NACA RM L57J15

C2

NACA

RESEARCH MEMORANDUM

LOW-SPEED BOUNDARY-LAYER-CONTROL INVESTIGATION ON A THIN RECTANGULAR SEMISPAN WING WITH LEADING-EDGE AND TRAILING-EDGE FLAPS

By Delwin R. Croom and Thomas R. Turner

Langley Aeronautical Laboratory
Langley Field, Va.

LIBRARY COPY

JAN 21 1958

LANGLEY AERONAUTICAL LABORATORY
LIBRARY, NACA
LANGLEY FIELD, VIRGINIA

NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

WASHINGTON

January 20, 1958

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

LOW-SPEED BOUNDARY-LAYER-CONTROL INVESTIGATION ON A THIN RECTANGULAR SEMISPAN WING WITH LEADING-EDGE AND TRAILING-EDGE FLAPS

By Delwin R. Croom and Thomas R. Turner

SUMMARY

A boundary-layer-control investigation was made in the Langley 300-MPH 7- by 10-foot tunnel to determine the longitudinal aerodynamic characteristics and the chordwise load distribution on a thin, untapered, unswept, semispan wing having an aspect ratio of 3.33 and NACA 65A004 airfoil sections. The wing was equipped with leading- and trailing-edge flaps. Boundary-layer control was obtained by blowing high-pressure air over the trailing-edge flaps.

Results of the investigation, without discussion, are presented in the form of longitudinal aerodynamic characteristics and tabulated pressure coefficients, section normal-force coefficients, and section pitching-moment coefficients.

INTRODUCTION

Considerable interest is being shown in the use of blowing high-energy air over wings and flaps as a means of controlling the boundary layer. Numerous investigations have been made to explore and verify the feasibility of increasing the lift by controlling this separation phenomenon; however, this work has been confined to wings having thickness ratios of 0.06 or greater.

The present investigation was made in the Langley 300-MPH 7- by 10-foot tunnel to obtain the static longitudinal aerodynamic characteristics and the chordwise pressure distribution of a thin low-aspect-ratio wing with leading- and trailing-edge flaps with and without blowing over the trailing-edge flaps.

The results of the investigation are presented without discussion. Pressure coefficients, section normal-force coefficients, and section pitching-moment coefficients are presented in tabular form.

SYMBOLS

c chord, ft

c_w plain-wing chord, ft

c_f trailing-edge-flap chord, ft

c_N leading-edge-flap chord, ft

S semispan wing area, sq ft

C_L lift coefficient, $\frac{\text{Lift}}{q_\infty S}$

C_D drag coefficient, $\frac{\text{Drag}}{q_\infty S}$

ΔC_D jet-boundary correction applied to drag coefficient

C_m pitching-moment coefficient of wing referred to wing quarter-chord, $\frac{\text{Pitching moment}}{q_\infty S c_w}$

C_p pressure coefficient, $\frac{p_{t,\infty} - p}{q_\infty}$ (Subscripts u and l denote upper and lower surface)

C_μ momentum coefficient, $\frac{w_j V_j}{g q_\infty S}$

l_f distance from wing quarter-chord to hinge line of trailing-edge flap measured parallel to trailing-edge-flap chord, ft

l_N distance from wing quarter-chord to hinge line of leading-edge flap measured parallel to leading-edge-flap chord, ft

x longitudinal distance, ft

x_f distance from hinge line of trailing-edge flap to center of load on trailing-edge flap, ft

x_N distance from hinge line of leading-edge flap to center of load on leading-edge flap, ft

$p_{t,\infty}$ free-stream total pressure, lb/sq ft

p local static pressure, lb/sq ft

q_∞ free-stream dynamic pressure, $\frac{\rho V_\infty^2}{2}$, lb/sq ft

ρ mass density of air, slugs/cu ft

V_∞ free-stream velocity, ft/sec

δ_f trailing-edge-flap deflection (positive direction, trailing edge down), deg

δ_N leading-edge-flap deflection (positive direction, nose of flap down), deg

α angle of attack of chord plane set in tunnel, deg

$\Delta\alpha$ jet-boundary correction applied to angle of attack, deg

α_c corrected angle of attack, deg

$c_{N,f}$ section normal-force coefficient of trailing-edge flap based on trailing-edge-flap chord

$c_{N,N}$ section normal-force coefficient of leading-edge flap based on leading-edge-flap chord

$c_{N,w'}$ section normal-force coefficient of that portion of wing between leading-edge and trailing-edge flaps based on plain-wing chord

$c_{N,w}$ wing section normal-force coefficient based on plain-wing chord (chord force of leading-edge flap and trailing-edge flap neglected), $c_{N,w'} + c_{N,f}\left(\frac{c_f}{c_w}\right)\cos\delta_f + c_{N,N}\left(\frac{c_N}{c_w}\right)\cos\delta_N$

$c_{m,f}$ section pitching-moment coefficient of trailing-edge flap based on trailing-edge-flap chord (moments taken about trailing-edge-flap hinge line)

$c_{m,N}$ section pitching-moment coefficient of leading-edge flap based on leading-edge-flap chord (moments taken about leading-edge-flap hinge line)

$c_{m,w'}$ section pitching-moment coefficient of that portion of wing between leading-edge and trailing-edge flaps based on plain-wing chord (moments taken about wing quarter-chord)

$c_{m,w}$ wing section pitching-moment coefficient based on plain-wing chord (moments taken about wing quarter-chord; chord force of leading- and trailing-edge flaps neglected),

$$c_{m,w'} - \frac{c_{N,f}\left(l_f + x_f\right)c_f}{c_w^2} + \frac{c_{N,N}\left(l_N + x_N\right)c_N}{c_w^2}$$

w_j weight rate of flow of jet, lb/sec

V_j jet velocity (isentropic expansion is assumed),

$$\sqrt{\frac{2\gamma}{\gamma - 1} RTg\left[1 - \left(\frac{p_\infty}{p_{t,p}}\right)^{\left(\frac{\gamma-1}{\gamma}\right)}\right]}, \text{ ft/sec}$$

γ ratio of specific heats of air, 1.4

p_∞ free-stream static pressure, lb/sq ft

$p_{t,p}$ total pressure in plenum chamber, lb/sq ft

R universal gas constant

T plenum-chamber temperature, $^{\circ}R$

g acceleration due to gravity, ft/sec^2

MODEL AND APPARATUS

The model was tested in the Langley 300-MPH 7- by 10-foot tunnel by means of the semispan technique with the ceiling of the tunnel as the reflection plane. The geometric characteristics of the semispan wing used in this investigation are given in figure 1. The wing had 0°

of sweep, a taper ratio of 1, an aspect ratio of 3.33, and NACA 65A004 airfoil sections parallel to the free airstream direction. The wing was equipped with leading-edge and trailing-edge flaps. The leading-edge-flap chord was 15 percent of the wing chord and the leading-edge flap pivoted about the lower surface along the 15-percent-chord line. For the deflected condition the break in the upper surface was faired to an arc of a circle. The trailing-edge flaps had chords of 20, 25, 33, and 40 percent of the wing chord. (See fig. 2.) It should be noted that the 40-percent-chord flap was obtained by the addition of a $\frac{1}{8}$- inch plate attached to the lower surface of the 33-percent-chord flap as shown in figure 2.

The leading-edge flap, the trailing-edge flaps, and the wing were constructed with flush-surface pressure orifices located at the 0.60-semispan position. The trailing-edge flaps were sealed on the lower surface and had a full-span 0.020-inch gap on the upper surface from which high-pressure air was blown over the flaps. The nose of this flap was hollow and high-pressure air was brought through it and ejected into the wing plenum chamber through 71 holes of $\frac{1}{16}$- inch diameter located at spanwise intervals of $\frac{1}{2}$ inch.

The compressed air was brought onto the balance frame through a $1\frac{1}{2}$- inch-diameter steel pipe (fig. 3). One end of the long pipe was fastened rigidly to the tunnel foundation and the other end was attached rigidly to the balance frame. The pipe was long enough to be considered as a very weak spring connecting the balance frame to the ground. The tare of this setup was determined experimentally to be within the accuracy of reading of the scales.

The weight rate of flow of air was determined by means of a calibrated sharp-edge orifice in the pipe line before the air came onto the balance frame, and the pressures and temperatures for determining the jet-exit velocities were measured in the plenum chamber in the wing.

TEST CONDITIONS

The tests were made in the Langley 300-MPH 7- by 10-foot tunnel at the dynamic pressures and ranges of leading- and trailing-edge-flap deflection and momentum coefficient given in the following table:

Trailing-edge-flap chord, c_f/c_w	Dynamic pressure, q, lb/sq ft	Trailing-edge-flap deflection range, δ_f, deg	Leading-edge-flap deflection range, δ_N, deg	Momentum-coefficient range, C_μ
0.20	12.5	60	0 to 25	0 to 0.25
.25	25	0 to 60	0 to 45	0
.33	12.5	0 to 75	0 to 45	0 to 0.31
.40	12.5	60	15 to 45	0 to 0.26

The angle-of-attack range for the investigation extended from about -32° to about 28°. Pressure distributions were obtained for the 20- and 25-percent-chord-flap configurations.

CORRECTIONS

The following jet-boundary corrections applied to the data of this paper are based on the 3-foot-chord model and were obtained by the method outlined in reference 1:

$$\Delta\alpha = 1.435C_L$$

$$\Delta C_D = 0.025C_L{}^2$$

Therefore, the jet-boundary corrections as applied to the 3.33-foot-chord model are 11 percent too small.

The blockage correction as applied to the dynamic pressure was obtained by the method outlined in reference 2.

RESULTS

The results of this investigation are presented without discussion. The order of presentation in the figures and tables is as follows:

Longitudinal aerodynamic data for the 0.20-, 0.25-, 0.33-, and 0.40-chord trailing-edge-flap configurations . . . . . Figures 4 to 29
(Data of figures 14 to 29 for the various momentum-coefficients were obtained from cross plots in order to have directly comparable figures.)
Tabulation and plots of pressure distribution over the 0.25-chord trailing-edge-flap configuration . . . . . Figures 30 to 52
Pressure coefficients for the 0.20-chord trailing-edge-flap configuration . . . . . Tables I to V
Section data for the 0.20-chord trailing-edge-flap configuration . . . . . Tables VI to X
Section data for the 0.25-chord trailing-edge-flap configuration . . . . . Tables XI to XIV

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., September 27, 1957.

REFERENCES

1. Polhamus, Edward C: Jet-Boundary-Induced-Upwash Velocities for Swept Reflection-Plane Models Mounted Vertically in 7- by 10-Foot, Closed, Rectangular Wind Tunnels. NACA TN 1752, 1948.

2. Herriot, John G.: Blockage Corrections for Three-Dimensional-Flow Closed-Throat Wind Tunnels, With Consideration of the Effect of Compressibility. NACA Rep. 995, 1950. (Supersedes NACA RM A7B28.)

TABLE I - PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE

(a) $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $c_N = 0^\circ$; $q = 12.5$ lb/sq ft, $C_\mu = 0$

Lower surface

x/c	C_p for -					
	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$
Nose						
.0453	1.703	1.859	.554	.130	.012	.019
.0927	1.692	1.641	.696	.243	.108	.056
.1852	1.692	1.429	.815	.384	.247	.198
.2786	1.709	1.320	.821	.446	.325	.259
.3706	1.703	1.250	.845	.497	.392	.339
.5565	1.715	1.173	.863	.548	.462	.420
.7425	1.697	1.135	.857	.616	.536	.469
.9272	1.622	1.103	.851	.627	.548	.512
Wing						
.1944	1.465	1.077	.851	.672	.608	.568
.2500	1.343	1.000	.845	.667	.614	.611
.3057	1.227	.949	.815	.678	.626	.593
.3612	1.128	.878	.792	.650	.645	.611
.4444	1.011	.795	.696	.587	.614	.593
.5278	.907	.647	.595	.525	.542	.555
.6115	.884	.506	.452	.384	.422	.420
.6678	.826	.506	.393	.316	.307	.371
.7094	.808	.519	.310	.220	.253	.235
.7372	.796	.506	.298	.226	.235	.228
.7647	.779	.506	.298	.264	.217	.222
.7925	.738	.487	.280	.249	.205	.222
Flap						
.0582	.760	.526	.333	.237	.229	.210
.1306	.779	.532	.357	.226	.205	.210
.2758	.698	.455	.238	.181	.263	.179
.4177	.665	.378	.230	.192	.235	.222
.5600	.698	.404	.310	.339	.343	.344
.7019	.721	.442	.494	.429	.428	.489
.8527	.808	.814	.744	.684	.723	.704
1.0000	1.680	1.904	1.857	1.672	1.723	1.839

Upper surface

x/c	C_p for -					
	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$
Nose						
.0000	1.825	2.327	.179	2.954	2.861	2.487
.0453	.116	.404	1.583	2.661	2.518	2.271
.0927	.256	.609	1.482	2.672	2.536	2.271
.1852	.419	.776	1.357	2.655	2.524	2.271
.2786	.500	.820	1.321	2.390	2.494	2.284
.3706	.581	.859	1.286	1.926	2.518	2.293
.5565	.680	.929	1.280	1.491	2.536	2.253
.7425	.738	.974	1.250	1.395	2.554	2.210
.9272	.796	1.045	1.298	1.412	2.566	2.203
Wing						
.1944	.860	1.077	1.286	1.373	2.440	2.234
.2500	.936	1.109	1.274	1.350	2.199	2.247
.3057	.988	1.160	1.286	1.344	1.910	2.216
.3612	1.041	1.179	1.321	1.350	1.669	2.185
.4444	1.116	1.288	1.375	1.361	1.536	2.109
.5278	1.362	1.635	1.518	1.644	1.691	2.185
.6115	1.379	1.673	1.571	1.722	1.709	2.055
.6678	1.448	1.731	1.589	1.695	1.727	1.981
.7094	1.448	1.750	1.625	1.712	1.673	1.889
.7372	1.483	1.769	1.661	1.678	1.673	1.926
.7647	1.569	2.000	1.732	1.695	1.673	1.944
.7925	1.690	2.019	1.857	1.864	1.818	2.000
Flap						
-.0420	1.700	2.000	1.880	1.900	1.982	2.055
-.0350	1.710	2.000	1.900	1.924	1.945	2.074
-.0240	1.730	2.096	1.911	1.949	1.800	2.055
-.0130	1.750	2.000	1.911	1.949	1.764	2.000
.0000	1.759	2.019	1.839	1.847	1.818	2.037
.0582	1.741	2.000	1.834	1.830	1.800	1.963
.1306	1.793	2.000	1.893	1.915	1.782	2.037
.2758	1.828	2.038	1.911	1.814	1.818	2.222
.4177	1.793	2.058	1.911	1.915	1.873	1.944
.5600	1.724	1.865	1.929	1.814	1.764	1.963
.7019	1.828	2.038	1.893	1.814	1.891	2.018
.8527	1.793	1.904	1.839	1.847	1.800	1.981
1.0000	.983	.923	.875	.964	.964	.889

TABLE I.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(b) $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 0^\circ$, $q \approx 12.5$ lb/sq ft; $C_\mu = 0.013$

Lower surface

x/c	C_p for -					
	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$
Nose						
.0453	1.860	1.398	.343	.019	.006	.012
.0927	1.879	1.325	.542	.151	.042	.043
.1852	1.879	1.199	.675	.314	.187	.161
.2786	1.891	1.139	.711	.384	.259	.255
.3706	1.866	1.090	.735	.440	.307	.311
.5565	1.745	1.036	.765	.522	.410	.379
.7425	1.588	1.012	.795	.579	.452	.441
.9272	1.406	.964	.789	.591	.476	.484
Wing						
.1944	1.170	.916	.819	.629	.548	.553
.2500	1.042	.873	.795	.648	.560	.571
.3057	.970	.849	.777	.692	.566	.571
.3612	.897	.777	.741	.641	.546	.590
.4444	.824	.687	.675	.597	.542	.565
.5278	.751	.578	.554	.503	.500	.534
.6115	.691	.410	.404	.396	.367	.410
.6678	.654	.398	.307	.283	.271	.286
.7094	.612	.422	.277	.224	.211	.311
.7372	.612	.422	.253	.214	.193	.217
.7647	.594	.440	.265	.201	.175	.199
.7925	.588	.434	.265	.214	.283	.180
Flap						
.0582	.600	.438	.380	.180	.350	.180
.1306	.618	.440	.289	.157	.187	.156
.2758	.521	.392	.217	.151	.235	.161
.4177	.539	.301	.193	.189	.175	.199
.5600	.539	.289	.283	.277	.301	.311
.7019	.606	.392	.440	.516	.410	.422
.8527	.776	.457	.451	.541	.578	.658
1.0000	1.618	1.593	1.649	1.667	1.554	1.758

Upper surface

x/c	C_p for -					
	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$
Nose						
.0000	1.873	1.036	.849	3.277	3.054	2.373
.0453	.182	.620	2.030	2.931	2.741	2.168
.0927	.351	.783	1.783	2.950	2.717	2.155
.1852	.497	.867	1.530	2.987	2.705	2.168
.2786	.582	.926	1.416	2.994	2.717	2.161
.3706	.654	.958	1.404	2.918	2.729	2.161
.5565	.758	1.012	1.373	2.528	2.723	2.161
.7425	.794	1.036	1.337	1.987	2.753	2.143
.9272	.909	1.133	1.416	1.717	2.759	2.174
Wing						
.1944	.964	1.145	1.361	1.616	2.887	2.224
.2500	1.030	1.169	1.361	1.591	2.512	2.235
.3057	1.067	1.217	1.386	1.591	2.181	2.211
.3612	1.121	1.253	1.404	1.616	1.916	2.186
.4444	1.248	1.367	1.494	1.660	1.723	2.155
.5278	1.527	1.709	1.891	2.094	1.854	2.315
.6115	1.900	2.100	1.945	2.200	1.927	2.450
.6678	2.000	2.054	2.054	2.245	1.964	2.500
.7094	1.854	1.991	2.091	2.340	2.164	2.074
.7372	1.964	1.964	2.182	2.415	2.327	2.000
.7647	2.273	2.218	2.418	2.679	2.564	2.074
.7925	2.745	2.691	2.945	3.321	3.273	2.074
Flap						
-.0420	7.272	6.618	6.872	7.717	8.436	3.481
-.0350	7.945	7.036	7.400	8.339	9.400	3.352
-.0240	7.454	6.545	6.909	8.056	9.381	3.407
-.0130	5.963	4.963	5.418	6.792	8.472	3.444
.0000	3.018	2.618	3.091	4.113	5.200	2.778
.0582	2.018	1.909	2.218	2.641	2.820	2.018
.1306	1.964	1.836	2.127	2.453	2.473	2.014
.2758	1.891	1.764	1.909	2.132	2.145	[illegible]
.4177	1.764	1.745	1.750	1.900	1.980	[illegible]
.5600	1.745	1.745	1.745	1.906	2.000	[illegible]
.7019	1.709	1.636	1.818	1.943	1.964	[illegible]
.8527	1.654	1.673	1.782	1.887	1.854	[illegible]
1.0000	1.127	.964	.927	1.207	1.036	[illegible]

TABLE I.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(c) $c_f = 0.20c_w$; $\delta_f = 30^\circ$; $\delta_N = 0^\circ$, q = 12.5 lb/sq ft; $C_L = 0.026$

Lower surface

x/c	C_p for -					
	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$
Nose						
.0453	1.471	.401	.054	.019	.012	.006
.0927	1.382	.575	.210	.051	.024	.048
.1852	1.218	.701	.371	.190	.163	.168
.2786	1.129	.707	.425	.272	.223	.257
.3706	1.082	.737	.485	.361	.259	.311
.5565	1.012	.760	.539	.449	.337	.383
.7425	.976	.766	.575	.443	.410	.449
.9272	.929	.743	.587	.481	.446	.479
Wing						
.1944	.871	.743	.623	.538	.500	.515
.2500	.835	.707	.605	.538	.534	.563
.3057	.776	.677	.599	.513	.548	.587
.3612	.706	.605	.575	.513	.534	.587
.4444	.588	.539	.521	.506	.512	.557
.5278	.494	.401	.425	.399	.476	.521
.6115	.435	.311	.299	.285	.349	.407
.6678	.406	.257	.240	.234	.241	.305
.7094	.418	.251	.240	.171	.199	.246
.7372	.447	.287	.210	.177	.163	.192
.7647	.471	.275	.228	.177	.175	.222
.7925	.394	.353	.210	.158	.181	.192
Flap						
.0582	.406	.305	.251	.158	.181	.168
.1306	.435	.287	.240	.127	.157	.174
.2758	.382	.216	.180	.108	.145	.144
.4177	.282	.120	.144	.139	.151	.216
.5600	.200	.144	.186	.146	.223	.311
.7019	.229	.216	.216	.234	.325	.443
.8527	.388	.407	.359	.354	.536	.635
1.0000	.671	.659	.731	.715	1.012	1.689

Upper surface

x/c	C_p for -					
	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$
Nose						
.0000	1.212	.437	3.293	3.525	2.934	2.263
.0453	.623	1.814	3.281	3.114	2.717	2.036
.0927	.812	1.629	3.263	3.108	2.741	2.042
.1852	.923	1.467	3.108	3.158	2.723	2.036
.2786	.965	1.419	2.665	3.158	2.717	2.036
.3706	1.018	1.389	2.132	3.190	2.723	2.042
.5565	1.104	1.377	1.748	3.183	2.741	2.048
.7425	1.135	1.365	1.761	3.108	2.765	2.060
.9272	1.229	1.449	1.743	2.861	2.771	2.072
Wing						
.1944	1.282	1.449	1.677	2.247	2.771	2.072
.2500	1.318	1.473	1.689	1.918	2.669	2.084
.3057	1.412	1.545	1.719	1.785	2.494	2.090
.3612	1.482	1.605	1.748	1.816	2.229	2.072
.4444	1.647	1.731	1.892	1.943	1.970	2.048
.5278	1.930	2.161	2.161	1.906	2.000	2.143
.6115	2.510	2.393	2.375	2.189	2.000	2.107
.6678	2.684	2.679	2.679	2.321	2.145	2.036
.7094	2.789	3.000	2.946	2.564	2.254	2.018
.7372	3.158	3.250	3.304	3.075	2.400	1.929
.7647	3.842	4.036	3.964	3.528	2.764	1.946
.7925	5.701	5.911	5.768	5.170	3.745	1.946
Flap						
-.0420	17.806	18.161	17.268	16.188	10.618	4.411
-.0350	21.227	21.750	20.250	18.829	12.054	4.071
-.0240	22.332	22.821	21.303	19.735	12.672	3.821
-.0130	22.402	22.893	21.428	19.942	12.872	3.893
.0000	14.789	15.107	14.143	13.018	8.418	2.857
.0582	5.666	5.821	5.500	4.868	3.345	1.964
.1306	3.947	4.071	3.821	3.490	2.473	1.929
.2758	2.579	2.732	2.607	2.377	1.854	2.000
.4177	2.017	2.071	1.964	1.566	1.491	1.875
.5600	1.614	1.679	1.571	1.283	1.382	1.786
.7019	1.368	1.357	1.357	1.113	1.454	1.786
.8527	.982	.964	1.018	.849	1.254	1.679
1.0000	.912	.982	.857	.604	.636	.696

TABLE I.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(d) $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 0^\circ$; q = 12.5 lb/sq ft; $C_\mu = 0.057$

Lower surface

x/c	C_p for -					
	α = -12°	α = -8°	α = -4°	α = 0°	α = 4°	α = 8°
Nose						
.0453	1.265	.318	.035	.007	.006	.006
.0927	1.212	.471	.171	.047	.042	.061
.1852	1.112	.588	.324	.200	.137	.188
.2786	1.041	.653	.406	.280	.214	.255
.3706	.976	.671	.441	.340	.292	.309
.5565	.929	.694	.512	.413	.363	.376
.7425	.912	.700	.559	.460	.423	.418
.9272	.876	.694	.547	.467	.452	.461
Wing						
.1944	.806	.700	.582	.513	.488	.533
.2500	.765	.676	.571	.513	.506	.551
.3057	.718	.623	.571	.547	.530	.582
.3612	.671	.623	.559	.487	.530	.570
.4444	.559	.529	.488	.480	.476	.545
.5278	.418	.435	.412	.387	.434	.521
.6115	.359	.306	.306	.233	.339	.400
.6678	.365	.247	.224	.187	.268	.309
.7094	.382	.253	.206	.147	.196	.218
.7372	.418	.265	.212	.147	.190	.194
.7647	.424	.259	.218	.100	.173	.206
.7925	.353	.241	.218	.147	.173	.170
Flap						
.0582	.359	.271	.206	.187	.155	.158
.1306	.412	.282	.229	.147	.149	.170
.2758	.335	.185	.276	.067	.137	.152
.4177	.235	.129	.165	.047	.131	.206
.5600	.182	.153	.159	.093	.208	.327
.7019	.194	.194	.194	.147	.280	.420
.8527	.288	.300	.312	.327	.494	.648
1.0000	.500	.594	.594	.573	.934	1.673

Upper surface

x/c	C_p for -					
	α = -12°	α = -8°	α = -4°	α = 0°	α = 4°	α = 8°
Nose						
.0000	.971	.976	3.464	3.573	2.051	2.273
.0453	.700	2.123	3.170	3.186	2.690	2.036
.0927	.859	1.823	3.170	3.226	2.684	2.042
.1852	.941	1.576	3.182	3.220	2.678	2.024
.2786	.982	1.506	2.923	3.220	2.702	2.030
.3706	1.029	1.465	2.506	3.246	2.690	2.012
.5565	1.088	1.453	1.894	3.293	2.678	2.036
.7425	1.141	1.459	1.718	3.246	2.720	2.036
.9272	1.218	1.500	1.712	3.186	2.732	2.048
Wing						
.1944	1.247	1.488	1.700	2.640	2.744	2.085
.2500	1.312	1.541	1.688	2.113	2.661	2.109
.3057	1.400	1.588	1.723	1.840	2.530	2.097
.3612	1.476	1.659	1.770	1.773	2.286	2.097
.4444	1.629	1.812	1.888	1.933	1.976	2.073
.5278	1.965	2.035	2.158	2.400	1.893	2.164
.6115	2.175	2.430	2.438	2.720	1.911	2.054
.6678	2.491	2.719	2.754	2.840	2.089	2.109
.7094	2.772	2.895	2.982	3.200	2.179	2.000
.7372	3.263	3.403	3.384	3.440	2.375	1.927
.7647	3.877	4.052	4.088	4.280	2.696	1.964
.7925	5.824	6.052	5.912	6.160	3.607	1.927
Flap						
-.0420	20.104	20.262	19.385	20.120	11.946	6.236
-.0350	23.490	23.701	22.262	23.180	13.107	5.491
-.0240	24.823	24.876	23.508	24.300	13.821	5.127
-.0130	25.279	25.560	24.087	24.880	14.339	5.291
.0000	16.315	16.368	15.526	15.940	9.018	3.436
.0582	5.824	5.824	5.649	5.720	3.304	2.018
.1306	4.070	4.088	3.912	3.960	2.446	1.945
.2758	2.789	2.789	2.631	2.720	1.911	1.964
.4177	2.105	2.158	2.017	2.040	1.679	1.891
.5600	1.561	1.702	1.614	1.600	1.464	1.752
.7019	1.210	1.220	1.250	1.420	1.500	1.875
.8527	1.700	1.035	.982	1.000	1.357	1.782
1.0000	.912	.930	.930	.960	.929	.927

TABLE I.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(e) $c_f = 0.20c_w$; $\delta_f = 60°$, $\delta_N = 0°$, q = 12.5 lb/sq ft; $C_\mu = 0.030$

Lower surface

x/c	C_p for -					
	α = -12°	α = -8°	α = -4°	α = 0°	α = 4°	α = 8°
Nose						
.0453	1.104	.263	.018	.013	.031	.013
.0927	1.087	.455	.139	.026	.062	.039
.1852	1.029	.587	.283	.162	.167	.169
.2786	.982	.611	.367	.260	.241	.227
.3706	.953	.653	.416	.299	.272	.292
.5565	.913	.683	.476	.364	.389	.383
.7425	.872	.701	.512	.429	.407	.422
.9272	.843	.695	.524	.467	.463	.480
Wing						
.1944	.791	.671	.548	.519	.500	.539
.2500	.715	.553	.554	.526	.525	.552
.3057	.703	.635	.560	.500	.537	.571
.3612	.622	.599	.530	.513	.531	.584
.4444	.558	.527	.500	.480	.488	.571
.5278	.424	.419	.422	.357	.451	.526
.6115	.326	.311	.295	.247	.352	.403
.6678	.326	.251	.235	.175	.247	.299
.7094	.343	.228	.175	.078	.204	.227
.7372	.366	.257	.199	.117	.191	.221
.7647	.389	.263	.217	.130	.291	.162
.7925	.308	.234	.175	.143	.179	.156
Flap						
.0582	.343	.246	.211	.156	.167	.156
.1306	.378	.294	.259	.162	.167	.175
.2758	.308	.257	.289	.123	.117	.130
.4177	.192	.096	.127	.084	.148	.162
.5600	.157	.152	.139	.104	.191	.260
.7019	.105	.263	.193	.169	.278	.390
.8527	.186	.413	.277	.325	.481	.604
1.0000	.500	.503	.506	.532	.815	1.636

Upper surface

x/c	C_p for -					
	α = -12°	α = -8°	α = -4°	α = 0°	α = 4°	α = 8°
Nose						
.0000	.517	2.012	3.331	3.597	2.987	2.448
.0453	.825	2.329	3.108	3.234	2.864	2.185
.0927	.936	1.946	3.120	3.221	2.845	2.188
.1852	1.011	1.701	3.187	3.227	2.833	2.182
.2786	1.064	1.605	3.114	3.247	2.851	2.188
.3706	1.081	1.557	2.898	3.234	2.851	2.175
.5565	1.145	1.545	2.283	3.299	2.858	2.182
.7425	1.192	1.521	1.849	3.285	2.870	2.182
.9272	1.250	1.557	1.735	3.285	2.895	2.195
Wing						
.1944	1.314	1.593	1.717	2.857	2.895	2.240
.2500	1.354	1.611	1.699	2.324	2.864	2.269
.3057	1.442	1.683	1.759	1.974	2.685	2.260
.3612	1.540	1.748	1.813	1.896	2.413	2.253
.4444	1.697	1.880	1.958	1.974	2.105	2.188
.5278	1.931	2.357	2.273	2.038	2.333	2.327
.6115	2.172	2.643	2.636	2.250	2.333	2.327
.6678	2.552	3.000	3.109	2.769	2.407	2.308
.7094	2.845	3.268	3.218	2.846	2.611	2.192
.7372	3.276	3.714	3.545	3.192	2.778	2.115
.7647	4.000	4.429	4.309	3.981	3.167	2.211
.7925	6.103	6.679	6.382	5.923	4.315	2.135
Flap						
-.0420	23.862	24.964	24.144	23.961	17.037	9.846
-.0350	26.930	28.125	27.072	26.672	18.074	8.538
-.0240	28.310	29.607	28.490	28.018	18.870	8.500
-.0130	29.689	30.928	29.835	29.422	19.999	8.461
.0000	18.258	19.214	18.308	17.749	11.889	5.000
.0582	6.034	6.500	6.200	5.577	3.926	2.269
.1306	4.103	4.536	4.291	3.827	2.889	2.050
.2758	2.845	3.107	3.018	2.365	2.185	2.154
.4177	2.172	2.429	2.273	1.827	1.778	2.077
.5600	1.690	1.857	1.818	1.461	1.593	1.961
.7019	1.500	1.630	1.610	1.100	1.600	2.077
.8527	1.103	1.286	1.218	.769	1.937	1.846
1.0000	1.000	.643	.709	.231	.963	1.577

TABLE I - PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(f) $c_f = 0.10c_w$; $c_f = 60°$; $\delta_N = 0°$; $q = 12.6$ lb/sq ft, $C_\mu = 0.115$

Lower surface

x/c	C_p for -					
	$\alpha = -12°$	$\alpha = -8°$	$\alpha = -4°$	$\alpha = 0°$	$\alpha = 4°$	$\alpha = 8°$
Nose						
.0453	1.006	.155	.018	.036	.019	
.0927	1.050	.329	.123	.021	.091	
.1852	1.026	.478	.270	.121	.118	
.2786	.969	.540	.350	.207	.186	
.3706	.937	.590	.393	.293	.242	
.5565	.918	.621	.454	.371	.311	
.7425	.868	.640	.497	.393	.379	
.9272	.843	.621	.554	.479	.398	
Wing						
.1944	.818	.640	.546	.521	.453	
.2500	.767	.615	.564	.557	.460	
.3057	.704	.609	.546	.521	.484	
.3612	.616	.578	.521	.521	.478	
.4444	.541	.478	.448	.464	.447	
.5278	.390	.391	.374	.343	.385	
.6115	.327	.279	.276	.293	.292	
.6678	.314	.224	.221	.150	.224	
.7094	.321	.217	.184	.150	.180	
.7372	.327	.224	.166	.121	.161	
.7647	.345	.255	.178	.136	.155	
.7925	.289	.205	.184	.157	.149	
Flap						
.0582	.314	.230	.190	.257	.243	
.1306	.363	.242	.215	.129	.130	
.2758	.252	.143	.135	.100	.093	
.4177	.201	.048	.098	.043	.099	
.5600	.113	.075	.129	.064	.155	
.7019	.188	.130	.178	.121	.193	
.8527	.226	.236	.258	.279	.404	
1.0000	.453	.466	.497	.450	.640	

Upper surface

x/c	C_p for -					
	$\alpha = -12°$	$\alpha = -8°$	$\alpha = -4°$	$\alpha = 0°$	$\alpha = 4°$	$\alpha = 8°$
Nose						
.0000	.182	2.477	3.797	3.971	3.050	
.0453	.981	2.429	2.957	3.707	2.944	
.0927	1.049	2.161	2.987	3.678	2.938	
.1852	1.107	1.863	3.024	3.721	2.944	
.2786	1.145	1.733	3.061	3.721	2.913	
.3706	1.163	1.671	3.061	3.742	2.938	
.5565	1.195	1.627	2.766	3.771	2.944	
.7425	1.239	1.596	2.355	3.842	2.949	
.9272	1.333	1.640	2.055	3.821	2.956	
Wing						
.1944	1.358	1.427	1.822	3.450	3.000	
.2500	1.421	1.640	1.810	2.835	2.956	
.3057	1.509	1.708	1.865	2.328	2.845	
.3612	1.604	1.770	1.914	2.107	2.609	
.4444	1.773	1.944	2.055	2.221	2.267	
.5278	2.377	2.407	2.454	2.872	2.333	
.6115	2.660	2.722	2.745	3.298	2.353	
.6678	3.056	3.130	3.273	3.723	2.333	
.7094	3.415	3.463	3.345	4.064	2.593	
.7372	3.868	3.926	3.836	4.574	2.778	
.7647	4.773	4.815	4.582	5.383	3.259	
.7925	7.320	7.278	6.872	8.042	4.630	
Flap						
-.0420	35.828	35.110	33.199	38.382	26.034	
-.0330	37.866	37.499	34.871	40.233	25.944	
-.0240	39.960	39.810	36.871	42.403	27.277	
-.0130	43.286	43.406	40.398	46.552	30.147	
.0000	24.546	24.351	22.563	25.957	16.129	
.0582	6.962	6.907	6.418	7.291	3.889	
.1306	4.736	4.704	4.454	5.042	2.889	
.2758	3.245	3.315	3.182	3.594	2.148	
.4177	2.566	2.537	2.509	2.830	1.833	
.5600	1.981	2.037	1.982	2.106	1.778	
.7019	1.717	1.667	1.673	1.830	1.685	
.8527	1.302	1.315	1.345	1.532	1.481	
1.0000	.585	.648	.655	.660	.759	

TABLE I.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(a) $c_f = 0.20c_w$; $\delta_f = 60°$; $\delta_N = 0°$; $q \approx 12.5$ lb/sq ft, $C_L = 0.179$

Lower surface

x/c	C_p for -					
	α = -12°	α = -8°	α = -4°	α = 0°	α = 4°	α = 8°
Nose						
.0453	.766	.105	.012	.000	.042	
.0927	.838	.278	.096	.006	.006	
.1852	.862	.438	.228	.101	.096	
.2786	.862	.506	.311	.182	.181	
.3706	.838	.543	.353	.258	.235	
.5565	.826	.611	.425	.333	.313	
.7425	.802	.617	.479	.358	.380	
.9272	.784	.617	.473	.390	.398	
Wing						
.1944	.749	.617	.503	.428	.458	
.2500	.683	.617	.527	.459	.470	
.3057	.647	.586	.503	.447	.488	
.3612	.599	.568	.485	.465	.482	
.4444	.503	.488	.449	.415	.452	
.5278	.407	.401	.353	.340	.404	
.6115	.287	.302	.275	.252	.313	
.6678	.257	.204	.204	.164	.211	
.7094	.299	.222	.192	.145	.175	
.7372	.323	.235	.186	.151	.175	
.7647	.347	.265	.216	.151	.181	
.7925	.347	.216	.180	.176	.175	
Flap						
.0582	.323	.222	.186	.170	.151	
.1306	.311	.259	.192	.145	.187	
.2758	.252	.136	.114	.101	.199	
.4177	.174	.093	.054	.063	.114	
.5600	.114	.117	.084	.112	.157	
.7019	.120	.154	.150	.189	.217	
.8527	.210	.191	.204	.245	.416	
1.0000	.449	.475	.461	.459	.596	

Upper surface

x/c	C_p for -					
	α = -12°	α = -8°	α = -4°	α = 0°	α = 4°	α = 8°
Nose						
.0000	.066	4.185	3.623	3.597	3.090	
.0453	1.246	3.012	3.150	3.459	3.006	
.0927	1.257	2.500	3.162	3.434	3.006	
.1852	1.204	2.074	3.174	3.484	3.018	
.2786	1.251	1.901	3.216	3.484	3.006	
.3706	1.246	1.796	3.240	3.497	3.012	
.5565	1.287	1.747	3.138	3.547	3.012	
.7425	1.317	1.691	2.802	3.572	3.006	
.9272	1.389	1.741	2.359	3.610	3.018	
Wing						
.1944	1.401	1.716	1.868	3.346	3.066	
.2500	1.479	1.716	1.766	2.855	3.030	
.3057	1.575	1.796	1.844	2.283	2.954	
.3612	1.647	1.858	1.898	1.994	2.729	
.4444	1.838	2.043	2.066	2.063	2.392	
.5278	2.250	2.593	2.429	2.358	2.291	
.6115	2.661	2.870	2.786	2.604	2.254	
.6678	3.000	3.167	3.143	2.924	2.527	
.7094	3.375	3.685	3.500	3.339	2.691	
.7372	3.964	4.204	3.946	3.905	2.945	
.7647	4.821	5.129	4.857	4.792	3.400	
.7925	7.464	7.870	7.304	7.245	4.927	
Flap						
-.0420	46.607	48.925	46.982	46.809	37.198	
-.0350	46.589	47.999	44.839	45.998	34.471	
-.0240	47.892	49.721	46.339	47.620	35.598	
-.0130	56.160	58.147	54.089	55.356	41.144	
.0000	28.071	29.129	27.035	27.829	20.072	
.0582	6.911	7.204	6.443	6.603	4.218	
.1306	4.643	4.907	4.554	4.509	3.018	
.2758	3.357	3.537	3.179	3.245	2.291	
.4177	2.607	2.704	2.571	2.453	1.982	
.5600	2.107	2.167	1.964	1.943	1.782	
.7019	1.732	1.759	1.625	1.585	1.709	
.8527	1.375	1.352	1.232	1.245	1.491	
1.0000	.643	.611	.607	.491	.782	

TABLE I.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Concluded

(h) $c_f = 0.26c_w$; $\delta_f = 60°$; $\delta_N = 0°$; $q = 12.6$ lb/sq ft; $C_\mu = 0.241$

Lower surface

x/c	C_p for -					
	$\alpha = -12°$	$\alpha = -8°$	$\alpha = -4°$	$\alpha = 0°$	$\alpha = 4°$	$\alpha = 8°$
Nose						
.0453	.582	.061	.006	.007	.037	
.0927	.706	.227	.042	.020	.006	
.1852	.788	.380	.198	.068	.062	
.2786	.753	.436	.269	.142	.161	
.3706	.765	.497	.317	.196	.199	
.5565	.753	.546	.389	.270	.279	
.7425	.729	.577	.437	.345	.323	
.9272	.729	.583	.449	.385	.379	
Wing						
.1944	.706	.595	.485	.419	.410	
.2500	.647	.577	.485	.432	.433	
.3057	.623	.546	.473	.426	.447	
.3612	.574	.534	.461	.432	.447	
.4444	.494	.448	.413	.399	.410	
.5278	.374	.350	.341	.318	.360	
.6115	.265	.239	.257	.223	.279	
.6678	.265	.215	.180	.142	.186	
.7094	.276	.202	.174	.135	.161	
.7372	.276	.209	.168	.088	.155	
.7647	.329	.233	.210	.108	.143	
.7925	.235	.288	.184	.122	.137	
Flap						
.0582	.282	.251	.150	.149	.155	
.1306	.329	.251	.174	.128	.193	
.2758	.224	.141	.204	.061	.087	
.4177	.118	.061	.078	.054	.112	
.5600	.129	.092	.120	.061	.124	
.7019	.129	.147	.150	.101	.186	
.8527	.229	.196	.222	.182	.311	
1.0000	.471	.558	.449	.439	.571	

Upper surface

x/c	C_p for -					
	$\alpha = -12°$	$\alpha = -8°$	$\alpha = -4°$	$\alpha = 0°$	$\alpha = 4°$	$\alpha = 8°$
Nose						
.0000	.106	5.490	3.808	3.878	3.230	
.0453	1.459	3.404	3.210	3.804	3.155	
.0927	1.412	2.760	3.228	3.790	3.143	
.1852	1.335	2.239	3.240	3.790	3.145	
.2786	1.312	2.018	3.273	3.804	3.168	
.3706	1.318	1.859	3.299	3.831	3.149	
.5565	1.353	1.779	3.245	3.851	3.174	
.7425	1.365	1.748	2.988	3.891	3.180	
.9272	1.441	1.791	2.587	3.939	3.199	
Wing						
.1944	1.471	1.754	2.042	3.709	3.230	
.2500	1.518	1.748	1.892	3.148	3.192	
.3057	1.582	1.822	1.922	2.520	3.124	
.3612	1.676	1.883	1.994	2.128	2.932	
.4444	1.853	2.061	2.144	2.148	2.528	
.5278	2.281	2.600	2.534	2.755	2.389	
.6115	2.614	2.891	2.911	3.184	2.352	
.6678	2.842	3.273	3.250	3.510	2.355	
.7094	3.403	3.673	3.661	4.041	2.833	
.7372	3.912	4.218	4.196	4.612	3.093	
.7647	4.702	5.236	5.071	5.612	3.685	
.7925	7.526	8.036	7.714	8.531	5.352	
Flap						
-.0420	55.489	57.179	55.589	63.183	51.537	
-.0350	46.857	48.452	47.428	54.244	44.906	
-.0240	52.875	54.616	52.357	59.346	44.962	
-.0130	67.312	69.506	66.747	75.693	55.073	
.0000	31.051	32.144	30.803	34.938	24.666	
.0582	6.807	7.200	6.911	7.633	4.537	
.1306	4.664	4.909	4.732	5.204	3.285	
.2758	3.281	3.509	3.482	3.633	2.481	
.4177	2.544	2.727	2.696	2.957	2.148	
.5600	1.982	2.218	2.232	2.367	1.870	
.7019	1.684	1.873	1.804	2.041	1.778	
.8527	1.439	1.473	1.446	1.531	1.530	
1.0000	.702	.709	.752	.653	.778	

TABLE II - PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE

(c) $c_f = 0.20c_w$, $\delta_f = 60^\circ$, $\delta_N = 4.75^\circ$; $q \approx 12.5$ lb/sq ft; $C_L = 0$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.811	1.654	1.873	1.038	.199	.003	.006	.006
.0927	1.786	1.648	1.885	1.069	.373	.142	.057	.071
.1852	1.799	1.648	1.885	1.044	.534	.296	.178	.208
.2786	1.799	1.654	1.824	.987	.596	.352	.248	.244
.3706	1.824	1.673	1.691	.969	.634	.407	.299	.301
.5565	1.824	1.660	1.382	.924	.665	.475	.357	.353
.7425	1.830	1.679	1.157	.868	.677	.506	.408	.397
.9272	1.836	1.698	1.036	.818	.590	.494	.420	.397
Wing								
.1944	1.824	1.711	1.042	.855	.665	.568	.497	.468
.2500	1.874	1.767	.994	.849	.714	.605	.535	.500
.3057	1.880	1.748	.927	.824	.702	.623	.573	.545
.3612	1.906	1.736	.909	.811	.677	.630	.573	.570
.4444	1.899	1.660	.818	.717	.665	.599	.580	.564
.5278	1.824	1.547	.727	.610	.565	.580	.535	.532
.6115	1.711	1.396	.612	.472	.441	.395	.427	.397
.6678	1.585	1.283	.576	.421	.329	.309	.325	.308
.7094	1.484	1.207	.551	.390	.236	.235	.255	.231
.7372	1.415	1.101	.527	.421	.230	.204	.204	.186
.7647	1.321	1.000	.492	.415	.242	.210	.204	.192
.7925	1.214	.874	.515	.390	.261	.185	.197	.179
Flap								
.0582	1.057	.918	.527	.396	.248	.191	.191	.167
.1306	1.075	.943	.600	.453	.267	.179	.210	.160
.2758	1.296	.887	.436	.340	.205	.148	.172	.128
.4177	1.170	.805	.448	.302	.211	.185	.236	.199
.5600	1.157	.780	.503	.327	.311	.265	.325	.282
.7019	1.132	.742	.594	.453	.497	.469	.490	.468
.8527	1.107	1.138	.854	.774	.689	.673	.707	.699
1.0000	1.629	1.698	1.842	1.899	1.873	1.765	1.885	1.904

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.836	1.761	1.824	.478	2.366	3.209	3.025	2.205
.0453	.013	.038	.242	1.000	3.317	2.969	2.688	1.891
.0927	.050	.176	.418	1.069	2.068	2.987	2.688	1.891
.1852	.157	.308	.594	1.107	1.826	3.012	2.713	1.885
.2786	.252	.440	.673	1.126	1.714	3.030	2.688	1.878
.3706	.340	.503	.758	1.157	1.671	3.037	2.688	1.878
.5565	.440	.641	.867	1.220	1.621	2.926	2.700	1.885
.7425	.560	.730	.951	1.289	1.615	2.811	2.688	1.885
.9272	.692	.868	1.109	1.428	1.764	2.154	2.688	1.897
Wing								
.1944	.767	.943	1.115	1.358	1.633	1.778	2.637	1.910
.2500	.824	.969	1.133	1.321	1.522	1.654	2.548	1.936
.3057	.899	1.013	1.151	1.327	1.497	1.605	2.427	1.923
.3612	.943	1.057	1.194	1.346	1.484	1.580	2.261	1.936
.4444	1.044	1.132	1.273	1.409	1.528	1.623	2.095	1.936
.5278	1.321	1.453	1.473	1.623	1.741	1.778	2.038	2.154
.6115	1.434	1.566	1.564	1.755	1.741	1.778	2.038	2.077
.6678	1.528	1.623	1.600	1.660	1.759	1.815	2.058	2.115
.7094	1.528	1.585	1.600	1.755	1.759	1.796	1.923	2.135
.7372	1.509	1.641	1.600	1.679	1.815	1.778	1.942	2.135
.7647	1.641	1.698	1.745	1.792	1.852	1.815	2.000	2.173
.7925	1.736	1.792	1.854	1.924	1.926	1.870	2.096	2.173
Flap								
-.0420	2.000	2.075	2.182	2.019	2.167	1.944	2.327	2.154
-.0350	1.943	1.962	2.073	2.000	2.093	1.852	2.269	2.115
-.0240	1.849	1.868	1.945	1.924	1.963	1.852	2.211	2.077
-.0130	1.736	1.755	1.854	1.924	1.907	1.815	2.115	2.135
.0000	1.811	1.773	1.891	1.943	1.870	1.833	2.154	2.077
.0582	1.549	1.849	1.873	1.981	2.037	1.852	2.173	2.135
.1306	1.773	1.868	1.909	2.019	2.018	1.852	2.211	2.115
.2758	2.000	1.849	2.018	2.000	2.074	1.889	2.385	2.192
.4177	1.906	1.868	1.927	1.962	2.000	1.944	2.211	2.115
.5600	1.887	1.773	1.873	1.962	1.944	1.889	2.211	2.058
.7019	1.924	1.887	1.927	1.981	2.018	1.963	2.211	2.115
.8527	1.830	1.849	1.945	1.924	1.907	1.815	2.115	1.904
1.0000	1.906	1.811	1.836	1.849	1.833	1.778	1.961	1.961

TABLE II.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(b) $c_f = 0.30c_w$; $\delta_f = 60^\circ$; $\delta_N = 4.75^\circ$; $q \approx 12.5$ lb/sq ft; $C_\mu = 0.013$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.639	1.325	1.935	.703	.078	.018	.013	.013
.0927	1.639	1.349	1.625	.806	.223	.094	.065	.070
.1852	1.639	1.366	1.386	.842	.392	.218	.159	.159
.2786	1.645	1.360	1.222	.830	.470	.271	.217	.229
.3706	1.633	1.383	1.164	.824	.488	.329	.280	.280
.5565	1.657	1.372	1.047	.812	.548	.394	.331	.344
.7425	1.669	1.383	.971	.764	.566	.406	.382	.369
.9272	1.681	1.395	.895	.715	.542	.435	.395	.401
Wing								
.1944	1.705	1.418	.889	.764	.602	.494	.446	.471
.2500	1.699	1.407	.871	.739	.620	.518	.490	.522
.3057	1.735	1.372	.836	.739	.651	.529	.522	.554
.3612	1.753	1.296	.778	.703	.633	.541	.548	.567
.4444	1.741	1.209	.684	.642	.598	.518	.516	.560
.5278	1.687	1.099	.591	.539	.524	.482	.497	.529
.6115	1.548	1.017	.485	.424	.404	.376	.382	.401
.6678	1.440	.965	.468	.303	.301	.282	.267	.299
.7094	1.367	.918	.468	.303	.241	.212	.210	.229
.7372	1.301	.872	.479	.327	.223	.200	.197	.178
.7647	1.211	.849	.462	.315	.241	.200	.197	.166
.7925	1.084	.779	.462	.315	.223	.182	.172	.185
Flap								
.0582	1.078	.814	.474	.315	.217	.188	.146	.166
.1306	1.120	.831	.520	.339	.229	.200	.146	.172
.2758	1.163	.796	.439	.242	.181	.141	.115	.146
.4177	1.114	.709	.427	.218	.187	.194	.166	.197
.5600	1.060	.715	.392	.273	.295	.265	.267	.280
.7019	1.012	.680	.433	.412	.428	.371	.382	.465
.8527	1.253	.959	.766	.661	.645	.565	.592	.675
1.0000	1.518	1.436	1.596	1.660	1.645	1.576	1.592	1.872

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.681	1.511	2.777	.048	4.602	3.464	2.942	2.165
.0453	.006	.105	.322	1.370	3.560	3.082	2.611	1.879
.0927	.090	.279	.520	1.321	2.747	3.082	2.599	1.917
.1852	.229	.395	.661	1.285	2.181	3.070	2.611	1.911
.2786	.343	.506	.772	1.247	1.934	3.088	2.605	1.892
.3706	.428	.581	.854	1.279	1.858	3.117	2.624	1.885
.5565	.548	.698	.930	1.339	1.777	3.153	2.611	1.892
.7425	.639	.796	1.029	1.388	1.747	3.106	2.611	1.911
.9272	.789	.942	1.164	1.533	1.867	2.870	2.592	1.911
Wing								
.1944	.873	.982	1.187	1.467	1.741	2.270	2.611	1.936
.2500	.946	1.011	1.199	1.430	1.614	1.923	2.579	1.949
.3057	1.006	1.075	1.234	1.436	1.633	1.782	2.471	1.968
.3612	1.060	1.104	1.257	1.473	1.626	1.718	2.350	1.955
.4444	1.169	1.209	1.351	1.527	1.669	1.765	2.172	1.936
.5278	1.527	1.431	1.719	1.764	1.782	1.789	2.288	2.173
.6115	1.673	1.552	1.807	1.854	1.873	1.965	2.211	2.154
.6678	2.036	1.707	1.754	2.018	1.945	2.070	2.173	2.231
.7094	1.782	1.707	1.930	1.945	2.091	2.140	2.115	2.115
.7372	1.982	1.828	2.035	2.109	2.236	2.281	2.154	2.154
.7647	2.182	2.017	2.210	2.327	2.491	2.456	2.211	2.154
.7925	2.836	2.527	2.667	2.836	3.036	3.088	2.365	2.077
Flap								
-.0420	7.054	6.414	6.544	6.727	7.091	7.245	4.565	3.442
-.0350	8.091	7.207	7.158	7.527	7.963	8.228	5.250	3.346
-.0240	7.872	6.879	6.719	6.909	7.527	8.000	5.500	3.365
-.0130	6.400	5.586	5.175	5.200	6.291	6.719	5.573	3.604
.0000	3.691	3.000	2.667	2.830	3.527	4.210	3.942	2.808
.0582	2.400	2.017	1.912	2.200	2.364	2.509	2.327	2.250
.1306	2.145	1.862	1.965	2.127	2.254	2.281	2.135	2.192
.2758	1.982	1.741	1.947	1.982	2.054	1.982	2.000	2.231
.4177	1.818	1.672	1.860	1.909	1.873	1.789	1.904	2.077
.5600	1.764	1.552	1.719	1.836	1.745	1.737	1.769	2.019
.7019	1.782	1.569	1.684	1.909	1.764	1.649	1.865	2.000
.8527	1.673	1.500	1.684	1.818	1.727	1.403	1.711	1.962
1.0000	1.745	1.465	1.596	1.673	1.654	1.368	1.692	1.904

TABLE II.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(c) $c_f = 0.20c_w$; $\delta_f = 60°$; $\delta_N = 4.75°$, $q = 12.5$ lb/sq ft, $C_\mu = 0.026$

Lower surface

x/c	C_p for -							
	$\alpha = -16°$	$\alpha = -12°$	$\alpha = -8°$	$\alpha = -4°$	$\alpha = 0°$	$\alpha = 4°$	$\alpha = 8°$	$\alpha = 12°$
Nose								
.0453	1.223	2.257	.770	.137	.030	.006	.034	.313
.0927	1.229	1.034	.861	.298	.138	.056	.057	.063
.1852	1.206	1.417	.848	.460	.264	.180	.166	.151
.2786	1.206	1.251	.836	.516	.327	.267	.234	.226
.3706	1.206	1.129	.806	.547	.384	.317	.257	.308
.5565	1.218	1.018	.794	.590	.440	.373	.345	.352
.7425	1.218	.945	.733	.571	.440	.416	.383	.377
.9272	1.229	.853	.691	.559	.472	.391	.383	.402
Wing								
.1944	1.218	.846	.703	.596	.522	.483	.446	.478
.2500	1.188	.822	.691	.609	.553	.503	.480	.509
.3057	1.171	.761	.642	.609	.547	.522	.503	.553
.3612	1.153	.705	.606	.571	.535	.503	.514	.566
.4444	1.088	.620	.545	.528	.497	.466	.509	.547
.5278	1.071	.503	.461	.422	.402	.441	.469	.478
.6115	1.018	.472	.321	.323	.308	.311	.363	.415
.6678	.982	.460	.303	.236	.233	.217	.291	.302
.7094	.976	.478	.309	.211	.201	.168	.211	.220
.7372	.918	.509	.315	.217	.201	.174	.200	.176
.7647	.841	.497	.351	.217	.189	.161	.206	.201
.7925	.747	.429	.297	.211	.201	.180	.177	.176
Flap								
.0582	.741	.472	.303	.205	.201	.155	.160	.164
.1306	.776	.528	.351	.230	.201	.137	.149	.129
.2758	.847	.429	.236	.149	.126	.099	.143	.151
.4177	.829	.356	.127	.112	.126	.130	.183	.214
.5600	.806	.331	.121	.161	.157	.168	.274	.308
.7019	.741	.429	.127	.230	.245	.261	.394	.459
.8527	.700	.429	.333	.373	.384	.416	.571	.667
1.0000	.700	.589	.606	.727	.748	.801	1.440	1.855

Upper surface

x/c	C_p for -							
	$\alpha = -16°$	$\alpha = -12°$	$\alpha = -8°$	$\alpha = -4°$	$\alpha = 0°$	$\alpha = 4°$	$\alpha = 8°$	$\alpha = 12°$
Nose								
.0000	1.323	2.589	.127	3.466	3.685	3.522	2.617	2.157
.0453	.124	.307	1.267	2.733	3.214	3.155	2.383	1.931
.0927	.276	.515	1.267	2.323	3.245	3.130	2.388	1.912
.1852	.429	.693	1.248	2.012	3.251	3.149	2.383	1.918
.2786	.535	.797	1.279	1.882	3.255	3.155	2.394	1.899
.3706	.623	.883	1.303	1.826	3.245	3.149	2.388	1.899
.5565	.765	1.000	1.357	1.801	3.094	3.205	2.366	1.912
.7425	.888	1.116	1.430	1.838	2.704	3.211	2.343	1.906
.9272	1.006	1.288	1.612	1.994	2.277	3.186	2.314	1.912
Wing								
.1944	1.071	1.330	1.533	1.845	1.994	2.832	2.343	1.918
.2500	1.147	1.337	1.521	1.783	1.924	2.478	2.326	1.937
.3057	1.206	1.399	1.588	1.807	1.924	2.155	2.246	1.912
.3612	1.300	1.478	1.660	1.845	1.950	1.994	2.166	1.899
.4444	1.453	1.644	1.806	1.969	2.057	1.932	2.011	1.931
.5278	1.684	2.164	2.218	2.278	2.434	2.167	1.983	2.151
.6115	1.965	2.436	2.436	2.593	2.660	2.296	1.879	2.151
.6678	2.246	2.691	2.727	2.870	2.887	2.518	1.948	2.226
.7094	2.491	2.909	3.054	3.130	3.113	2.695	1.879	2.151
.7372	2.842	3.291	3.418	3.481	3.434	2.926	1.810	2.113
.7647	3.439	3.891	4.182	4.149	4.000	3.463	1.897	2.170
.7925	5.245	5.909	6.109	6.204	5.698	4.741	2.000	2.057
Flap								
-.0420	16.420	17.599	18.509	18.203	16.697	13.666	4.672	4.302
-.0350	19.981	21.544	22.472	22.148	20.093	16.296	4.827	4.113
-.0240	21.017	22.690	23.617	23.333	21.150	17.148	4.983	3.830
-.0130	21.192	22.872	23.708	23.499	21.301	17.296	5.190	3.887
.0000	13.894	15.072	15.636	15.351	13.905	11.296	3.500	2.943
.0582	5.175	5.854	5.945	5.981	5.490	4.407	2.000	2.226
.1306	3.596	4.054	4.127	4.130	3.868	3.148	1.845	2.189
.2758	2.544	2.954	2.800	2.852	2.623	2.222	1.759	2.264
.4177	1.895	2.182	2.073	2.130	1.981	1.704	1.724	2.151
.5600	1.509	1.764	1.636	1.593	1.679	1.352	1.586	2.038
.7019	1.246	1.491	1.364	1.289	1.358	1.133	1.638	2.019
.8527	1.017	1.109	.982	1.074	1.094	1.074	1.500	1.924
1.0000	.667	.709	.655	.796	.830	.852	1.431	1.887

TABLE II.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(d) $c_f = 0.20c_W$, $\alpha_f = 60^\circ$; $\delta_N = 4.75^\circ$; $q = 12.5$ lb/sq ft; $C_\mu = 0.057$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.220	1.738	.693	.099	.012	.036	.012	.000
.0927	1.232	1.536	.783	.247	.083	.060	.036	.041
.1852	1.196	1.333	.789	.389	.220	.192	.164	.176
.2786	1.220	1.167	.783	.481	.286	.263	.230	.247
.3706	1.220	1.071	.771	.500	.327	.305	.291	.282
.5565	1.220	.982	.741	.543	.411	.377	.345	.329
.7425	1.226	.899	.675	.555	.417	.401	.394	.388
.9272	1.232	.792	.681	.543	.399	.395	.394	.388
Wing								
.1944	1.208	.821	.651	.562	.470	.455	.455	.429
.2500	1.190	.780	.657	.580	.488	.491	.485	.482
.3057	1.173	.720	.633	.580	.506	.527	.515	.529
.3612	1.137	.655	.554	.568	.500	.503	.533	.547
.4444	1.101	.583	.512	.506	.446	.479	.503	.529
.5278	1.054	.464	.410	.407	.393	.407	.455	.494
.6115	.994	.429	.301	.302	.304	.311	.358	.388
.6678	.982	.429	.277	.247	.220	.251	.289	.306
.7094	.925	.440	.271	.185	.185	.204	.212	.229
.7372	.911	.476	.271	.216	.167	.216	.170	.212
.7647	.827	.482	.343	.198	.167	.192	.145	.212
.7925	.744	.393	.253	.198	.167	.174	.115	.194
Flap								
.0582	.726	.429	.253	.198	.196	.174	.152	.165
.1306	.750	.482	.277	.228	.208	.180	.194	.153
.2758	.827	.405	.163	.148	.113	.126	.152	.153
.4177	.821	.310	.072	.123	.113	.126	.200	.200
.5600	.780	.244	.072	.167	.155	.150	.291	.300
.7019	.768	.226	.169	.216	.226	.251	.412	.447
.8527	.673	.381	.277	.327	.339	.377	.600	.647
1.0000	.637	.530	.548	.568	.601	.715	1.521	1.776

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.292	2.702	.024	4.697	3.655	3.401	2.745	2.053
.0453	.173	.345	1.349	3.098	3.303	3.138	2.503	1.859
.0927	.310	.565	1.343	2.555	3.291	3.126	2.527	1.841
.1852	.482	.738	1.323	2.154	3.315	3.138	2.527	1.829
.2786	.571	.827	1.301	1.987	3.339	3.138	2.527	1.829
.3706	.667	.899	1.331	1.901	3.351	3.114	2.503	1.847
.5565	.792	1.012	1.386	1.858	3.256	3.162	2.491	1.829
.7425	.911	1.131	1.446	1.870	2.827	3.198	2.466	1.835
.9272	1.054	1.286	1.602	2.000	2.363	3.180	2.448	1.847
Wing								
.1944	1.137	1.321	1.566	1.889	1.976	2.952	2.460	1.847
.2500	1.184	1.357	1.584	1.839	1.899	2.623	2.418	1.870
.3057	1.274	1.423	1.596	1.833	1.887	2.275	2.376	1.865
.3612	1.339	1.494	1.645	1.889	1.922	2.048	2.309	1.882
.4444	1.506	1.637	1.783	2.000	2.018	1.958	2.163	1.876
.5278	1.768	2.019	2.273	2.315	2.286	2.194	2.164	2.053
.6115	2.107	2.304	2.545	2.611	2.536	2.411	2.036	2.017
.6678	2.393	2.518	2.891	2.889	2.821	2.607	2.109	2.123
.7094	2.625	2.893	3.145	3.130	3.018	2.839	2.018	1.982
.7372	3.000	3.250	3.545	3.574	3.322	2.964	2.018	2.000
.7647	3.679	3.982	4.363	4.352	4.000	3.589	2.091	2.093
.7925	5.571	5.946	6.309	6.352	5.804	4.982	2.200	1.982
Flap								
-.0420	18.893	19.553	20.617	20.536	18.250	15.321	6.672	5.386
-.0350	22.607	23.482	24.781	24.610	21.553	18.000	6.691	5.087
-.0240	23.857	24.893	26.108	25.851	22.786	18.928	6.672	4.719
-.0130	24.446	25.268	26.672	26.407	23.286	19.500	7.018	4.702
.0000	15.661	16.268	17.181	16.925	14.893	12.428	4.363	3.193
.0582	5.554	5.893	6.254	6.204	5.464	4.571	2.164	2.017
.1306	3.857	4.089	4.363	4.333	3.839	3.179	1.964	1.947
.2758	2.696	2.839	3.054	2.926	2.589	2.286	1.982	1.982
.4177	2.000	2.125	2.327	2.222	2.018	1.839	1.891	1.965
.5600	1.571	1.804	1.818	1.685	1.571	1.500	1.764	1.912
.7019	1.375	1.482	1.454	1.463	1.304	1.411	1.727	1.912
.8527	1.089	1.161	1.200	.963	1.054	1.196	1.691	1.842
1.0000	.643	.589	.655	.630	.679	.804	1.582	1.807

TABLE II.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(e) $c_f = 0.20c_w$; $\delta_f = 60°$, $\delta_N = 4.75°$; q = 12.5 lb/sq ft; $C_\mu = 0.034$

Lower surface

x/c	C_p for -							
	α = -16°	α = -12°	α = -8°	α = -4°	α = 0°	α = 4°	α = 8°	α = 12°
Nose								
.0453	1.216	1.587	.559	.063	.018	.012	.018	.000
.0927	1.234	1.413	.673	.189	.078	.030	.061	.049
.1852	1.198	1.192	.738	.337	.210	.120	.158	.154
.2786	1.204	1.075	.726	.417	.263	.205	.206	.235
.3706	1.222	.965	.714	.443	.317	.235	.285	.278
.5565	1.222	.918	.696	.469	.359	.313	.339	.358
.7425	1.246	.849	.655	.503	.401	.343	.364	.358
.9272	1.234	.762	.651	.480	.401	.355	.388	.395
Wing								
.1944	1.216	.756	.625	.514	.449	.440	.448	.451
.2500	1.174	.727	.625	.526	.461	.452	.485	.500
.3057	1.132	.668	.601	.526	.485	.470	.485	.525
.3612	1.072	.616	.571	.514	.461	.446	.521	.543
.4444	1.015	.517	.488	.469	.449	.446	.497	.537
.5278	.964	.419	.405	.389	.365	.392	.461	.500
.6115	.934	.395	.286	.269	.263	.271	.364	.383
.6678	.916	.384	.238	.211	.180	.209	.273	.296
.7094	.892	.395	.262	.206	.150	.163	.206	.235
.7372	.868	.430	.268	.200	.168	.169	.176	.198
.7647	.772	.434	.321	.194	.180	.175	.194	.191
.7925	.677	.337	.224	.189	.174	.139	.158	.191
Flap								
.0582	.689	.378	.262	.194	.150	.139	.158	.185
.1306	.737	.424	.310	.211	.162	.157	.170	.167
.2758	.737	.360	.179	.160	.126	.096	.139	.136
.4177	.731	.279	.069	.114	.108	.108	.176	.216
.5600	.701	.203	.095	.137	.120	.163	.285	.309
.7019	.665	.192	.149	.194	.198	.211	.376	.451
.8527	.653	.308	.238	.251	.305	.386	.545	.667
1.0000	.605	.465	.476	.503	.509	.560	1.467	1.822

Upper surface

x/c	C_p for -							
	α = -16°	α = -12°	α = -8°	α = -4°	α = 0°	α = 4°	α = 8°	α = 12°
Nose								
.0000	1.240	2.232	.113	5.645	3.790	3.355	2.727	2.197
.0453	.162	.395	1.236	3.600	3.535	3.151	2.472	1.975
.0927	.311	.645	1.458	2.891	3.335	3.151	2.472	1.987
.1852	.497	.791	1.369	2.287	3.359	3.157	2.448	1.994
.2786	.617	.889	1.363	2.017	3.389	3.151	2.448	1.969
.3706	.689	.953	1.375	1.897	3.383	3.163	2.485	1.975
.5565	.826	1.070	1.440	1.863	3.347	3.169	2.454	1.975
.7425	.916	1.158	1.500	1.823	3.236	3.223	2.485	1.987
.9272	1.072	1.314	1.678	1.971	2.533	3.223	2.491	1.981
Wing								
.1944	1.156	1.383	1.619	1.868	2.042	3.066	2.491	2.343
.2500	1.240	1.401	1.595	1.806	1.934	2.801	2.503	2.037
.3057	1.287	1.465	1.655	1.817	1.922	2.428	2.436	2.037
.3612	1.359	1.546	1.714	1.957	1.958	2.151	2.382	2.049
.4444	1.527	1.686	1.833	1.948	2.030	1.994	2.260	2.061
.5278	1.875	1.983	2.214	2.224	2.256	2.291	2.254	2.204
.6115	2.196	2.241	2.518	2.483	2.536	2.527	2.127	2.241
.6678	2.536	2.586	2.875	2.845	2.875	2.764	2.127	2.296
.7094	2.786	2.828	3.179	3.103	3.071	2.982	2.018	2.185
.7372	3.161	3.241	3.482	3.534	3.464	3.236	2.273	2.222
.7647	3.893	3.948	4.339	4.276	4.179	3.818	2.091	2.185
.7925	5.946	6.086	6.500	6.396	6.143	5.382	2.309	2.148
Flap								
-.0420	23.214	23.310	24.786	23.689	23.161	20.290	9.381	8.555
-.0350	26.857	27.293	28.982	27.568	26.232	23.054	9.236	8.000
-.2240	28.375	28.879	30.607	29.155	27.571	24.363	9.290	7.537
-.0130	29.768	30.275	32.018	30.482	29.000	25.653	9.763	7.592
.0000	17.982	18.379	19.464	18.551	17.553	15.226	5.491	4.518
.0582	5.875	6.017	6.411	6.207	5.786	4.963	2.236	2.296
.1306	4.000	4.238	4.411	4.276	4.000	3.509	2.036	2.222
.2758	2.529	2.948	3.236	3.034	2.722	2.527	1.964	2.333
.4177	2.143	2.224	2.321	2.289	2.143	1.964	1.945	2.222
.5600	1.768	1.741	1.875	1.793	1.696	1.745	1.836	2.167
.7019	1.429	1.483	1.554	1.530	1.429	1.454	1.854	2.074
.8527	1.179	1.155	1.250	1.172	1.107	1.291	1.745	2.074
1.0000	.679	.569	.518	.621	.607	.818	1.454	1.889

TABLE II.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(f) $c_f = 0.30c_w$, $\delta_f = 60^\circ$; $\delta_N = 4.75^\circ$; $q = 12.5$ lb/sq ft; $C_\mu = 0.117$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.236	1.494	.389	.024	.030	.006	.018	.006
.0927	1.248	1.339	.545	.141	.036	.006	.041	.031
.1852	1.230	1.154	.635	.294	.157	.088	.147	.141
.2786	1.230	1.025	.665	.385	.217	.181	.200	.196
.3706	1.224	.944	.665	.394	.277	.228	.253	.258
.5565	1.224	.863	.653	.453	.343	.275	.312	.313
.7425	1.218	.815	.635	.459	.355	.327	.341	.350
.9272	1.200	.753	.581	.447	.375	.310	.347	.380
Wing								
.1944	1.157	.716	.599	.494	.410	.392	.412	.448
.2500	1.103	.704	.587	.494	.440	.415	.459	.460
.3057	1.012	.654	.587	.494	.446	.446	.485	.503
.3612	.945	.593	.557	.488	.452	.421	.500	.521
.4444	.854	.488	.461	.441	.361	.415	.494	.515
.5278	.812	.383	.365	.365	.349	.374	.441	.472
.6115	.794	.352	.234	.259	.253	.287	.335	.374
.6678	.770	.346	.222	.176	.181	.193	.229	.313
.7094	.751	.370	.234	.182	.145	.170	.224	.239
.7372	.727	.395	.234	.176	.139	.164	.182	.200
.7647	.667	.407	.293	.200	.169	.164	.188	.178
.7925	.570	.309	.192	.188	.157	.140	.153	.190
Flap								
.0582	.582	.315	.216	.188	.157	.152	.141	.178
.1306	.594	.370	.228	.188	.169	.164	.141	.166
.2758	.618	.346	.156	.141	.175	.170	.141	.141
.4177	.576	.265	.079	.071	.181	.076	.147	.178
.5600	.558	.173	.102	.118	.120	.129	.241	.301
.7019	.509	.148	.126	.165	.181	.193	.359	.417
.8527	.461	.222	.210	.224	.271	.287	.512	.595
1.0000	.491	.407	.371	.447	.458	.474	1.223	1.613

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.236	1.500	.653	4.535	3.837	3.321	2.723	2.221
.0453	.218	.574	1.928	4.253	3.554	3.198	2.470	2.092
.0927	.370	.759	1.737	3.941	3.542	3.175	2.465	2.104
.1852	.515	.876	1.575	2.741	3.530	3.175	2.476	2.092
.2786	.624	.963	1.515	2.335	3.536	3.187	2.488	2.086
.3706	.703	1.025	1.503	2.135	3.560	3.192	2.476	2.067
.5565	.842	1.136	1.551	1.976	3.594	3.198	2.512	2.073
.7425	.945	1.234	1.599	1.953	3.367	3.245	2.506	2.092
.9272	1.103	1.432	1.754	2.028	2.970	3.257	2.517	2.104
Wing								
.1944	1.188	1.457	1.683	1.965	2.265	3.128	2.559	2.110
.2500	1.224	1.481	1.659	1.894	2.024	2.894	2.576	2.135
.3057	1.309	1.562	1.737	1.900	2.000	2.532	2.535	2.147
.3612	1.388	1.636	1.772	1.953	2.036	2.234	2.453	2.141
.4444	1.570	1.815	1.922	2.082	2.132	2.046	2.347	2.153
.5278	2.000	2.148	2.304	2.386	2.545	2.246	2.316	2.345
.6115	2.291	2.444	2.643	2.684	2.891	2.403	2.140	2.309
.6678	2.782	2.833	3.018	3.052	3.218	2.719	2.263	2.236
.7094	3.018	3.241	3.339	3.438	3.600	2.912	2.105	2.309
.7372	3.382	3.630	3.768	3.824	3.927	3.228	2.123	2.273
.7647	4.218	4.518	4.732	4.732	4.745	3.877	2.228	2.309
.7925	6.509	7.055	7.161	7.122	7.036	5.666	2.403	2.236
Flap								
-.0420	32.926	34.536	33.982	33.121	32.217	27.806	16.175	14.781
-.0350	35.507	37.684	37.357	36.261	35.017	29.841	15.175	13.399
-.0240	37.380	39.610	39.214	38.241	36.889	31.683	15.420	12.854
-.0130	41.253	43.814	43.125	41.945	40.416	34.805	18.824	13.381
.0000	22.581	23.944	25.750	23.227	22.435	18.753	8.245	6.672
.0582	6.218	6.648	6.768	6.684	6.509	5.158	2.368	2.309
.1306	4.291	4.574	4.518	4.579	4.563	3.614	2.158	2.254
.2758	3.018	3.241	3.250	3.245	3.218	2.561	2.140	2.382
.4177	2.327	2.444	2.429	2.509	2.545	2.017	2.017	2.254
.5600	1.854	1.889	1.946	1.930	2.018	1.614	1.912	2.200
.7019	1.418	1.500	1.554	1.596	1.564	1.456	1.912	2.127
.8527	1.236	1.204	1.286	1.298	1.309	1.228	1.824	2.018
1.0000	.582	.500	.536	.614	.655	.667	1.386	1.654

TABLE II.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(c) $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 4.75^\circ$; $q = 12.5$ lb/sq ft, $C_\mu = 0.187$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.281	1.276	.267	.035	.000	.036	.018	.000
.0927	1.269	1.178	.424	.129	.024	.018	.030	.007
.1852	1.251	1.046	.535	.275	.121	.109	.144	.100
.2786	1.263	.960	.581	.339	.206	.145	.192	.147
.3706	1.281	.885	.587	.386	.248	.206	.240	.233
.5565	1.263	.839	.575	.427	.303	.285	.317	.293
.7425	1.234	.770	.593	.439	.351	.327	.341	.333
.9272	1.192	.690	.541	.433	.345	.309	.347	.333
Wing								
.1944	1.132	.695	.558	.468	.376	.370	.401	.413
.2500	1.012	.667	.558	.485	.436	.412	.455	.453
.3057	.952	.621	.541	.491	.424	.430	.467	.503
.3612	.856	.557	.517	.474	.412	.418	.455	.507
.4444	.754	.471	.465	.433	.400	.400	.461	.507
.5278	.671	.379	.378	.357	.315	.364	.419	.447
.6115	.623	.345	.267	.269	.230	.273	.311	.360
.6678	.605	.333	.233	.199	.152	.212	.234	.260
.7094	.593	.368	.215	.199	.164	.182	.192	.180
.7372	.605	.379	.198	.193	.139	.170	.180	.173
.7647	.557	.414	.273	.199	.164	.152	.180	.147
.7925	.461	.287	.198	.175	.145	.145	.180	.173
Flap								
.0582	.461	.299	.215	.175	.139	.145	.168	.133
.1306	.461	.345	.256	.199	.170	.145	.156	.140
.2758	.503	.310	.128	.117	.188	.085	.120	.100
.4177	.461	.207	.058	.070	.218	.109	.132	.127
.5600	.437	.149	.099	.117	.121	.121	.216	.227
.7019	.401	.126	.128	.164	.164	.188	.305	.180
.8527	.317	.207	.192	.222	.218	.327	.491	.560
1.0000	.449	.391	.442	.433	.370	.467	1.060	1.400

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.269	1.161	1.197	3.602	3.866	3.515	2.760	2.560
.0453	.246	.644	2.174	3.327	3.612	3.448	2.605	2.453
.0927	.419	.822	1.866	3.315	3.588	3.448	2.599	2.446
.1852	.557	.931	1.649	3.321	3.606	3.454	2.599	2.440
.2786	.665	1.017	1.587	3.228	3.618	3.454	2.625	2.440
.3706	.772	1.069	1.570	2.883	3.654	3.454	2.611	2.453
.5565	.890	1.167	1.604	2.175	3.684	3.478	2.611	2.473
.7425	1.006	1.253	1.651	1.941	3.630	3.479	2.623	2.480
.9272	1.180	1.425	1.825	2.023	3.430	3.527	2.635	2.486
Wing								
.1944	1.263	1.437	1.721	1.976	2.630	3.466	2.659	2.466
.2500	1.335	1.471	1.686	1.930	2.163	3.278	2.635	2.480
.3057	1.395	1.540	1.761	1.941	2.036	2.885	2.611	2.493
.3612	1.485	1.621	1.837	1.994	2.079	2.521	2.575	2.460
.4444	1.671	1.799	1.994	2.117	2.212	2.230	2.479	2.453
.5278	1.900	2.155	2.379	2.491	2.600	2.436	2.375	2.620
.6115	2.357	2.465	2.672	2.772	2.891	2.600	2.268	2.640
.6678	2.679	2.879	3.052	3.175	3.218	2.891	2.268	2.740
.7094	3.018	3.138	3.396	3.526	3.600	3.200	2.107	2.620
.7372	3.412	3.655	3.914	4.017	4.091	3.618	2.143	2.620
.7647	4.411	4.569	4.776	4.877	4.945	4.345	2.143	2.600
.7925	6.857	7.086	7.396	7.508	7.454	6.400	2.446	2.600
Flap								
-.0420	46.250	45.913	45.344	46.349	46.071	47.934	26.178	27.420
-.0350	46.292	46.223	46.129	46.506	46.598	41.998	22.750	23.430
-.0240	47.053	47.396	47.533	47.840	48.052	43.689	22.875	22.600
-.0130	55.375	56.085	55.913	56.067	56.307	50.889	25.750	24.500
.0000	27.071	27.241	27.361	27.507	27.544	24.781	11.518	11.200
.0582	6.321	6.500	6.707	6.684	6.672	5.709	2.268	2.640
.1306	4.286	4.465	4.500	4.631	4.636	3.873	1.964	2.560
.2758	3.018	3.190	3.310	3.316	3.327	2.836	1.982	2.500
.4177	2.429	2.500	2.500	2.596	2.600	2.200	1.964	2.480
.5600	1.821	2.052	2.086	2.035	2.000	1.873	1.839	2.460
.7019	1.464	1.517	1.483	1.579	1.509	1.491	1.750	2.200
.8527	1.196	1.121	1.362	1.333	1.364	1.254	1.625	2.100
1.0000	.607	.638	.741	.632	.636	.640	.821	1.360

TABLE II.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Concluded

(b) $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 4.75^\circ$; $q = 12.5$ lb/sq ft, $C_\mu = 0.261$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.427	1.063	.212	.000	.000	.074	.000	.042
.0927	1.403	1.066	.376	.084	.012	.040	.018	.018
.1852	1.392	.976	.494	.247	.136	.097	.108	.090
.2786	1.427	.904	.518	.301	.204	.154	.151	.145
.3706	1.392	.855	.559	.361	.241	.194	.211	.187
.5565	1.333	.819	.559	.398	.315	.240	.271	.253
.7425	1.257	.759	.553	.428	.333	.309	.313	.319
.9272	1.152	.663	.518	.416	.327	.331	.331	.313
Wing								
.1944	.982	.663	.553	.458	.383	.383	.392	.380
.2500	.842	.651	.541	.476	.376	.411	.440	.440
.3057	.754	.596	.541	.464	.407	.423	.464	.458
.3612	.678	.566	.518	.458	.383	.429	.464	.476
.4444	.608	.482	.447	.410	.383	.411	.440	.464
.5278	.555	.361	.376	.343	.327	.366	.416	.446
.6115	.468	.301	.253	.253	.247	.291	.319	.331
.6678	.485	.313	.212	.193	.179	.211	.223	.217
.7094	.526	.313	.224	.169	.136	.200	.169	.175
.7372	.503	.337	.212	.169	.160	.194	.151	.187
.7647	.515	.373	.276	.187	.167	.230	.145	.127
.7925	.409	.265	.194	.163	.142	.171	.139	.157
Flap								
.0582	.433	.289	.224	.157	.148	.160	.102	.151
.1306	.479	.337	.265	.181	.173	.160	.220	.220
.2758	.398	.289	.129	.072	.086	.149	.114	.114
.4177	.386	.193	.076	.060	.056	.103	.145	.139
.5600	.357	.138	.124	.096	.123	.166	.187	.211
.7019	.339	.108	.153	.169	.130	.200	.259	.301
.8527	.339	.181	.212	.193	.210	.249	.424	.524
1.0000	.456	.428	.488	.470	.463	.486	.922	1.151

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.380	.398	2.088	4.181	3.913	3.463	2.837	2.464
.0453	.263	.867	2.229	3.554	3.740	3.417	2.729	2.392
.0927	.444	1.012	2.041	3.584	3.734	3.394	2.705	2.410
.1852	.585	1.072	1.794	3.626	3.753	3.417	2.705	2.404
.2786	.690	1.126	1.682	3.512	3.753	3.417	2.705	2.426
.3706	.772	1.169	1.670	3.139	3.759	3.411	2.723	2.410
.5565	.930	1.271	1.688	2.343	3.814	3.440	2.729	2.428
.7425	1.017	1.361	1.741	2.054	3.827	3.463	2.741	2.434
.9272	1.204	1.542	1.918	2.120	3.703	3.508	2.735	2.446
Wing								
.1944	1.280	1.566	1.812	2.102	3.006	3.463	2.759	2.476
.2500	1.333	1.578	1.800	2.024	2.395	3.251	2.765	2.464
.3057	1.403	1.639	1.847	2.066	2.148	2.914	2.741	2.446
.3612	1.497	1.747	1.900	2.096	2.123	2.548	2.681	2.404
.4444	1.672	1.904	2.070	2.229	2.259	2.217	2.584	2.428
.5278	2.298	2.400	2.438	2.673	2.773	2.414	2.600	2.618
.6115	2.596	2.764	2.772	3.054	3.093	2.655	2.473	2.600
.6678	2.947	3.254	3.193	3.416	3.389	2.965	2.400	2.636
.7094	3.368	3.618	3.596	3.782	3.870	3.259	2.418	2.545
.7372	3.842	4.054	4.088	4.382	4.370	3.655	2.418	2.582
.7647	4.737	5.145	5.087	5.382	5.370	4.433	2.491	2.636
.7925	7.315	7.909	7.736	8.145	8.037	6.655	2.927	2.691
Flap								
-.0420	55.559	58.379	59.927	57.579	57.906	51.930	47.471	47.525
-.0350	45.524	47.207	45.542	47.243	47.610	45.085	30.653	28.344
-.0240	53.366	57.361	56.155	57.207	56.554	48.671	29.580	26.053
-.0130	64.979	71.397	69.230	71.160	71.183	59.861	34.508	29.181
.0000	30.420	32.417	31.595	32.362	32.166	27.206	14.545	12.545
.0582	6.614	7.109	6.982	7.254	7.129	5.798	2.673	2.727
.1306	4.631	4.891	4.737	5.018	4.981	4.034	2.339	2.509
.2758	3.316	3.582	3.421	3.600	3.649	2.983	2.273	2.600
.4177	2.737	2.818	2.667	2.854	2.889	2.483	2.327	2.582
.5600	2.105	2.291	2.193	2.145	2.278	1.948	2.109	2.418
.7019	1.667	1.727	1.579	1.691	1.722	1.569	2.054	2.364
.8527	1.526	1.473	1.491	1.400	1.574	1.448	1.909	2.182
1.0000	.860	.855	.825	1.054	.778	.759	1.073	1.327

TABLE III.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE

(a) $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $i_N = 10^\circ$; $q = 12.5$ lb/sq ft; $C_\mu = 0$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.585	1.636	1.434	1.537	.515	.054	.012	.000
.0927	1.595	1.636	1.434	1.371	.626	.205	.048	.018
.1852	1.595	1.624	1.440	1.143	.707	.355	.193	.121
.2786	1.590	1.636	1.434	1.040	.707	.392	.259	.176
.3706	1.606	1.648	1.458	.971	.707	.446	.301	.230
.5565	1.623	1.660	1.454	.891	.684	.470	.367	.303
.7425	1.623	1.654	1.458	.789	.620	.458	.367	.315
.9272	1.645	1.660	1.445	.651	.652	.422	.319	.315
Wing								
.1944	1.634	1.709	1.411	.737	.626	.512	.428	.394
.2500	1.639	1.721	1.286	.766	.661	.554	.476	.436
.3057	1.683	1.745	1.184	.793	.684	.584	.518	.473
.3612	1.705	1.763	1.095	.743	.661	.590	.542	.521
.4444	1.721	1.733	.994	.691	.631	.566	.536	.509
.5278	1.683	1.673	.911	.549	.485	.530	.500	.473
.6115	1.634	1.491	.881	.429	.403	.380	.380	.358
.6678	1.574	1.357	.815	.389	.351	.277	.277	.273
.7094	1.508	1.248	.774	.377	.322	.217	.217	.206
.7372	1.470	1.139	.720	.377	.333	.205	.205	.206
.7647	1.399	1.030	.708	.400	.316	.205	.181	.176
.7925	1.300	.933	.679	.411	.327	.205	.187	.133
Flap								
.0582	1.262	.976	.726	.446	.345	.169	.175	.170
.1306	1.279	1.012	.780	.417	.391	.193	.169	.139
.2758	1.388	.945	.619	.377	.310	.145	.151	.121
.4177	1.333	.867	.613	.309	.257	.181	.217	.176
.5600	1.377	.836	.619	.331	.287	.241	.289	.297
.7019	1.344	.770	.661	.434	.403	.416	.434	.412
.8527	1.300	1.242	1.012	.686	.696	.645	.626	.654
1.0000	1.535	1.594	1.708	1.794	1.801	1.741	1.759	1.866

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.645	1.745	1.577	1.886	.187	3.187	3.446	3.254
.0453	.011	.030	.125	.503	1.649	3.277	3.036	2.888
.0927	.076	.139	.310	.686	1.555	3.253	3.042	2.921
.1852	.169	.242	.488	.834	1.485	3.108	3.054	2.945
.2786	.268	.345	.595	.949	1.444	2.524	3.042	2.945
.3706	.344	.455	.684	1.017	1.456	2.024	3.066	2.933
.5565	.448	.606	.845	1.154	1.514	1.813	3.102	2.788
.7425	.579	.727	.964	1.257	1.602	1.898	3.078	2.648
.9272	.770	.957	1.220	1.520	1.889	2.108	2.940	2.533
Wing								
.1944	.809	.994	1.149	1.360	1.602	1.879	2.494	2.327
.2500	.863	1.000	1.119	1.337	1.526	1.741	2.187	2.188
.3057	.912	1.024	1.155	1.326	1.503	1.663	1.964	2.079
.3612	.951	1.079	1.167	1.343	1.473	1.633	1.825	2.000
.4444	1.038	1.151	1.232	1.406	1.509	1.602	1.717	1.927
.5278	1.213	1.364	1.571	1.569	1.667	1.782	1.891	1.909
.6115	1.246	1.236	1.571	1.603	1.667	1.782	1.879	1.909
.6678	1.229	1.200	1.536	1.517	1.596	1.673	1.727	1.782
.7094	1.344	1.418	1.643	1.586	1.737	1.764	1.891	1.909
.7372	1.377	1.509	1.661	1.638	1.702	1.745	1.927	1.894
.7647	1.459	1.564	1.714	1.724	1.631	1.800	1.927	1.927
.7925	1.492	1.654	1.732	1.810	1.842	1.854	1.964	1.927
Flap								
-.0420	1.607	1.727	1.911	2.000	1.930	1.964	2.073	2.036
-.0350	1.590	1.709	2.000	1.931	1.965	1.909	1.982	2.054
-.0240	1.557	1.673	1.875	1.828	1.860	1.836	1.927	2.036
-.0130	1.541	1.709	1.786	1.828	1.807	1.836	1.891	2.036
.0000	1.574	1.709	1.786	1.828	1.789	1.873	1.891	2.018
.0582	1.672	1.800	1.875	1.897	1.842	1.945	1.964	2.054
.1306	1.656	1.764	1.857	1.931	1.842	1.891	2.018	2.054
.2758	1.688	1.782	1.804	1.879	2.017	1.818	2.109	2.000
.4177	1.705	1.764	1.893	1.948	1.930	1.964	2.036	2.000
.5600	1.738	1.654	1.839	1.776	1.877	1.709	1.891	2.073
.7019	1.705	1.764	1.857	1.948	1.912	1.927	2.073	2.073
.8527	1.656	1.564	1.750	1.879	1.842	1.709	1.909	1.964
1.0000	1.738	1.691	1.625	1.724	1.631	1.618	1.709	1.818

