the body were made with the survey rake and the directional probe shown in figure 1. The surveys with the rake to determine vortex center positions were made at $x/d = 4.5, 6.0, 7.5,$ and 10.0 for all test conditions. Surveys with the directional probe to determine local flow directions in the vortex wake were made only at $\alpha = 20^\circ$ for $M = 0.30, R = 0.44 \times 10^6$.

Wind-tunnel wall corrections were considered negligible because the model was small compared to the test section. The sting deflection due to aerodynamic loads is estimated to be less than 0.1° for all test conditions. No corrections were applied to the data to account for this deflection.

RESULTS AND DISCUSSION

The results of the wind-tunnel tests have been reduced in a manner to parallel the information contained in references 2 and 3. Results pertaining to the loading over the body are contained in figures 3 through 8. The results pertaining to the vortex wake measurements and the calculated vortex strength are contained in figures 9 through 12.

Pressure Distribution and Normal Force

The circumferential pressure distribution measured at several longitudinal positions on the ogival nose and cylindrical afterbody are shown in figure 3. The results for a Mach number of 1.98 (ref. 2) have been included in this figure for comparison. It can be seen from these results that, within the subsonic speed range, Mach number effects are confined principally to the nose of the body. At angles of attack greater than 10° , large differences between the pressure coefficient measured at subsonic speeds and at $M = 1.98$ are evident over the entire body. The corresponding changes in the local normal-force coefficient are illustrated in figure 4 where the results of integrations of the pressures in the crossflow planes are presented.

The effects of increasing Reynolds number from 0.44×10^6 to 3.0×10^6 on the loading at a Mach number of 0.30 are shown in figure 6. Limitations imposed by the wind tunnel did not permit data to be obtained at Reynolds numbers significantly greater than 0.44×10^6 at the higher subsonic Mach numbers. The differences between the circumferential pressure distributions obtained at the two Reynolds numbers are small. The integrated effect of these changes, however, was sufficient to change the loading on the afterbody by 100 percent or more. (See fig. 7.) As noted in the section on tests, the model support system was sufficiently stiff that differences in angle of attack caused by the increased dynamic pressure at the larger Reynolds number were negligible.

Vortex-Core Location and Strength

Characteristically, the flow field near the upper surface of a body of revolution with a pointed nose at an angle of attack has a pair

of vortices extending downstream from some starting point behind the nose. Measurements of pitot pressure in the wake provided patterns which indicated the positions of the vortex pair at several positions along the body. These data, however, do not identify the origin of the vortex. As pointed out in reference 3, this origin can be detected from the surface pressure distribution. The surface pressure data were analyzed by a technique similar to that described in reference 3 to locate the approximate origin of the vortices presented in figure 9 as a function of angle of attack of the body. Also shown is the most forward position at which the vortex was identified from the wake surveys. Further information on the position of the vortex core is given in figure 10 and table I. The axial position where the vortex was considered to originate was approximately the same at all subsonic speeds and was not markedly different at $M = 1.98$. (See fig. 9.) The vortex-core position further downstream, however, was consistently closer to the plane of symmetry at subsonic speeds than at $M = 1.98$.

The measured vortex-core positions in the crossflow plane and the measured normal force were used to calculate the circulation strength by equation (10) of reference 3. In this computation, the integrated normal force from the nose to a given crossflow plane, in excess of the value given by slender-body theory, is assumed to be related to the vortex strength according to the Kutta-Joukowski law. The results of this computation are shown in figure 11. As would be anticipated from the measured normal-force characteristics, the vortex strength at a Mach number of 0.30 was greatly increased by a change in Reynolds number from 0.44×10^8 to 3.0×10^8 (cf. figs. 11(a) and 11(b)). Increases in the Mach number at subsonic speeds tended, in general, to increase the vortex strength. The much larger normal force over the afterbody at $M = 1.98$ results in much greater vortex strength than at subsonic speeds. This is illustrated in figure 11(d) where the vortex strength for $\alpha = 20^\circ$ at $M = 1.98$ is shown to be of the order of twice that at the same angle of attack at $M = 0.95$.

One set of measurements was made with the survey probe shown in figure 1(d) to determine both the direction and magnitude of the local velocities within the vortex wake. The total and local vortex strength within the wake were calculated from these measurements. The results showed that the total circulation over the upper half of the crossflow plane agreed well with the value calculated from the normal force and core position (see fig. 11(a)). However, the circulation was found to be distributed over nearly the entire region of the survey rather than concentrated in the region of the vortex core. This result is to be expected since the vortex is in reality the rolled-up vortex sheet resulting from separation of the boundary layer over the body. The flow angles measured during the survey are compared in figure 12 with results of calculations based on two-dimensional cylinder potential with either

potential or viscous vortices. (See eqs. (4) through (7) of ref. 3.) It can be seen that the velocities induced by the rolled-up vortex sheet are approximated fairly well over the entire wake if viscous vortices are used in the computation.

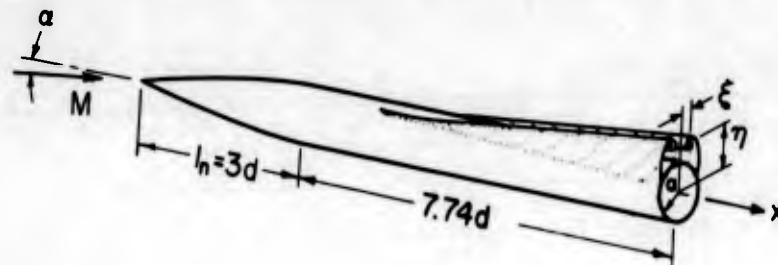
Ames Research Center
National Aeronautics and Space Administration
Moffett Field, Calif., Jan. 9, 1962

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1. Nielsen, Jack N., and Kaattari, George E.: The Effects of Vortex and Shock-Expansion Fields on Pitch and Yaw Instabilities of Supersonic Airplanes. IAS Preprint No. 743, 1957.
2. Perkins, Edward W., and Jorgensen, Leland H.: Comparison of Experimental and Theoretical Normal-Force Distributions (Including Reynolds Number Effects) of an Ogive-Cylinder Body at Mach Number 1.98. NACA TN 3716, 1956.
3. Jorgensen, Leland H., and Perkins, Edward W.: Investigation of Some Wake Vortex Characteristics of an Inclined Ogive-Cylinder Body at Mach Number 2. NACA Rep. 1371, 1958.

A
4
6
C

TABLE I.- VORTEX-CORE POSITIONS DETERMINED BY WAKE SURVEYS



M	$R \times 10^{-6}$	x/d	α , deg	Right side		Left side	
				ξ/a	η/a	$-\xi/a$	η/a
0.30	3.0	4.5	10	No core	No core	No core	No core
			15	0.49	1.02	.42	1.04
			20	.51	1.12	.44	1.13
			24	.73	1.31	.47	1.21
		6.0	10	No core	No core	No core	No core
			15	No data	No data	No data	No data
			20	.52	1.33	.45	1.35
			24	.53	1.42	.45	1.43
		7.5	10	No data	No data	.45	1.15
			15	No data	No data	.49	1.33
			20	No data	No data	.51	1.51
			24	No data	No data	.54	1.53
		10.0	10	.45	1.27	.41	1.29
			15	.46	1.51	.36	1.52
			20	.53	1.55	.42	1.53
			24	.69	1.58	.45	1.61
.30	.44	4.5	15	No core	No core	No core	No core
			20	.39	1.17	.28	1.21
			24	.29	1.27	.20	1.28
		6.0	15	No core	No core	No core	No core
			20	.37	1.16	.32	1.18
			24	.37	1.25	.32	1.25
		7.5	10	No core	No core	No core	No core
			15	No data	No data	.29	1.31
			20	.39	1.24	.37	1.25
			24	.44	1.44	.31	1.43
		10.0	10	.43	1.33	.39	1.35
			15	.52	1.43	.41	1.44
			20	.47	1.52	.45	1.46
			24	.40	1.56	.41	1.57

TABLE I.- VORTEX-CORE POSITIONS DETERMINED BY WAKE SURVEYS - Concluded

M	$R \times 10^{-6}$	x/d	α , deg	Right side		Left side	
				ξ/a	η/a	$-\xi/a$	η/a
0.80	0.44	4.5	15	No core	No core	No core	No core
			20	0.67	1.11	0.58	1.16
			24	.77	1.08	.70	1.14
		6.0	15	No core	No core	No core	No core
			20	.56	1.42	.45	1.46
			24	.67	1.80	.58	1.79
		7.5	10	No core	No core	No core	No core
			15	.36	1.40	.31	1.42
			20	.44	1.65	.40	1.67
		10.0	24	.47	2.21	.45	2.15
			10	.41	1.34	.36	1.35
			15	.45	1.48	.38	1.50
			20	.41	1.65	.32	1.67
			24	.45	2.20	.30	2.18
			.95	.44	4.5	10	No core
15	.68	.95				.64	.98
20	.76	1.08				.69	1.10
6.0	24	.83			1.06	.81	1.09
	15	No data			No data	No data	No data
	20	.59			1.51	.53	1.51
7.5	24	.80			1.83	.70	1.87
	10	No core			No core	No core	No core
	15	.48			1.32	.39	1.35
10.0	20	.47			1.80	.42	1.77
	24	.59			2.33	.55	2.31
	10	.41			1.35	.34	1.37
	15	.43			1.55	.34	1.59
	20	.45			1.80	.37	1.80
	24	.51			2.60	.36	2.53

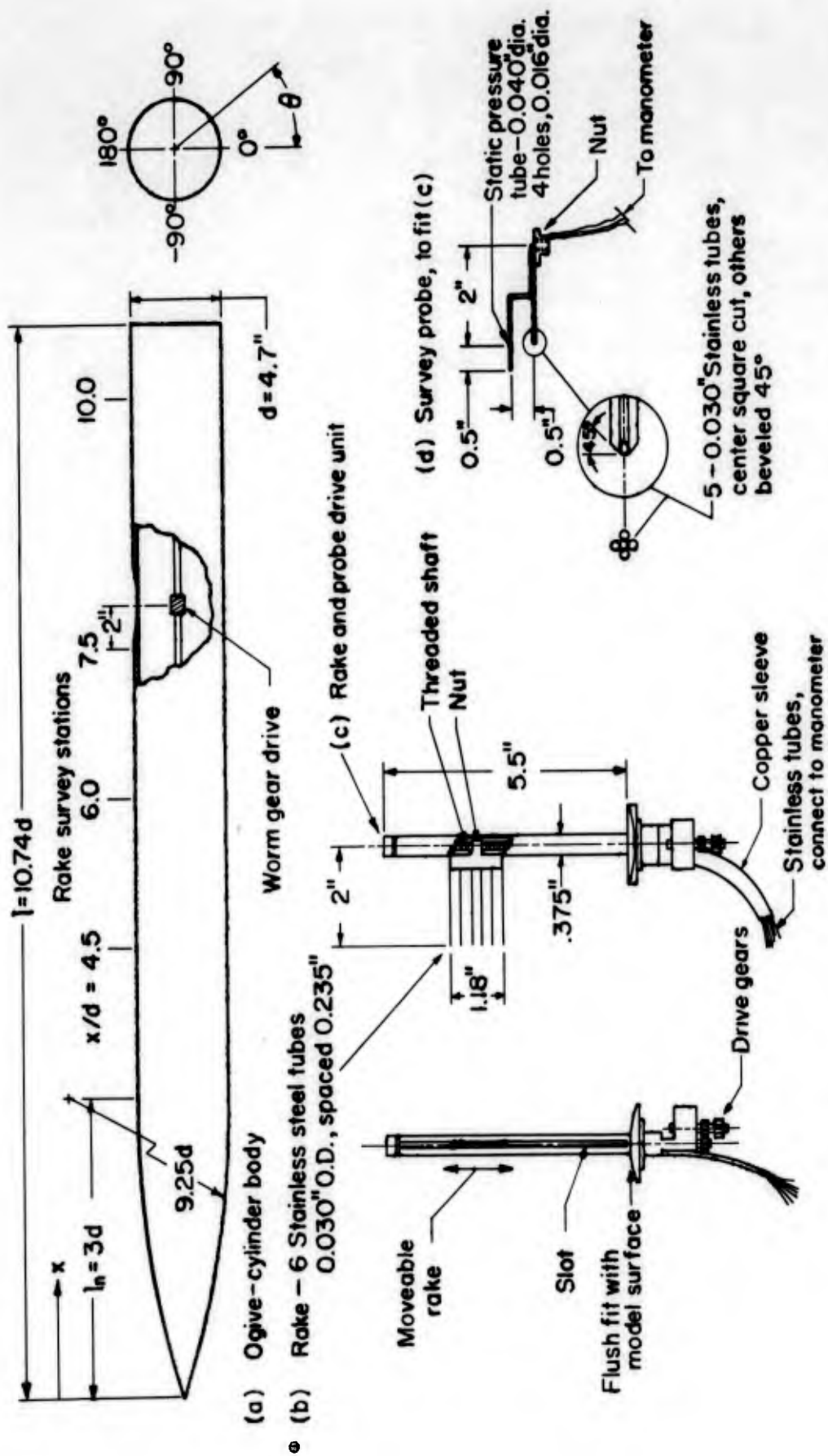
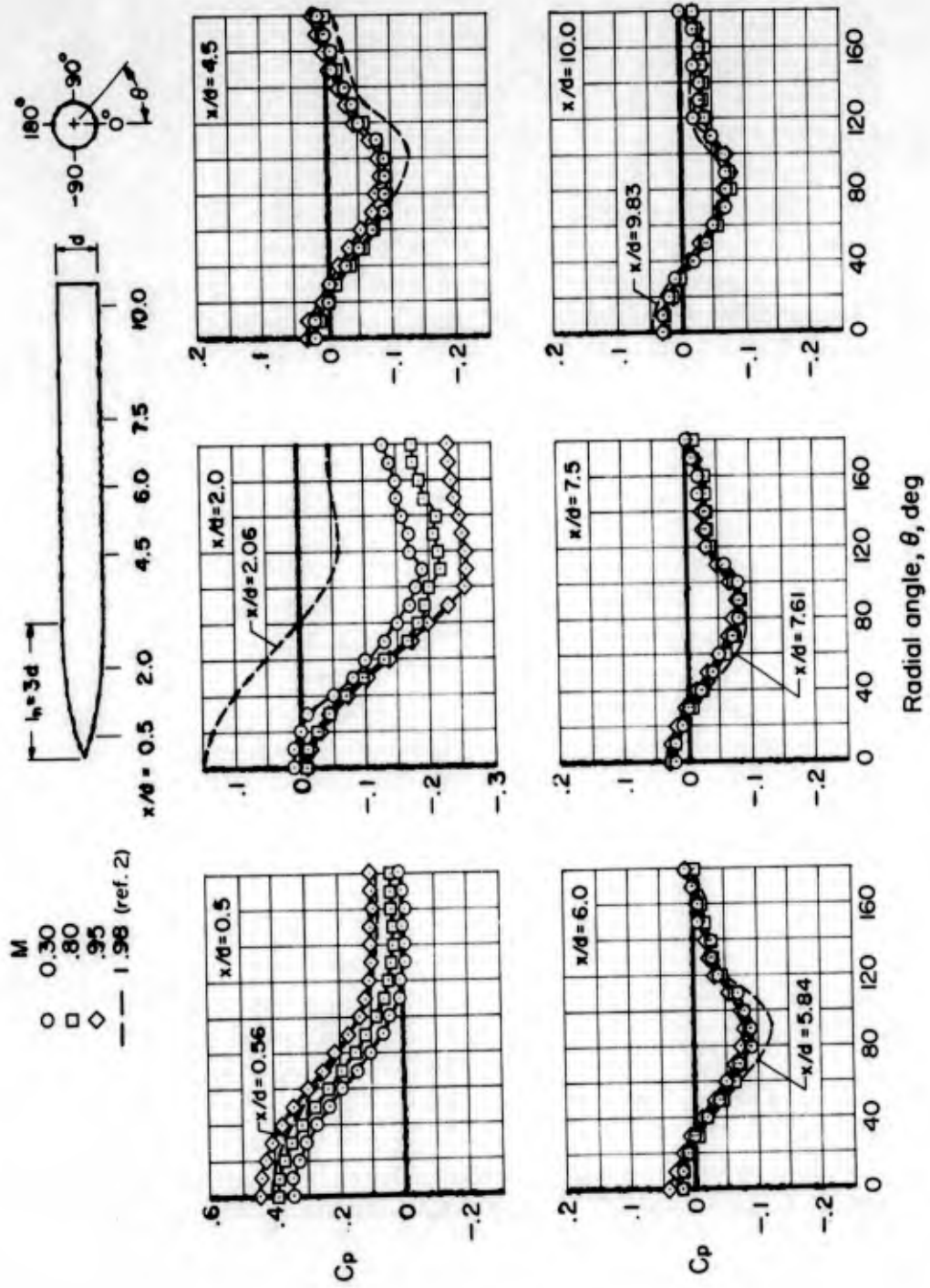


Figure 1.- The ogive-cylinder body dimensions and details of the vortex survey probe and rake.



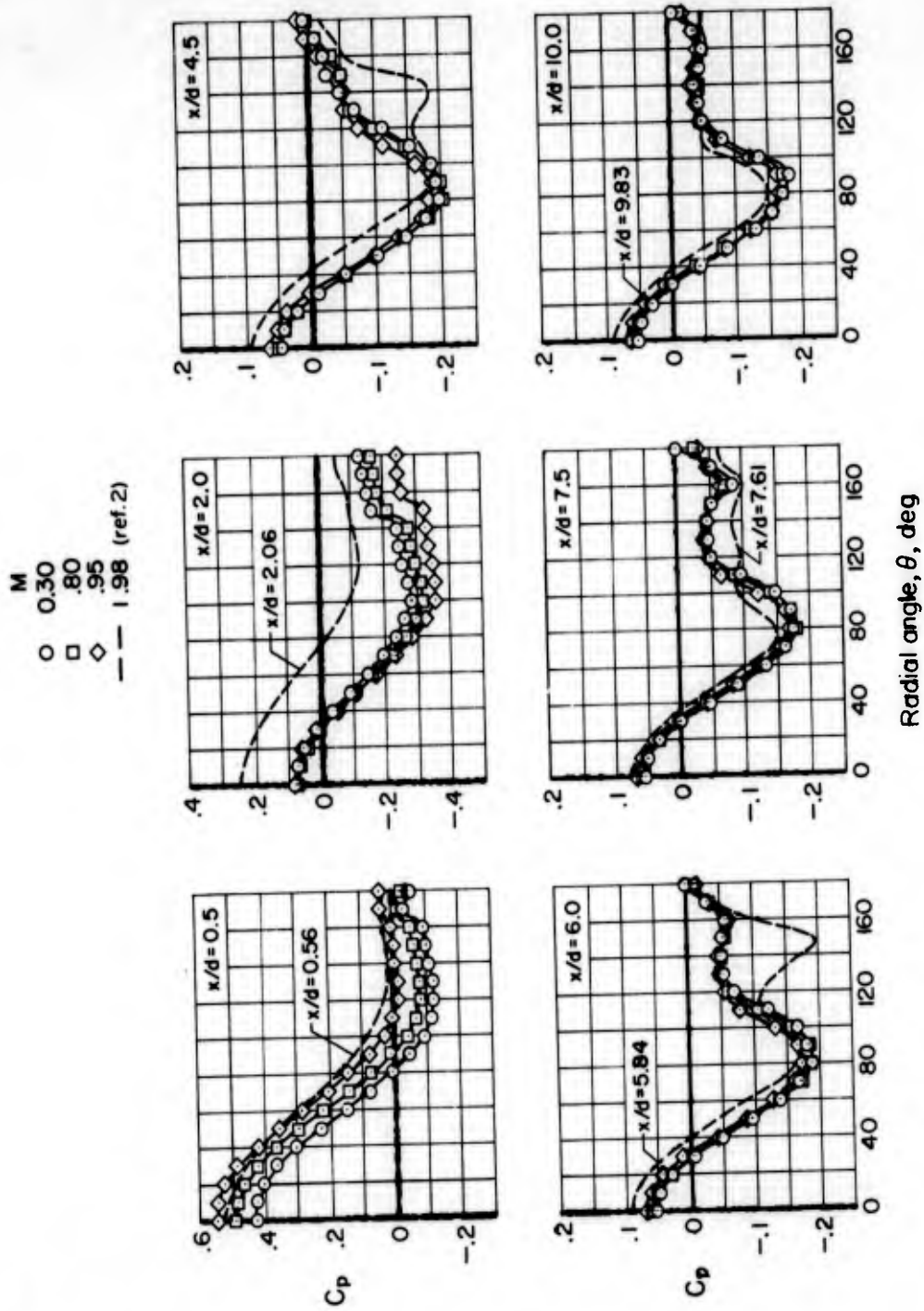
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Figure 2.- The ogive-cylinder body of revolution in the wind tunnel.



(a) $\alpha = 10^\circ$; $R = 0.44 \times 10^6$

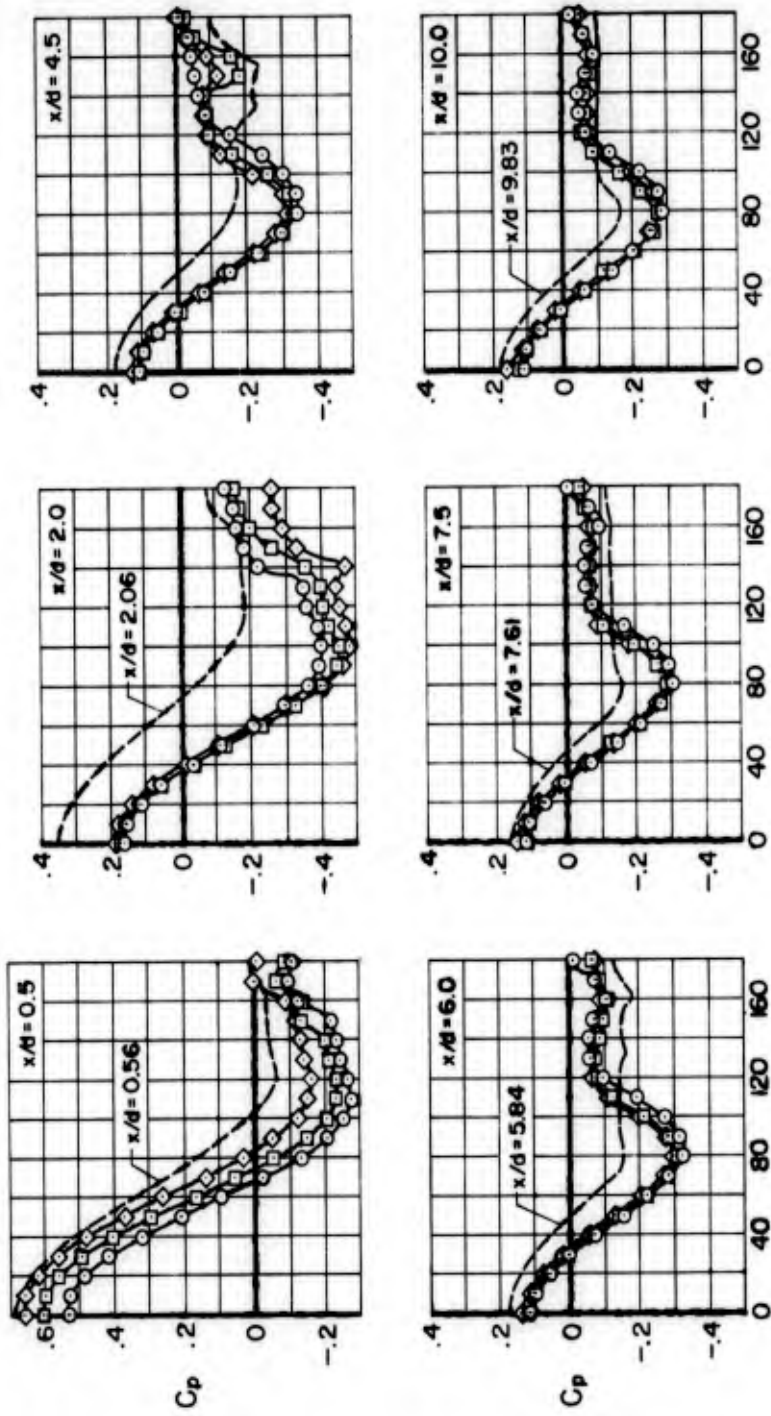
Figure 3.- Circumferential pressure distributions determined experimentally for several Mach numbers.