TABLE III.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(b) $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 10^\circ$; $q \approx 12.5$ lb/sq ft; $C_\mu = 0.012$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.547	1.461	1.418	1.196	.321	.000	.000	.000
.0927	1.536	1.443	1.405	1.159	.478	.117	.073	.048
.1852	1.553	1.467	1.424	1.097	.584	.265	.145	.169
.2786	1.558	1.473	1.430	.969	.615	.321	.261	.217
.3706	1.558	1.467	1.449	.920	.621	.376	.285	.259
.5565	1.581	1.473	1.443	.840	.615	.420	.327	.325
.7425	1.575	1.479	1.424	.754	.590	.407	.364	.331
.9272	1.592	1.479	1.399	.644	.578	.401	.339	.313
Wing								
.1944	1.609	1.527	1.272	.730	.584	.475	.424	.392
.2500	1.631	1.581	1.095	.730	.640	.525	.448	.470
.3057	1.659	1.593	1.000	.748	.677	.568	.509	.518
.3612	1.659	1.599	.905	.742	.596	.549	.515	.506
.4444	1.681	1.527	.835	.669	.578	.555	.509	.518
.5278	1.648	1.467	.753	.509	.516	.500	.479	.482
.6115	1.581	1.299	.703	.368	.366	.376	.327	.422
.6678	1.503	1.222	.620	.350	.304	.265	.279	.289
.7094	1.475	1.144	.595	.362	.329	.210	.200	.253
.7372	1.408	1.094	.576	.356	.286	.191	.188	.217
.7647	1.324	.994	.576	.344	.317	.173	.133	.151
.7925	1.246	.910	.525	.350	.292	.198	.139	.169
Flap								
.0582	1.223	.964	.576	.386	.279	.185	.133	.175
.1304	1.268	1.000	.639	.429	.286	.222	.133	.151
.2758	1.285	.916	.519	.313	.279	.123	.127	.133
.4177	1.223	.856	.481	.288	.217	.173	.182	.187
.5600	1.268	.820	.494	.258	.267	.253	.255	.343
.7019	1.229	.808	.563	.393	.354	.401	.370	.446
.8527	1.357	1.174	.728	.662	.634	.611	.558	.626
1.0000	1.486	1.467	1.563	1.638	1.708	1.703	1.394	1.747

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.581	1.539	1.449	1.221	1.366	4.271	3.515	2.645
.0453	.017	.024	.285	.767	2.366	3.407	3.139	2.717
.0927	.061	.144	.443	.939	2.025	3.456	3.121	2.741
.1852	.162	.293	.620	1.049	1.783	3.543	3.139	2.843
.2786	.251	.377	.746	1.098	1.720	3.481	3.127	2.807
.3706	.330	.485	.854	1.172	1.708	3.222	3.163	2.590
.5565	.452	.653	1.000	1.300	1.770	2.481	3.194	2.193
.7425	.614	.784	1.139	1.423	1.832	2.197	3.188	2.012
.9272	.804	1.000	1.449	1.711	2.149	2.277	3.109	2.036
Wing								
.1944	.855	1.024	1.367	1.540	1.863	2.098	2.769	2.060
.2500	.905	1.042	1.316	1.466	1.752	1.950	2.545	2.048
.3057	.961	1.066	1.373	1.472	1.714	1.895	2.273	2.048
.3612	1.011	1.138	1.399	1.515	1.714	1.864	2.054	2.042
.4444	1.112	1.218	1.487	1.552	1.765	1.876	1.945	2.024
.5278	1.250	1.536	1.641	1.673	2.000	2.037	1.945	2.200
.6115	1.350	1.643	1.811	1.800	2.055	2.185	2.018	2.182
.6678	1.400	1.679	1.924	1.927	2.130	2.315	2.073	2.091
.7094	1.533	1.821	2.151	2.000	2.241	2.296	2.164	2.091
.7372	1.633	1.929	2.283	2.109	2.352	2.500	2.418	2.018
.7647	1.750	2.089	2.509	2.418	2.611	2.815	2.564	2.091
.7925	2.217	2.589	3.207	3.227	3.333	3.648	3.400	2.054
Flap								
-.0420	5.266	5.857	7.830	7.709	7.852	8.555	8.181	3.509
-.0350	5.616	6.589	9.037	8.781	9.055	9.907	9.581	3.473
-.0240	5.133	5.911	8.773	8.254	8.426	9.759	9.818	3.454
-.0130	3.117	3.964	7.302	6.836	6.963	8.759	9.545	3.582
.0000	1.850	2.357	4.434	3.927	4.278	5.629	6.472	2.782
.0582	1.733	2.000	2.792	2.654	2.759	3.018	2.909	2.273
.1306	1.717	1.839	2.490	2.382	2.444	2.407	2.309	2.218
.2758	1.483	1.946	2.189	1.836	2.259	2.241	2.054	2.200
.4177	1.583	1.786	1.952	1.854	1.981	2.130	1.727	2.091
.5600	1.500	1.732	1.924	1.764	1.889	2.000	1.636	1.964
.7019	1.550	1.607	1.830	1.782	1.981	1.926	1.545	1.909
.8527	1.483	1.536	1.887	1.854	1.907	1.926	1.654	1.909
1.0000	1.633	1.607	1.585	1.454	1.611	1.518	1.291	1.891

TABLE III.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(c) $c_f = 0.20c_w$; $b_f = 60°$; $\delta_N = 10°$; $q = 12.5$ lb/sq ft; $C_\mu = 0.022$

Lower surface

x/c	C_p for -							
	$\alpha = -16°$	$\alpha = -12°$	$\alpha = -8°$	$\alpha = -4°$	$\alpha = 0°$	$\alpha = 4°$	$\alpha = 8°$	$\alpha = 12°$
Nose								
.0453	1.114	1.227	1.407	.395	.034	.018	.012	.000
.0927	1.114	1.227	1.290	.521	.155	.055	.050	.038
.1852	1.114	1.227	1.129	.599	.264	.153	.155	.126
.2786	1.126	1.245	1.018	.623	.333	.239	.211	.195
.3706	1.120	1.245	.920	.623	.379	.276	.273	.252
.5565	1.150	1.257	.846	.593	.414	.337	.335	.314
.7425	1.138	1.276	.747	.551	.402	.325	.348	.333
.9272	1.150	1.264	.630	.533	.379	.301	.298	.308
Wing								
.1944	1.162	1.270	.716	.557	.437	.380	.398	.394
.2500	1.174	1.190	.728	.581	.471	.405	.460	.434
.3057	1.222	1.096	.716	.581	.483	.423	.484	.497
.3612	1.186	1.012	.667	.557	.477	.405	.491	.509
.4444	1.234	.932	.580	.479	.454	.448	.494	.509
.5278	1.263	.816	.469	.359	.362	.411	.447	.478
.6115	1.293	.816	.333	.311	.270	.276	.348	.371
.6678	1.293	.730	.333	.269	.190	.154	.279	.283
.7094	1.257	.724	.358	.287	.201	.159	.250	.252
.7372	1.186	.705	.352	.293	.184	.129	.168	.201
.7647	1.084	.693	.358	.293	.195	.147	.149	.132
.7925	.982	.552	.352	.299	.213	.141	.168	.157
Flap								
.0582	1.000	.656	.420	.323	.178	.141	.143	.245
.1306	1.018	.736	.401	.305	.178	.153	.124	.151
.2758	1.096	.528	.327	.240	.080	.080	.137	.126
.4177	1.132	.528	.259	.144	.109	.098	.174	.195
.5600	1.144	.497	.160	.096	.155	.129	.298	.283
.7019	1.090	.534	.235	.198	.224	.251	.385	.447
.8527	1.012	.745	.420	.347	.351	.331	.565	.654
1.0000	1.120	.644	.654	.633	.667	.730	1.267	1.792

Upper surface

x/c	C_p for -							
	$\alpha = -16°$	$\alpha = -12°$	$\alpha = -8°$	$\alpha = -4°$	$\alpha = 0°$	$\alpha = 4°$	$\alpha = 8°$	$\alpha = 12°$
Nose								
.0000	1.246	1.288	1.370	.515	3.810	4.006	3.428	3.352
.0453	.042	.172	.593	1.964	3.552	3.662	3.168	3.123
.0927	.192	.395	.815	1.796	3.488	3.674	3.174	3.151
.1852	.359	.601	.988	1.653	2.948	3.711	3.168	3.207
.2786	.479	.693	1.049	1.629	2.287	3.717	3.192	3.182
.3706	.593	.810	1.148	1.617	2.017	3.729	3.205	3.132
.5565	.743	.959	1.315	1.677	1.914	3.662	3.205	2.874
.7425	.880	1.129	1.444	1.805	2.017	3.312	3.236	2.660
.9272	1.150	1.405	1.778	2.114	2.264	2.730	3.205	2.509
Wing								
.1944	1.186	1.363	1.611	1.868	2.023	2.251	3.000	2.346
.2500	1.216	1.262	1.586	1.778	1.885	2.092	2.764	2.195
.3057	1.293	1.417	1.636	1.820	1.874	2.061	2.528	2.069
.3612	1.359	1.503	1.703	1.826	1.869	2.030	2.310	1.987
.4444	1.515	1.626	1.839	1.934	1.948	2.079	2.093	1.937
.5278	1.768	1.782	2.167	2.107	2.276	2.345	2.333	2.237
.6115	2.071	1.982	2.993	2.232	2.379	2.527	2.296	2.189
.6678	2.214	2.273	2.833	2.607	2.828	2.636	2.500	2.264
.7094	2.393	2.527	3.204	2.750	3.034	2.909	2.389	2.132
.7372	2.893	2.891	3.407	3.248	3.276	3.163	2.518	2.113
.7647	3.643	3.582	4.241	4.000	3.810	3.763	2.759	2.189
.7925	5.375	5.400	6.222	5.821	5.638	5.271	3.296	2.283
Flap								
-.0420	16.500	16.581	18.111	17.357	16.034	15.236	8.407	5.332
-.0350	20.178	20.054	22.018	21.107	19.275	18.145	9.296	5.283
-.0240	21.411	21.217	23.222	22.303	20.413	19.054	9.518	5.188
-.0130	21.607	21.344	23.296	22.321	20.465	19.199	9.778	5.377
.0000	14.268	14.336	15.407	14.714	13.526	12.959	6.537	3.698
.0582	5.375	5.327	6.074	5.714	5.448	4.873	2.981	2.388
.1306	3.661	3.636	4.092	3.893	3.793	3.363	2.481	2.189
.2758	2.607	2.436	2.944	2.643	2.569	2.345	2.074	2.226
.4177	2.089	1.782	2.148	1.964	1.897	1.745	1.963	2.189
.5600	1.786	1.527	1.796	1.571	1.621	1.545	1.870	2.019
.7019	1.571	1.164	1.530	1.214	1.310	1.309	1.741	2.075
.8527	1.161	.891	1.130	.929	1.138	1.036	1.593	2.113
1.0000	1.107	.655	.926	.571	.845	.784	1.463	1.736

TABLE III.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(d) $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 10^\circ$; $q \approx 12.5$ lb/sq ft; $C_\mu = 0.038$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.094	1.259	1.107	.261	.017	.012	.018	.006
.0927	1.082	1.259	1.083	.438	.128	.018	.006	.018
.1852	1.082	1.271	.964	.549	.273	.104	.139	.108
.2786	1.094	1.247	.875	.555	.326	.159	.205	.175
.3706	1.094	1.271	.839	.575	.360	.233	.241	.229
.5565	1.100	1.290	.756	.562	.407	.294	.283	.289
.7425	1.106	1.284	.667	.529	.395	.307	.319	.325
.9272	1.118	1.259	.565	.490	.360	.282	.283	.301
Wing								
.1944	1.135	1.185	.619	.516	.424	.356	.337	.386
.2500	1.176	1.025	.619	.555	.459	.386	.416	.446
.3057	1.194	.901	.619	.549	.477	.436	.452	.470
.3612	1.212	.790	.589	.529	.471	.442	.458	.482
.4444	1.265	.734	.512	.471	.424	.442	.440	.482
.5278	1.271	.648	.399	.340	.355	.417	.404	.446
.6115	1.265	.574	.310	.222	.256	.270	.343	.380
.6678	1.188	.537	.280	.190	.203	.209	.229	.283
.7094	1.153	.537	.310	.229	.151	.172	.187	.229
.7372	1.082	.518	.333	.209	.163	.135	.151	.187
.7647	.971	.512	.357	.268	.174	.153	.145	.193
.7925	.906	.438	.304	.235	.186	.172	.133	.197
Flap								
.0582	.994	.500	.315	.261	.198	.166	.151	.151
.1306	.912	.475	.327	.242	.198	.166	.181	.133
.2758	.965	.401	.250	.157	.076	.061	.127	.127
.4177	1.000	.364	.119	.033	.099	.098	.133	.199
.5600	1.012	.302	.131	.072	.151	.159	.211	.289
.7019	.965	.321	.131	.157	.198	.209	.307	.422
.8527	.847	.506	.286	.255	.296	.331	.482	.626
1.0000	.788	.475	.524	.542	.552	.632	1.145	1.717

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.176	1.278	.619	1.307	3.848	4.061	3.271	3.289
.0453	.041	.235	.750	2.451	3.459	3.656	2.994	3.096
.0927	.188	.420	.905	2.065	3.488	3.644	2.994	3.126
.1852	.335	.611	1.000	1.830	3.435	3.587	3.018	3.163
.2786	.447	.728	1.071	1.764	3.145	3.680	3.024	3.163
.3706	.547	.796	1.149	1.745	2.691	3.711	3.042	3.114
.5565	.718	.975	1.286	1.823	2.093	3.734	3.066	2.946
.7425	.871	1.129	1.446	1.921	2.064	3.558	3.090	2.723
.9272	1.100	1.420	1.756	2.313	2.314	3.128	3.084	2.584
Wing								
.1944	1.141	1.383	1.613	2.013	2.139	2.484	2.928	2.343
.2500	1.176	1.395	1.583	1.895	2.023	2.233	2.777	2.193
.3057	1.253	1.469	1.631	1.915	1.982	2.116	2.536	2.072
.3612	1.329	1.518	1.684	1.928	2.011	2.110	2.307	2.012
.4444	1.471	1.666	1.798	2.072	2.081	2.159	2.084	1.994
.5278	1.895	2.111	2.214	2.588	2.172	2.364	2.127	2.073
.6115	2.140	2.370	2.518	2.980	2.445	2.509	2.127	2.036
.6678	2.561	2.593	2.821	3.274	2.621	2.650	2.127	2.164
.7094	2.667	2.907	3.036	3.392	2.948	2.982	2.164	2.109
.7372	3.052	3.333	3.429	3.941	3.310	3.345	2.273	2.091
.7647	3.719	3.926	4.196	4.725	3.965	3.873	2.545	2.054
.7925	5.649	6.037	6.232	7.000	5.931	5.527	3.127	2.218
Flap								
-.0420	18.631	20.074	20.107	22.234	18.689	17.672	10.236	6.800
-.0350	22.315	23.870	24.053	26.509	22.189	20.635	10.963	6.618
-.0240	23.701	25.222	25.357	27.920	23.379	21.744	11.263	6.327
-.0130	24.332	25.888	25.857	28.665	23.913	22.290	11.909	6.582
.0000	15.596	16.555	16.696	18.313	15.276	14.181	7.181	4.182
.0582	5.509	5.926	6.071	6.921	5.552	5.163	2.891	2.182
.1306	3.912	4.185	4.179	4.686	3.879	3.545	2.273	2.182
.2758	2.596	2.815	2.946	3.157	2.552	2.473	1.945	2.054
.4177	2.000	2.222	2.196	2.470	2.034	1.891	1.727	2.000
.5600	1.561	1.704	1.589	1.843	1.517	1.564	1.600	1.945
.7019	1.439	1.444	1.393	1.627	1.276	1.309	1.618	1.945
.8527	1.175	1.148	1.107	1.294	1.000	1.073	1.400	1.909
1.0000	.772	.647	.607	.686	.569	.764	1.218	1.654

TABLE III - PRESSURE COEFFICIENT C_p ON THE NOSE WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(e) $c_f = 0.200c_w$; $\delta_f = 60^\circ$, $\delta_N = 10^\circ$; $q \approx 12.5$ lb/sq ft; $C_\mu = 0.060$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.109	1.484	1.025	.234	.025	.012	.013	.013
.0927	1.103	1.484	1.019	.370	.099	.037	.038	.071
.1852	1.103	1.471	.943	.506	.222	.154	.138	.167
.2786	1.103	1.439	.868	.545	.290	.210	.189	.230
.3706	1.115	1.414	.824	.558	.321	.265	.252	.295
.5565	1.115	1.293	.761	.545	.370	.321	.283	.340
.7425	1.127	1.172	.667	.519	.383	.346	.314	.359
.9272	1.127	1.019	.535	.461	.370	.346	.289	.340
Wing								
.1944	1.164	.904	.629	.526	.426	.395	.371	.417
.2500	1.182	.841	.635	.545	.444	.407	.415	.481
.3057	1.212	.783	.629	.506	.457	.451	.421	.506
.3612	1.230	.701	.591	.500	.469	.451	.439	.545
.4444	1.279	.643	.522	.455	.451	.426	.421	.538
.5278	1.267	.560	.453	.390	.346	.370	.358	.500
.6115	1.230	.510	.283	.305	.272	.284	.296	.353
.6678	1.164	.490	.252	.234	.198	.204	.195	.301
.7094	1.115	.471	.277	.221	.179	.127	.164	.218
.7372	1.030	.433	.308	.240	.191	.148	.157	.205
.7647	.897	.446	.340	.318	.185	.148	.119	.167
.7925	.812	.433	.340	.234	.173	.160	.157	.199
Flap								
.0582	.836	.490	.377	.266	.160	.198	.170	.179
.1306	.861	.548	.365	.299	.210	.160	.138	.167
.2798	.927	.357	.201	.188	.086	.105	.075	.109
.4177	.945	.350	.138	.078	.086	.123	.126	.205
.5600	.964	.344	.107	.071	.148	.136	.220	.282
.7019	.945	.331	.138	.130	.173	.210	.302	.404
.8527	.800	.503	.245	.214	.235	.296	.463	.596
1.0000	.715	.516	.453	.429	.457	.518	1.132	1.756

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.188	1.369	.717	1.896	4.049	4.074	3.358	2.526
.0453	.073	.229	.906	2.370	3.561	3.666	3.050	2.340
.0927	.212	.433	1.019	2.156	3.574	3.697	3.088	2.404
.1852	.376	.650	1.113	1.935	3.604	3.703	3.113	2.442
.2786	.479	.764	1.145	1.857	3.574	3.728	3.107	2.385
.3706	.564	.879	1.226	1.857	3.284	3.740	3.132	2.333
.5565	.745	1.038	1.358	1.889	2.456	3.765	3.163	2.218
.7425	.897	1.210	1.547	2.026	2.173	3.679	3.176	2.173
.9272	1.194	1.541	1.874	2.376	2.395	3.358	3.176	2.192
Wing								
.1944	1.200	1.465	1.717	2.078	2.222	2.710	3.069	2.199
.2500	1.236	1.478	1.685	2.000	2.117	2.352	2.943	2.218
.3057	1.309	1.535	1.755	1.987	2.000	2.210	2.729	2.218
.3612	1.382	1.635	1.792	2.019	2.098	2.148	2.497	2.199
.4444	1.539	1.771	1.912	2.156	2.173	2.210	2.270	2.192
.5278	2.018	2.298	2.434	2.654	2.537	2.463	2.321	2.346
.6115	2.236	2.692	2.679	2.865	2.722	2.704	2.264	2.269
.6678	2.600	3.096	2.868	3.288	2.981	3.055	2.302	2.327
.7094	2.854	3.308	3.321	3.481	3.407	3.241	2.340	2.327
.7372	3.327	3.846	3.773	3.904	3.741	3.648	2.472	2.346
.7647	3.945	4.519	4.415	4.692	4.352	4.278	2.755	2.192
.7925	6.127	6.596	6.830	6.934	6.592	6.148	3.396	2.234
Flap								
-.0420	23.381	24.884	25.206	25.672	23.777	22.999	12.830	8.027
-.0350	26.872	28.999	28.357	30.018	27.610	24.981	13.433	8.077
-.0240	28.453	30.480	31.055	31.768	29.055	26.481	13.811	7.634
-.0130	29.671	31.806	32.319	32.864	30.221	27.388	14.509	7.538
.0000	18.199	19.557	19.848	20.345	18.499	16.740	8.377	4.461
.0582	5.982	6.538	6.679	6.894	6.259	5.592	2.943	2.327
.1306	4.109	4.425	4.604	4.731	4.278	4.111	2.321	2.250
.2798	2.927	3.192	3.170	3.231	3.018	2.741	2.019	2.269
.4177	2.254	2.404	2.528	2.519	2.315	2.296	1.962	2.250
.5600	1.818	1.885	1.906	1.965	1.667	1.870	1.587	2.135
.7019	1.654	1.615	1.623	1.711	1.500	1.593	1.717	2.154
.8527	1.218	1.327	1.415	1.288	1.185	1.315	1.660	1.865
1.0000	.909	.615	.698	.558	.611	.833	1.264	1.711

TABLE III.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(c) $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 10^\circ$, q = 12.5 lb/sq ft, $C_\mu = 0.118$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.072	1.772	.737	.113	.018	.012	.024	.019
.0927	1.090	1.678	.811	.250	.077	.047	.018	.045
.1852	1.096	1.532	.766	.387	.202	.116	.121	.159
.2786	1.084	1.368	.743	.440	.262	.198	.170	.236
.3706	1.084	1.216	.703	.458	.315	.233	.242	.274
.5565	1.102	1.000	.629	.458	.363	.279	.279	.318
.7425	1.108	.883	.617	.446	.352	.291	.303	.350
.9272	1.133	.783	.463	.421	.327	.256	.303	.325
Wing								
.1944	1.151	.707	.509	.446	.399	.326	.351	.395
.2500	1.169	.684	.560	.488	.423	.372	.388	.465
.3057	1.139	.661	.537	.482	.429	.401	.436	.490
.3612	1.096	.608	.531	.476	.423	.389	.443	.510
.4444	1.120	.520	.440	.429	.387	.395	.404	.510
.5278	1.108	.427	.343	.345	.310	.320	.351	.427
.6115	1.078	.327	.263	.250	.256	.244	.291	.363
.6678	.994	.339	.251	.209	.173	.151	.230	.306
.7094	.958	.363	.240	.209	.185	.140	.170	.197
.7372	.904	.403	.263	.223	.179	.140	.149	.185
.7647	.807	.403	.314	.268	.190	.116	.127	.146
.7925	.681	.339	.253	.214	.143	.145	.145	.172
Flap								
.0582	.699	.368	.263	.256	.179	.122	.133	.134
.1306	.711	.409	.297	.244	.202	.140	.133	.134
.2758	.819	.351	.240	.155	.089	.070	.097	.121
.4177	.819	.287	.131	.054	.054	.087	.133	.185
.5600	.807	.263	.103	.125	.113	.122	.158	.287
.7019	.777	.228	.143	.185	.155	.140	.255	.452
.8527	.699	.287	.229	.208	.214	.273	.442	.605
1.0000	.614	.374	.400	.458	.405	.419	.897	1.516

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.145	1.696	.091	3.512	4.035	3.852	3.248	2.656
.0453	.072	.281	1.137	2.649	3.583	3.598	3.046	2.407
.0927	.229	.474	1.189	2.315	3.607	3.604	3.040	2.427
.1852	.392	.672	1.234	2.030	3.643	3.610	3.078	2.420
.2786	.530	.801	1.280	1.922	3.649	3.621	3.078	2.433
.3706	.645	.930	1.331	1.887	3.607	3.639	3.103	2.401
.5565	.831	1.070	1.451	1.934	3.143	3.651	3.097	2.414
.7425	.946	1.251	1.577	2.006	2.547	3.621	3.121	2.376
.9272	1.259	1.567	1.903	2.375	2.387	3.470	3.139	2.395
Wing								
.1944	1.289	1.491	1.726	2.083	2.292	2.860	3.066	2.414
.2500	1.301	1.491	1.680	1.982	2.184	2.447	2.975	2.407
.3057	1.373	1.555	1.714	1.994	2.161	2.197	2.836	2.388
.3612	1.458	1.649	1.777	2.024	2.190	2.116	2.642	2.388
.4444	1.626	1.795	1.931	2.149	2.268	2.151	2.376	2.363
.5278	2.036	2.210	2.190	2.429	2.411	2.465	2.254	2.519
.6115	2.327	2.421	2.500	2.768	2.679	2.759	2.327	2.481
.6678	2.673	2.842	2.862	3.321	3.107	2.931	2.382	2.461
.7094	3.054	3.123	3.138	3.339	3.250	3.155	2.382	2.461
.7372	3.563	3.719	3.556	3.786	3.786	3.638	2.509	2.461
.7647	4.218	4.368	4.500	4.679	4.589	4.345	2.691	2.385
.7925	6.636	6.842	6.914	7.261	7.036	6.362	3.473	2.500
Flap								
-.0420	32.980	33.051	32.499	33.196	32.553	29.999	20.926	16.634
-.0350	35.417	35.981	35.499	36.375	35.214	31.965	20.163	14.596
-.0240	37.071	37.788	37.241	38.321	37.196	33.706	20.708	14.153
-.0130	40.289	41.016	40.361	41.410	40.214	36.551	22.363	14.423
.0000	22.417	22.806	22.724	23.268	22.625	20.241	11.509	7.461
.0582	6.254	6.386	6.396	6.696	6.375	5.724	3.073	2.500
.1306	4.254	4.403	4.396	4.696	4.482	4.103	2.418	2.346
.2758	3.145	3.123	3.069	3.304	3.089	2.828	2.254	2.385
.4177	2.454	2.368	2.379	2.500	2.375	2.259	1.873	2.346
.5600	1.909	1.860	1.810	2.054	1.857	1.741	1.854	2.139
.7019	1.691	1.544	1.655	1.625	1.536	1.431	1.891	2.115
.8527	1.309	1.210	1.328	1.357	1.054	1.241	1.654	2.096
1.0000	.855	.596	.690	.696	.571	.724	.982	1.577

TABLE II.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(g) $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\alpha_N = 10^\circ$; $q = 12.5$ lb/sq ft; $C_\mu = 0.128$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	.983	1.757	.598	.086	.013	.043	.036	
.0927	.978	1.463	.678	.233	.038	.025	.018	
.1852	.967	1.181	.695	.368	.176	.112	.126	
.2786	.978	1.039	.678	.423	.239	.174	.168	
.3706	.989	.949	.672	.448	.277	.211	.216	
.5565	.994	.825	.615	.466	.314	.261	.269	
.7425	1.022	.717	.557	.466	.346	.286	.299	
.9272	1.022	.621	.483	.423	.314	.279	.281	
Wing								
.1944	1.000	.604	.523	.485	.384	.348	.347	
.2500	1.003	.599	.534	.539	.428	.348	.419	
.3057	1.011	.597	.534	.491	.443	.404	.431	
.3612	.978	.542	.506	.442	.440	.416	.437	
.4444	.978	.475	.443	.393	.390	.348	.431	
.5278	.955	.345	.333	.350	.302	.317	.359	
.6115	.911	.305	.261	.245	.252	.224	.269	
.6678	.889	.305	.236	.221	.164	.149	.192	
.7094	.822	.328	.253	.184	.126	.137	.162	
.7372	.778	.362	.270	.184	.132	.124	.168	
.7647	.722	.395	.305	.251	.151	.143	.184	
.7925	.644	.437	.351	.319	.252	.224	.228	
Flap								
.0582	.667	.384	.316	.282	.226	.199	.192	
.1306	.667	.339	.282	.215	.182	.143	.132	
.2758	.700	.288	.230	.135	.082	.062	.090	
.4177	.722	.237	.144	.061	.094	.087	.108	
.5600	.689	.130	.086	.092	.113	.130	.174	
.7019	.683	.153	.109	.147	.164	.124	.246	
.8527	.600	.203	.178	.196	.189	.236	.377	
1.0000	.556	.384	.408	.417	.453	.404	.796	

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.055	2.712	.155	5.447	4.748	4.050	3.186	
.0453	.144	.322	1.362	3.196	3.729	3.913	3.066	
.0927	.278	.525	1.345	2.656	3.786	3.913	3.078	
.1852	.439	.729	1.339	2.251	3.887	3.913	3.084	
.2786	.561	.830	1.362	2.098	3.962	3.919	3.084	
.3706	.655	.915	1.402	2.055	3.905	3.938	3.126	
.5565	.828	1.079	1.526	2.067	3.371	3.969	3.114	
.7425	.944	1.237	1.635	2.147	2.786	3.987	3.126	
.9272	1.228	1.570	1.971	2.490	2.578	3.925	3.156	
Wing								
.1944	1.255	1.508	1.741	2.196	2.453	3.447	3.108	
.2500	1.255	1.514	1.707	2.110	2.333	2.938	3.030	
.3057	1.350	1.582	1.730	2.092	2.314	2.559	2.958	
.3612	1.411	1.655	1.799	2.116	2.333	2.373	2.778	
.4444	1.594	1.813	1.960	2.239	2.440	2.366	2.503	
.5278	1.900	2.233	2.483	2.654	2.566	2.722	2.500	
.6115	2.233	2.610	2.759	2.945	2.906	2.889	2.393	
.6678	2.617	2.915	3.069	3.382	3.377	3.074	2.464	
.7094	2.967	3.186	3.414	3.691	3.641	3.481	2.536	
.7372	3.317	3.678	3.965	4.254	4.179	4.037	2.625	
.7647	4.183	4.661	4.810	5.163	5.056	4.833	2.929	
.7925	6.483	7.169	7.362	7.927	7.986	7.389	3.732	
Flap								
-.0420	44.598	45.050	45.706	48.071	48.413	45.851	30.410	
-.0350	42.948	45.695	46.533	49.416	48.696	45.628	27.410	
-.0240	42.082	46.644	47.585	50.543	49.998	46.999	27.928	
-.0130	49.565	54.118	55.326	58.689	57.620	54.036	31.607	
.0000	24.299	26.983	27.413	29.162	28.640	26.981	15.036	
.0582	5.650	6.610	6.621	7.091	7.300	6.500	3.143	
.1306	4.083	4.627	4.586	4.982	4.924	4.592	2.607	
.2758	2.933	3.220	3.207	3.473	3.358	3.296	2.304	
.4177	2.283	2.441	2.603	2.745	2.623	2.510	2.161	
.5600	1.900	2.034	2.000	2.145	2.094	1.981	1.982	
.7019	1.367	1.678	1.672	1.745	1.811	1.574	1.857	
.8527	1.300	1.339	1.345	1.400	1.358	1.389	1.804	
1.0000	.850	.644	.690	.673	.679	.593	1.089	

TABLE III.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Concluded

(h) $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 10^\circ$; $q \approx 12.5$ lb/sq ft; $C_\mu = 0.345$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.029	1.517	.494	.048	.012	.036	.057	
.0927	1.023	1.343	.626	.175	.042	.006	.032	
.1852	1.023	1.093	.665	.313	.138	.071	.108	
.2786	1.035	.992	.653	.373	.216	.167	.152	
.3706	1.052	.889	.635	.404	.251	.190	.190	
.5565	1.052	.773	.630	.429	.305	.250	.266	
.7425	1.052	.709	.553	.434	.323	.262	.266	
.9272	1.081	.581	.489	.392	.311	.238	.253	
Wing								
.1944	1.087	.651	.529	.440	.347	.315	.316	
.2500	1.075	.645	.559	.464	.395	.369	.367	
.3057	1.035	.605	.541	.476	.419	.369	.380	
.3612	1.011	.512	.506	.458	.401	.393	.392	
.4444	.982	.459	.453	.422	.371	.363	.367	
.5278	.901	.349	.294	.325	.269	.333	.335	
.6115	.866	.314	.235	.241	.246	.196	.247	
.6678	.837	.326	.235	.205	.216	.155	.152	
.7094	.796	.355	.241	.205	.144	.101	.127	
.7372	.738	.372	.276	.223	.120	.089	.108	
.7647	.709	.360	.312	.223	.132	.101	.139	
.7925	.645	.378	.300	.203	.162	.137	.139	
Flap								
.0582	.668	.343	.271	.277	.168	.155	.101	
.1306	.523	.337	.253	.241	.138	.129	.108	
.2758	.668	.308	.253	.133	.072	.083	.089	
.4177	.668	.198	.076	.060	.048	.060	.038	
.5600	.663	.163	.106	.120	.096	.095	.114	
.7019	.634	.110	.129	.145	.156	.143	.209	
.8527	.453	.238	.200	.181	.250	.238	.278	
1.0000	.529	.395	.447	.458	.443	.411	.671	

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.122	2.215	.247	6.584	4.533	3.970	3.525	
.0453	.203	.372	1.659	3.693	3.374	3.857	3.329	
.0927	.343	.639	1.576	2.976	3.904	3.863	3.342	
.1852	.517	.773	1.518	2.446	3.964	3.863	3.304	
.2786	.616	.884	1.518	2.235	4.018	3.869	3.329	
.3706	.709	1.000	1.553	2.157	4.024	3.863	3.329	
.5565	.878	1.134	1.623	2.139	3.677	3.893	3.342	
.7425	1.052	1.296	1.759	2.211	3.024	3.922	3.354	
.9272	1.337	1.633	2.112	2.572	2.605	3.887	3.380	
Wing								
.1944	1.349	1.529	1.859	2.277	2.395	3.464	3.405	
.2500	1.389	1.564	1.812	2.145	2.299	2.976	3.316	
.3057	1.459	1.628	1.882	2.145	2.251	2.547	3.146	
.3612	1.546	1.680	1.941	2.193	2.275	2.327	2.937	
.4444	1.721	1.860	2.088	2.301	2.377	2.309	2.690	
.5278	1.897	2.224	2.316	2.509	2.714	2.643	2.660	
.6115	2.293	2.379	2.667	2.800	2.964	2.893	2.585	
.6678	2.534	2.724	2.877	3.145	3.446	3.089	2.604	
.7094	2.965	3.172	3.421	3.563	3.661	3.518	2.736	
.7372	3.483	3.690	3.965	4.182	4.393	4.107	2.830	
.7647	4.465	4.655	4.965	5.163	5.296	4.875	3.283	
.7925	7.121	7.310	7.614	7.927	7.957	7.321	4.415	
Flap								
-.0420	53.706	54.309	55.576	48.234	57.160	56.178	30.431	
-.0350	45.896	46.516	47.085	49.670	48.892	47.910	39.243	
-.0240	50.861	51.809	52.524	54.943	52.607	51.071	40.262	
-.0130	62.188	63.619	65.576	68.324	64.939	62.535	47.809	
.0000	29.827	30.482	31.525	32.835	31.250	30.018	22.037	
.0582	6.448	6.707	6.807	7.181	7.089	6.411	3.981	
.1306	4.345	4.569	4.634	4.854	4.786	4.446	3.038	
.2758	2.896	3.069	3.193	3.127	3.268	2.982	2.509	
.4177	2.396	2.414	2.667	2.618	2.732	2.518	2.340	
.5600	1.945	1.931	2.035	2.036	2.071	1.982	2.189	
.7019	1.483	1.500	1.579	1.527	1.625	1.625	2.075	
.8527	1.293	1.224	1.491	1.344	1.393	1.375	1.849	
1.0000	.862	.741	.895	.764	.679	.788	1.000	

TABLE IV.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE

(a) $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 20^\circ$; $q = 12.5$ lb/sq ft, $C_L = 0$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.804	1.679	1.481	1.204	1.411	.433	.060	.000
.0927	1.797	1.648	1.494	1.192	1.250	.573	.169	.013
.1852	1.797	1.679	1.500	1.192	1.030	.618	.337	.132
.2786	1.804	1.697	1.512	1.192	.917	.618	.337	.182
.3706	1.810	1.691	1.500	1.204	.845	.586	.373	.226
.5565	1.816	1.721	1.524	1.204	.720	.554	.386	.258
.7425	1.823	1.715	1.512	1.220	.595	.516	.337	.245
.9272	1.829	1.721	1.531	1.220	.482	.318	.331	.226
Wing								
.1944	1.778	1.757	1.580	1.263	.613	.459	.367	.308
.2500	1.791	1.782	1.611	1.234	.631	.541	.458	.384
.3057	1.791	1.800	1.636	1.168	.649	.592	.494	.440
.3612	1.804	1.830	1.642	1.072	.661	.592	.530	.453
.4444	1.823	1.860	1.623	.958	.612	.599	.524	.472
.5278	1.829	1.794	1.524	.844	.565	.529	.470	.447
.6115	1.810	1.709	1.389	.784	.446	.414	.380	.371
.6678	1.766	1.618	1.302	.731	.399	.312	.301	.289
.7094	1.766	1.570	1.216	.713	.375	.293	.277	.226
.7372	1.728	1.473	1.129	.695	.357	.325	.277	.220
.7647	1.696	1.382	1.043	.665	.369	.293	.265	.239
.7925	1.639	1.303	.981	.665	.339	.280	.253	.245
Flap								
.0582	1.650	1.350	.988	.677	.351	.293	.265	.239
.1306	1.650	1.370	1.062	.731	.381	.312	.277	.239
.2758	1.772	1.285	.932	.587	.286	.248	.247	.176
.4177	1.703	1.248	.895	.545	.274	.210	.217	.173
.5600	1.753	1.250	.858	.569	.321	.306	.301	.270
.7019	1.778	1.248	.870	.617	.434	.465	.434	.409
.8527	1.462	1.527	1.278	.904	.714	.707	.578	.641
1.0000	1.835	1.648	1.611	1.695	1.792	1.886	1.819	1.799

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.734	1.709	1.537	1.228	1.470	.338	3.699	4.924
.0453	.082	.018	.031	.156	.518	1.860	3.783	4.044
.0927	.044	.012	.062	.329	.714	1.701	3.638	4.082
.1852	.006	.073	.191	.509	.851	1.643	3.072	4.182
.2786	.076	.170	.352	.635	.994	1.624	2.373	4.295
.3706	.139	.267	.463	.743	1.089	1.669	2.102	4.226
.5565	.310	.448	.654	.958	1.286	1.809	2.157	3.572
.7425	.468	.642	.876	1.180	1.536	1.993	2.349	2.906
.9272	.813	1.006	1.296	1.689	2.047	2.630	2.952	2.836
Wing								
.1944	.880	1.012	1.197	1.479	1.714	2.070	2.331	2.516
.2500	.905	1.006	1.142	1.359	1.559	1.828	2.018	2.239
.3057	.949	1.030	1.173	1.347	1.518	1.751	1.879	2.069
.3612	.968	1.075	1.185	1.347	1.492	1.688	1.819	1.962
.4444	1.051	1.151	1.247	1.341	1.488	1.650	1.741	1.855
.5278	1.358	1.418	1.518	1.589	1.768	2.000	1.873	1.924
.6115	1.453	1.454	1.518	1.661	1.821	1.961	1.836	1.868
.6678	1.490	1.473	1.537	1.714	1.786	1.942	1.854	1.830
.7094	1.415	1.491	1.537	1.679	1.750	1.846	1.818	1.773
.7372	1.528	1.509	1.593	1.679	1.750	1.904	1.818	1.736
.7647	1.525	1.527	1.630	1.750	1.786	1.981	1.854	1.821
.7925	1.679	1.636	1.741	1.750	1.875	2.000	1.891	1.868
Flap								
-.0420	1.887	1.891	1.963	1.857	2.179	2.231	2.091	2.038
-.0350	1.830	1.800	1.870	1.946	2.054	2.135	2.036	1.924
-.0240	1.736	1.727	1.778	1.839	1.893	2.077	1.982	1.868
-.0130	1.679	1.691	1.722	1.804	1.946	2.019	1.909	1.868
.0000	1.679	1.673	1.685	1.857	1.911	2.000	1.873	1.849
.0582	1.679	1.691	1.722	1.821	1.929	2.000	1.891	1.830
.1306	1.736	1.654	1.722	1.786	1.857	2.038	1.964	1.924
.2758	1.830	1.618	1.734	1.804	1.804	2.038	1.927	1.962
.4177	1.755	1.600	1.704	1.821	1.929	2.058	1.927	1.906
.5600	1.755	1.673	1.722	1.786	1.857	2.038	1.964	1.906
.7019	1.811	1.673	1.741	1.804	1.964	2.019	1.945	1.906
.8527	1.717	1.709	1.704	1.804	1.875	2.058	1.945	1.924
1.0000	1.887	1.709	1.741	1.643	1.643	1.808	1.830	1.773

TABLE IV.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(b) $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 20^\circ$; $q = 12.5$ lb/sq ft; $C_\mu = 0.012$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.429	1.325	1.217	1.244	.874	.169	.019	.019
.0927	1.447	1.349	1.217	1.208	.868	.313	.070	.012
.1852	1.459	1.361	1.224	1.202	.796	.428	.185	.099
.2786	1.476	1.367	1.230	1.196	.766	.446	.248	.160
.3706	1.453	1.373	1.236	1.167	.707	.452	.274	.198
.5565	1.465	1.386	1.230	1.065	.593	.428	.280	.247
.7425	1.465	1.386	1.242	.994	.503	.392	.241	.210
.9272	1.459	1.416	1.242	.887	.359	.331	.255	.210
Wing								
.1944	1.482	1.434	1.273	.732	.485	.404	.293	.259
.2500	1.500	1.464	1.292	.625	.551	.482	.382	.333
.3057	1.506	1.482	1.304	.637	.581	.512	.414	.358
.3612	1.523	1.506	1.342	.613	.563	.530	.433	.407
.4444	1.553	1.518	1.323	.595	.545	.506	.433	.426
.5278	1.553	1.518	1.304	.530	.479	.440	.420	.395
.6115	1.547	1.458	1.261	.446	.359	.331	.331	.321
.6678	1.476	1.434	1.224	.405	.311	.301	.242	.235
.7094	1.500	1.367	1.174	.375	.293	.271	.210	.210
.7372	1.459	1.337	1.130	.351	.299	.295	.217	.185
.7647	1.406	1.217	1.068	.351	.305	.283	.248	.198
.7925	1.335	1.163	1.031	.357	.329	.283	.255	.235
Flap								
.0582	1.347	1.157	1.012	.363	.305	.277	.261	.222
.1306	1.359	1.193	1.050	.269	.317	.295	.255	.222
.2758	1.423	1.205	1.056	.268	.263	.211	.185	.198
.4177	1.400	1.157	1.006	.315	.210	.205	.134	.160
.5600	1.482	1.181	.975	.339	.257	.277	.204	.216
.7019	1.494	1.133	.919	.434	.359	.373	.318	.309
.8527	1.476	1.343	1.106	.679	.629	.596	.503	.506
1.0000	1.823	1.398	1.329	1.631	1.725	1.819	1.860	1.518

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.482	1.398	1.298	1.149	.449	1.849	4.611	4.493
.0453	.076	.012	.062	.327	1.012	2.482	3.974	3.888
.0927	.000	.030	.193	.530	1.114	2.145	3.974	3.901
.1852	.100	.181	.366	.720	1.210	1.958	4.000	3.913
.2786	.171	.283	.509	.845	1.299	1.886	3.974	3.938
.3706	.265	.396	.627	.958	1.383	1.892	3.732	3.938
.5565	.418	.578	.838	1.131	1.539	2.024	2.879	3.938
.7425	.623	.765	1.075	1.393	1.790	2.205	2.548	3.703
.9272	.965	1.199	1.547	1.911	2.401	2.879	3.032	3.259
Wing								
.1944	1.006	1.205	1.416	1.696	1.958	2.277	2.599	2.740
.2500	1.012	1.181	1.366	1.577	1.766	2.006	2.306	2.456
.3057	1.071	1.211	1.379	1.571	1.731	1.928	2.197	2.290
.3612	1.118	1.259	1.404	1.553	1.701	1.873	2.095	2.136
.4444	1.206	1.343	1.466	1.625	1.743	1.886	2.070	2.049
.5278	1.403	1.454	1.741	1.821	1.964	2.018	2.288	2.148
.6115	1.579	1.654	1.852	1.982	2.071	2.109	2.385	2.132
.6678	1.719	1.782	2.037	2.018	2.232	2.218	2.615	2.148
.7094	1.860	1.891	2.148	2.179	2.304	2.382	2.538	2.185
.7372	1.965	2.145	2.315	2.393	2.500	2.564	2.711	2.278
.7647	2.316	2.418	2.611	2.714	2.821	2.891	3.058	2.426
.7925	3.386	3.545	3.741	3.786	3.857	3.945	3.884	2.926
Flap								
-.0420	9.614	10.000	10.185	10.250	10.411	10.745	10.115	6.889
-.0350	11.298	11.781	12.037	11.964	11.857	12.272	11.538	7.555
-.0240	11.789	12.327	12.407	12.393	12.286	12.672	11.826	7.833
-.0130	11.233	11.818	12.000	11.786	11.678	12.036	11.423	7.796
.0000	7.614	7.963	8.074	7.964	7.750	8.127	7.692	5.222
.0582	2.982	3.018	3.167	3.143	3.179	3.218	3.192	2.444
.1306	2.017	2.073	2.204	2.286	2.250	2.327	2.385	1.963
.2758	1.737	1.654	1.759	1.946	1.804	1.909	1.981	1.593
.4177	1.614	1.618	1.815	1.929	1.964	2.018	2.096	1.556
.5600	1.649	1.709	1.778	1.929	2.054	2.109	2.077	1.481
.7019	1.561	1.691	1.889	1.929	1.982	2.000	2.058	1.518
.8527	1.544	1.564	1.741	1.893	2.018	2.091	2.115	1.518
1.0000	1.789	1.509	1.593	1.643	1.821	1.800	1.904	1.426

TABLE IV - PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(c) $c_f = 0.30c_w$; $\delta_f = 60^\circ$; $\delta_N = 20^\circ$; $q = 12.5$ lb/sq ft; $C_L = 0.027$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.241	1.011	1.093	1.232	.379	.031	.020	.012
.0927	1.241	1.000	1.081	1.087	.516	.135	.026	.012
.1852	1.247	1.017	1.081	.907	.559	.245	.124	.093
.2786	1.241	1.034	1.093	.791	.553	.285	.176	.167
.3706	1.247	1.017	1.087	.721	.540	.301	.222	.191
.5565	1.223	1.046	1.104	.605	.503	.313	.248	.204
.7425	1.253	1.052	1.093	.494	.447	.282	.229	.200
.9272	1.271	1.057	1.104	.372	.286	.294	.216	.185
Wing								
.1944	1.294	1.115	1.029	.459	.404	.307	.288	.259
.2500	1.318	1.138	.907	.523	.466	.374	.327	.321
.3057	1.329	1.167	.796	.546	.516	.429	.399	.370
.3612	1.341	1.195	.703	.523	.503	.429	.392	.383
.4444	1.376	1.230	.616	.488	.478	.423	.379	.395
.5278	1.376	1.276	.558	.401	.429	.362	.340	.370
.6115	1.400	1.293	.506	.302	.317	.298	.261	.278
.6678	1.347	1.236	.436	.267	.267	.227	.209	.210
.7094	1.306	1.172	.442	.267	.255	.221	.196	.198
.7372	1.282	1.103	.430	.250	.267	.184	.196	.185
.7647	1.229	.977	.424	.279	.267	.233	.190	.198
.7925	1.206	.943	.430	.279	.273	.233	.209	.185
Flap								
.0582	1.188	.931	.471	.291	.248	.209	.190	.185
.1306	1.235	1.017	.523	.291	.267	.196	.203	.198
.2758	1.271	.994	.314	.198	.180	.098	.131	.130
.4177	1.288	1.029	.302	.140	.143	.055	.078	.111
.5600	1.282	1.000	.305	.203	.250	.200	.150	.250
.7019	1.329	.983	.308	.256	.317	.374	.203	.302
.8527	1.282	.965	.535	.600	.366	.319	.359	.562
1.0000	1.494	1.069	.581	1.500	1.745	1.546	.752	1.210

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.271	1.098	1.122	1.325	.466	6.533	5.437	4.222
.0453	.010	.017	.250	.575	1.882	4.073	4.398	3.851
.0927	.012	.109	.419	.744	1.752	3.239	4.457	3.864
.1852	.129	.282	.616	.907	1.689	2.576	4.561	3.864
.2786	.218	.391	.756	1.035	1.665	2.355	4.607	3.870
.3706	.315	.506	.860	1.128	1.733	2.270	4.405	3.870
.5565	.494	.718	1.070	1.349	1.894	2.368	3.437	3.870
.7425	.723	.943	1.302	1.581	2.155	2.527	2.954	3.679
.9272	1.118	1.385	1.872	2.197	2.814	3.245	3.228	3.272
Wing								
.1944	1.147	1.316	1.674	1.889	2.292	2.576	2.888	2.753
.2500	1.153	1.259	1.587	1.732	2.056	2.282	2.575	2.506
.3057	1.200	1.305	1.604	1.726	2.019	2.165	2.431	2.333
.3612	1.276	1.356	1.633	1.732	1.994	2.098	2.359	2.197
.4444	1.388	1.494	1.773	1.814	2.050	2.122	2.353	2.074
.5278	1.649	1.776	2.000	2.121	2.333	2.418	2.627	2.185
.6115	1.824	1.965	2.207	2.241	2.537	2.527	2.823	2.185
.6678	1.965	2.207	2.483	2.431	2.741	2.673	2.961	2.185
.7094	2.105	2.483	2.690	2.690	2.889	2.854	3.255	2.241
.7372	2.403	2.879	3.103	2.931	3.130	3.145	3.568	2.278
.7647	2.842	3.465	3.741	3.414	3.592	3.618	4.157	2.444
.7925	4.105	5.138	5.569	4.948	5.074	5.072	5.706	2.815
Flap								
-.0420	13.034	15.517	16.965	15.534	15.296	15.108	16.588	7.741
-.0350	15.210	18.413	20.344	18.327	17.925	17.472	19.274	8.111
-.0240	15.596	19.086	21.413	19.241	18.685	18.272	20.234	8.537
-.0130	15.333	18.879	21.293	19.206	18.481	18.090	20.195	8.907
.0000	10.157	12.707	14.069	12.431	12.037	11.799	13.254	5.574
.0582	3.614	4.724	5.379	4.655	4.444	4.436	5.176	2.500
.1306	2.456	3.207	3.793	3.345	3.167	3.218	3.647	2.037
.2758	1.842	2.382	2.586	2.224	2.389	2.364	2.235	1.704
.4177	1.579	1.914	1.948	2.052	2.241	2.127	1.941	1.667
.5600	1.649	1.517	1.517	2.103	2.259	2.236	1.588	1.648
.7019	1.719	1.310	1.207	2.086	2.296	2.164	1.294	1.593
.8527	1.614	1.017	.931	2.034	2.204	2.018	1.176	1.574
1.0000	1.649	.983	.621	1.552	1.815	1.745	.765	1.389

TABLE IV.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(d) $c_f = 0.20c_w$, $\delta_f = 60^\circ$; $\delta_N = 20^\circ$; $q = 12.5$ lb/sq ft; $C_\mu = 0.037$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.088	1.012	1.079	.946	.210	.024	.012	.031
.0927	1.123	1.018	1.073	.917	.341	.120	.006	.019
.1852	1.106	1.023	1.097	.798	.429	.246	.104	.099
.2786	1.112	1.023	1.067	.738	.431	.275	.153	.173
.3706	1.100	1.047	1.097	.702	.431	.299	.209	.185
.5565	1.123	1.047	1.091	.548	.395	.305	.221	.241
.7425	1.135	1.076	1.079	.452	.365	.275	.215	.210
.9272	1.153	1.065	1.048	.345	.251	.275	.190	.198
Wing								
.1944	1.159	1.118	1.006	.446	.353	.323	.251	.272
.2500	1.165	1.141	.903	.506	.401	.377	.307	.327
.3057	1.212	1.182	.794	.512	.443	.407	.362	.370
.3612	1.241	1.206	.709	.512	.425	.395	.356	.383
.4444	1.306	1.247	.618	.484	.413	.395	.380	.414
.5278	1.388	1.294	.564	.393	.341	.359	.325	.370
.6115	1.400	1.265	.521	.292	.240	.269	.251	.284
.6678	1.359	1.223	.479	.262	.228	.228	.172	.228
.7094	1.300	1.141	.424	.238	.204	.216	.184	.191
.7372	1.223	1.053	.412	.262	.192	.222	.196	.198
.7647	1.118	.894	.436	.280	.228	.228	.215	.204
.7925	1.035	.859	.418	.327	.240	.263	.196	.198
Flap								
.0582	1.088	.876	.430	.236	.228	.234	.195	.198
.1306	1.135	.900	.479	.268	.210	.222	.209	.210
.2758	1.159	.929	.327	.173	.144	.132	.153	.265
.4177	1.212	.953	.315	.095	.102	.066	.104	.272
.5600	1.218	1.123	.320	.150	.126	.100	.150	.350
.7019	1.206	.988	.327	.167	.168	.216	.166	.420
.8527	1.129	.871	.467	.286	.246	.389	.301	.443
1.0000	1.123	.753	.533	.488	.503	.515	.607	1.062

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.135	1.123	1.097	.667	1.425	5.377	5.214	4.253
.0453	.012	.059	.267	.809	2.263	4.802	4.330	3.851
.0927	.082	.176	.485	.964	1.934	4.299	4.349	3.870
.1852	.200	.329	.654	1.077	1.784	3.066	4.441	3.870
.2786	.312	.465	.788	1.167	1.731	2.605	4.521	3.888
.3706	.429	.565	.921	1.280	1.778	2.407	4.355	3.901
.5565	.582	.765	1.139	1.500	1.904	2.407	3.527	3.932
.7425	.841	1.000	1.394	1.726	2.150	2.587	2.944	3.759
.9272	1.265	1.476	1.970	2.369	2.850	3.245	3.104	3.370
Wing								
.1944	1.253	1.406	1.733	2.030	2.287	2.597	2.833	2.845
.2500	1.253	1.323	1.612	1.839	2.048	2.323	2.521	2.635
.3057	1.300	1.412	1.679	1.821	1.994	2.222	2.386	2.432
.3612	1.394	1.447	1.709	1.845	1.988	2.150	2.319	2.302
.4444	1.529	1.594	1.848	1.940	2.050	2.210	2.313	2.185
.5278	1.895	1.789	2.109	2.321	2.464	2.536	2.236	2.241
.6115	2.158	2.017	2.436	2.536	2.714	2.750	2.382	2.259
.6678	2.438	2.298	2.654	2.821	2.911	2.911	2.636	2.259
.7094	2.789	2.579	2.945	3.018	3.161	3.258	2.854	2.333
.7372	3.052	2.930	3.327	3.500	3.571	3.607	3.145	2.444
.7647	3.877	3.596	4.054	4.214	4.232	4.268	3.673	2.574
.7925	5.824	5.421	6.091	6.321	6.375	6.196	5.327	3.037
Flap								
-.0420	19.431	18.964	20.545	21.000	20.803	20.036	17.635	10.537
-.0350	22.999	22.034	24.035	24.428	24.232	23.214	20.090	10.796
-.0240	24.016	23.350	25.253	25.678	25.518	24.518	21.163	11.370
-.0130	24.069	23.788	25.562	26.107	25.857	24.821	21.508	11.963
.0000	15.876	15.105	16.472	16.857	16.678	16.000	13.599	7.129
.0582	5.684	5.280	5.818	6.089	6.099	5.893	4.854	2.796
.1306	3.912	3.684	4.000	4.214	4.232	4.071	3.345	2.296
.2758	2.649	2.386	2.745	2.839	2.839	2.679	2.491	1.963
.4177	2.281	1.877	2.109	2.179	2.179	2.000	1.600	1.815
.5600	1.807	1.526	1.691	1.768	1.804	1.679	1.436	1.741
.7019	1.579	1.246	1.400	1.429	1.411	1.357	1.109	1.722
.8527	1.281	1.035	1.036	1.107	.982	1.036	.873	1.667
1.0000	1.140	.684	.564	.661	.696	.643	.527	1.556

TABLE IV.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(c) $c_f = 0.20c_w$; $\delta_f = 30^\circ$; $\delta_N = 20^\circ$; $q = 12.5$ lb/sq ft; $C_L = 0.061$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453		1.006	1.138	.818	.156	.017	.031	.023
.0927		.994	1.138	.818	.275	.086	.006	.041
.1852		1.006	1.144	.745	.389	.195	.086	.110
.2786		1.018	1.126	.703	.389	.241	.129	.157
.3706		1.023	1.126	.642	.407	.264	.159	.198
.5565		1.029	1.060	.570	.383	.259	.202	.244
.7425		1.053	1.036	.461	.359	.253	.190	.215
.9272		1.047	.976	.333	.275	.259	.172	.209
Wing								
.1944		1.112	.892	.430	.347	.276	.233	.267
.2500		1.141	.766	.473	.383	.351	.319	.314
.3057		1.153	.695	.491	.419	.374	.325	.360
.3612		1.176	.653	.491	.431	.374	.331	.384
.4444		1.218	.569	.455	.407	.351	.350	.378
.5278		1.259	.503	.359	.335	.322	.325	.360
.6115		1.247	.455	.261	.251	.253	.221	.296
.6678		1.188	.431	.236	.216	.201	.159	.238
.7094		1.094	.413	.242	.228	.201	.184	.221
.7372		1.000	.389	.248	.222	.201	.172	.221
.7647		.865	.395	.279	.234	.224	.190	.221
.7925		.812	.359	.285	.246	.241	.209	.221
Flap								
.0582		.859	.401	.273	.246	.218	.190	.203
.1306		.859	.443	.261	.246	.213	.184	.215
.2758		.871	.437	.182	.144	.121	.196	.163
.4177		.912	.389	.103	.084	.069	.166	.140
.5600		.947	.413	.150	.150	.103	.170	.250
.7019		.871	.401	.176	.174	.167	.178	.314
.8527		.876	.431	.273	.216	.218	.270	.453
1.0000		.718	.461	.455	.443	.437	.485	1.070

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000		1.094	1.084	.261	2.030	4.615	5.521	3.906
.0453		.047	.287	1.000	2.305	3.741	4.416	3.674
.0927		.176	.491	1.121	2.144	3.747	4.478	3.656
.1852		.353	.695	1.194	1.952	3.736	4.588	3.668
.2786		.459	.826	1.309	1.904	3.368	4.680	3.668
.3706		.571	.922	1.388	1.934	2.810	4.613	3.685
.5565		.771	1.162	1.594	2.084	2.293	3.819	3.691
.7425		1.000	1.437	1.860	2.323	2.448	3.110	3.587
.9272		1.494	2.036	2.521	3.048	3.075	3.147	3.255
Wing								
.1944		1.406	1.772	2.127	2.443	2.557	2.889	2.813
.2500		1.365	1.671	1.957	2.204	2.276	2.576	2.598
.3057		1.412	1.689	1.933	2.132	2.184	2.472	2.436
.3612		1.465	1.731	1.933	2.126	2.149	2.374	2.290
.4444		1.606	1.868	2.086	2.216	2.195	2.380	2.151
.5278		1.895	2.268	2.418	2.500	2.451	2.673	2.092
.6115		2.175	2.589	2.709	2.732	2.690	2.800	2.069
.6678		2.386	2.857	3.145	3.018	2.931	3.054	2.089
.7094		2.702	3.161	3.273	3.357	3.172	3.291	2.103
.7372		3.052	3.643	3.782	3.768	3.569	3.636	2.190
.7647		3.754	4.411	4.545	4.589	4.293	4.254	2.396
.7925		5.754	6.571	6.872	6.857	6.262	6.054	2.828
Flap								
-.0420		23.157	25.036	26.035	25.839	24.017	22.763	11.965
-.0350		25.806	28.157	29.271	28.982	26.775	24.835	11.586
-.0240		26.929	29.357	30.708	30.446	28.155	25.926	11.724
-.0130		28.069	30.339	31.762	31.464	29.086	26.908	12.482
.0000		17.104	18.786	19.599	19.411	17.879	16.490	7.069
.0582		5.544	6.196	6.436	6.446	5.948	5.563	2.414
.1306		3.807	4.214	4.436	4.464	4.121	3.909	2.069
.2758		2.667	2.964	3.073	3.018	2.776	2.909	1.793
.4177		2.035	2.339	2.327	2.321	2.155	2.091	1.707
.5600		1.649	1.821	1.945	1.875	1.776	1.654	1.621
.7019		1.368	1.554	1.491	1.518	1.414	1.327	1.586
.8527		1.070	1.286	1.182	1.196	1.052	1.182	1.552
1.0000		.684	.607	.655	.625	.586	.655	1.190

TABLE IV - PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(f) $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 20^\circ$; $q = 12.5$ lb/sq ft; $C_\mu = 0.119$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.066	.957	1.553	.612	.068	.000	.054	
.0927	1.054	.975	1.423	.679	.198	.061	.018	
.1852	1.078	.975	1.244	.654	.309	.166	.072	
.2786	1.084	.981	1.119	.600	.358	.215	.120	
.3706	1.078	.981	.988	.570	.370	.227	.156	
.5565	1.084	1.012	.809	.521	.346	.239	.174	
.7425	1.078	1.018	.679	.412	.309	.245	.174	
.9272	1.114	1.037	.589	.297	.296	.227	.168	
Wing								
.1944	1.239	1.096	.613	.388	.321	.264	.229	
.2500	1.151	1.086	.571	.418	.389	.331	.281	
.3057	1.193	1.067	.571	.448	.407	.356	.317	
.3612	1.253	1.049	.518	.455	.407	.374	.323	
.4444	1.398	1.024	.476	.436	.389	.356	.323	
.5278	1.494	1.006	.399	.358	.333	.313	.269	
.6115	1.464	.957	.345	.255	.247	.245	.216	
.6678	1.386	.902	.327	.218	.204	.209	.162	
.7094	1.283	.840	.327	.230	.204	.190	.156	
.7372	1.169	.810	.333	.230	.216	.196	.156	
.7647	.970	.699	.315	.279	.235	.196	.180	
.7925	.934	.650	.333	.273	.247	.258	.180	
Flap								
.0582	.916	.638	.339	.255	.228	.221	.180	
.1306	.904	.656	.357	.248	.216	.215	.180	
.2758	1.066	.681	.244	.158	.142	.098	.096	
.4177	1.133	.675	.190	.095	.074	.074	.060	
.5600	1.175	.675	.179	.097	.099	.117	.108	
.7019	1.169	.638	.190	.127	.148	.147	.144	
.8527	1.036	.626	.321	.212	.191	.209	.228	
1.0000	.801	.552	.434	.442	.414	.405	.415	

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.102	1.086	1.464	.048	3.876	4.834	4.832	
.0453	.012	.098	.345	1.248	2.821	4.208	4.575	
.0927	.096	.239	.536	1.309	2.487	4.296	4.569	
.1852	.247	.436	.750	1.357	2.210	4.232	4.575	
.2786	.355	.570	.881	1.418	2.136	4.183	4.635	
.3706	.470	.669	1.000	1.497	2.111	3.785	4.647	
.5565	.681	.902	1.232	1.703	2.259	2.742	4.575	
.7425	.898	1.135	1.524	1.976	2.506	2.656	4.036	
.9272	1.337	1.650	2.119	2.648	3.290	3.300	3.311	
Wing								
.1944	1.367	1.564	1.863	2.230	2.654	2.834	2.898	
.2500	1.361	1.509	1.762	2.036	2.370	2.539	2.611	
.3057	1.434	1.558	1.774	2.030	2.315	2.429	2.503	
.3612	1.500	1.613	1.827	2.060	2.284	2.385	2.413	
.4444	1.639	1.760	1.952	2.157	2.389	2.466	2.429	
.5278	1.745	2.018	2.268	2.454	2.667	2.709	2.714	
.6115	2.291	2.345	2.607	2.745	3.037	3.036	2.893	
.6678	2.800	2.618	2.929	3.036	3.333	3.273	3.107	
.7094	2.927	2.945	3.304	3.418	3.704	3.600	3.500	
.7372	3.436	3.473	3.732	3.945	4.241	4.054	3.804	
.7647	4.200	4.163	4.607	4.854	5.111	4.945	4.607	
.7925	6.527	6.418	7.089	7.436	7.796	7.436	6.750	
Flap								
-.0420	35.889	36.180	36.696	37.962	39.277	36.798	33.893	
-.0350	36.128	36.107	37.196	38.489	39.925	37.198	33.714	
-.0240	36.817	36.871	38.321	39.798	41.314	38.598	34.821	
-.0130	39.580	39.725	41.160	42.780	44.295	41.225	37.375	
.0000	22.254	22.163	23.232	24.144	25.073	23.399	21.018	
.0582	6.145	5.982	6.482	6.828	7.129	6.745	5.982	
.1306	4.182	4.073	4.393	4.673	4.852	4.591	4.250	
.2758	3.018	2.836	3.143	3.182	3.352	3.273	2.857	
.4177	2.327	2.182	2.446	2.527	2.630	2.527	2.268	
.5600	1.854	1.854	1.946	2.054	2.167	2.018	1.768	
.7019	1.545	1.418	1.589	1.618	1.657	1.636	1.536	
.8527	1.209	1.200	1.268	1.295	1.370	1.230	1.250	
1.0000	.964	.600	.518	.582	.519	.582	.643	

TABLE IV.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(d) $c_f = 0.200c_r$, $\delta_f = 60^\circ$; $\delta_N = 20^\circ$, $q = 12.6$ lb/sq ft, $C_\mu = 0.186$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.024	1.000	1.497	.491	.042	.031	.061	
.0927	1.024	1.018	1.293	.584	.158	.044	.012	
.1852	1.024	1.024	1.030	.609	.261	.138	.074	
.2786	1.054	1.043	.898	.584	.327	.182	.129	
.3706	1.060	1.031	.802	.547	.327	.220	.147	
.5565	1.078	1.037	.659	.484	.327	.245	.178	
.7425	1.072	1.061	.569	.410	.291	.226	.172	
.9272	1.096	1.037	.491	.273	.291	.208	.172	
Wing								
.1944	1.102	1.061	.587	.391	.291	.270	.233	
.2500	1.132	1.043	.611	.441	.388	.321	.252	
.3057	1.250	1.012	.593	.460	.394	.358	.307	
.3612	1.222	.939	.569	.447	.376	.377	.356	
.4444	1.353	.877	.503	.429	.352	.352	.307	
.5278	1.431	.816	.419	.360	.303	.308	.294	
.6115	1.407	.791	.359	.255	.242	.252	.221	
.6678	1.317	.718	.347	.224	.212	.201	.159	
.7094	1.180	.693	.323	.199	.200	.170	.153	
.7372	1.054	.632	.323	.224	.206	.164	.159	
.7647	.910	.607	.353	.236	.200	.239	.184	
.7925	.832	.577	.353	.273	.224	.258	.202	
Flap								
.0582	.790	.607	.371	.236	.230	.208	.178	
.1306	.784	.656	.407	.217	.215	.182	.159	
.2758	.940	.540	.246	.149	.127	.082	.074	
.4177	1.024	.521	.198	.068	.085	.044	.074	
.5600	1.042	.540	.160	.093	.109	.107	.098	
.7019	1.006	.515	.174	.137	.127	.138	.153	
.8527	.856	.644	.383	.186	.194	.182	.221	
1.0000	.641	.534	.431	.422	.424	.421	.393	

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.048	1.216	2.078	.366	5.157	5.050	4.913	
.0453	.006	.098	.371	1.553	3.303	4.327	4.735	
.0927	.084	.276	.605	1.497	2.763	4.383	4.754	
.1852	.234	.472	.796	1.478	2.363	4.503	4.748	
.2786	.265	.570	.904	1.553	2.256	4.471	4.785	
.3706	.473	.705	1.054	1.615	2.230	4.207	4.815	
.5565	.689	.926	1.299	1.807	2.351	3.302	4.785	
.7425	.922	1.215	1.581	2.074	2.582	2.918	4.539	
.9272	1.383	1.730	2.204	2.776	3.327	3.302	3.920	
Wing								
.1944	1.359	1.638	1.952	2.310	2.666	2.937	3.226	
.2500	1.347	1.589	1.830	2.118	2.412	2.648	2.895	
.3057	1.407	1.638	1.856	2.087	2.327	2.547	2.711	
.3612	1.479	1.705	1.910	2.118	2.303	2.497	2.589	
.4444	1.641	1.859	2.054	2.248	2.424	2.560	2.613	
.5278	2.018	2.182	2.339	2.574	2.936	2.924	2.818	
.6115	2.393	2.545	2.714	2.907	3.163	3.189	3.159	
.6678	2.679	2.891	3.000	3.222	3.400	3.698	3.400	
.7094	3.036	3.200	3.411	3.648	3.818	3.981	3.727	
.7372	3.464	3.709	3.964	4.167	4.400	4.509	4.254	
.7647	4.357	4.636	4.911	5.167	5.382	5.415	5.054	
.7925	6.786	7.218	7.643	7.944	8.181	8.151	7.436	
Flap								
-.0420	50.910	53.125	53.285	54.517	48.216	55.412	51.343	
-.0350	45.607	47.798	47.982	49.610	49.889	49.998	46.034	
-.0240	44.982	47.671	48.875	49.943	50.761	50.564	45.780	
-.0130	50.464	53.779	54.964	56.258	57.107	56.865	51.416	
.0000	25.607	27.053	27.875	28.573	29.126	28.951	26.181	
.0582	6.000	6.509	6.804	6.981	7.254	7.207	6.382	
.1306	4.161	4.473	4.661	4.778	4.963	5.019	4.436	
.2758	3.000	3.163	3.321	3.463	3.491	3.660	3.182	
.4177	2.339	2.436	2.536	2.611	2.691	2.773	2.491	
.5600	1.946	1.964	2.125	2.222	2.327	2.453	1.927	
.7019	1.571	1.564	1.607	1.759	1.709	2.038	1.618	
.8527	1.429	1.418	1.393	1.407	1.473	1.490	1.291	
1.0000	.875	.727	.554	.667	.618	.528	.636	

TABLE IV - PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Concluded

(h) $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 20^\circ$; $q = 12.5$ lb/sq ft, $C_\mu = 0.250$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.006	1.035	1.341	.376	.012	.042	.077	
.0927	1.011	1.035	1.156	.471	.098	.036	.019	
.1852	1.023	1.052	.922	.529	.215	.121	.071	
.2786	1.029	1.064	.796	.494	.276	.158	.096	
.3706	1.035	1.075	.743	.471	.294	.200	.147	
.5565	1.046	1.093	.599	.447	.288	.206	.173	
.7425	1.064	1.093	.491	.418	.258	.206	.160	
.9272	1.081	1.081	.461	.271	.270	.194	.122	
Wing								
.1944	1.081	1.058	.455	.353	.270	.242	.224	
.2500	1.116	.982	.509	.424	.331	.315	.276	
.3057	1.157	.884	.509	.424	.356	.321	.327	
.3612	1.227	.825	.491	.412	.368	.339	.327	
.4444	1.349	.773	.431	.388	.368	.321	.321	
.5278	1.430	.727	.347	.329	.288	.279	.301	
.6115	1.383	.686	.269	.247	.215	.218	.224	
.6678	1.285	.639	.269	.229	.184	.200	.141	
.7094	1.168	.593	.269	.212	.190	.170	.135	
.7372	1.023	.581	.251	.212	.178	.188	.122	
.7647	.866	.500	.305	.259	.209	.218	.173	
.7925	.767	.494	.275	.241	.239	.230	.173	
Flap								
.0582	.785	.523	.257	.212	.196	.200	.147	
.1306	.785	.581	.311	.212	.172	.170	.141	
.2758	.907	.424	.216	.147	.135	.097	.045	
.4177	.982	.424	.132	.059	.049	.073	.032	
.5600	1.000	.442	.102	.094	.061	.125	.083	
.7019	.965	.401	.216	.141	.129	.158	.122	
.8527	.884	.593	.269	.159	.159	.170	.179	
1.0000	.628	.488	.443	.424	.411	.436	.410	

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.052	1.110	1.659	.576	7.349	5.393	4.974	
.0453	.012	.145	.461	1.694	3.950	4.290	4.782	
.0927	.134	.320	.659	1.618	3.153	4.351	4.788	
.1852	.279	.500	.874	1.576	2.601	4.472	4.801	
.2786	.401	.634	.988	1.559	2.386	4.490	4.814	
.3706	.494	.756	1.108	1.647	2.337	4.290	4.840	
.5565	.709	.988	1.347	1.812	2.435	3.363	4.840	
.7425	.965	1.250	1.611	2.070	2.650	2.921	4.724	
.9272	1.413	1.785	2.275	2.770	3.417	3.242	4.301	
Wing								
.1944	1.395	1.651	1.976	2.365	2.760	2.975	3.493	
.2500	1.366	1.587	1.868	2.159	2.497	2.666	3.139	
.3057	1.442	1.622	1.880	2.129	2.411	2.563	2.865	
.3612	1.500	1.692	1.922	2.135	2.392	2.521	2.724	
.4444	1.686	1.860	2.078	2.259	2.484	2.594	2.724	
.5278	1.931	2.293	2.429	2.474	2.745	2.836	3.077	
.6115	2.310	2.603	2.804	2.807	3.091	3.182	3.404	
.6678	2.621	2.914	3.268	3.368	3.582	3.509	3.615	
.7094	3.052	3.328	3.536	3.561	3.873	3.909	4.058	
.7372	3.483	3.914	4.071	4.105	4.345	4.418	4.442	
.7647	4.345	4.741	5.071	5.123	5.418	5.436	5.461	
.7925	6.862	7.345	7.839	7.894	8.327	8.145	7.980	
Flap								
-.0420	61.619	62.533	65.160	64.699	67.088	65.942	68.228	
-.0350	49.309	49.706	52.500	51.892	53.652	56.034	59.498	
-.0240	52.378	53.188	55.785	55.067	57.107	56.016	57.825	
-.0130	60.826	62.981	64.392	64.593	69.088	66.379	66.690	
.0000	28.482	29.775	31.410	31.577	32.835	31.508	31.441	
.0582	5.948	6.396	6.893	6.982	7.363	7.181	6.961	
.1306	4.138	4.431	4.732	4.772	5.000	4.963	4.846	
.2758	2.914	3.069	3.357	3.351	3.618	3.473	3.346	
.4177	2.345	2.552	2.607	2.544	2.709	2.691	2.596	
.5600	1.845	2.103	2.000	2.053	2.236	2.145	2.000	
.7019	1.483	1.655	1.643	1.509	1.709	1.618	1.596	
.8527	1.241	1.500	1.429	1.421	1.473	1.400	1.461	
1.0000	.776	.810	.643	.649	.582	.655	.712	

TABLE V.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE

(a) $c_f = 0.20c_w$, $\delta_f = 60^\circ$; $\delta_N = 25^\circ$; $q = 12.5$ lb/sq ft, $C_\mu = 0$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.694	1.617	1.488	1.190	1.240	.733	.119	.006
.0927	1.694	1.629	1.482	1.202	1.227	.776	.245	.032
.1852	1.700	1.617	1.530	1.208	1.234	.691	.365	.146
.2786	1.718	1.629	1.500	1.221	1.221	.648	.402	.197
.3706	1.700	1.641	1.524	1.196	1.162	.612	.396	.248
.5565	1.718	1.659	1.518	1.215	1.092	.545	.358	.242
.7425	1.723	1.665	1.524	1.227	.980	.400	.333	.210
.9272	1.706	1.677	1.524	1.227	.957	.285	.245	.210
Wing								
.1944	1.682	1.719	1.595	1.300	.773	.436	.314	.248
.2500	1.706	1.713	1.613	1.319	.695	.515	.402	.338
.3057	1.694	1.748	1.625	1.343	.682	.539	.465	.401
.3612	1.718	1.796	1.673	1.343	.643	.551	.516	.401
.4444	1.755	1.796	1.661	1.307	.629	.545	.491	.452
.5278	1.718	1.766	1.571	1.239	.552	.503	.465	.408
.6115	1.718	1.701	1.476	1.190	.448	.394	.377	.325
.6678	1.718	1.653	1.381	1.135	.383	.297	.264	.280
.7094	1.718	1.611	1.262	1.067	.370	.285	.270	.210
.7372	1.706	1.533	1.208	1.024	.312	.261	.239	.223
.7647	1.682	1.437	1.095	.981	.331	.291	.264	.210
.7925	1.647	1.389	1.030	.957	.305	.261	.233	.217
Flap								
.0582	1.400	1.401	1.036	.969	.305	.273	.233	.223
.1306	1.394	1.461	1.083	.926	.383	.285	.239	.223
.2758	1.718	1.401	1.006	.908	.227	.230	.182	.159
.4177	1.676	1.359	.940	.877	.247	.200	.176	.146
.5600	1.723	1.395	.929	.834	.318	.291	.277	.223
.7019	1.776	1.341	.869	.810	.461	.400	.428	.401
.8527	1.770	1.485	1.220	.988	.740	.697	.679	.624
1.0000	1.753	1.653	1.583	1.583	1.863	1.812	1.906	1.892

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.670	1.635	1.518	1.276	1.143	.345	3.031	5.044
.0453	.182	.036	.024	.025	.279	1.085	3.497	4.860
.0927	.024	.024	.024	.141	.474	1.157	2.264	4.879
.1852	.012	.036	.155	.319	.701	1.242	2.063	4.783
.2786	.047	.132	.262	.466	.851	1.309	1.975	3.879
.3706	.118	.198	.381	.589	.961	1.400	2.031	2.815
.5565	.253	.389	.577	.804	1.227	1.594	2.163	2.509
.7425	.447	.587	.815	1.073	1.454	1.879	2.428	2.783
.9272	.771	.994	1.280	1.638	2.214	2.624	3.245	3.611
Wing								
.1944	.847	1.006	1.214	1.421	1.944	1.982	2.421	2.669
.2500	.882	1.006	1.149	1.331	1.630	1.770	2.069	2.306
.3057	.912	1.030	1.167	1.288	1.571	1.679	1.924	2.108
.3612	.959	1.060	1.167	1.282	1.526	1.630	1.824	2.000
.4444	1.035	1.126	1.202	1.325	1.519	1.618	1.742	1.898
.5278	1.281	1.196	1.375	1.527	1.846	1.727	1.962	2.000
.6115	1.403	1.321	1.411	1.527	1.846	1.709	1.962	1.981
.6678	1.403	1.232	1.446	1.454	1.904	1.764	1.943	1.942
.7094	1.422	1.357	1.375	1.564	1.769	1.673	1.924	1.865
.7372	1.439	1.393	1.464	1.527	1.904	1.764	1.887	1.846
.7647	1.526	1.429	1.518	1.600	1.885	1.727	1.962	1.942
.7925	1.596	1.518	1.554	1.636	2.000	1.800	2.000	1.961
Flap								
-.0420	1.807	1.786	1.768	1.800	2.154	1.982	2.189	2.173
-.0350	1.719	1.661	1.696	1.782	2.077	1.927	2.132	2.096
-.0240	1.614	1.518	1.625	1.673	1.961	1.836	2.057	2.019
-.0130	1.561	1.536	1.607	1.673	1.961	1.818	2.000	1.981
.0000	1.561	1.571	1.589	1.654	1.981	1.764	2.019	1.961
.0582	1.561	1.607	1.625	1.636	2.019	1.782	2.000	2.038
.1306	1.544	1.607	1.625	1.673	2.000	1.800	2.000	1.961
.2758	1.614	1.589	1.625	1.709	2.000	1.854	1.906	2.077
.4177	1.649	1.589	1.625	1.673	1.981	1.836	2.000	2.000
.5600	1.650	1.589	1.589	1.654	1.923	1.764	1.924	2.000
.7019	1.649	1.554	1.661	1.654	1.923	1.836	2.038	2.077
.8527	1.631	1.696	1.643	1.600	1.904	1.782	2.000	2.038
1.0000	1.913	1.696	1.661	1.636	1.808	1.764	1.924	1.827

TABLE V.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(b) $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 25^\circ$; $q = 12.5$ lb/sq ft; $C_\mu = 0.013$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.506	1.401	1.176	1.210	1.218	.376	.018	.012
.0927	1.512	1.383	1.176	1.216	1.080	.475	.096	.012
.1852	1.537	1.389	1.188	1.228	.914	.531	.217	.087
.2786	1.531	1.407	1.188	1.234	.782	.518	.271	.143
.3706	1.531	1.383	1.188	1.240	.724	.494	.283	.174
.5565	1.524	1.395	1.194	1.269	.506	.457	.283	.193
.7425	1.537	1.407	1.188	1.257	.488	.420	.235	.174
.9272	1.543	1.401	1.182	1.222	.374	.247	.247	.155
Wing								
.1944	1.512	1.457	1.212	1.090	.494	.389	.277	.199
.2500	1.543	1.500	1.241	.934	.540	.444	.331	.286
.3057	1.562	1.506	1.241	.802	.580	.512	.404	.329
.3612	1.605	1.518	1.276	.707	.580	.531	.440	.373
.4444	1.599	1.543	1.288	.647	.557	.512	.428	.391
.5278	1.580	1.568	1.276	.605	.506	.438	.380	.360
.6115	1.592	1.549	1.259	.527	.408	.352	.301	.286
.6678	1.568	1.531	1.218	.467	.328	.284	.235	.230
.7094	1.543	1.475	1.200	.425	.305	.235	.205	.193
.7372	1.518	1.395	1.129	.407	.293	.265	.193	.193
.7647	1.432	1.308	1.065	.389	.305	.259	.193	.217
.7925	1.438	1.271	1.012	.395	.351	.272	.223	.205
Flap								
.0582	1.389	1.278	1.029	.461	.328	.265	.217	.168
.1306	1.389	1.302	1.035	.527	.333	.259	.223	.174
.2758	1.487	1.339	1.047	.257	.247	.228	.175	.130
.4177	1.506	1.296	1.023	.275	.247	.179	.139	.118
.5600	1.531	1.352	1.012	.353	.316	.228	.217	.217
.7019	1.549	1.333	.953	.425	.397	.346	.331	.304
.8527	1.500	1.420	1.065	.707	.615	.611	.548	.484
1.0000	1.944	1.549	1.294	1.629	1.759	1.919	1.879	1.851

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.518	1.395	1.194	1.222	1.264	.370	5.054	5.118
.0453	.074	.000	.000	.174	.592	1.833	4.614	4.683
.0927	.000	.012	.088	.359	.782	1.697	3.512	4.702
.1852	.037	.117	.235	.563	.948	1.629	2.645	4.689
.2786	.099	.235	.376	.695	1.080	1.666	2.373	4.584
.3706	.179	.327	.506	.808	1.201	1.728	2.307	4.304
.5565	.358	.543	.718	1.072	1.420	1.944	2.373	3.450
.7425	.586	.778	.965	1.371	1.695	2.222	2.632	2.851
.9272	.994	1.278	1.529	2.024	2.425	3.030	3.476	3.329
Wing								
.1944	1.062	1.271	1.400	1.731	1.965	2.370	2.608	2.514
.2500	1.086	1.222	1.306	1.581	1.776	2.086	2.235	2.447
.3057	1.136	1.259	1.312	1.539	1.718	1.975	2.102	2.273
.3612	1.167	1.290	1.353	1.545	1.707	1.938	2.018	2.161
.4444	1.284	1.370	1.400	1.623	1.747	1.950	1.982	2.099
.5278	1.574	1.574	1.684	1.929	1.741	2.037	2.164	2.222
.6115	1.685	1.778	1.772	2.089	1.845	2.204	2.254	2.278
.6678	1.796	1.889	1.930	2.179	1.965	2.333	2.291	2.278
.7094	1.981	2.093	2.035	2.304	2.086	2.426	2.436	2.444
.7372	2.167	2.241	2.175	2.482	2.293	2.630	2.654	2.574
.7647	2.518	2.611	2.491	2.875	2.638	3.018	2.982	2.815
.7925	3.592	3.611	3.509	3.929	3.638	4.092	3.945	3.537
Flap								
-.0420	10.240	10.240	9.543	10.571	10.034	10.889	10.127	8.666
-.0350	12.111	12.074	11.175	12.286	11.845	12.685	11.654	9.722
-.0240	12.426	12.518	11.613	12.786	12.241	13.166	12.054	9.981
-.0150	11.777	12.037	11.122	12.036	11.569	12.314	11.345	9.518
.0000	8.018	8.037	7.488	8.143	7.724	8.278	7.654	6.426
.0582	3.130	3.130	2.912	3.321	3.017	3.333	3.182	2.889
.1306	2.130	2.204	2.035	2.339	2.121	2.407	2.309	2.259
.2758	1.667	1.548	1.649	2.000	1.690	2.018	2.164	1.944
.4177	1.722	1.648	1.631	2.018	1.776	2.167	2.073	1.944
.5600	1.722	1.593	1.737	2.054	1.810	2.093	2.109	2.018
.7019	1.759	1.685	1.684	2.071	1.776	2.000	2.000	1.981
.8527	1.593	1.556	1.631	2.000	1.707	2.074	1.982	1.907
1.0000	1.907	1.611	1.544	1.804	1.569	1.796	1.818	1.833

TABLE [illegible] - PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(c) $\delta_f = 0.50\delta_w$; $\delta_f = 60^\circ$, $\delta_N = 25^\circ$; $q = 12.5$ lb/sq ft; $C_L = 0.023$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.333	1.199	1.182	1.236	.704	.117	.006	.025
.0927	1.327	1.199	1.170	1.224	.741	.239	.032	.000
.1852	1.333	1.205	1.188	1.188	.685	.350	.139	.074
.2786	1.345	1.193	1.194	1.133	.617	.368	.190	.136
.3706	1.321	1.205	1.200	1.054	.580	.362	.215	.148
.5565	1.352	1.199	1.224	.891	.525	.325	.222	.185
.7425	1.352	1.205	1.224	.727	.401	.325	.215	.160
.9272	1.358	1.205	1.236	.624	.265	.209	.190	.148
Wing								
.1944	1.364	1.247	1.188	.558	.401	.301	.234	.191
.2500	1.389	1.253	1.151	.539	.469	.380	.291	.272
.3057	1.413	1.277	1.121	.545	.506	.436	.354	.296
.3612	1.438	1.307	1.091	.533	.494	.448	.380	.321
.4444	1.463	1.319	1.030	.503	.475	.436	.354	.358
.5278	1.481	1.337	.964	.436	.407	.393	.329	.333
.6115	1.457	1.361	.925	.358	.333	.301	.228	.253
.6678	1.420	1.331	.861	.321	.265	.251	.184	.179
.7094	1.383	1.301	.824	.309	.247	.215	.177	.154
.7372	1.339	1.271	.770	.285	.247	.221	.158	.173
.7647	1.296	1.193	.697	.291	.272	.233	.203	.173
.7925	1.259	1.175	.691	.291	.302	.239	.196	.185
Flap								
.0582	1.247	1.181	.739	.315	.284	.258	.190	.179
.1306	1.259	1.223	.800	.351	.290	.233	.171	.179
.2758	1.358	1.205	.806	.218	.185	.129	.127	.111
.4177	1.352	1.229	.624	.248	.173	.129	.070	.117
.5600	1.345	1.259	.594	.255	.210	.209	.101	.136
.7019	1.407	1.235	.558	.351	.309	.331	.177	.228
.8527	1.241	1.133	.782	.442	.395	.399	.266	.370
1.0000	1.574	1.392	1.406	1.770	1.950	1.981	.747	.734

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.308	1.223	1.224	1.224	.296	2.730	5.506	5.123
.0453	.012	.012	.042	.359	1.142	2.497	4.202	4.715
.0927	.000	.054	.182	.551	1.234	2.276	4.221	4.728
.1852	.056	.181	.370	.739	1.308	2.079	4.253	4.765
.2786	.167	.307	.509	.891	1.395	2.012	3.835	4.789
.3706	.259	.410	.636	1.030	1.494	2.055	3.177	4.796
.5565	.451	.639	.867	1.260	1.716	2.214	2.614	4.401
.7425	.704	.880	1.151	1.594	2.031	2.521	2.835	3.456
.9272	1.160	1.398	1.721	2.266	2.864	3.441	3.664	3.259
Wing								
.1944	1.210	1.355	1.570	1.903	2.277	2.613	2.924	2.907
.2500	1.210	1.301	1.473	1.757	2.031	2.288	2.544	2.543
.3057	1.259	1.337	1.467	1.721	1.957	2.165	2.399	2.401
.3612	1.321	1.373	1.485	1.721	1.950	2.116	2.297	2.290
.4444	1.420	1.458	1.582	1.824	1.994	2.135	2.304	2.234
.5278	1.685	1.782	1.909	2.091	2.278	2.309	2.585	2.574
.6115	1.889	1.927	2.000	2.291	2.518	2.509	2.698	2.648
.6678	1.963	2.018	2.200	2.364	2.778	2.818	2.906	2.815
.7094	2.241	2.218	2.309	2.582	2.796	2.782	3.151	2.907
.7372	2.444	2.454	2.545	2.854	3.018	3.018	3.453	3.185
.7647	2.870	2.818	2.945	3.273	3.518	3.382	4.132	3.537
.7925	4.111	4.000	4.182	4.654	4.852	4.745	5.887	4.759
Flap								
-.0420	13.518	12.999	13.345	14.236	14.926	13.999	17.301	13.759
-.0350	15.666	14.981	15.417	16.472	17.240	16.145	20.207	15.759
-.0240	16.111	15.708	16.145	17.272	18.037	16.908	21.433	16.796
-.0130	15.851	15.472	15.890	16.982	17.703	16.508	21.112	16.722
.0000	10.259	9.963	10.327	11.054	11.518	10.854	14.018	10.870
.0582	3.537	3.418	3.509	3.963	4.130	3.963	5.415	4.222
.1306	2.463	2.400	2.545	2.873	3.000	2.873	3.792	3.093
.2758	1.907	1.818	1.927	2.218	2.389	2.291	2.528	2.130
.4177	1.704	1.654	1.818	2.145	2.278	2.254	1.906	1.685
.5600	1.741	1.745	1.891	2.364	2.333	2.364	1.528	1.574
.7019	1.833	1.836	2.000	2.327	2.370	2.382	1.283	1.537
.8527	1.648	1.709	1.818	2.236	2.352	2.345	1.094	1.278
1.0000	1.630	1.654	1.527	2.036	2.037	2.000	.472	.926

TABLE V.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(d) $c_f = 0.20c_w$; $\delta_f = 60°$; $\delta_N = 25°$; $q = 12.5$ lb/sq ft, $C_\mu = 0.038$

Lower surface

x/c	C_p for -							
	$\alpha = -16°$	$\alpha = -12°$	$\alpha = -8°$	$\alpha = -4°$	$\alpha = 0°$	$\alpha = 4°$	$\alpha = 8°$	$\alpha = 12°$
Nose								
.0453	1.127	1.017	1.077	1.292	.441	.037	.012	.042
.0927	1.133	1.029	1.071	1.123	.528	.166	.024	.006
.1852	1.139	1.029	1.083	.906	.553	.270	.120	.066
.2786	1.145	1.040	1.113	.783	.516	.325	.169	.120
.3706	1.133	1.048	1.101	.684	.484	.319	.187	.157
.5565	1.164	1.040	1.137	.573	.441	.307	.181	.169
.7425	1.176	1.052	1.137	.474	.354	.282	.169	.157
.9272	1.176	1.063	1.107	.392	.224	.245	.181	.157
Wing								
.1944	1.188	1.103	1.089	.550	.335	.294	.205	.211
.2500	1.206	1.144	.988	.585	.410	.356	.283	.259
.3057	1.236	1.167	.899	.567	.435	.399	.307	.301
.3612	1.273	1.213	.827	.544	.441	.368	.337	.337
.4444	1.370	1.322	.792	.491	.416	.380	.343	.331
.5278	1.442	1.362	.744	.421	.354	.331	.301	.307
.6115	1.448	1.351	.708	.345	.267	.264	.229	.265
.6678	1.400	1.282	.655	.298	.230	.221	.169	.205
.7094	1.327	1.184	.601	.281	.217	.215	.169	.181
.7372	1.230	1.092	.565	.304	.224	.196	.157	.181
.7647	1.139	.937	.524	.292	.224	.202	.169	.175
.7925	1.091	.874	.524	.316	.217	.233	.175	.193
Flap								
.0582	1.054	.902	.595	.327	.230	.202	.169	.193
.1306	1.067	.914	.661	.363	.230	.202	.187	.181
.2758	1.194	.960	.393	.222	.143	.135	.096	.133
.4177	1.224	1.006	.434	.181	.143	.086	.060	.108
.5600	1.218	1.052	.470	.181	.124	.129	.102	.157
.7019	1.236	.994	.423	.205	.168	.184	.169	.229
.8527	1.188	.891	.673	.409	.267	.245	.247	.361
1.0000	1.091	.753	.554	.503	.528	.509	.530	.633