(b) $\alpha = 15^\circ$; $R = 0.44 \times 10^6$

Figure 3.- Continued.

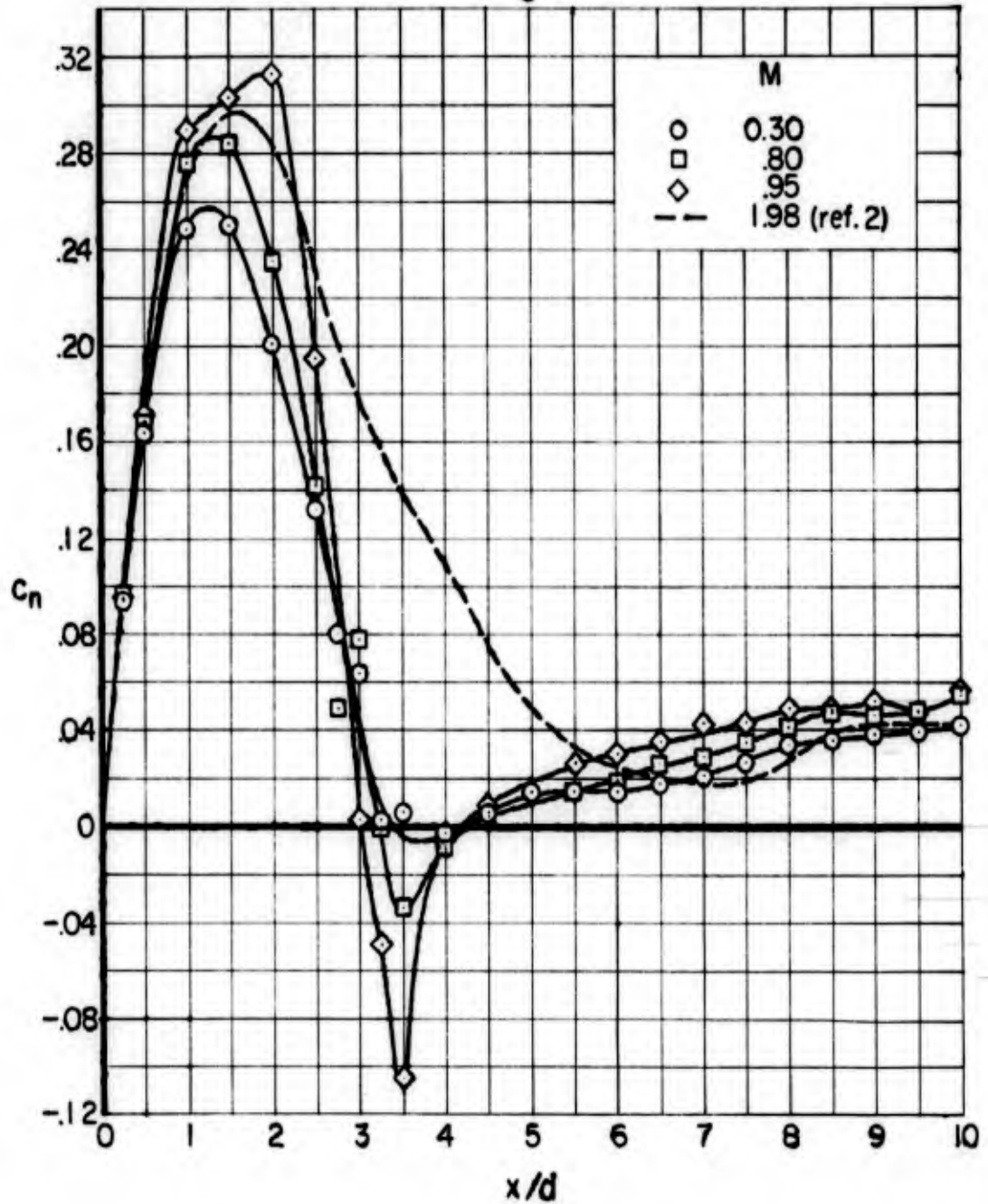
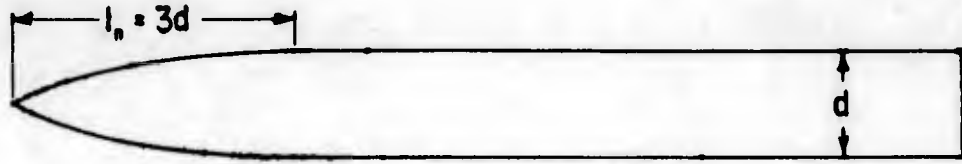
M
 O 0.30
 □ .80
 ◇ .95
 --- 1.98 (ref. 2)



Radial angle, θ , deg

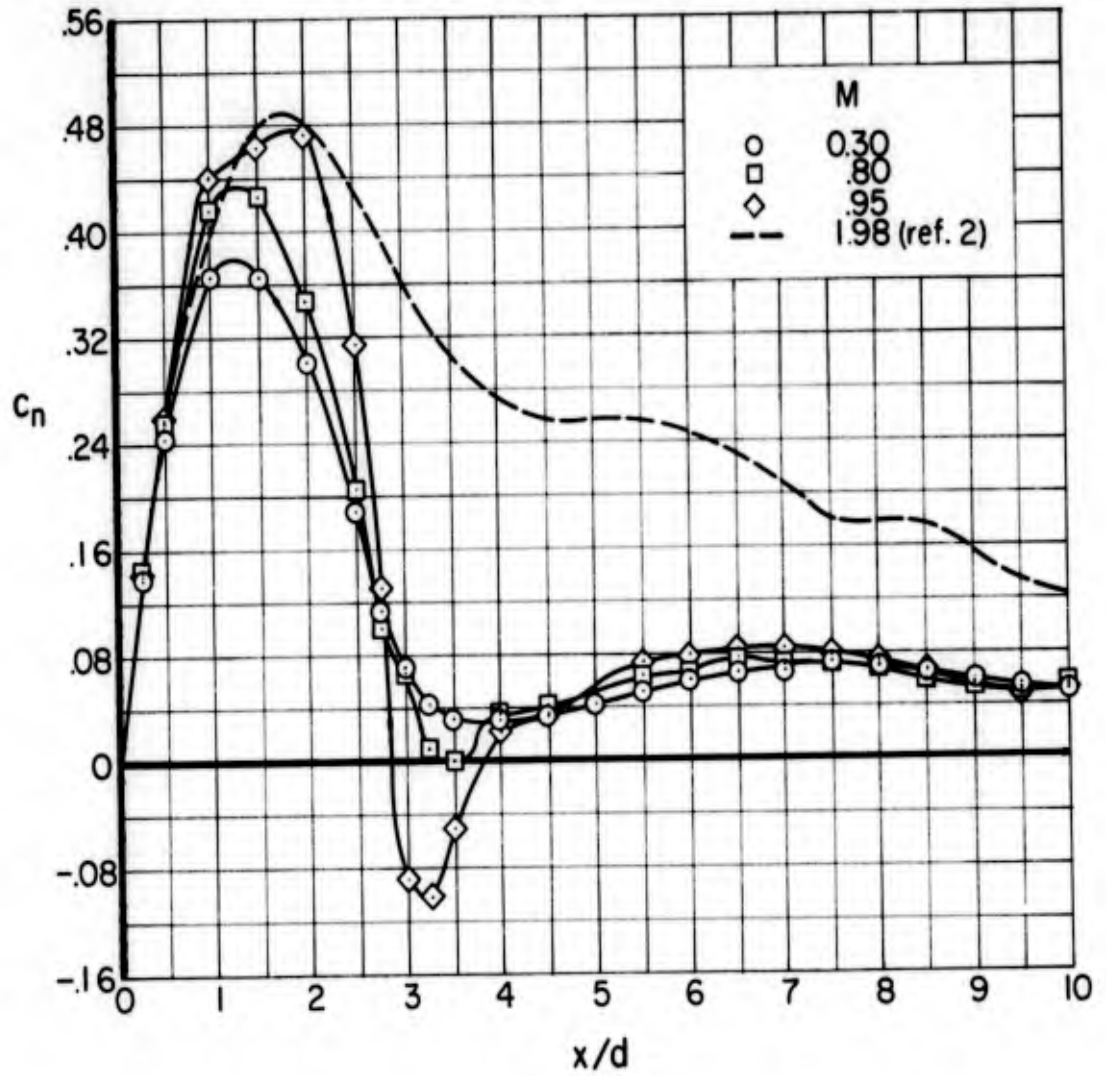
(c) $\alpha = 20^\circ$; $R = 0.44 \times 10^6$

Figure 3.- Concluded.