Upper surface

x/c	C_p for -							
	$\alpha = -16°$	$\alpha = -12°$	$\alpha = -8°$	$\alpha = -4°$	$\alpha = 0°$	$\alpha = 4°$	$\alpha = 8°$	$\alpha = 12°$
Nose								
.0000	1.139	1.063	1.161	1.672	.155	4.674	5.072	5.096
.0453	.030	.011	.119	.427	1.602	3.239	4.108	4.711
.0927	.000	.098	.274	.655	1.540	2.736	4.175	4.735
.1852	.085	.247	.440	.830	1.522	2.380	4.265	4.771
.2786	.218	.385	.577	.982	1.571	2.282	4.120	4.789
.3706	.303	.488	.708	1.123	1.683	2.270	3.578	4.807
.5565	.521	.707	.929	1.351	1.582	2.405	2.765	4.416
.7425	.776	.948	1.232	1.661	2.224	2.699	2.789	3.506
.9272	1.267	1.488	1.827	2.421	3.081	3.662	3.542	3.247
Wing								
.1944	1.303	1.454	1.631	2.011	2.416	2.773	2.904	2.946
.2500	1.285	1.397	1.530	1.836	2.180	2.441	2.542	2.608
.3057	1.327	1.425	1.542	1.824	2.093	2.313	2.379	2.434
.3612	1.394	1.488	1.589	1.848	2.068	2.282	2.289	2.337
.4444	1.557	1.592	1.708	1.930	2.149	2.319	2.307	2.277
.5278	2.036	1.862	2.071	2.281	2.555	2.673	2.545	2.436
.6115	2.327	2.089	2.304	2.544	2.833	2.927	2.692	2.509
.6678	2.636	2.293	2.554	2.807	3.105	3.200	2.784	2.691
.7094	2.927	2.586	2.357	3.105	3.444	3.436	3.236	2.954
.7372	3.345	2.896	3.179	3.491	3.833	3.836	3.600	3.127
.7647	4.054	3.569	3.875	4.228	4.555	4.618	4.145	3.618
.7925	6.091	5.483	5.750	6.333	6.666	6.782	6.000	4.873
Flap								
-.0420	21.017	19.034	19.678	20.666	21.814	21.799	19.126	15.908
-.0350	24.690	22.206	22.982	24.227	25.555	25.526	22.126	17.908
-.0240	26.290	23.655	24.464	25.771	27.147	27.144	23.526	19.093
-.0130	26.799	24.034	24.786	26.016	27.333	27.362	23.744	19.363
.0000	17.236	15.310	15.857	16.859	17.703	17.636	15.217	12.272
.0582	6.072	5.310	5.518	5.930	6.315	6.236	5.454	4.327
.1306	4.345	3.690	3.875	4.123	4.444	4.436	3.873	3.091
.2758	3.054	2.500	2.589	2.877	3.037	3.000	2.527	2.127
.4177	2.400	1.897	2.071	2.158	2.296	2.292	1.982	1.709
.5600	2.054	1.534	1.714	1.807	1.815	1.964	1.564	1.454
.7019	1.927	1.448	1.571	1.561	1.593	1.654	1.418	1.491
.8527	1.545	1.086	1.161	1.105	1.167	.982	1.236	1.182
1.0000	1.164	.776	.679	.702	.848	.545	.619	.818

TABLE 7 - PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(e) $c_f = 0.20c_w$; $\delta_f = 60^\circ$, $\delta_N = 25^\circ$, q = 12.6 lb/sq ft; $C_L = 0.063$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.074	1.036	1.096	1.204	.339	.032	.019	.067
.0927	1.074	1.061	1.096	1.041	.469	.117	.025	.012
.1852	1.063	1.054	1.102	.865	.494	.247	.089	.073
.2786	1.074	1.054	1.120	.778	.469	.286	.153	.109
.3706	1.086	1.054	1.139	.672	.457	.292	.197	.133
.5565	1.091	1.079	1.145	.567	.395	.292	.210	.188
.7425	1.114	1.079	1.139	.450	.333	.273	.191	.164
.9272	1.114	1.097	1.108	.357	.210	.273	.178	.164
Wing								
.1944	1.131	1.151	1.048	.479	.333	.273	.210	.206
.2500	1.160	1.151	.958	.497	.395	.377	.287	.279
.3057	1.189	1.230	.831	.485	.407	.383	.312	.321
.3612	1.217	1.273	.747	.485	.420	.390	.357	.321
.4444	1.326	1.364	.699	.444	.383	.370	.344	.351
.5278	1.412	1.400	.633	.368	.333	.351	.293	.321
.6115	1.411	1.418	.584	.310	.247	.266	.236	.255
.6678	1.394	1.309	.512	.275	.198	.195	.178	.206
.7094	1.286	1.224	.488	.269	.185	.195	.172	.206
.7372	1.194	1.121	.458	.246	.173	.195	.178	.206
.7647	1.074	.957	.446	.251	.173	.208	.185	.188
.7925	1.011	.957	.458	.310	.210	.234	.210	.182
Flap								
.0582	.994	1.012	.476	.281	.185	.214	.185	.194
.1306	1.034	.951	.512	.287	.185	.188	.159	.200
.2758	1.120	.988	.343	.187	.123	.104	.076	.133
.4177	1.189	1.054	.319	.158	.074	.078	.064	.097
.5600	1.240	1.085	.373	.146	.099	.091	.089	.127
.7019	1.211	.994	.349	.170	.154	.149	.153	.206
.8527	1.120	1.024	.572	.263	.179	.195	.242	.315
1.0000	.983	.794	.518	.427	.426	.435	.452	.582

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.080	1.097	1.120	1.292	.383	6.675	5.477	5.193
.0453	.034	.018	.145	.485	1.765	3.974	4.420	4.512
.0927	.057	.115	.301	.707	1.666	3.198	4.497	4.848
.1852	.160	.255	.482	.854	1.611	2.649	4.630	4.878
.2786	.263	.388	.626	1.000	1.654	2.467	4.573	4.890
.3706	.354	.485	.735	1.109	1.722	2.428	4.191	4.939
.5565	.537	.709	1.048	1.374	1.926	2.578	3.178	4.800
.7425	.800	1.006	1.373	1.690	2.216	2.863	3.013	3.951
.9272	1.251	1.563	2.018	2.415	3.135	3.883	3.681	3.381
Wing								
.1944	1.303	1.473	1.759	2.041	2.469	2.961	3.095	2.994
.2500	1.280	1.418	1.651	1.859	2.210	2.571	2.713	2.685
.3057	1.337	1.460	1.675	1.842	2.111	2.454	2.548	2.503
.3612	1.383	1.515	1.699	1.848	2.105	2.389	2.458	2.388
.4444	1.531	1.660	1.831	1.953	2.166	2.428	2.484	2.351
.5278	1.897	1.964	2.218	2.298	2.593	2.923	2.750	2.545
.6115	2.172	2.254	2.473	2.631	2.833	3.115	2.961	2.727
.6678	2.621	2.527	2.764	2.877	3.148	3.481	3.250	2.873
.7094	2.828	2.518	3.034	3.175	3.481	3.731	3.558	3.054
.7372	3.103	3.254	3.491	3.594	3.852	4.192	4.019	3.382
.7647	3.896	4.018	4.200	4.280	4.667	4.981	4.711	3.763
.7925	6.017	6.391	6.403	6.456	7.000	7.404	6.750	5.272
Flap								
-.0420	23.724	24.435	24.835	25.051	26.129	27.249	25.172	23.072
-.0350	26.689	27.399	27.962	28.139	29.462	30.576	27.749	21.399
-.0240	28.396	29.071	29.580	30.034	31.332	32.614	29.595	22.763
-.0130	29.206	30.090	30.417	30.858	32.184	33.383	30.345	23.472
.0000	19.086	18.417	18.745	19.087	19.999	20.730	18.788	14.199
.0582	5.793	5.872	5.927	6.140	6.518	6.865	6.134	4.454
.1306	4.103	4.091	4.109	4.280	4.595	4.769	4.346	3.309
.2758	2.931	2.636	2.873	2.842	3.111	3.250	2.961	2.218
.4177	2.310	2.127	2.236	2.228	2.407	2.538	2.288	1.818
.5600	1.931	1.654	1.982	1.824	1.944	2.038	1.808	1.564
.7019	1.800	1.430	1.854	1.550	1.600	1.700	1.650	1.400
.8527	1.448	1.218	1.109	1.193	1.315	1.404	1.404	1.291
1.0000	1.103	.782	.600	.544	.611	.808	.788	.727

TABLE V.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(f) $c_f = 0.20c_w$; $\delta_f = 60°$; $\delta_N = 25°$; $q = 12.5$ lb/sq ft; $C_\mu = 0.121$

Lower surface

x/c	C_p for -							
	$\alpha = -16°$	$\alpha = -12°$	$\alpha = -8°$	$\alpha = -4°$	$\alpha = 0°$	$\alpha = 4°$	$\alpha = 8°$	$\alpha = 12°$
Nose								
.0453	1.104	.970	1.042	1.018	.279	.030	.039	.049
.0927	1.110	.970	1.042	.934	.376	.077	.013	.012
.1852	1.116	.970	1.024	.825	.455	.202	.091	.049
.2786	1.123	.970	1.059	.741	.424	.244	.137	.086
.3706	1.129	1.000	1.071	.633	.430	.268	.157	.105
.5565	1.141	1.000	1.083	.536	.388	.262	.196	.142
.7425	1.147	1.030	1.048	.428	.382	.220	.183	.148
.9272	1.172	1.030	1.000	.301	.242	.238	.170	.142
Wing								
.1944	1.178	1.089	.923	.404	.358	.244	.209	.185
.2500	1.190	1.119	.798	.464	.394	.286	.288	.235
.3057	1.221	1.125	.696	.488	.442	.315	.288	.278
.3612	1.325	1.167	.613	.464	.418	.339	.314	.272
.4444	1.454	1.232	.536	.434	.406	.315	.301	.309
.5278	1.527	1.298	.464	.355	.376	.304	.288	.278
.6115	1.521	1.244	.405	.259	.279	.208	.229	.210
.6678	1.441	1.143	.375	.247	.230	.196	.170	.179
.7094	1.319	1.030	.363	.259	.230	.173	.190	.160
.7372	1.202	.934	.351	.247	.224	.155	.170	.142
.7647	1.012	.792	.369	.259	.248	.179	.196	.179
.7925	.963	.750	.349	.283	.255	.220	.209	.192
Flap								
.0582	.939	.738	.357	.259	.267	.232	.183	.185
.1306	.939	.768	.369	.247	.224	.185	.150	.179
.2758	1.086	.815	.268	.151	.158	.101	.046	.099
.4177	1.172	.875	.244	.114	.109	.065	.059	.086
.5600	1.233	.857	.262	.108	.109	.089	.105	.117
.7019	1.208	.792	.262	.151	.182	.125	.144	.160
.8527	1.000	.768	.393	.199	.230	.179	.190	.259
1.0000	.834	.583	.393	.404	.436	.399	.405	.537

Upper surface

x/c	C_p for -							
	$\alpha = -16°$	$\alpha = -12°$	$\alpha = -8°$	$\alpha = -4°$	$\alpha = 0°$	$\alpha = 4°$	$\alpha = 8°$	$\alpha = 12°$
Nose								
.0000	1.098	1.030	1.059	.892	1.127	5.035	6.025	5.318
.0453	.004	.006	.214	.717	2.169	4.863	4.888	4.802
.0927	.043	.149	.381	.898	1.939	4.214	4.993	4.827
.1852	.186	.304	.589	1.060	1.806	3.053	5.130	4.839
.2786	.288	.434	.738	1.157	1.782	2.613	5.163	4.888
.3706	.368	.542	.869	1.277	1.854	2.482	4.895	4.827
.5565	.589	.780	1.143	1.542	2.073	2.583	3.816	4.357
.7425	.853	1.030	1.417	1.861	2.388	2.875	3.372	3.746
.9272	1.386	1.601	2.131	2.705	3.297	3.797	3.888	3.222
Wing								
.1944	1.423	1.530	1.905	2.205	2.606	2.952	3.366	2.901
.2500	1.386	1.482	1.774	2.012	2.315	2.589	2.993	2.703
.3057	1.435	1.524	1.792	1.976	2.242	2.464	2.810	2.556
.3612	1.503	1.577	1.821	2.012	2.236	2.434	2.725	2.512
.4444	1.675	1.708	1.964	2.108	2.309	2.488	2.751	2.500
.5278	2.000	1.982	2.393	2.600	2.618	2.732	2.941	2.481
.6115	2.400	2.339	2.714	2.909	3.036	2.911	3.274	2.611
.6678	2.654	2.625	2.946	3.236	3.309	3.411	3.568	2.796
.7094	3.036	2.982	3.339	3.582	3.673	3.732	3.941	3.037
.7372	3.491	3.411	3.857	4.109	4.145	4.125	4.372	3.370
.7647	4.345	4.161	4.661	4.963	5.018	5.018	5.372	3.907
.7925	6.672	6.411	7.179	7.509	7.563	7.589	7.764	5.444
Flap								
-.0420	35.889	34.768	34.232	37.217	37.598	34.732	38.077	29.629
-.0350	36.144	34.928	34.910	37.944	38.416	37.482	38.214	28.351
-.0240	37.998	36.732	39.018	40.216	40.671	39.910	40.586	29.942
-.0130	40.416	39.160	41.482	42.616	43.089	42.285	43.096	32.035
.0000	22.908	22.053	23.643	24.417	24.653	24.286	24.826	17.629
.0582	6.218	5.893	6.554	6.854	6.872	6.875	7.020	4.630
.1306	4.363	4.089	4.518	4.782	4.800	4.732	4.941	3.407
.2758	3.000	2.679	3.179	3.273	3.309	3.179	3.451	2.407
.4177	2.418	2.143	2.482	2.600	2.582	2.536	2.686	1.963
.5600	2.109	1.714	2.018	2.054	2.184	2.161	2.274	1.907
.7019	1.820	1.500	1.600	1.700	1.750	1.800	1.900	1.800
.8527	1.309	1.232	1.375	1.382	1.418	1.196	1.471	1.593
1.0000	1.164	.750	.661	.655	.673	.554	.647	.685

TABLE V.- PRESSURE COEFFICIENT C_p ON THE NOSE, WING, AND FLAP THROUGH THE ANGLE-OF-ATTACK RANGE - Concluded

(g) $c_f = 0.20c_w$; $\delta_f = 60^\circ$, $\delta_N = 20^\circ$; $q = 12.5$ lb/sq ft, $C_\mu = 0.189$

Lower surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0453	1.054	.956	.994	.783	.099	.000	.082	.082
.0927	1.042	.938	.994	.748	.217	.043	.013	.019
.1852	1.059	.975	1.012	.678	.323	.167	.063	.075
.2786	1.065	.975	.994	.620	.348	.216	.127	.113
.3706	1.071	.988	1.012	.567	.348	.241	.146	.119
.5565	1.113	1.012	1.012	.456	.311	.216	.177	.170
.7425	1.095	1.019	.994	.357	.292	.191	.177	.164
.9272	1.113	1.037	.921	.269	.174	.204	.171	.145
Wing								
.1944	1.155	1.118	.836	.363	.292	.253	.234	.214
.2500	1.167	1.155	.673	.421	.323	.333	.247	.264
.3057	1.202	1.174	.588	.433	.360	.346	.291	.308
.3612	1.286	1.193	.533	.421	.360	.339	.316	.333
.4444	1.440	1.248	.491	.398	.329	.339	.304	.314
.5278	1.530	1.248	.424	.322	.286	.302	.253	.302
.6115	1.480	1.231	.345	.234	.261	.278	.228	.239
.6678	1.357	1.106	.309	.240	.286	.216	.165	.189
.7094	1.244	.975	.291	.234	.186	.191	.158	.164
.7372	1.113	.863	.285	.246	.199	.198	.152	.164
.7647	.929	.733	.291	.269	.224	.204	.203	.176
.7925	.893	.683	.291	.257	.211	.228	.203	.176
Flap								
.0582	.893	.764	.327	.257	.242	.216	.228	.208
.1306	.599	.758	.333	.251	.174	.179	.158	.176
.2758	.994	.705	.230	.152	.106	.074	.063	.094
.4177	1.077	.752	.236	.082	.099	.080	.063	.082
.5600	1.113	.795	.218	.099	.099	.111	.139	.113
.7019	1.089	.683	.242	.117	.149	.123	.158	.182
.8527	.976	.520	.285	.152	.168	.278	.203	.252
1.0000	.667	.540	.448	.392	.404	.407	.392	.528

Upper surface

x/c	C_p for -							
	$\alpha = -16^\circ$	$\alpha = -12^\circ$	$\alpha = -8^\circ$	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$
Nose								
.0000	1.083	1.075	.888	.444	2.043	5.117	5.747	4.931
.0453	.036	.081	.297	.930	2.310	4.092	5.392	4.692
.0927	.077	.158	.497	1.052	2.174	4.166	5.443	4.698
.1852	.196	.366	.691	1.169	1.988	4.104	5.491	4.704
.2786	.310	.491	.867	1.298	1.988	3.512	5.519	4.723
.3706	.411	.615	.982	1.380	2.000	2.876	5.468	4.740
.5565	.631	.876	1.273	1.625	2.261	2.623	4.677	4.742
.7425	.887	1.161	1.594	1.953	2.584	2.929	3.557	4.497
.9272	1.411	1.764	2.327	2.760	3.553	3.839	3.715	4.006
Wing								
.1944	1.434	1.633	2.060	2.292	2.807	3.211	3.605	3.472
.2500	1.417	1.584	1.897	2.082	2.509	2.722	3.019	3.157
.3057	1.452	1.633	1.921	2.046	2.410	2.580	2.823	2.931
.3612	1.536	1.689	1.976	2.070	2.379	2.524	2.797	2.780
.4444	1.684	1.832	2.091	2.175	2.478	2.605	2.797	2.635
.5278	1.929	2.185	2.309	2.544	2.759	2.815	2.943	2.849
.6115	2.214	2.444	2.618	2.842	2.944	3.148	3.302	2.962
.6678	2.446	2.667	2.891	3.175	3.315	3.407	3.472	3.019
.7094	2.929	3.148	3.291	3.509	3.648	3.889	3.962	3.283
.7372	3.446	3.633	3.927	4.140	4.278	4.407	4.509	3.566
.7647	4.250	4.481	4.909	5.017	5.278	5.370	5.434	4.132
.7925	6.732	7.237	7.600	7.666	8.166	8.259	8.169	5.641
Flap								
-.0420	51.589	53.665	54.634	53.068	55.980	55.832	54.431	49.488
-.0350	44.982	46.739	48.071	47.085	50.017	49.591	48.356	37.376
-.0240	46.017	47.925	49.907	48.910	52.110	51.721	50.054	38.696
-.0130	51.392	53.628	55.888	54.664	58.128	57.665	55.601	42.658
.0000	25.893	26.888	28.344	27.928	29.981	29.666	28.980	21.225
.0582	6.107	6.296	6.782	6.912	7.241	7.389	7.169	4.736
.1306	4.214	4.426	4.727	4.807	5.055	5.074	5.019	3.622
.2758	2.921	2.889	3.127	3.158	3.463	3.370	3.472	2.472
.4177	2.268	2.389	2.636	2.631	2.704	2.685	2.792	2.283
.5600	1.982	1.963	1.982	2.070	2.241	2.093	2.207	2.094
.7019	1.700	1.650	1.700	1.700	1.800	1.800	1.800	1.900
.8527	1.286	1.426	1.491	1.298	1.593	1.481	1.525	1.679
1.0000	.929	.685	.782	.614	.944	.444	.698	.943

TABLE VI.- INTEGRATED SECTION DATA

$c_f = 0.20c_w$; $\delta_f = 60^o$; $\delta_N = 0^o$; $q \approx 12.5$ lb/sq ft

α, deg	$c_{N,w}$	$c_{m,w}$	$c_{N,N}$	$c_{m,N}$	$c_{N,f}$	$c_{m,f}$
$C_\mu = 0$						
-12	.056	-.191	-1.019	.607	.999	-.486
-8	.542	-.285	-.347	.304	1.384	-.625
-4	.772	-.263	.549	-.289	1.390	-.614
0	1.012	-.258	1.404	-.869	1.431	-.630
4	1.319	-.243	2.153	-1.108	1.378	-.611
8	1.505	-.318	1.910	-.999	1.573	-.675
$C_\mu = 0.013$						
-12	.426	-.308	-.959	.628	1.401	-.520
-8	.764	-.331	-.075	.136	1.464	-.566
-4	1.043	-.344	.792	-.443	1.663	-.611
0	1.461	-.370	2.067	-1.204	1.972	-.696
4	1.678	-.349	2.416	-1.243	2.085	-.711
8	1.598	-.360	1.851	-.962	1.650	-.674
$C_\mu = 0.026$						
-12	1.289	-.548	.009	.112	3.121	-.634
-8	1.566	-.566	.776	-.411	3.298	-.678
-4	1.784	-.532	1.769	-1.073	3.118	-.644
0	1.899	-.449	2.772	-1.441	2.771	-.524
4	1.824	-.374	2.463	-1.263	2.082	-.526
8	1.459	-.321	1.740	-.902	1.552	-.588
$C_\mu = 0.037$						
-12	1.340	-.568	.104	.042	3.477	-.746
-8	1.637	-.578	.945	-.513	3.476	-.688
-4	1.853	-.547	1.892	-1.142	3.295	-.655
0	2.223	-.568	2.909	-1.488	3.468	-.708
4	1.824	-.378	2.409	-1.237	2.268	-.600
8	1.481	-.333	1.731	-.897	1.682	-.632

α, deg	$c_{N,w}$	$c_{m,w}$	$c_{N,N}$	$c_{m,N}$	$c_{N,f}$	$c_{m,f}$
$C_\mu = 0.060$						
-12	1.437	-.593	.216	-.028	3.760	-.742
-8	1.847	-.654	1.051	-.580	4.026	-.789
-4	2.029	-.613	2.089	-1.235	3.885	-.772
0	2.177	-.522	2.965	-1.513	3.419	-.549
4	2.079	-.466	2.566	-1.310	2.795	-.681
8	1.657	-.387	1.883	-.976	2.095	-.757
$C_\mu = 0.115$						
-12	1.760	-.733	.309	-.086	4.804	-.844
-8	2.037	-.730	1.231	-.685	4.846	-.868
-4	2.246	-.684	2.319	-1.292	4.578	-.844
0	2.912	-.776	3.484	-1.768	5.254	-.961
4	2.244	-.509	2.694	-1.377	3.280	-.719
$C_\mu = 0.179$						
-12	1.881	-.763	.510	-.215	5.389	-.884
-8	2.240	-.796	1.430	-.818	5.634	-.907
-4	2.387	-.721	2.621	-1.421	5.241	-.839
0	2.671	-.694	3.281	-1.657	5.270	-.813
4	2.357	-.554	2.760	-1.412	3.967	-.738
$C_\mu = 0.241$						
-12	1.956	-.776	.678	-.320	5.731	-.875
-8	2.350	-.826	1.587	-.919	6.053	-.945
-4	2.573	-.783	2.788	-1.481	5.876	-.933
0	3.140	-.853	3.632	-1.838	6.583	-1.019
4	2.590	-.623	2.945	-1.502	4.785	-.806

TABLE VII.- INTEGRATED SECTION DATA

$c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 4.75^\circ$; $q \approx 12.5$ lb/sq ft

α, deg	$c_{N,w}$	$c_{m,w}$	$c_{N,N}$	$c_{m,N}$	$c_{N,f}$	$c_{m,f}$
$C_\mu = 0$						
-16	-.492	-.076	-1.341	.773	.771	-.448
-12	-.202	-.142	-1.034	.621	.971	-.530
-8	.435	-.265	-.600	.504	1.341	-.648
-4	.772	-.285	.328	-.081	1.502	-.705
0	1.073	-.286	1.275	-.716	1.600	-.725
4	1.310	-.263	2.403	-1.307	1.542	-.705
8	1.674	-.326	2.404	-1.235	1.836	-.829
12	1.442	-.352	1.600	-.830	1.740	-.776
$C_\mu = 0.013$						
-16	-.171	-.165	-1.103	.654	1.082	-.428
-12	.154	-.204	-.679	.435	1.126	-.443
-8	.705	-.320	-.240	.291	1.504	-.605
-4	1.024	-.346	.617	-.260	1.766	-.700
0	1.303	-.336	1.595	-.914	1.834	-.673
4	1.612	-.322	2.764	-1.420	1.816	-.608
8	1.771	-.355	2.343	-1.201	1.830	-.701
12	1.470	-.359	1.631	-.845	1.814	-.766
$C_\mu = 0.026$						
-16	.658	-.388	-.480	.341	2.471	-.414
-12	1.283	-.562	-.199	.306	3.191	-.650
-8	1.594	-.584	.639	-.251	3.362	-.675
-4	1.862	-.586	1.490	-.796	3.383	-.682
0	2.069	-.544	2.610	-1.423	3.153	-.667
4	2.036	-.447	2.885	-1.461	2.614	-.579
8	1.550	-.305	2.094	-1.079	1.578	-.599
12	1.467	-.363	1.633	-.847	1.861	-.767
$C_\mu = 0.037$						
-16	.765	-.429	-.446	.327	2.792	-.465
-12	1.322	-.569	-.077	.215	3.392	-.685
-8	1.705	-.629	.713	-.303	3.757	-.770
-4	1.951	-.607	1.614	-.888	3.618	-.699
0	2.083	-.538	2.757	-1.491	3.234	-.650
4	2.114	-.471	2.869	-1.482	2.821	-.631
8	1.693	-.344	2.217	-1.145	1.818	-.667
12	1.402	-.340	1.569	-.810	1.759	-.714

α, deg	$c_{N,w}$	$c_{m,w}$	$c_{N,N}$	$c_{m,N}$	$c_{N,f}$	$c_{m,f}$
$C_\mu = 0.064$						
-16	.904	-.485	-.433	.317	3.256	-.546
-12	1.416	-.592	.044	.137	3.718	-.707
-8	1.778	-.644	.837	-.376	4.060	-.787
-4	1.983	-.615	1.733	-.980	3.910	-.754
0	2.192	-.574	2.875	-1.533	3.676	-.706
4	2.286	-.532	2.941	-1.490	3.329	-.712
8	1.754	-.368	2.206	-1.127	2.032	-.701
12	1.577	-.390	1.711	-.885	2.147	-.815
$C_\mu = 0.117$						
-16	1.143	-.582	-.402	.304	4.077	-.637
-12	1.678	-.692	.169	.066	4.615	-.775
-8	1.981	-.713	1.042	-.511	4.722	-.827
-4	2.238	-.693	2.055	-1.202	4.677	-.835
0	2.511	-.681	3.160	-1.657	4.572	-.829
4	2.368	-.556	2.994	-1.516	3.798	-.701
8	1.872	-.407	2.252	-1.146	2.518	-.756
12	1.696	-.418	1.839	-.948	2.494	-.825
$C_\mu = 0.187$						
-16	1.339	-.659	-.375	.301	4.855	-.684
-12	1.761	-.722	.276	-.087	5.164	-.798
-8	2.108	-.752	1.191	-.606	5.312	-.884
-4	2.390	-.743	2.231	-1.290	5.394	-.859
0	2.694	-.732	3.364	-1.719	5.333	-.846
4	2.703	-.655	3.258	-1.649	4.792	-.770
8	1.980	-.429	2.376	-1.209	2.832	-.677
12	2.071	-.513	2.243	-1.144	3.311	-.923
$C_\mu = 0.251$						
-16	1.594	-.761	-.418	.343	5.577	-.846
-12	2.052	-.836	.444	-.109	6.090	-.951
-8	2.269	-.811	1.326	-.677	5.928	-.911
-4	2.643	-.831	2.445	-1.423	6.256	-.957
0	2.938	-.812	3.534	-1.790	6.176	-.977
4	2.750	-.682	3.234	-1.632	5.215	-.823
8	2.208	-.515	2.513	-1.281	3.694	-.845
12	2.105	-.536	2.219	-1.129	3.702	-.960

TABLE VIII. - INTEGRATED SECTION DATA

$c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 10^\circ$; q ≈ 12.5 lb/sq ft

α, deg	$c_{N,w}$	$c_{m,w}$	$c_{N,N}$	$c_{m,N}$	$c_{N,f}$	$c_{m,f}$
$C_\mu = 0$						
-16	-.471	-.028	-1.146	.660	.396	-.296
-12	-.307	-.090	-1.051	.635	.785	-.444
-8	.256	-.214	-.616	.430	1.102	-.556
-4	.748	-.276	.225	.084	1.428	-.675
0	.994	-.275	1.023	-.438	1.470	-.683
4	1.273	-.267	1.988	-1.090	1.508	-.673
8	1.581	-.276	2.867	-1.419	1.651	-.741
12	1.602	-.294	2.643	-1.349	1.691	-.764
$C_\mu = 0.012$						
-16	-.355	-.062	-1.092	.639	.465	-.252
-12	-.013	-.174	-.834	.529	1.005	-.445
-8	.676	-.346	-.409	.339	1.818	-.660
-4	1.013	-.353	.486	-.070	1.816	-.663
0	1.382	-.381	1.413	-.666	2.041	-.745
4	1.711	-.387	2.569	-1.418	2.220	-.763
8	1.852	-.347	2.990	-1.470	2.037	-.636
12	1.595	-.341	2.204	-1.179	1.793	-.740
$C_\mu = 0.023$						
-16	.614	-.354	-.403	.320	2.385	-.403
-12	.890	-.435	-.285	.273	2.609	-.460
-8	1.580	-.604	.445	-.013	3.369	-.718
-4	1.734	-.540	1.319	-.587	3.140	-.612
0	1.952	-.538	2.110	-1.143	3.101	-.647
4	2.171	-.489	3.297	-1.713	2.854	-.610
8	2.042	-.393	3.057	-1.496	2.168	-.701
12	1.741	-.246	2.750	-1.430	1.977	-.796
$C_\mu = 0.038$						
-16	.698	-.397	-.396	.315	2.612	-.414
-12	1.212	-.566	-.291	.276	3.377	-.665
-8	1.645	-.611	.564	-.104	3.548	-.706
-4	2.136	-.688	1.552	-.719	4.043	-.829
0	2.052	-.545	2.385	-1.308	3.273	-.640
4	2.269	-.510	3.458	-1.748	3.112	-.641
8	1.965	-.368	2.931	-1.428	2.125	-.630
12	1.724	-.329	2.793	-1.440	1.939	-.735

α, deg	$c_{N,w}$	$c_{m,w}$	$c_{N,N}$	$c_{m,N}$	$c_{N,f}$	$c_{m,f}$
$C_\mu = 0.060$						
-16	.831	-.456	-.370	.309	3.179	-.524
-12	1.493	-.659	-.226	.314	3.909	-.746
-8	1.835	-.680	.687	-.168	4.195	-.846
-4	2.192	-.698	1.654	-.768	4.292	-.845
0	2.309	-.627	2.657	-1.464	3.921	-.758
4	2.461	-.587	3.506	-1.734	3.707	-.785
8	2.138	-.420	3.016	-1.471	2.434	-.724
12	1.712	-.393	2.079	-1.035	2.141	-.796
$C_\mu = 0.118$						
-16	1.064	-.546	-.298	.274	3.963	-.622
-12	1.616	-.683	-.037	.268	4.393	-.752
-8	1.898	-.685	.899	-.309	4.500	-.820
-4	2.252	-.714	1.791	-.864	4.712	-.841
0	2.459	-.663	2.934	-1.585	4.491	-.780
4	2.537	-.614	3.498	-1.722	4.144	-.765
8	2.213	-.453	2.981	-1.460	2.916	-.747
12	1.917	-.448	2.227	-1.104	2.609	-.837
$C_\mu = 0.188$						
-16	1.183	-.565	-.188	.209	4.277	-.601
-12	1.800	-.742	.170	.147	5.177	-.843
-8	2.097	-.761	1.034	-.392	5.296	-.866
-4	2.504	-.804	1.988	-.989	5.674	-.926
0	2.759	-.772	3.197	-1.720	5.588	-.905
4	2.929	-.717	3.861	-1.884	5.270	-.858
8	2.355	-.511	3.007	-1.446	3.584	-.816
$C_\mu = 0.245$						
-16	1.292	-.628	-.160	.204	4.988	-.642
-12	1.836	-.749	.296	.069	5.481	-.802
-8	2.199	-.785	1.215	-.503	5.755	-.896
-4	2.534	-.794	2.169	-1.109	5.852	-.868
0	2.834	-.793	3.359	-1.800	5.863	-.897
4	2.953	-.737	3.824	-1.866	5.587	-.864
8	2.678	-.610	3.260	-1.589	4.627	-.910

TABLE IX.- INTEGRATED SECTION DATA

$c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 20^\circ$; $q \approx 12.5$ lb/sq ft

α, deg	$c_{N,w}$	$c_{m,w}$	$c_{N,N}$	$c_{m,N}$	$c_{N,f}$	$c_{m,f}$
$C_\mu = 0$						
-16	-.575	-.007	-1.457	.825	.129	-.194
-12	-.431	-.042	-1.207	.723	.371	-.298
-8	-.103	-.122	-.801	.560	.756	-.443
-4	.460	-.225	-.182	.265	1.132	-.568
0	.971	-.295	.671	-.034	1.453	-.679
4	1.303	-.310	1.645	-.608	1.633	-.754
8	1.484	-.276	2.574	-1.234	1.570	-.722
12	1.723	-.251	3.631	-1.880	1.563	-.718
$C_\mu = 0.012$						
-16	-.088	-.141	-.970	.605	.950	-.220
-12	.091	-.183	-.743	.514	1.243	-.339
-8	.431	-.255	-.311	.240	1.565	-.479
-4	1.128	-.414	.215	.142	2.223	-.710
0	1.429	-.423	1.193	-.316	2.304	-.768
4	1.677	-.421	2.082	-.854	2.392	-.786
8	2.060	-.425	3.346	-1.701	2.436	-.824
12	1.980	-.319	3.801	-1.870	1.713	-.574
$C_\mu = 0.027$						
-16	.276	-.238	-.666	.478	1.507	-.326
-12	.624	-.327	-.234	.292	2.070	-.334
-8	1.289	-.515	.113	.157	2.898	-.561
-4	1.594	-.536	.882	-.123	3.102	-.851
0	1.921	-.550	1.833	-.670	3.191	-.920
4	2.145	-.530	2.742	-1.261	3.154	-.890
8	2.507	-.537	3.871	-1.988	2.992	-.640
12	2.026	-.336	3.780	-1.849	1.876	-.638
$C_\mu = 0.037$						
-16	.669	-.377	-.429	.374	2.567	-.411
-12	.728	-.366	-.178	.266	2.441	-.340
-8	1.435	-.571	.210	.124	3.311	-.625
-4	1.865	-.616	1.134	-.260	3.613	-.724
0	2.128	-.614	1.991	-.786	3.611	-.716
4	2.336	-.582	2.975	-1.451	3.428	-.680
8	2.356	-.478	3.830	-1.970	2.885	-.551
12	2.120	-.369	3.822	-1.858	2.162	-.886

α, deg	$c_{N,w}$	$c_{m,w}$	$c_{N,N}$	$c_{m,N}$	$c_{N,f}$	$c_{m,f}$
$C_\mu = 0.061$						
-12	.842	-.413	-.152	.257	2.866	-.427
-8	1.608	-.633	.257	.125	3.733	-.699
-4	2.059	-.678	1.305	-.351	4.066	-.783
0	2.293	-.660	2.196	-.880	4.061	-.781
4	2.366	-.599	2.996	-1.480	3.755	-.723
8	2.629	-.577	4.023	-2.071	3.542	-.700
12	2.016	-.343	3.618	-1.753	2.078	-.635
$C_\mu = 0.119$						
-16	.889	-.470	-.320	.332	3.599	-.470
-12	1.207	-.547	.007	.194	3.875	-.571
-8	1.868	-.705	.508	.090	4.541	-.789
-4	2.241	-.736	1.522	-.471	4.811	-.848
0	2.609	-.768	2.485	-1.039	5.025	-.889
4	2.765	-.711	3.522	-1.775	4.744	-.841
8	2.825	-.642	4.370	-2.198	4.288	-.780
$C_\mu = 0.186$						
-16	1.013	-.529	-.289	.318	4.297	-.548
-12	1.502	-.669	.020	.201	4.896	-.709
-8	2.040	-.770	.714	-.008	5.338	-.842
-4	2.437	-.807	1.700	-.571	5.629	-.920
0	2.763	-.820	2.642	-1.146	5.698	-.937
4	3.053	-.818	3.808	-1.933	5.855	-.993
8	3.123	-.728	4.703	-2.305	5.181	-.841
$C_\mu = 0.250$						
-16	1.064	-.549	-.235	.293	4.663	-.510
-12	1.637	-.724	.036	.193	5.390	-.788
-8	2.220	-.825	.851	-.091	5.894	-.867
-4	2.491	-.818	1.775	-.632	5.943	-.873
0	2.771	-.883	2.293	-.916	6.485	-.977
4	3.086	-.822	3.824	-1.948	6.096	-.906
8	3.388	-.817	4.862	-2.342	6.089	-.903

TABLE X.- INTEGRATED SECTION DATA

$c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 25^\circ$; $q \approx 12.5$ lb/sq ft

α, deg	$c_{N,w}$	$c_{m,w}$	$c_{N,N}$	$c_{m,N}$	$c_{N,f}$	$c_{m,f}$
$C_\mu = 0$						
-16	-.542	.002	-1.378	.772	.027	-.149
-12	-.481	-.011	-1.194	.714	.242	-.239
-8	-.183	-.085	-.838	.587	.637	-.391
-4	.171	-.133	-.250	.348	.772	-.427
0	.972	-.323	.394	.142	1.540	-.704
4	1.184	-.278	1.501	-.390	1.433	-.661
8	1.541	-.299	2.592	-1.011	1.627	-.749
12	1.807	-.283	3.763	-1.797	1.691	-.775
$C_\mu = 0.013$						
-16	-.053	-.159	-1.067	.659	1.013	-.236
-12	.105	-.180	-.739	.542	1.127	-.277
-8	.384	-.228	-.320	.375	1.377	-.415
-4	1.072	-.437	.050	.245	2.322	-.768
0	1.311	-.379	1.108	-.157	2.081	-.637
4	1.722	-.441	2.068	-.681	2.469	-.796
8	1.960	-.420	3.103	-1.346	2.424	-.797
12	2.152	-.381	4.136	-2.038	2.239	-.772
$C_\mu = 0.023$						
-16	.243	-.238	-.768	.544	1.520	-.329
-12	.409	-.252	-.431	.414	1.585	-.386
-8	.797	-.376	-.169	.320	2.240	-.635
-4	1.510	-.525	.655	.058	2.987	-.914
0	1.858	-.550	1.695	-.448	3.147	-.952
4	2.076	-.522	2.648	-.975	3.059	-.944
8	2.464	-.545	3.718	-1.718	3.113	-.628
12	2.460	-.468	4.481	-2.213	2.660	-.654
$C_\mu = 0.038$						
-16	.745	-.428	-.498	.429	2.971	-.530
-12	.732	-.364	-.178	.303	2.512	-.383
-8	1.218	-.522	.025	.234	3.120	-.615
-4	1.807	-.615	1.040	-.106	3.556	-.725
0	2.235	-.661	2.045	-.633	3.825	-.771
4	2.484	-.651	2.972	-1.166	3.825	-.763
8	2.507	-.563	3.791	-1.781	3.347	-.698
12	2.461	-.465	4.492	-2.220	2.767	-.629

α, deg	$c_{N,w}$	$c_{m,w}$	$c_{N,N}$	$c_{m,N}$	$c_{N,f}$	$c_{m,f}$
$C_\mu = 0.063$						
-16	.748	-.417	-.400	.374	3.006	-.489
-12	.838	-.422	-.173	.311	2.955	-.419
-8	1.466	-.607	.146	.212	3.710	-.716
-4	1.907	-.644	1.096	-.136	3.907	-.752
0	2.328	-.696	2.148	-.697	4.221	-.831
4	2.718	-.723	3.248	-1.325	4.434	-.891
8	2.793	-.648	4.145	-1.976	4.044	-.830
12	2.599	-.503	4.704	-2.311	3.081	-.661
$C_\mu = 0.121$						
-16	.890	-.485	-.388	.388	3.745	-.529
-12	1.056	-.502	-.054	.252	3.675	-.520
-8	1.808	-.716	.343	.129	4.585	-.807
-4	2.251	-.760	1.293	-.261	4.878	-.879
0	2.517	-.754	2.370	-.813	4.932	-.895
4	2.797	-.734	3.455	-1.508	4.862	-.874
8	3.187	-.757	4.653	-2.254	5.090	-.956
12	2.833	-.623	4.161	-2.262	3.774	-.798
$C_\mu = 0.189$						
-16	.948	-.504	-.299	.344	4.245	-.517
-12	1.279	-.594	.071	.210	4.658	-.637
-8	2.017	-.775	.568	.043	5.412	-.862
-4	2.372	-.789	1.595	-.386	5.449	-.877
0	2.781	-.827	2.696	-.960	5.866	-.981
4	3.008	-.807	3.691	-1.649	5.758	-.915
8	3.331	-.792	5.024	-2.496	5.727	-.959
12	2.982	-.633	4.822	-2.284	4.509	-.877

TABLE XI.- INTEGRATED SECTION DATA

$c_f = 0.25c_w$; $\delta_N = 0^\circ$; $C_L = 0$; $q \approx 25$ lb/sq ft

α, deg	$c_{N,w}$	$c_{m,w}$	$c_{N,N}$	$c_{m,N}$	$c_{N,f}$	$c_{m,f}$
$\delta_f = 5^\circ$						
-4	-.0317	-.0438	-.4820	-.3169	.0074	-.0195
0	.1910	-.0396	.3087	.1964	.1497	-.0480
4	.5278	-.0426	1.3873	.6911	.2580	-.0870
8	.7507	-.0439	1.6367	.8627	.2272	-.0778
12	.8809	-.1074	1.5091	.7978	.3876	-.1458
16	1.0803	-.1376	1.8315	.9573	.5180	-.2005
$\delta_f = 10^\circ$						
-8	-.1252	-.1060	-1.3379	-.7884	.2776	-.0822
-4	.1324	-.1007	-.3868	-.2631	.3248	-.0973
0	.4119	-.0975	.5365	.3298	.3748	-.1144
4	.6817	-.0806	1.5747	.9858	.3917	-.1226
8	.9402	-.0921	1.7830	.9378	.3924	-.1366
12	1.0163	-.1382	1.6922	.8902	.5366	-.2011
16	1.0608	-.1762	1.6033	.8418	.6868	-.2675
$\delta_f = 15^\circ$						
-10	-.0779	-.1315	-1.2610	-.6911	.3854	-.1110
-8	.0766	-.1496	-.8903	-.6613	.4634	-.1338
-4	.3656	-.1516	-.0440	-.0498	.5456	-.1589
0	.5852	-.1416	.7241	.4413	.5522	-.1621
4	.8549	-.1177	1.8877	1.0920	.5494	-.1706
8	1.0421	-.1252	1.8102	.9476	.5148	-.1800
12	1.0991	-.1623	1.7915	.9391	.6668	-.2488
16	1.0007	-.1978	1.3590	.7209	.7878	-.3123

α, deg	$c_{N,w}$	$c_{m,w}$	$c_{N,N}$	$c_{m,N}$	$c_{N,f}$	$c_{m,f}$
$\delta_f = 30^\circ$						
-12	.1248	-.2093	-1.2019	-.6902	.8133	-.2893
-8	.4626	-.2508	-.4829	-.3413	1.0079	-.3504
-4	.6810	-.2370	.3604	.1987	1.0468	-.4005
0	.7719	-.1807	1.0552	.6520	.8182	-.3350
4	1.1241	-.1748	2.0490	1.0680	.8753	-.3501
8	1.2586	-.2052	1.8382	.9569	.8713	-.3440
12	1.1235	-.2407	1.3990	.7396	1.0300	-.4157
16	1.1718	-.2526	1.4595	.7658	1.0916	-.4494
$\delta_f = 45^\circ$						
-12	.0791	-.1987	-1.1599	-.6596	.8367	-.3757
-8	.4636	-.2284	-.3105	-.2427	1.0340	-.4525
-4	.6999	-.2337	.4125	.2311	1.1122	-.4744
0	.9650	-.2147	1.4413	.9004	1.1000	-.4698
4	1.2673	-.2116	2.0740	1.0720	1.1592	-.4960
8	1.3858	-.2591	1.8173	.9436	1.1626	-.4922
12	1.2381	-.2767	1.4601	.7684	1.2676	-.5357
16	1.2966	-.2864	1.5519	.8102	1.3255	-.5651
$\delta_f = 60^\circ$						
-16	-.3411	-.0905	-1.1802	-.6370	.6881	-.3490
-12	.3003	-.2331	-.9439	-.5840	1.1975	-.5528
-8	.6226	-.2527	-.0428	-.0760	1.3497	-.6165
-4	.8509	-.2439	.7646	.4462	1.3466	-.6038
0	1.0807	-.2147	1.8307	1.0778	1.2757	-.5699
4	1.3834	-.2249	2.1197	1.0951	1.3445	-.5936
8	1.3755	-.2655	1.7315	.8982	1.3497	-.5962
12	1.2977	-.2775	1.5668	.8169	1.4539	-.6405
16	1.3561	-.2887	1.6403	.8480	1.5182	-.6742

TABLE XII.- INTEGRATED SECTION DATA

$c_f = 0.25c_w$; $\delta_N = 15^\circ$; $C_\mu = 0$; $q \approx 25$ lb/sq ft

α, deg	$c_{N,w}$	$c_{m,w}$	$c_{N,N}$	$c_{m,N}$	$c_{N,f}$	$c_{m,f}$
$\delta_f = 0^\circ$						
-4	-.3445	-.0021	-.9562	-.5702	-.0273	.0154
0	-.0303	-.0269	-.6760	-.4813	.0102	-.0029
4	.2245	-.0180	.2528	-.0227	.0417	-.0134
8	.4981	-.0152	1.1873	.5760	.0986	-.0362
12	.7805	-.0043	2.2661	1.3049	.1407	-.0502
16	.9745	-.0542	2.2256	1.1493	.2533	-.0955
20	.9962	-.1191	1.7882	.9227	.4682	-.1842
$\delta_f = 15^\circ$						
-8	-.1417	-.0675	-.8293	-.3076	.2739	-.0699
-4	.2388	-.1519	-.6469	-.4369	.4620	-.1350
0	.5537	-.1604	.1673	-.0871	.5354	-.1557
4	.8366	-.1564	1.1257	.5102	.5816	-.1747
8	1.0562	-.1366	2.0229	1.1267	.5717	-.1782
12	1.2331	-.1154	2.8340	1.4938	.5582	-.1938
16	1.2465	-.1776	2.1515	1.1151	.7332	-.2853
$\delta_f = 30^\circ$						
-8	-.1092	-.0972	-.8362	-.5124	.5020	-.2358
-4	.4337	-.2013	-.4469	-.3476	.7370	-.3093
0	.7502	-.2109	.4609	.0853	.8526	-.3528
4	1.0095	-.2022	1.3513	.6467	.8737	-.3627
8	1.2450	-.1744	2.3796	1.3511	.8356	-.3442
12	1.4189	-.1812	2.7326	1.4067	.8770	-.3510
16	1.3075	-.2635	1.6799	.8618	1.0955	-.4442

α, deg	$c_{N,w}$	$c_{m,w}$	$c_{N,N}$	$c_{m,N}$	$c_{N,f}$	$c_{m,f}$
$\delta_f = 45^\circ$						
-8	.0502	-.1216	-.5828	-.3844	.6459	-.3083
-4	.6608	-.2494	-.1637	-.2684	1.0851	-.4664
4	1.1589	-.2341	1.5910	.8000	1.1502	-.4950
8	1.3468	-.2067	2.5163	1.4093	1.1173	-.4851
12	1.5000	-.2280	2.5674	1.3084	1.1728	-.5019
16	1.2816	-.2772	1.5319	.7836	1.2672	-.5386
$\delta_f = 60^\circ$						
-12	-.2037	-.0896	-.9409	-.5551	.6793	-.3400
-8	.2075	-.1456	-.4041	-.2947	.8350	-.4035
-4	.7902	-.2579	.1581	-.0978	1.3146	-.5934
0	1.0168	-.2451	1.0129	.4289	1.3165	-.5931
4	1.2660	-.2325	1.9124	1.0107	1.3304	-.5976
8	1.4551	-.2024	2.9485	1.6116	1.3011	-.5866
12	1.5752	-.2594	2.3610	1.1996	1.4189	-.6346
16	1.3498	-.2813	1.6481	.8400	1.4804	-.6600

TABLE XIII. - INTEGRATED SECTION DATA

$\delta_f = 0.3c_w$; $\delta_N = 30^\circ$; $C_\mu = 0$; $q = 25$ lb/sq ft

α, deg	$c_{N,w}$	$c_{m,w}$	$c_{N,N}$	$c_{m,N}$	$c_{N,f}$	$c_{m,f}$
$\delta_f = 0^\circ$						
-8	-.5449	.0828	-1.2559	-.6778	-.3399	.1346
-4	-.4214	.0488	-1.0007	-.6191	-.1688	.0611
0	-.2522	-.0179	-.7061	-.4902	.0418	-.0174
4	.1455	-.0509	-.2743	-.2893	.0718	-.0240
8	.4509	-.0480	.4303	-.0302	.1052	-.0370
12	.7238	-.0457	1.4186	.5307	.1560	-.0552
16	.9286	-.0441	2.1719	1.0164	.2018	-.0718
20	1.0302	-.0542	2.6481	1.4009	.3105	-.1184
$\delta_f = 5^\circ$						
-8	-.4240	.0466	-1.0560	-.6311	-.2013	.0890
-4	-.2706	.0054	-.8824	-.5627	-.0110	.0070
0	.0137	-.0577	-.5421	-.4129	.1764	-.0594
4	.3366	-.1015	-.1445	-.2267	.2421	-.0784
8	.6345	-.0944	.6585	.1004	.2744	-.0890
12	.8363	-.0933	1.5914	.6436	.3218	-.1058
16	1.0971	-.0291	2.3511	1.1316	.3656	-.1243
20	1.1538	-.0901	2.9221	1.5058	.4640	-.1755
$\delta_f = 10^\circ$						
-8	-.3168	.0119	-1.0167	-.6213	-.0780	.0616
-4	-.1349	-.0342	-.7879	-.5249	.1268	-.0354
0	.1800	-.1040	-.4212	-.3569	.3256	-.1046
4	.5484	-.1477	-.0501	-.2089	.3974	-.1208
8	.8390	-.1389	.9994	.2738	.4424	-.1374
12	1.1311	-.1410	1.9481	.8396	.5024	-.1578
16	1.2230	-.1289	2.4487	1.2102	.5116	-.1762
20	1.2392	-.1220	2.9218	1.5698	.6050	-.2371
$\delta_f = 15^\circ$						
-8	-.1624	-.0375	-.9136	-.5733	.0675	.0179
-4	.0052	-.0690	-.6429	-.4533	.2414	-.0664
0	.3558	-.1555	-.3067	-.3036	.4832	-.1469
4	.7378	-.1859	.2079	-.0907	.5536	-.1677
8	.9996	-.1773	1.2117	.4000	.5872	-.1814
12	1.2357	-.1728	2.0383	.8978	.6195	-.1962
14	1.3266	-.1629	2.4770	1.1791	.6219	-.2075
16	1.3211	-.1541	2.6142	1.3182	.6330	-.2275
20	1.2989	-.1488	2.9373	1.5800	.7251	-.2866

α, deg	$c_{N,w}$	$c_{m,w}$	$c_{N,N}$	$c_{m,N}$	$c_{N,f}$	$c_{m,f}$
$\delta_f = 30^\circ$						
-8	-.2593	-.0932	-.9695	-.5898	.2319	-.1306
-4	.0077	-.0922	-.6757	-.4702	.4525	-.2274
0	.5268	-.2137	-.2360	-.2671	.7572	-.3146
4	.9273	-.2302	.5385	.0209	.8458	-.3488
8	1.1826	-.2205	1.4978	.5649	.8603	-.3510
12	1.3617	-.1989	2.3197	1.0778	.8388	-.3451
14	1.4637	-.1921	2.7743	1.4044	.8690	-.3590
16	1.4633	-.2098	2.8041	1.4653	1.0091	-.4085
20	1.4249	-.2149	2.8978	1.5778	1.0431	-.4350
$\delta_f = 45^\circ$						
-8	-.1758	-.0699	-.9475	-.5827	.4473	-.2274
-4	.1214	-.1009	-.4791	-.3751	.5158	-.2645
0	.7445	-.2407	.1195	-.0942	1.0094	-.4416
4	1.1034	-.2663	.8696	.1978	1.1499	-.4942
8	1.3214	-.2574	1.7404	.7213	1.1935	-.5138
12	1.5241	-.2472	2.5683	1.2609	1.2370	-.5406
14	1.5921	-.2329	2.9393	1.5289	1.2205	-.5323
16	1.5812	-.2367	3.0525	1.6307	1.2947	-.5472
20	1.5978	-.2654	2.9051	1.5364	1.3262	-.5733
$\delta_f = 60^\circ$						
-8	-.0765	-.0831	-.8011	-.5147	.5562	-.2821
-4	.2324	-.1270	-.3553	-.3160	.7493	-.3795
0	.8905	-.2626	.2678	-.0382	1.2784	-.5752
4	1.1693	-.2617	1.1070	.3369	1.3204	-.5914
8	1.3882	-.2549	1.9278	.8324	1.3718	-.6131
12	1.5524	-.2332	2.7495	1.3982	1.3626	-.6136
16	1.6209	-.2173	3.3659	1.8200	1.4149	-.6317
20	1.6701	-.2717	2.9313	1.5231	1.5281	-.6806
24	1.5143	-.3104	1.9671	.9898	1.6271	-.7302
12	1.5678	-.2300	2.8149	1.4409	1.3402	-.5947
14	1.6737	-.2277	3.2512	1.7200	1.3821	-.6218

TABLE XIV.- INTEGRATED SECTION DATA

$c_f = 0.25c_w$; $\delta_N = 45^\circ$; $C_\mu = 0$; $q \approx 25$ lb/sq ft

α, deg	$c_{N,w}$	$c_{m,w}$	$c_{N,N}$	$c_{m,N}$	$c_{N,f}$	$c_{m,f}$
$\delta_f = 0^\circ$						
-4	-.4087	.0705	-1.1220	-.6747	-.2918	.1197
0	-.2613	.0360	-.9116	-.5893	-.1273	.0434
4	-.0334	-.0177	-.6605	-.4862	.0608	-.0259
8	.2869	-.0784	-.3518	-.3498	.1363	-.0475
12	.6397	-.1479	-.4199	-.4400	.2572	-.0918
16	.7586	-.1940	.0080	-.2000	.5599	-.2248
20	.7801	-.1691	.6453	.1387	.5599	-.2366
24	.8909	-.1676	1.1793	.4418	.5876	-.2502
$\delta_f = 15^\circ$						
-12	-.1807	-.0362	-1.1425	-.6440	.0844	.0146
-10	-.1589	-.0347	-1.0960	-.6347	.0734	.0128
-8	-.1333	-.0353	-1.0547	-.6267	.0702	.0131
-4	-.0379	-.0415	-.8439	-.5471	.0947	-.0038
0	.1010	-.0645	-.6191	-.4658	.1921	-.0509
4	.4446	-.1639	-.3315	-.3458	.5088	-.1675
8	.8010	-.2093	-.3469	-.4960	.5250	-.1741
12	.8531	-.2556	-.2897	-.3840	.7643	-.2955
16	.8761	-.2391	.2178	-.0964	.8174	-.3338
20	.9637	-.2266	.8059	.2560	.8296	-.3411
$\delta_f = 30^\circ$						
-8	-.3454	.0212	-1.1925	-.6871	-.0848	.0296
-4	-.1516	-.0346	-.9393	-.5858	.2216	-.1314
0	.1046	-.0716	-.5923	-.4520	.3204	-.1765
4	.6771	-.2253	-.1255	-.2493	.7929	-.3306
8	1.1015	-.2681	.1321	-.2889	.8657	-.3538
12	.9247	-.2932	-.2039	-.3613	1.0352	-.4294
16	1.0283	-.2751	.5516	.0662	1.0486	-.4373
20	1.1253	-.2709	1.1473	.4169	1.0888	-.4562

α, deg	$c_{N,w}$	$c_{m,w}$	$c_{N,N}$	$c_{m,N}$	$c_{N,f}$	$c_{m,f}$
$\delta_f = 45^\circ$						
-8	-.3153	-.0056	-1.1689	-.6716	.1056	-.0710
-4	-.0458	-.0650	-.8459	-.5462	.3897	-.2091
0	.2015	-.0994	-.4922	-.4009	.4593	-.2699
4	.8208	-.2524	.0193	-.1907	1.0051	-.4434
8	1.1950	-.2793	.4777	-.0858	1.0789	-.4664
12	1.4418	-.3186	1.0468	.2907	1.2624	-.5354
16	1.1336	-.2864	.9113	.2676	1.2274	-.5282
20	1.4517	-.3195	1.5731	.6844	1.3907	-.6018
$\delta_f = 60^\circ$						
0	.4005	-.1534	-.2669	-.3004	.8813	-.4376
4	.9786	-.2665	.3206	-.0493	1.2339	-.5584
8	1.2672	-.2722	.8548	.1996	1.2659	-.5691
12	1.5228	-.3124	1.2453	.4076	1.4808	-.6560
16	1.2655	-.2883	1.1522	.4018	1.4306	-.6426
20	1.3330	-.3062	1.4290	.6404	1.5406	-.6920
24	1.4044	-.2974	1.8689	.9658	1.5225	-.6819
26	1.5002	-.2940	2.3601	1.2884	1.5544	-.6920
28	1.5580	-.2943	2.6990	1.4484	1.5928	-.7118
30	1.7356	-.3069	2.9576	1.5227	1.6509	-.7411
32	1.8050	-.3140	3.1809	1.6267	1.7456	-.7896

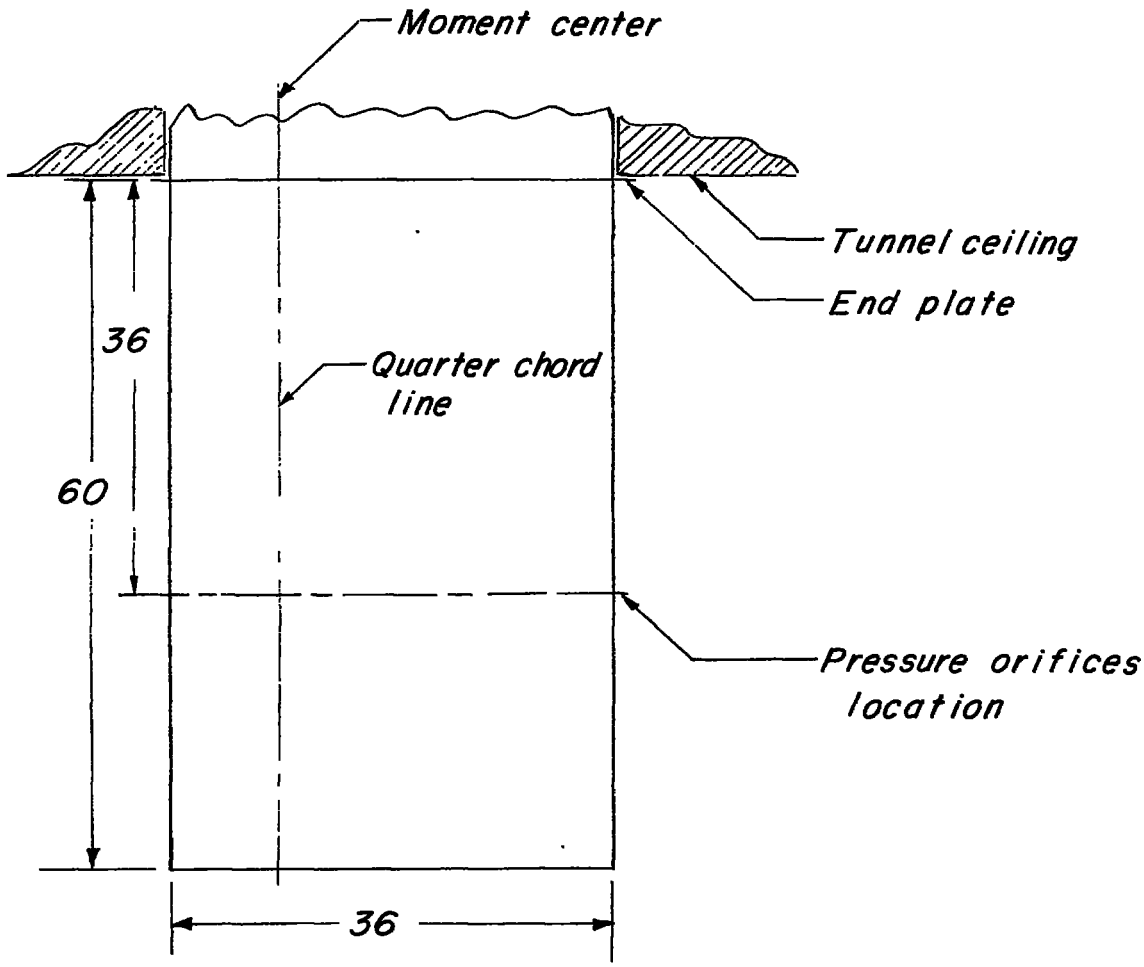


Figure 1.- General characteristics of model. (All dimensions are in inches unless otherwise noted.)

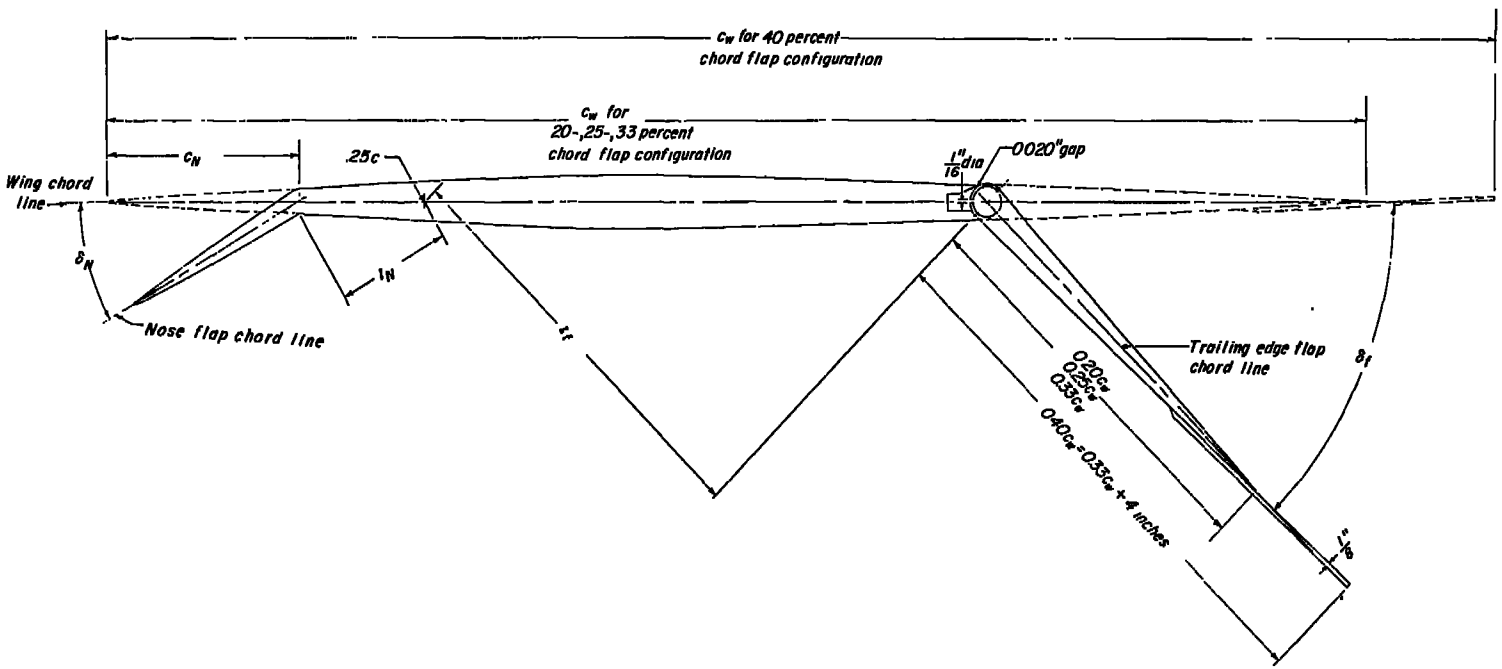


Figure 2.- Details of leading- and trailing-edge flaps investigated. (Note chord of wing increases when $c_f = 0.40c_w$.)

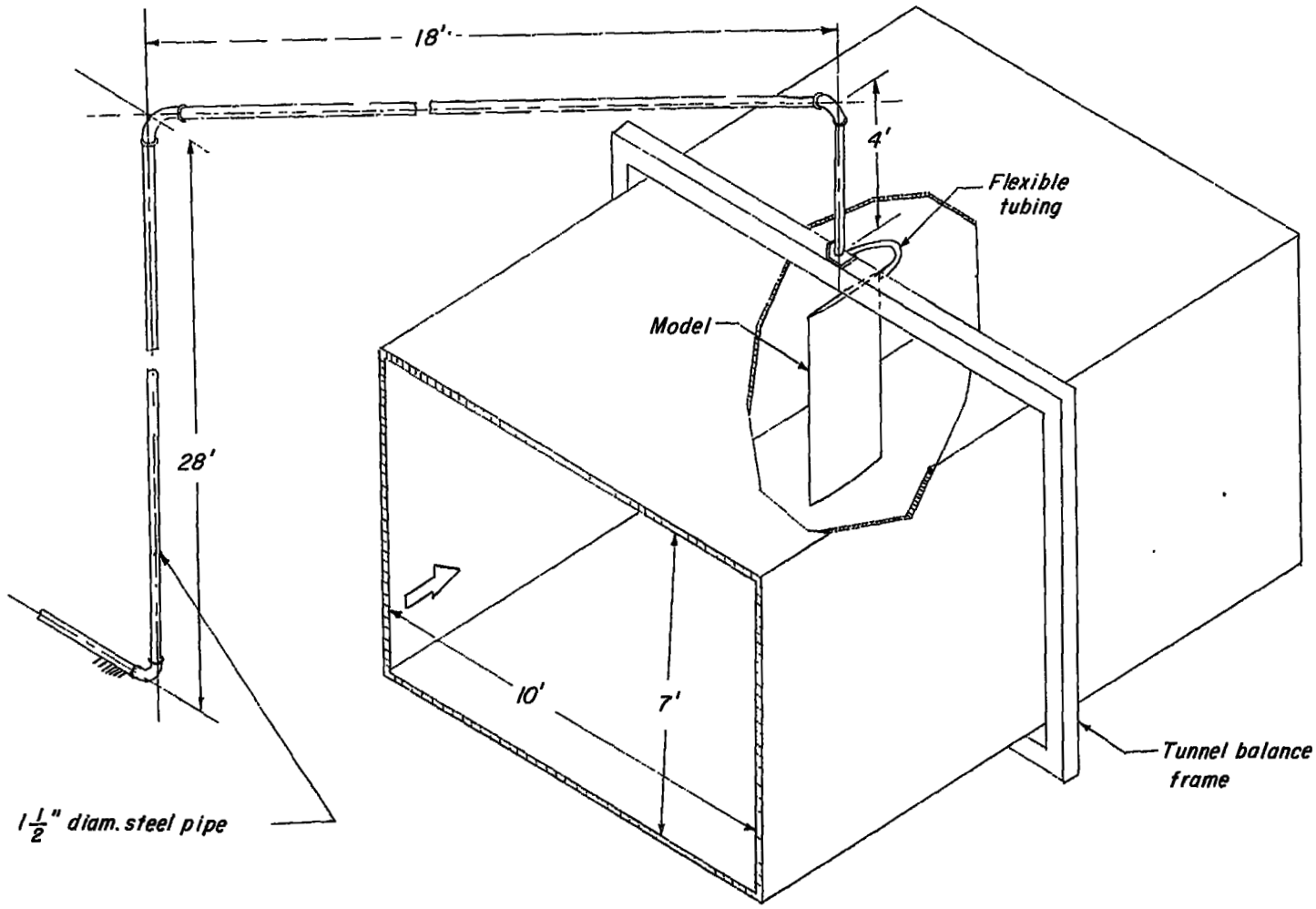
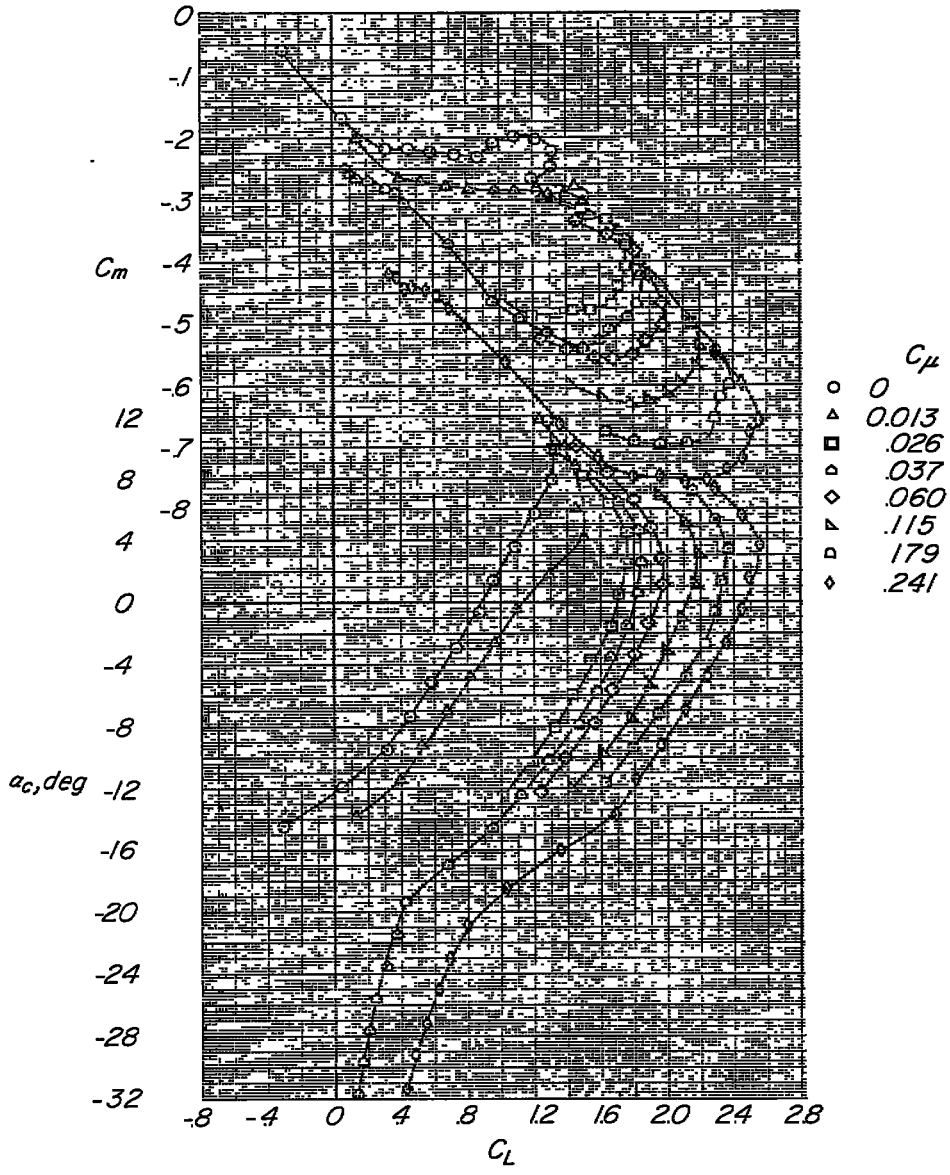
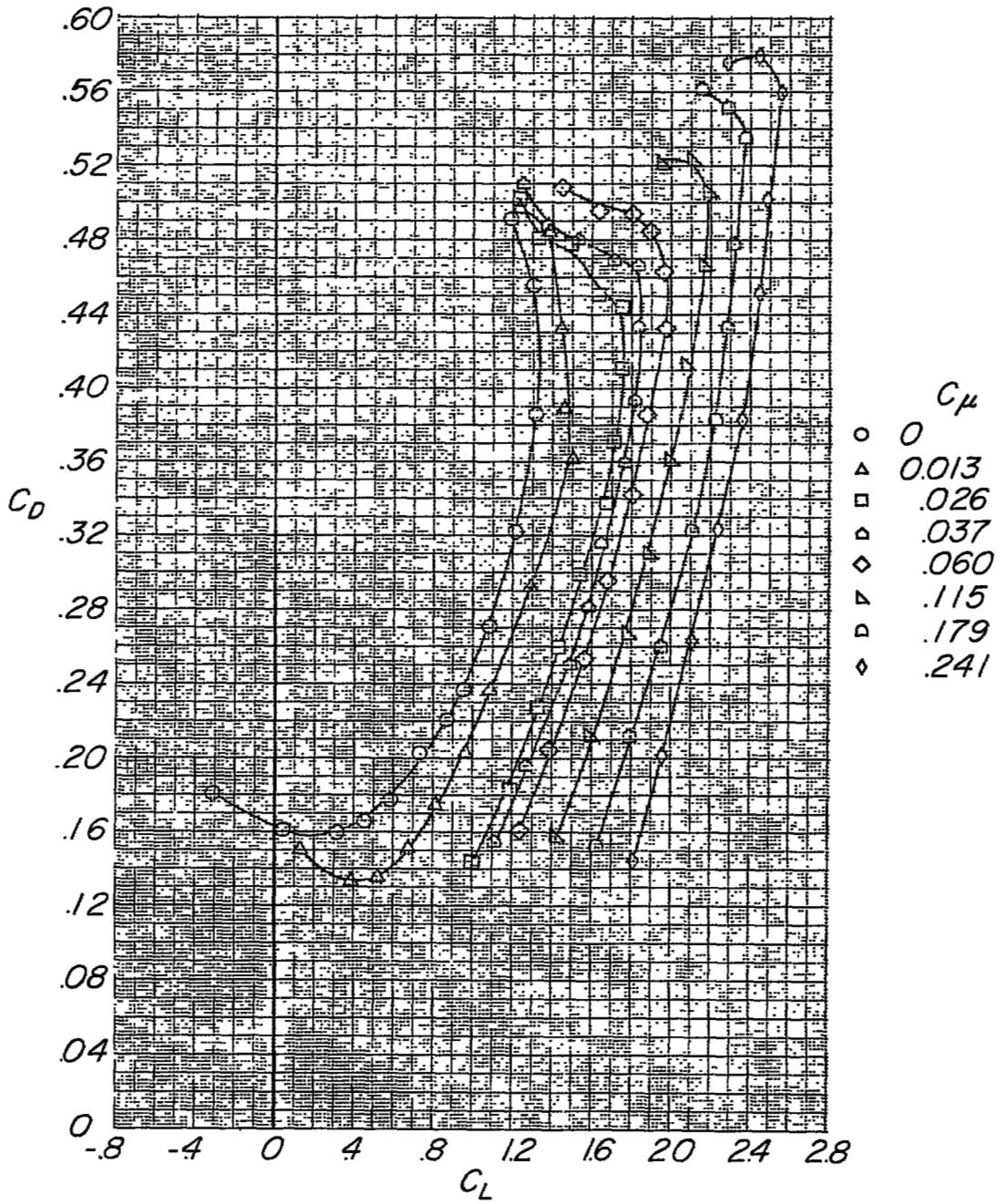


Figure 3.- Schematic diagram of apparatus used to introduce high-pressure air to model.



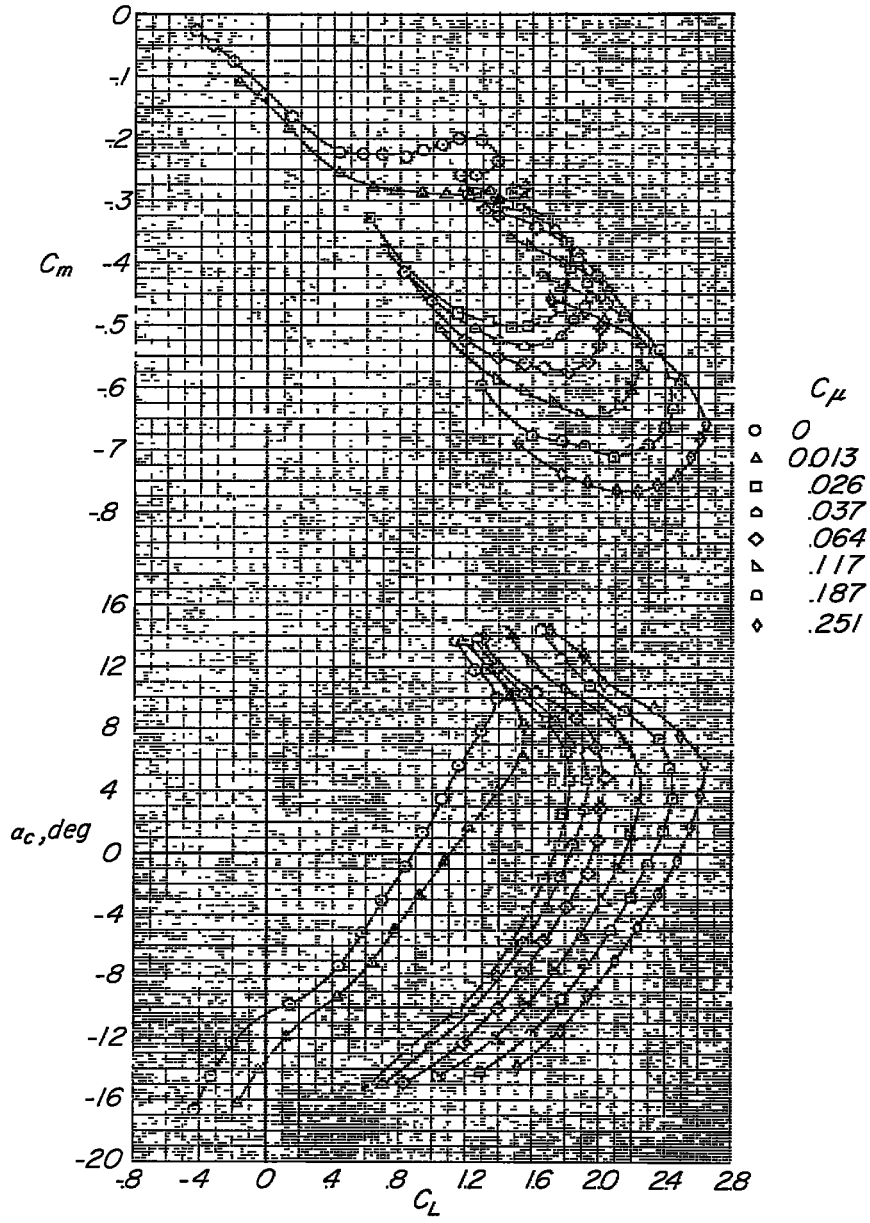
(a) C_m and α_c against C_L.

Figure 4.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 0^\circ$; $q \approx 12.5$ lb/sq ft.



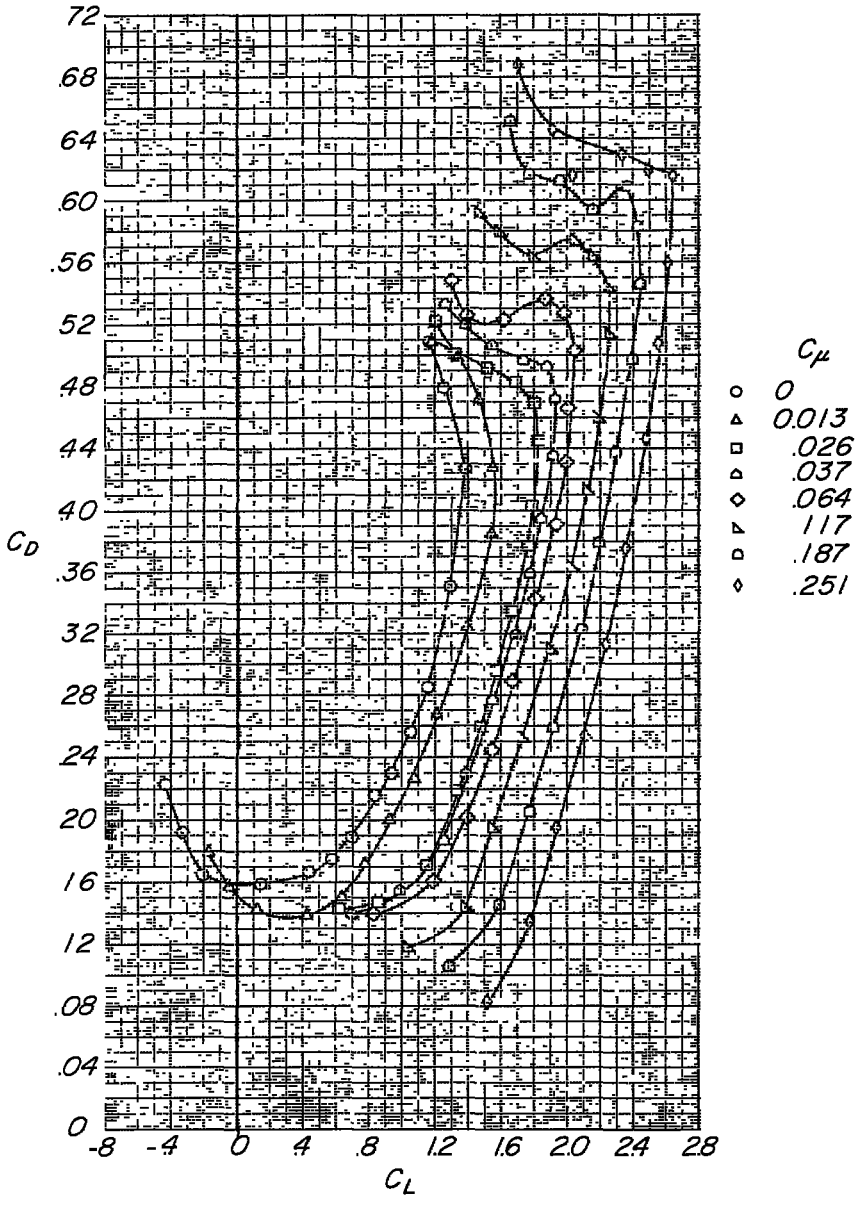
(b) C_D against C_L.

Figure 4.- Concluded.



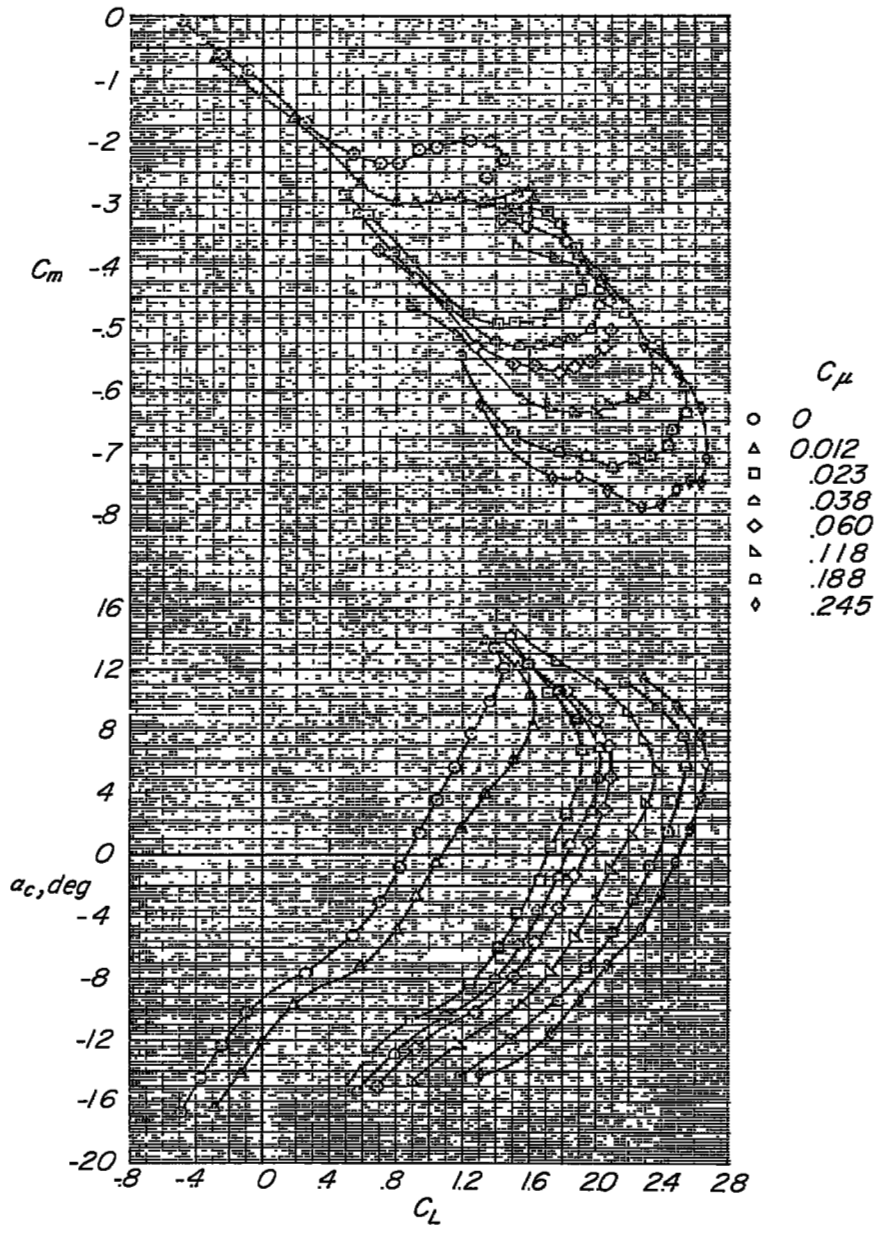
(a) C_m and α_c against C_L.

Figure 5.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 4.75^\circ$; $q \approx 12.5$ lb/sq ft.



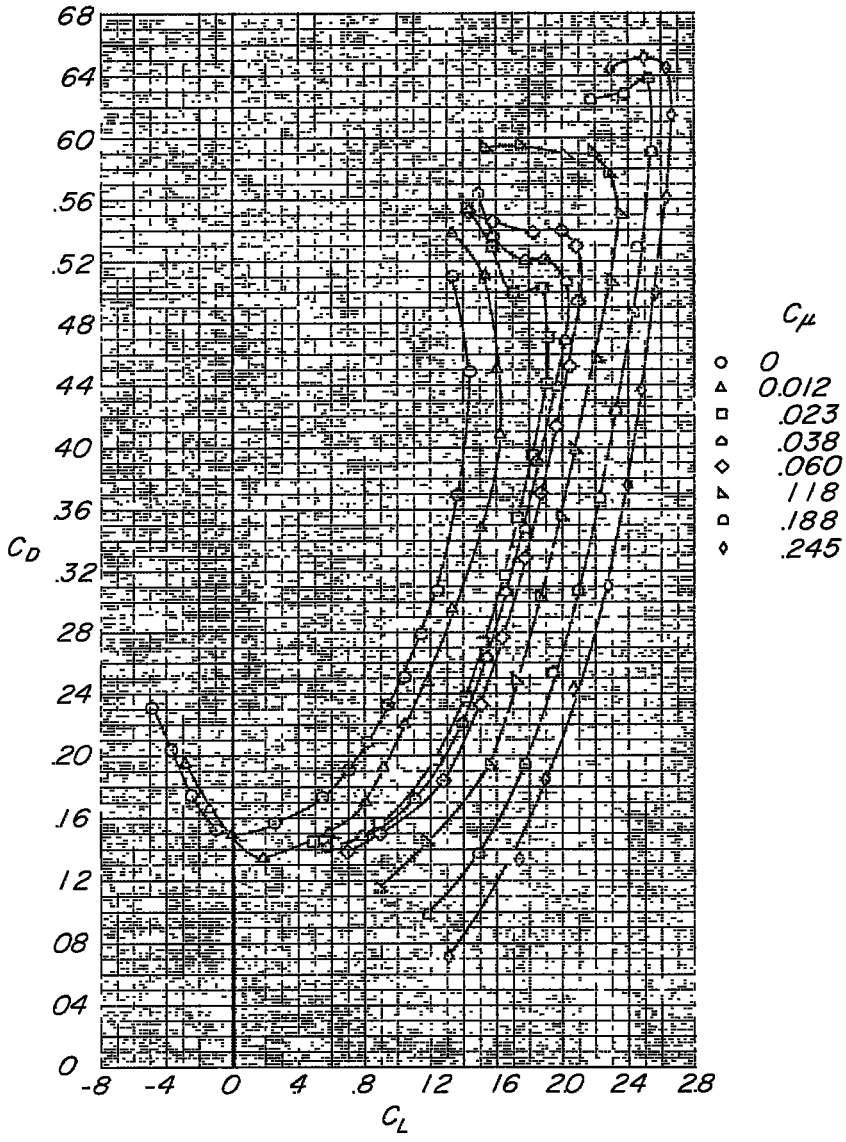
(b) C_D against C_L.

Figure 5.- Concluded.



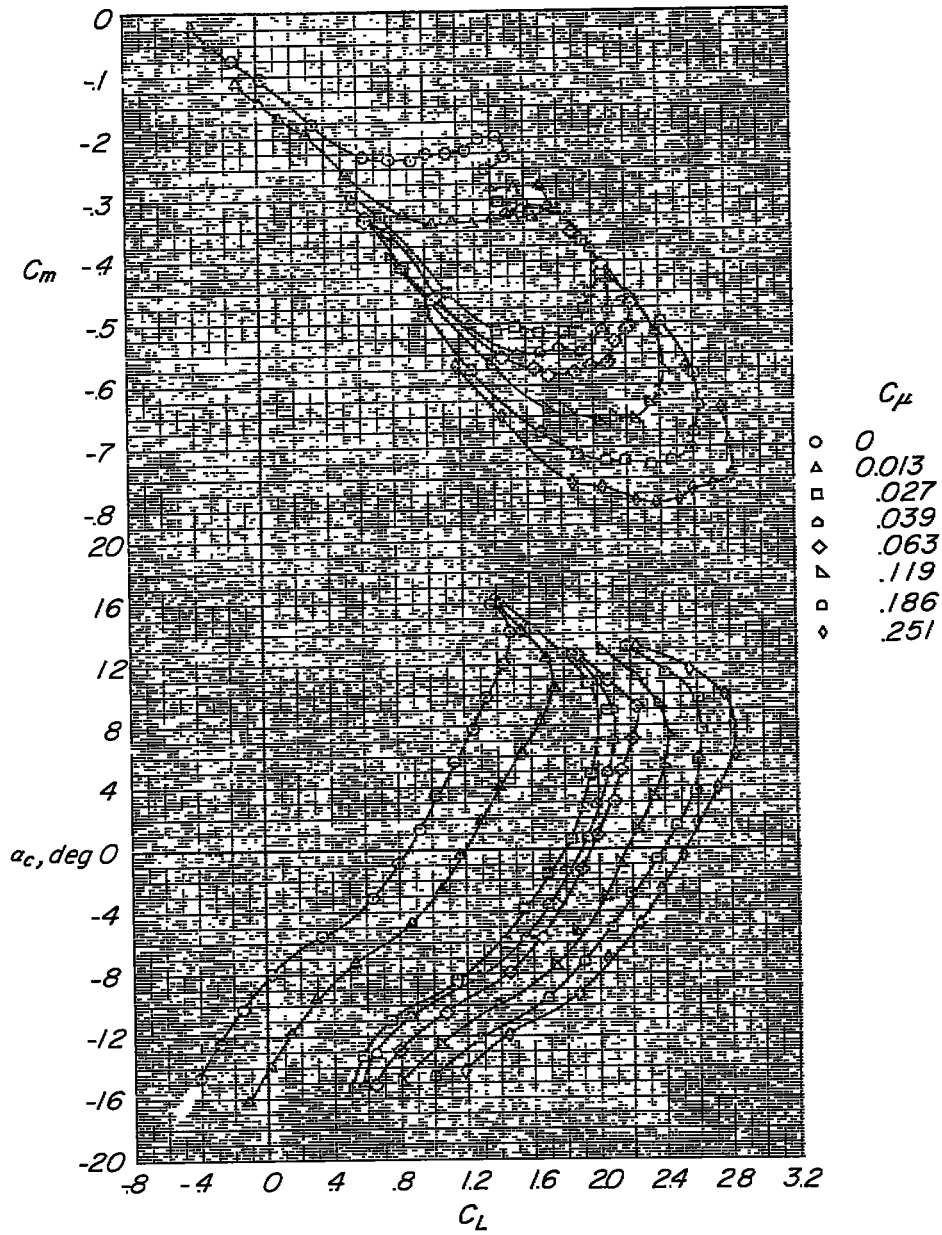
(a) C_m and α_c against C_L.

Figure 6.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 10^\circ$; $q \approx 12.5$ lb/sq ft.



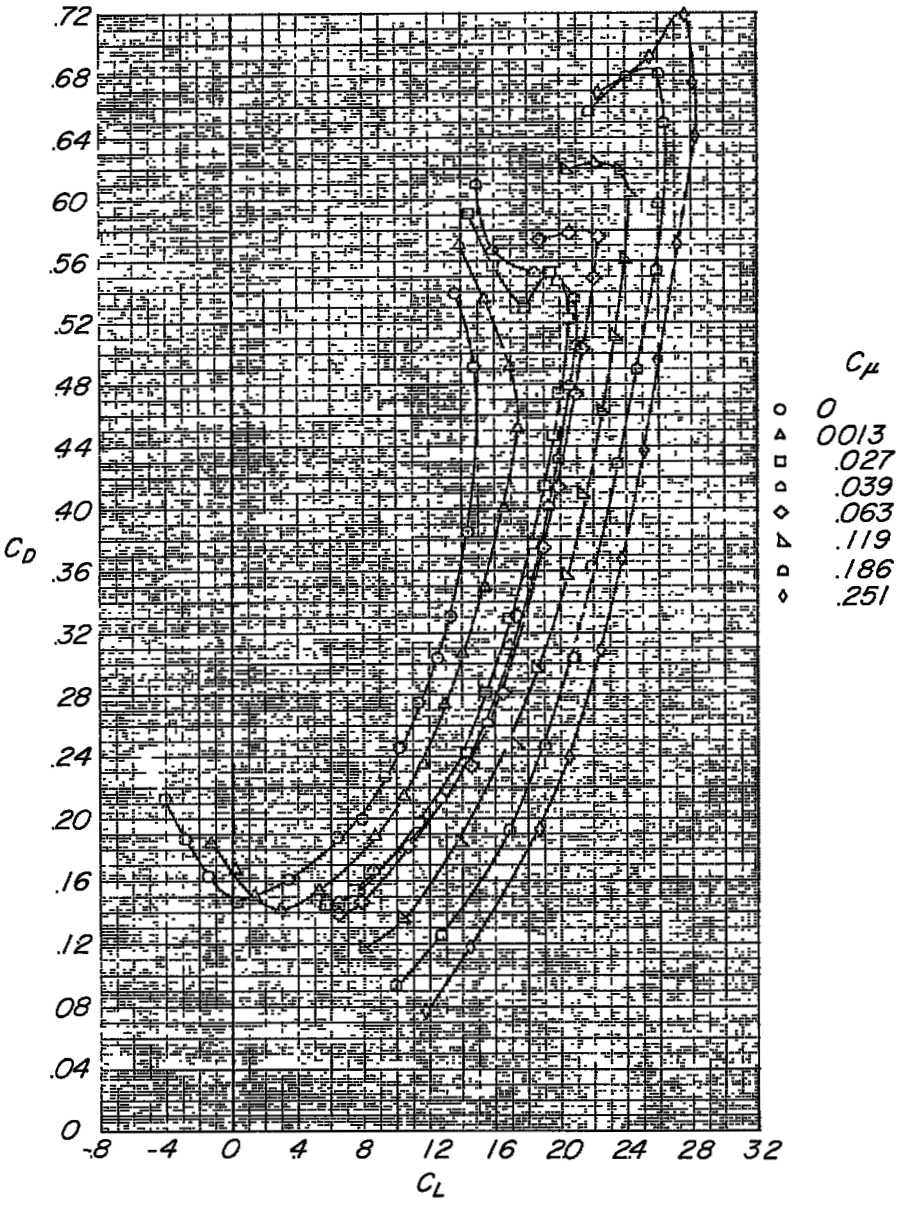
(b) C_D against C_L.

Figure 6.- Concluded.



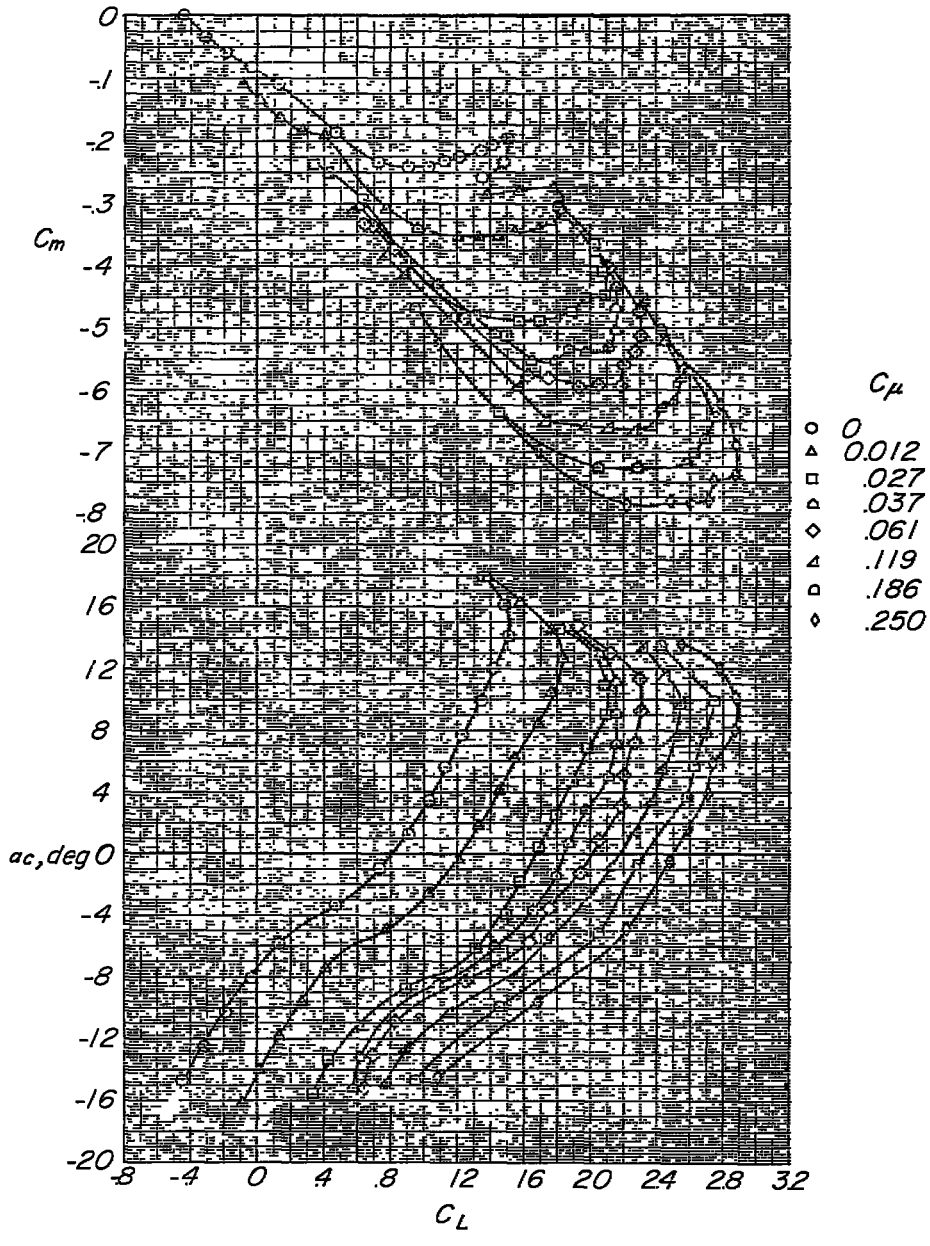
(a) C_m and α_c against C_L.

Figure 7.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 15^\circ$; $q \approx 12.5$ lb/sq ft.



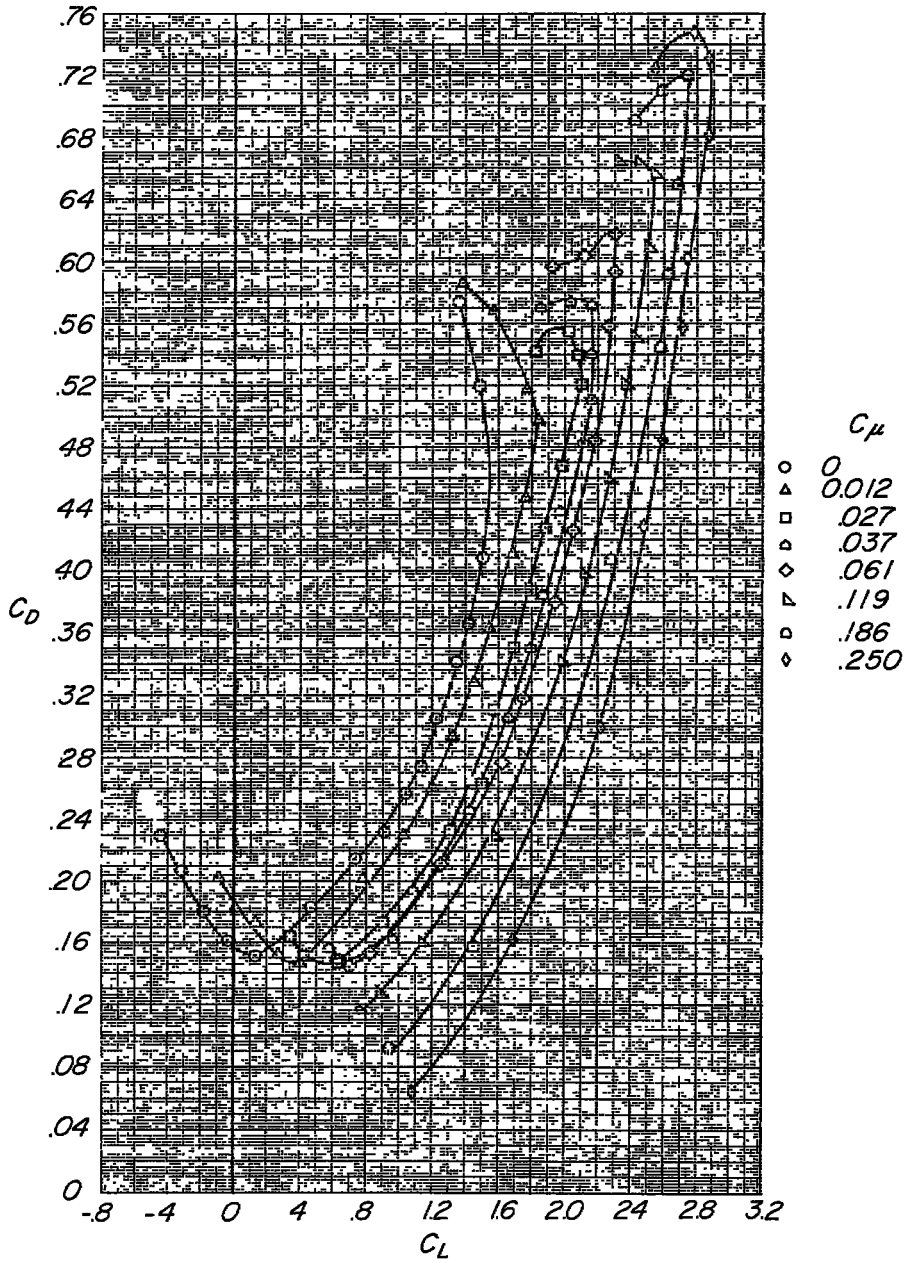
(b) C_D against C_L.

Figure 7.- Concluded.



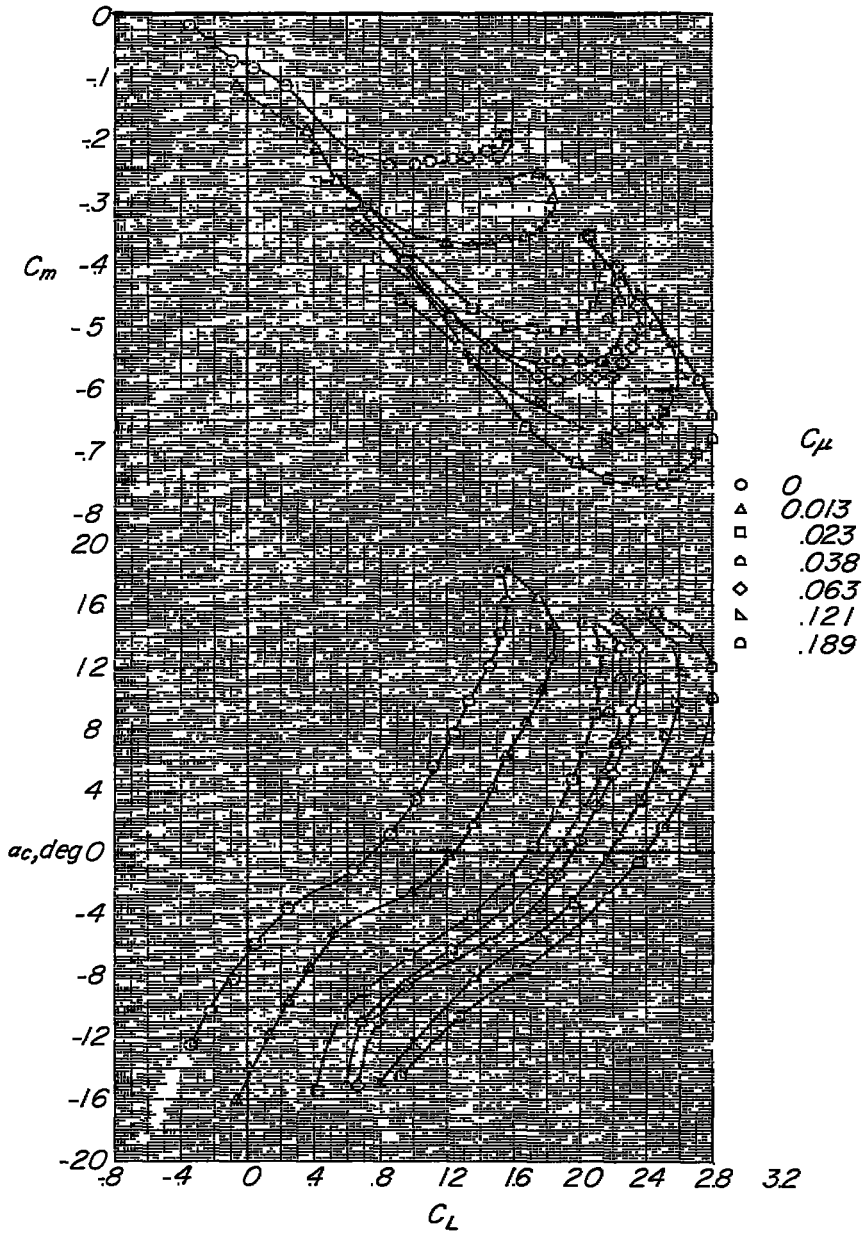
(a) C_m and α_c against C_L.

Figure 8.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 20^\circ$; $q \approx 12.5$ lb/sq ft.



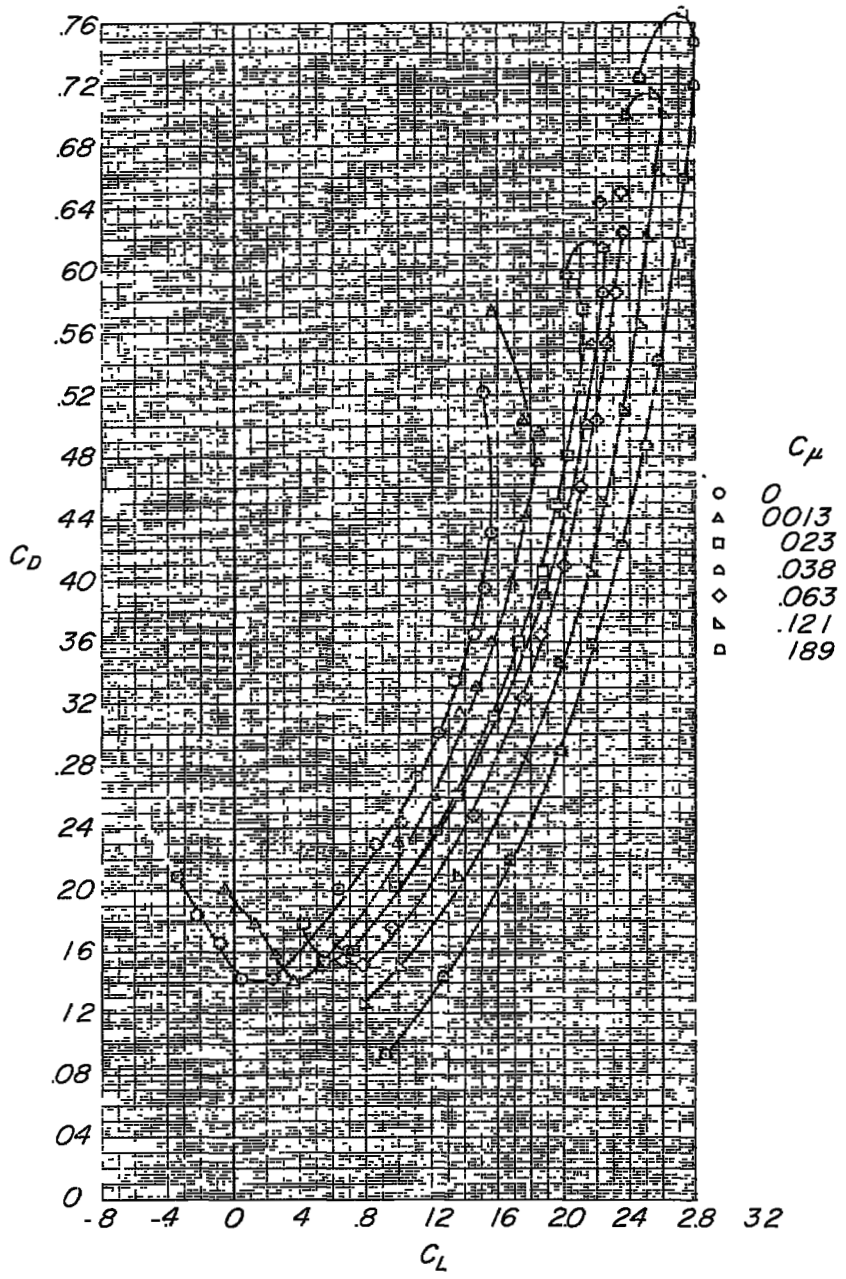
(b) C_D against C_L.

Figure 8.- Concluded.



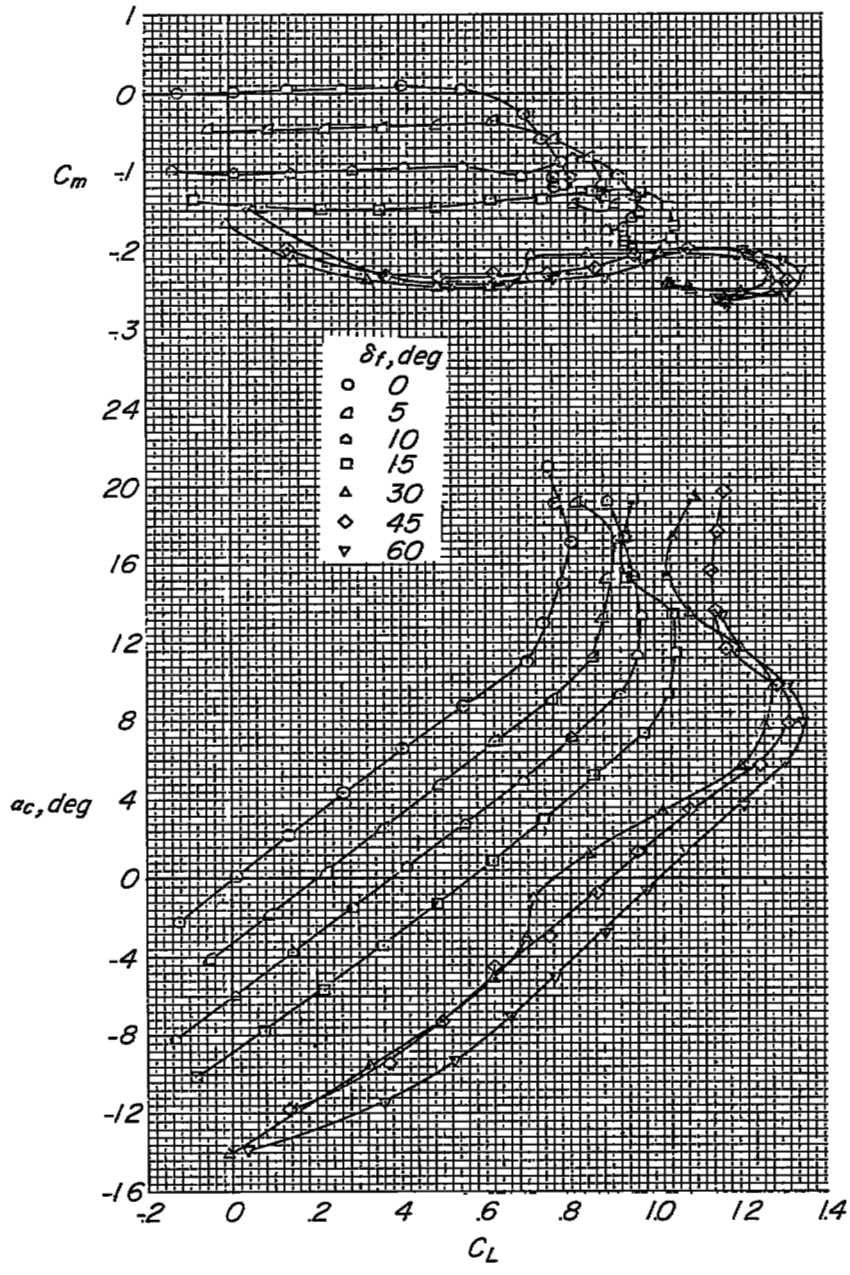
(a) C_m and α_c against C_L.

Figure 9.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.20c_w$; $\delta_f = 60^\circ$; $\delta_N = 25^\circ$; $q \approx 12.5$ lb/sq ft.



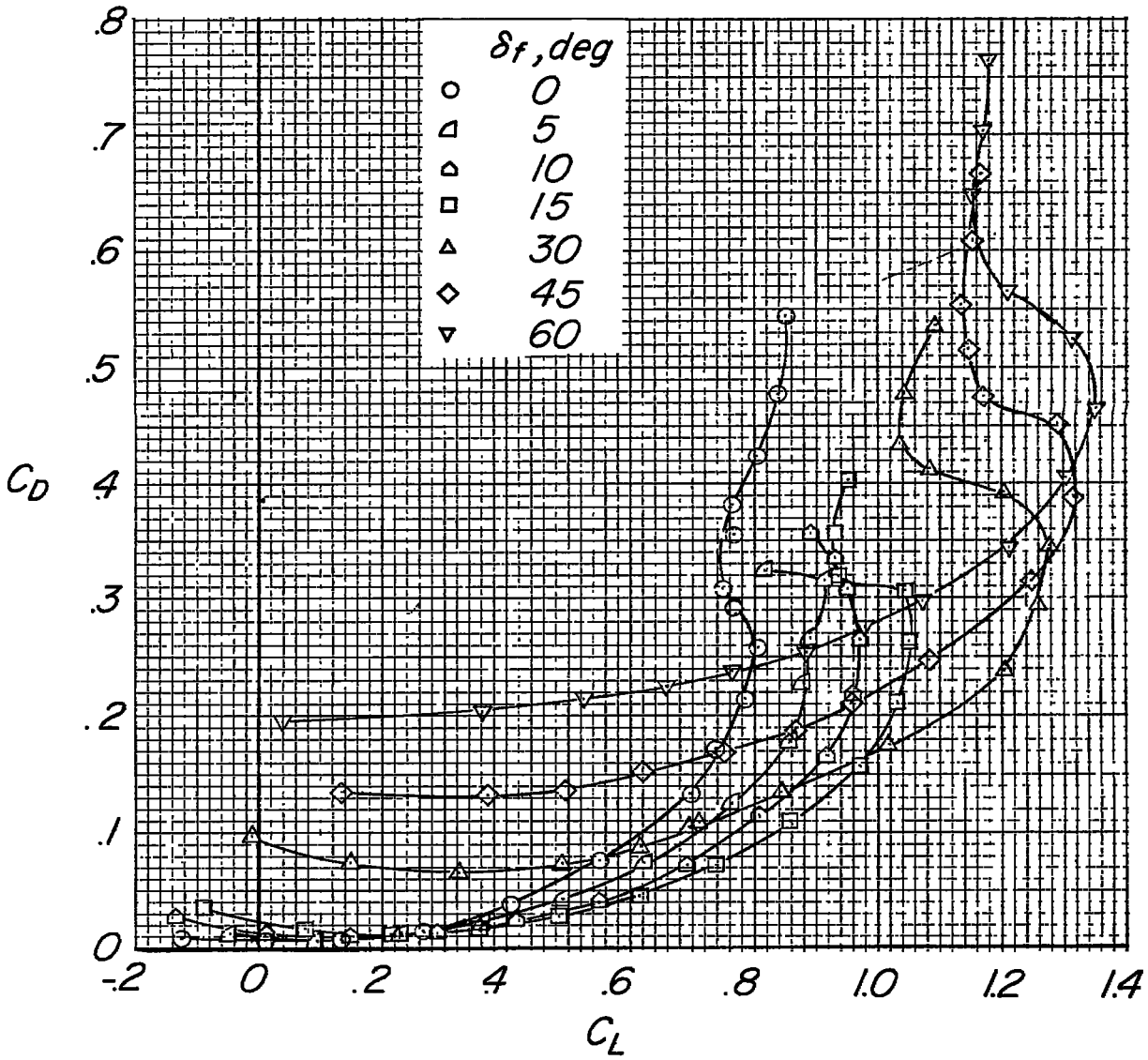
(b) C_D against C_L.

Figure 9.- Concluded.



(a) C_m and α_c against C_L.

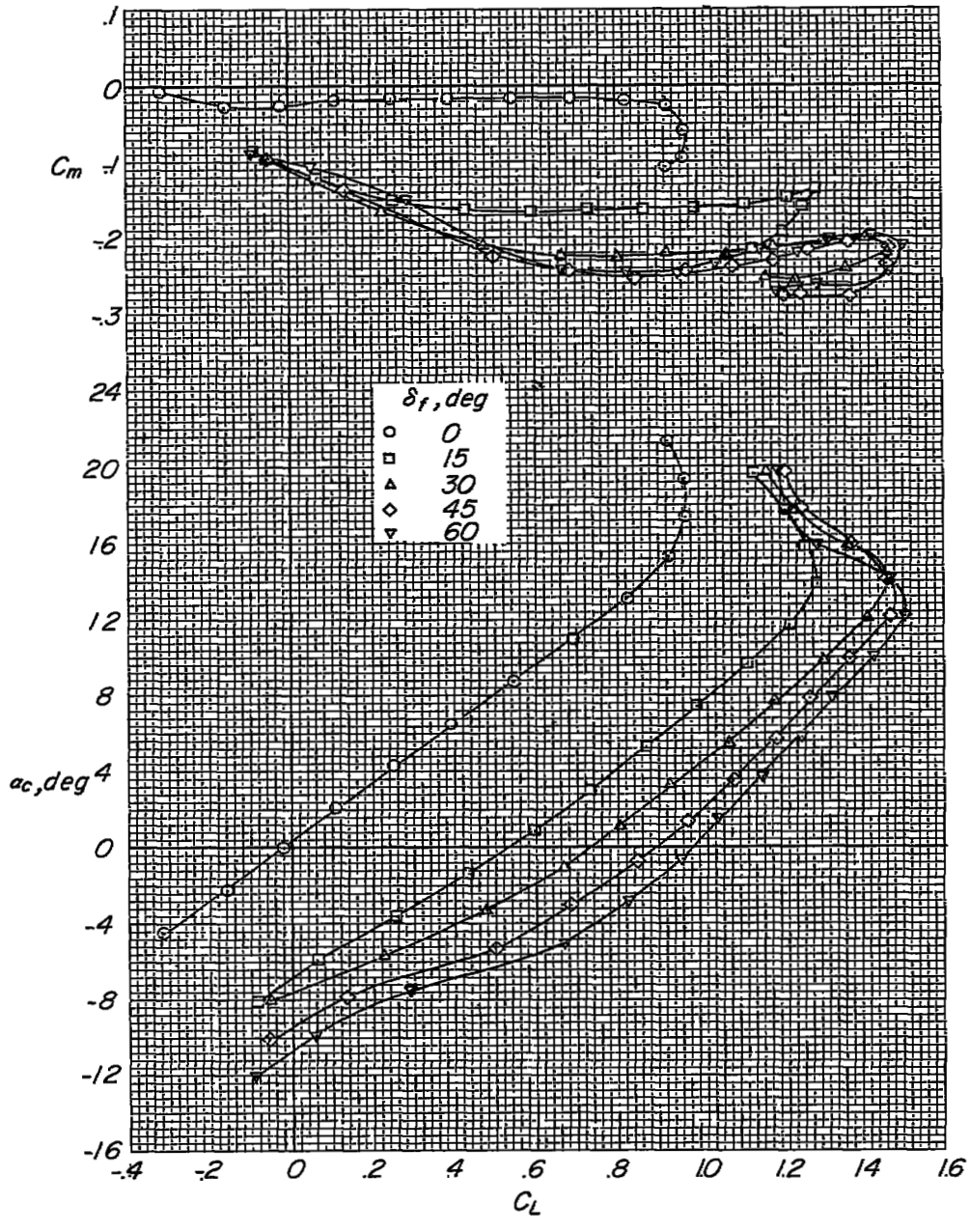
Figure 10.- Effect of flap deflection on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.25c_w$; $\delta_N = 0^o$; $C_\mu = 0$; $q \approx 25$ lb/sq ft.



(b) C_D against C_L.

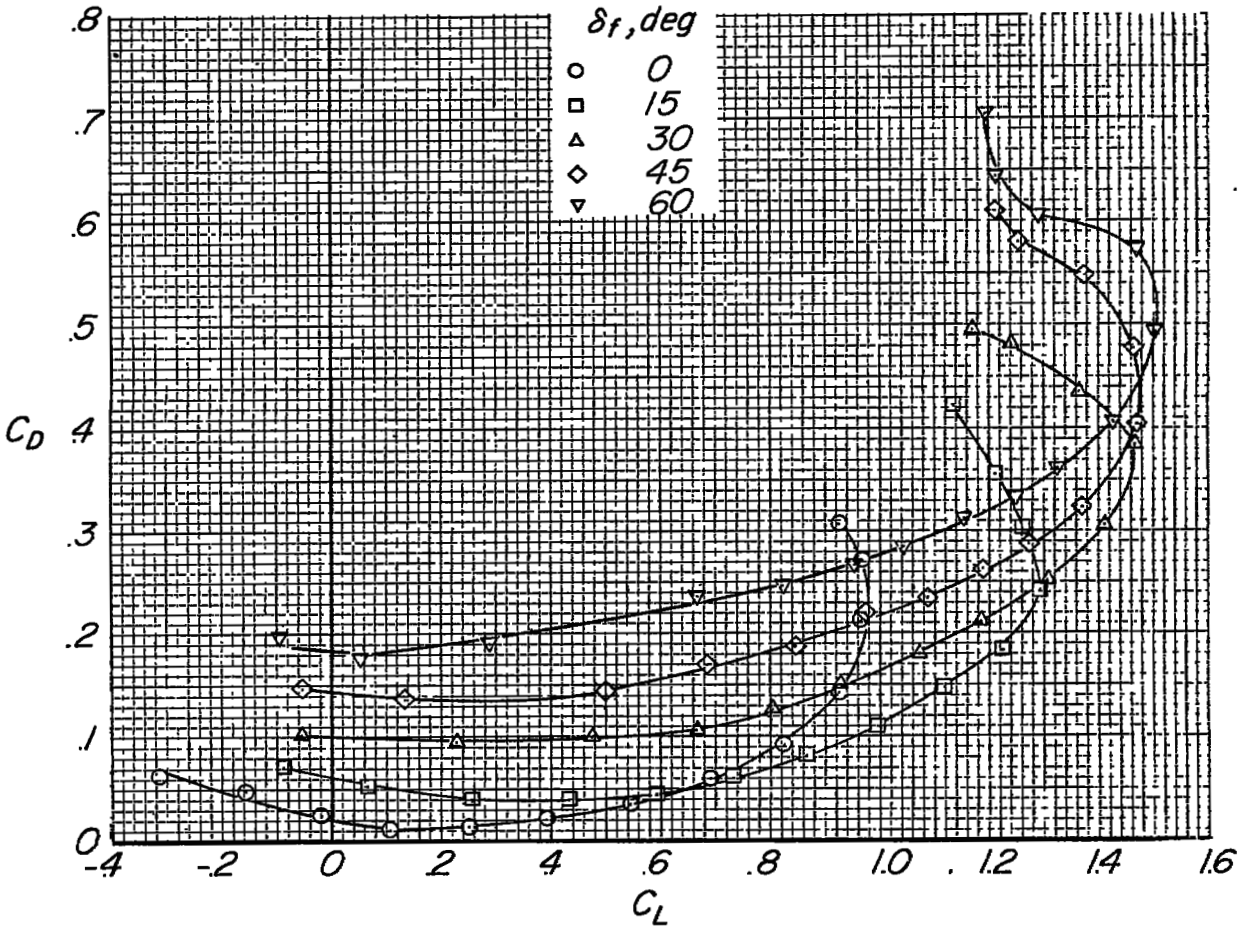
Figure 10.- Concluded.

10D



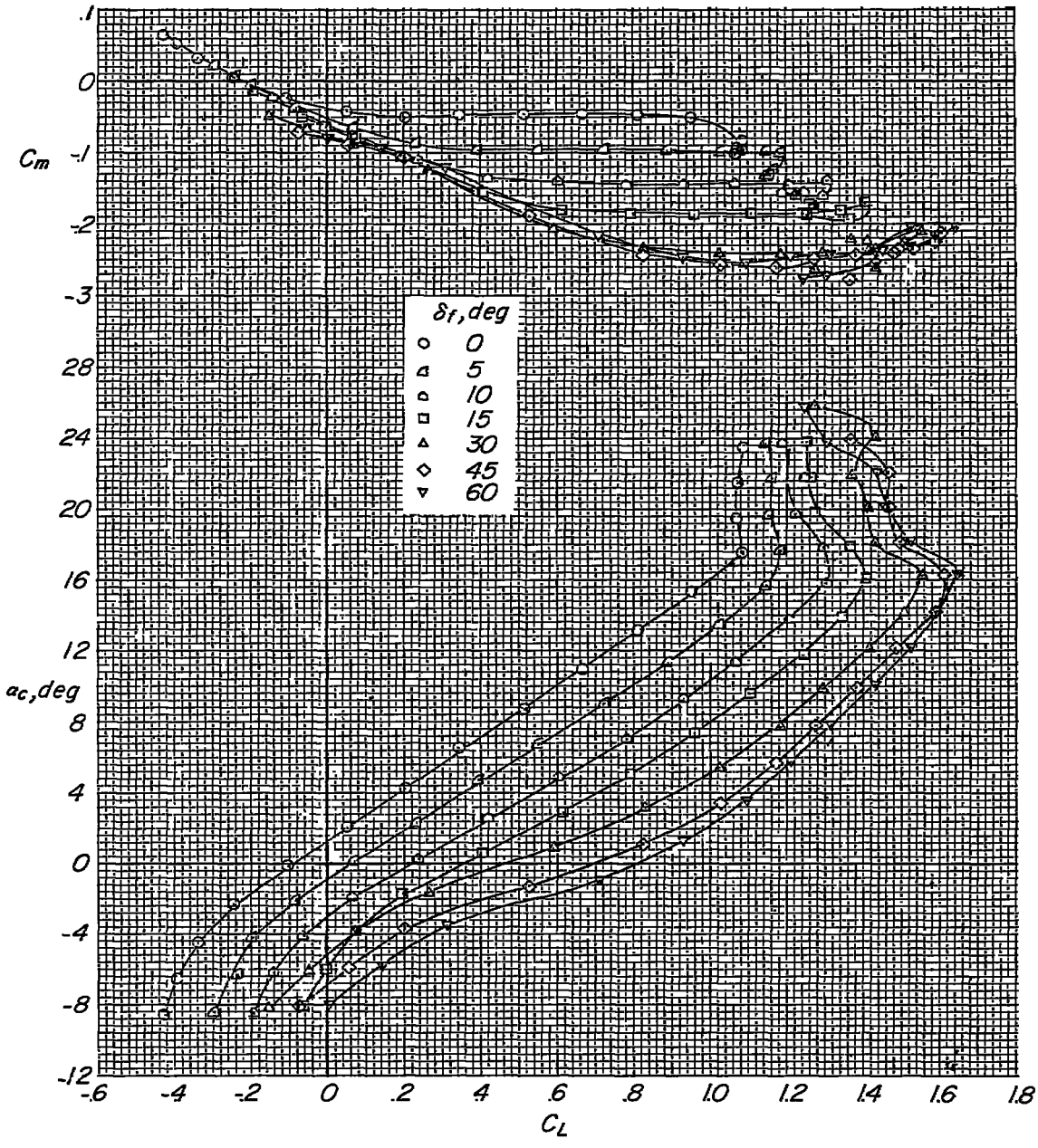
(a) C_m and α_c against C_L.

Figure 11.- Effect of flap deflection on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.25c_w$; $\delta_N = 15^\circ$; $C_\mu = 0$; $q \approx 25$ lb/sq ft.



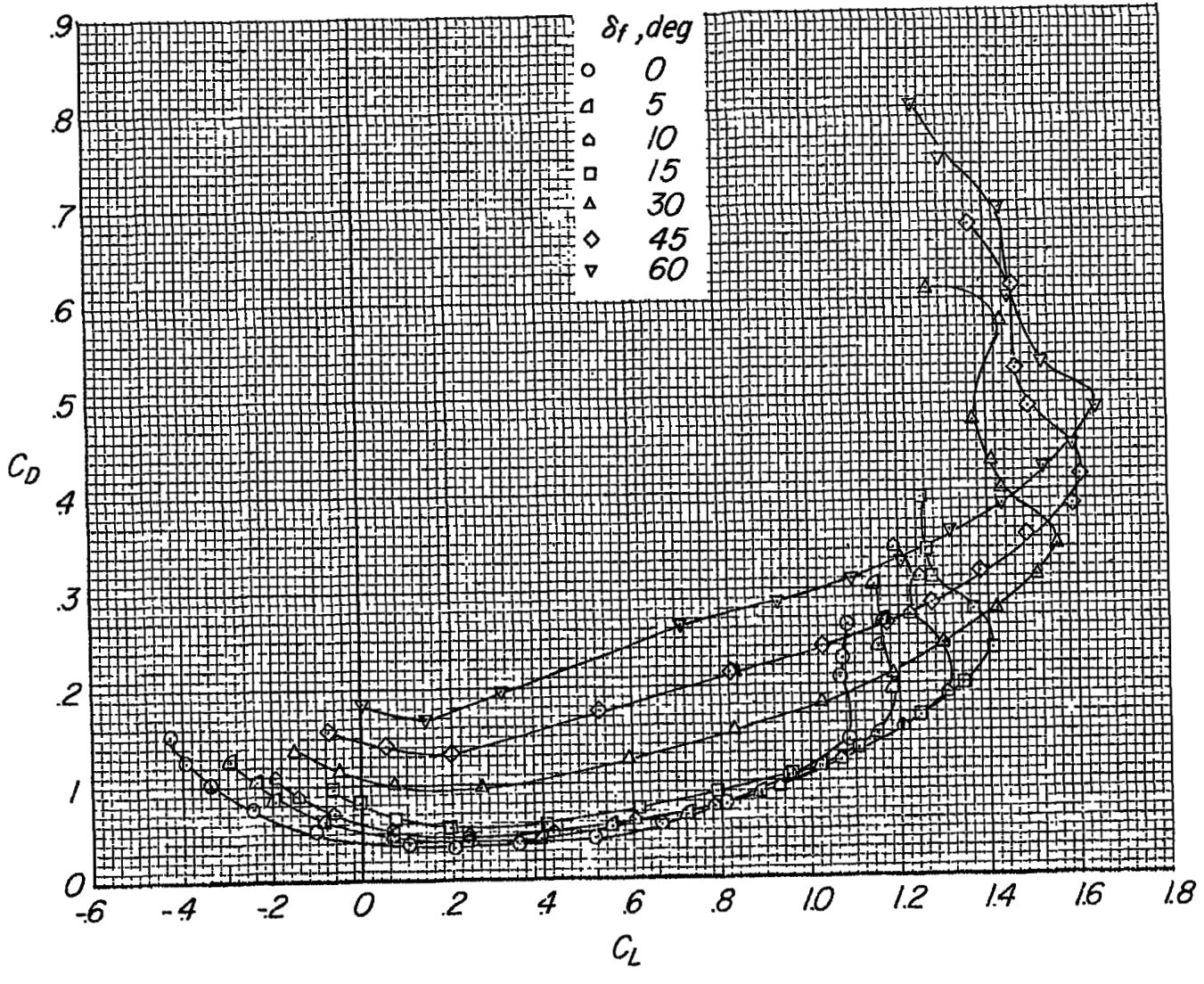
(b) C_D against C_L.

Figure 11.- Concluded.



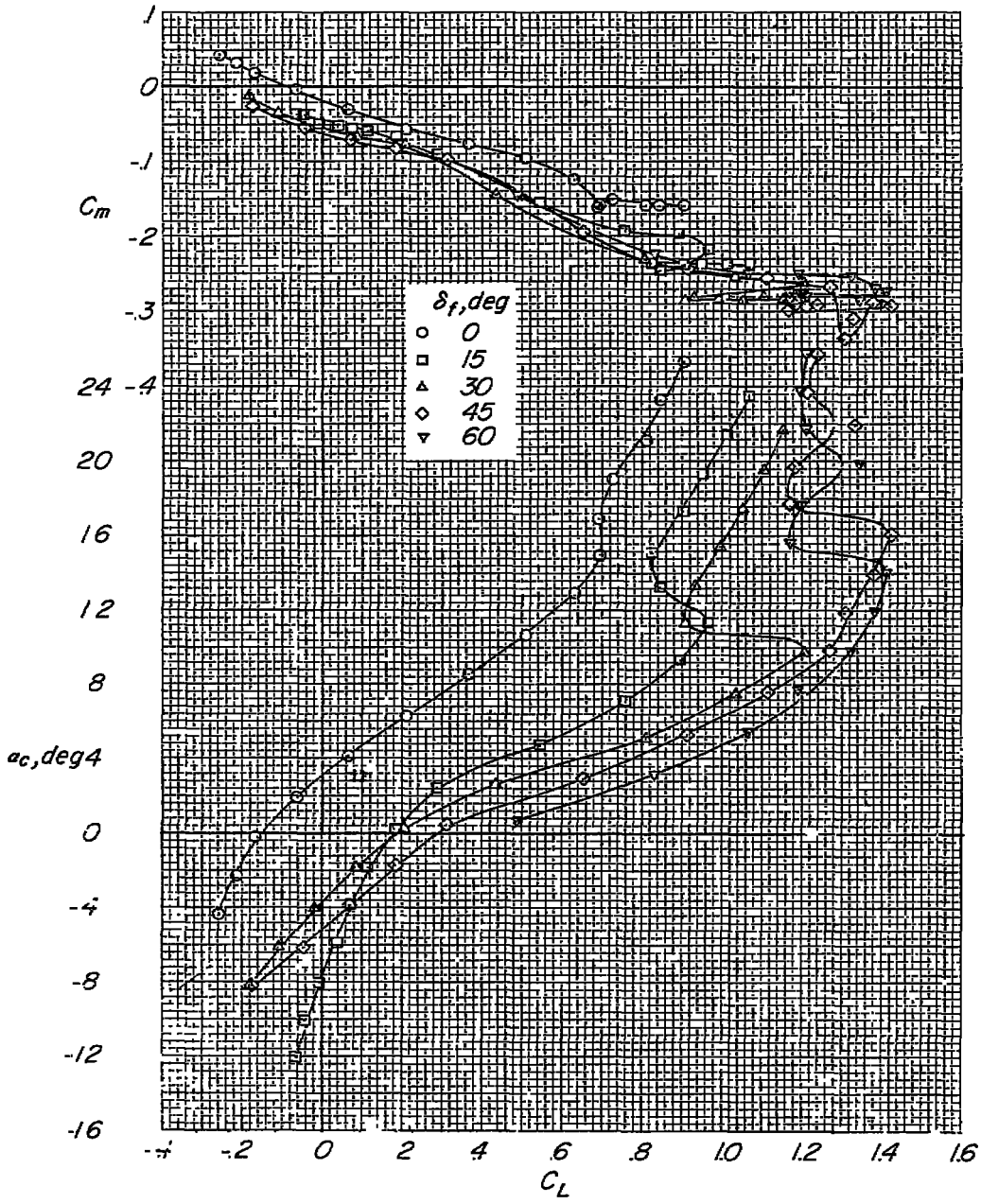
(a) C_m and α_c against C_L.

Figure 12.- Effect of flap deflection on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.25c_w$; $\delta_N = 30^\circ$; $C_\mu = 0$; $q \approx 25$ lb/sq ft.



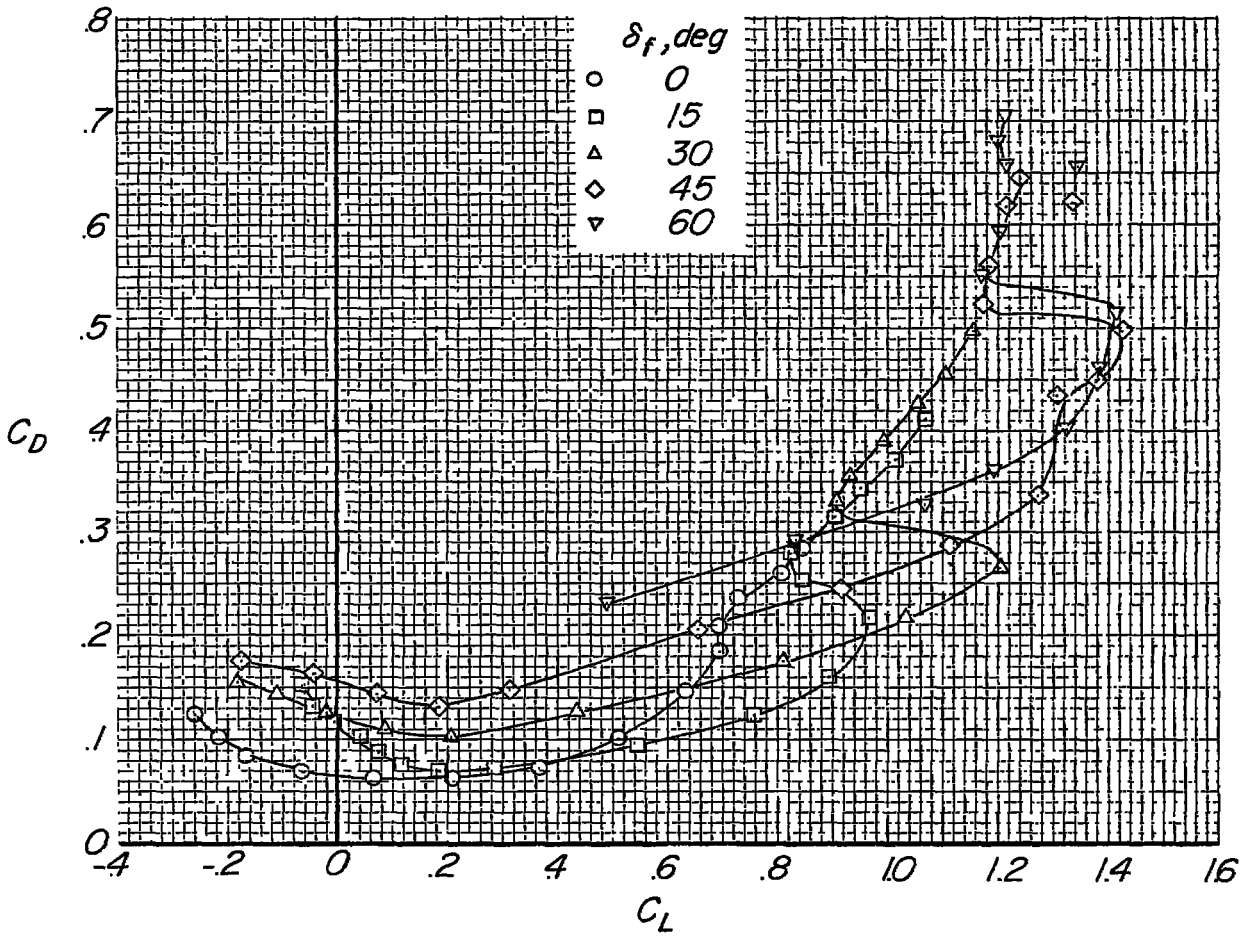
(b) C_D against C_L.

Figure 12.- Concluded.



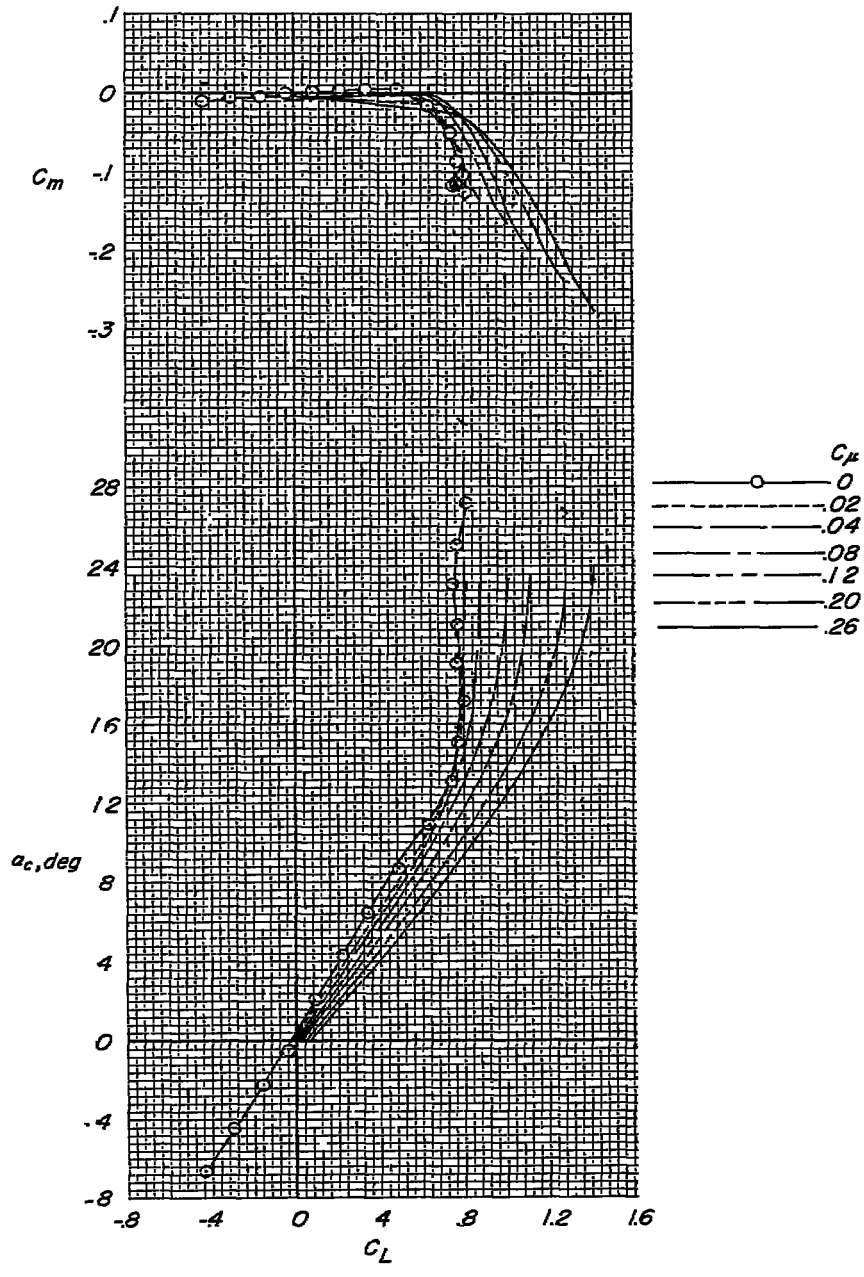
(a) C_m and α_c against C_L.

Figure 13.- Effect of flap deflection on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.25c_w$; $\delta_N = 45^\circ$; $C_\mu = 0$; $q \approx 25$ lb/sq ft.



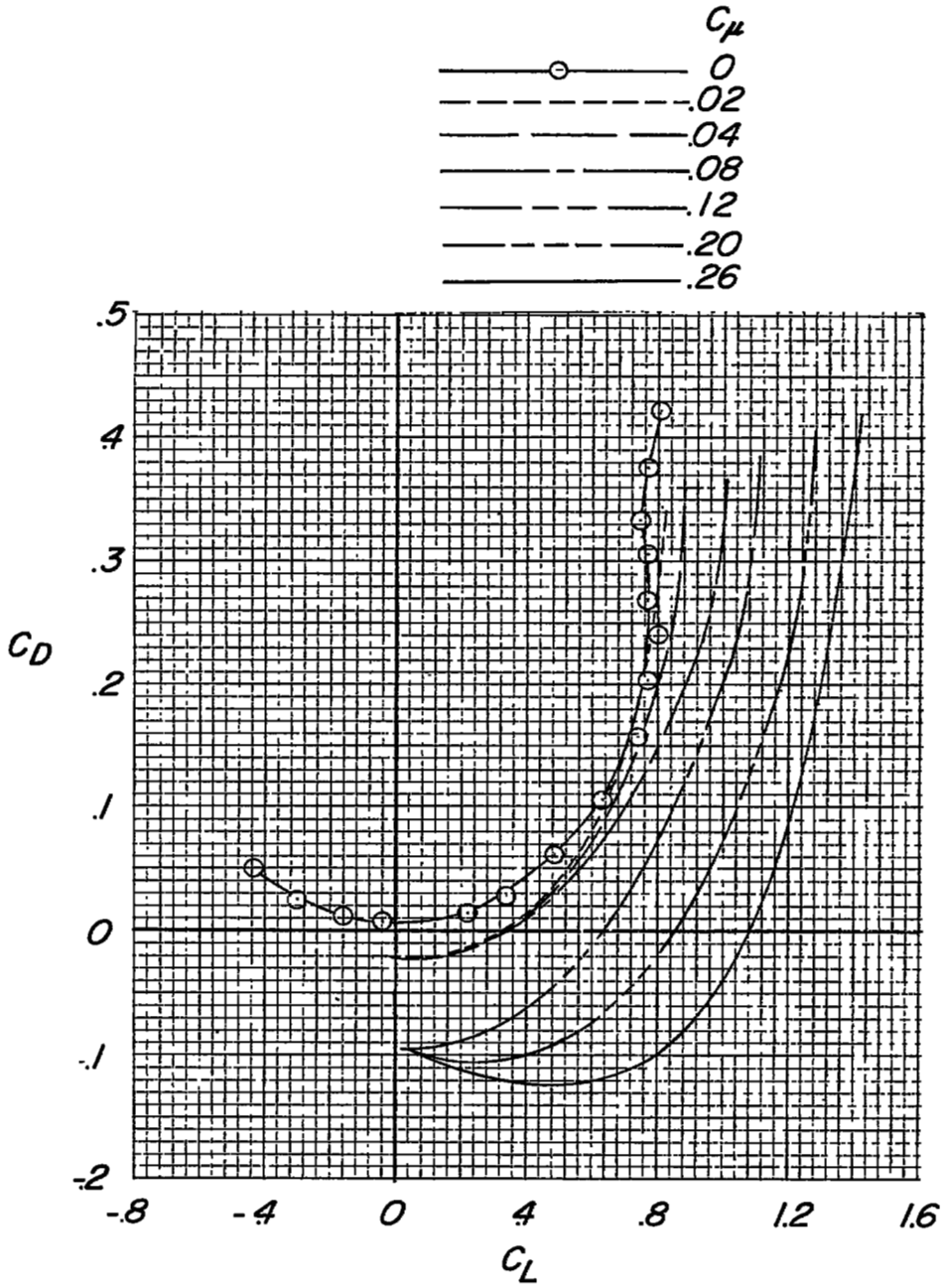
(b) C_D against C_L.

Figure 13.- Concluded.



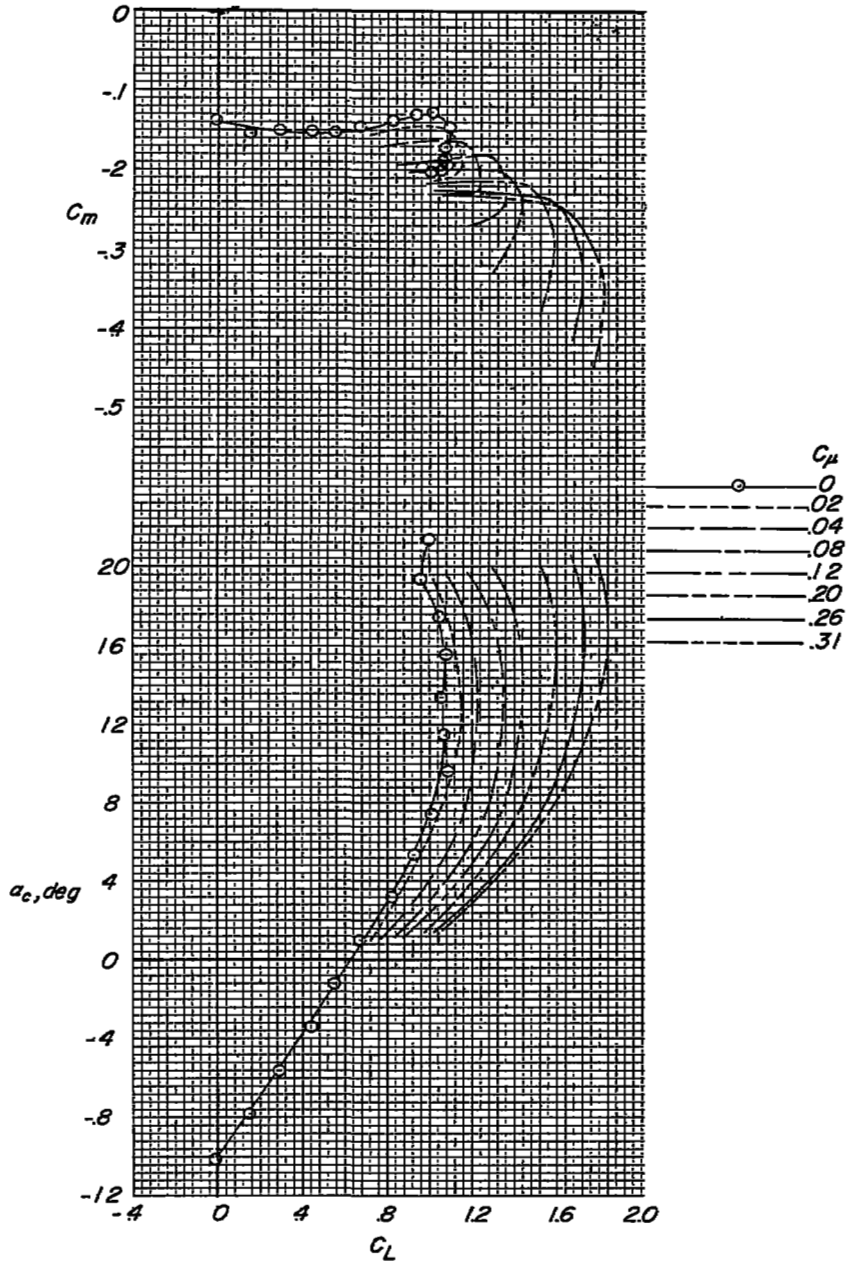
(a) C_m and α_c against C_L.

Figure 14.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.33c_w$; $\delta_f = 0^\circ$; $\delta_N = 0^\circ$; $q \approx 12.5$ lb/sq ft.



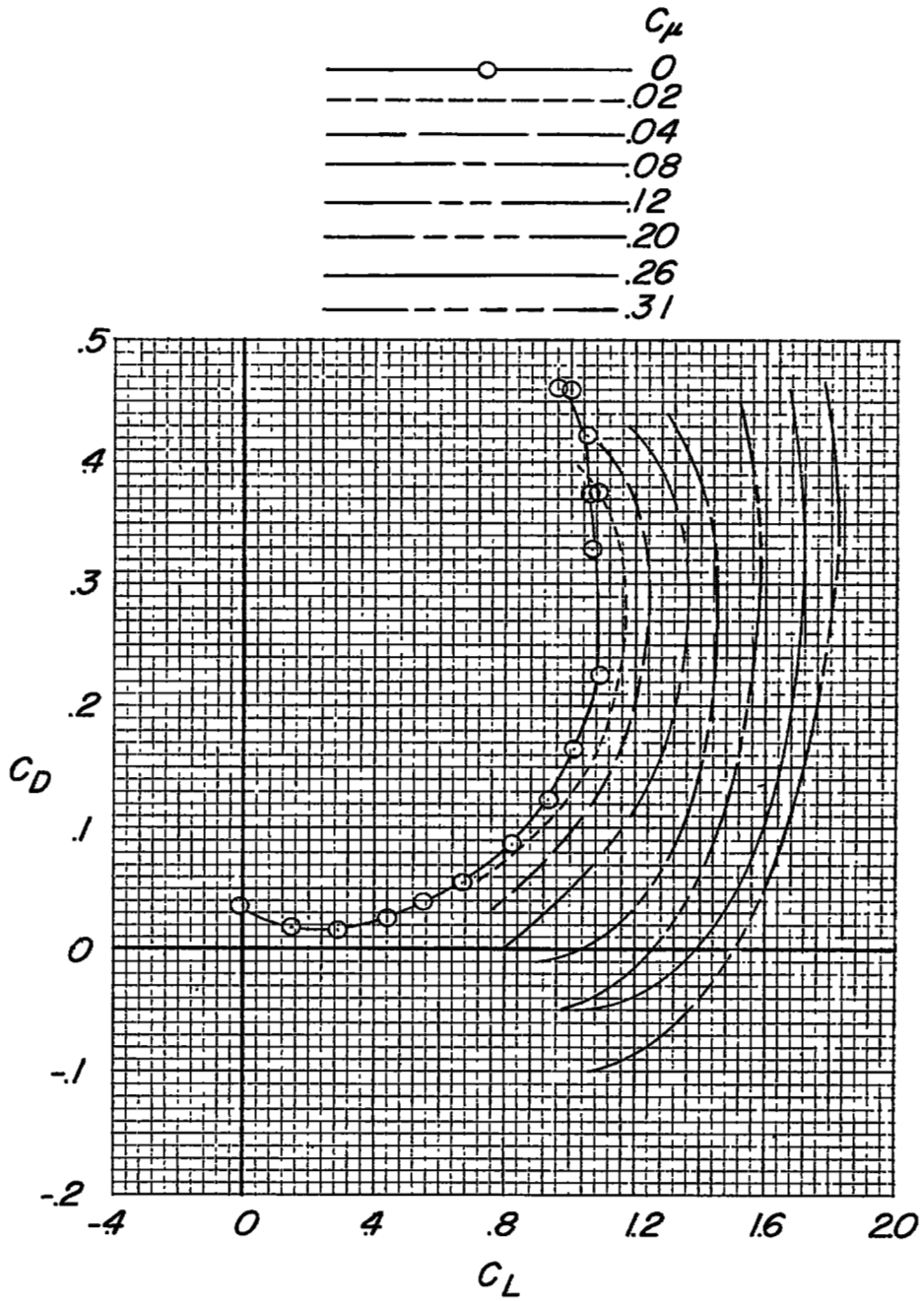
(b) C_D against C_L.

Figure 14.- Concluded.



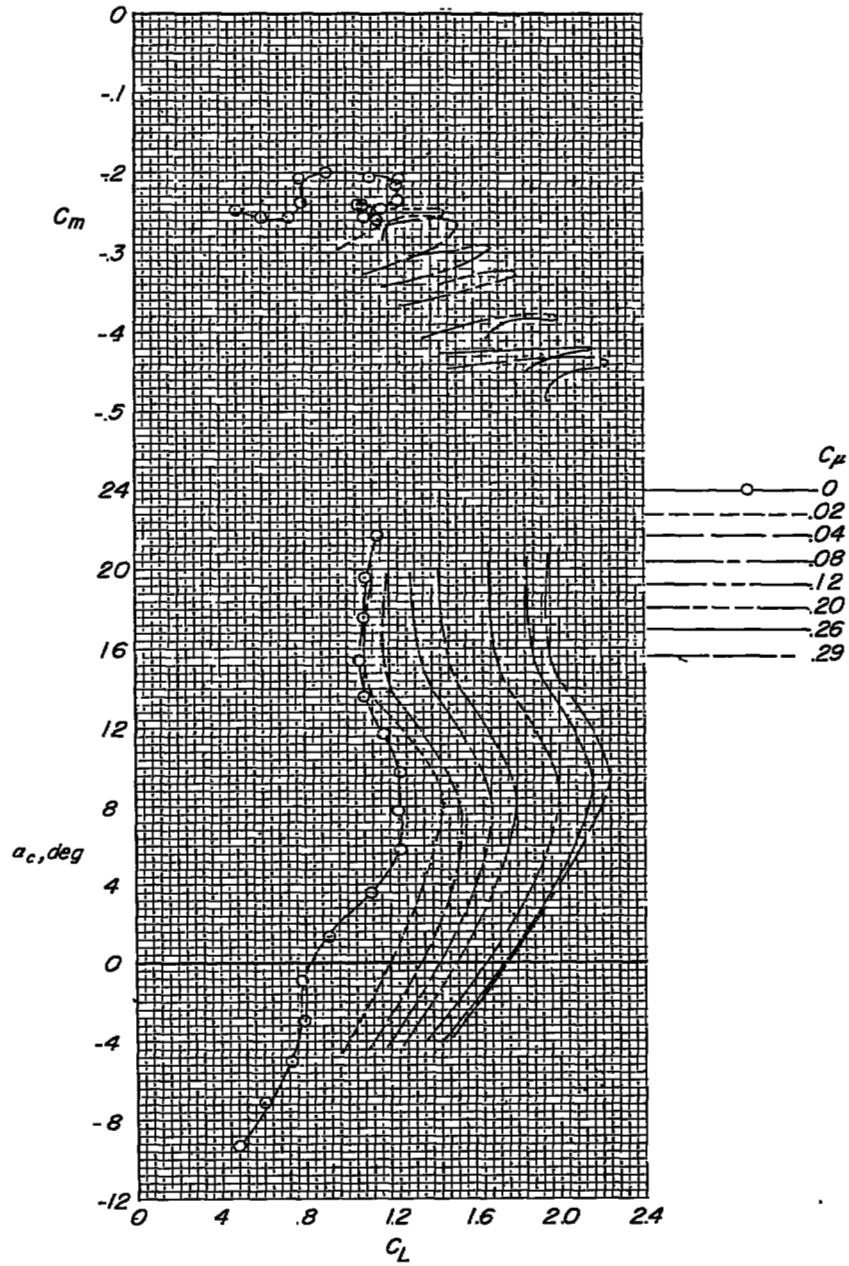
(a) C_m and α_c against C_L.

Figure 15.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.33c_w$; $\delta_f = 15^\circ$; $\delta_N = 0^\circ$; $q \approx 12.5$ lb/sq ft.



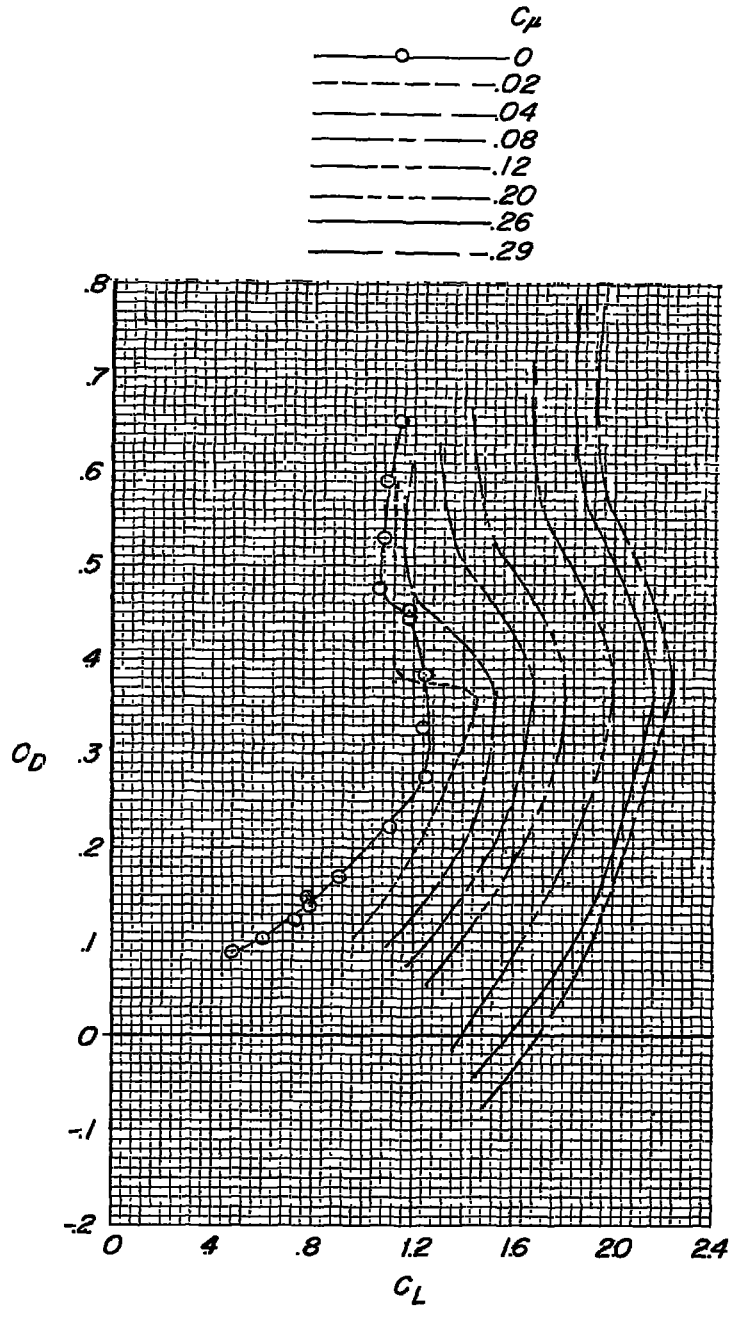
(b) C_D against C_L.

Figure 15.- Concluded.



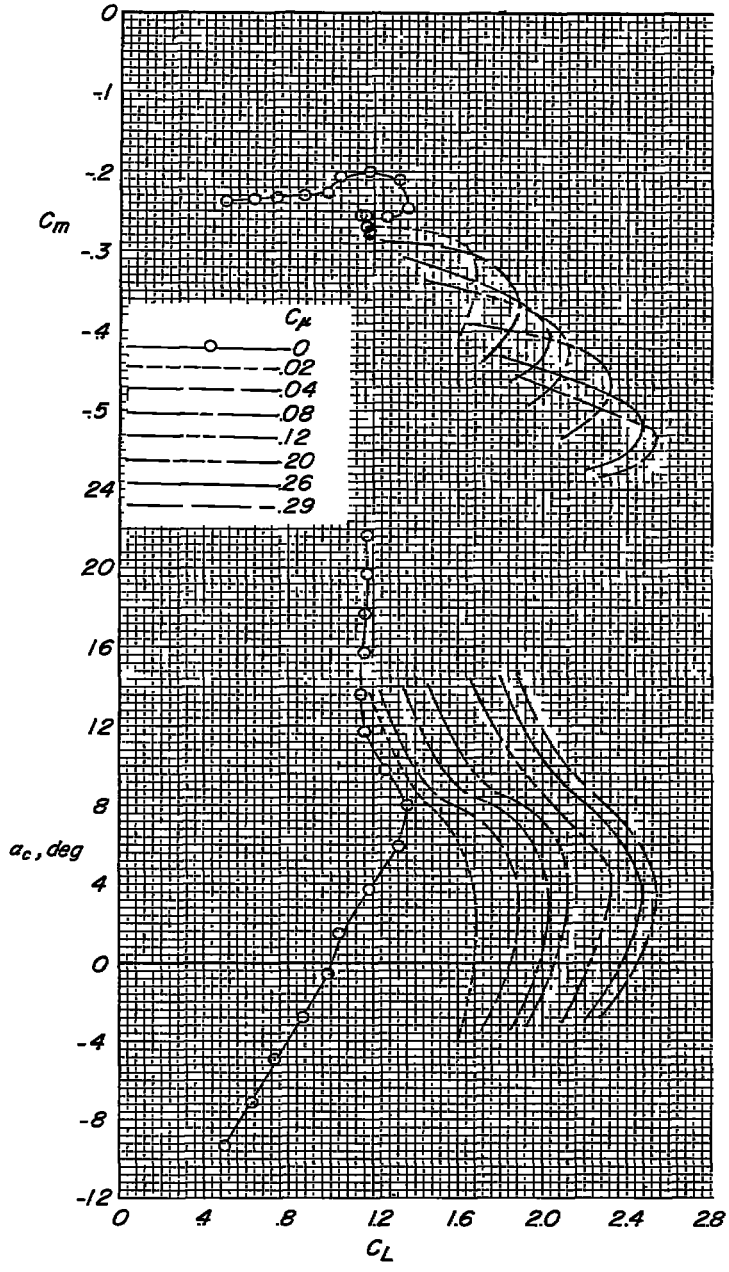
(a) C_m and α_c against C_L.

Figure 16.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.33c_w$; $\delta_f = 30^\circ$; $\delta_N = 0^\circ$; $q \approx 12.5$ lb/sq ft.



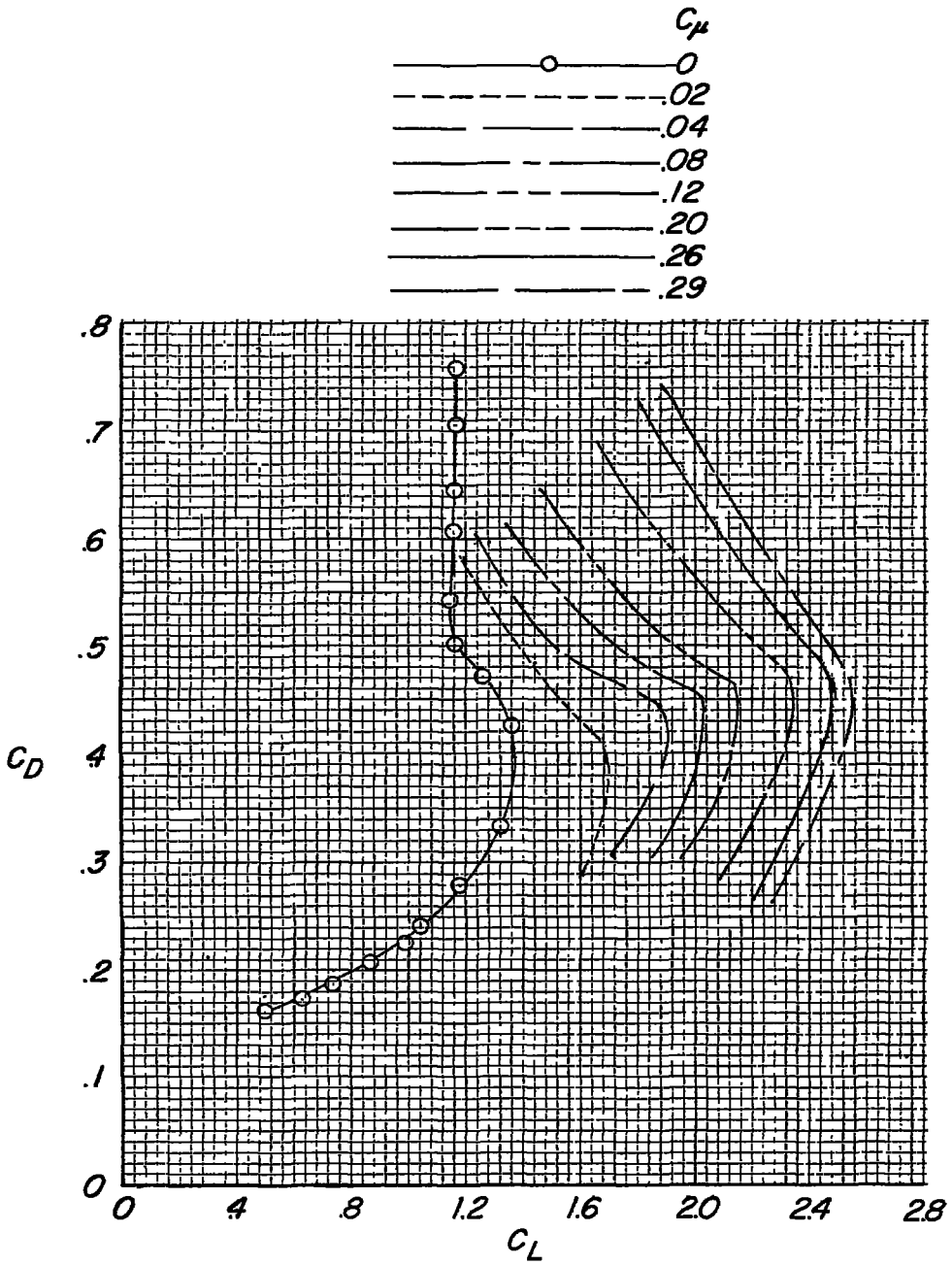
(b) C_D against C_L.

Figure 16.- Concluded.



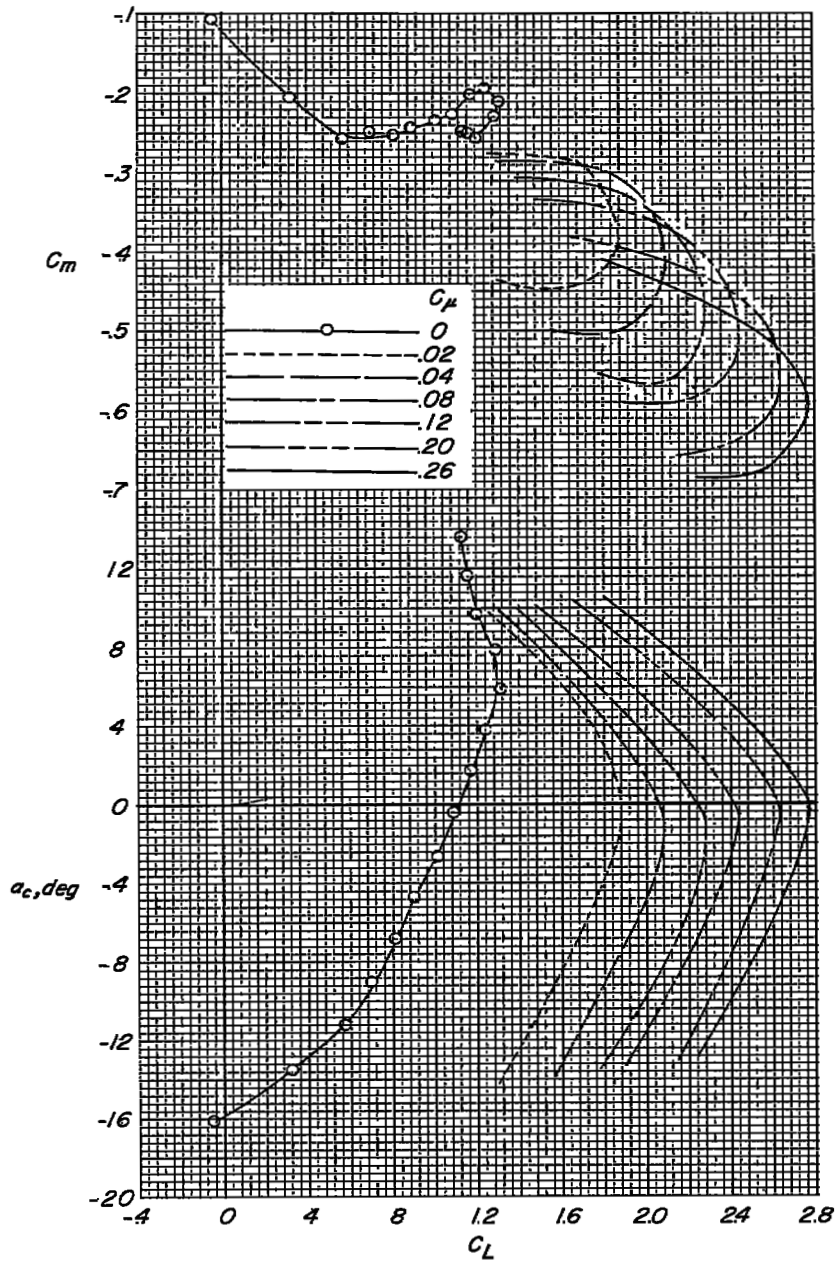
(a) C_m and α_c against C_L.

Figure 17.- Effect of momentum coefficients on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.33c_w$; $\delta_f = 45^\circ$; $\delta_N = 0^\circ$; $q \approx 12.5$ lb/sq ft.



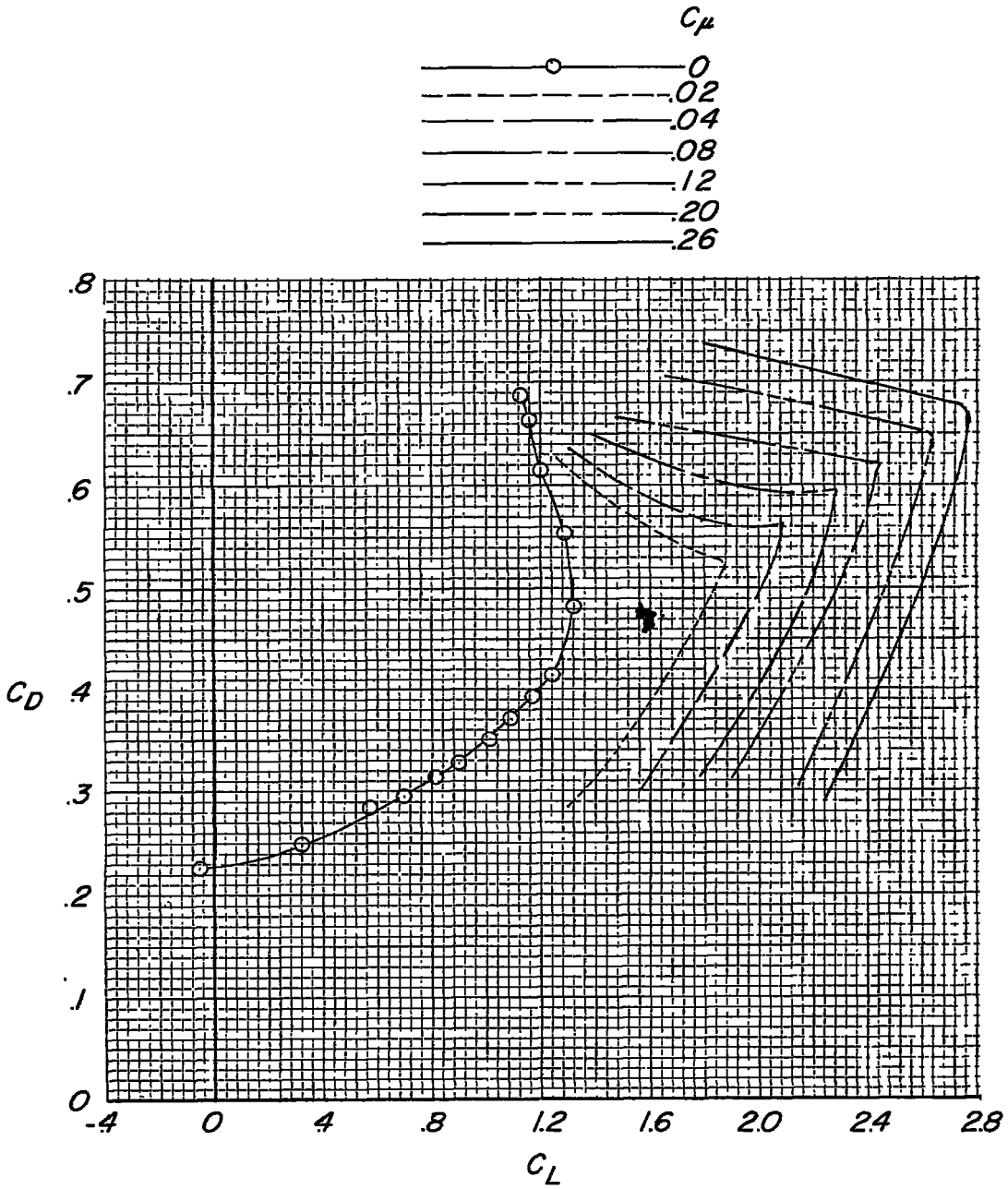
(b) C_D against C_L.

Figure 17.- Concluded.



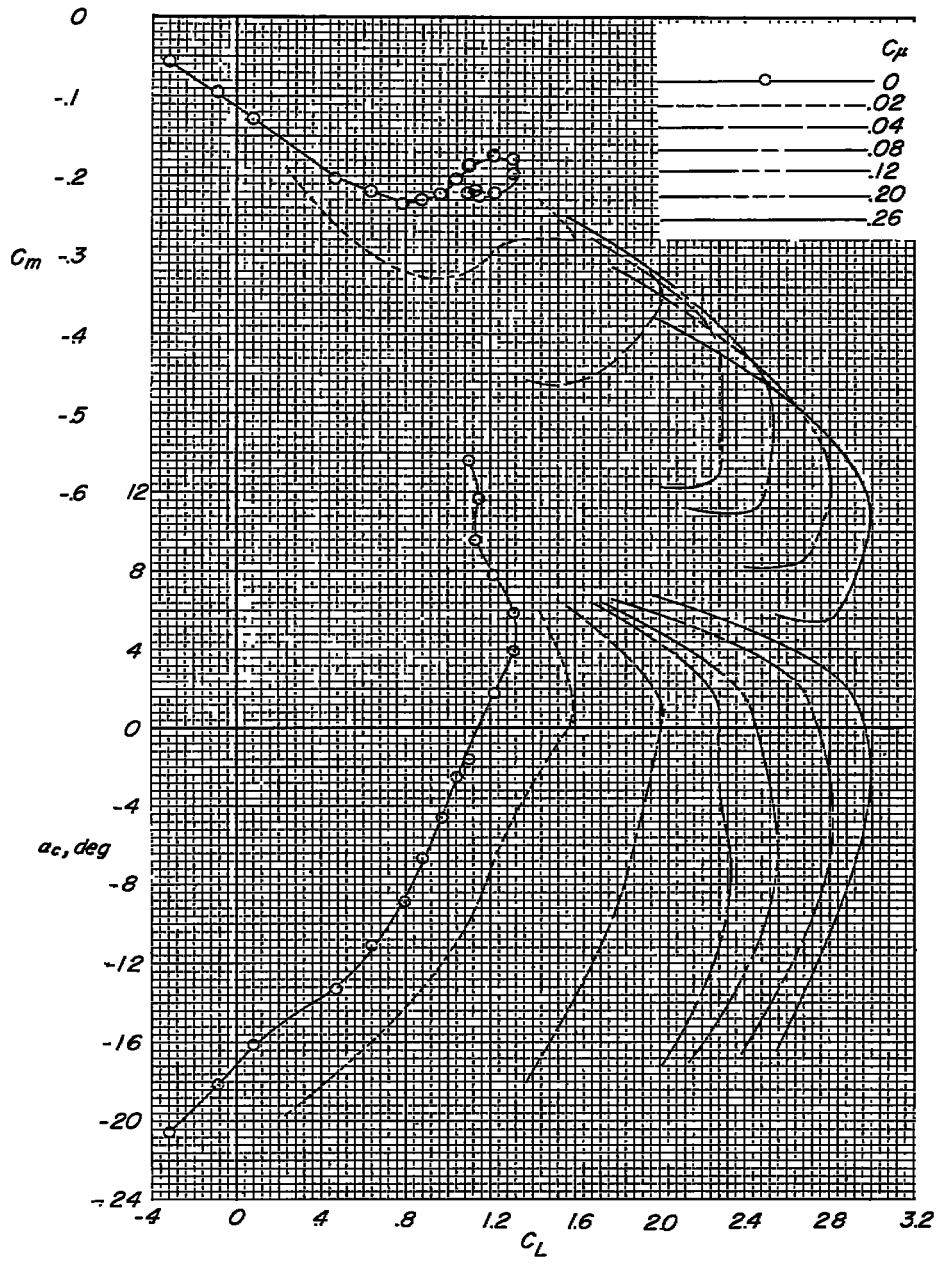
(a) C_m and α_c against C_L.