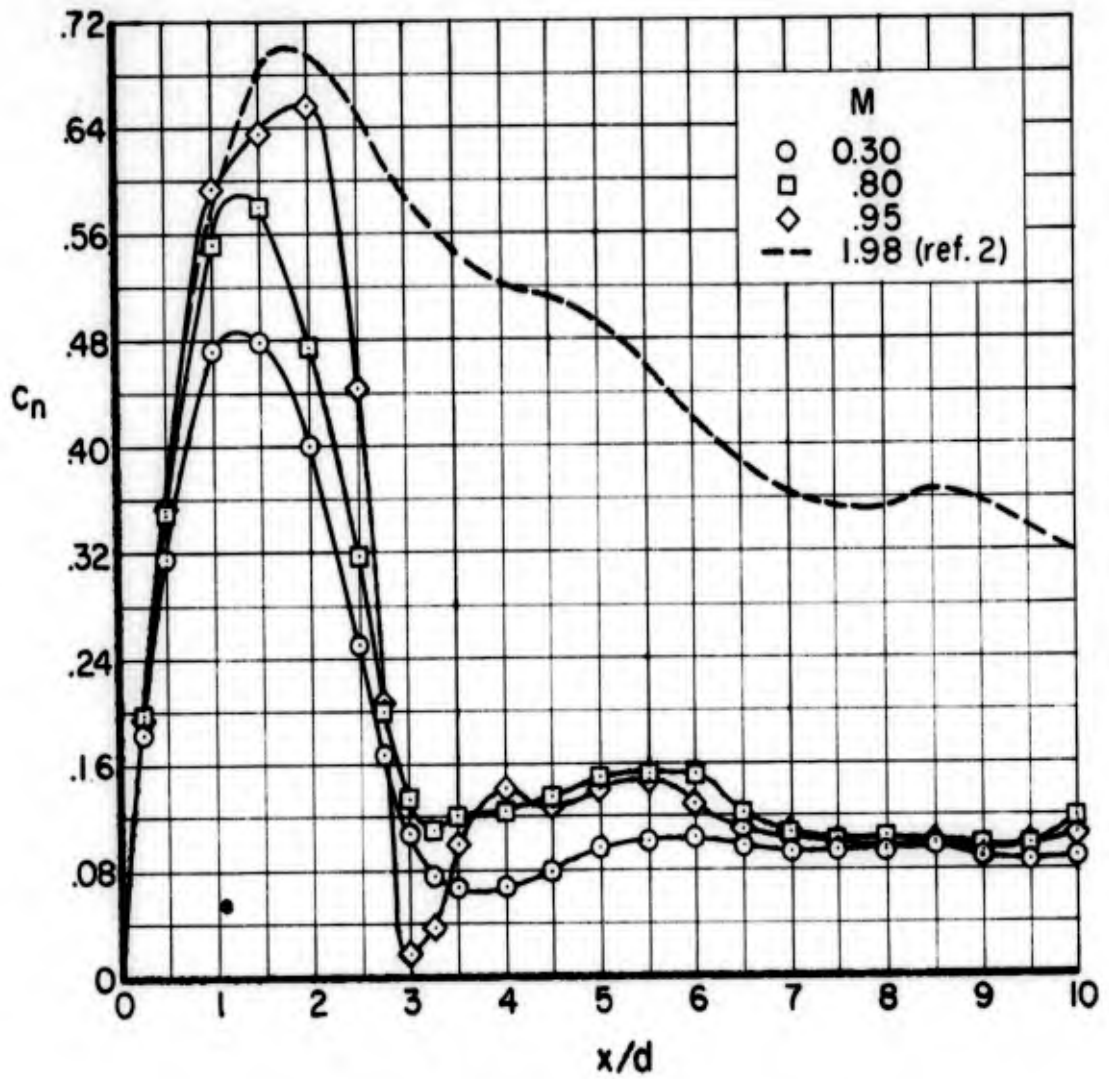
(a) $\alpha = 10^\circ$

Figure 4.- Normal-force distributions determined experimentally for several Mach numbers; $R = 0.44 \times 10^6$.



(b) $\alpha = 15^\circ$

Figure 4.- Continued.



(c) $\alpha = 20^\circ$

Figure 4.- Concluded.

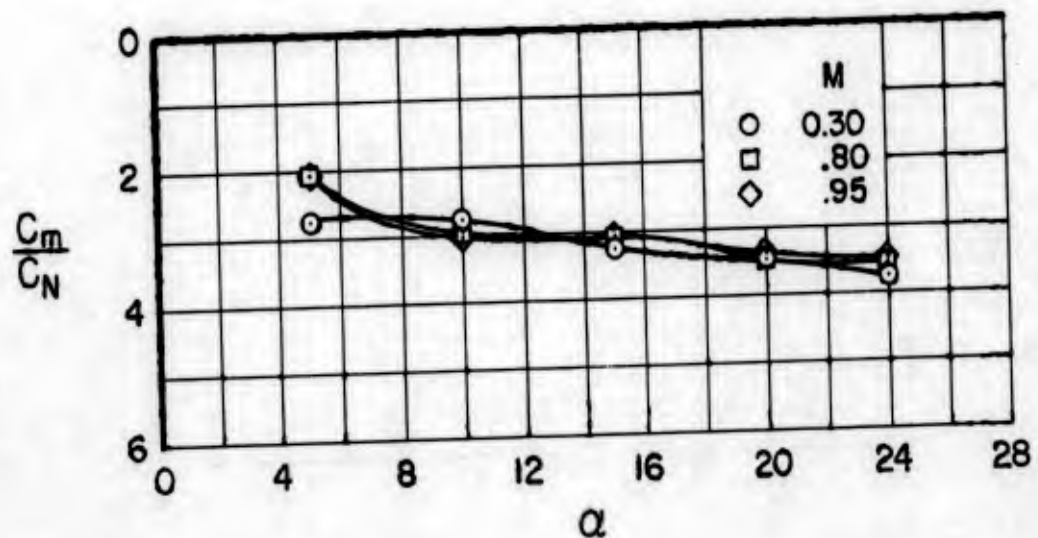
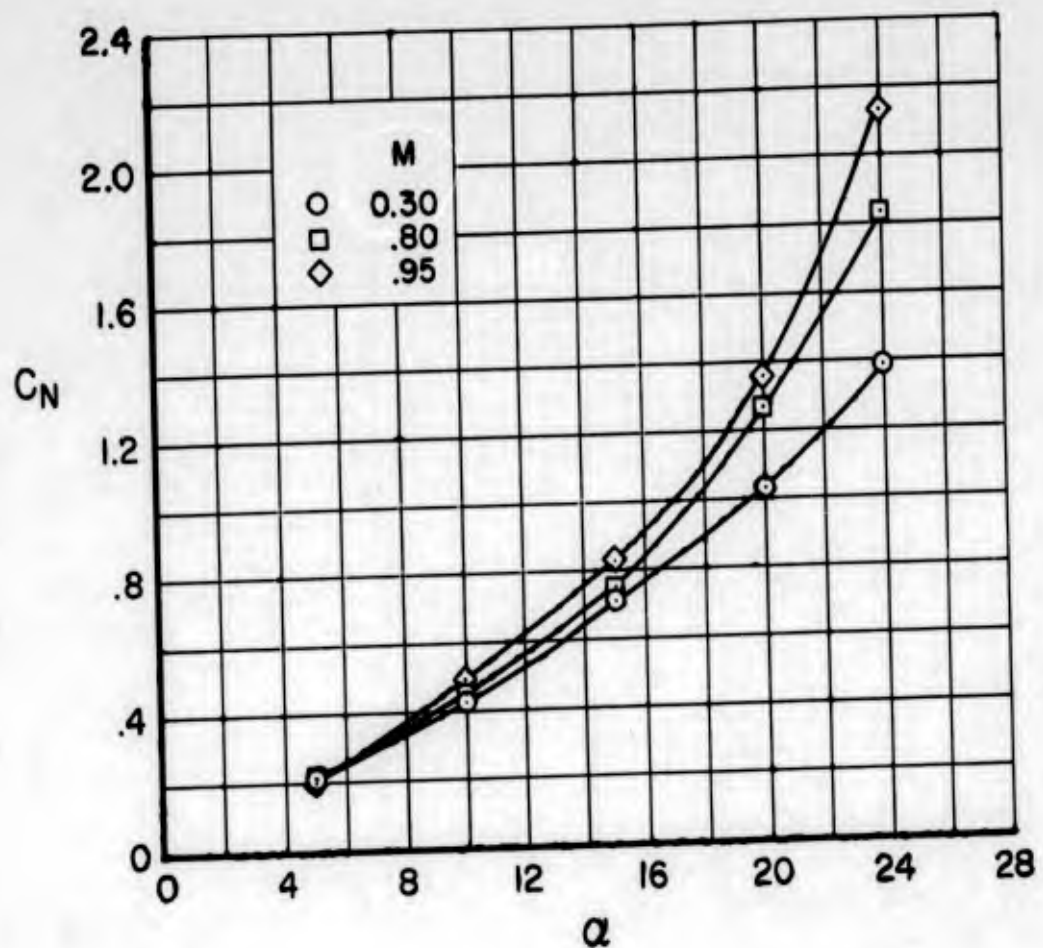
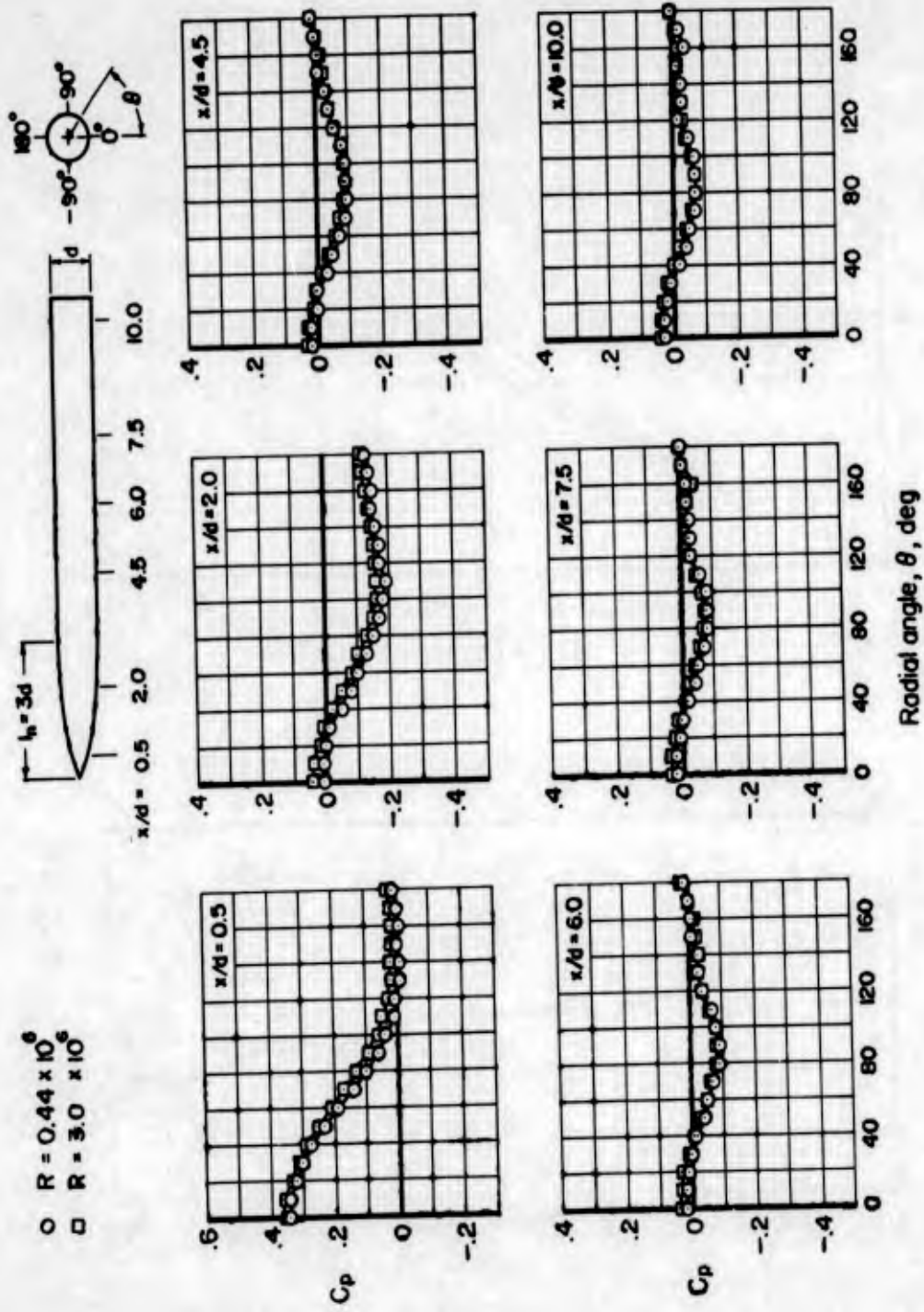
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Figure 5.- Normal-force coefficients and the ratio of pitching-moment to normal-force coefficients versus angle of attack for several Mach numbers; $R = 0.44 \times 10^8$.



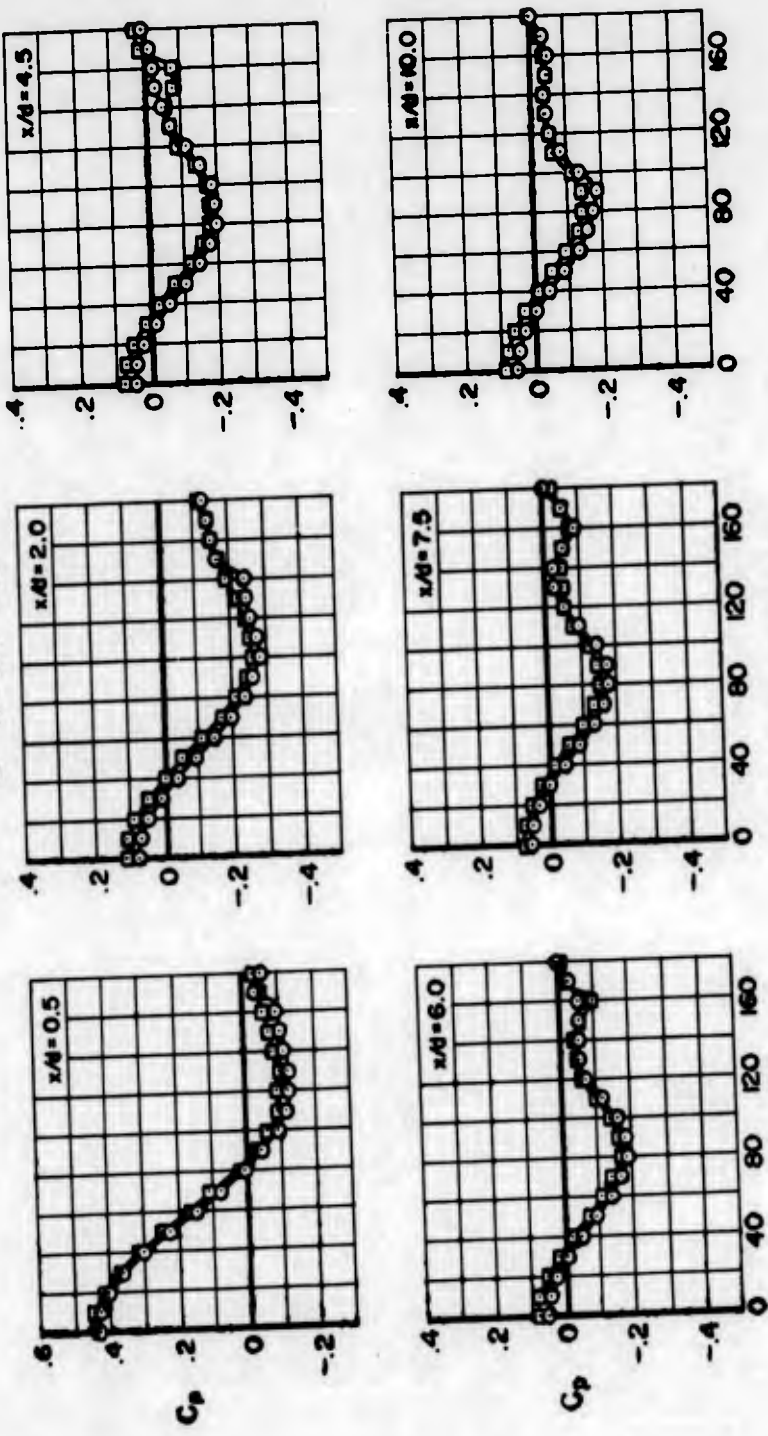
○ $R = 0.44 \times 10^6$
 □ $R = 3.0 \times 10^6$

(a) $\alpha = 10^\circ$

Figure 6.- The effect of Reynolds number on the circumferential pressure distribution; $M = 0.30$.

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0

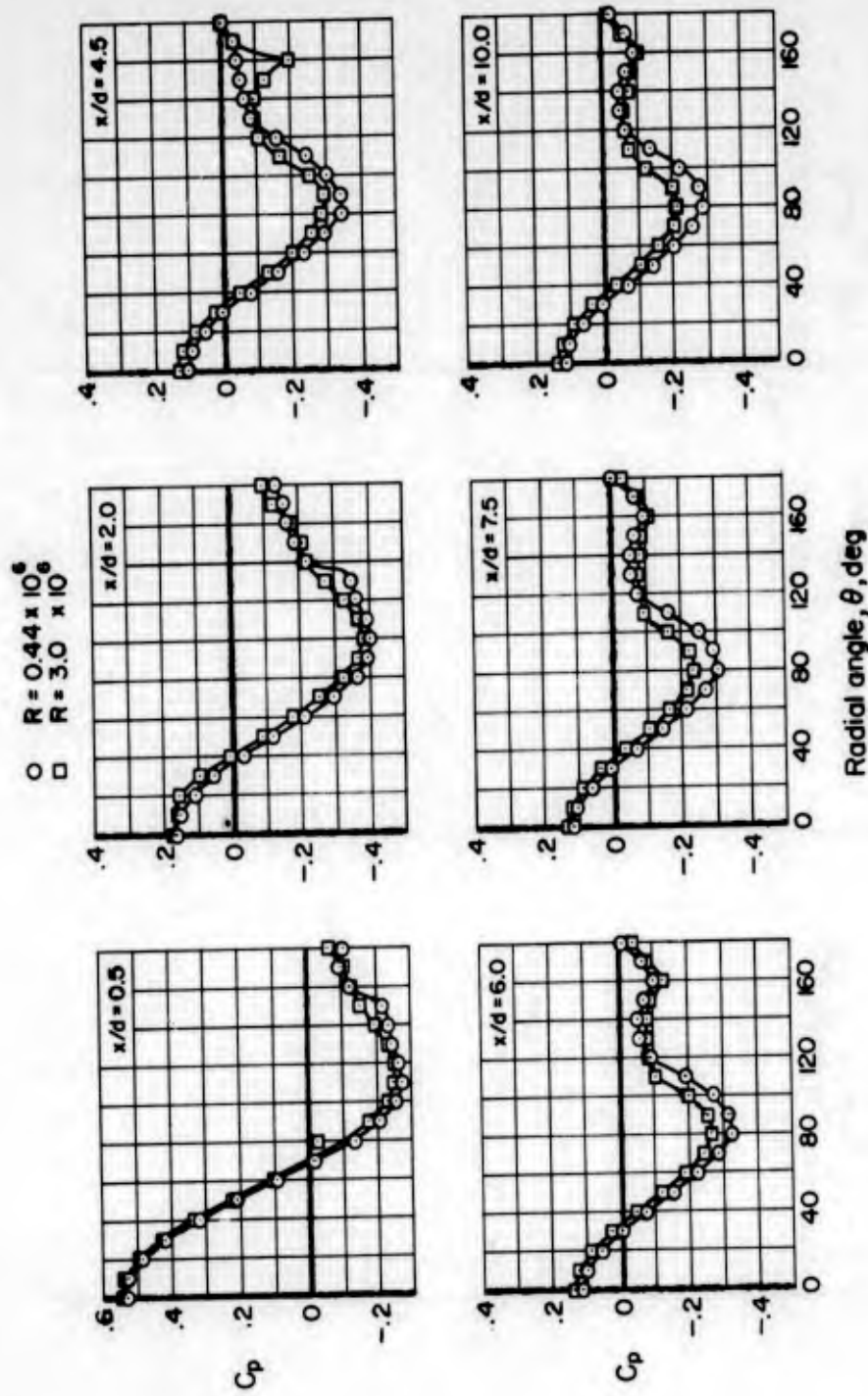
○ $R = 0.44 \times 10^6$
□ $R = 3.0 \times 10^6$



Radial angle, θ , deg

(b) $\alpha = 15^\circ$

Figure 6.- Continued.



(c) $\alpha = 20^\circ$

Figure 6.- Concluded.