Figure 18.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.33c_w$; $\delta_f = 60^\circ$; $\delta_N = 0^\circ$; $q \approx 12.5$ lb/sq ft.



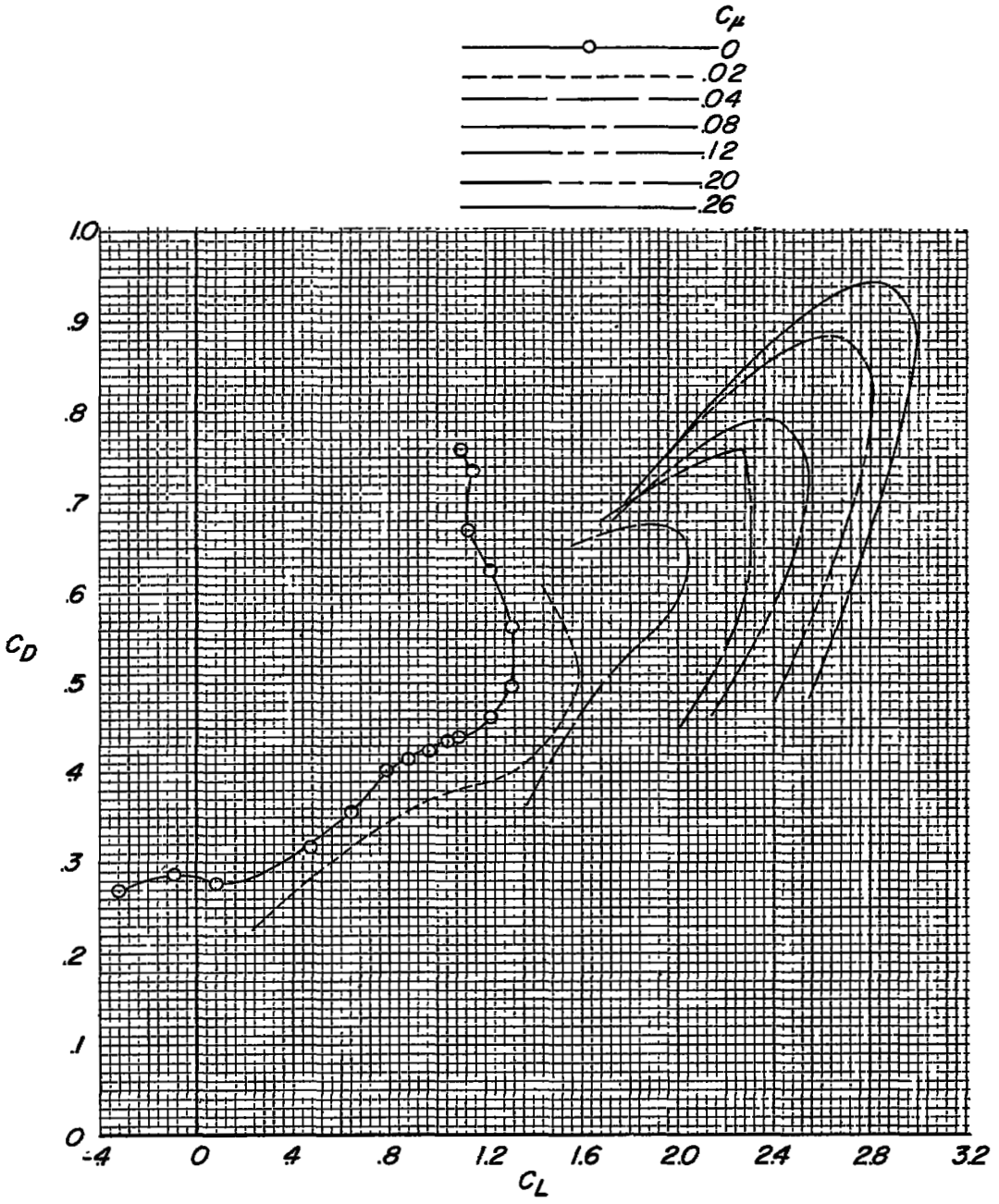
(b) C_D against C_L.

Figure 18.- Concluded.



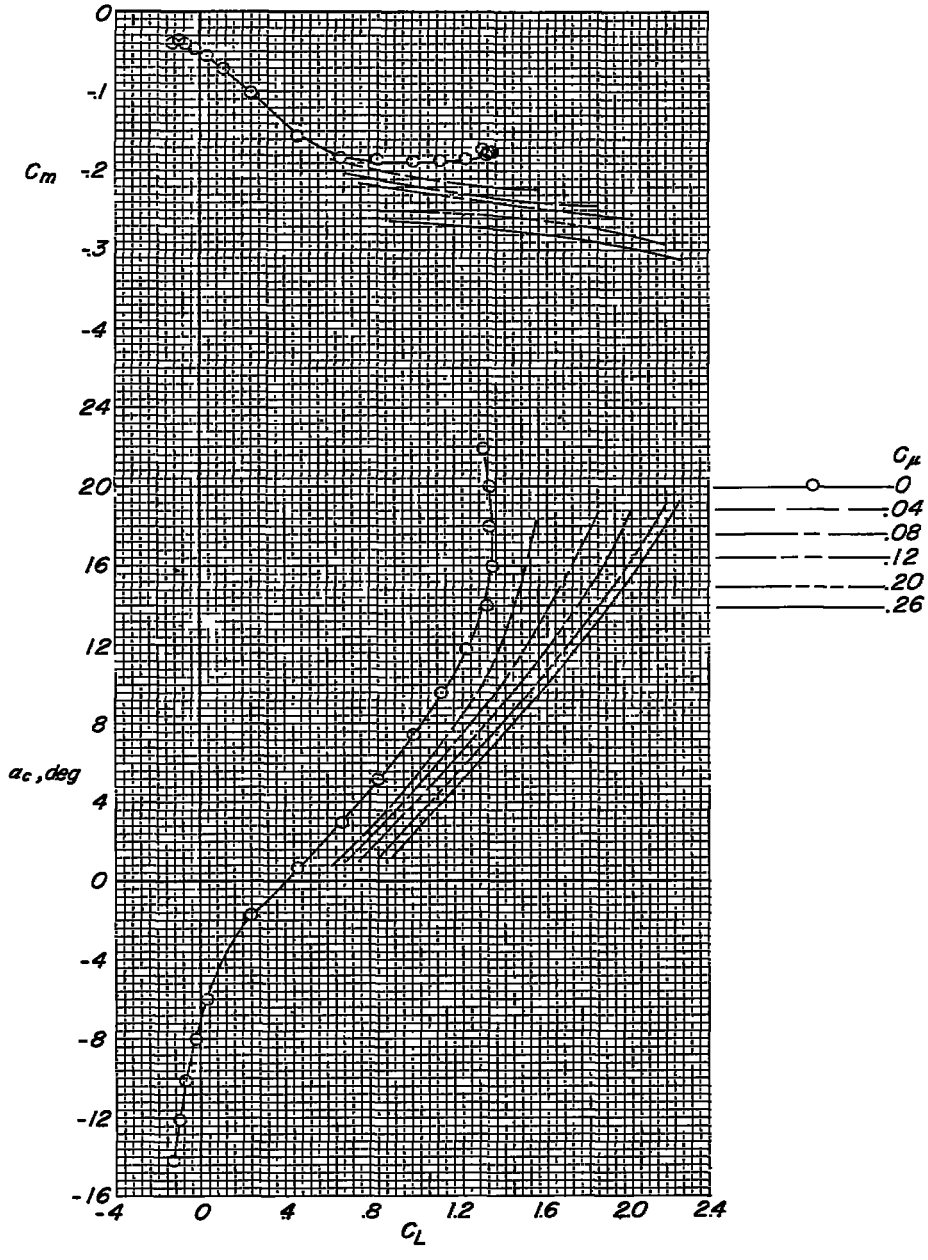
(a) C_m and α_c against C_L.

Figure 19.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.33c_w$; $\delta_f = 75^\circ$; $\delta_N = 0^\circ$; $q \approx 12.5$ lb/sq ft.



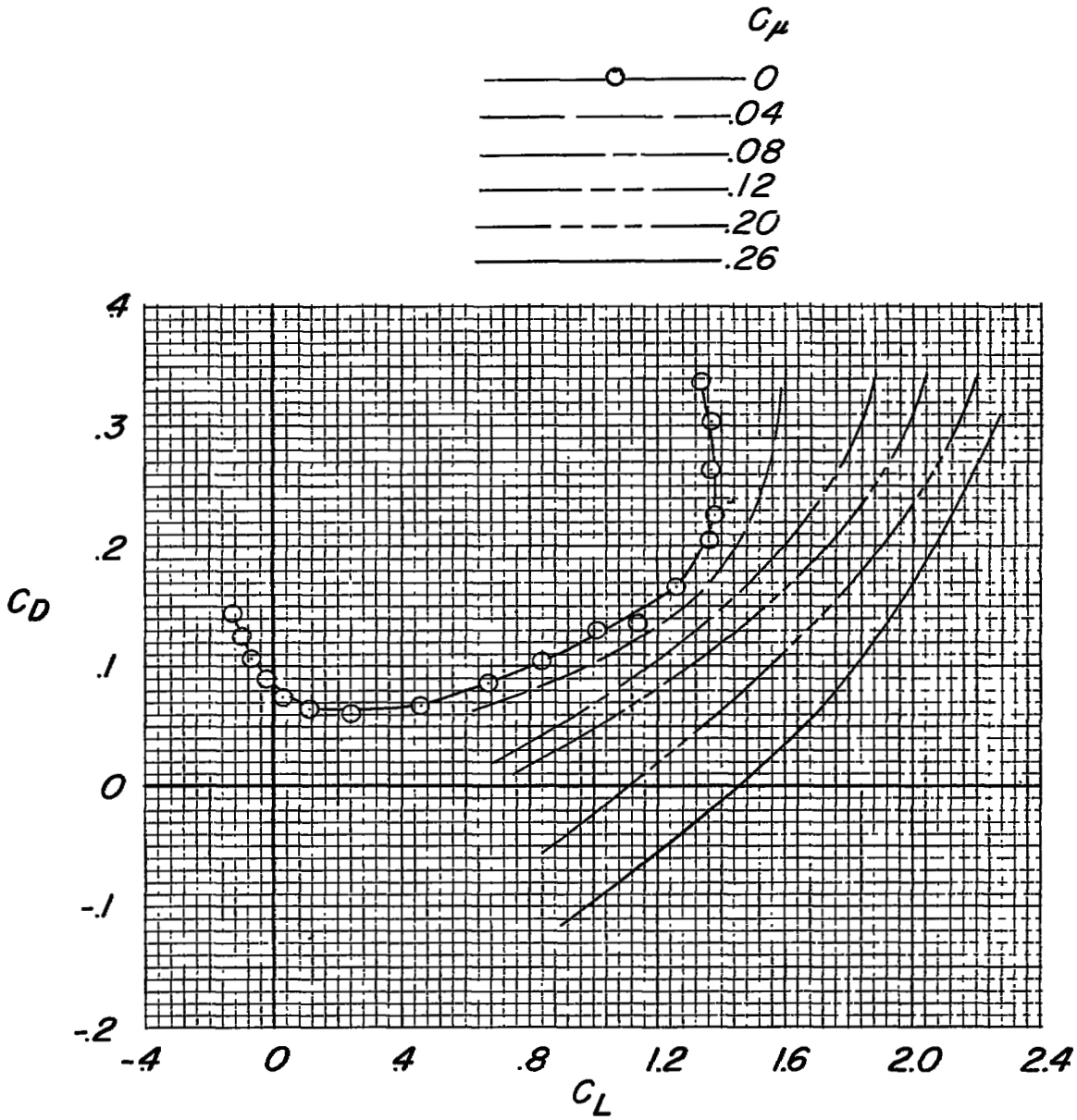
(b) C_D against C_L.

Figure 19.- Concluded.



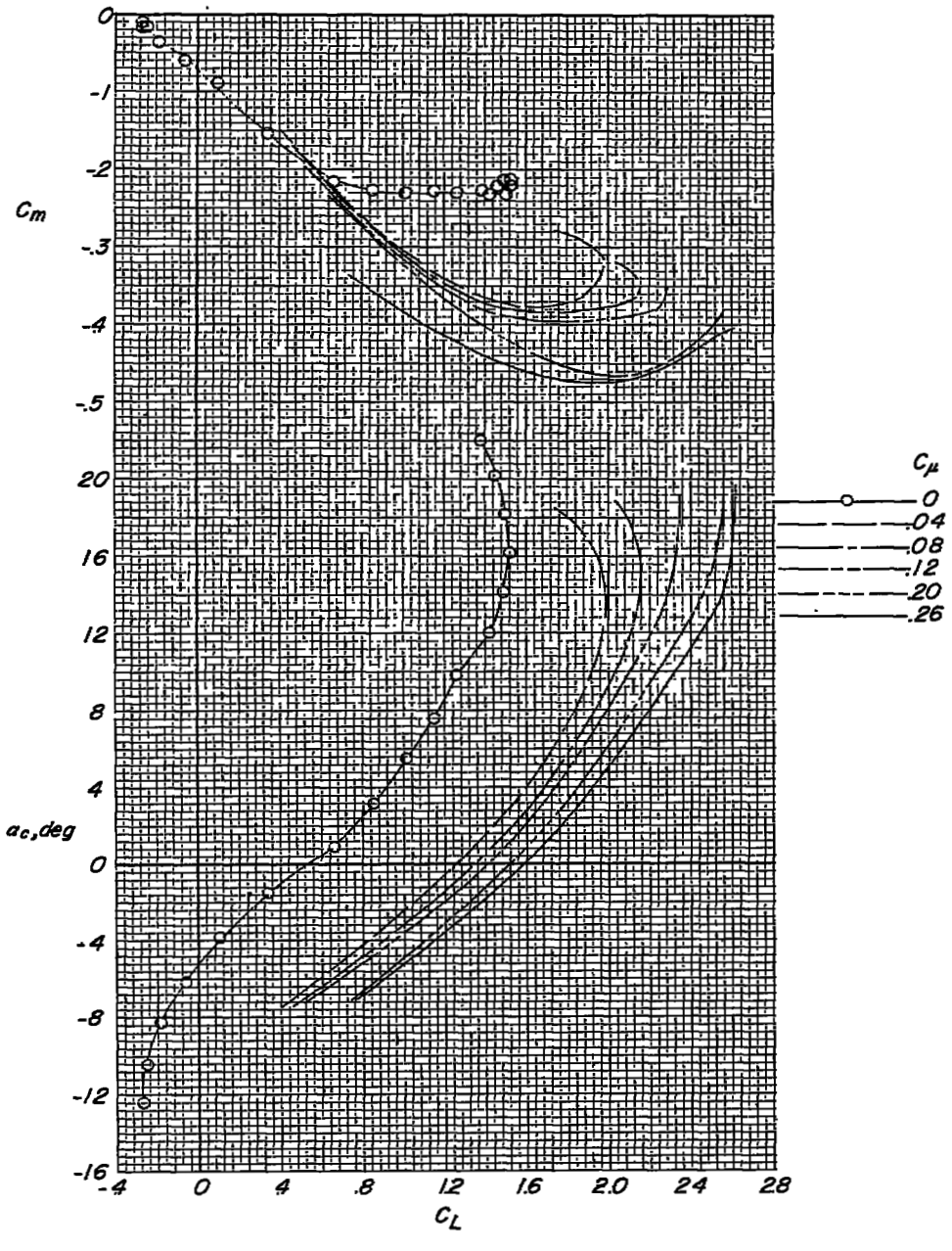
(a) C_m and α_c against C_L.

Figure 20.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.33c_w$; $\delta_f = 15^\circ$; $\delta_N = 30^\circ$; $q \approx 12.5$ lb/sq ft.



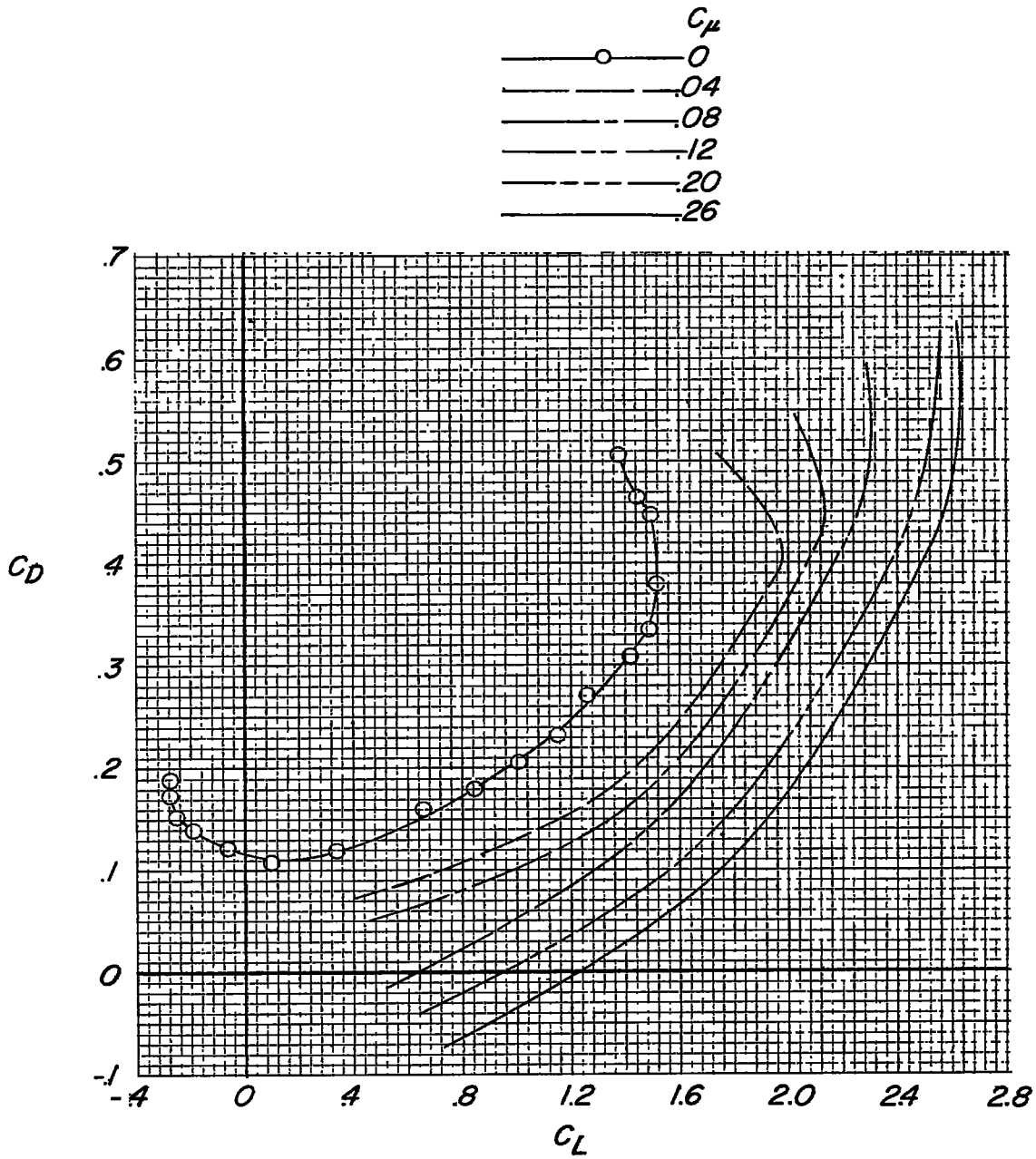
(b) C_D against C_L.

Figure 20.- Concluded.



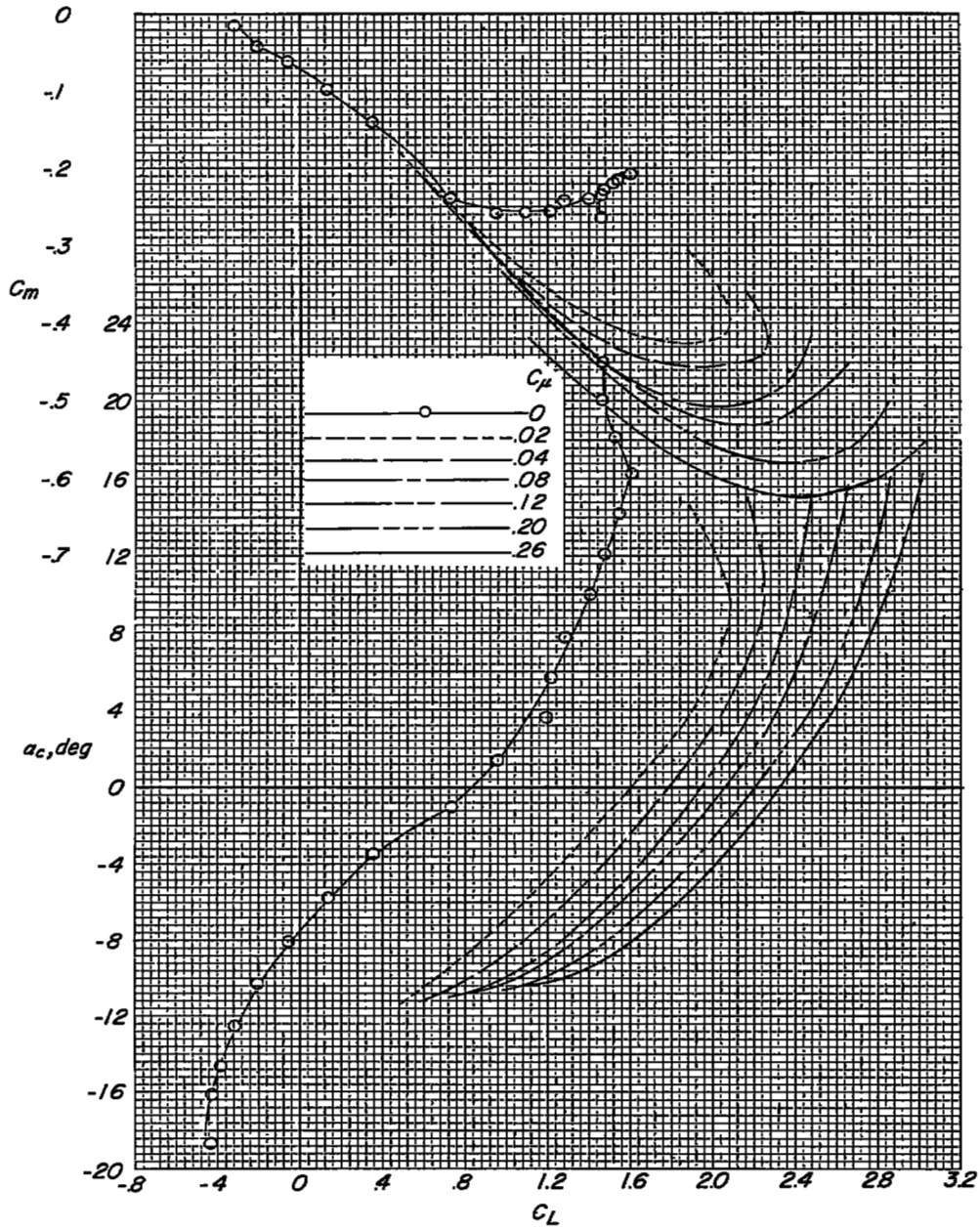
(a) C_m and α_c against C_L.

Figure 21.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.33c_w$; $\delta_f = 30^\circ$; $\delta_N = 30^\circ$; $q \approx 12.5$ lb/sq ft.



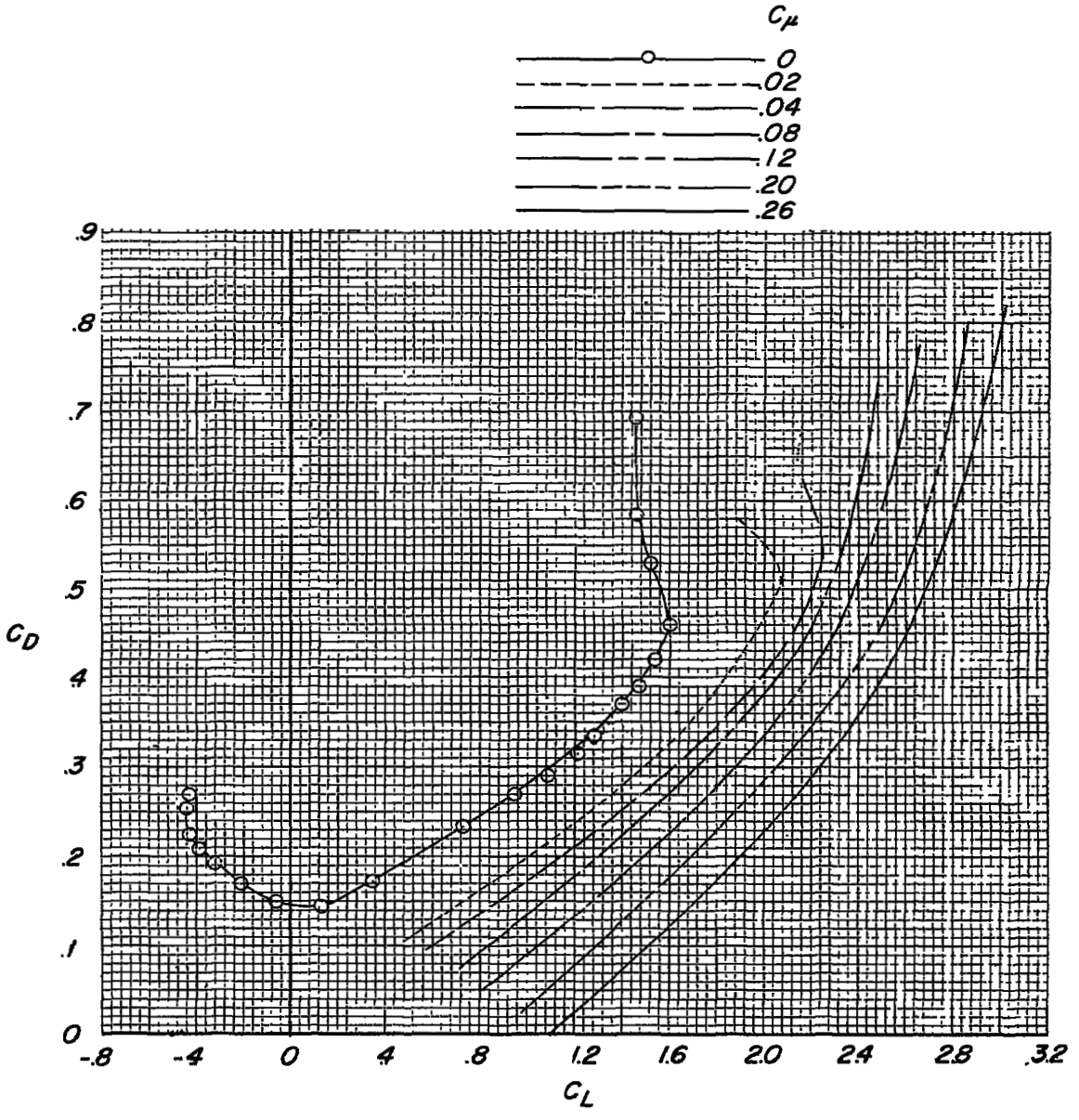
(b) C_D against C_L.

Figure 21.- Concluded.



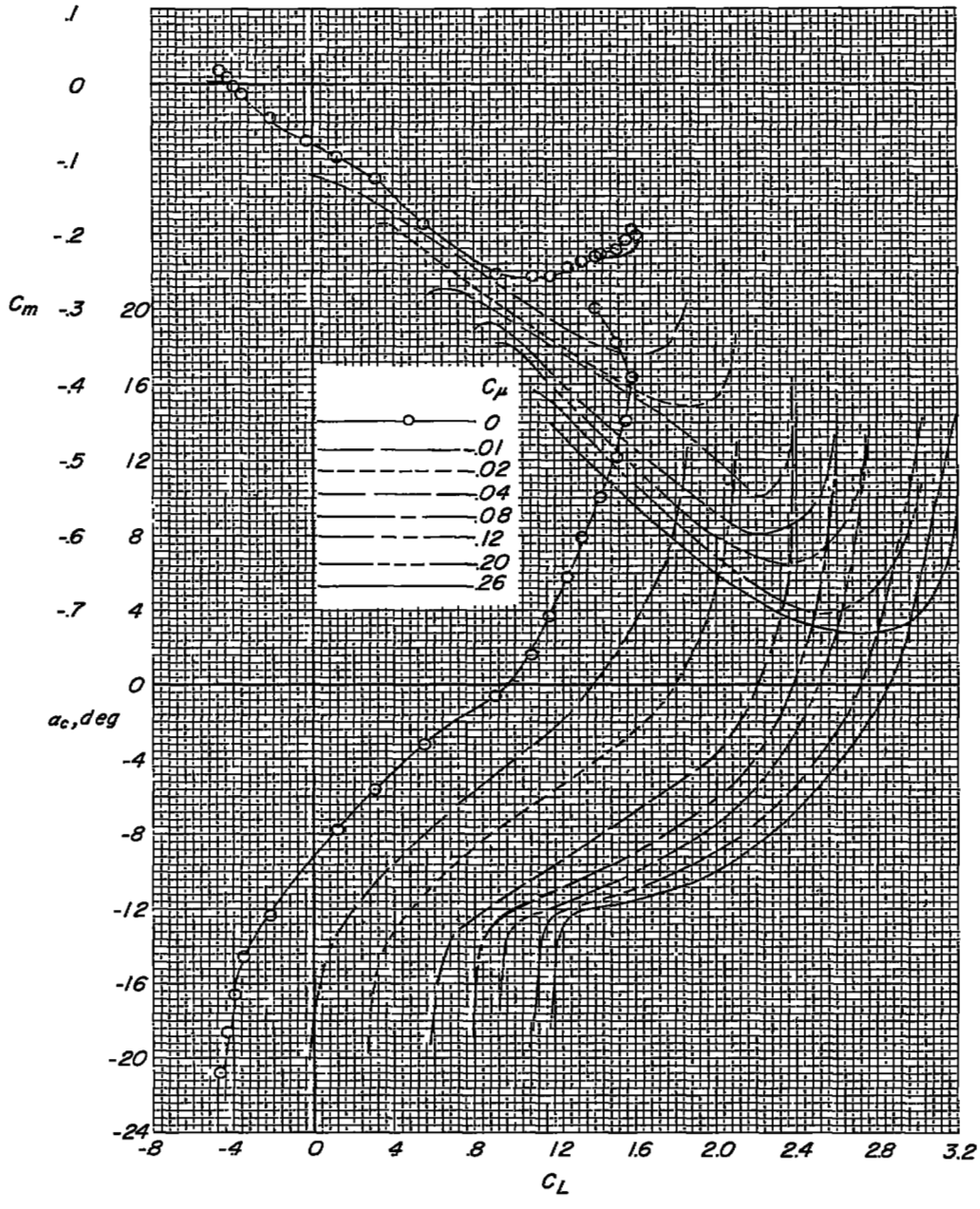
(a) C_m and α_c against C_L.

Figure 22.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.33c_w$; $\delta_f = 45^\circ$; $\delta_N = 30^\circ$; $q \approx 12.5$ lb/sq ft.



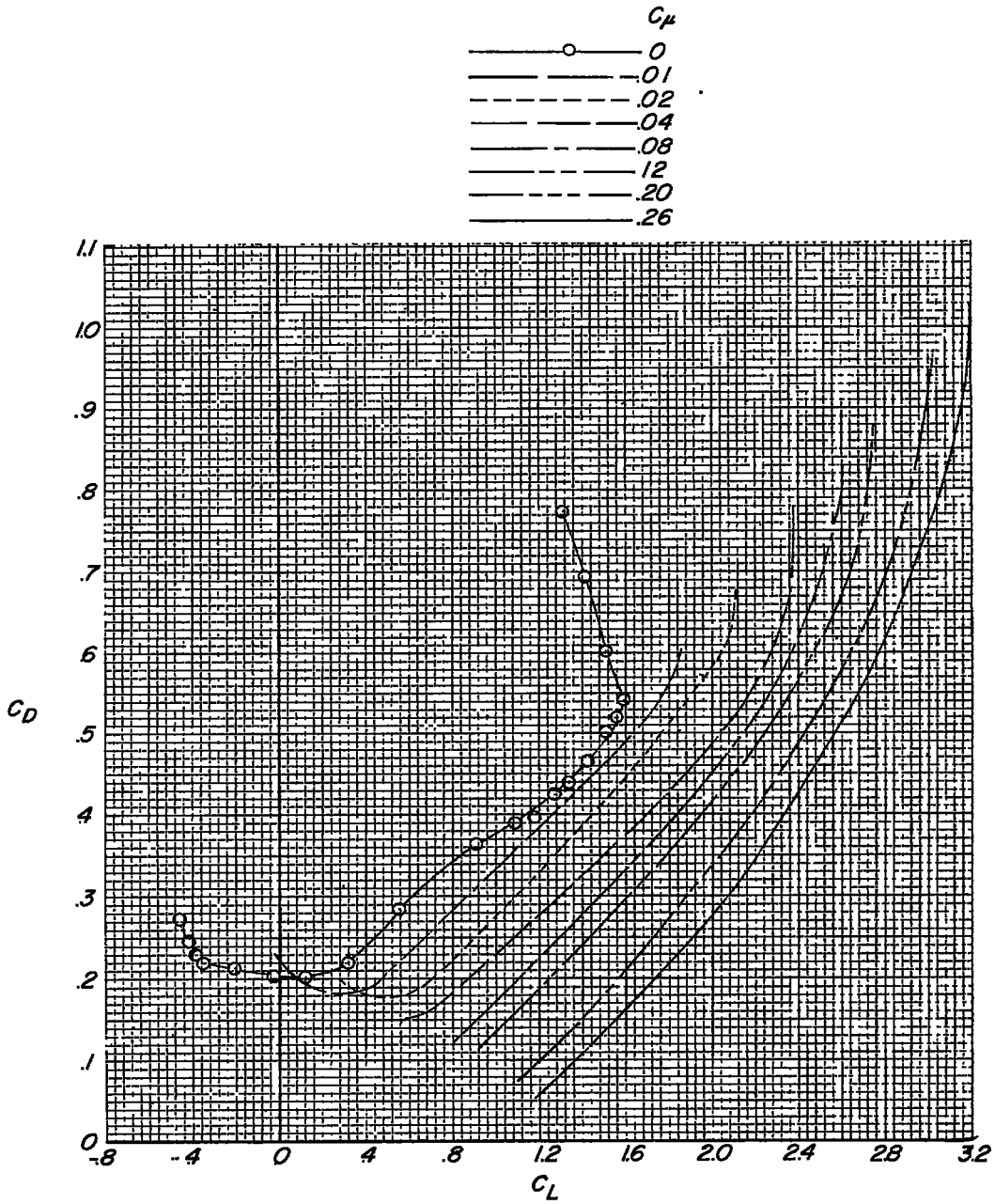
(b) C_D against C_L.

Figure 22.- Concluded.



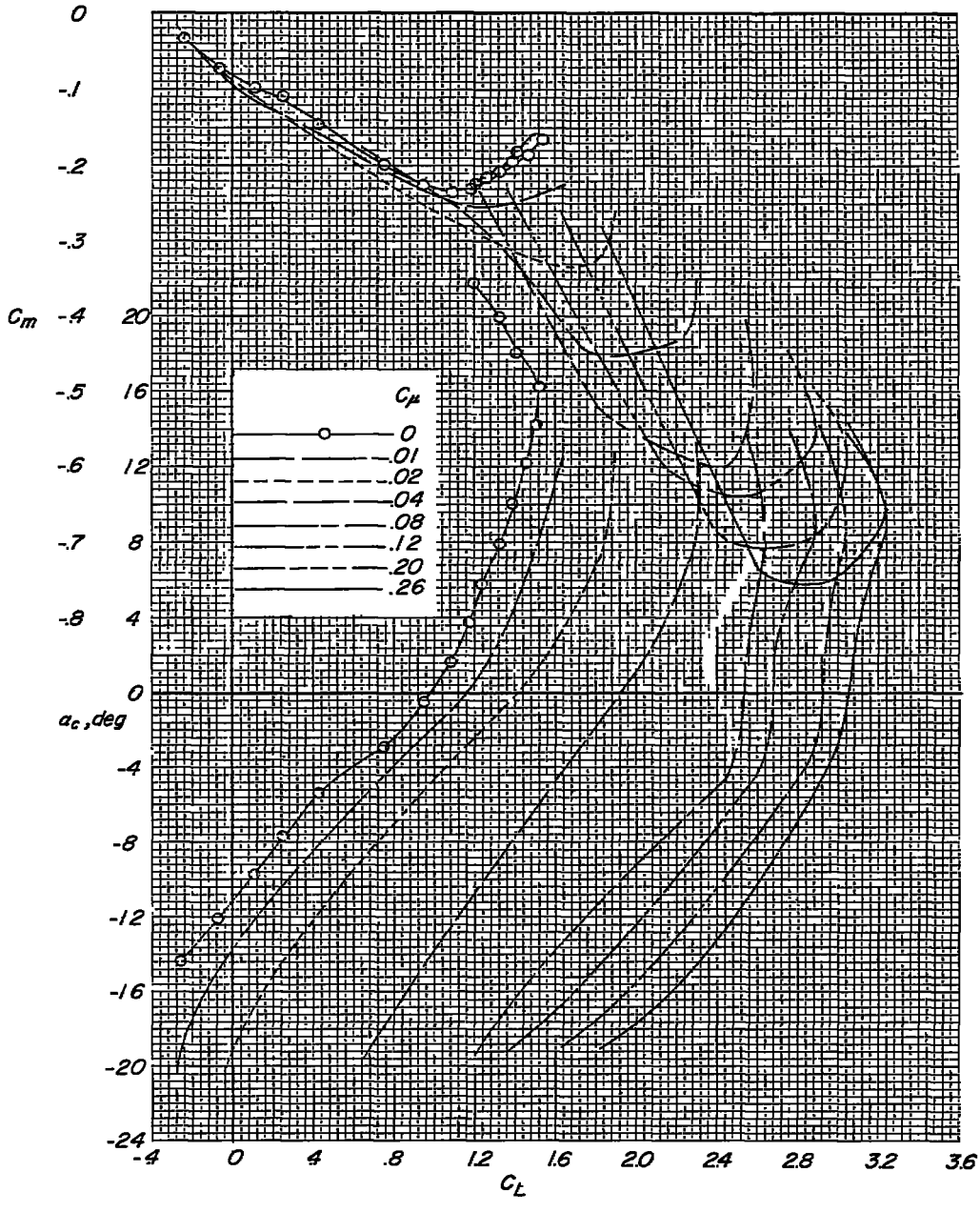
(a) C_m and α_c against C_L.

Figure 23.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.33c_w$; $\delta_f = 60^\circ$; $\delta_N = 30^\circ$; $q \approx 12.5$ lb/sq ft.



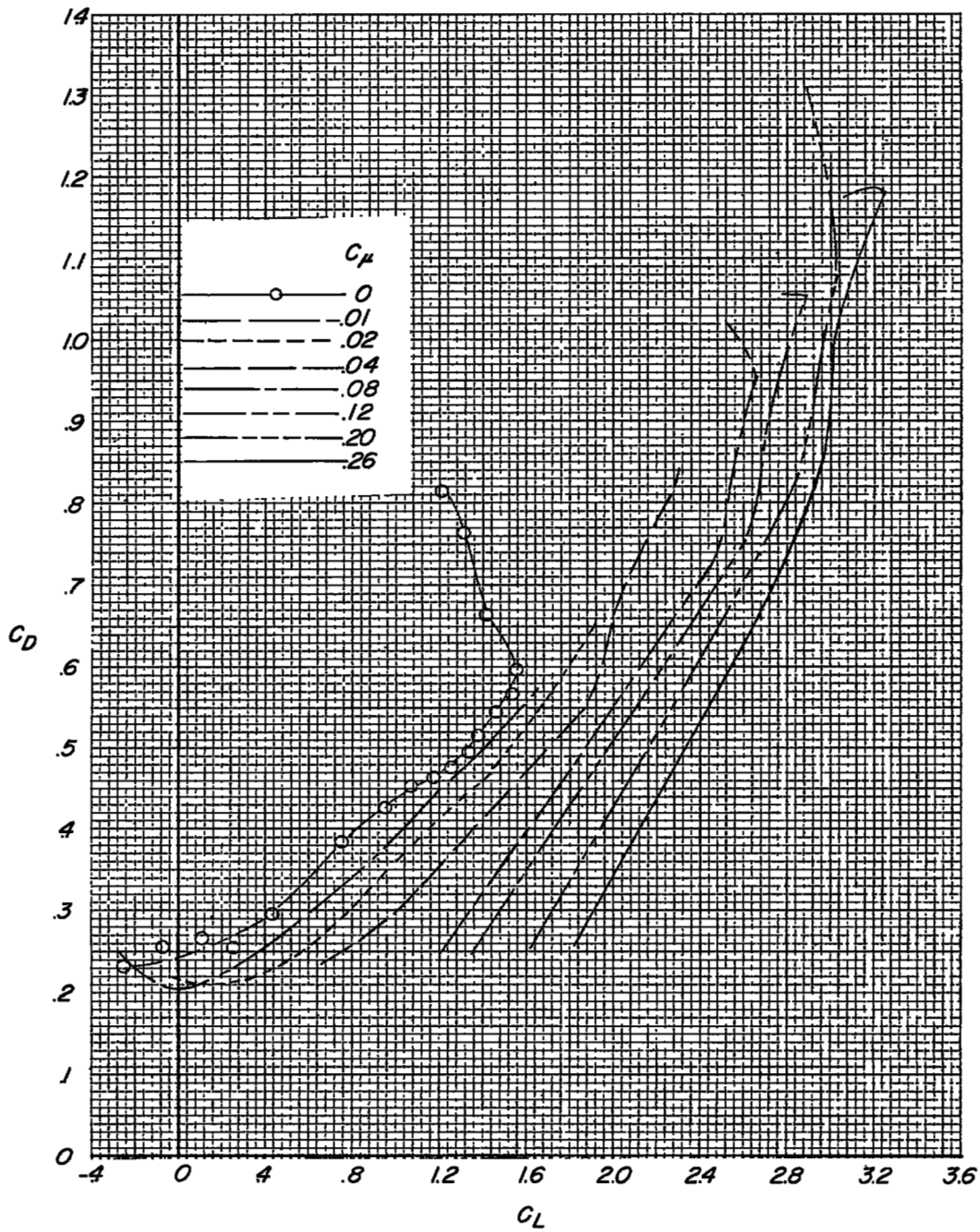
(b) C_D against C_L.

Figure 23.- Concluded.



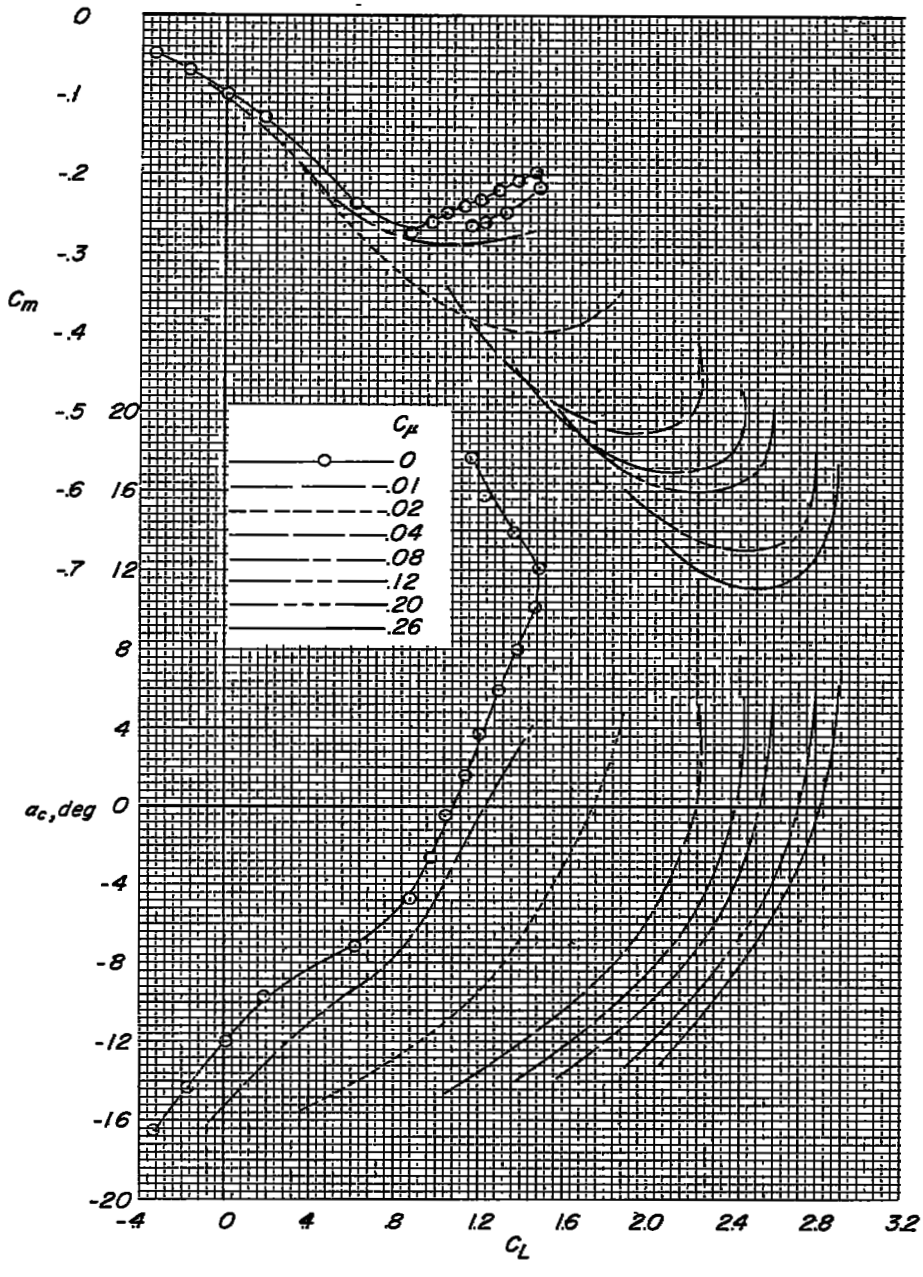
(a) C_m and α_c against C_L.

Figure 24.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.33c_w$; $\delta_f = 75^\circ$; $\delta_N = 30^\circ$; $q \approx 12.5$ lb/sq ft.



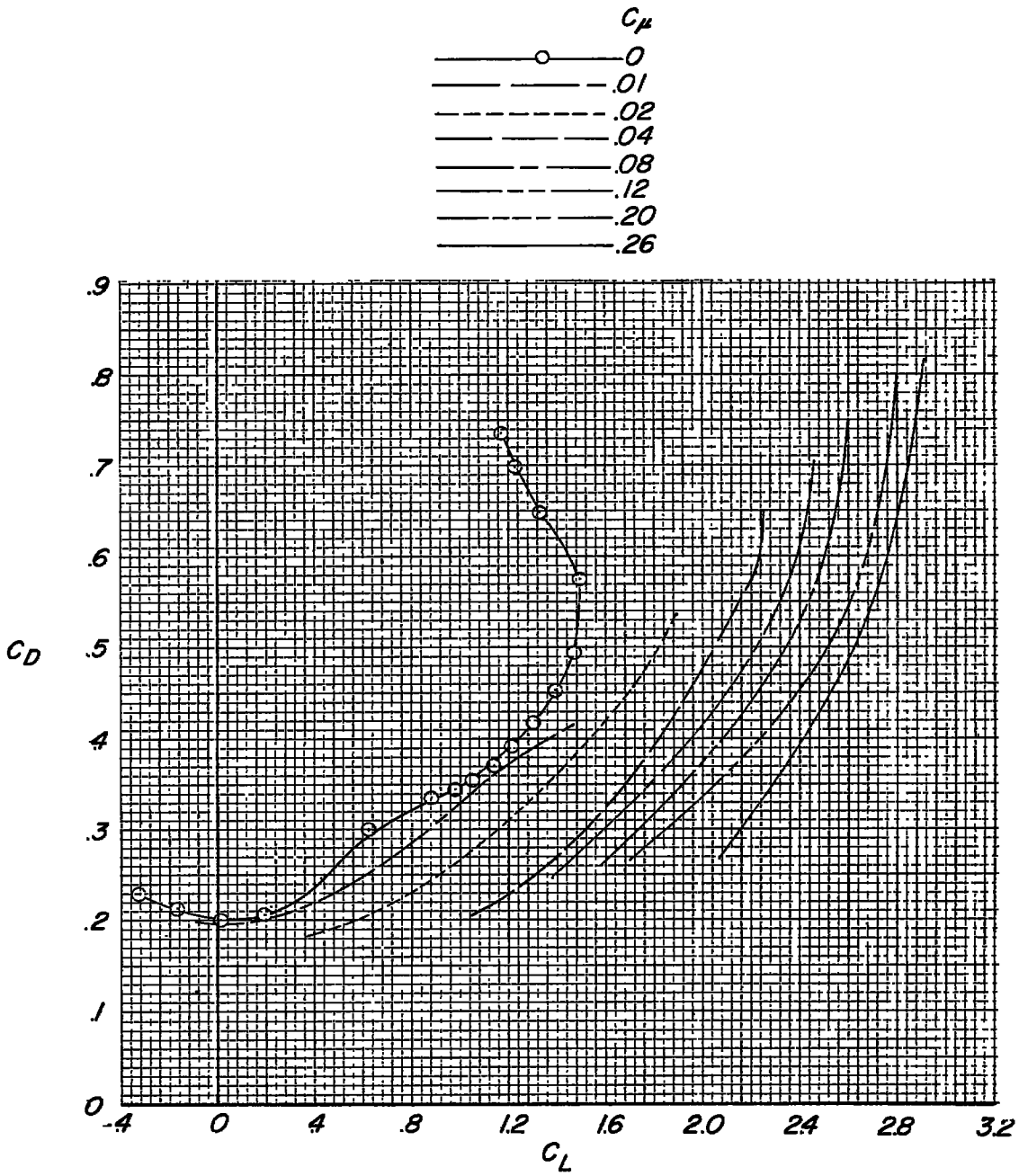
(b) C_D against C_L.

Figure 24.- Concluded.



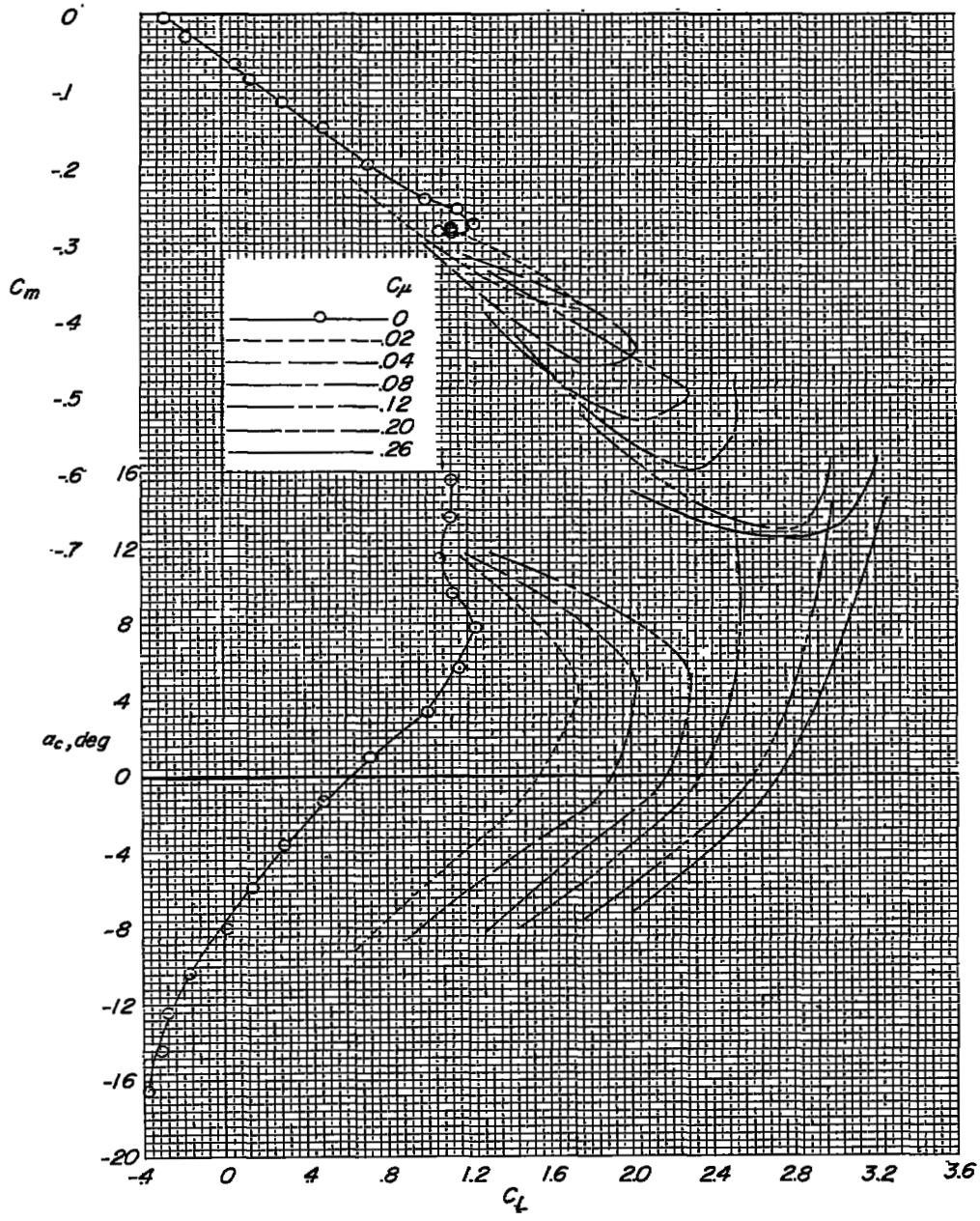
(a) C_m and α_c against C_L.

Figure 25.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.33c_w$; $\delta_f = 60^\circ$; $\delta_N = 15^\circ$; $q \approx 12.5$ lb/sq ft.



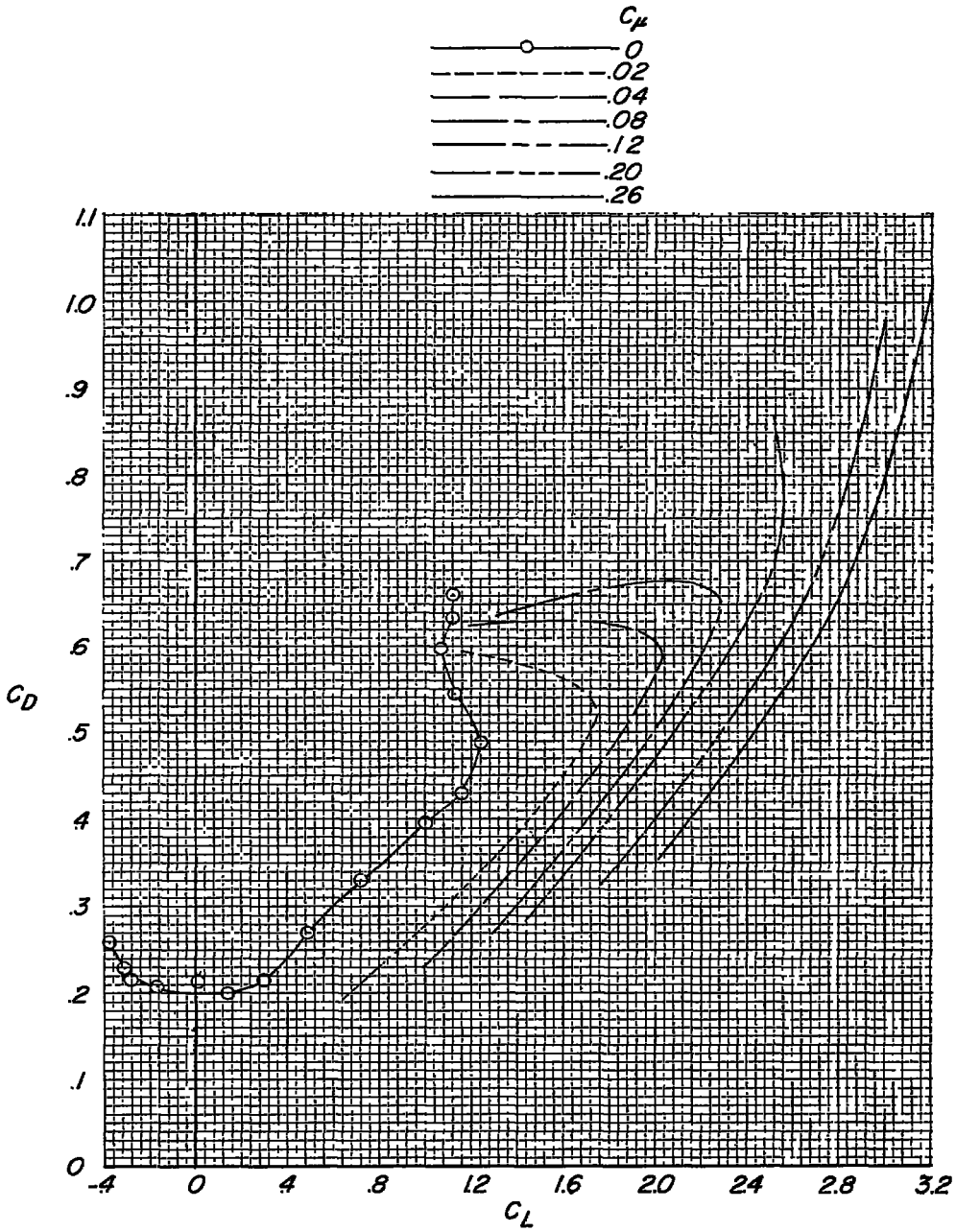
(b) C_D against C_L.

Figure 25.- Concluded.



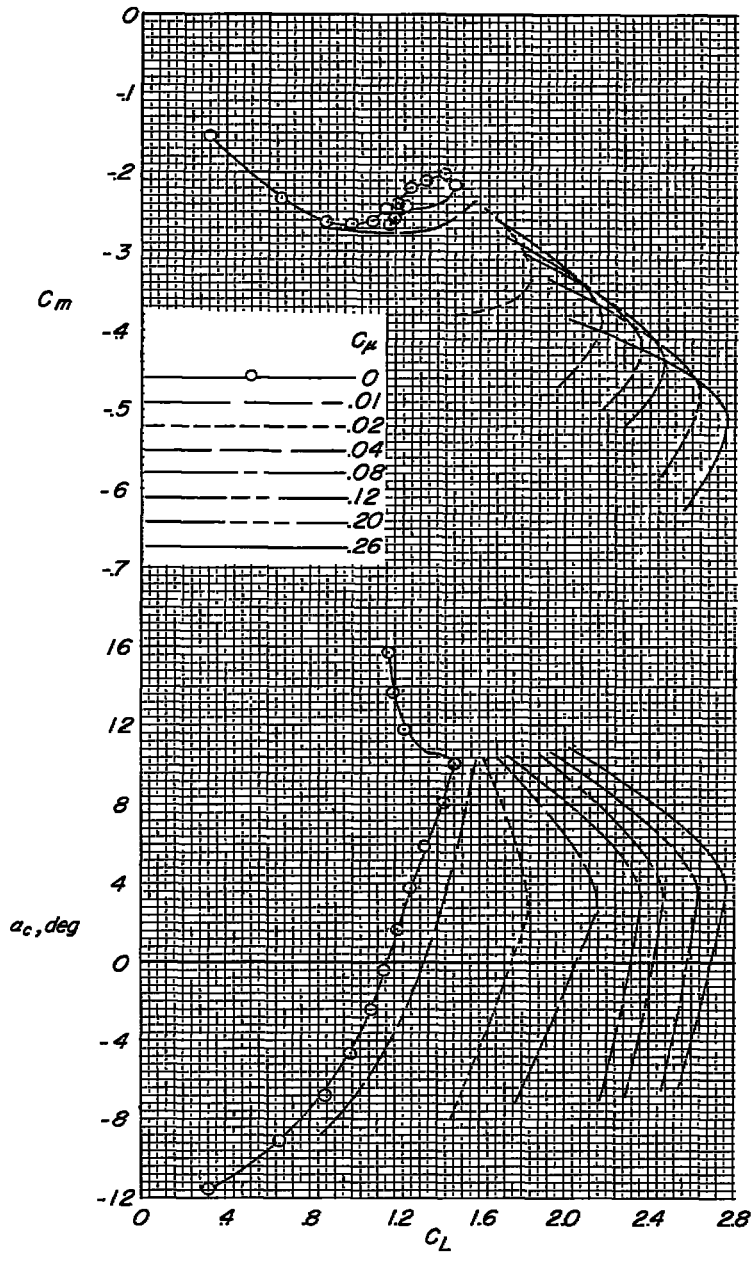
(a) C_m and α_c against C_L.

Figure 26.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.33c_w$; $\delta_f = 60^\circ$; $\delta_N = 45^\circ$; $q \approx 12.5$ lb/sq ft.



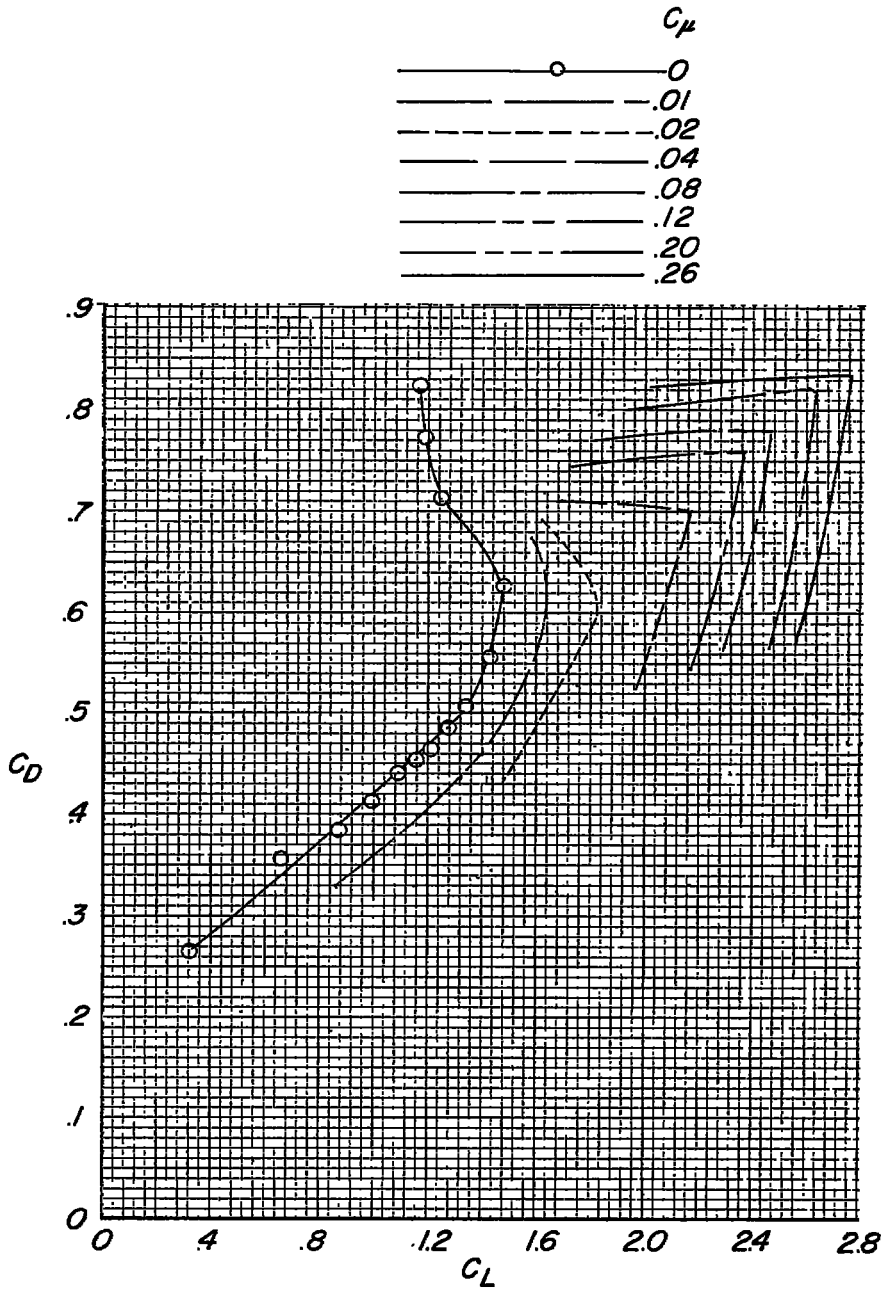
(b) C_D against C_L.

Figure 26.- Concluded.



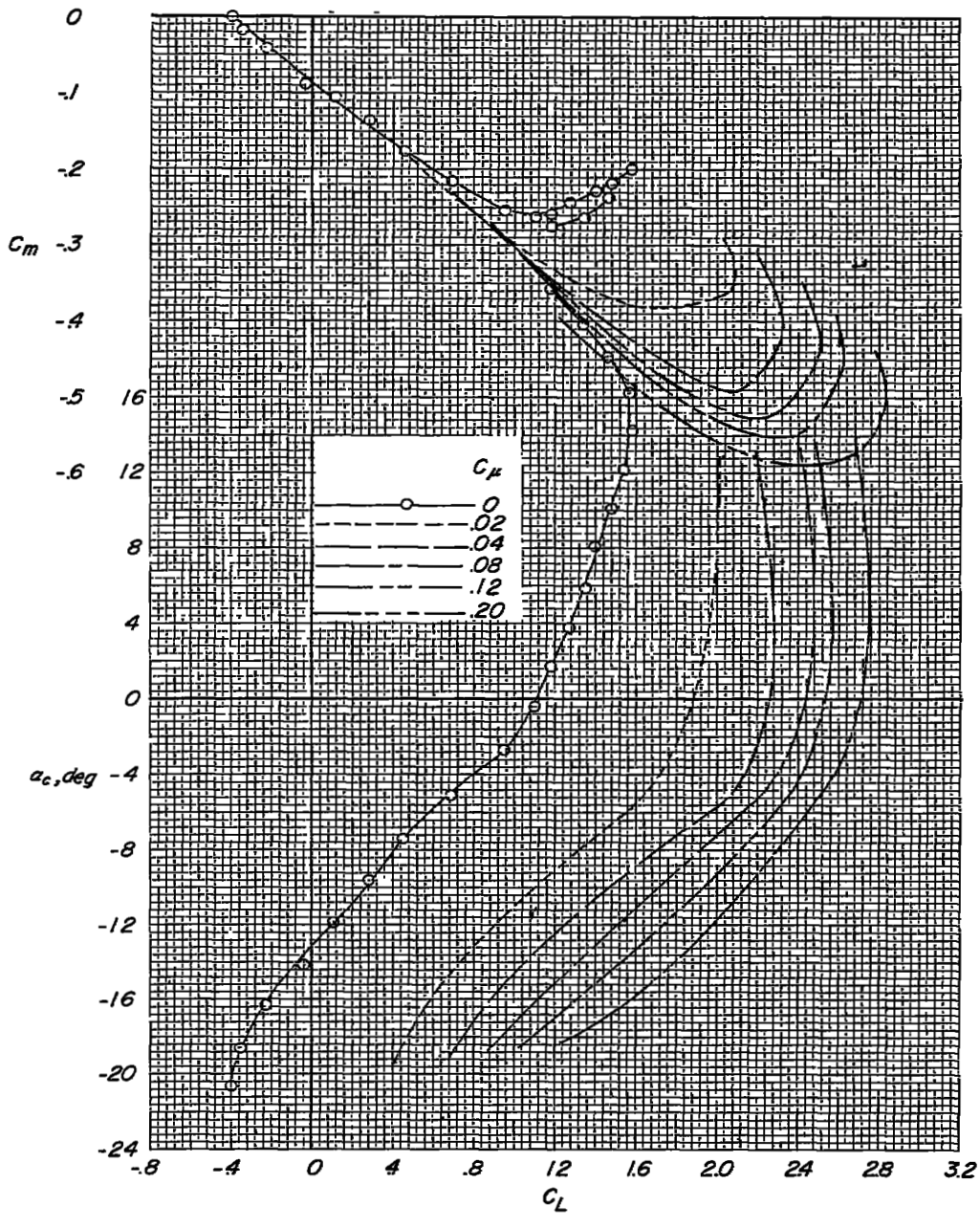
(a) C_m and α_c against C_L.

Figure 27.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.40c_w$; $\delta_f = 60^\circ$; $\delta_N = 15^\circ$; $q \approx 12.5$ lb/sq ft.



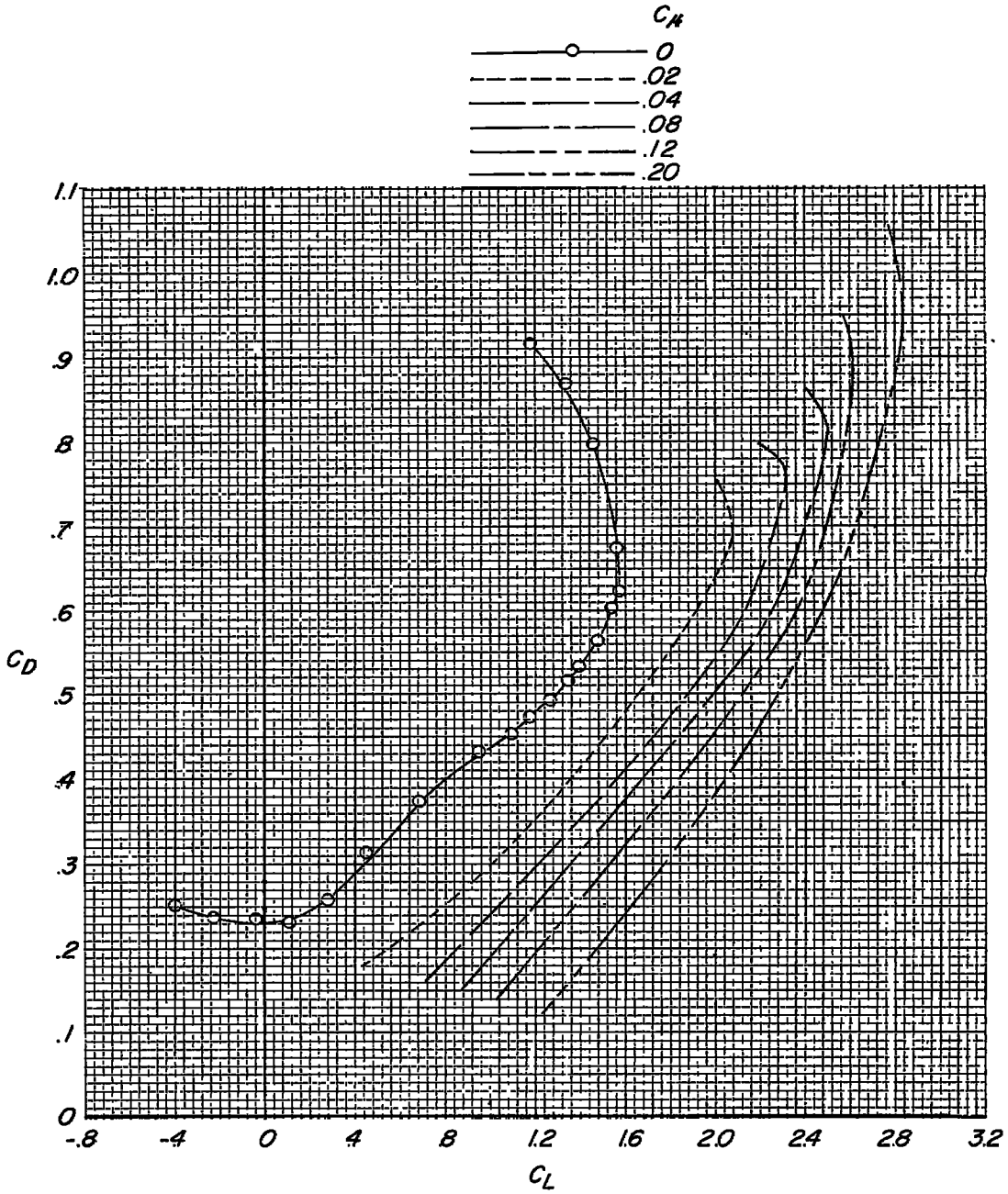
(b) C_D against C_L.

Figure 27.- Concluded.



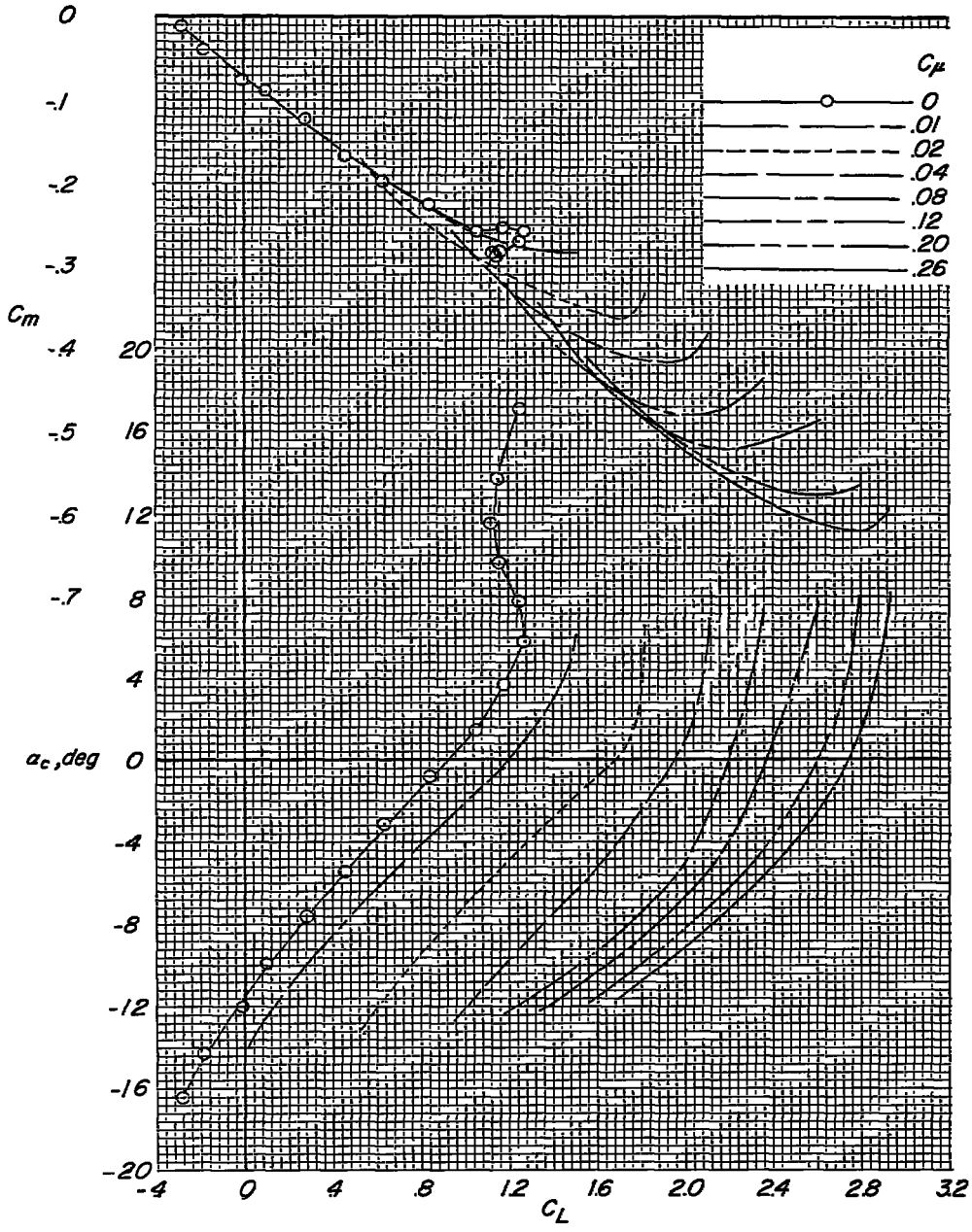
(a) C_m and α_c against C_L.

Figure 28.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.40c_w$; $\delta_f = 60^\circ$; $\delta_N = 30^\circ$; $q \approx 12.5$ lb/sq ft.



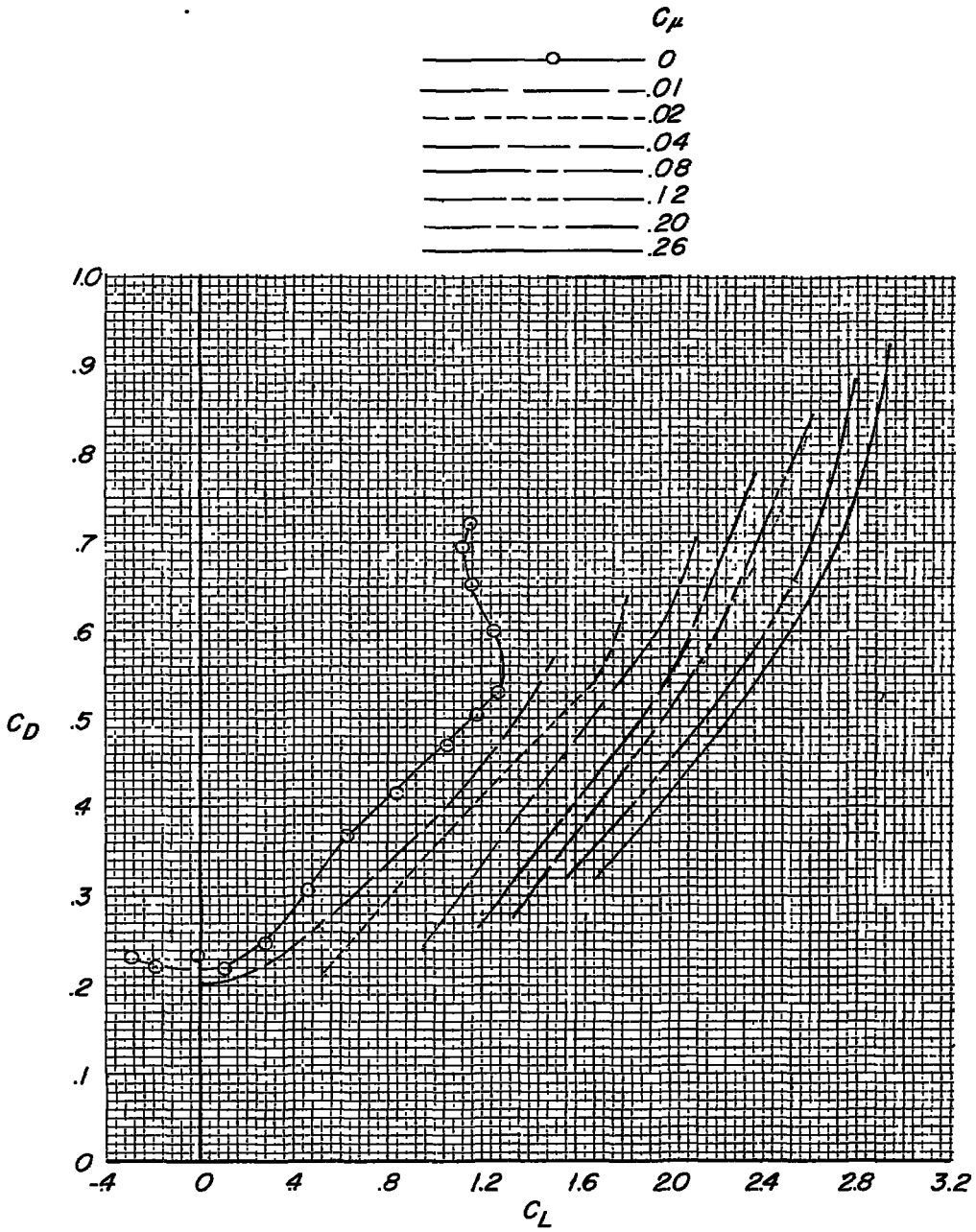
(b) C_D against C_L.

Figure 28.- Concluded.



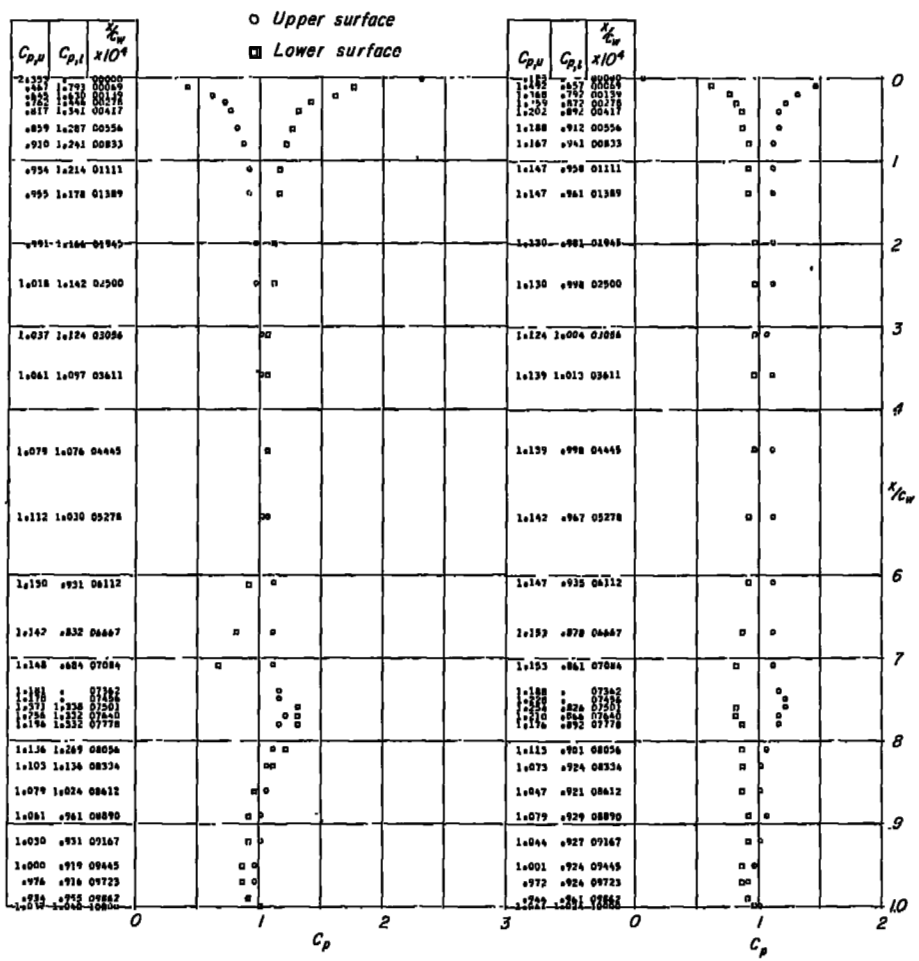
(a) C_m and α_c against C_L.

Figure 29.- Effect of momentum coefficient on the static longitudinal aerodynamic characteristics of the model. $c_f = 0.40c_w$; $\delta_f = 60^\circ$; $\delta_N = 45^\circ$; $q \approx 12.5$ lb/sq ft.



(b) C_D against C_L.

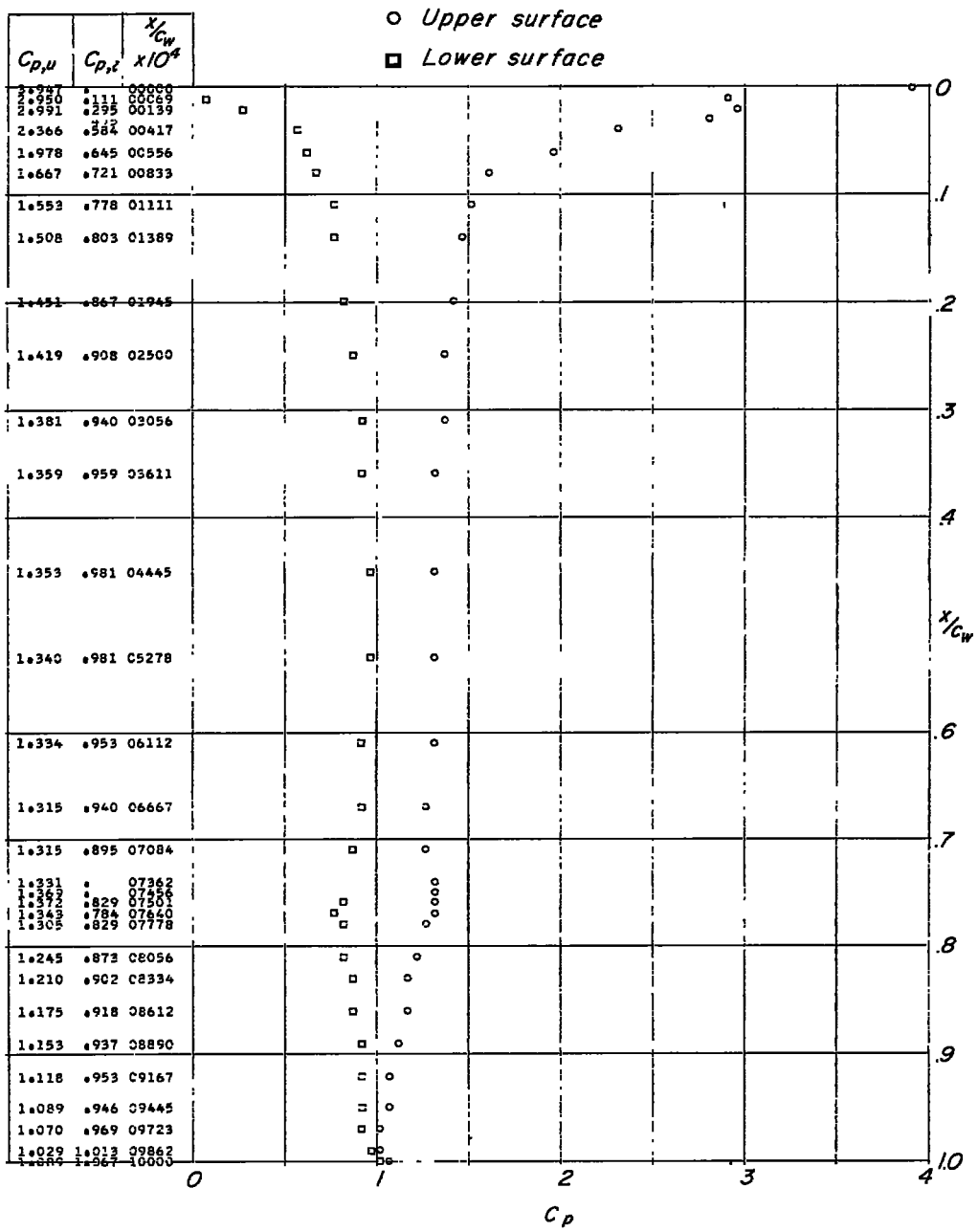
Figure 29.- Concluded.



(a) $\alpha = -4^{\circ}$.

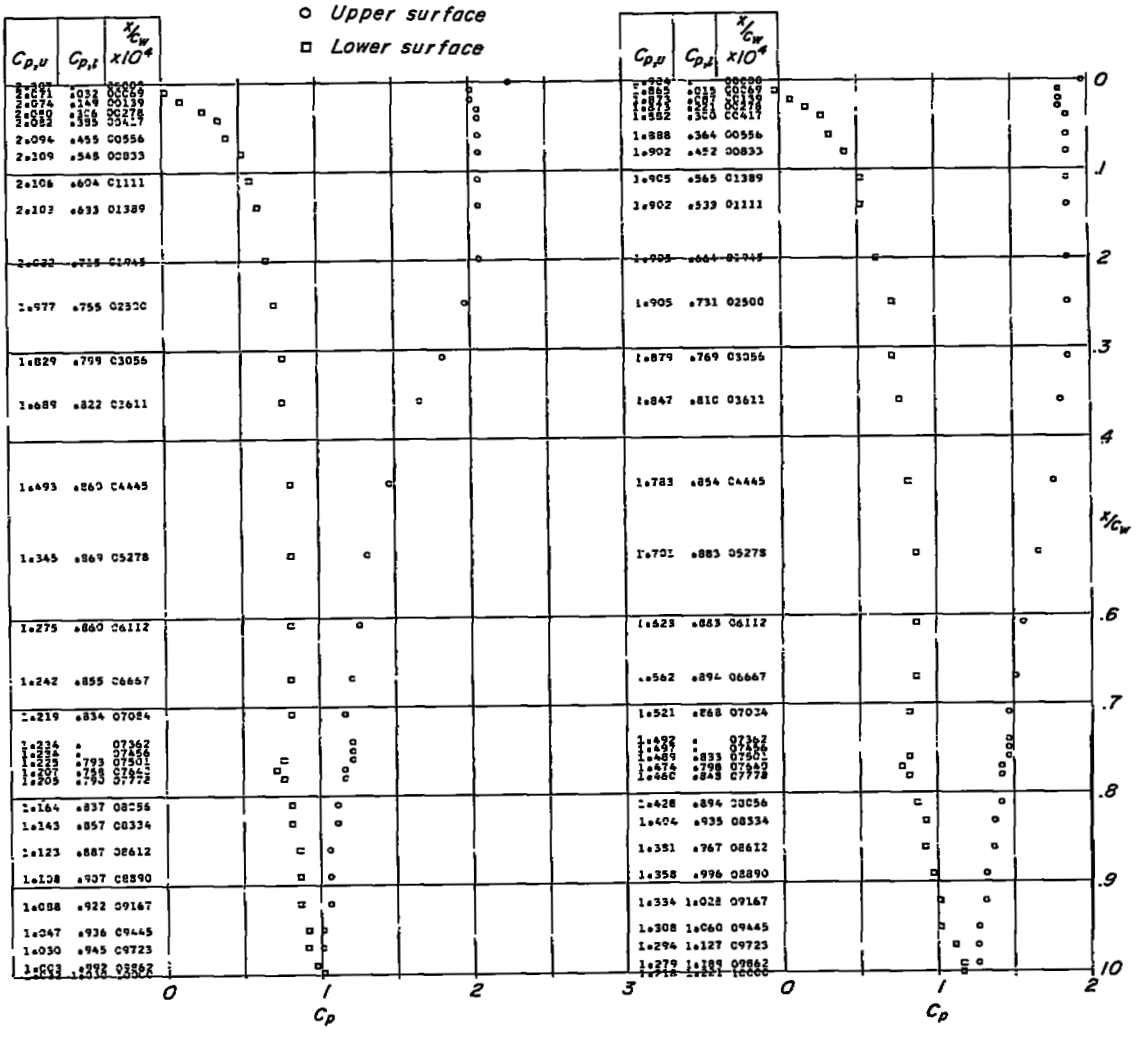
(b) $\alpha = 0^{\circ}$.