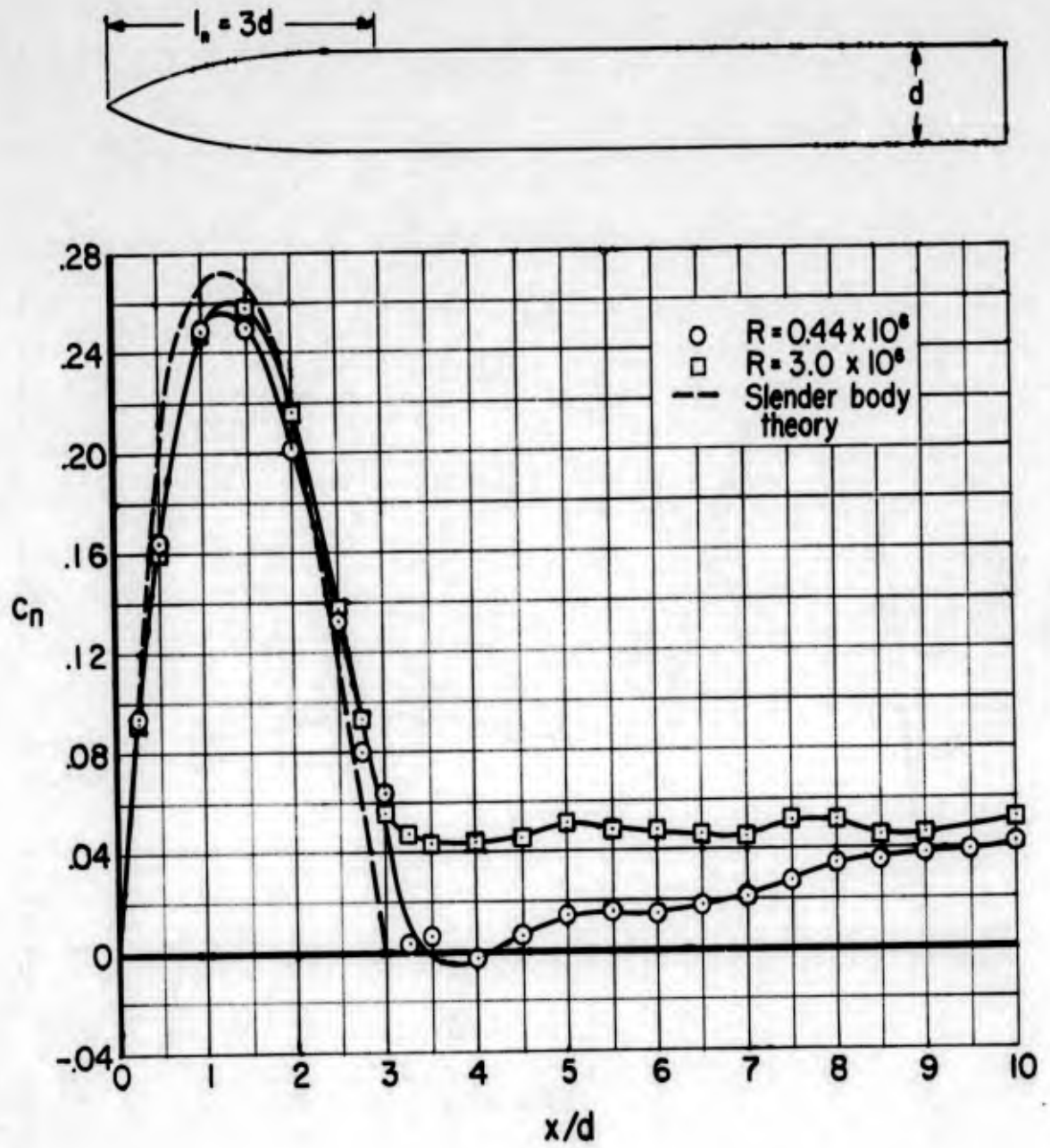
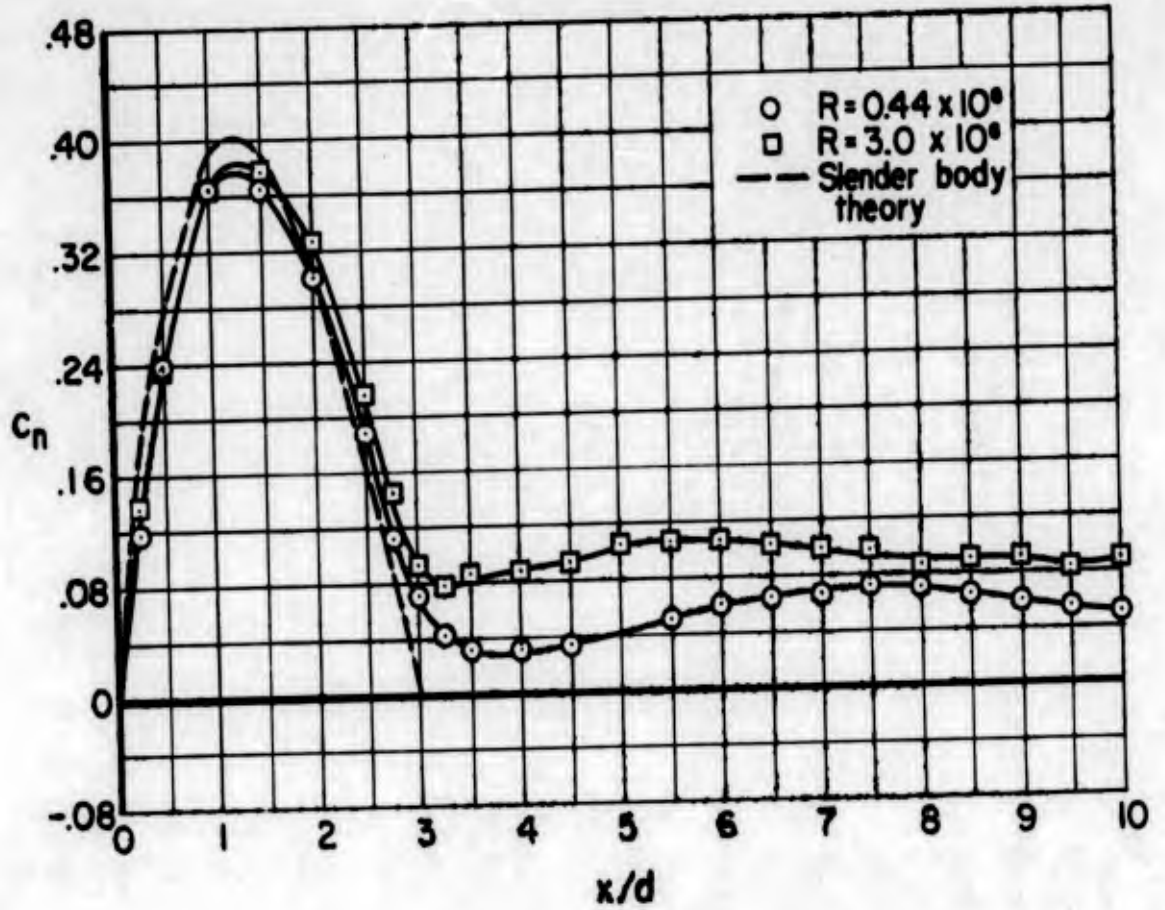
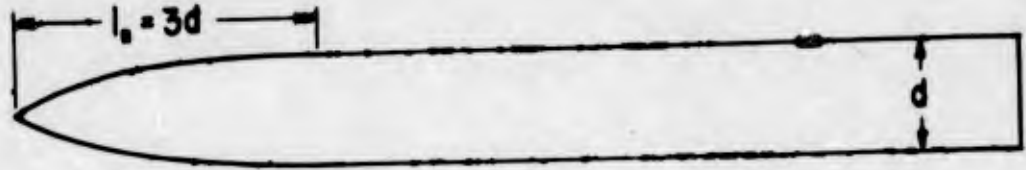
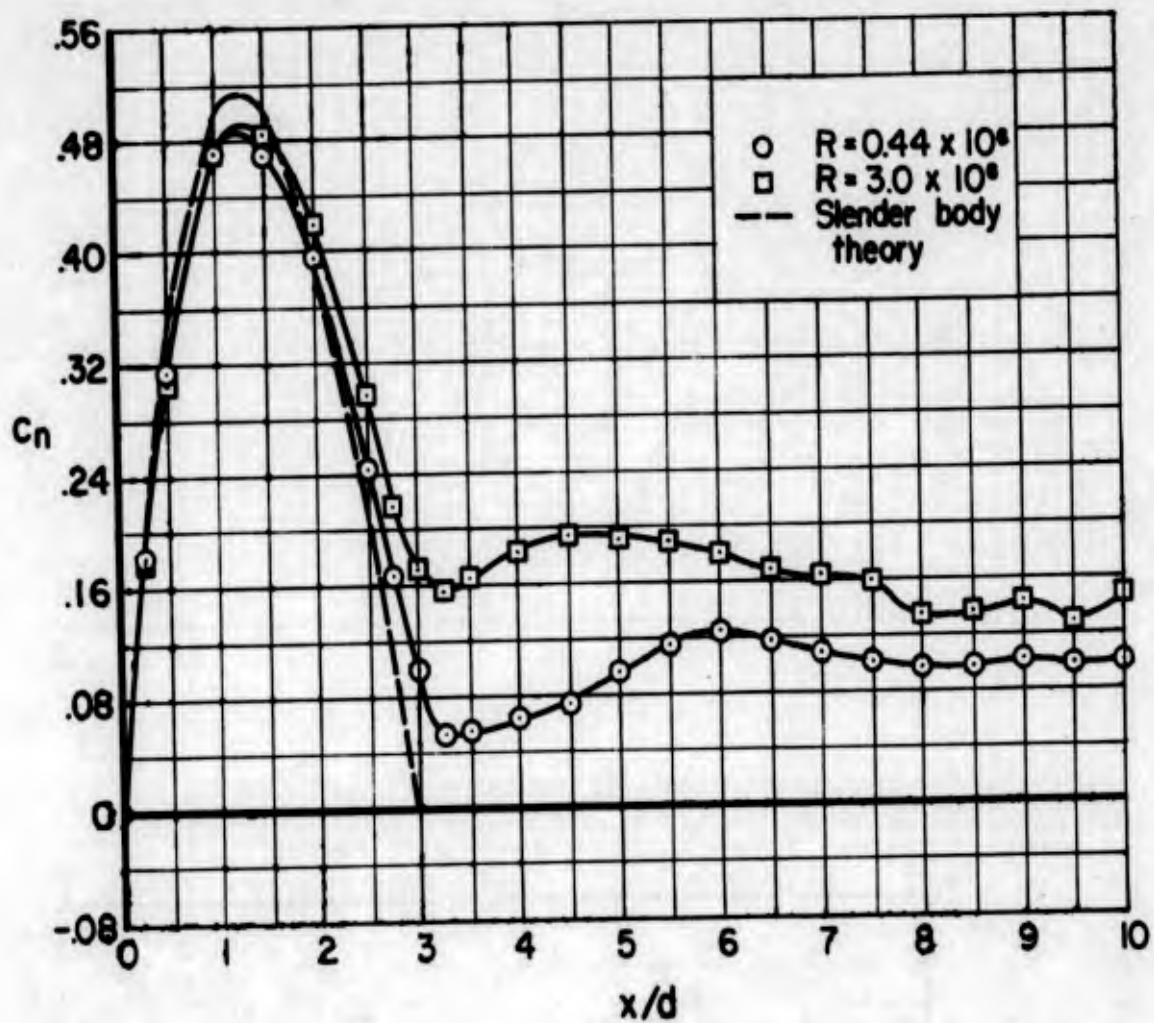
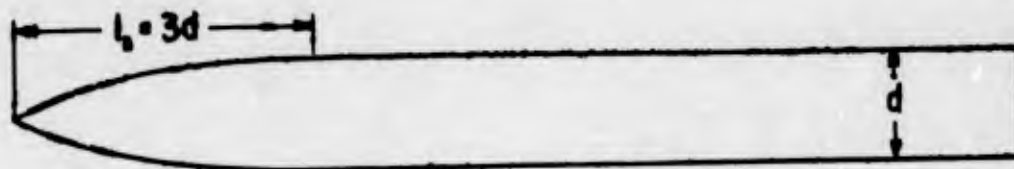
(a) $\alpha = 10^\circ$

Figure 7.- The experimental normal-force distribution at two Reynolds numbers compared with that calculated by slender-body theory; $M = 0.30$.