Figure 30.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 0^{\circ}$; $\delta_f = 5^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = 4^{\circ}$.

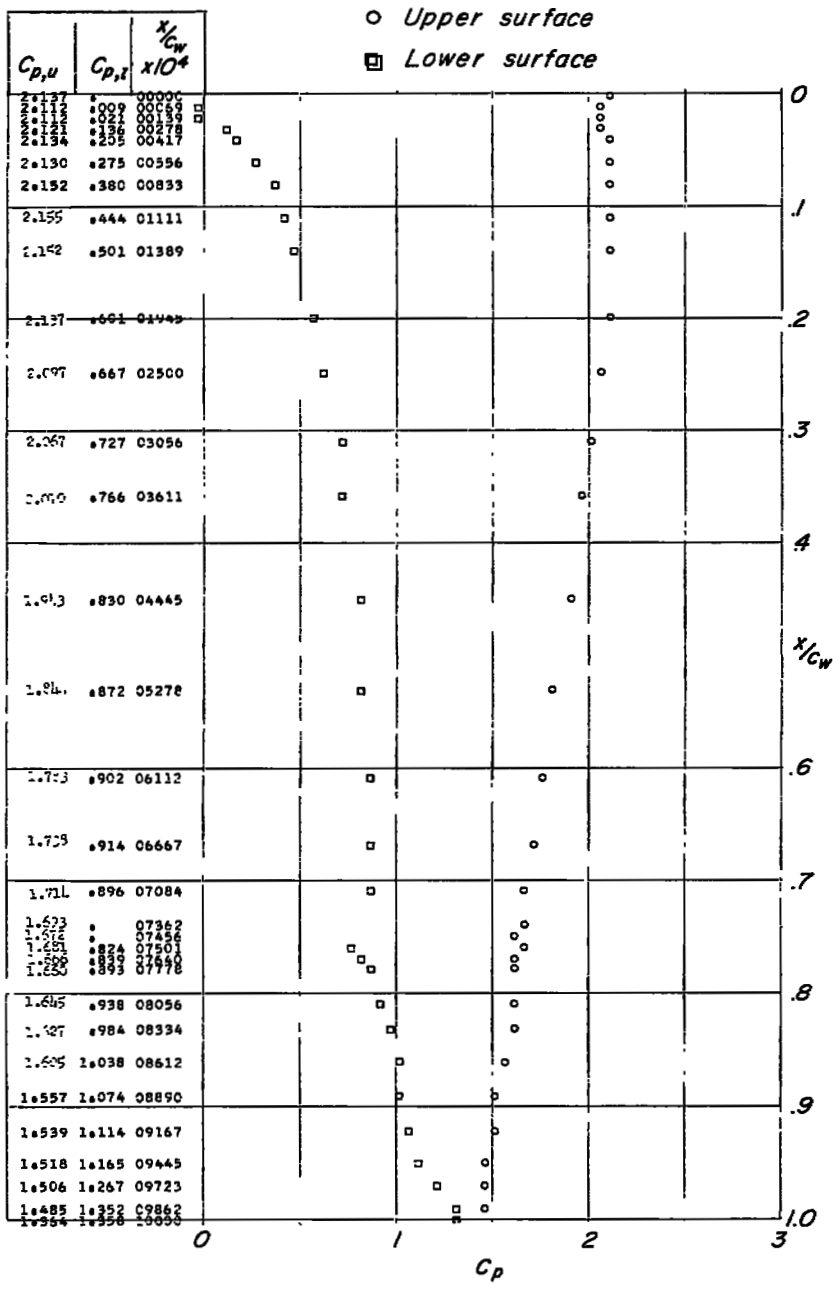
Figure 30.- Continued.



(d) $\alpha = 8^{\circ}$.

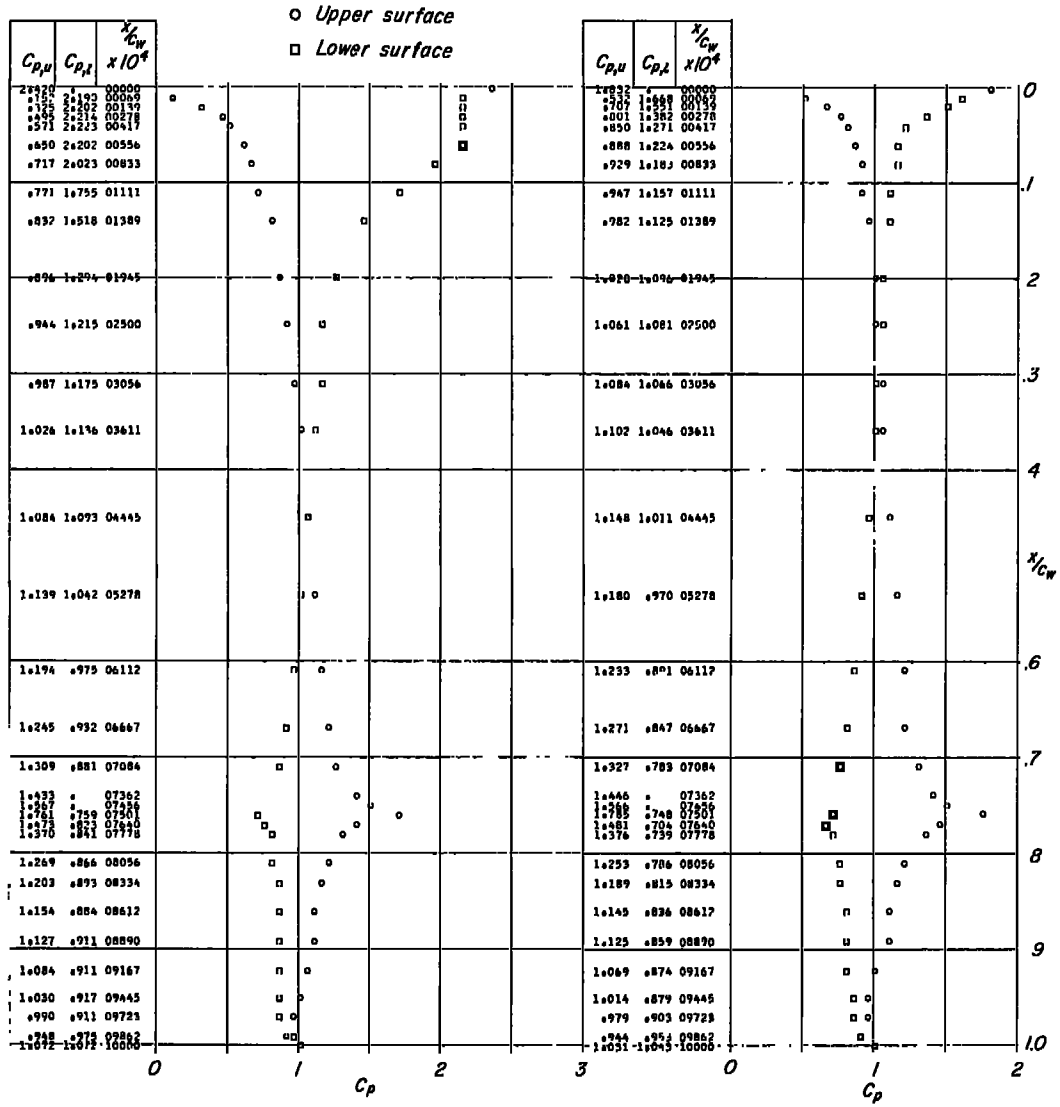
(e) $\alpha = 12^{\circ}$.

Figure 30.- Continued.



(f) $\alpha = 16^{\circ}$.

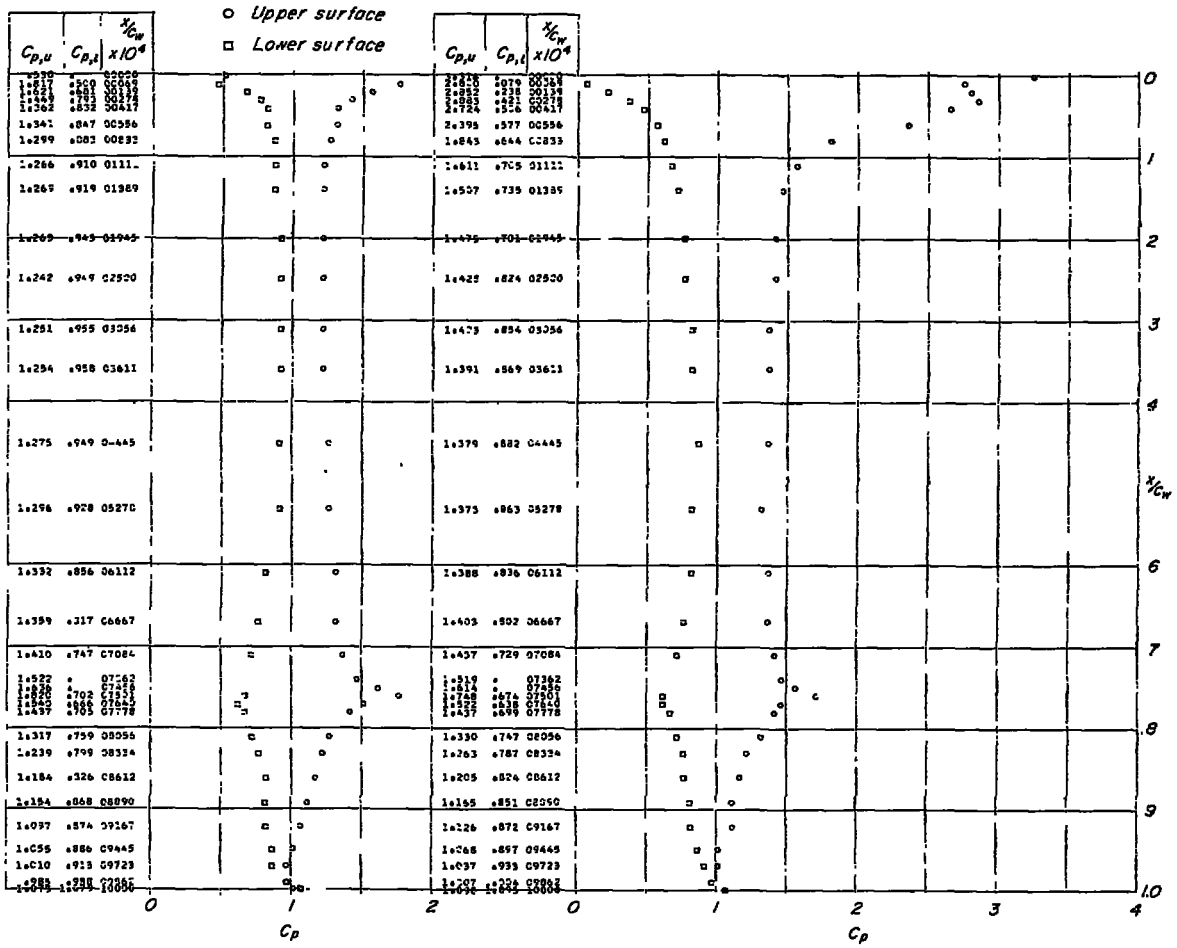
Figure 30.- Concluded.



(a) $\alpha = -8^{\circ}$.

(b) $\alpha = -4^{\circ}$.

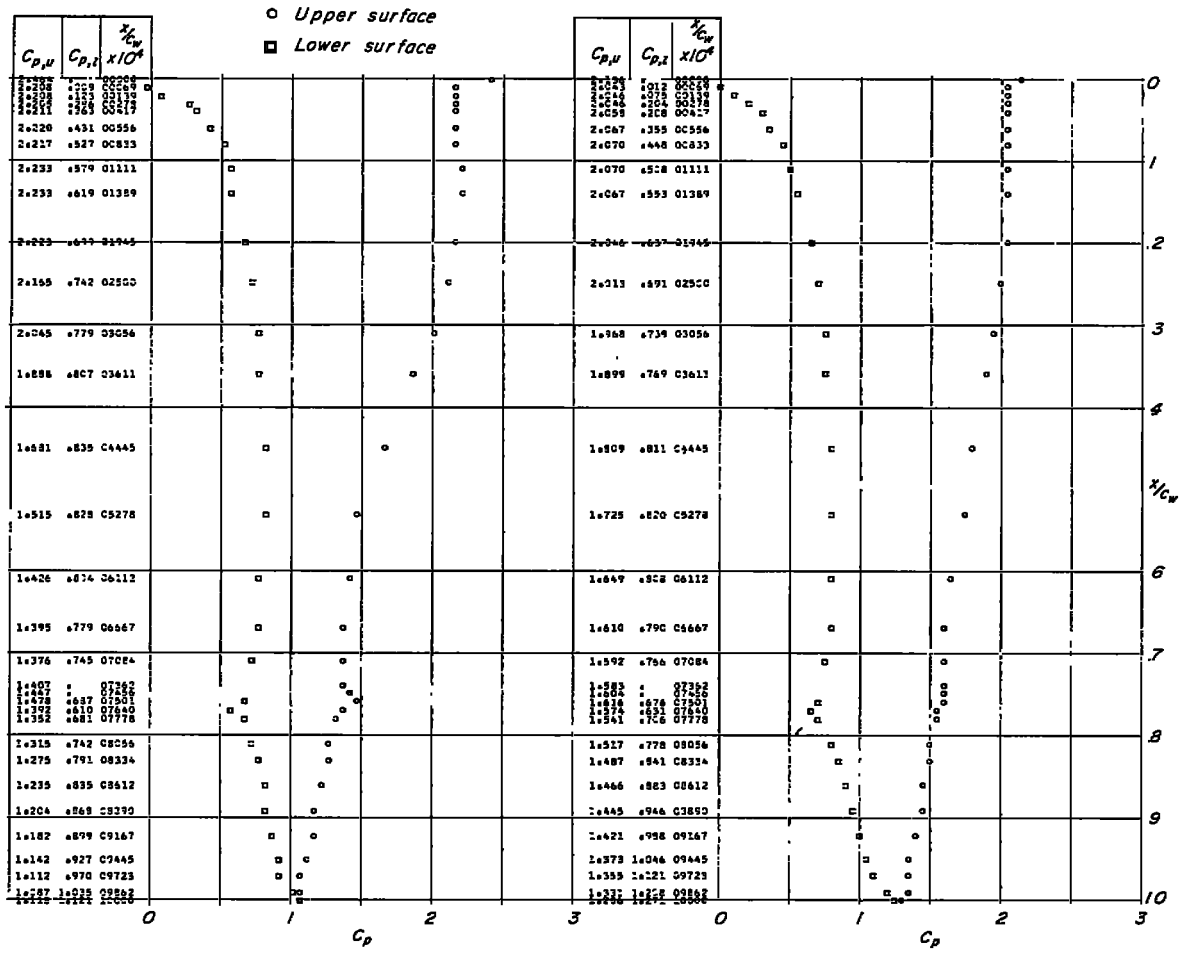
Figure 31.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 0^{\circ}$; $\delta_f = 10^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = 0^{\circ}$.

(d) $\alpha = 4^{\circ}$.

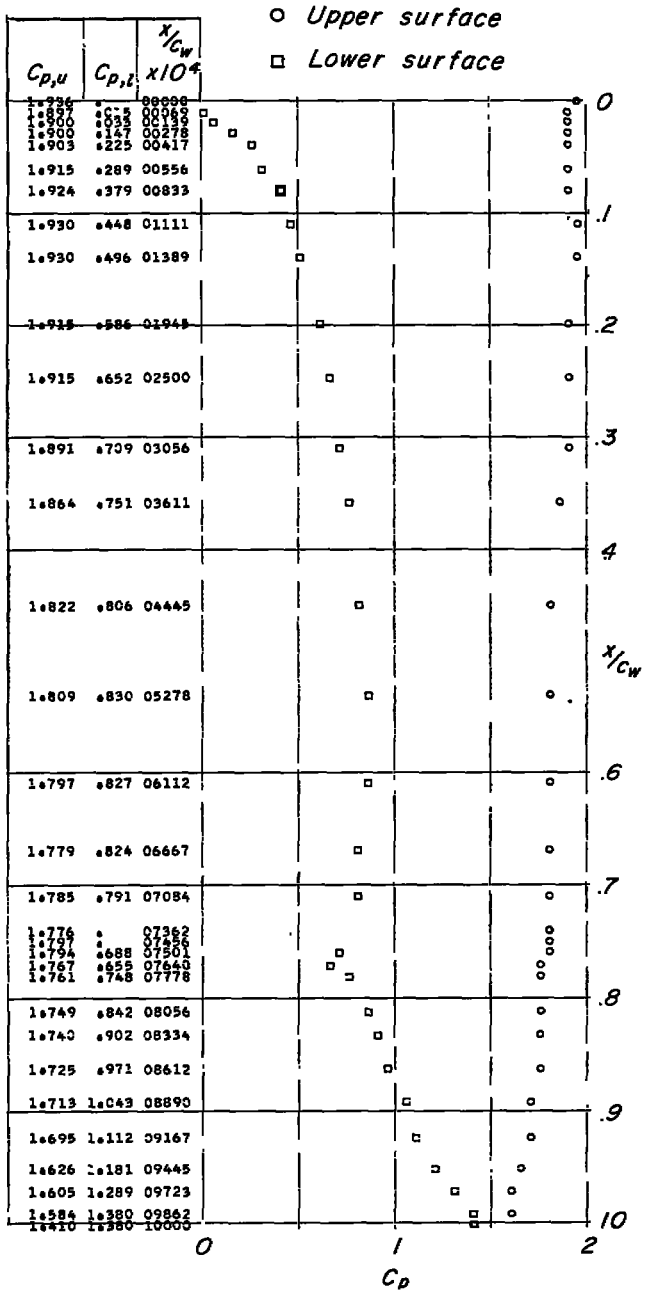
Figure 31.- Continued.



(e) $\alpha = 8^{\circ}$.

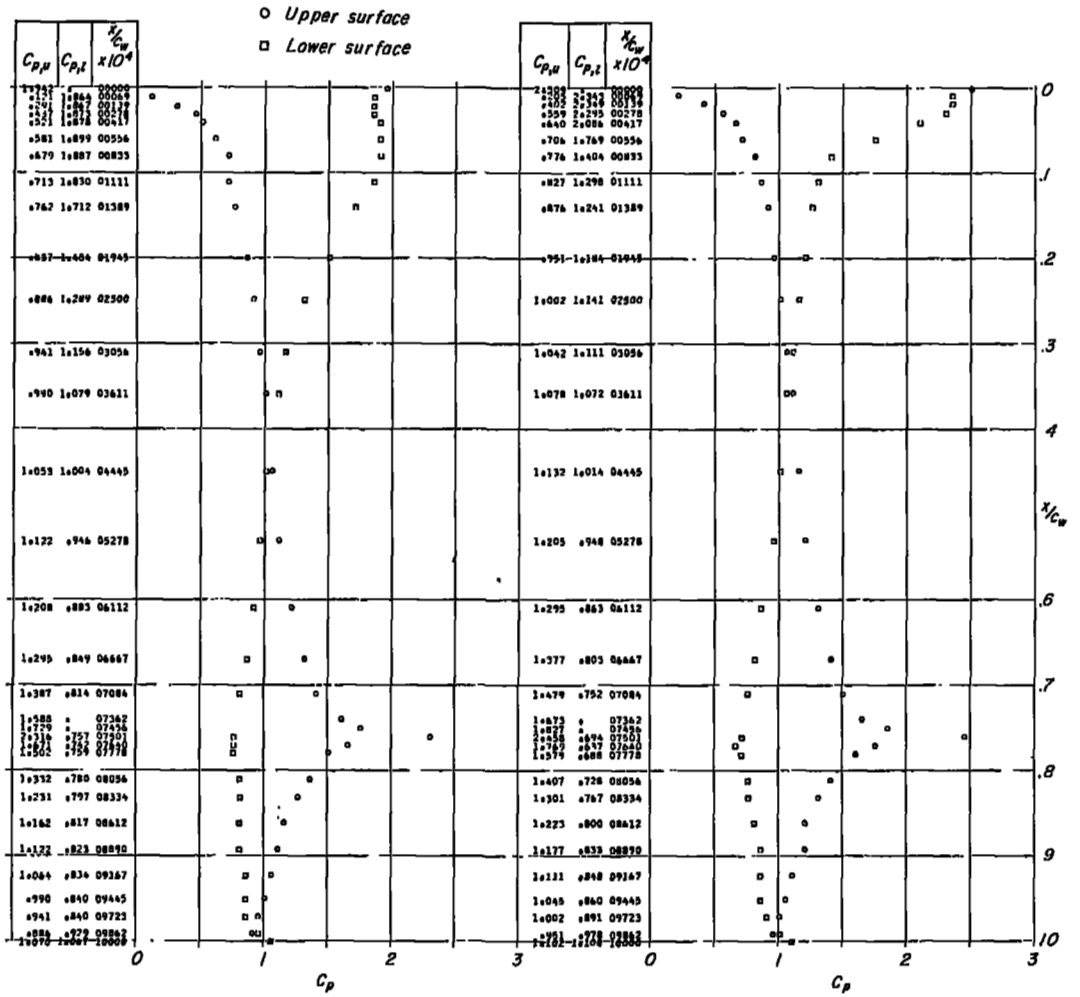
(f) $\alpha = 12^{\circ}$.

Figure 31.- Continued.



(g) $\alpha = 16^{\circ}$.

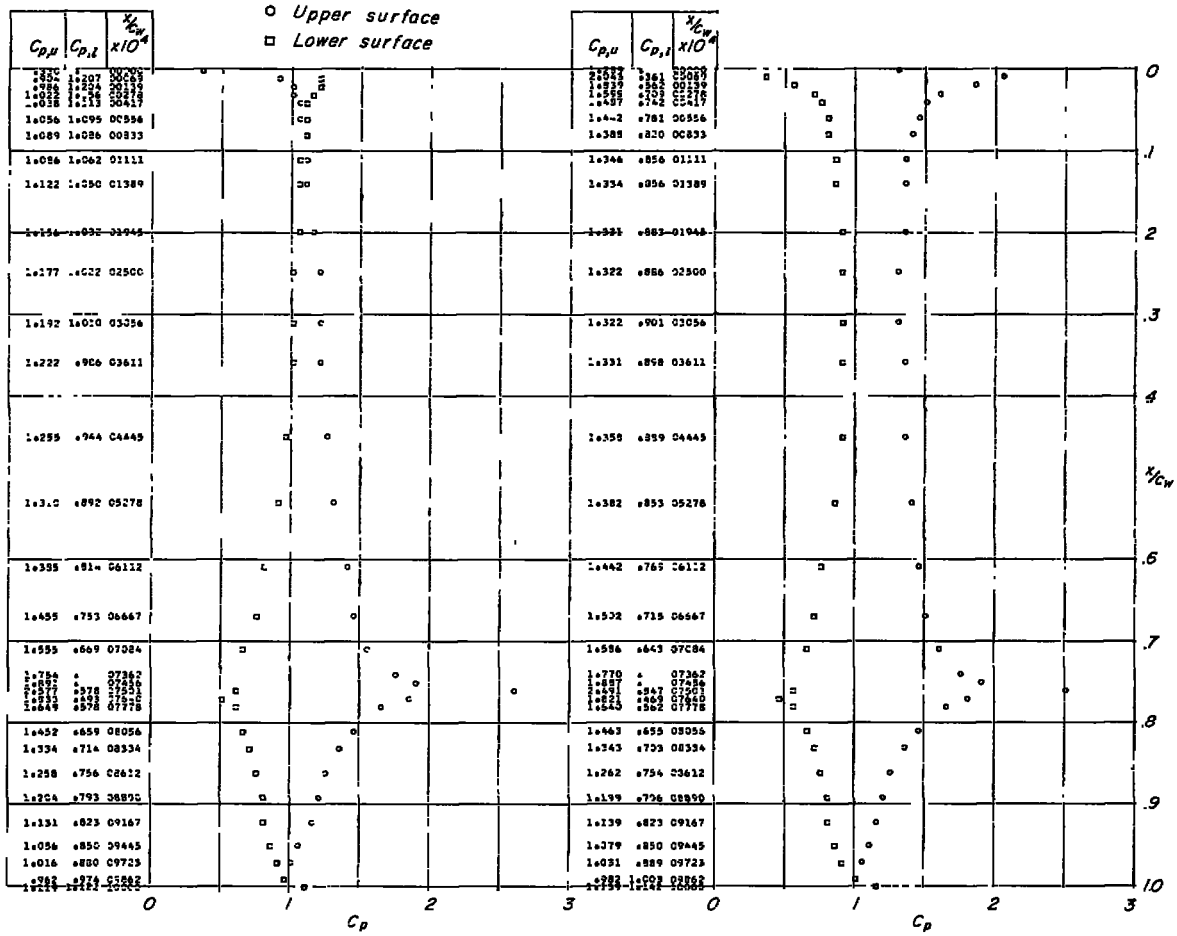
Figure 31.- Concluded.



(a) $\alpha = -10^{\circ}$.

(b) $\alpha = -8^{\circ}$.

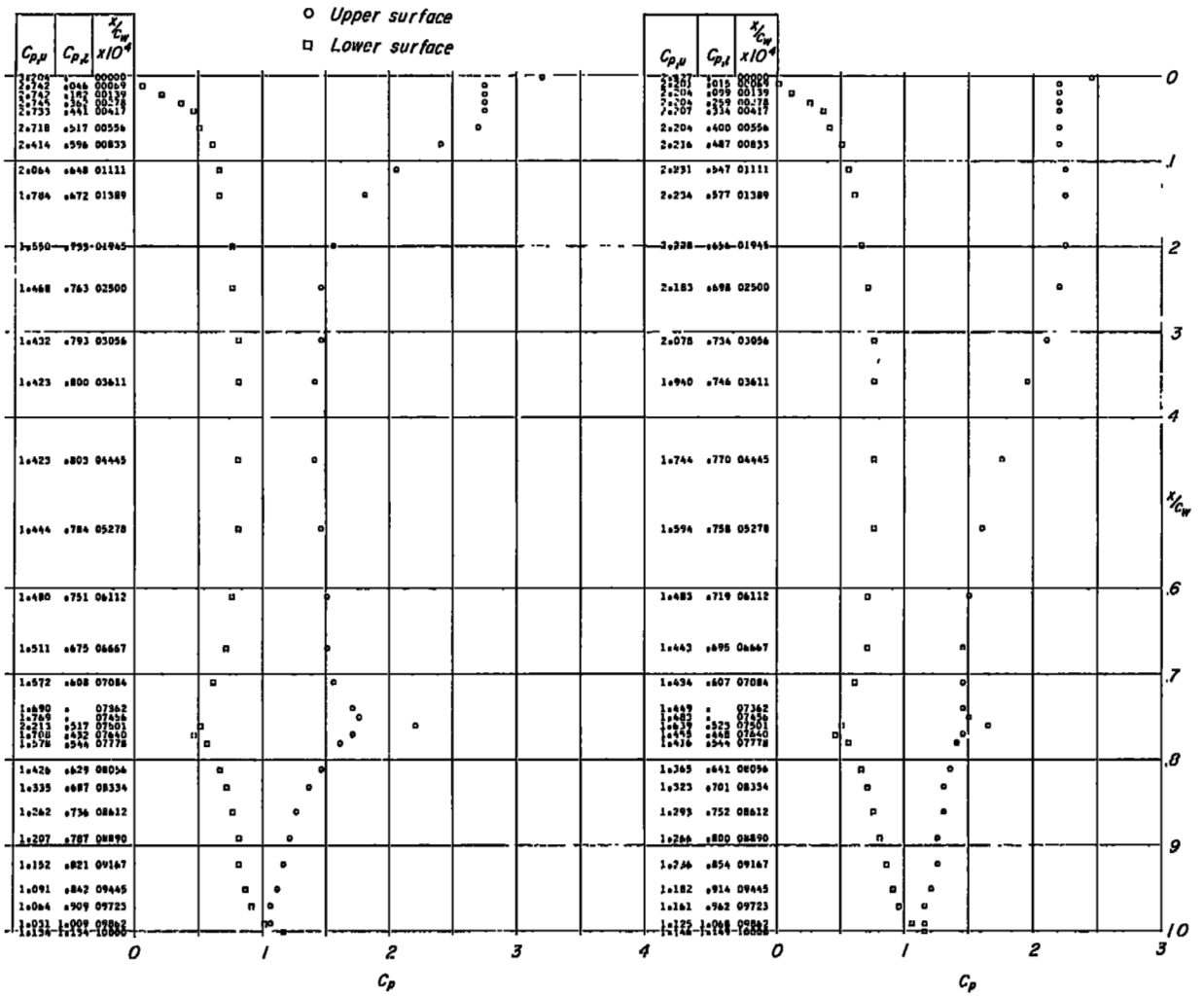
Figure 32.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 0^{\circ}$; $\delta_f = 15^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = -4^{\circ}$.

(d) $\alpha = 0^{\circ}$.

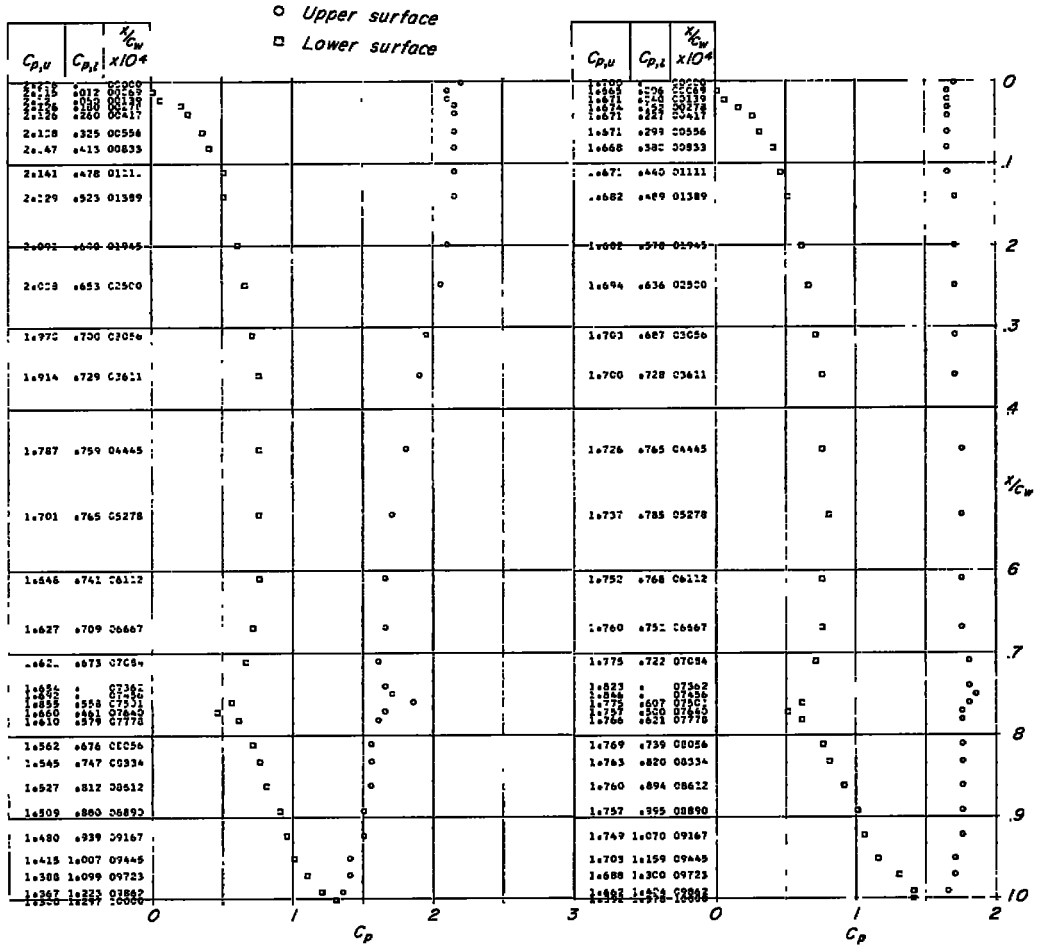
Figure 32.- Continued.



(e) $\alpha = 4^{\circ}$.

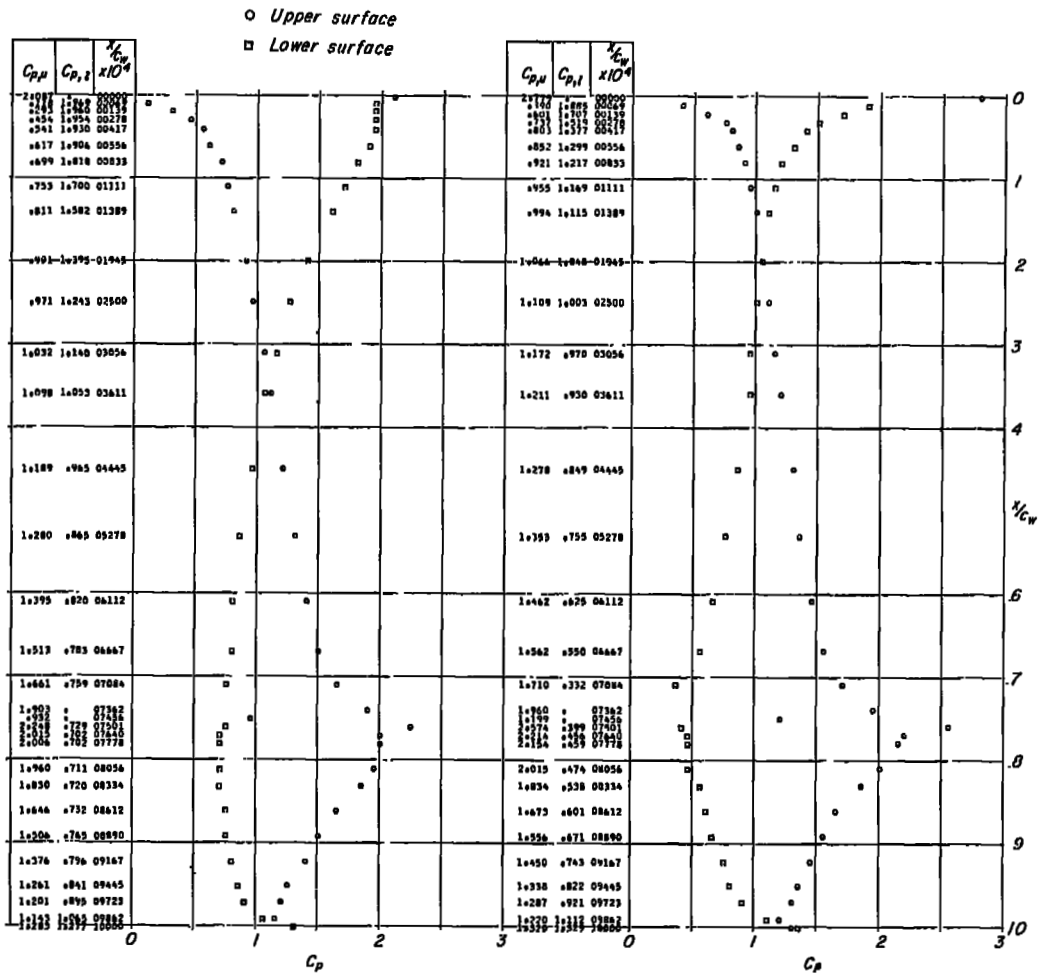
(f) $\alpha = 8^{\circ}$.

Figure 32.- Continued.



(g) $\alpha = 12^{\circ}$. (h) $\alpha = 16^{\circ}$.

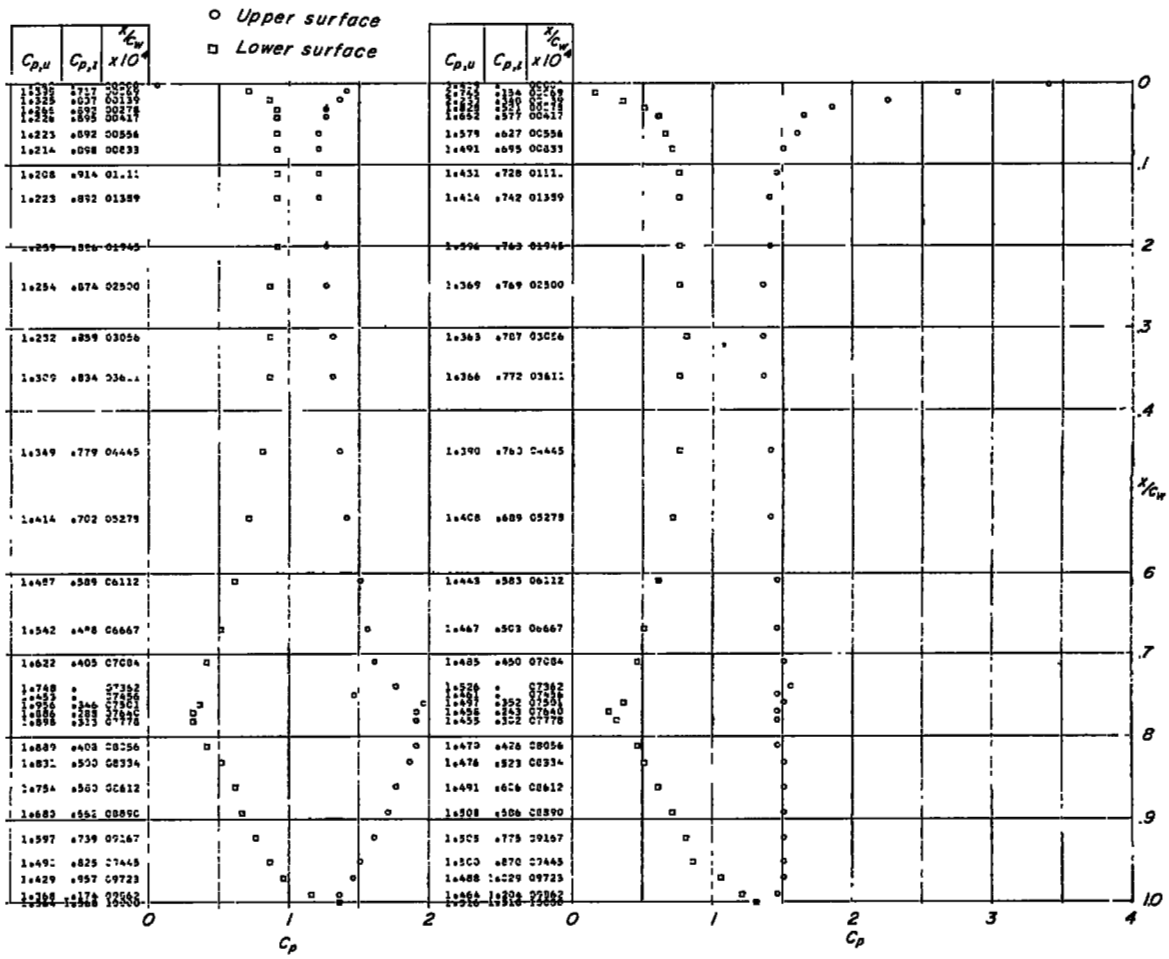
Figure 32.- Concluded.



(a) $\alpha = -12^{\circ}$.

(b) $\alpha = -8^{\circ}$.

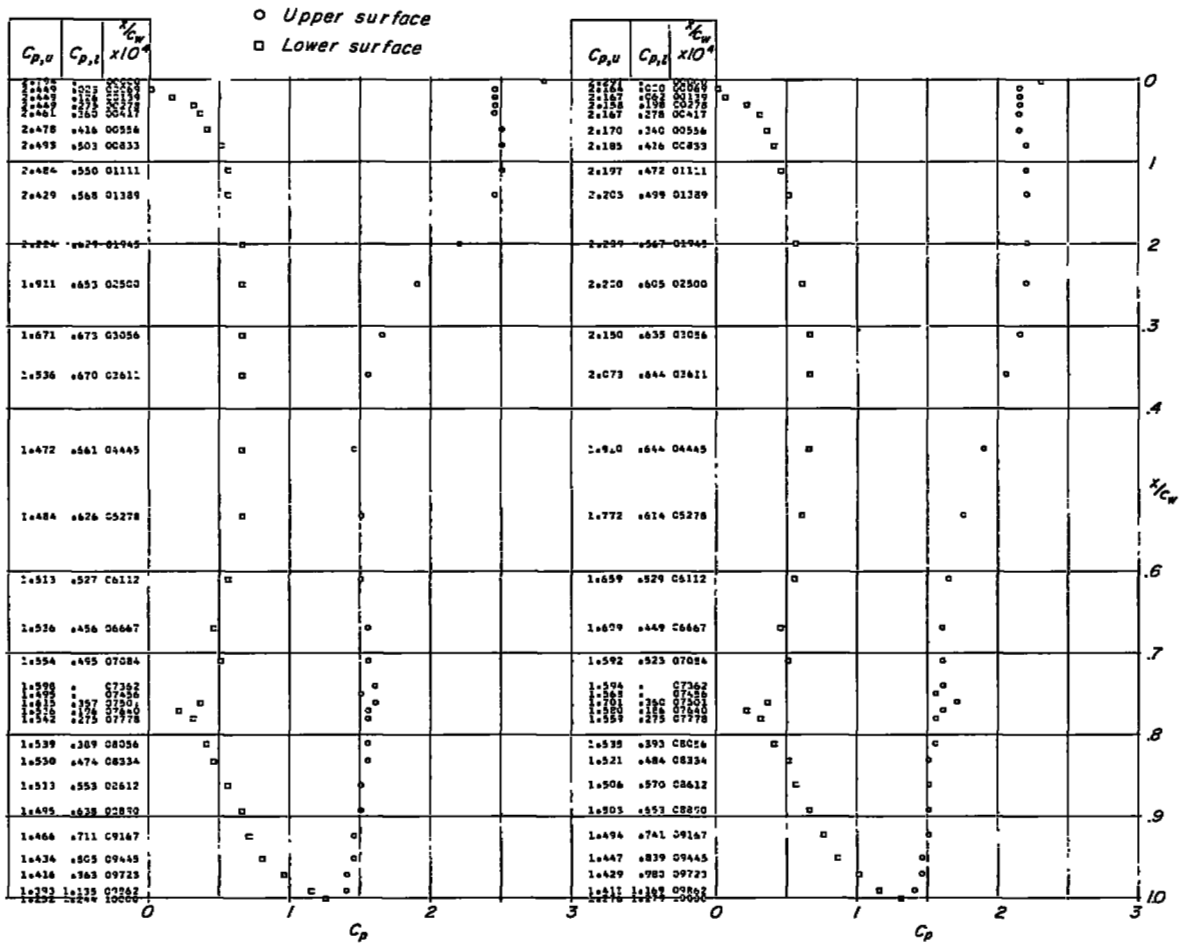
Figure 33.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 0^{\circ}$; $\delta_f = 30^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = -4^{\circ}$.

(d) $\alpha = 0^{\circ}$.

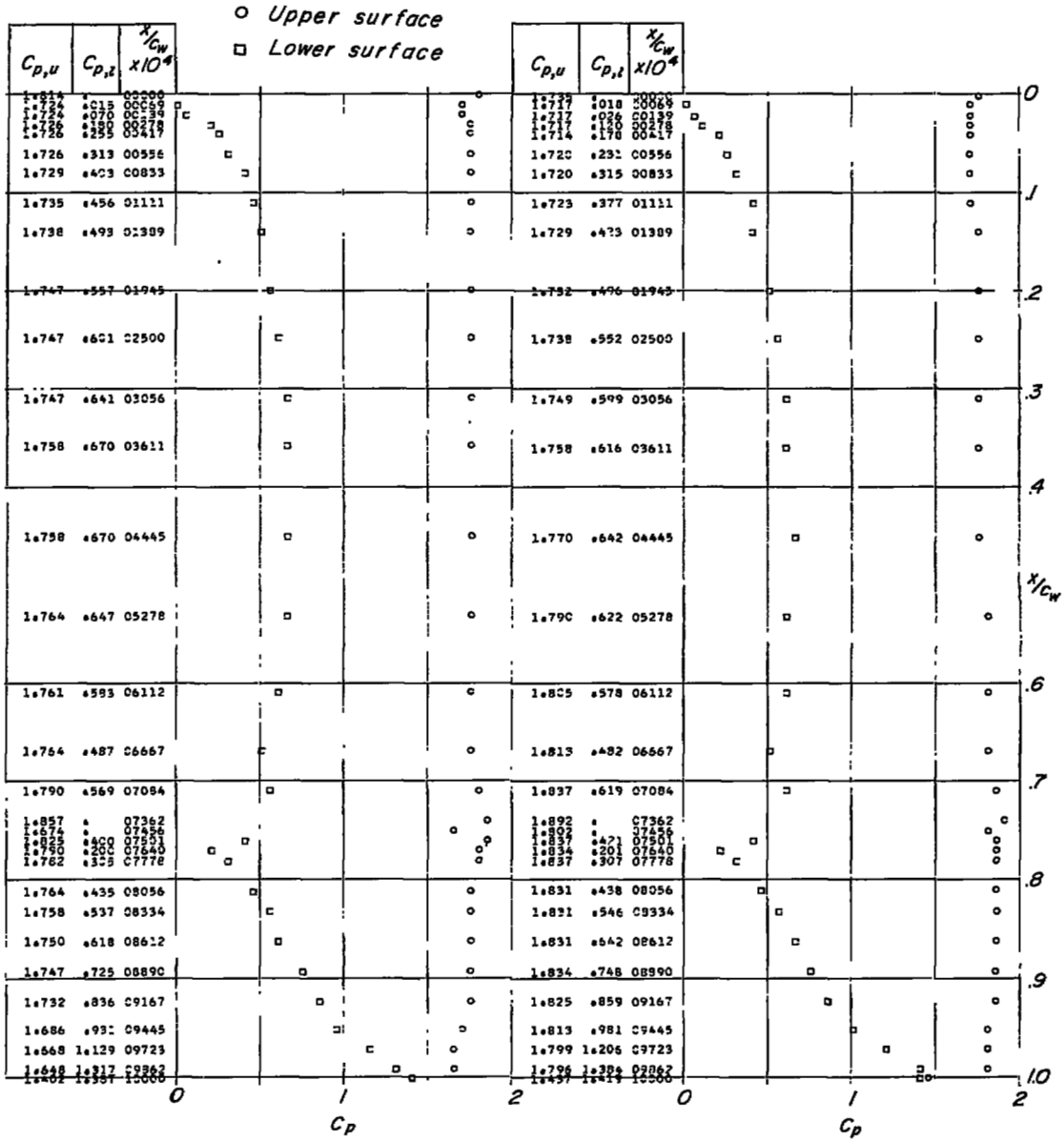
Figure 33.- Continued.



(e) $\alpha = 4^{\circ}$.

(f) $\alpha = 8^{\circ}$.

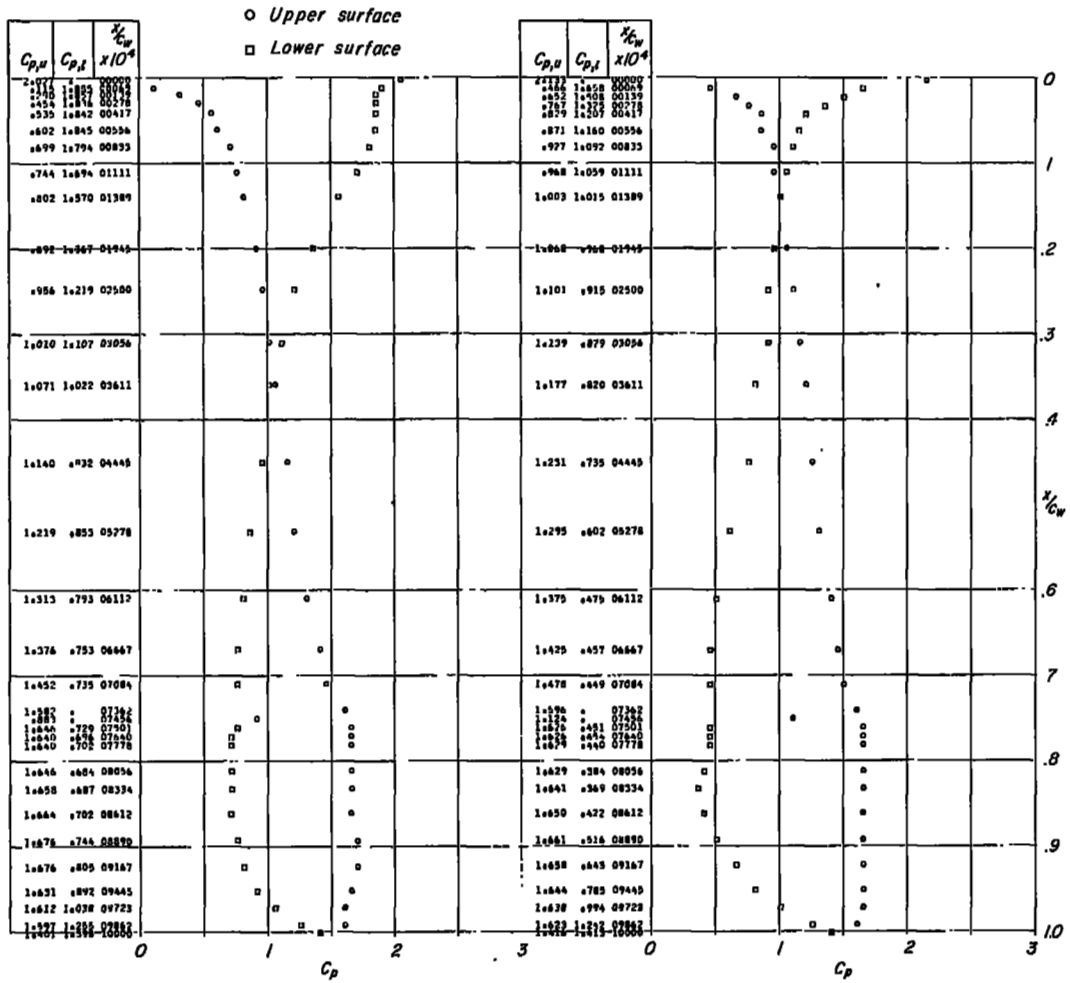
Figure 33.- Continued.



(g) $\alpha = 12^{\circ}$.

(h) $\alpha = 16^{\circ}$.

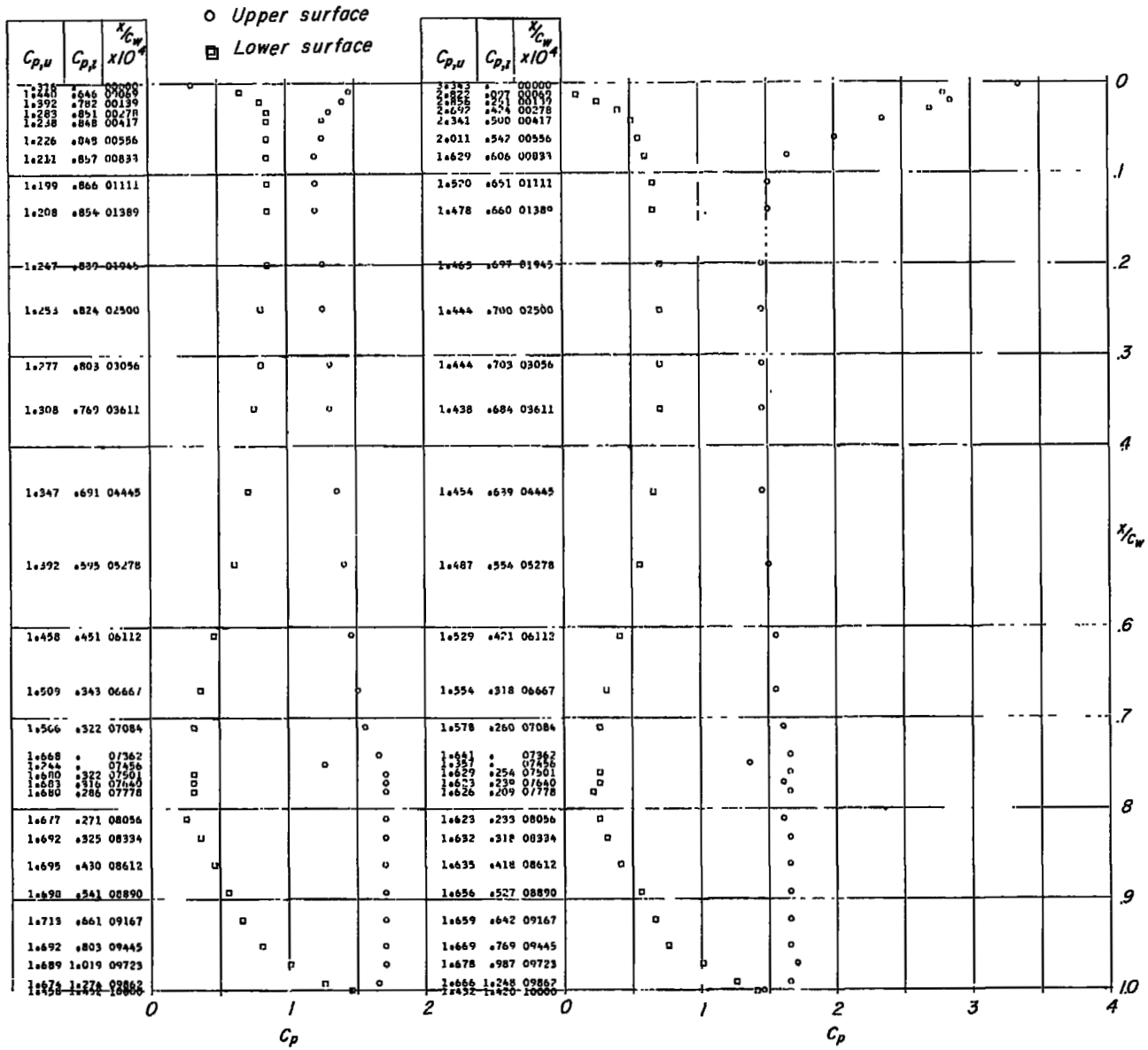
Figure 33.- Concluded.



(a) $\alpha = -12^{\circ}$.

(b) $\alpha = -8^{\circ}$.

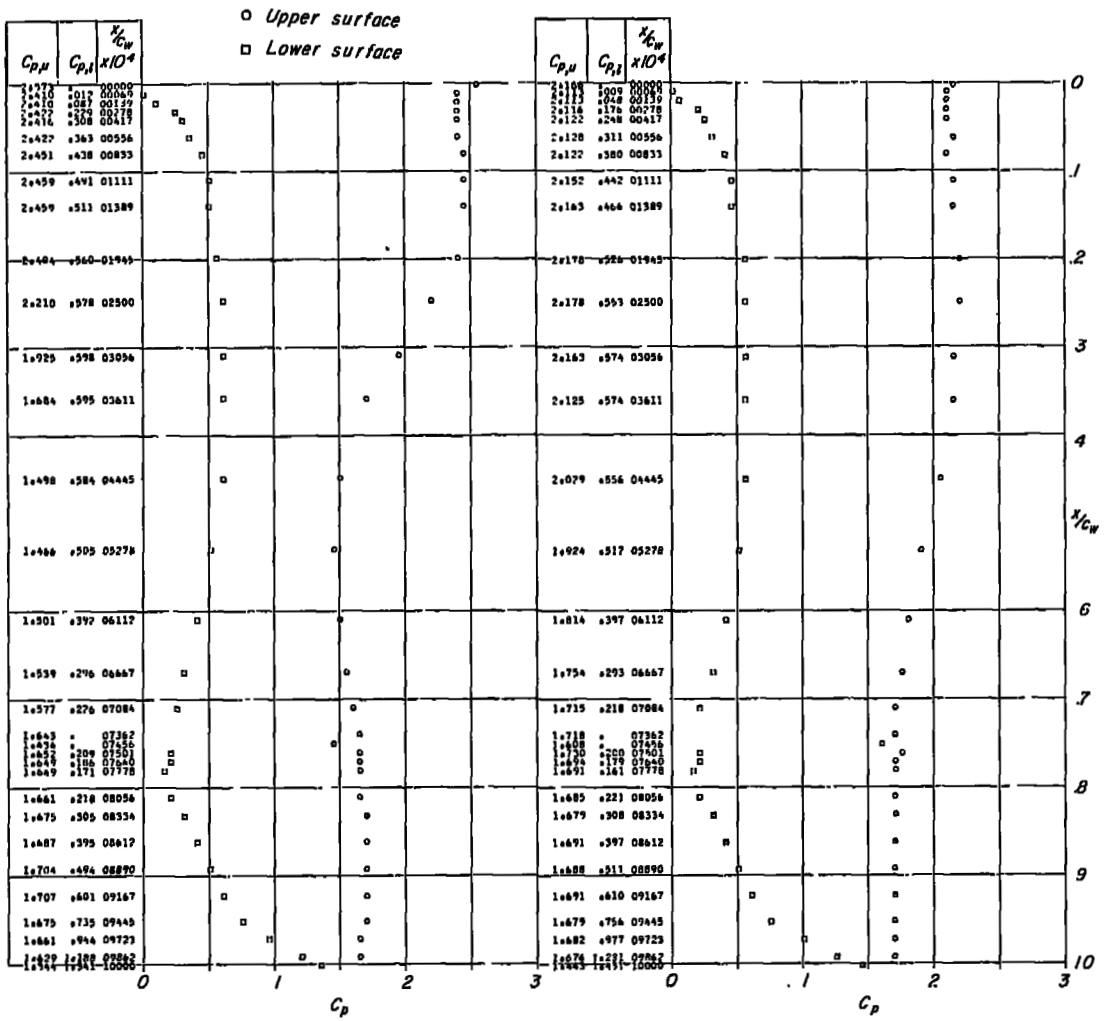
Figure 34.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 0^{\circ}$; $\delta_f = 45^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = -4^{\circ}$.

(d) $\alpha = 0^{\circ}$.

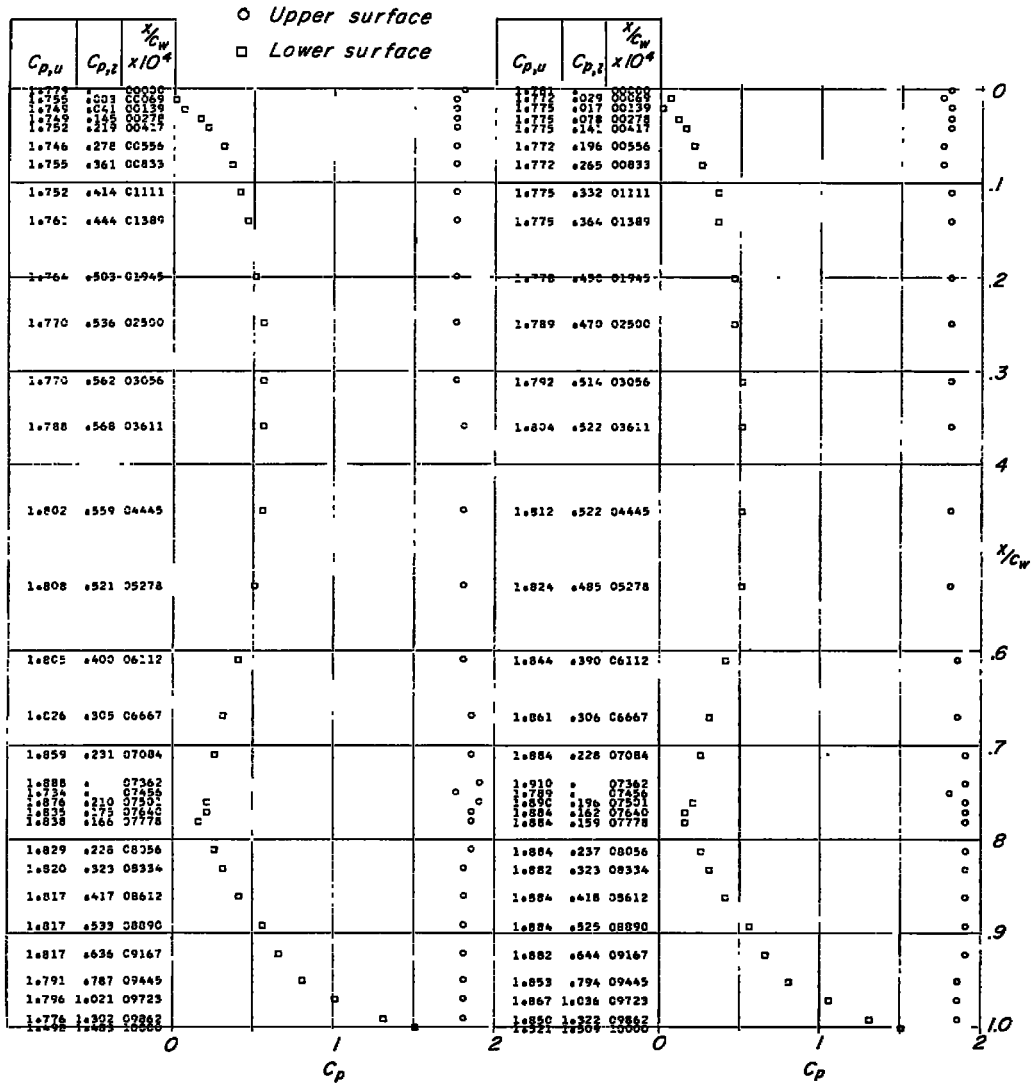
Figure 34.- Continued.



(e) $\alpha = 4^{\circ}$.

(f) $\alpha = 8^{\circ}$.

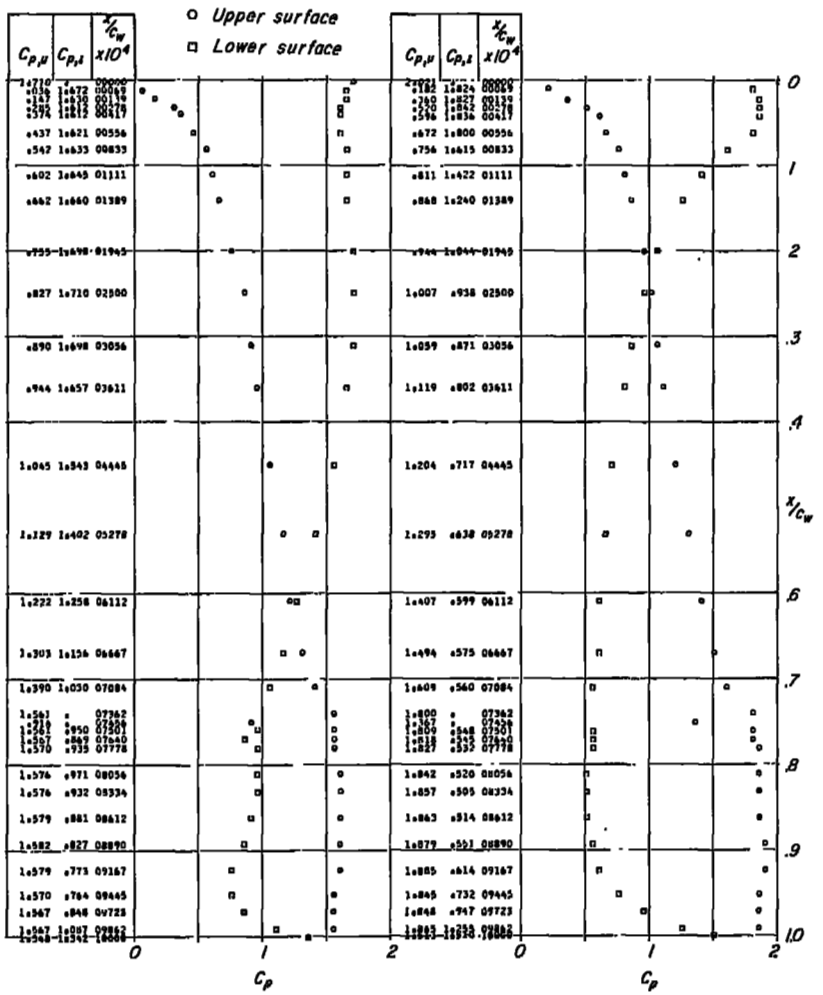
Figure 34.- Continued.



(g) $\alpha = 12^{\circ}$.

(h) $\alpha = 16^{\circ}$.

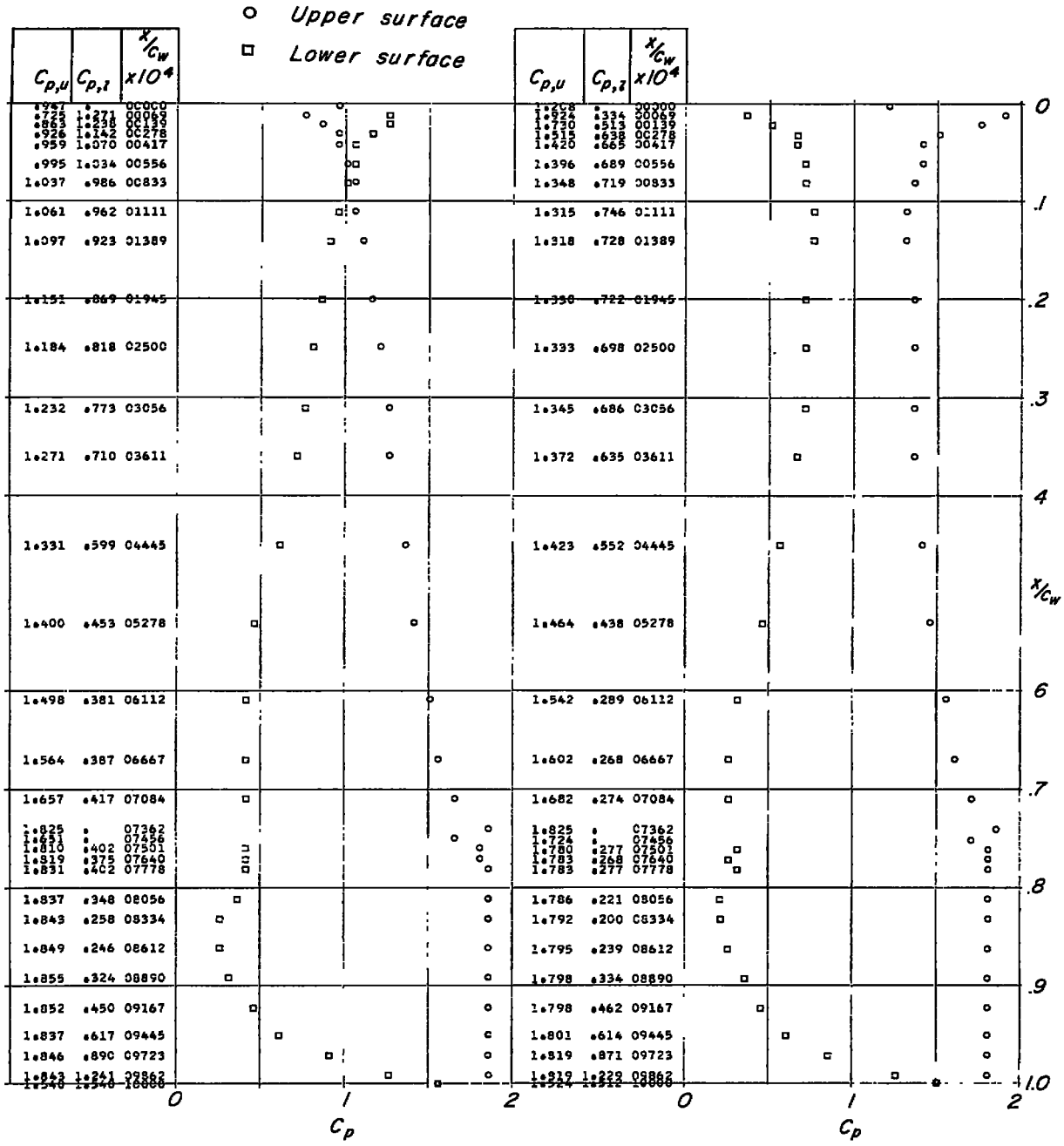
Figure 34.- Concluded.



(a) $\alpha = -16^{\circ}$.

(b) $\alpha = -12^{\circ}$.

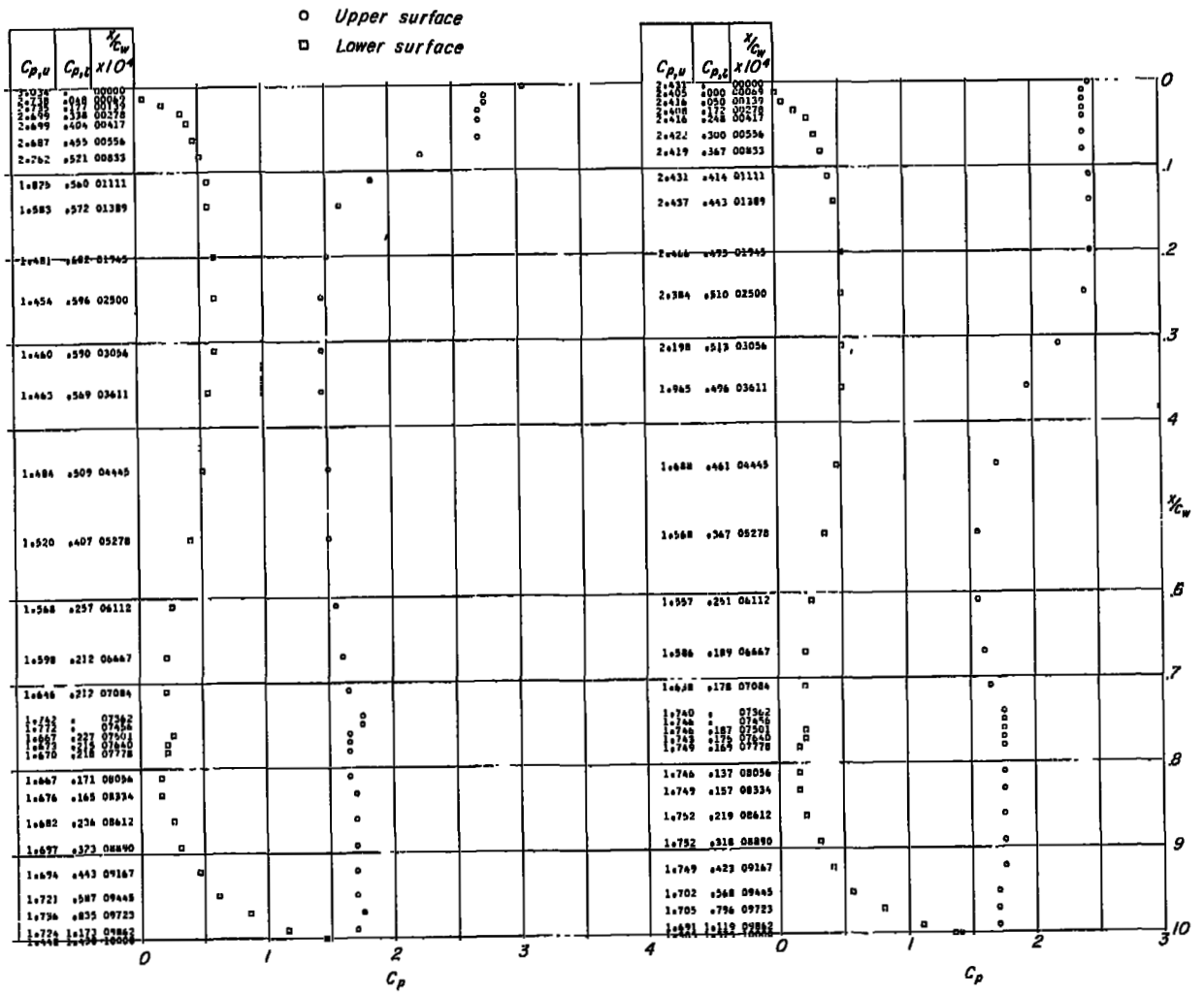
Figure 35.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 0^{\circ}$; $\delta_f = 60^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = -8^{\circ}$.

(d) $\alpha = -4^{\circ}$.

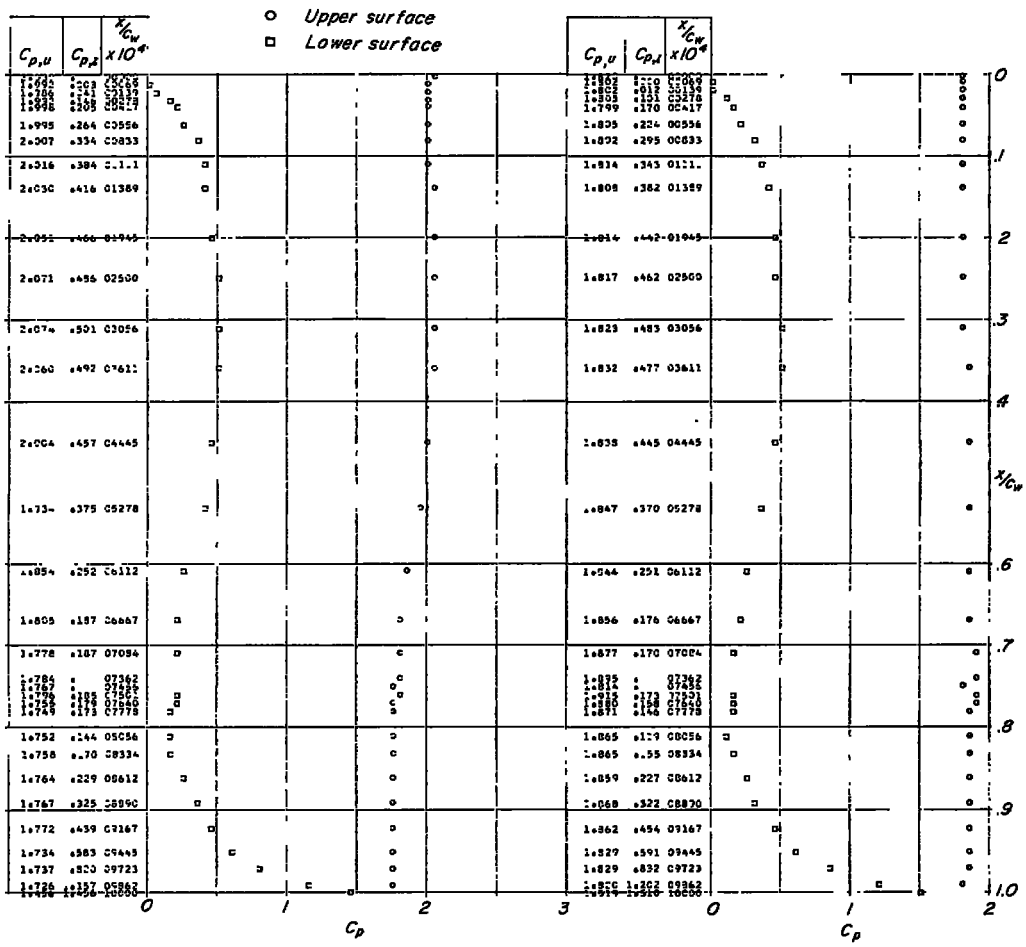
Figure 35.- Continued.



(e) $\alpha = 0^{\circ}$.

(f) $\alpha = 4^{\circ}$.

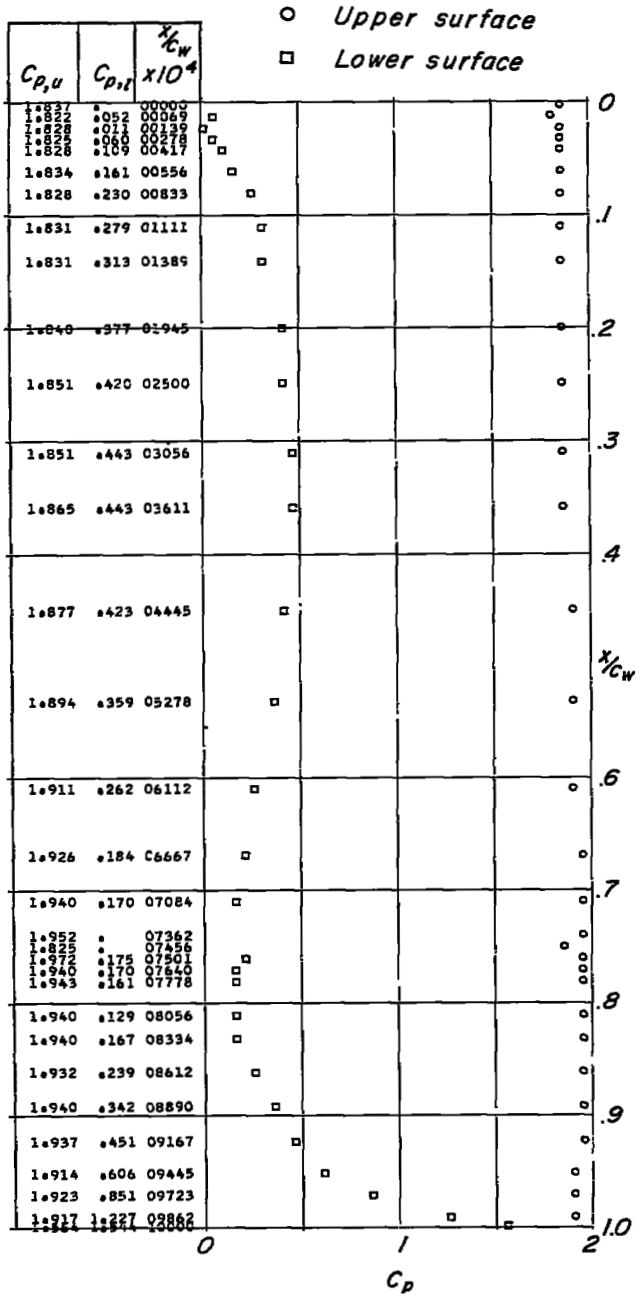
Figure 35.- Continued.



(g) $\alpha = 8^{\circ}$.

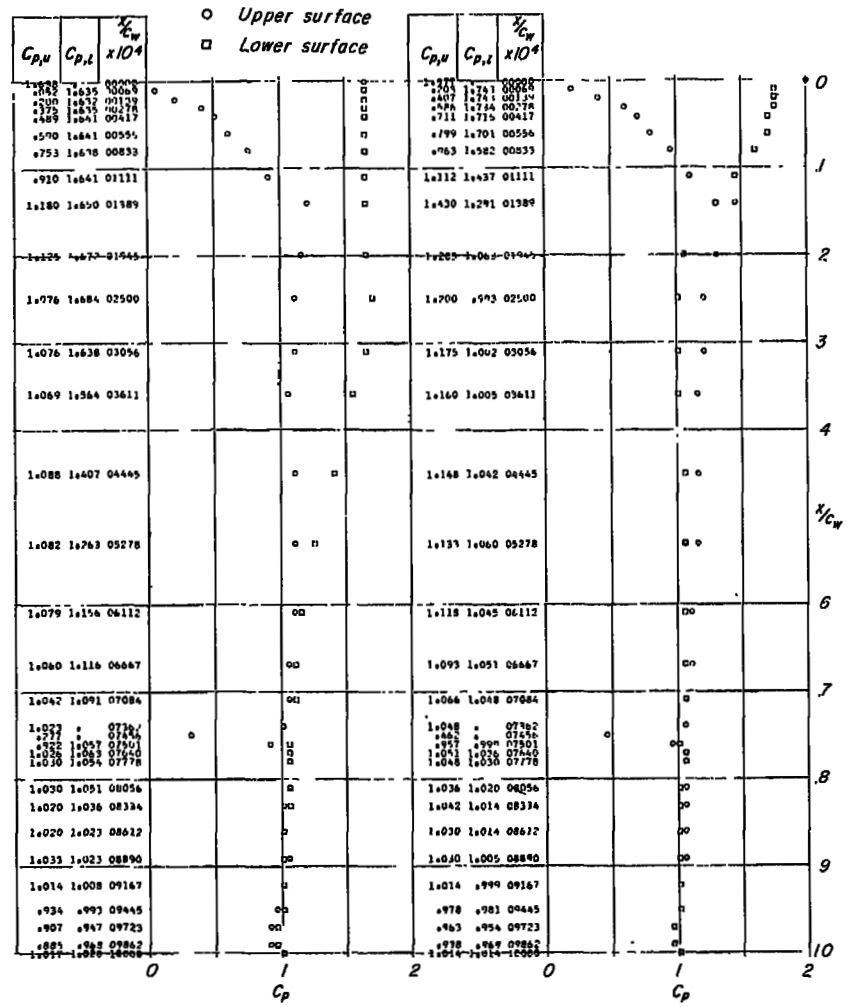
(h) $\alpha = 12^{\circ}$.

Figure 35.- Continued.



(i) $\alpha = 16^{\circ}$.

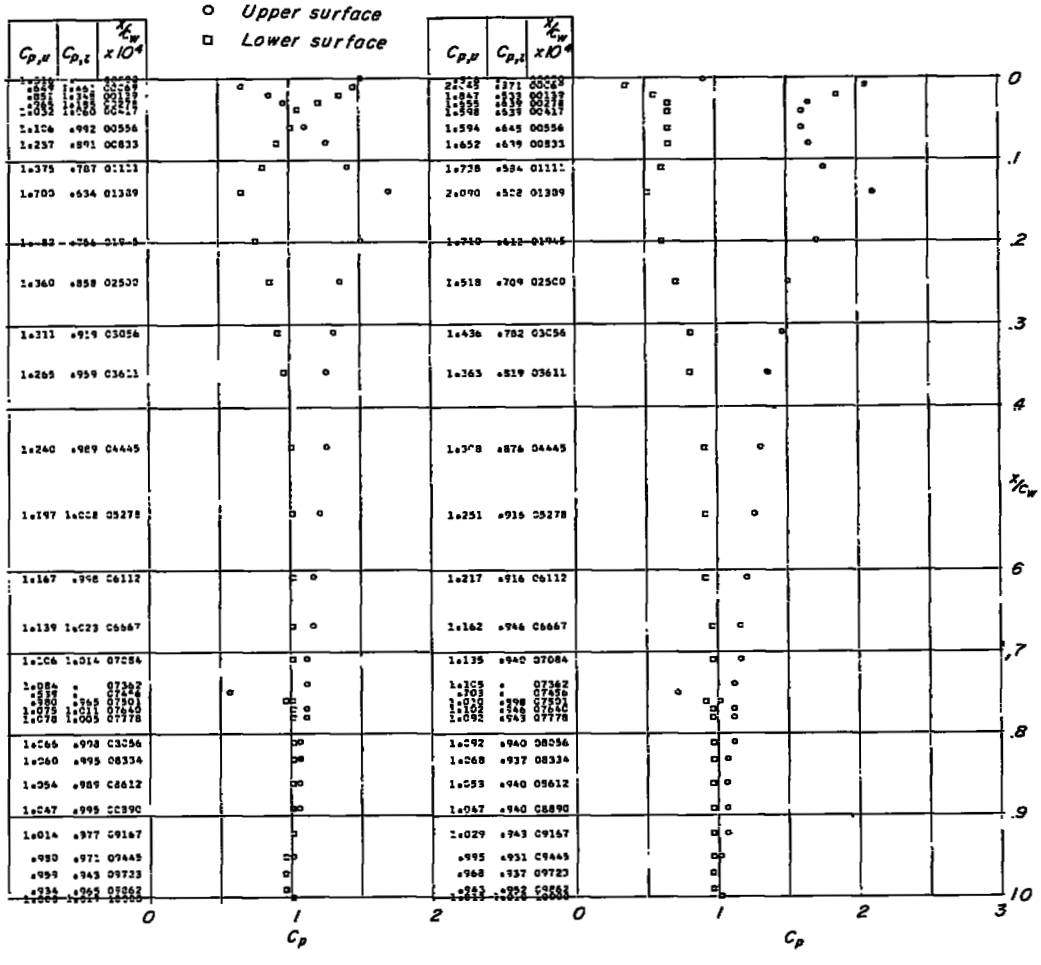
Figure 35.- Concluded.



(a) $\alpha = -4^{\circ}$.

(b) $\alpha = 0^{\circ}$.

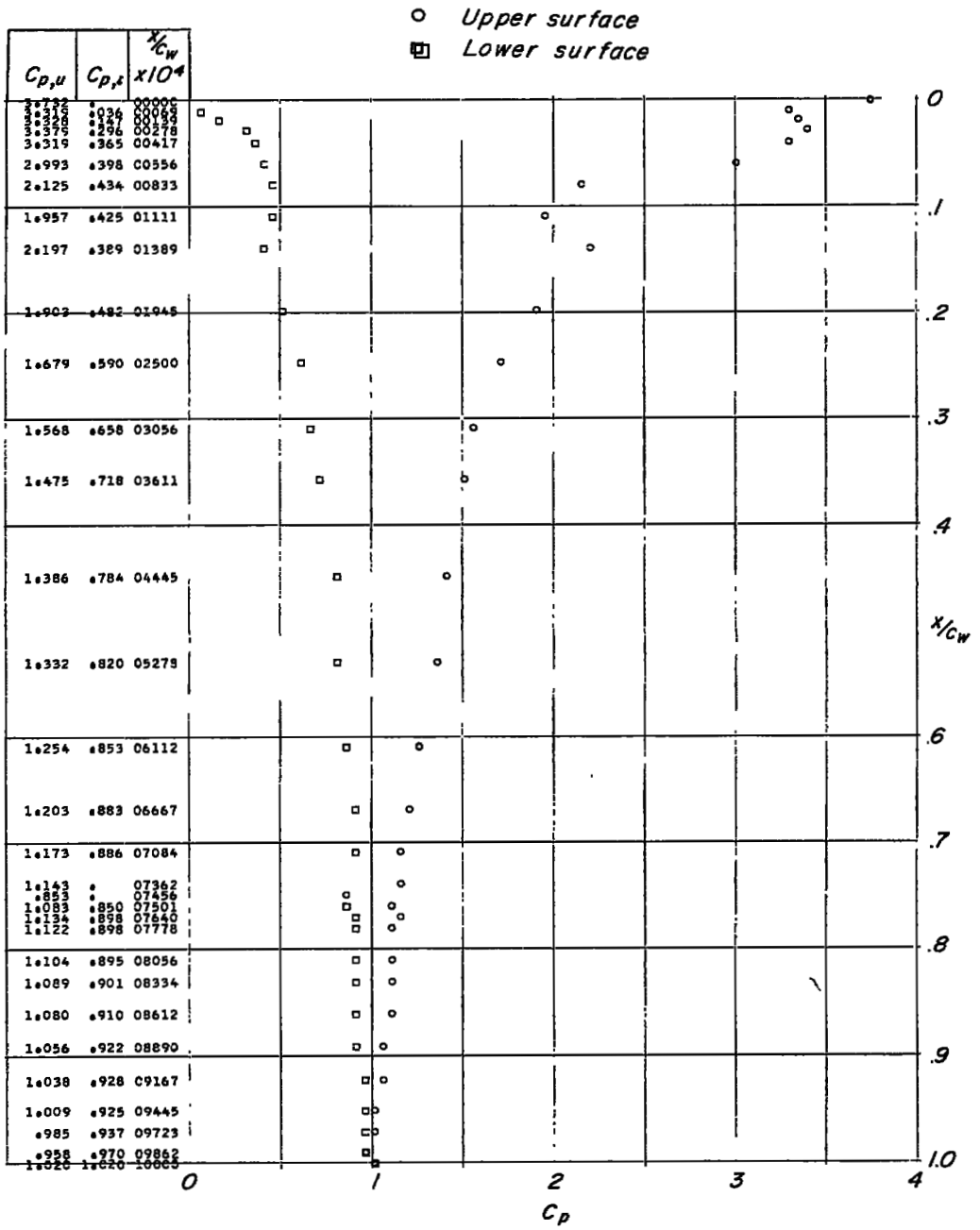
Figure 36.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 15^{\circ}$; $\delta_f = 0^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = 4^{\circ}$.

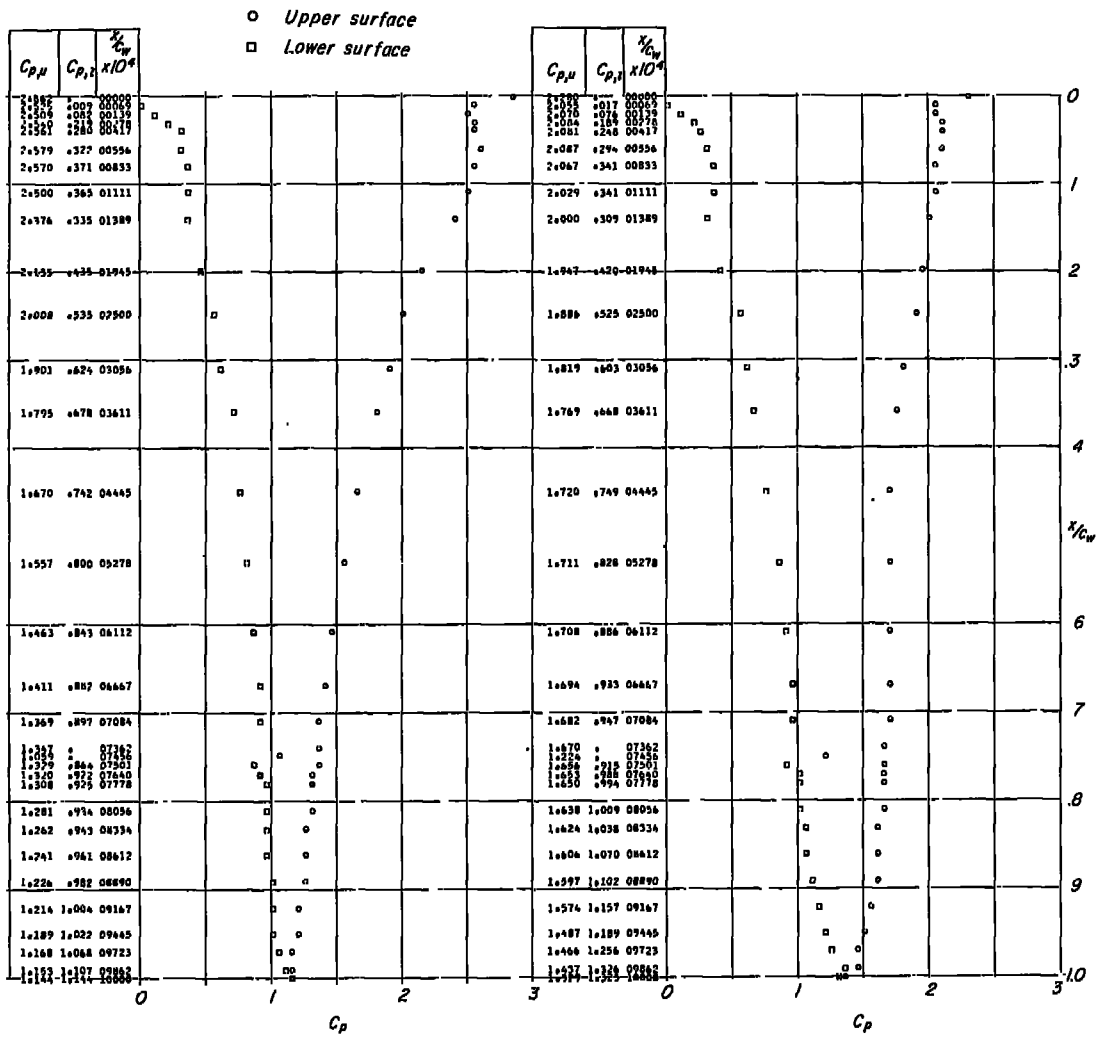
(d) $\alpha = 8^{\circ}$.

Figure 36.- Continued.



(e) $\alpha = 12^{\circ}$.

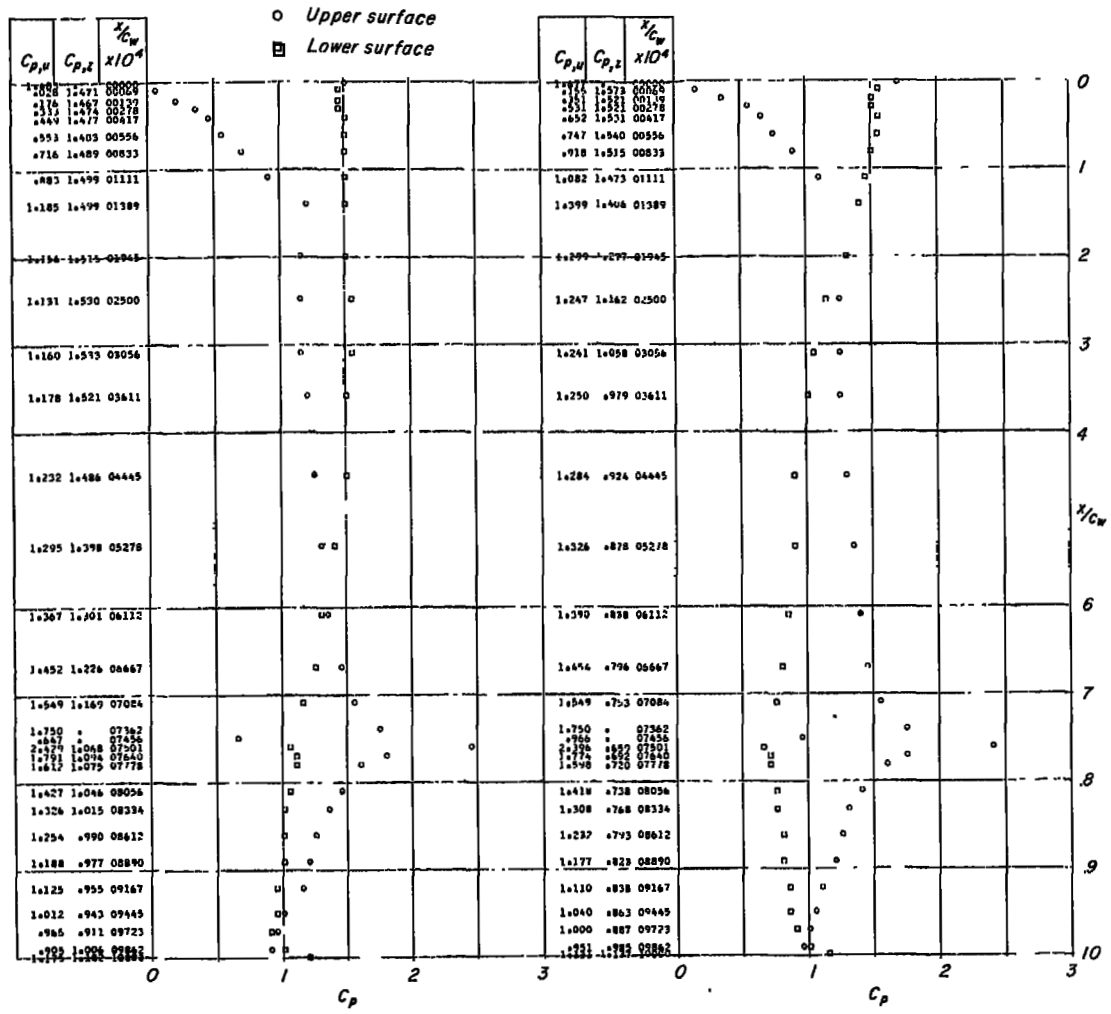
Figure 36.- Continued.



(f) $\alpha = 16^{\circ}$.

(g) $\alpha = 20^{\circ}$.

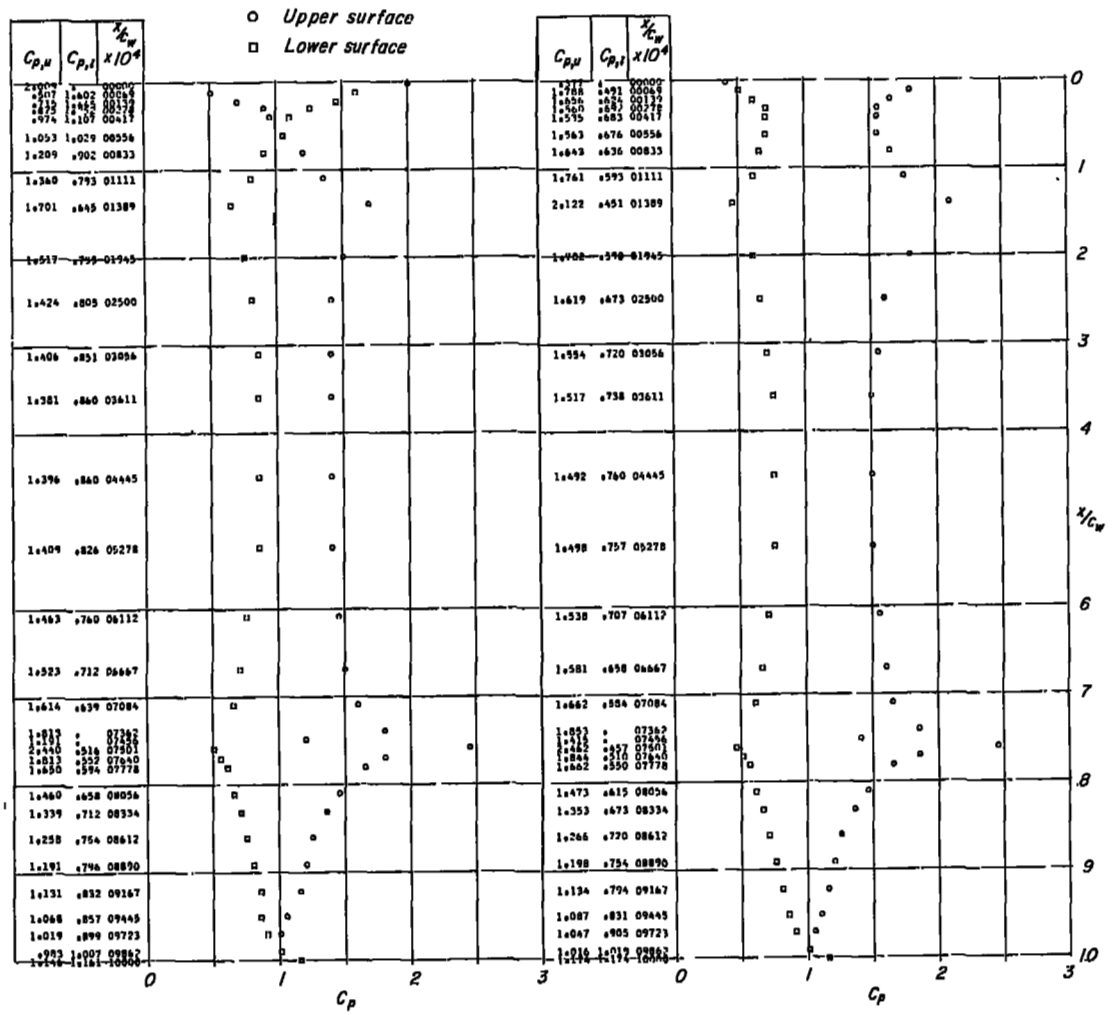
Figure 36.- Concluded.



(a) $\alpha = -8^{\circ}$.

(b) $\alpha = -4^{\circ}$.

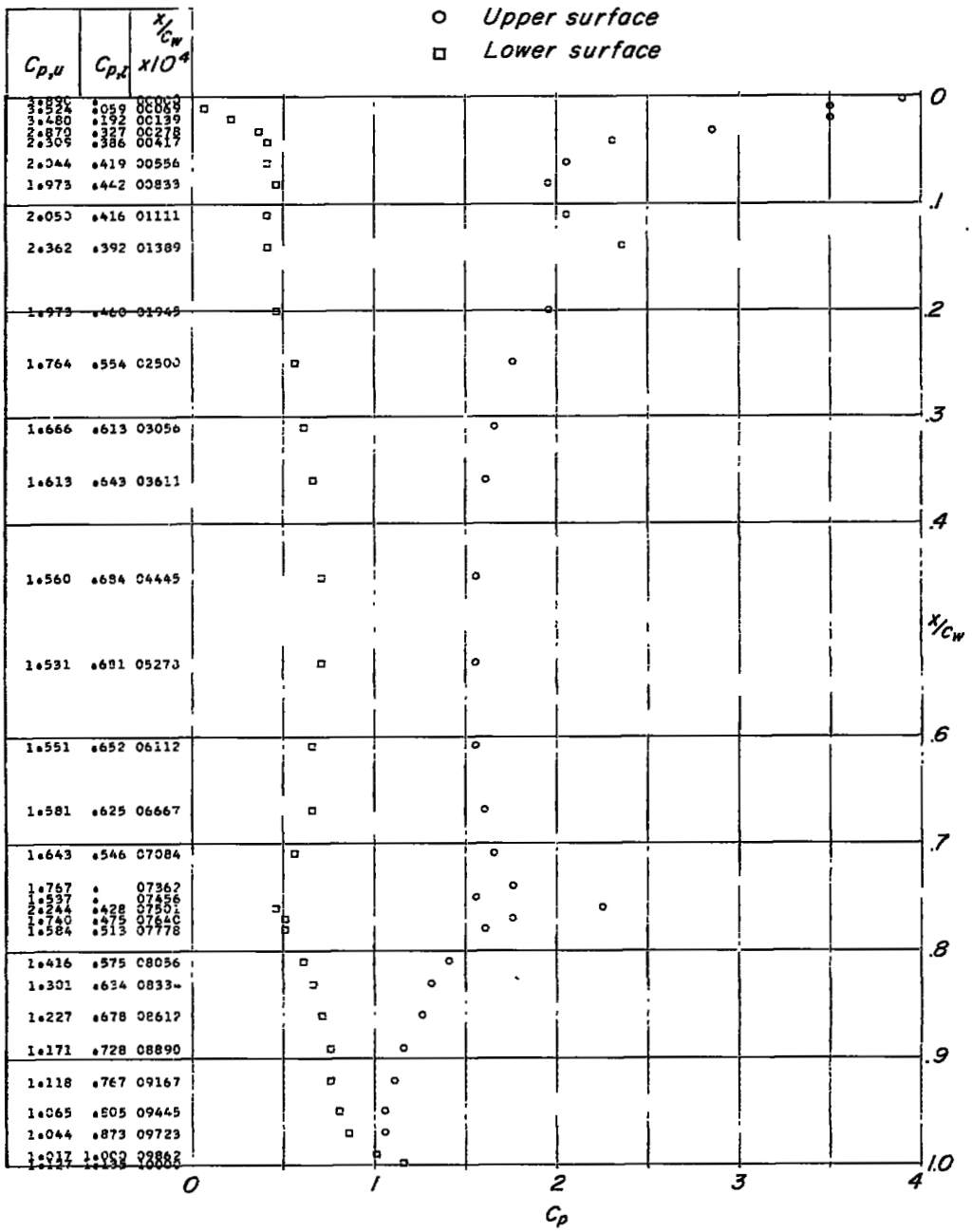
Figure 37.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 15^{\circ}$; $\delta_f = 15^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = 0^{\circ}$.

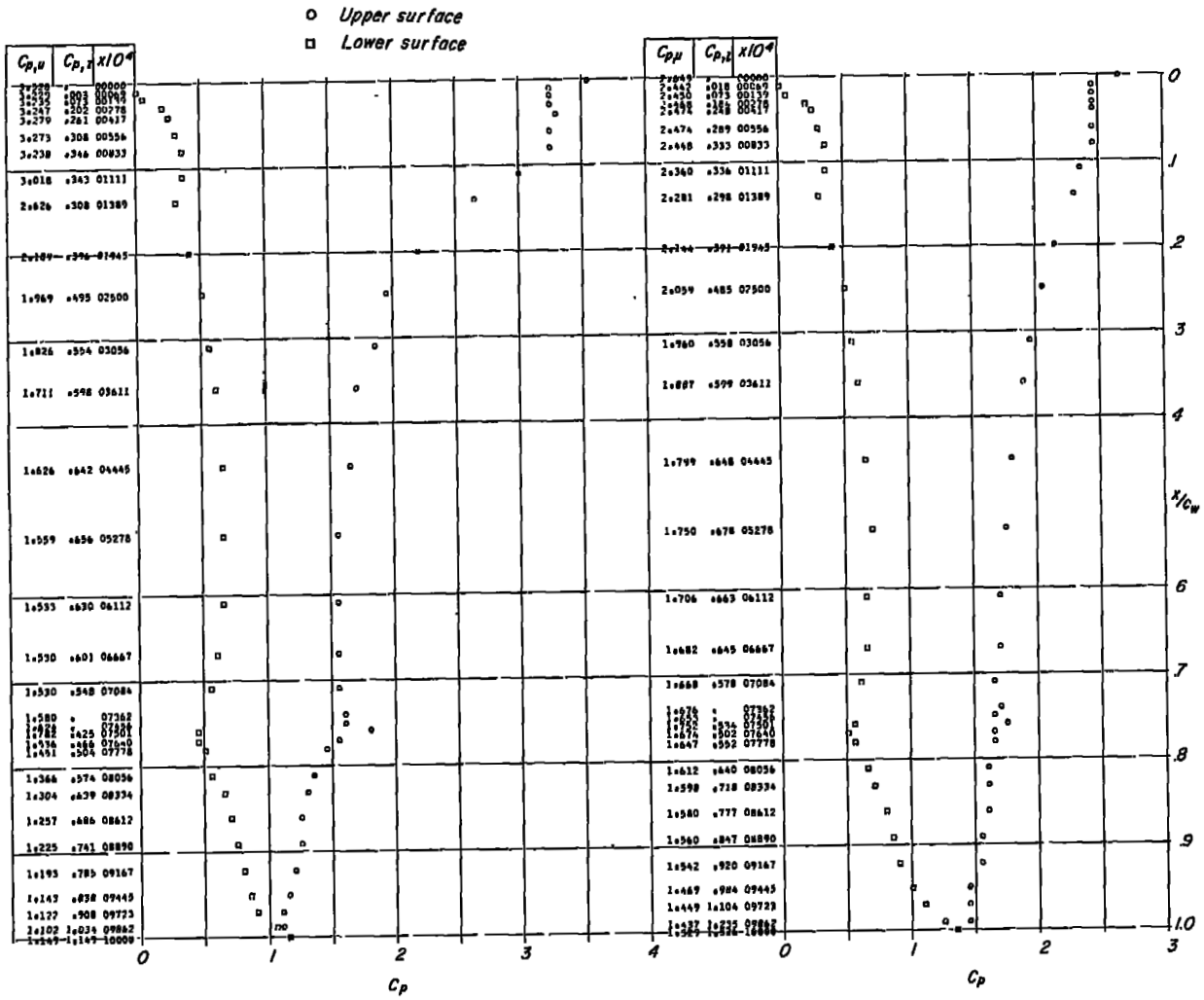
(d) $\alpha = 4^{\circ}$.

Figure 37.- Continued.



(e) $\alpha = 8^{\circ}$.

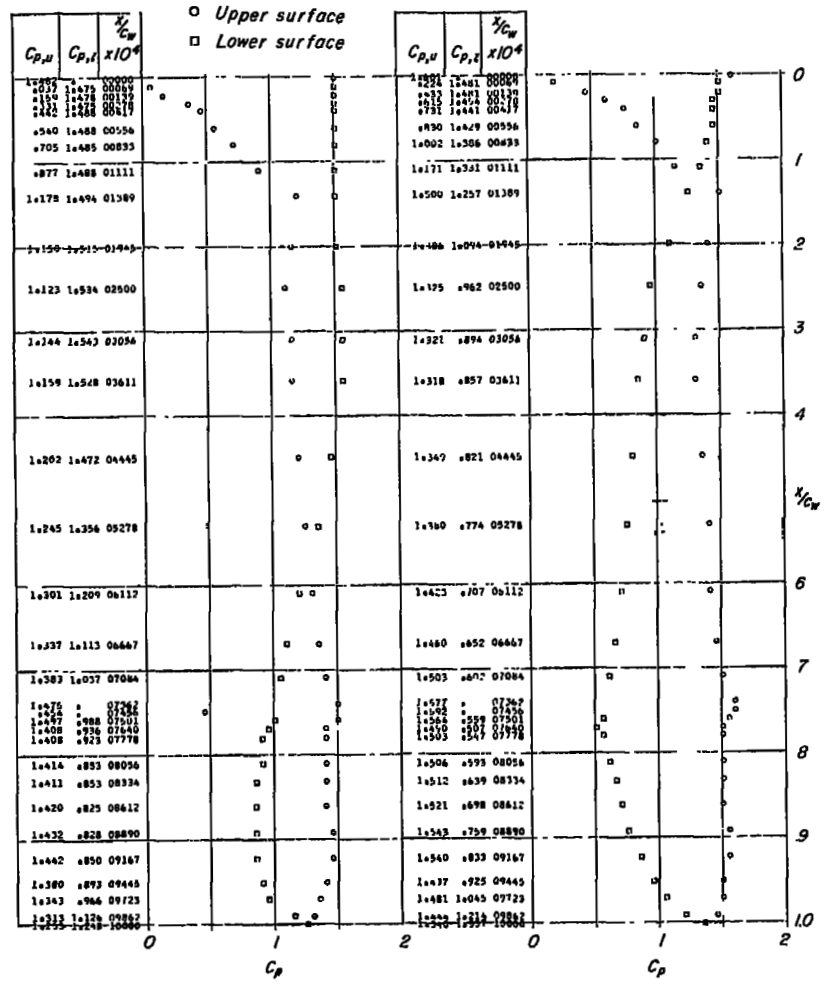
Figure 37.- Continued.



(f) $\alpha = 12^{\circ}$.

(g) $\alpha = 16^{\circ}$.

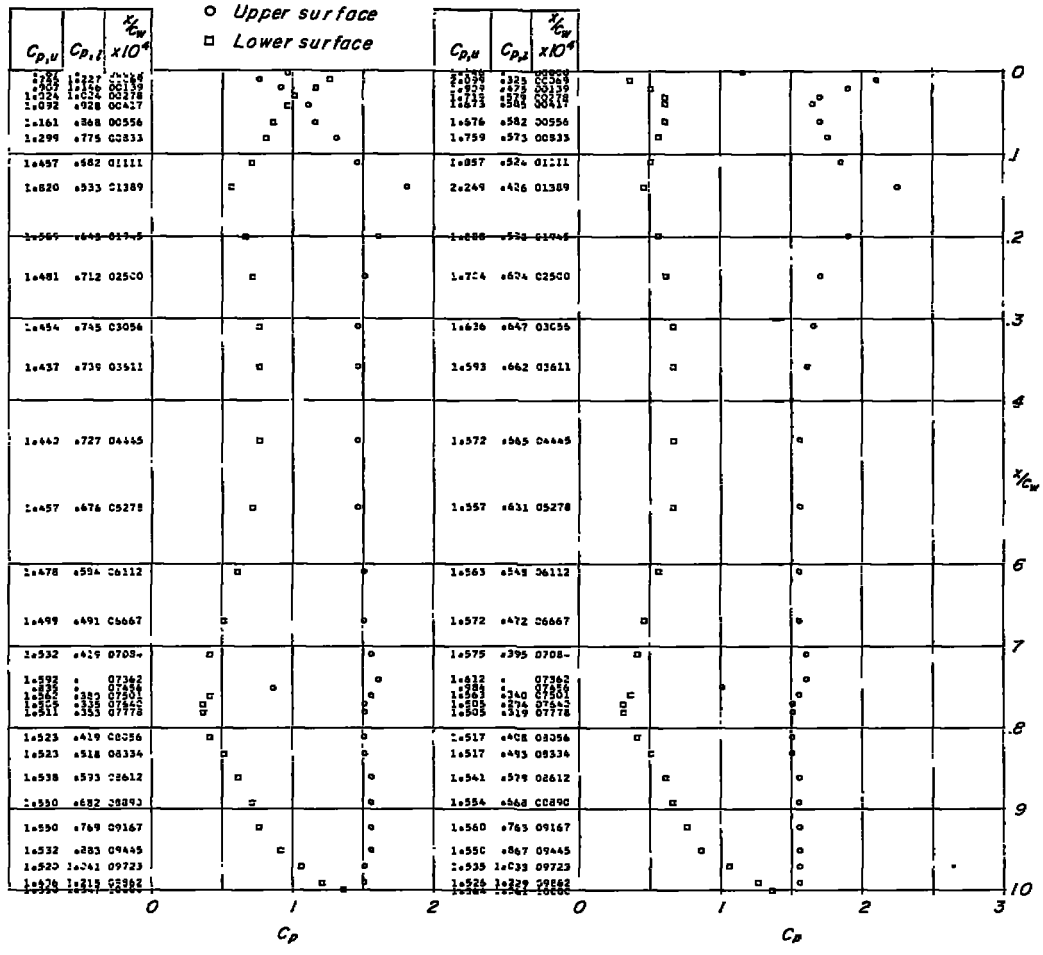
Figure 37.- Concluded.



(a) $\alpha = -8^{\circ}$.

(b) $\alpha = -4^{\circ}$.

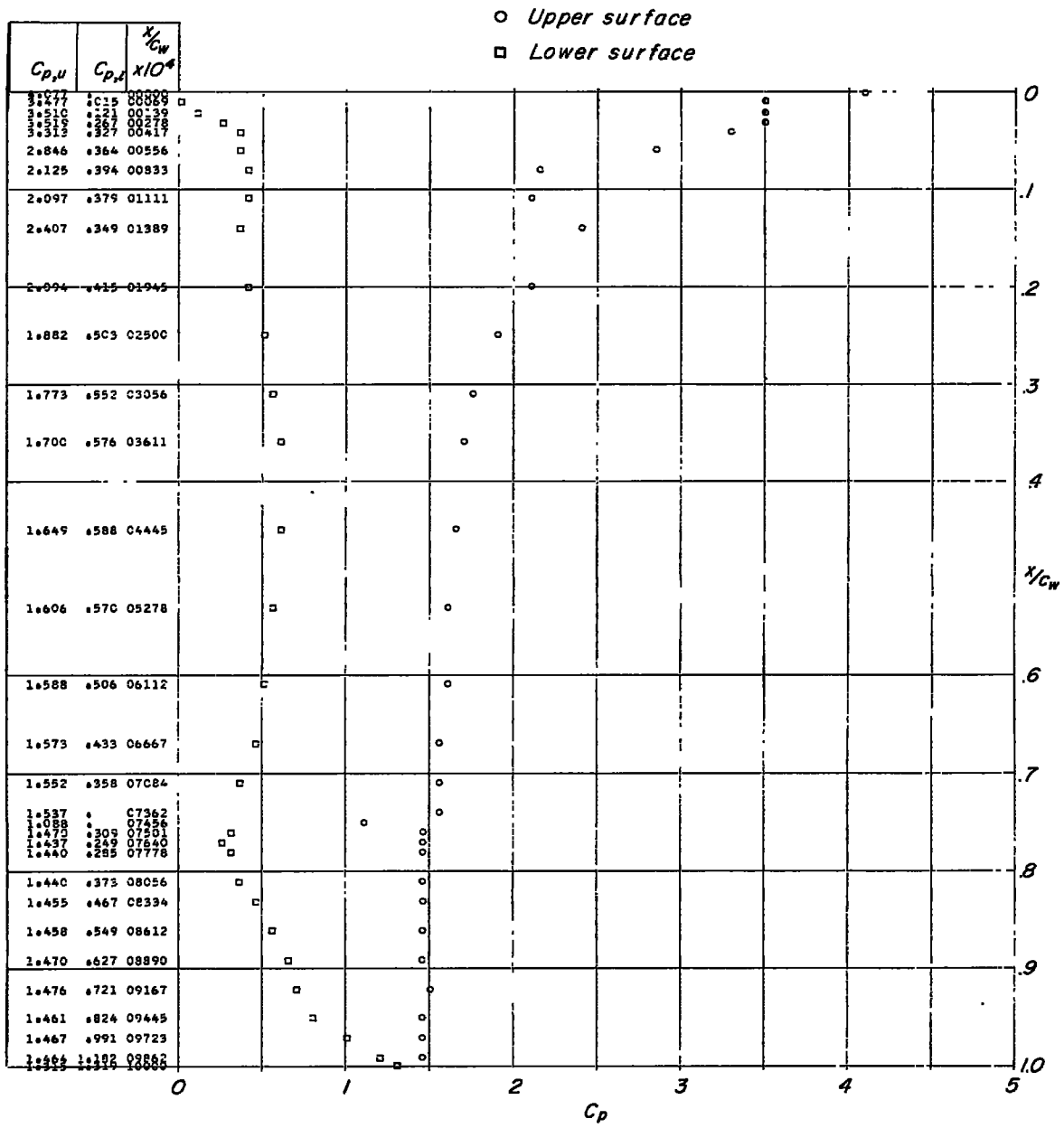
Figure 38.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 15^{\circ}$; $\delta_f = 30^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = 0^{\circ}$.

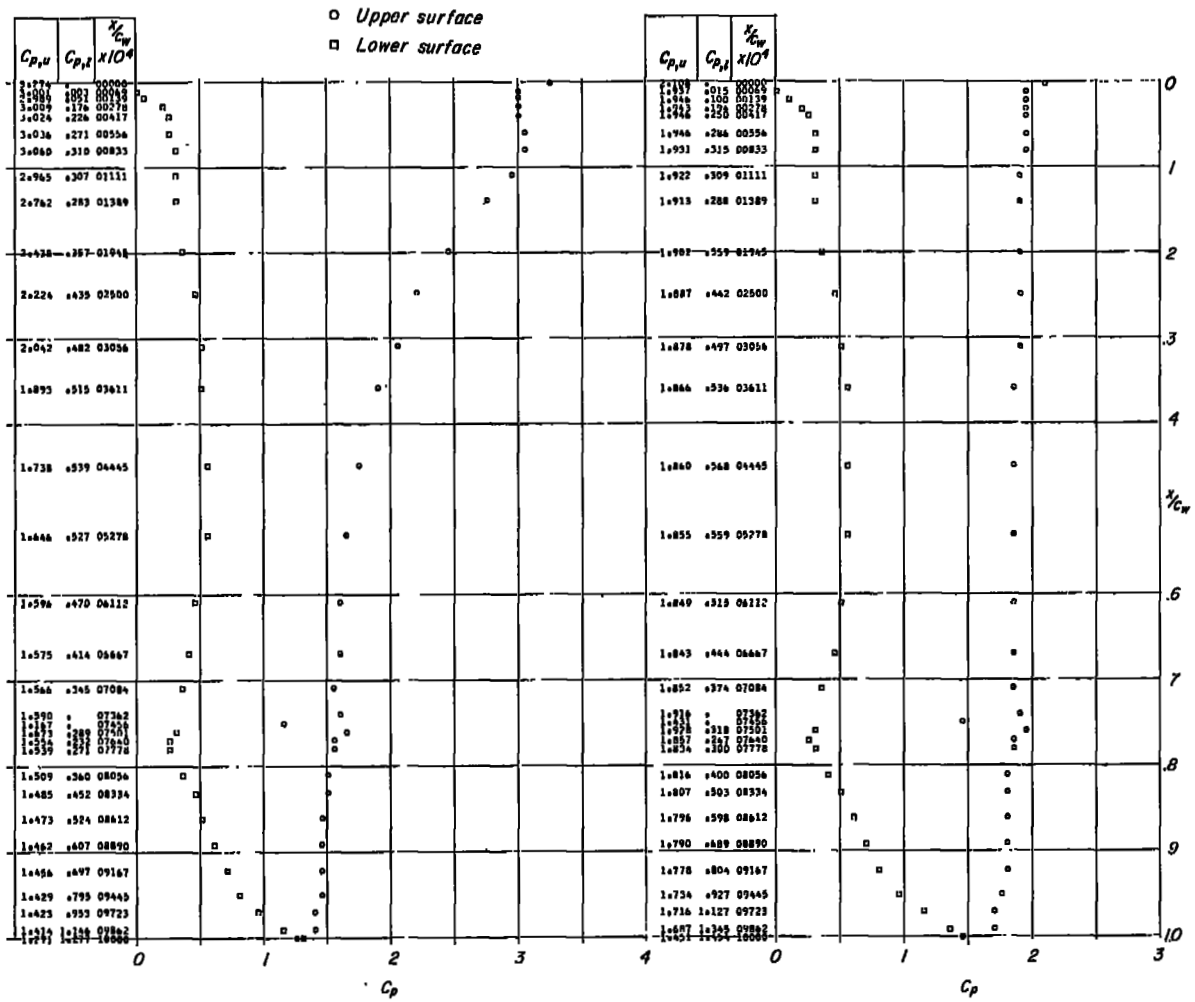
(d) $\alpha = 4^{\circ}$.

Figure 38.- Continued.



(e) $\alpha = 8^{\circ}$.

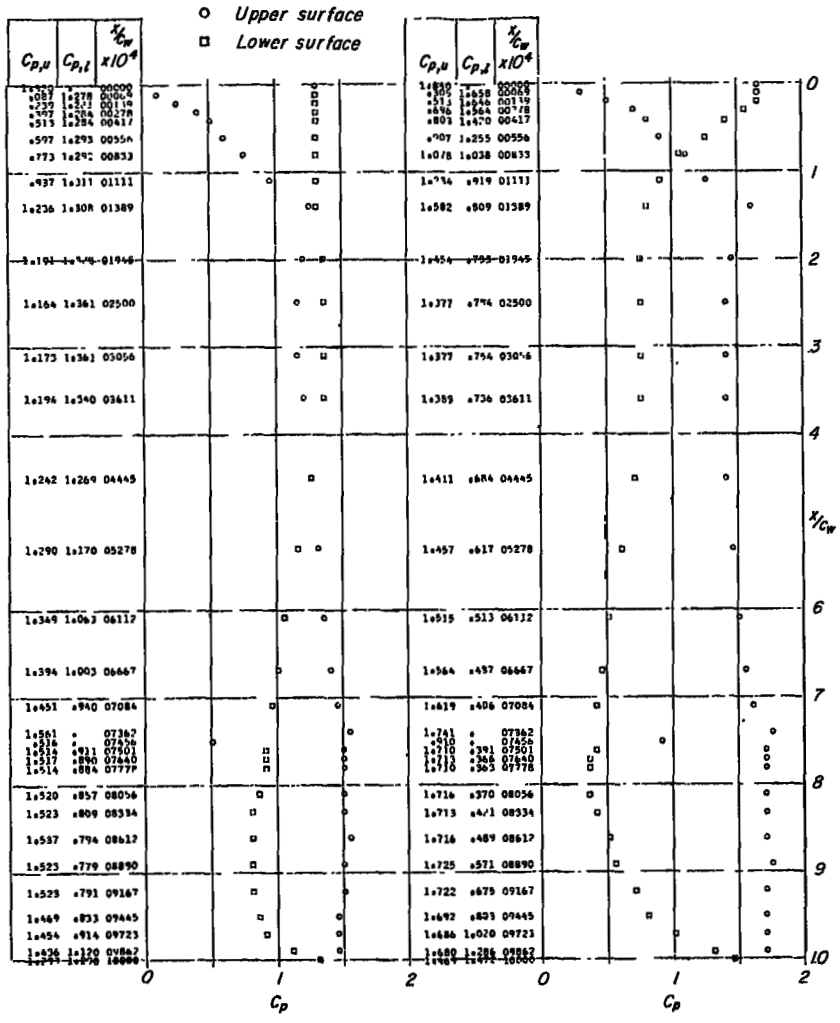
Figure 38.- Continued.



(f) $\alpha = 12^{\circ}$.

(g) $\alpha = 16^{\circ}$.

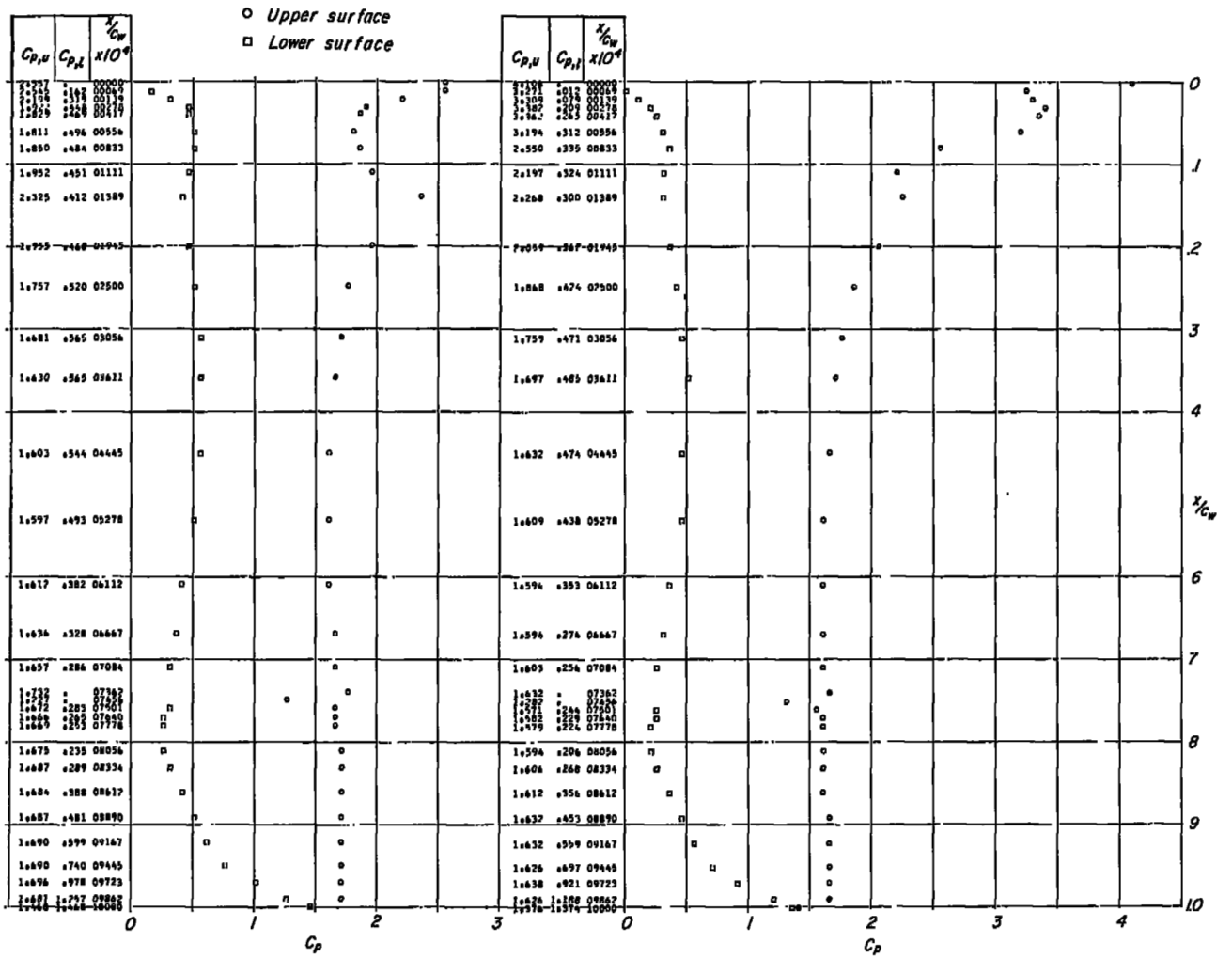
Figure 38.- Concluded.



(a) $\alpha = -8^{\circ}$.

(b) $\alpha = -4^{\circ}$.

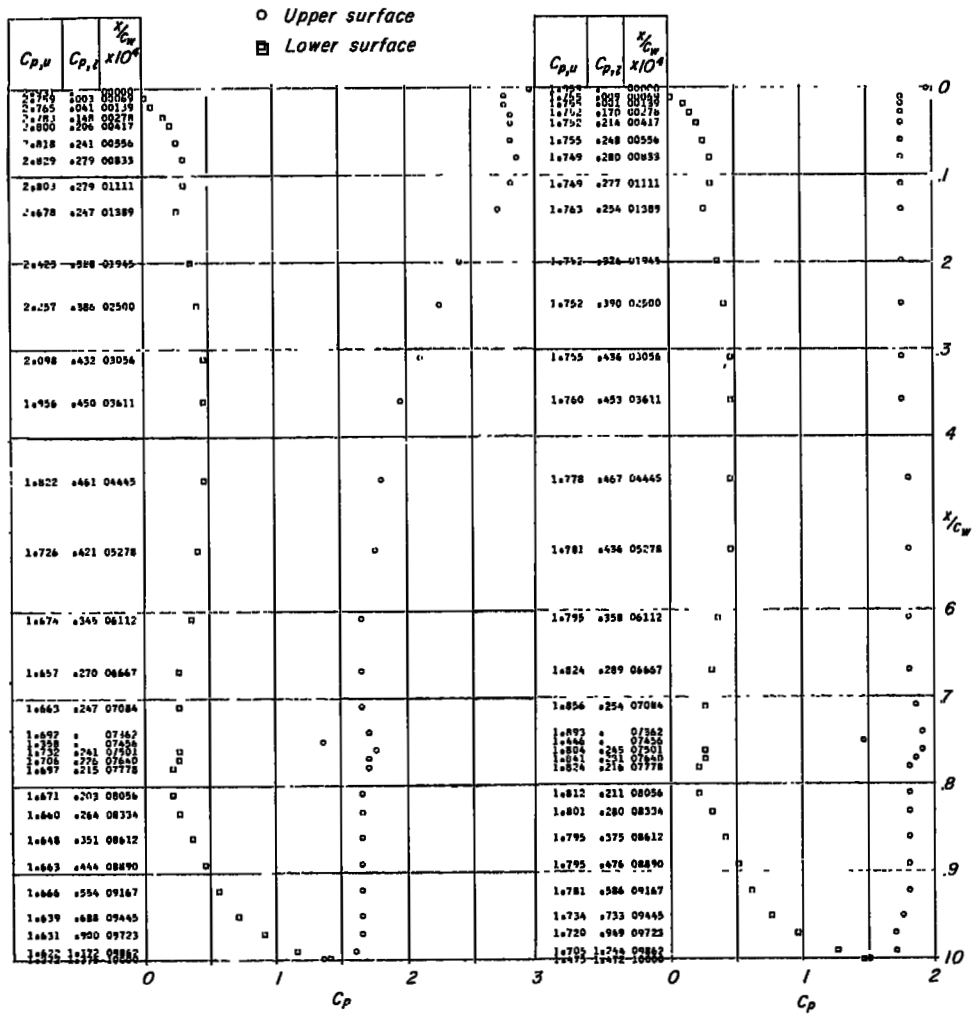
Figure 39.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 15^{\circ}$; $\delta_f = 45^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = 4^{\circ}$.

(d) $\alpha = 8^{\circ}$.

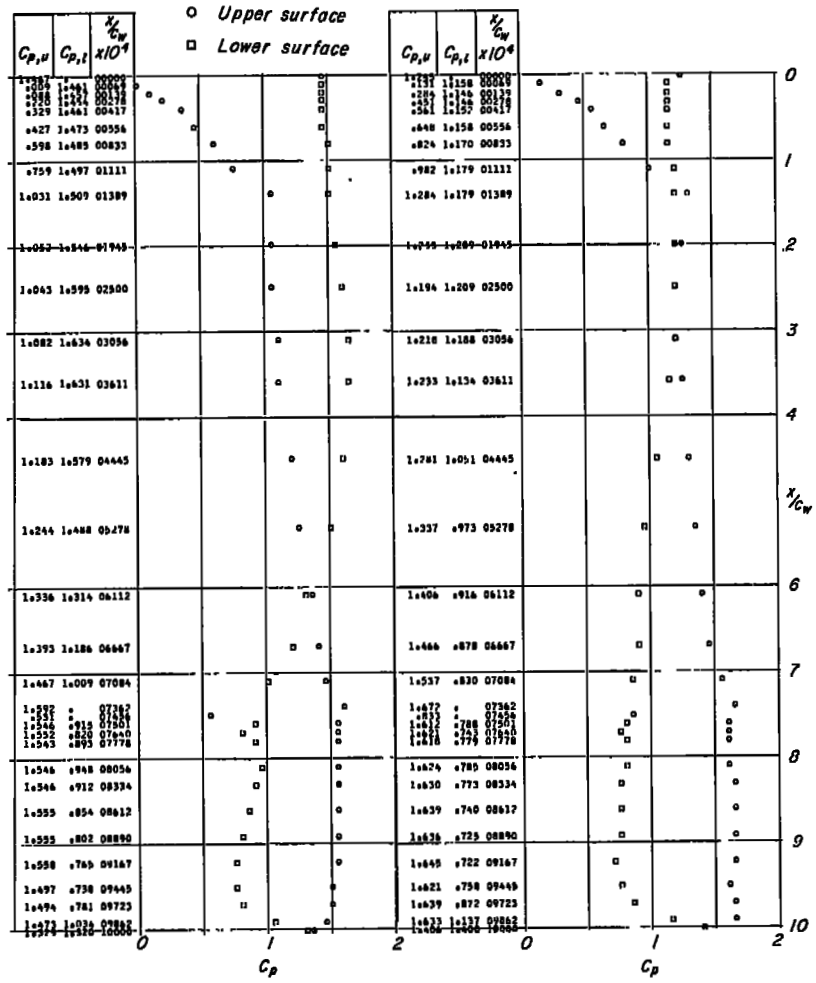
Figure 39.- Continued.



(e) $\alpha = 12^{\circ}$.

(f) $\alpha = 16^{\circ}$.

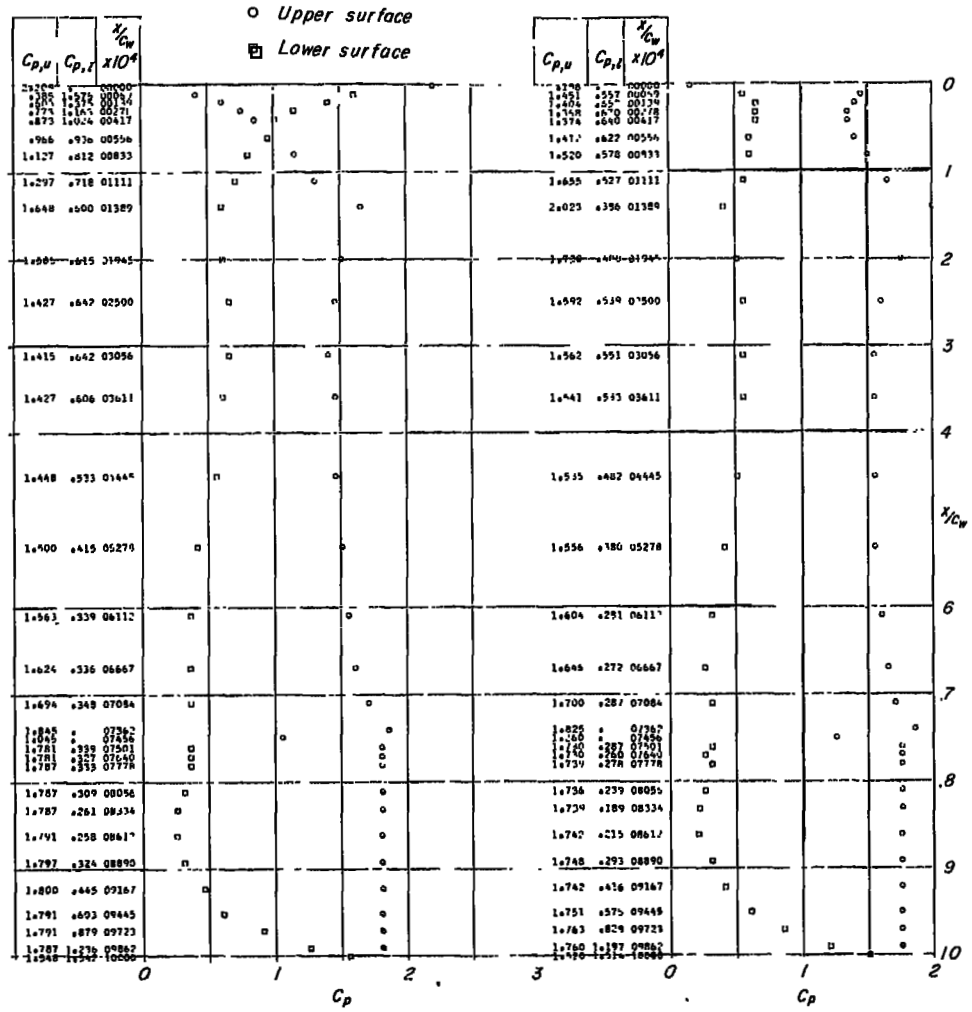
Figure 39.- Concluded.



(a) $\alpha = -12^{\circ}$.

(b) $\alpha = -8^{\circ}$.

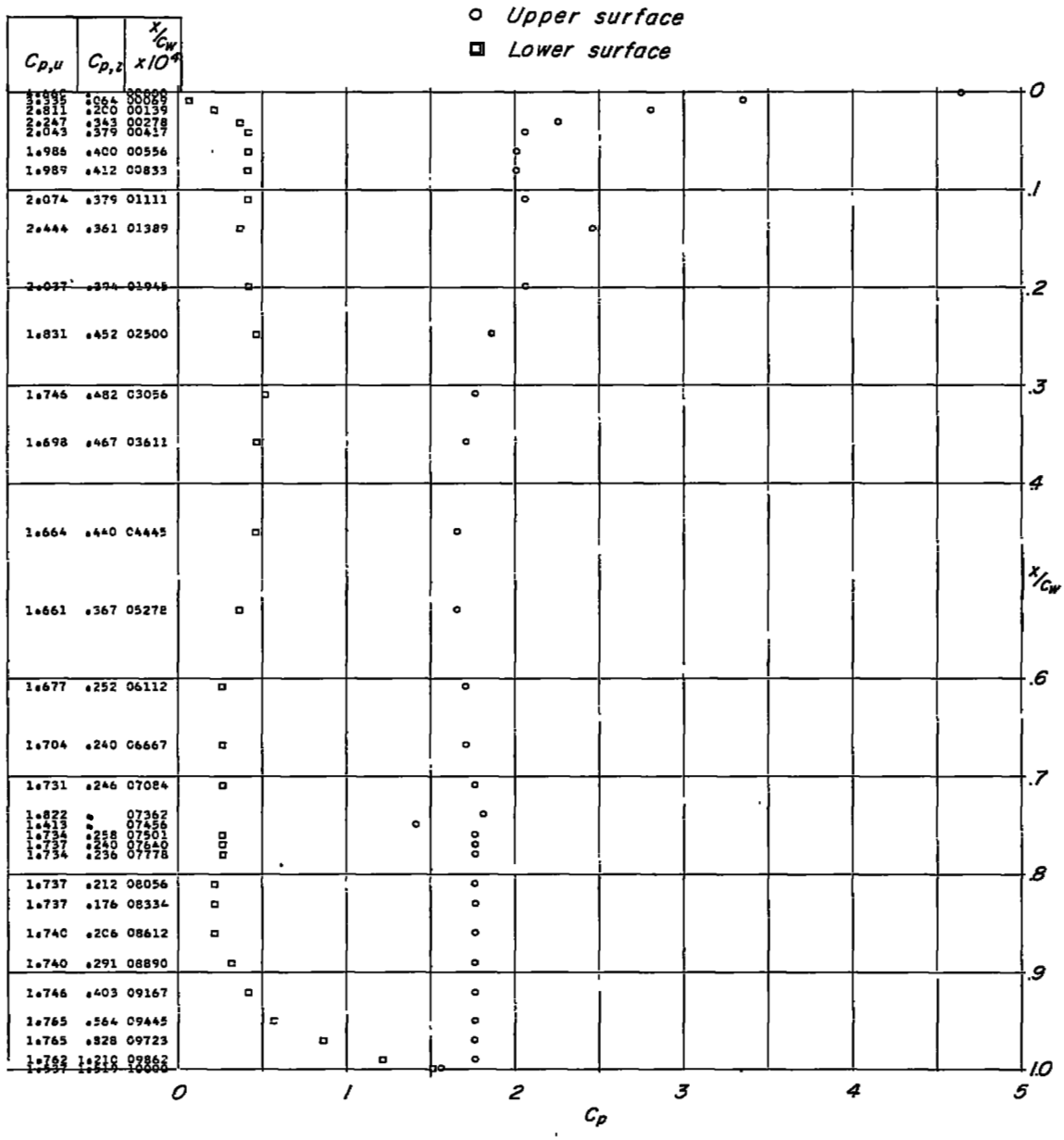
Figure 40.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 15^{\circ}$; $\delta_f = 60^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = -4^{\circ}$.

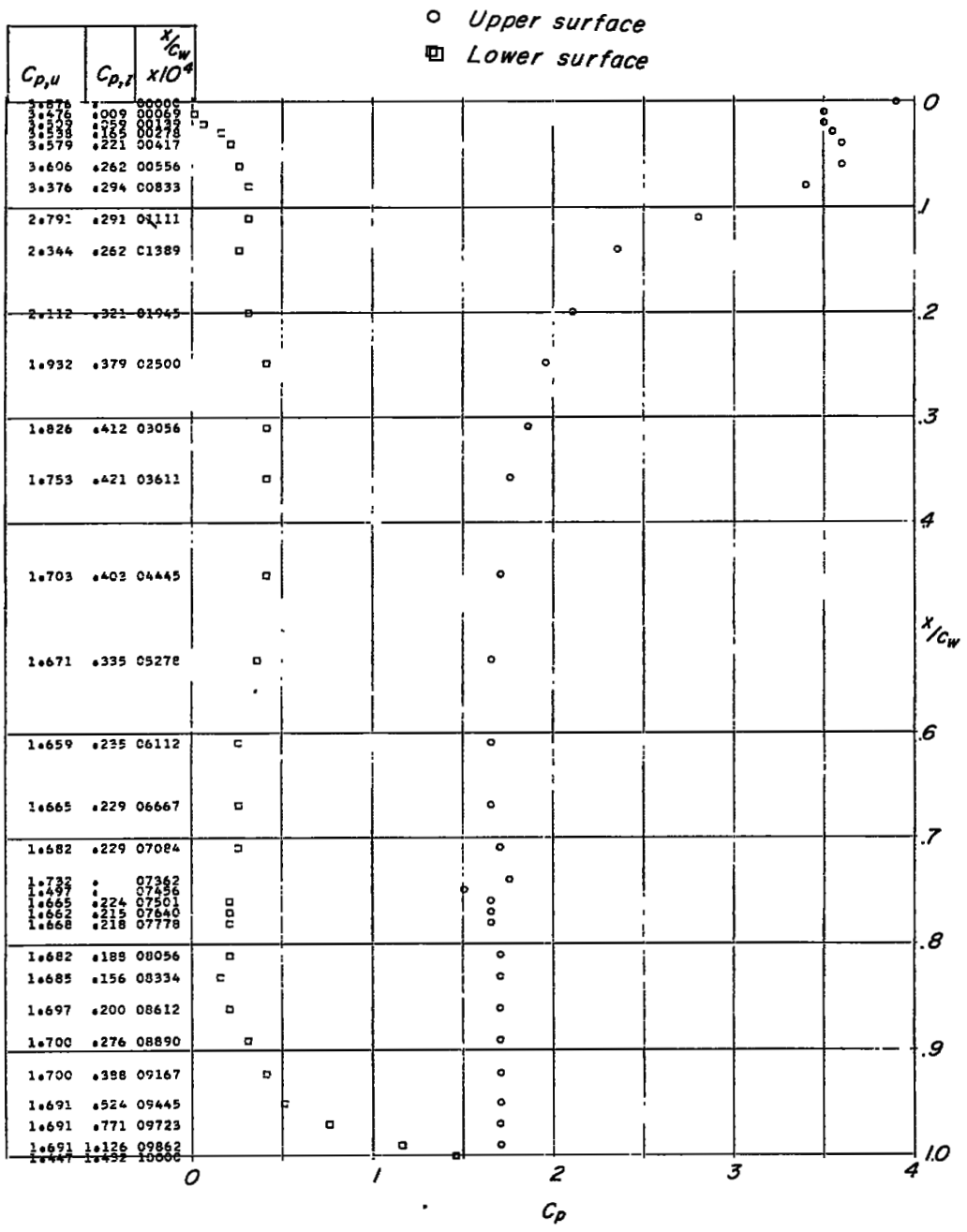
(d) $\alpha = 0^{\circ}$.

Figure 40.- Continued.



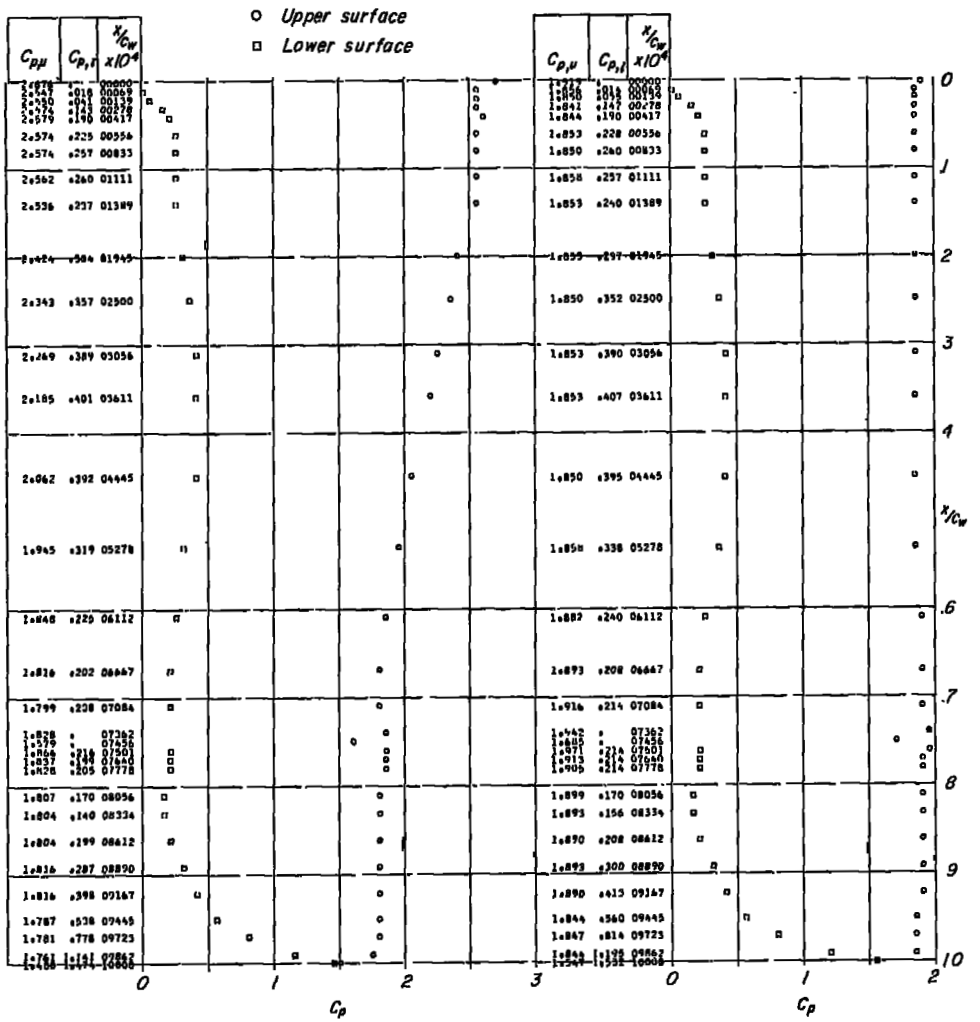
(e) $\alpha = 4^{\circ}$.

Figure 40.- Continued.



(f) $\alpha = 8^{\circ}$.

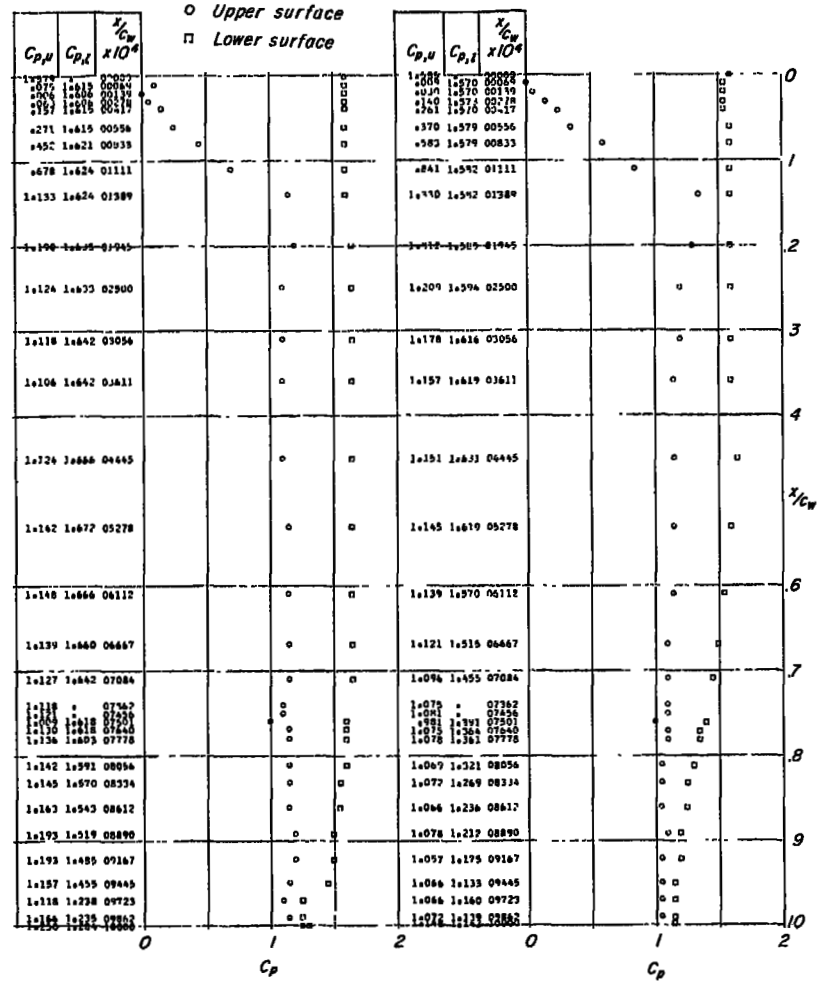
Figure 40.- Continued.



(g) $\alpha = 12^{\circ}$.

(h) $\alpha = 16^{\circ}$.

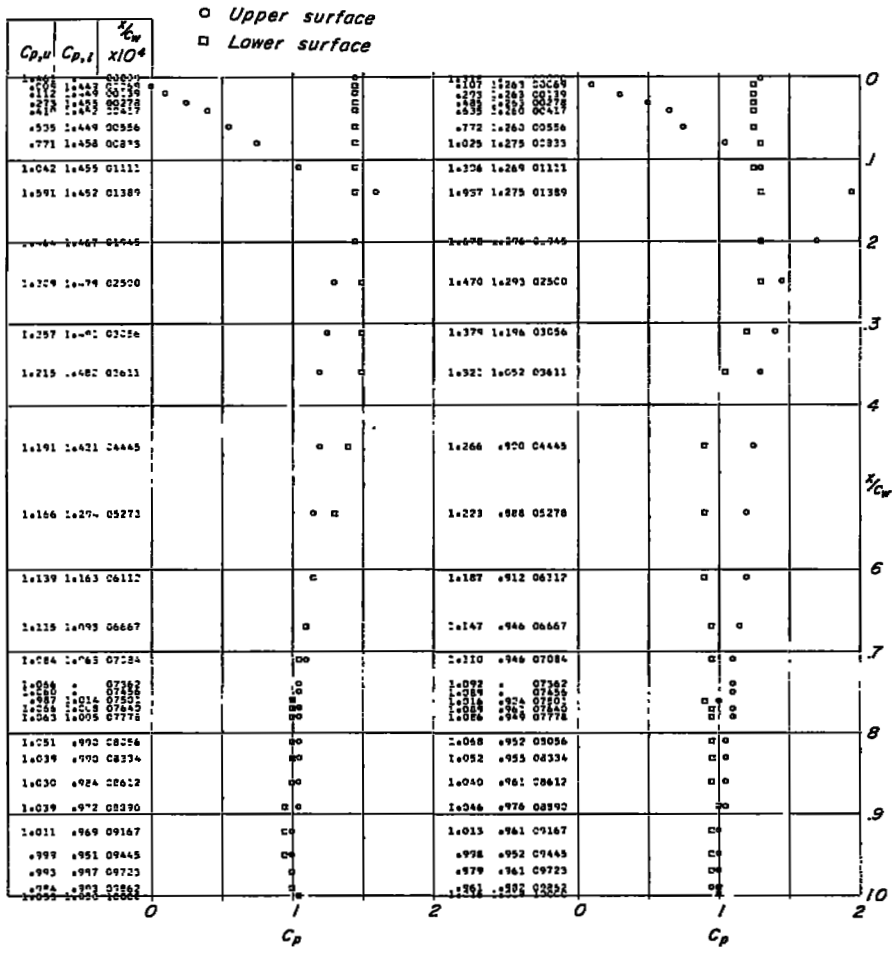
Figure 40.- Concluded.



(a) $\alpha = -8^{\circ}$.

(b) $\alpha = -4^{\circ}$.

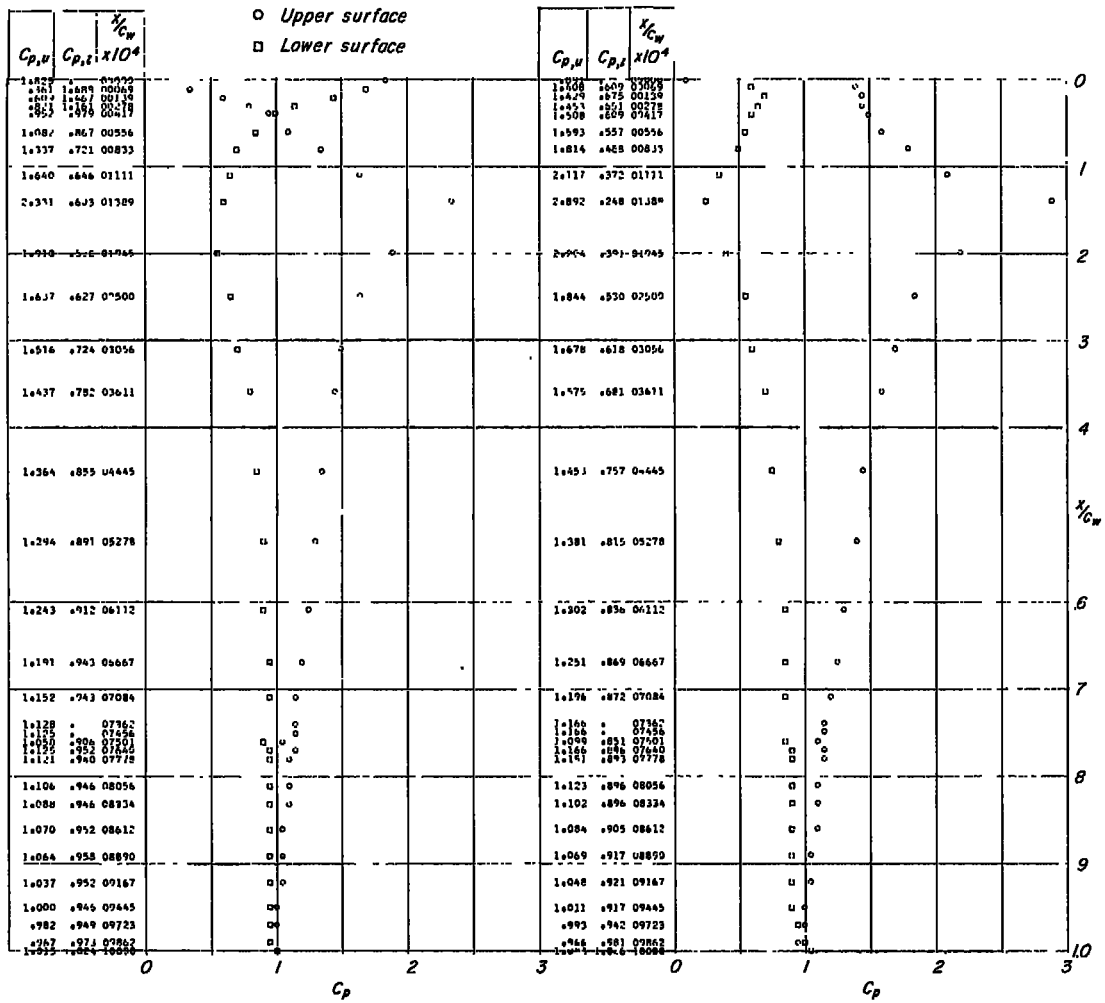
Figure 41.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 30^{\circ}$; $\delta_f = 0^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = 0^{\circ}$.

(d) $\alpha = 4^{\circ}$.

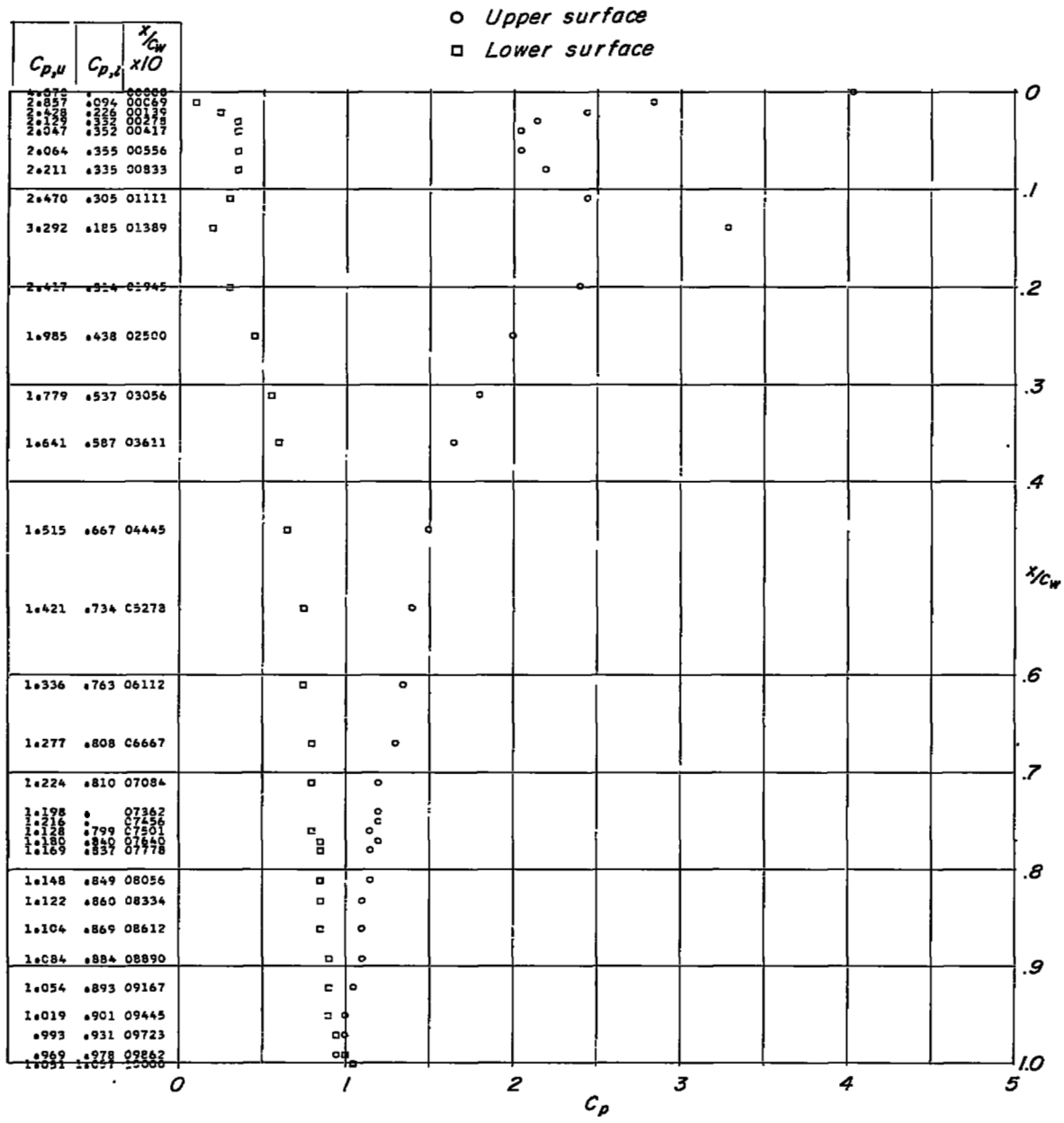
Figure 41.- Continued.



(e) $\alpha = 8^{\circ}$.

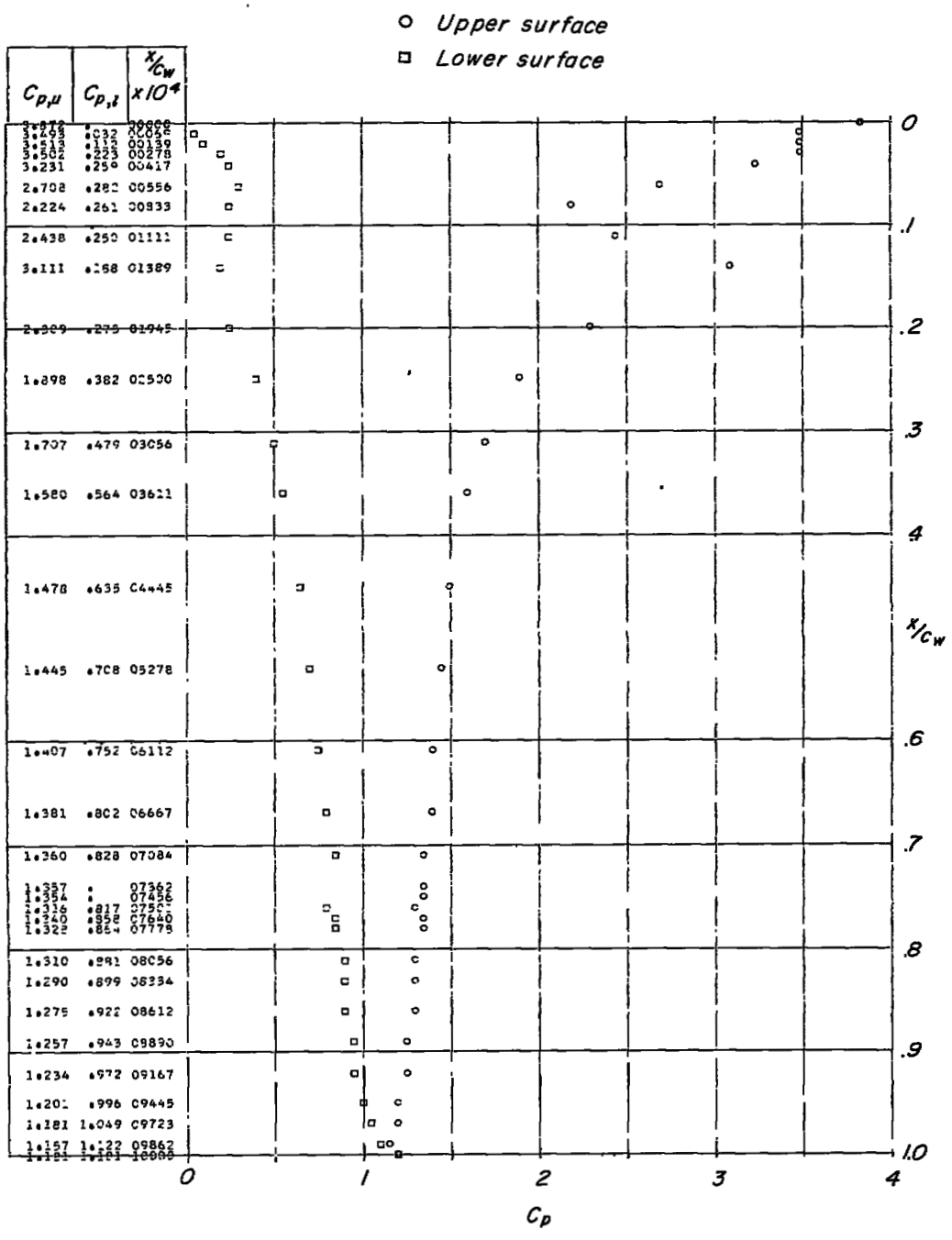
(f) $\alpha = 12^{\circ}$.

Figure 41.- Continued.



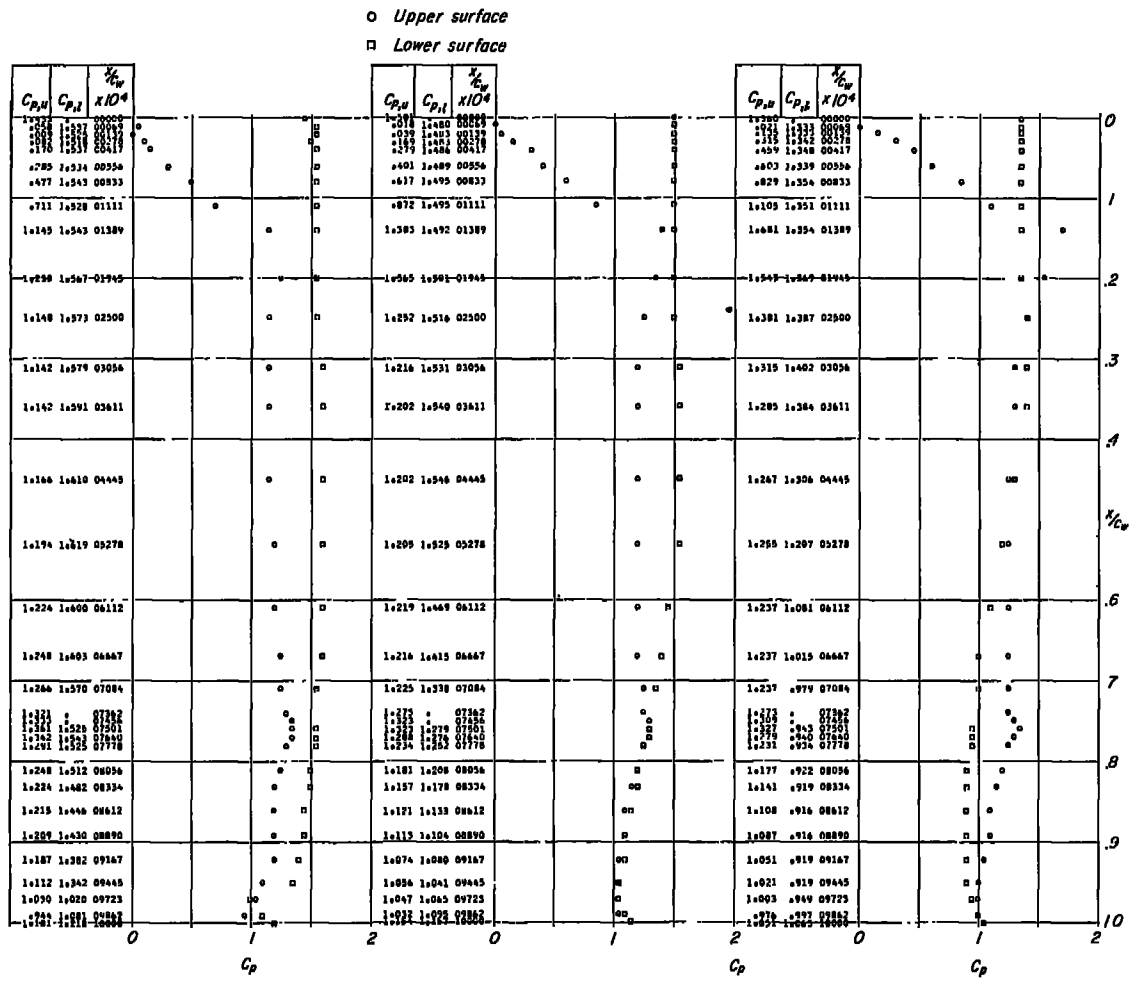
(g) $\alpha = 16^{\circ}$.

Figure 41.- Continued.



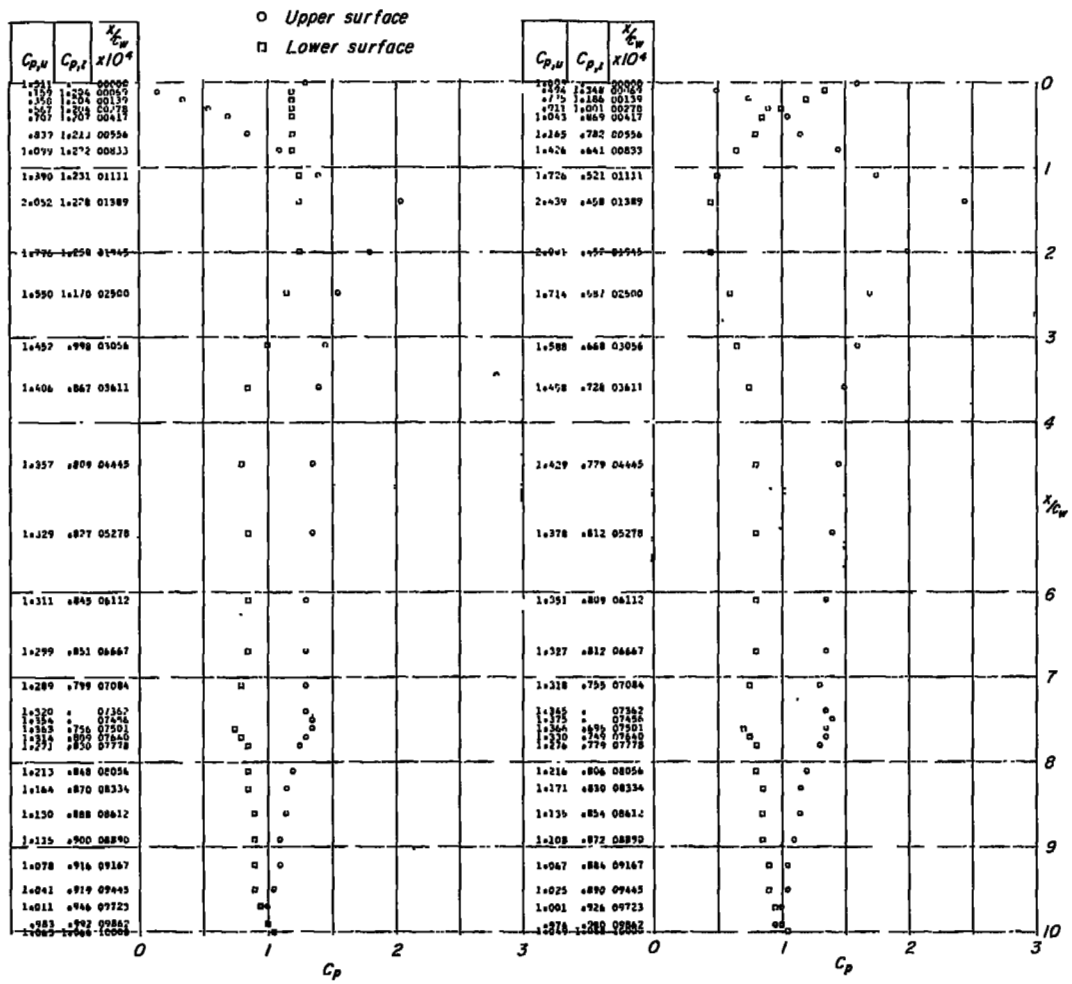
(h) $\alpha = 20^{\circ}$.

Figure 41.- Concluded.



(a) $\alpha = -8^{\circ}$. (b) $\alpha = -4^{\circ}$. (c) $\alpha = 0^{\circ}$.

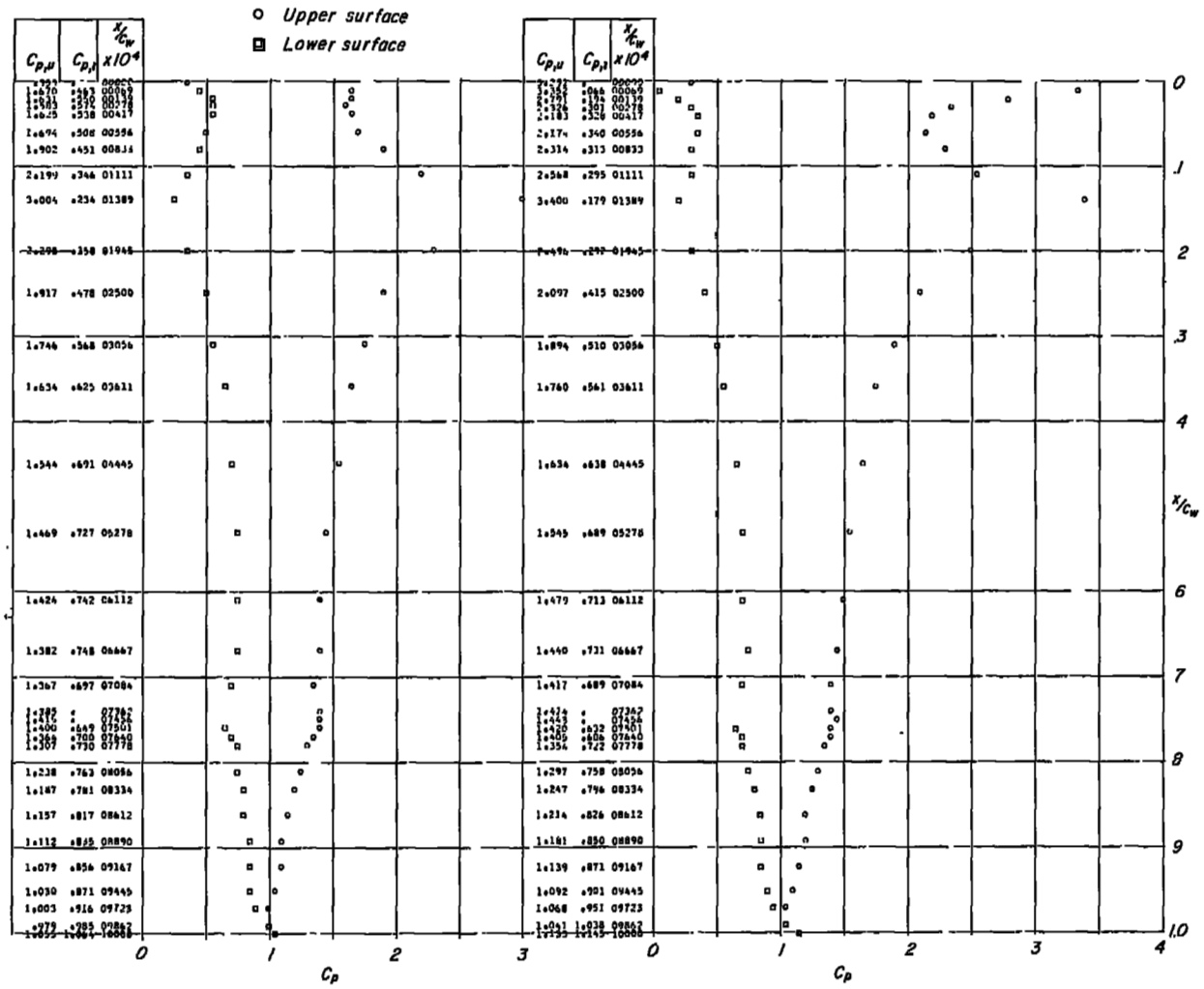
Figure 42.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 30^{\circ}$; $\delta_f = 5^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(d) $\alpha = 4^{\circ}$.

(e) $\alpha = 8^{\circ}$.

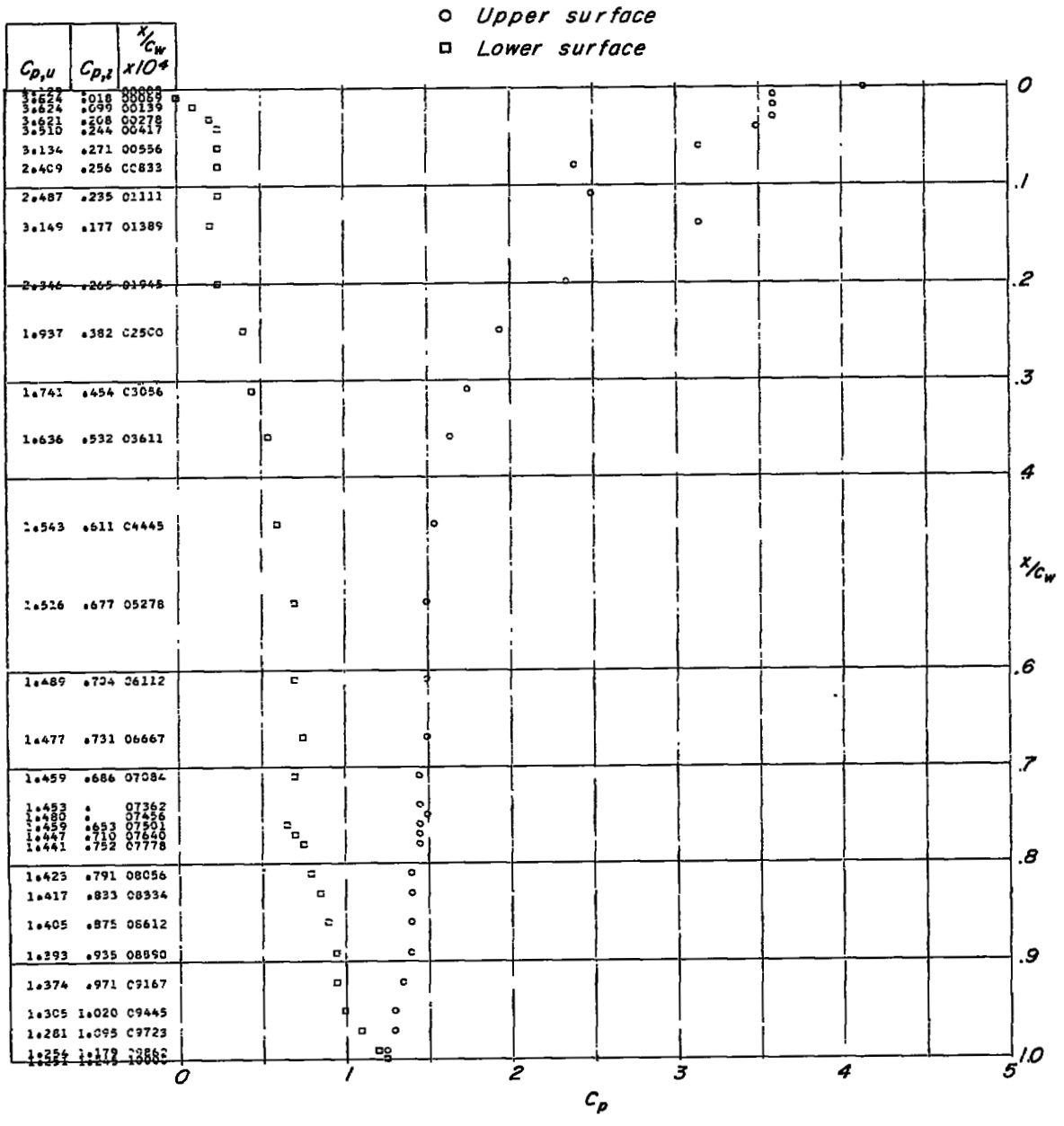
Figure 42.- Continued.



(f) $\alpha = 12^{\circ}$.

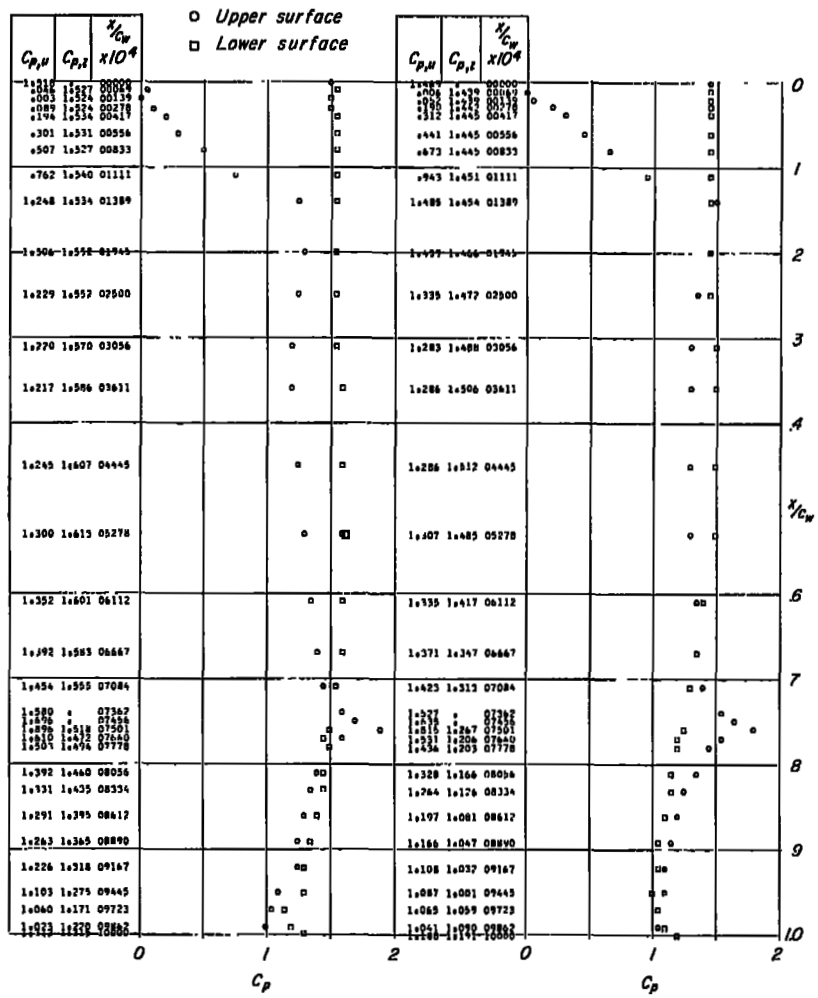
(g) $\alpha = 16^{\circ}$.

Figure 42.- Continued.



(h) $\alpha = 20^{\circ}$.