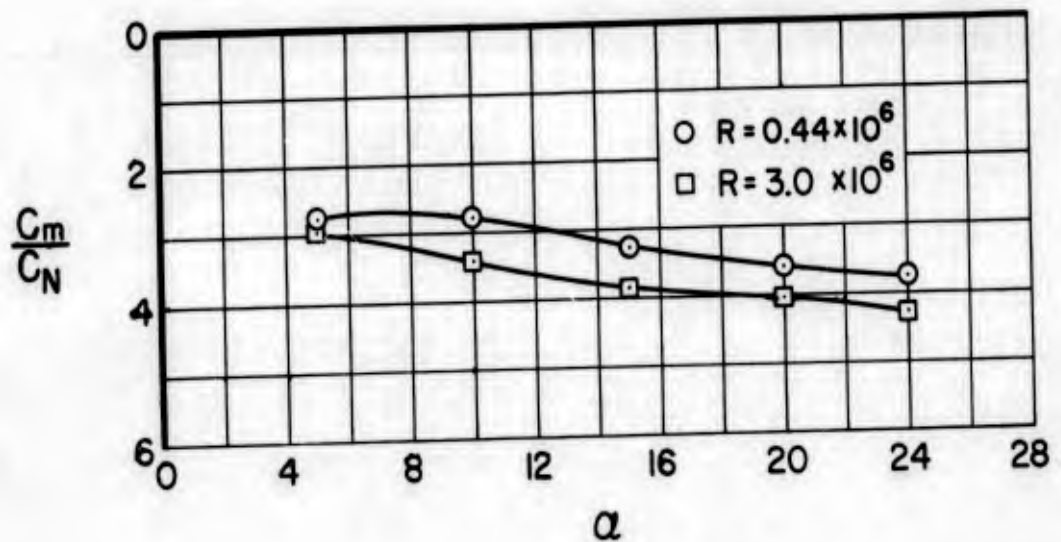
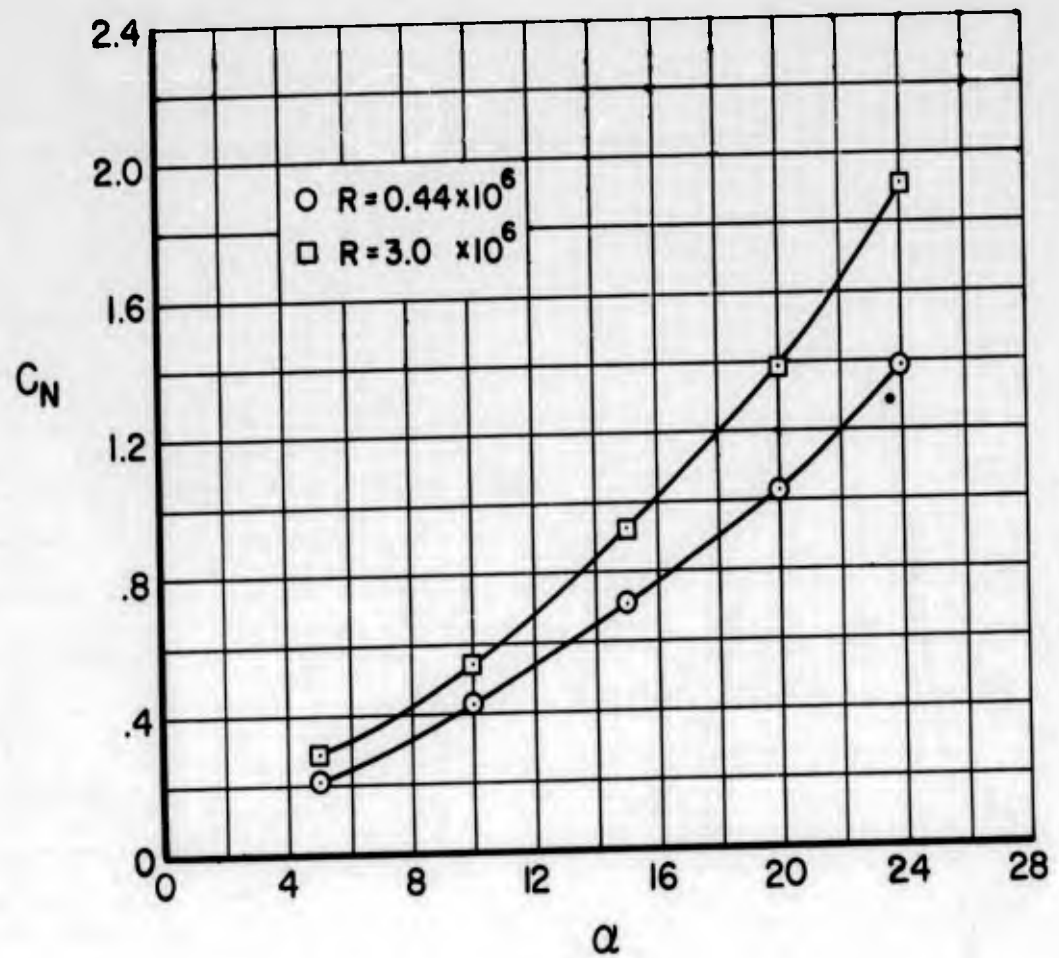
(b) $\alpha = 15^\circ$

Figure 7.- Continued.



(c) $\alpha = 20^\circ$

Figure 7.- Concluded.



—Figure 8.— Normal force and ratio of pitching-moment to normal-force coefficients versus angle of attack at two Reynolds numbers; $M = 0.30$.

○ Most forward position at which vortex was identified by wake survey

▨ Region in which vortex originates (separation occurs) as estimated by the method of ref. 3

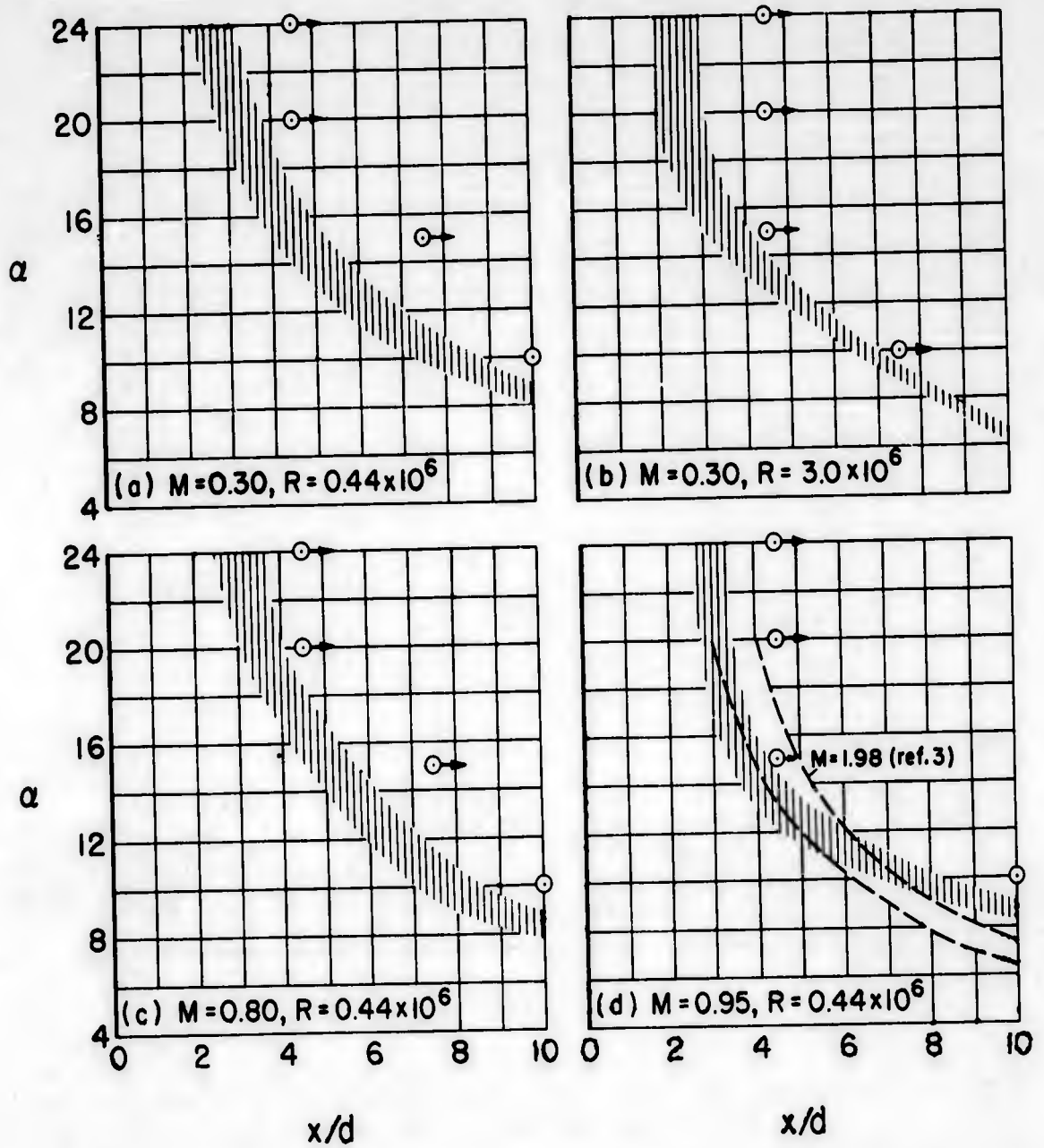


Figure 9.- The approximate positions of vortex separation and the most forward axial positions at which separated vortices were detected from pressure surveys.



	M	Rx10 ⁻⁶
○	0.30	0.44
□	0.30	3.00
◇	0.80	0.44
◇	0.95	0.44
+	1.98	0.44 (ref.3)

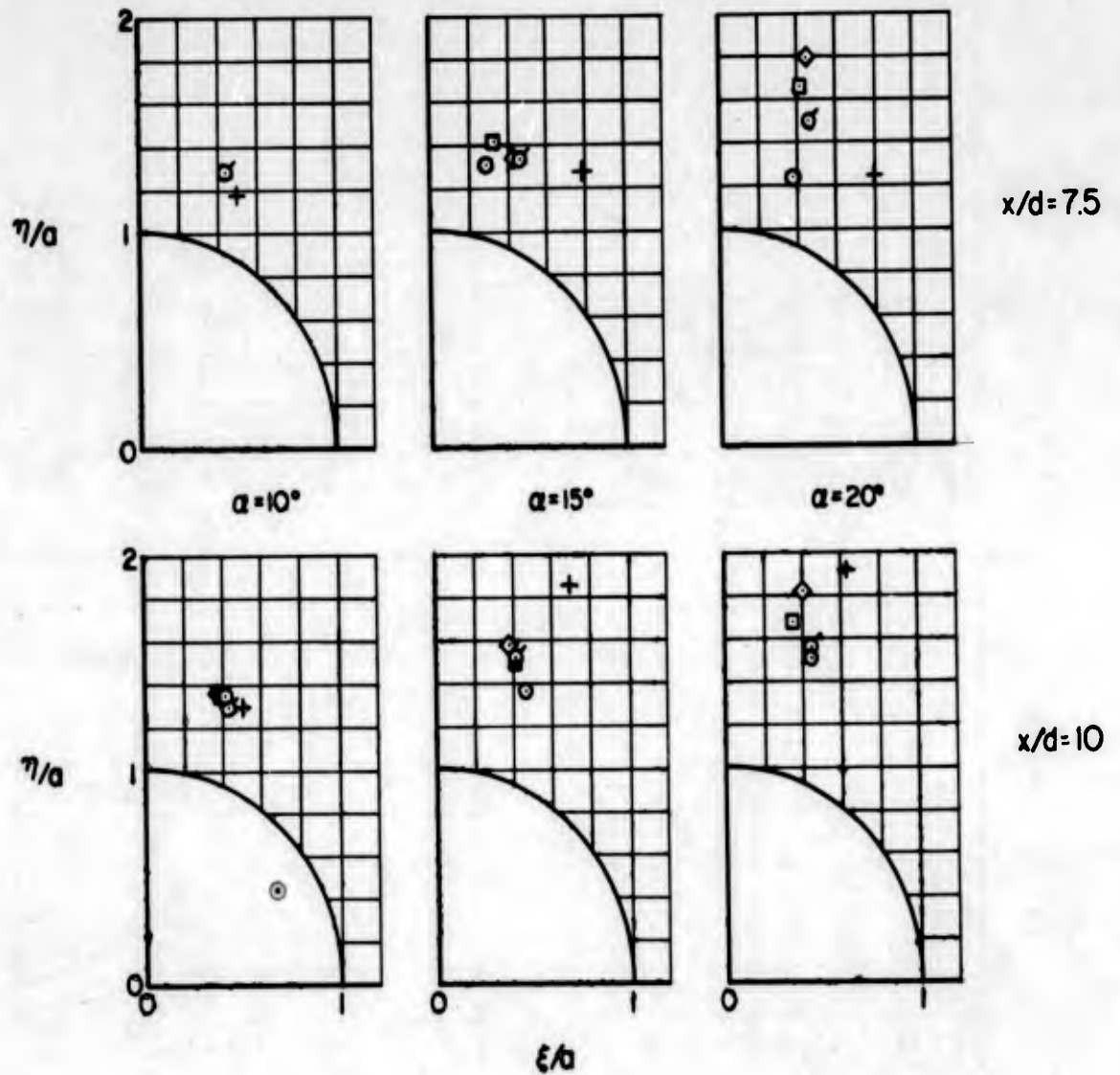


Figure 10.- Vortex core positions (average of right and left vortices) at two axial stations for several angles of attack.

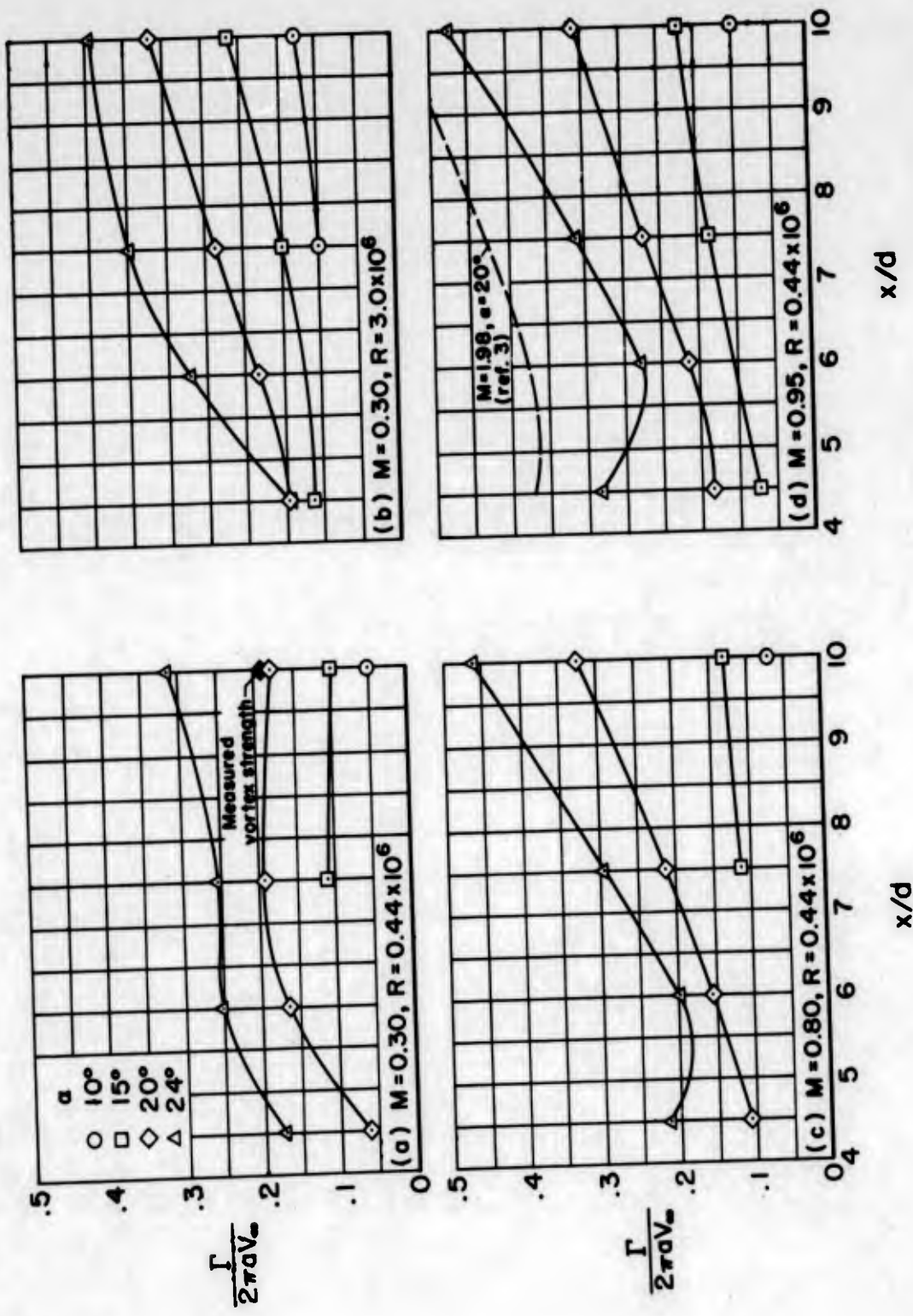
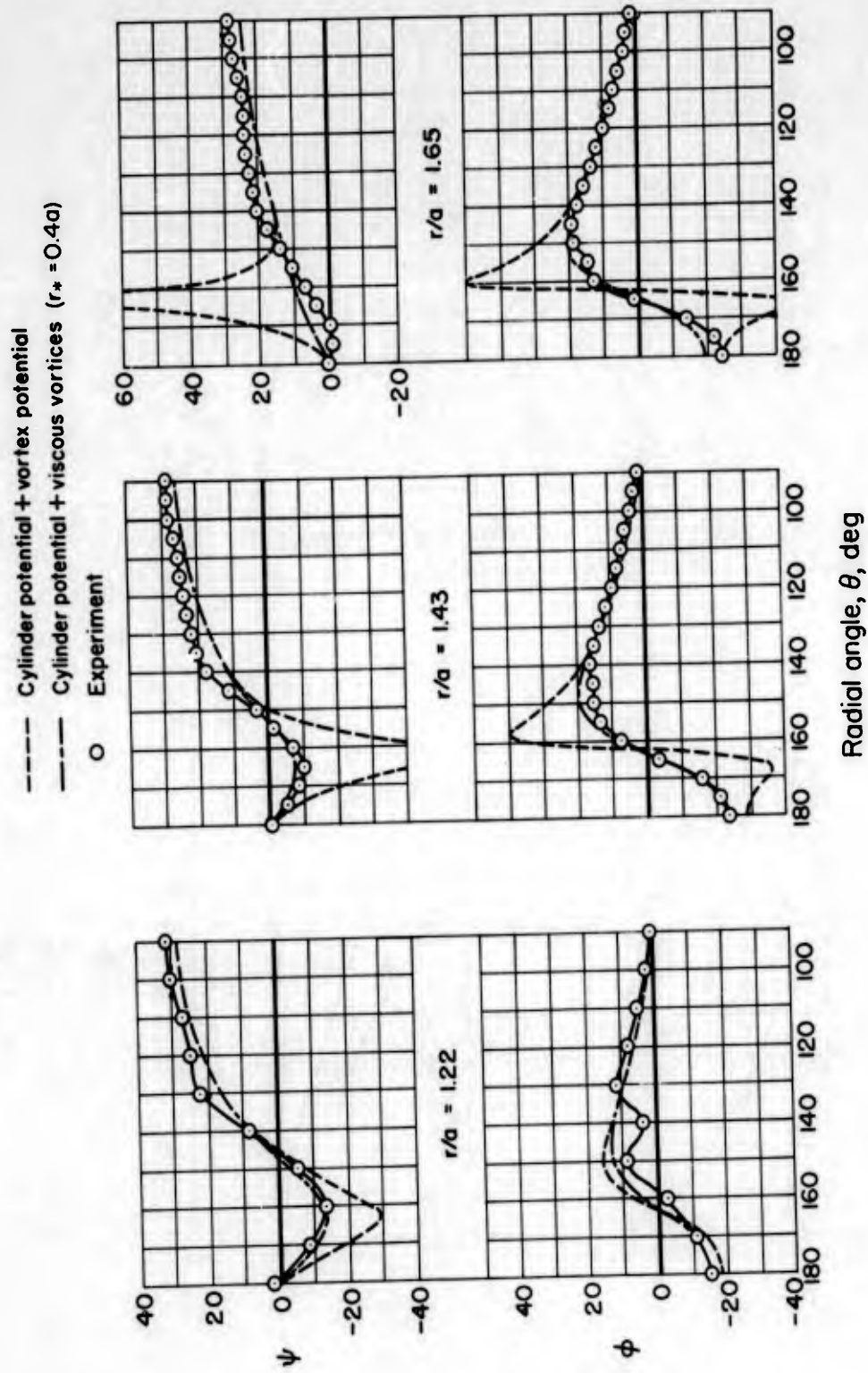
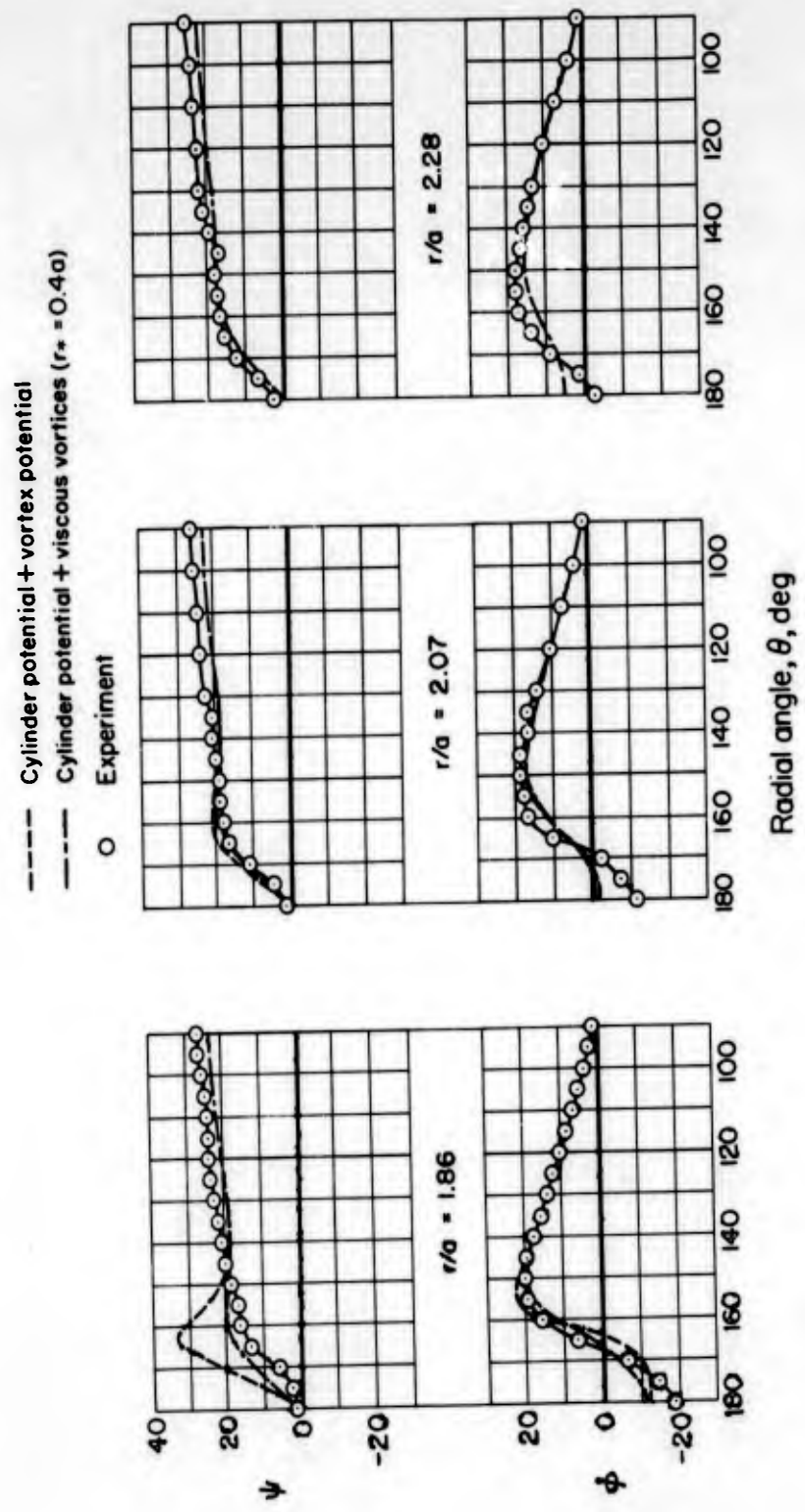


Figure 11.- Vortex circulation strengths computed from experimental values of normal-force distributions and vortex-center positions.



(a) $r/a = 1.22, 1.43, \text{ and } 1.65$

Figure 12.- Measured flow angles compared with computed flow angles for cylinder potential plus either vortex potential or viscous potential flows at several radial distances from the body surface; $x/d = 10, M = 0.30, R = 0.44 \times 10^6, \alpha = 20^\circ$.



(b) $r/a = 1.86, 2.07, \text{ and } 2.28$

Figure 12.- Concluded.

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50, Stability and control;
51, Stresses and loads.)

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