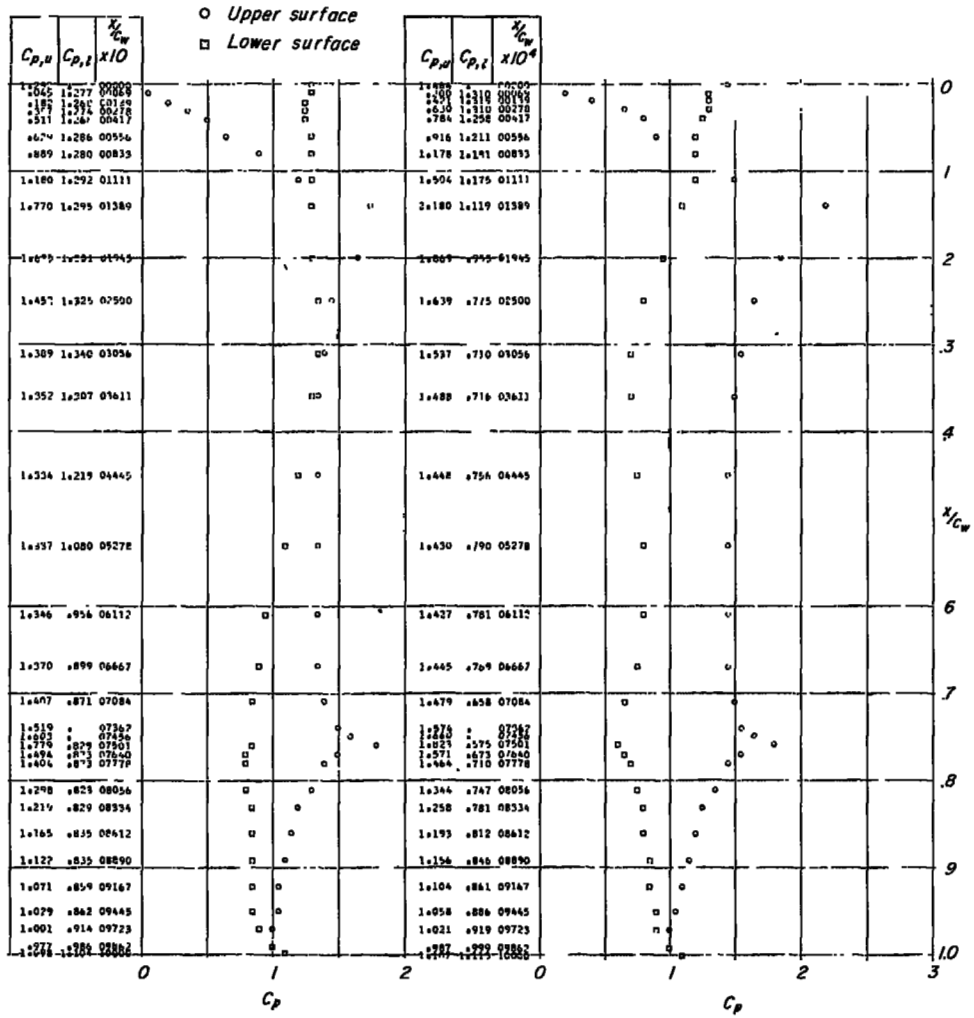
Figure 42.- Concluded.



(a) $\alpha = -8^{\circ}$.

(b) $\alpha = -4^{\circ}$.

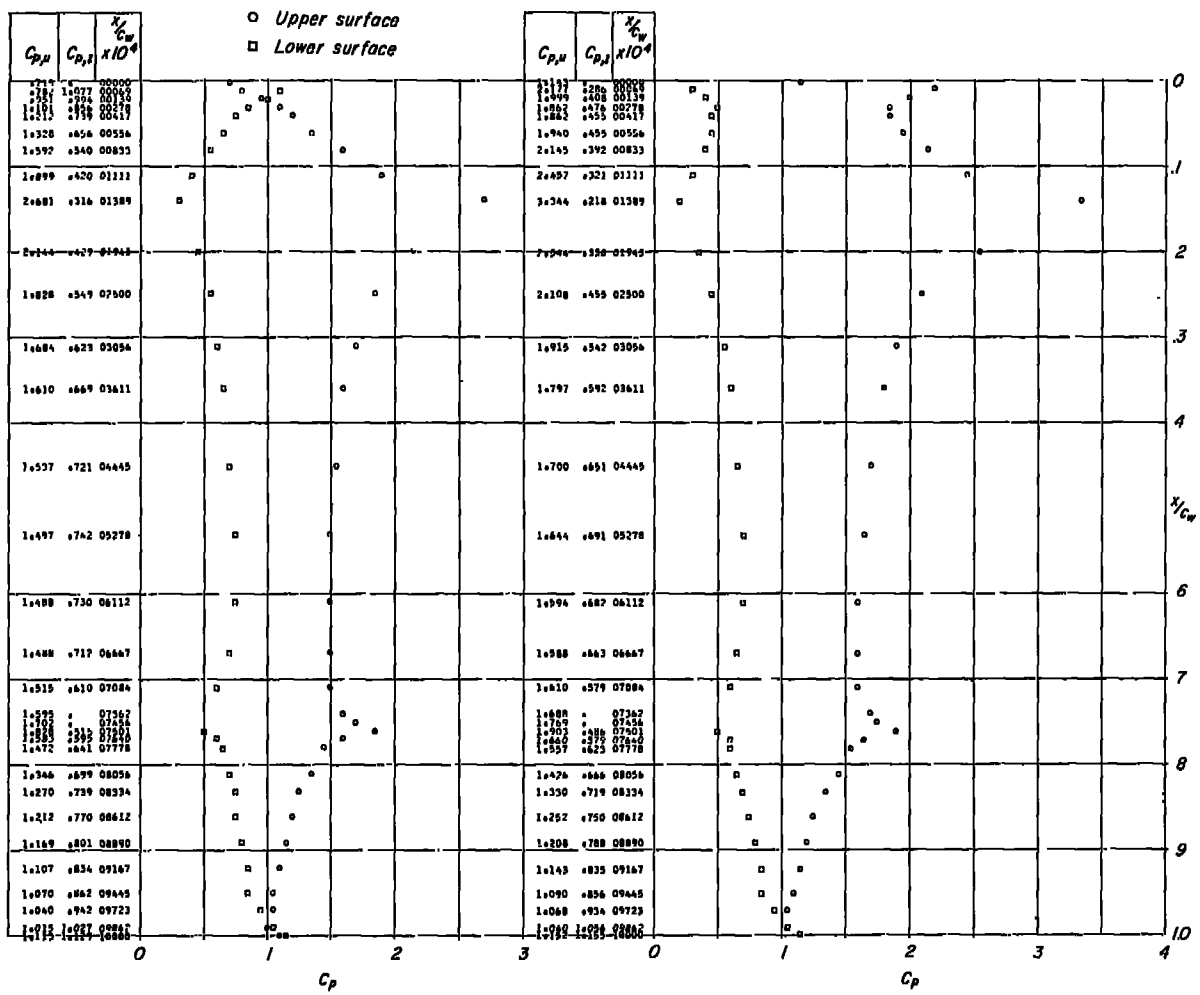
Figure 43.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 30^{\circ}$; $\delta_f = 10^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = 0^{\circ}$.

(d) $\alpha = 4^{\circ}$.

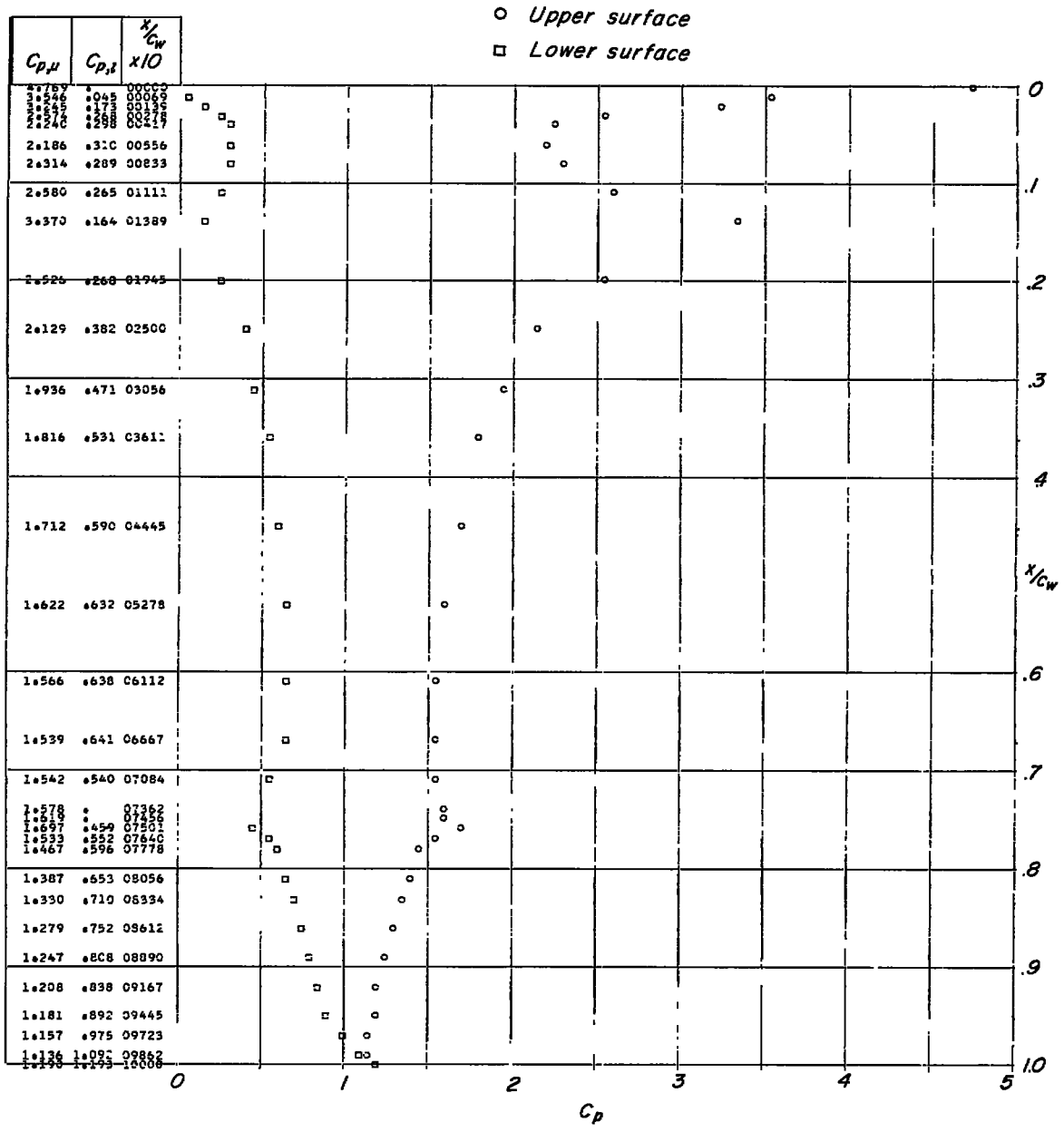
Figure 43.- Continued.



(e) $\alpha = 8^{\circ}$.

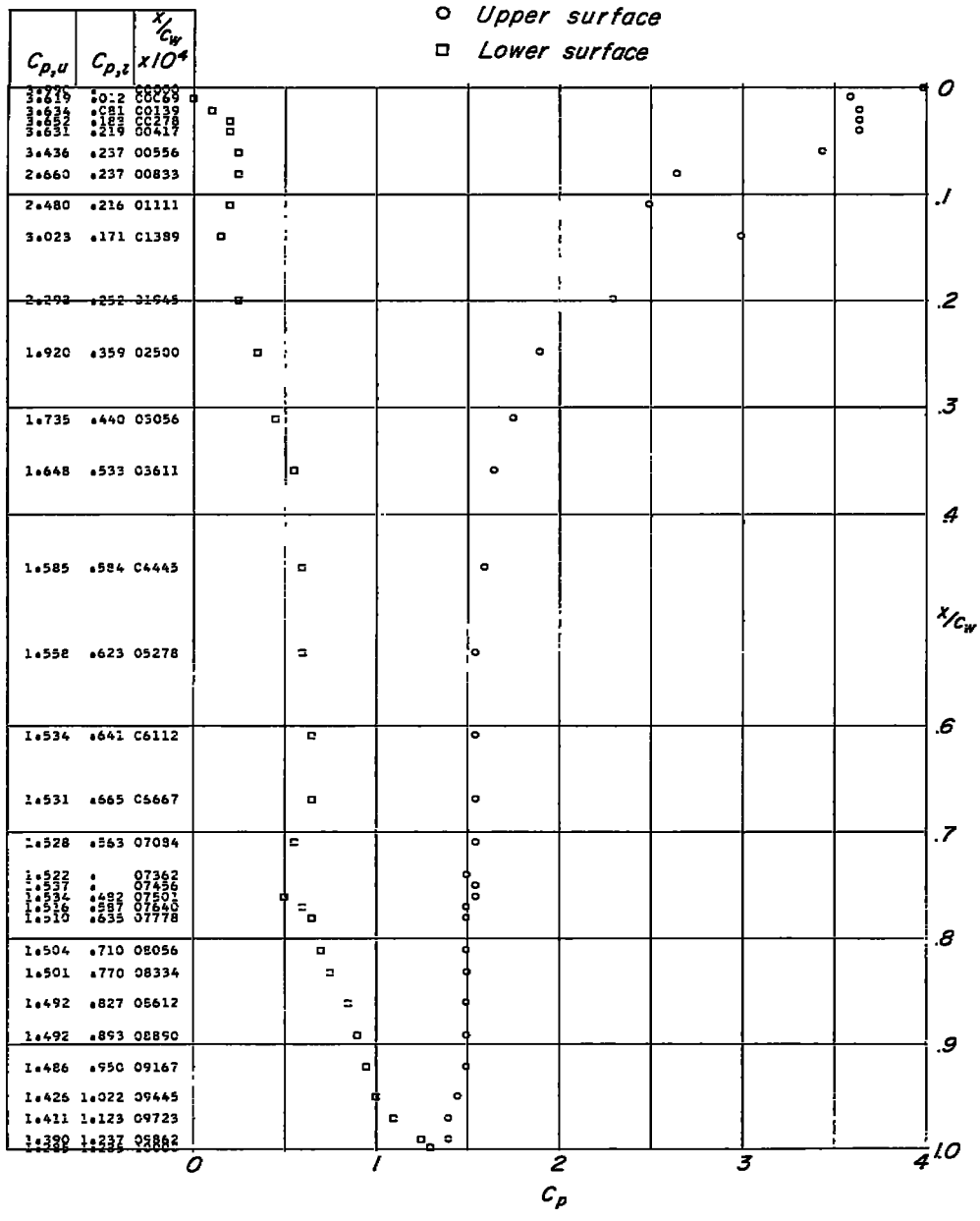
(f) $\alpha = 12^{\circ}$.

Figure 43.- Continued.



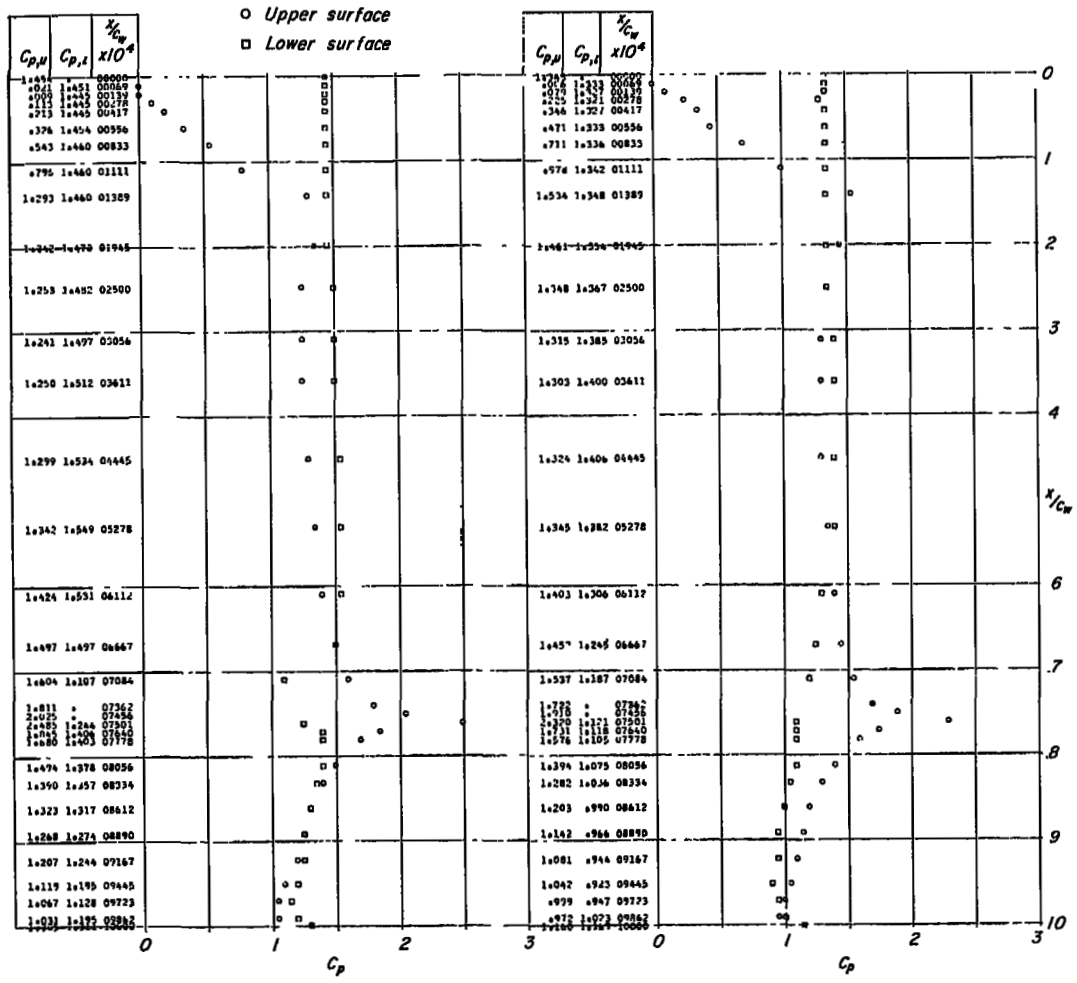
(g) $\alpha = 16^{\circ}$.

Figure 43.- Continued.



(h) $\alpha = 20^{\circ}$.

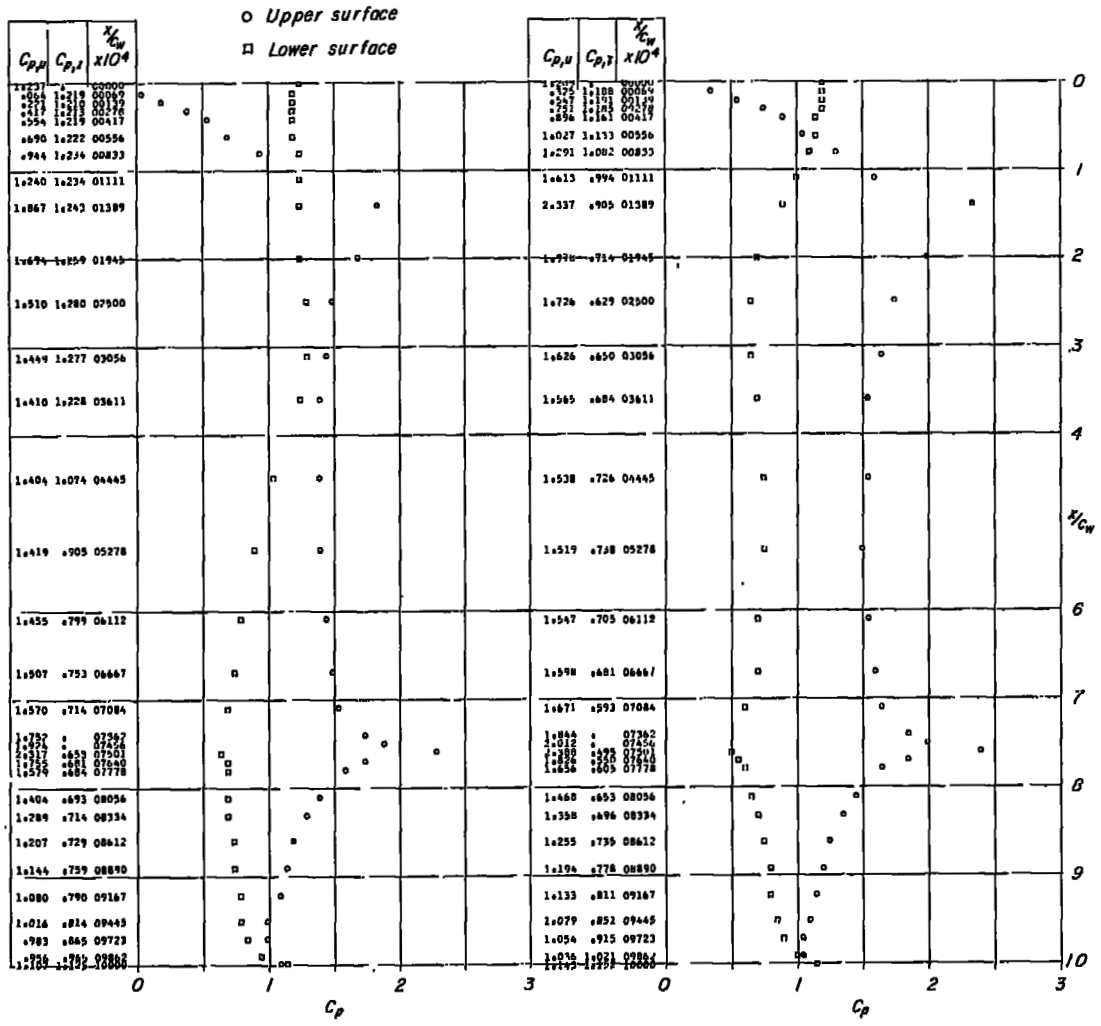
Figure 43.- Concluded.



(a) $\alpha = -8^{\circ}$.

(b) $\alpha = -4^{\circ}$.

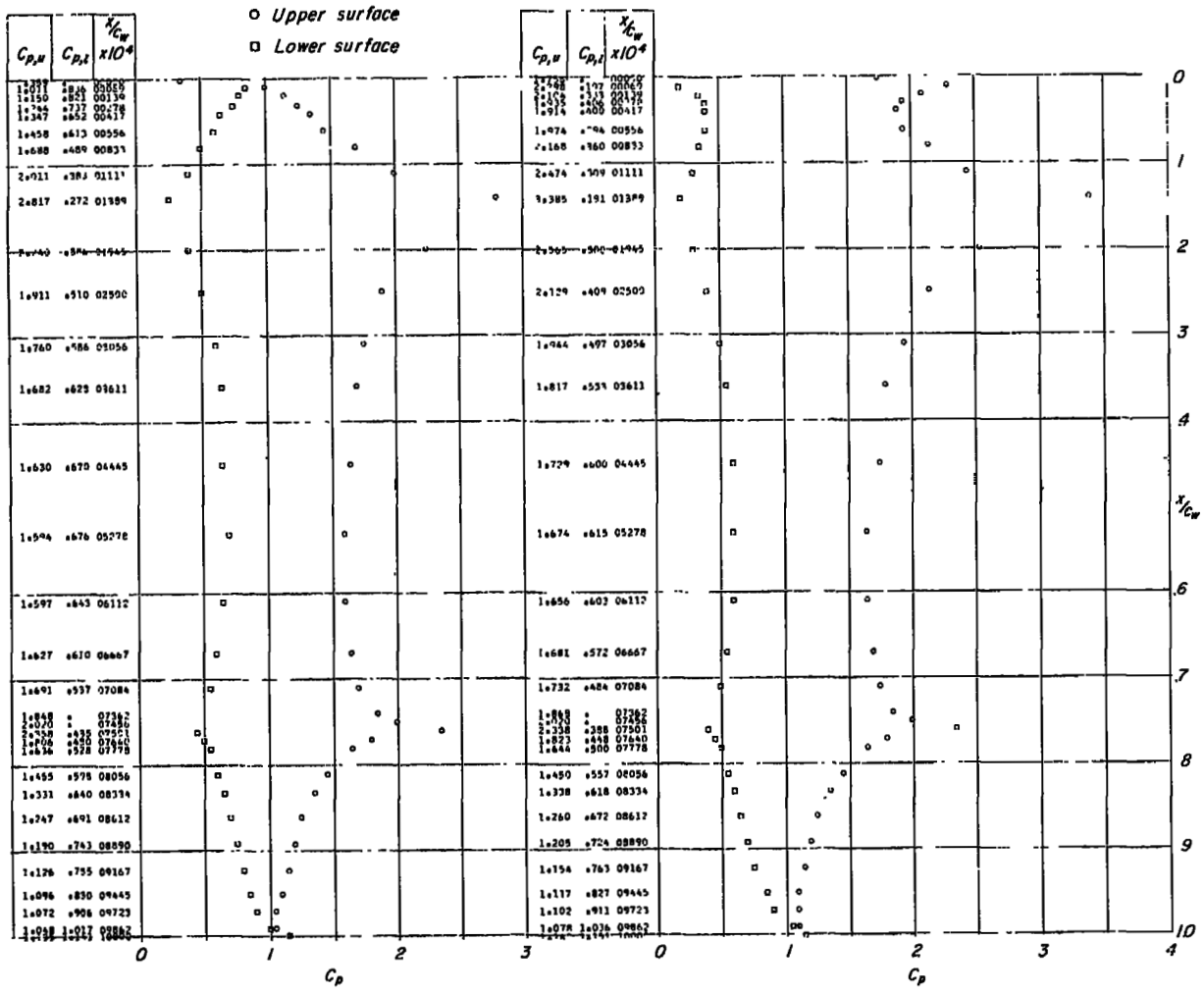
Figure 44.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 30^{\circ}$; $\delta_f = 15^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = 0^{\circ}$.

(d) $\alpha = 4^{\circ}$.

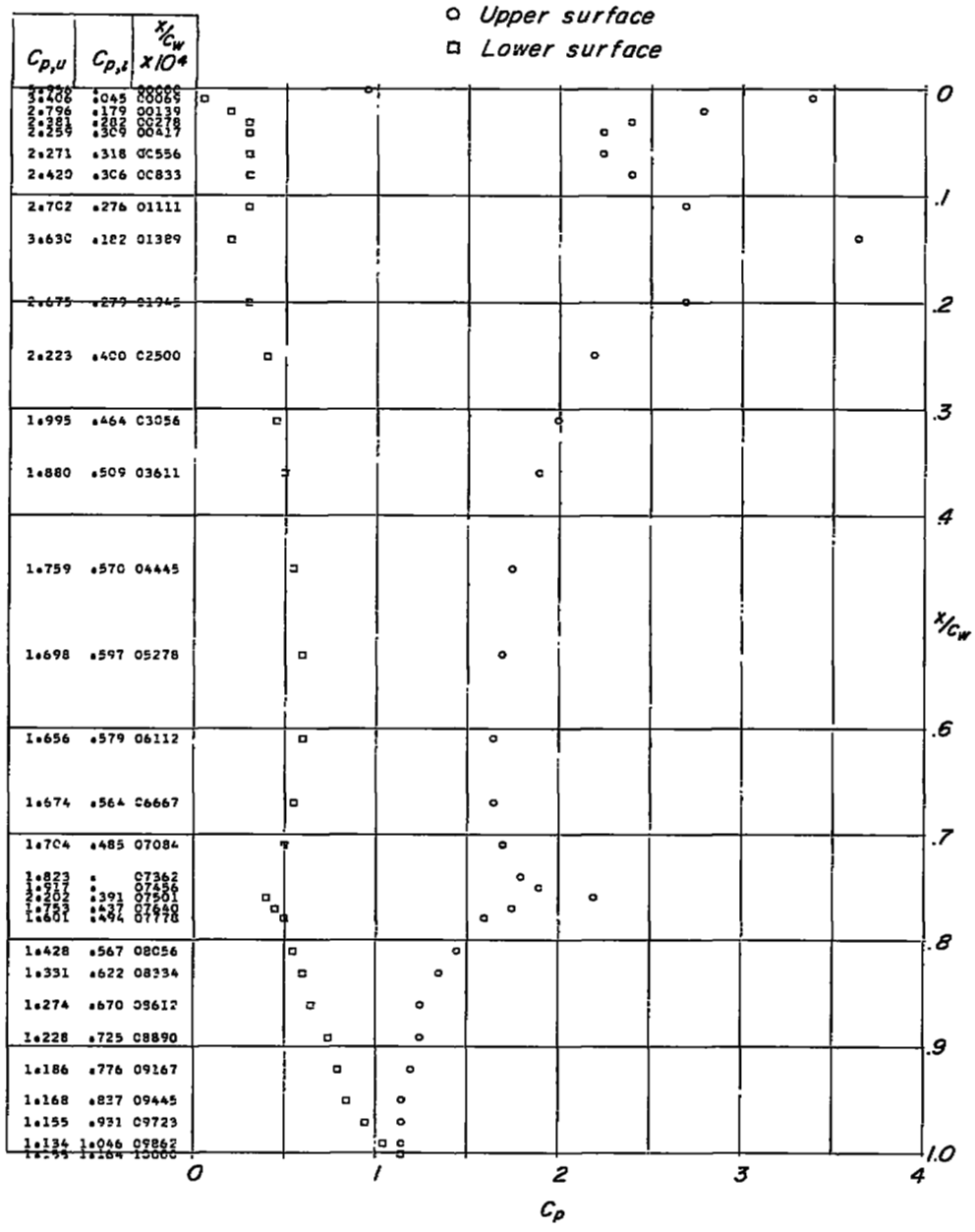
Figure 44.- Continued.



(e) $\alpha = 8^{\circ}$.

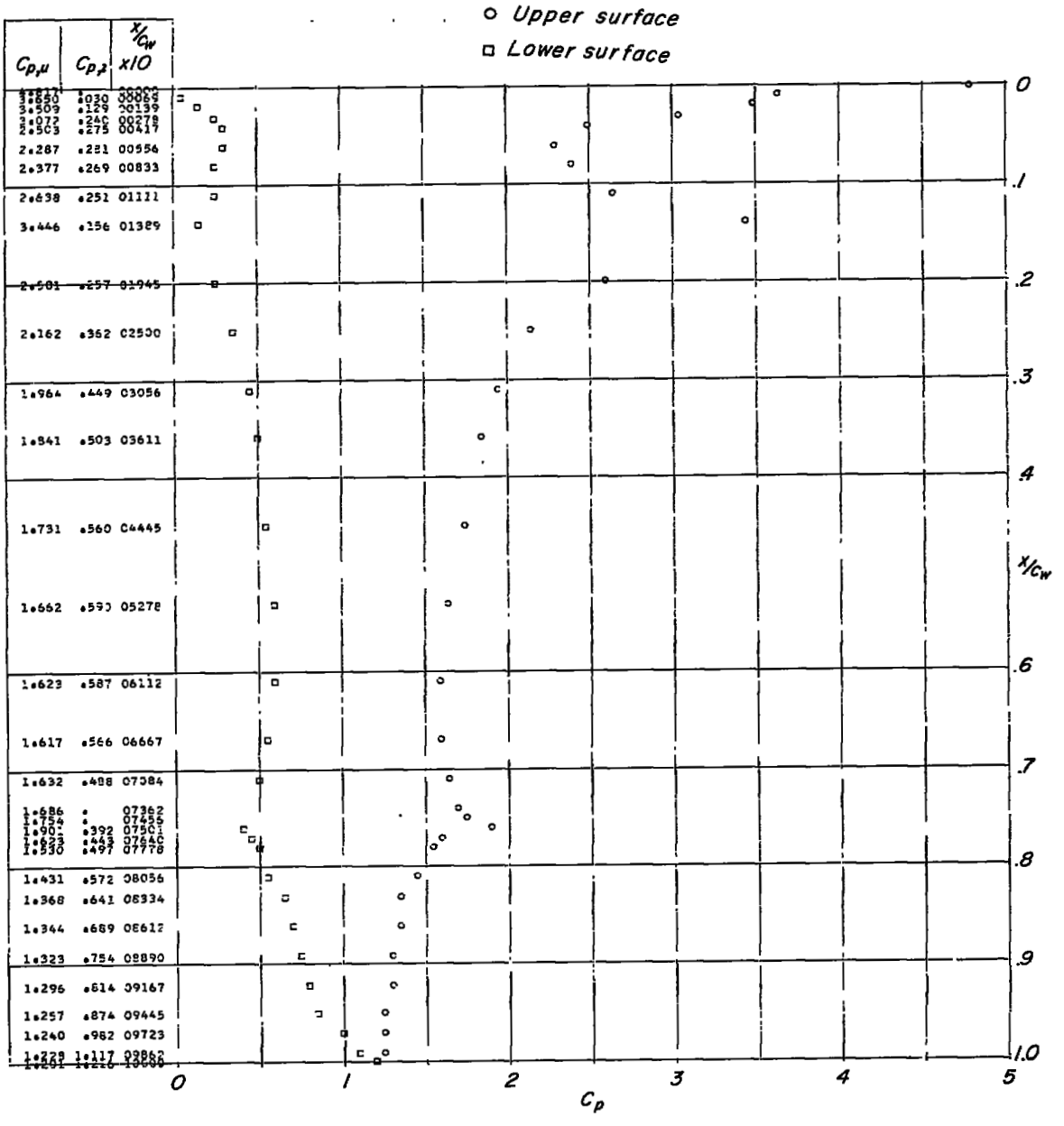
(f) $\alpha = 12^{\circ}$.

Figure 44.- Continued.



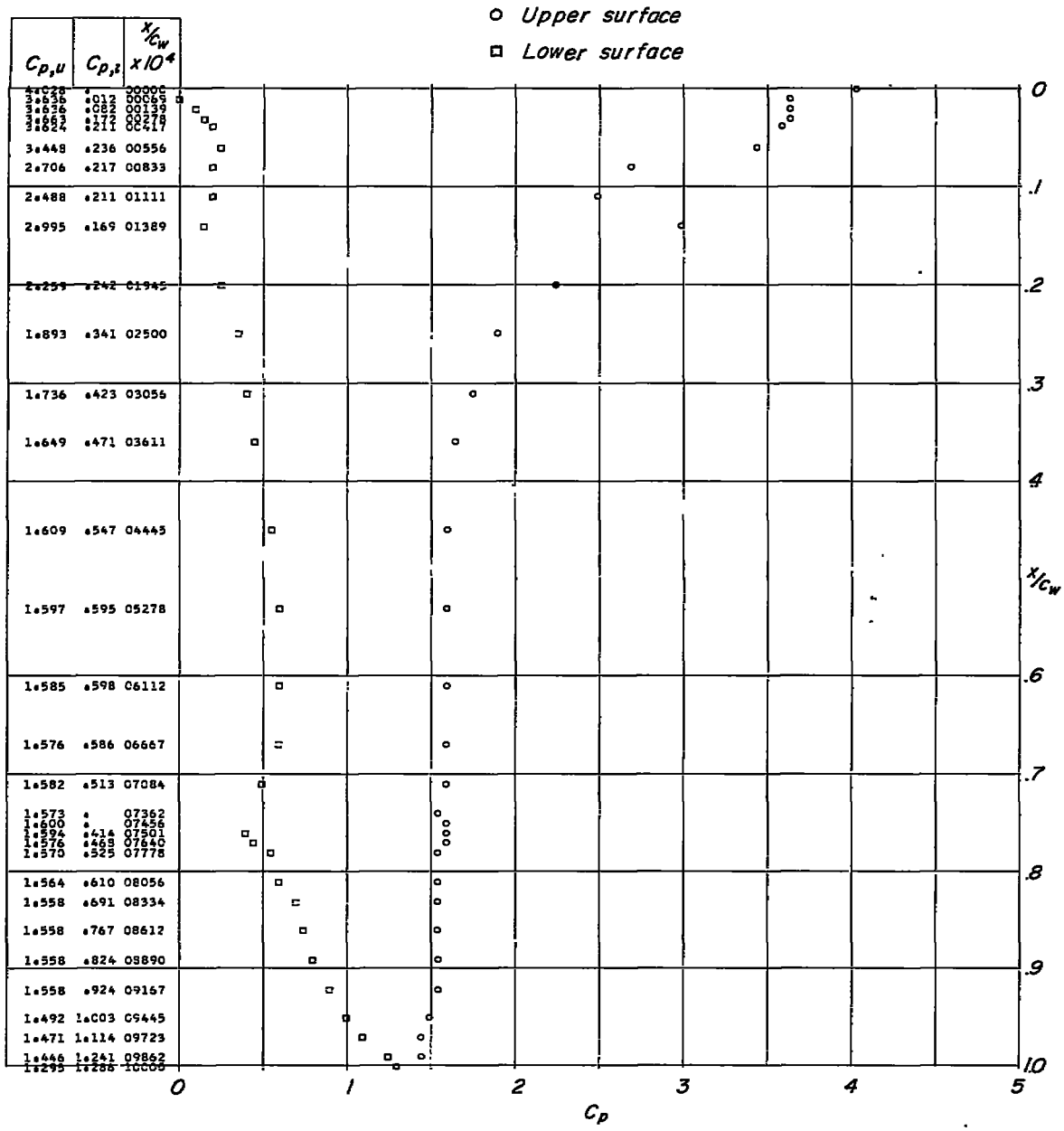
(g) $\alpha = 14^{\circ}$.

Figure 44.- Continued.



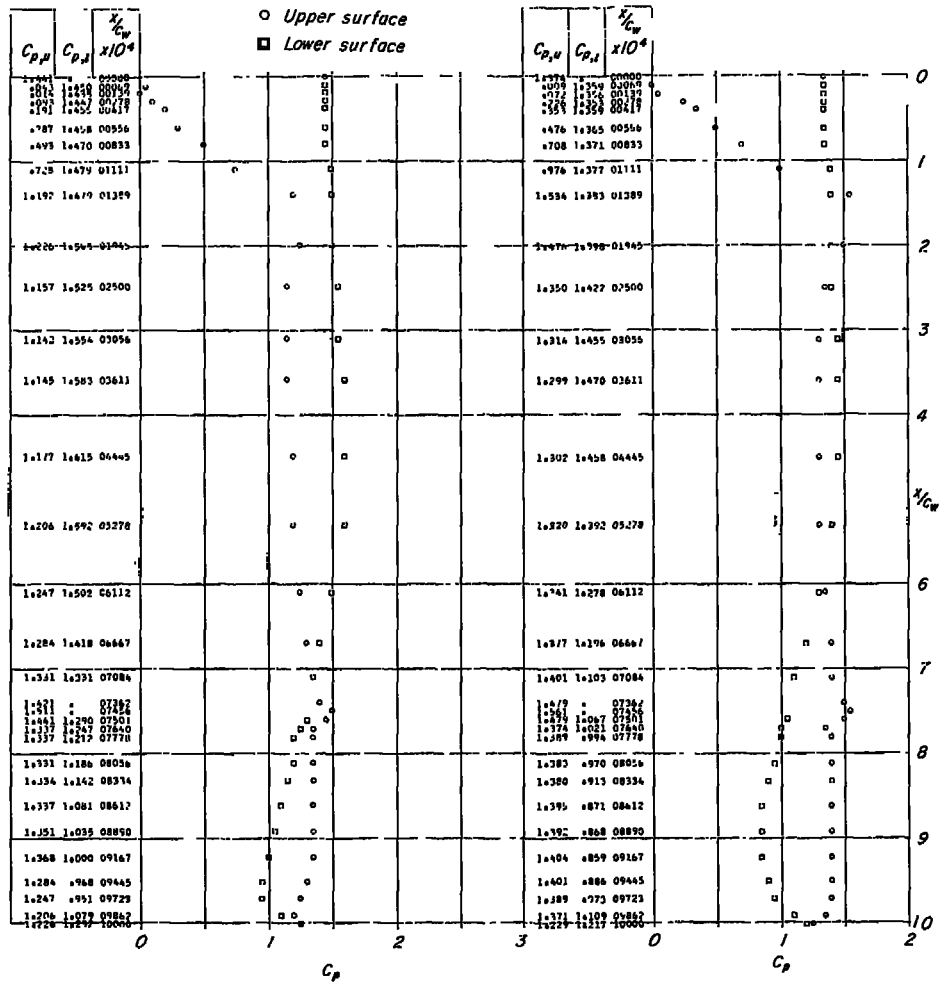
(h) $\alpha = 16^{\circ}$.

Figure 44.- Continued.



(i) $\alpha = 20^{\circ}$.

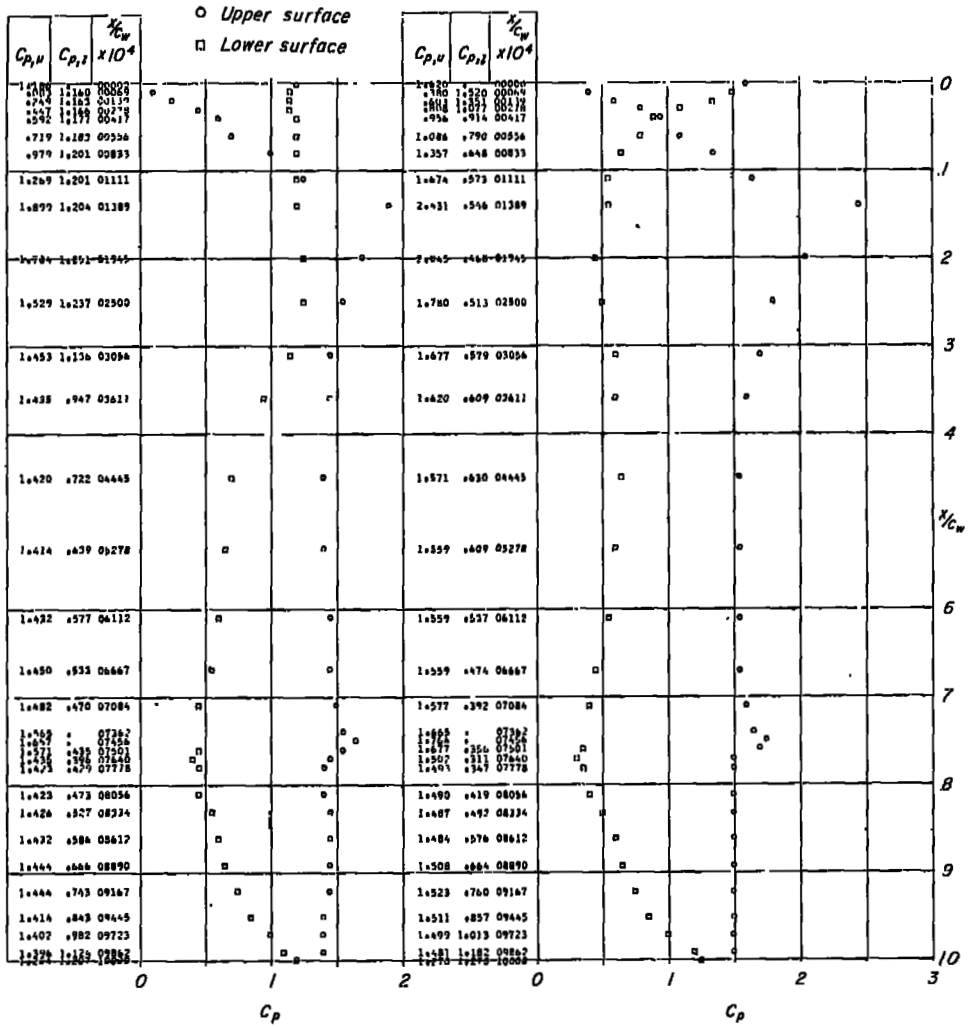
Figure 44.- Concluded.



(a) $\alpha = -8^{\circ}$.

(b) $\alpha = -4^{\circ}$.

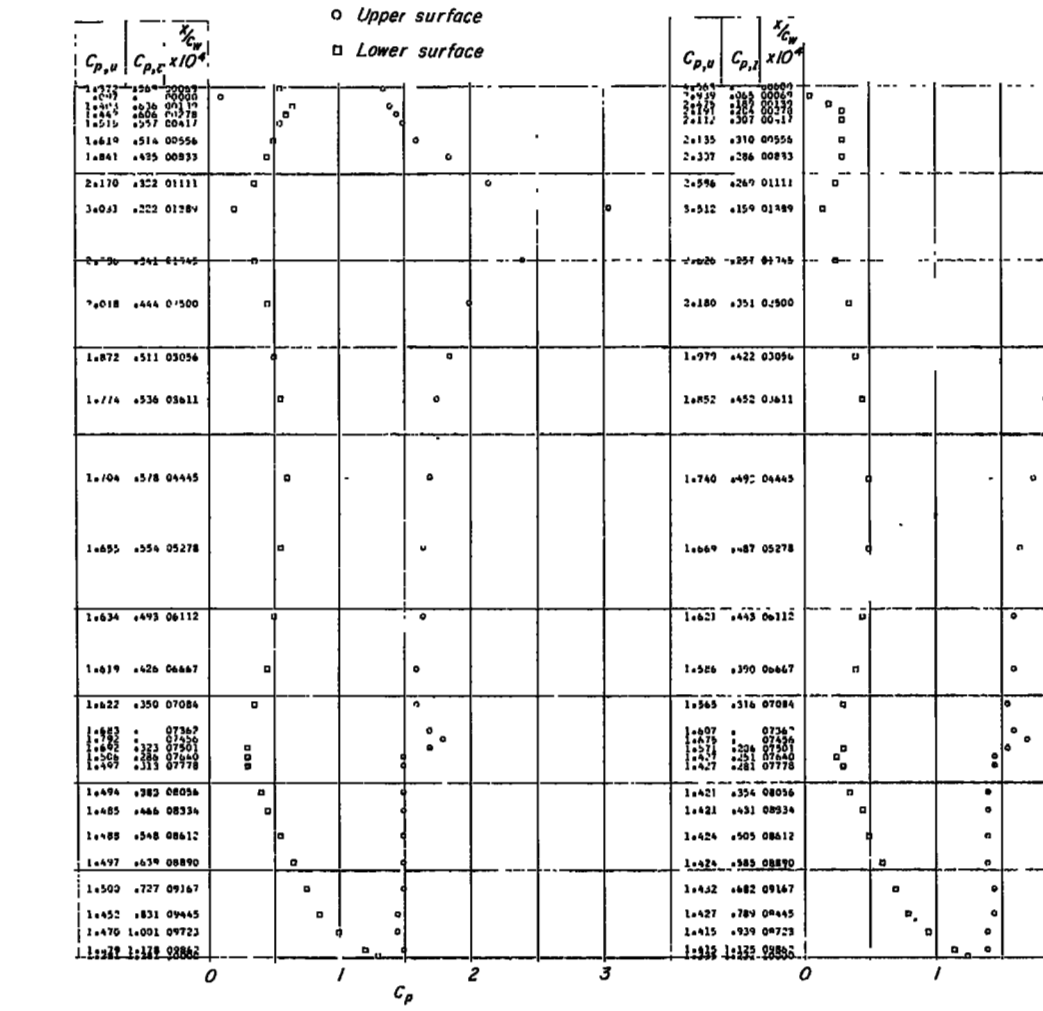
Figure 45.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 30^{\circ}$; $\delta_f = 30^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = 0^{\circ}$.

(d) $\alpha = 4^{\circ}$.

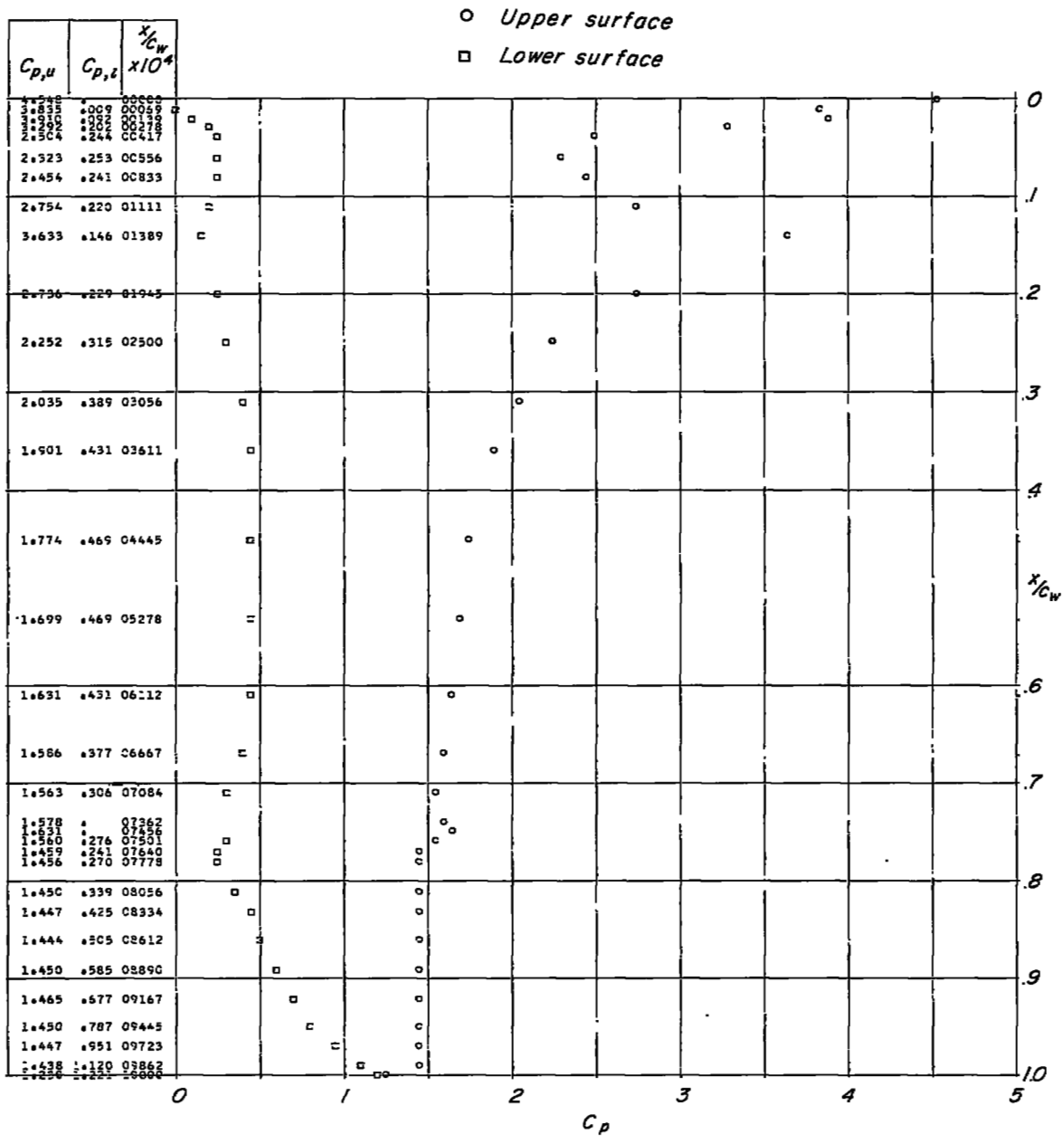
Figure 45.- Continued.



(e) $\alpha = 8^{\circ}$.

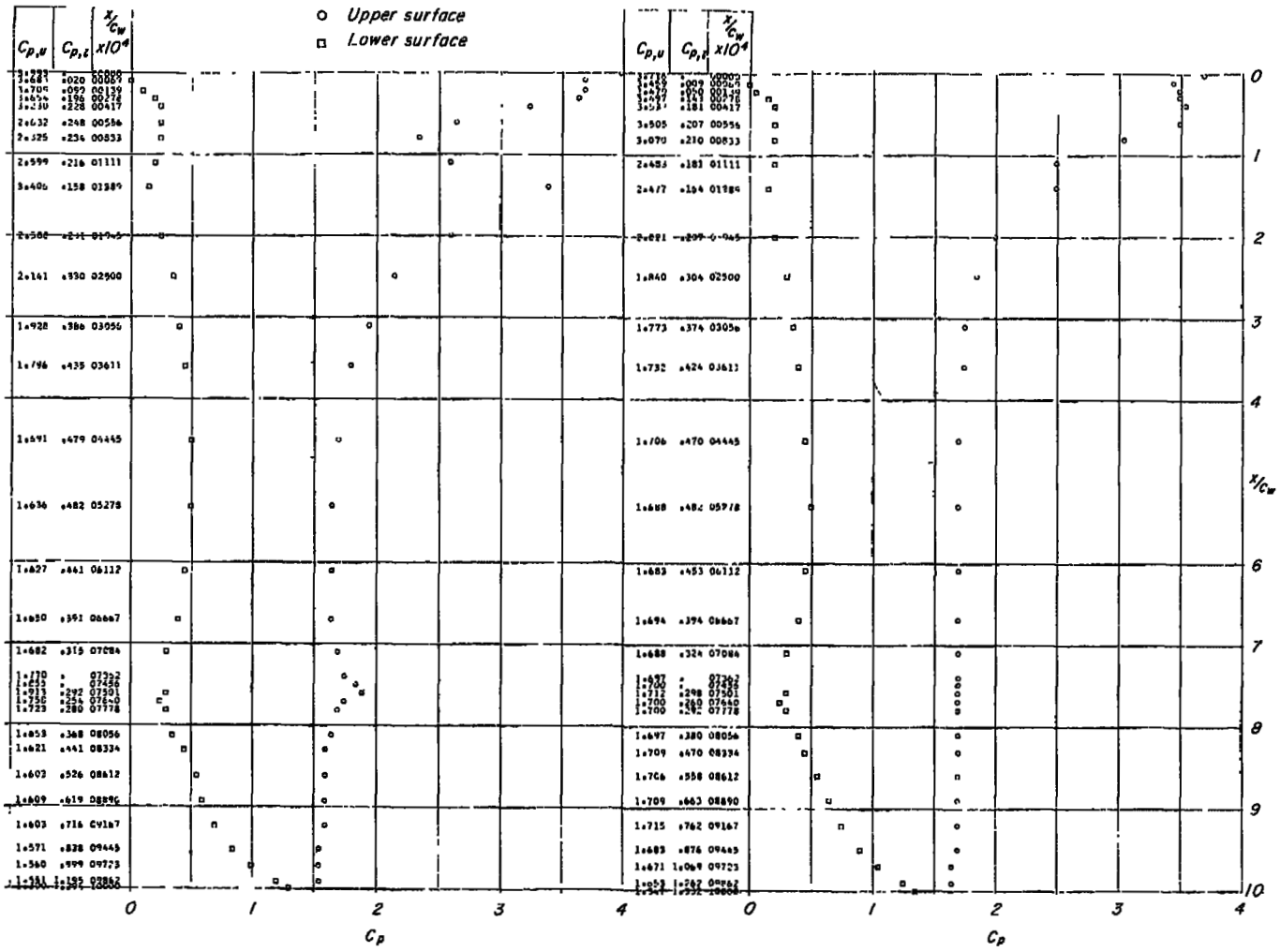
(f) $\alpha = 12^{\circ}$.

Figure 45.- Continued.



(g) $\alpha = 14^{\circ}$.

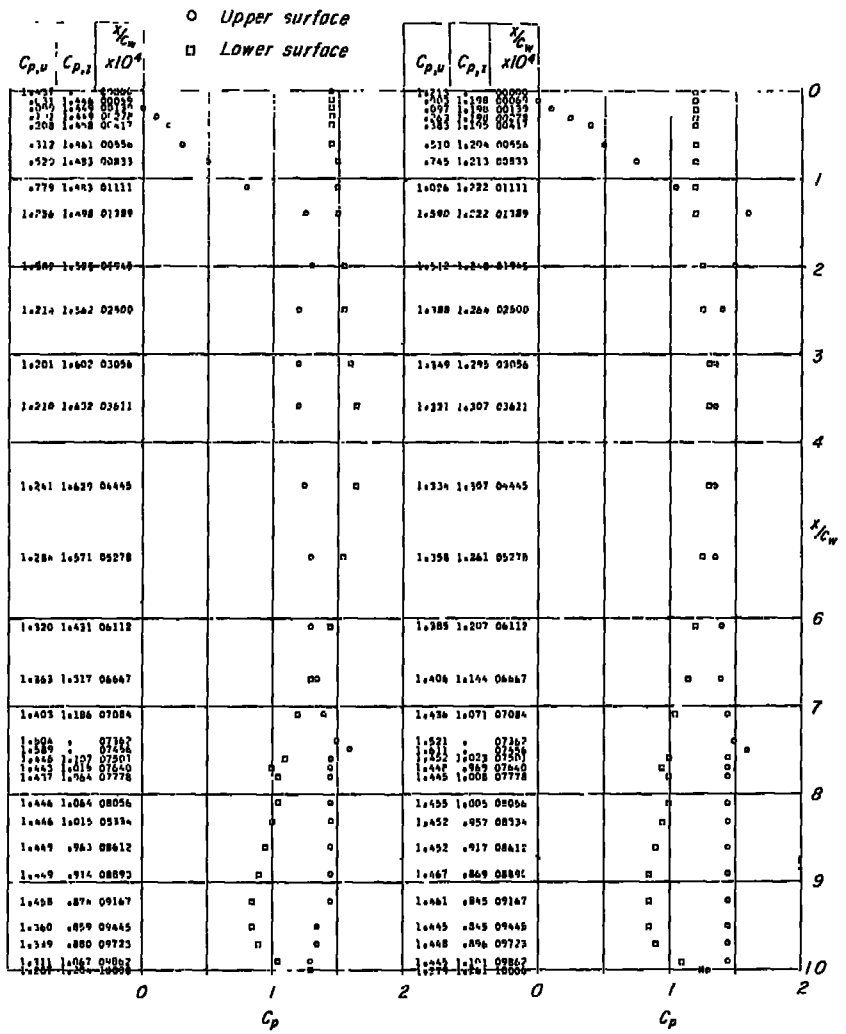
Figure 45.- Continued.



(h) $\alpha = 16^{\circ}$.

(i) $\alpha = 20^{\circ}$.

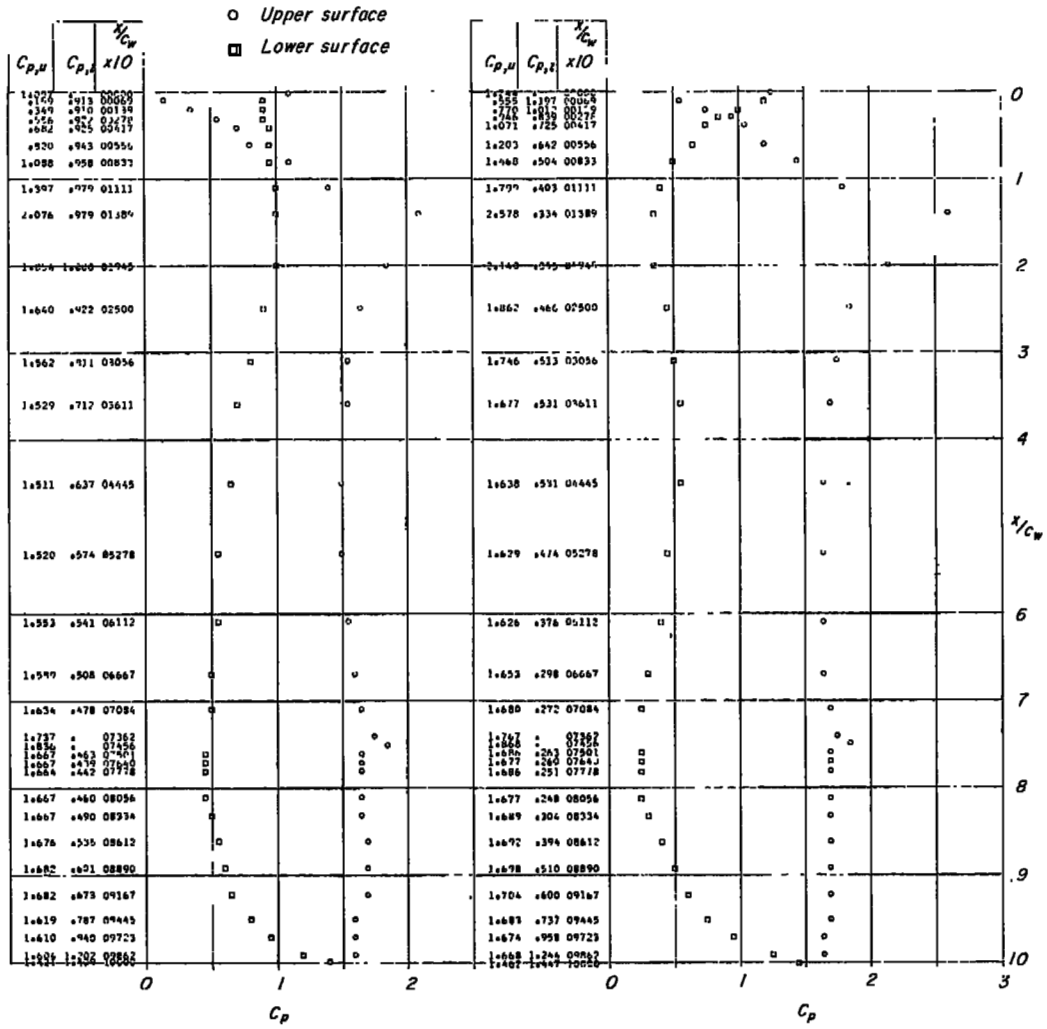
Figure 45.- Concluded.



(a) $\alpha = -8^{\circ}$.

(b) $\alpha = -4^{\circ}$.

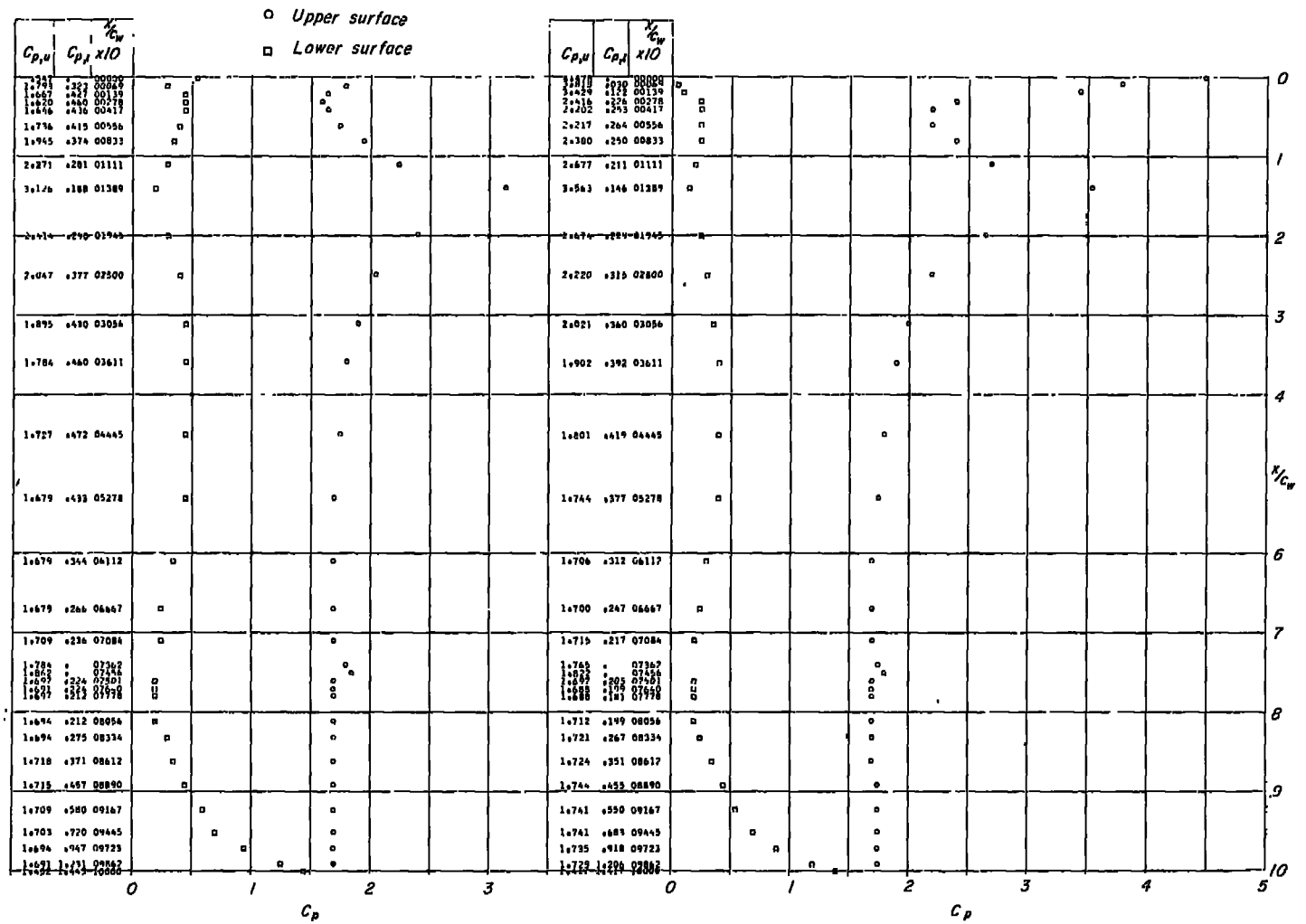
Figure 46.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 30^{\circ}$; $\delta_f = 45^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = 0^{\circ}$.

(d) $\alpha = 4^{\circ}$.

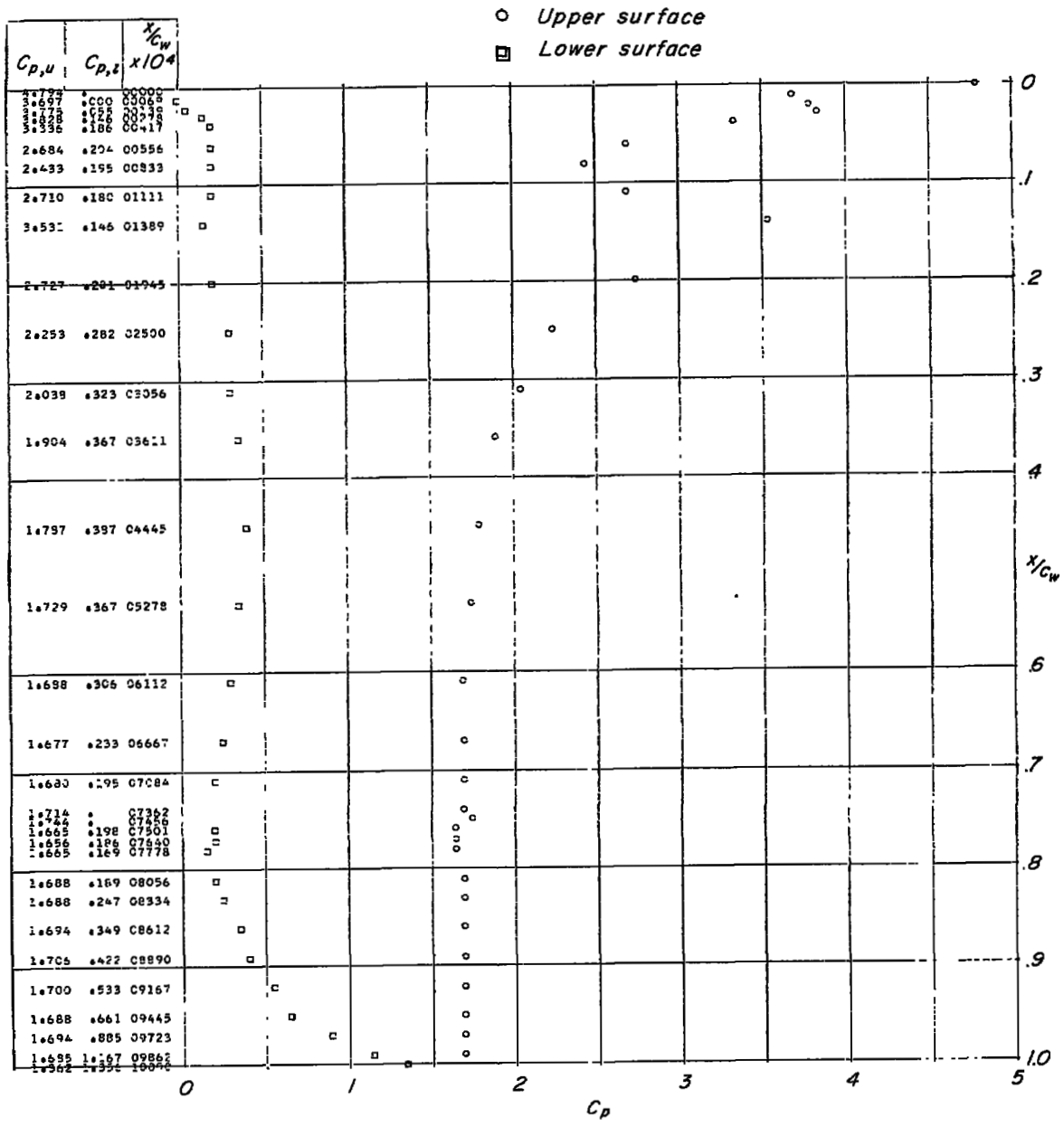
Figure 46.- Continued.



(e) $\alpha = 8^{\circ}$.

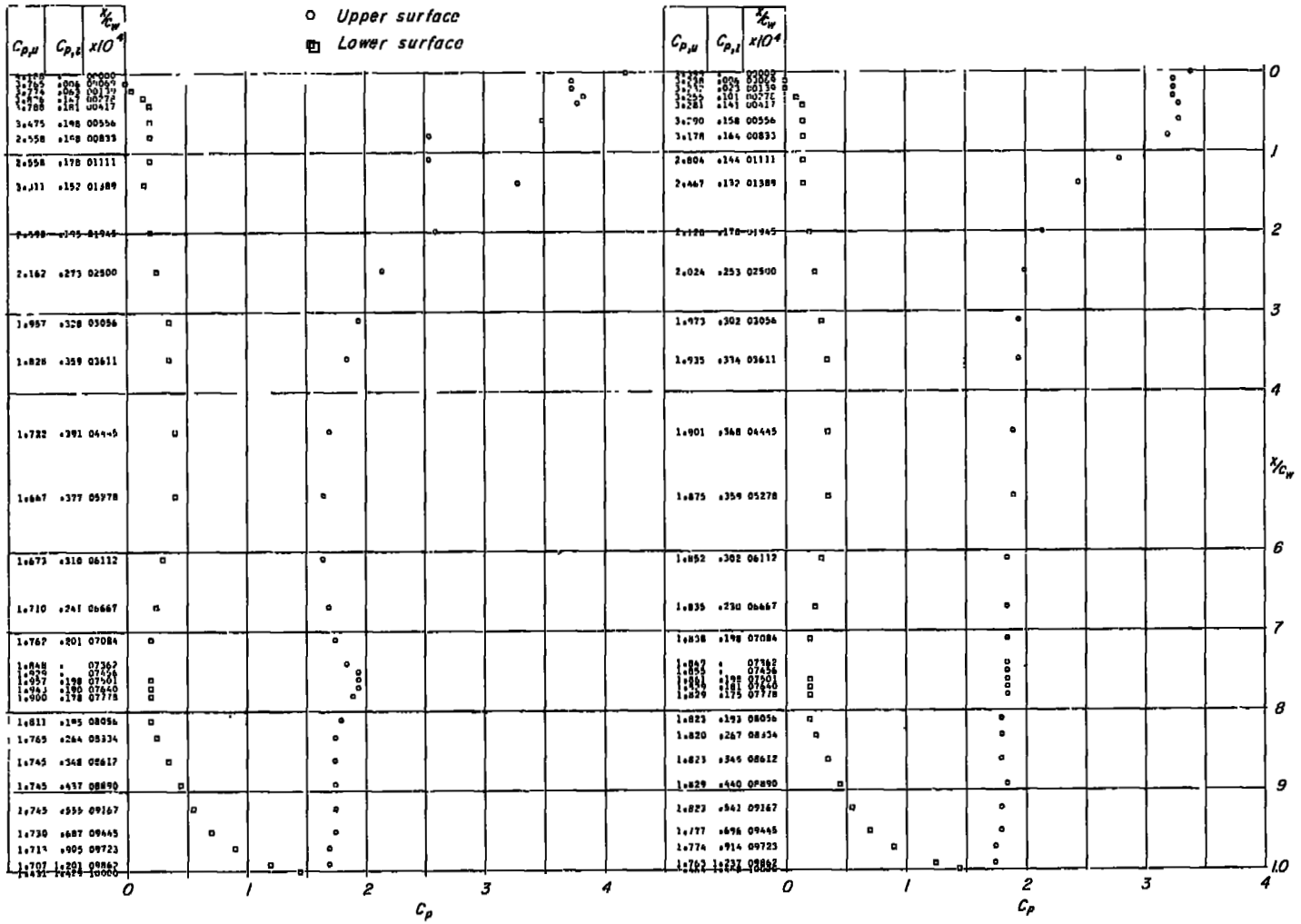
(f) $\alpha = 12^{\circ}$.

Figure 46.- Continued.



(g) $\alpha = 14^{\circ}$.

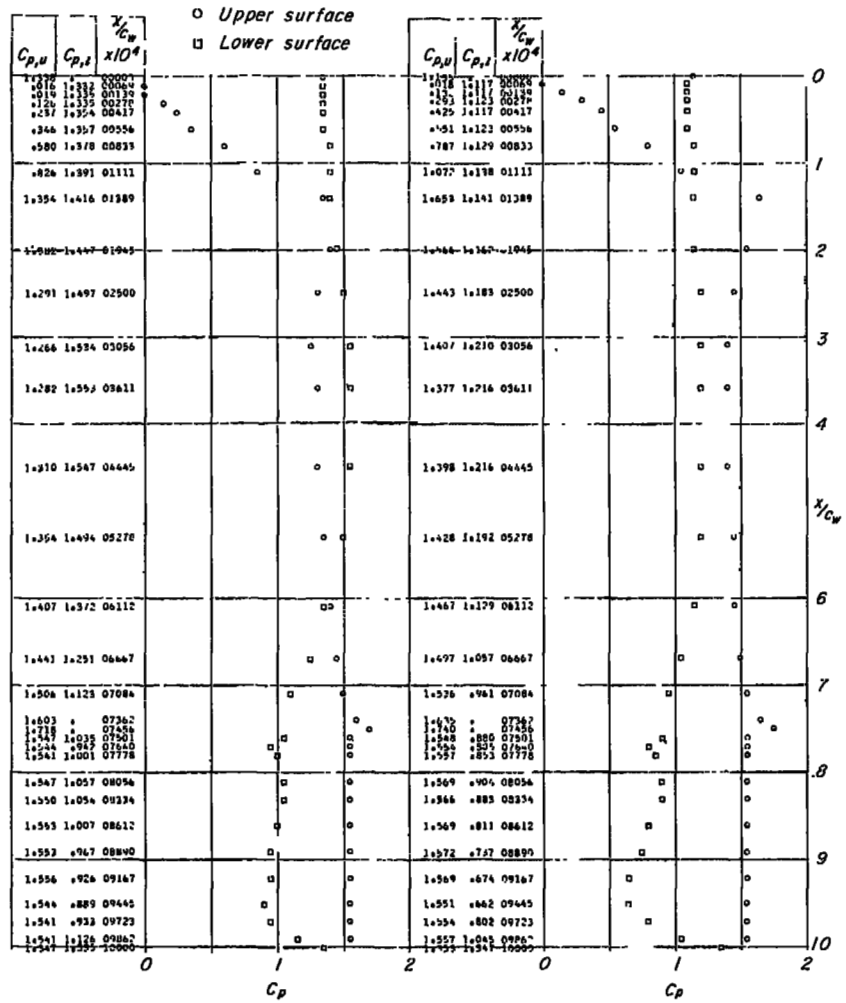
Figure 46.- Continued.



(h) $\alpha = 16^{\circ}$.

(i) $\alpha = 20^{\circ}$.

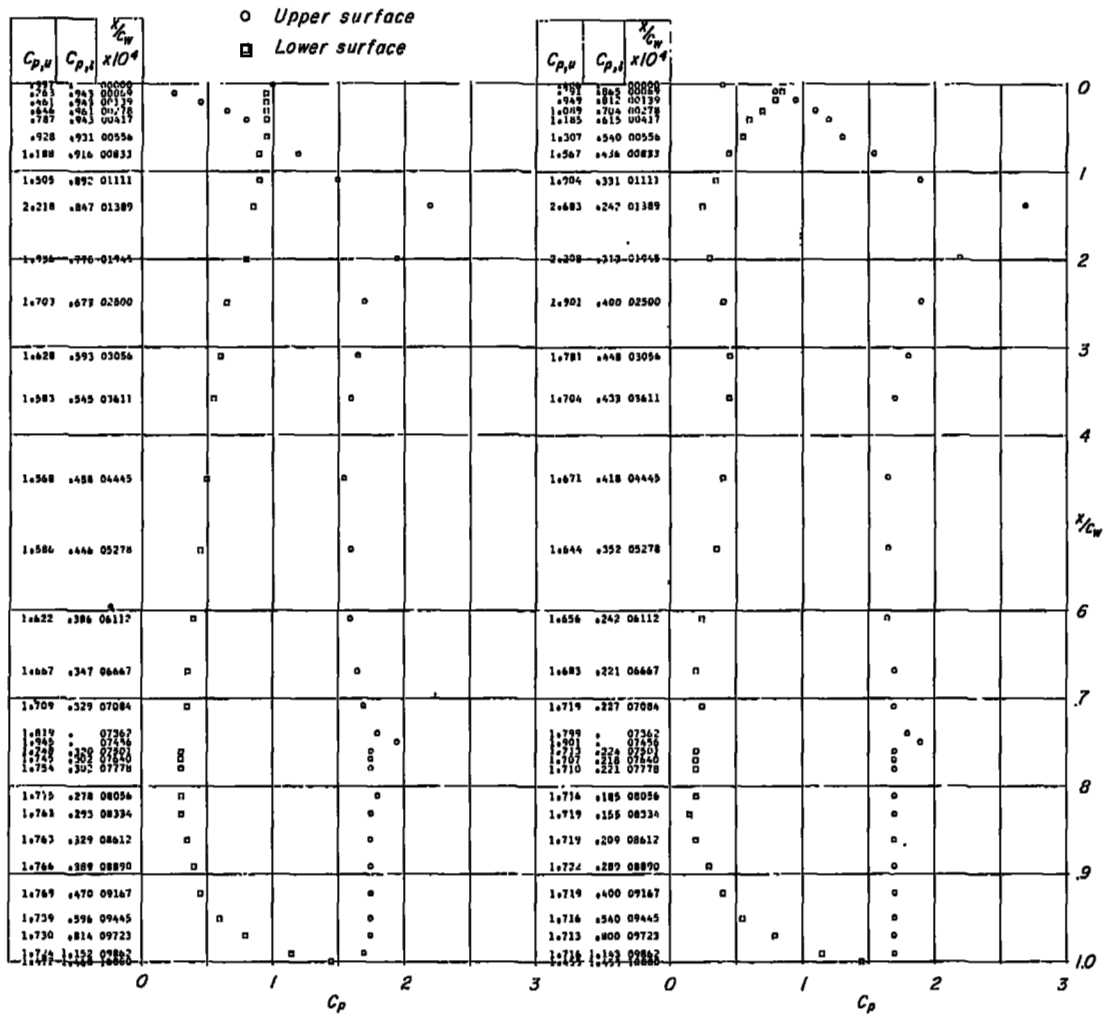
Figure 46.- Concluded.



(a) $\alpha = -8^{\circ}$.

(b) $\alpha = -4^{\circ}$.

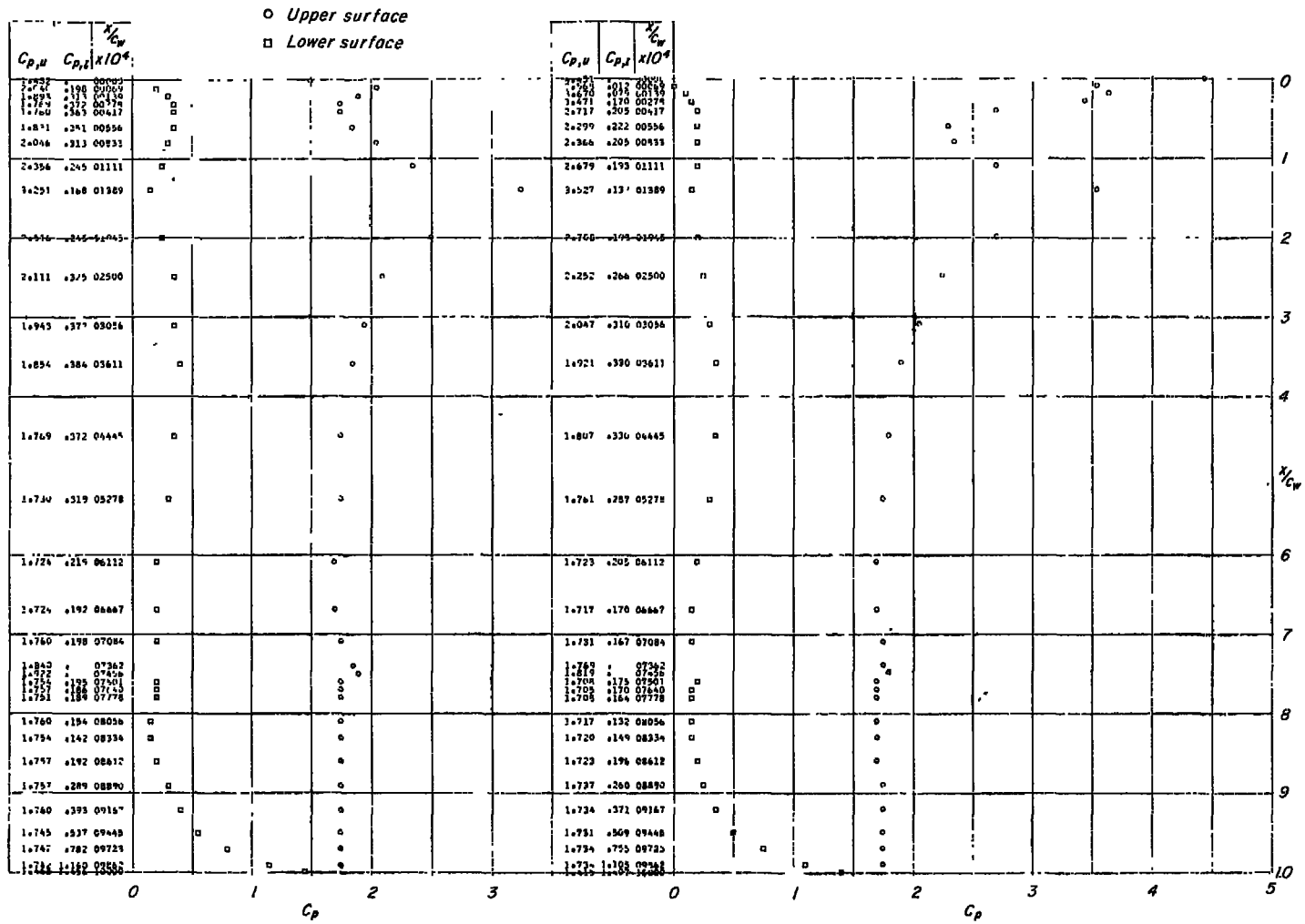
Figure 47.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 30^{\circ}$; $\delta_f = 60^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = 0^{\circ}$.

(d) $\alpha = 4^{\circ}$.

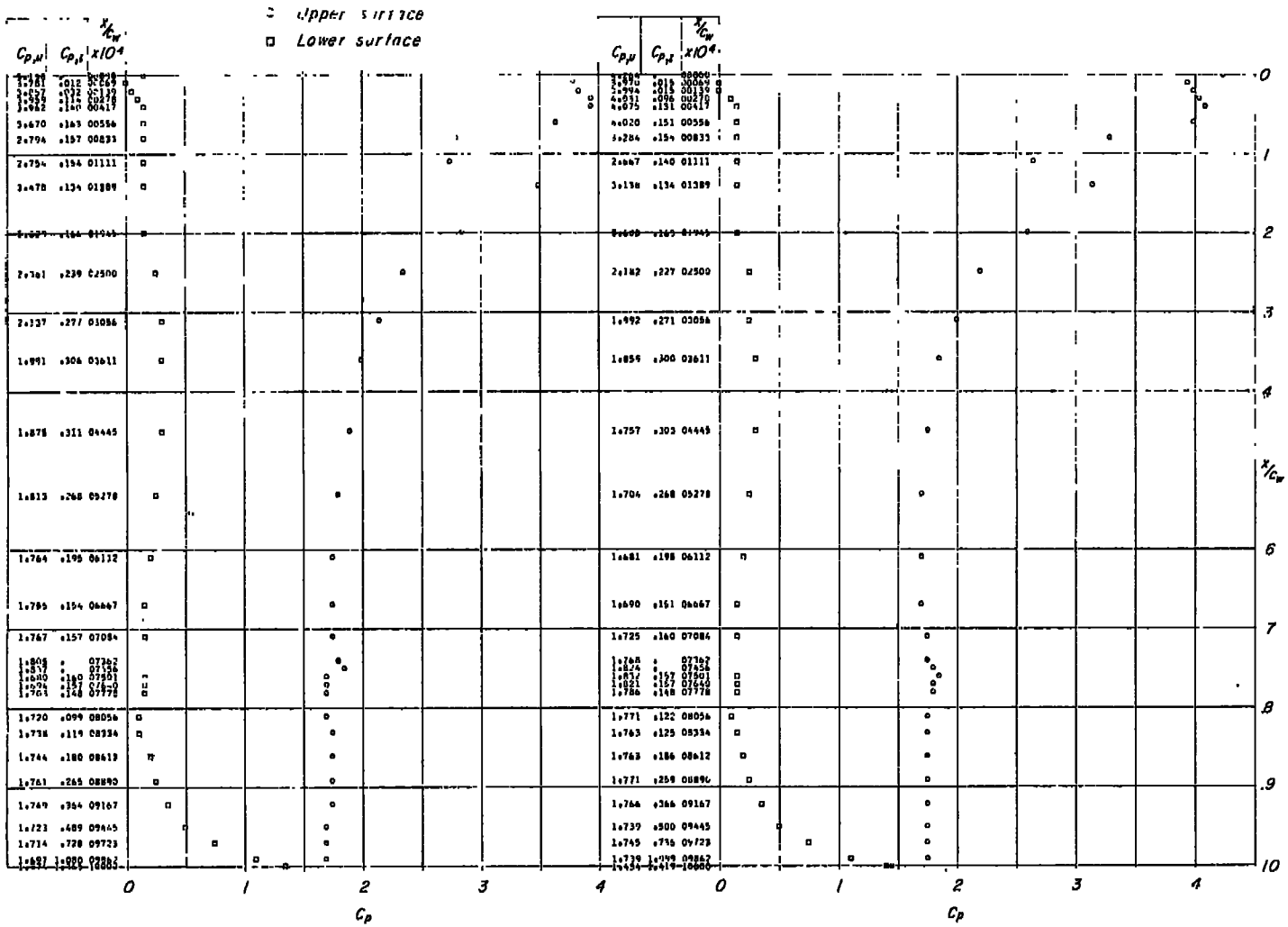
Figure 47.- Continued.



(e) $\alpha = 8^{\circ}$.

(f) $\alpha = 12^{\circ}$.

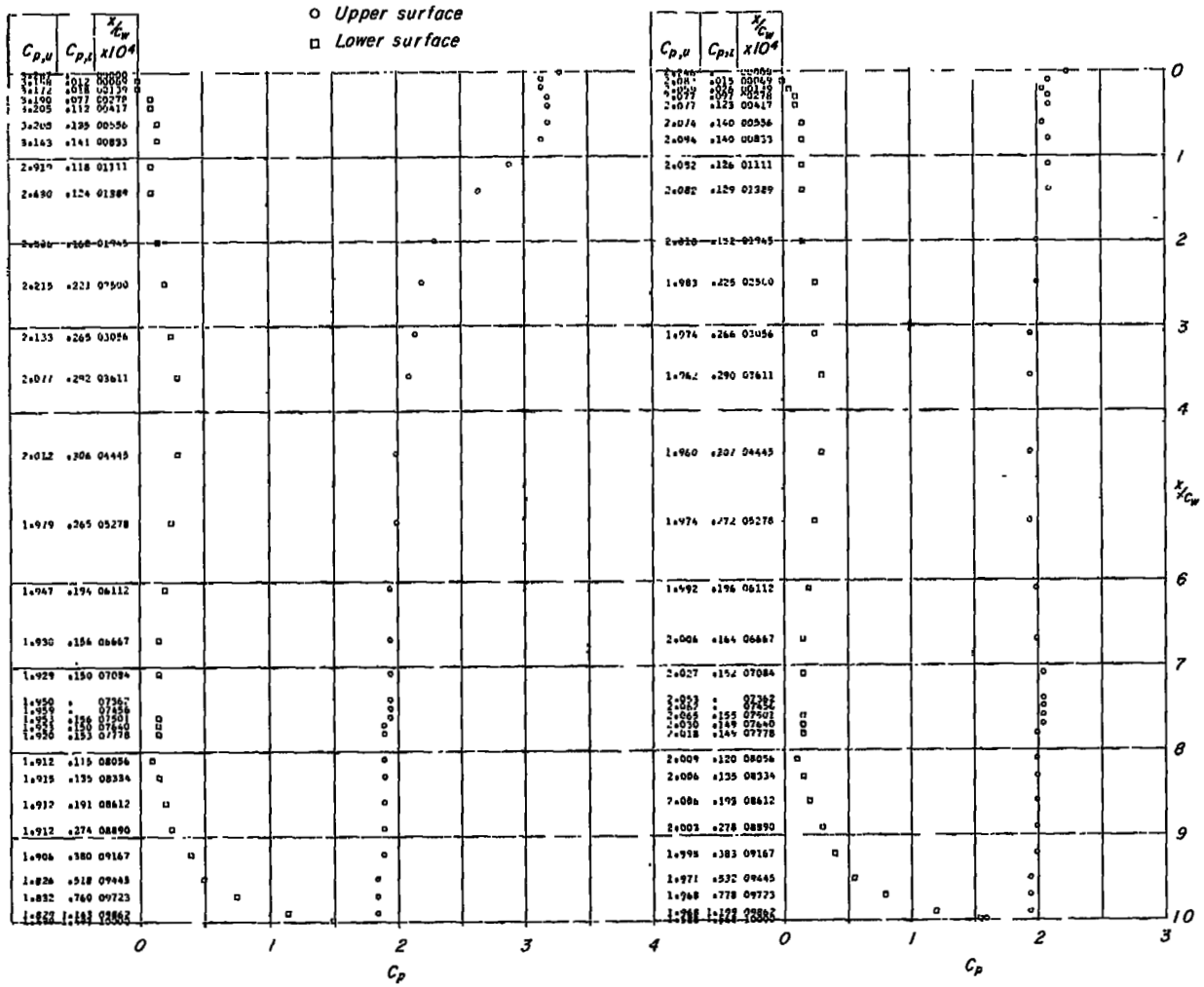
Figure 47.- Continued.



(g) $\alpha = 14^{\circ}$.

(h) $\alpha = 16^{\circ}$.

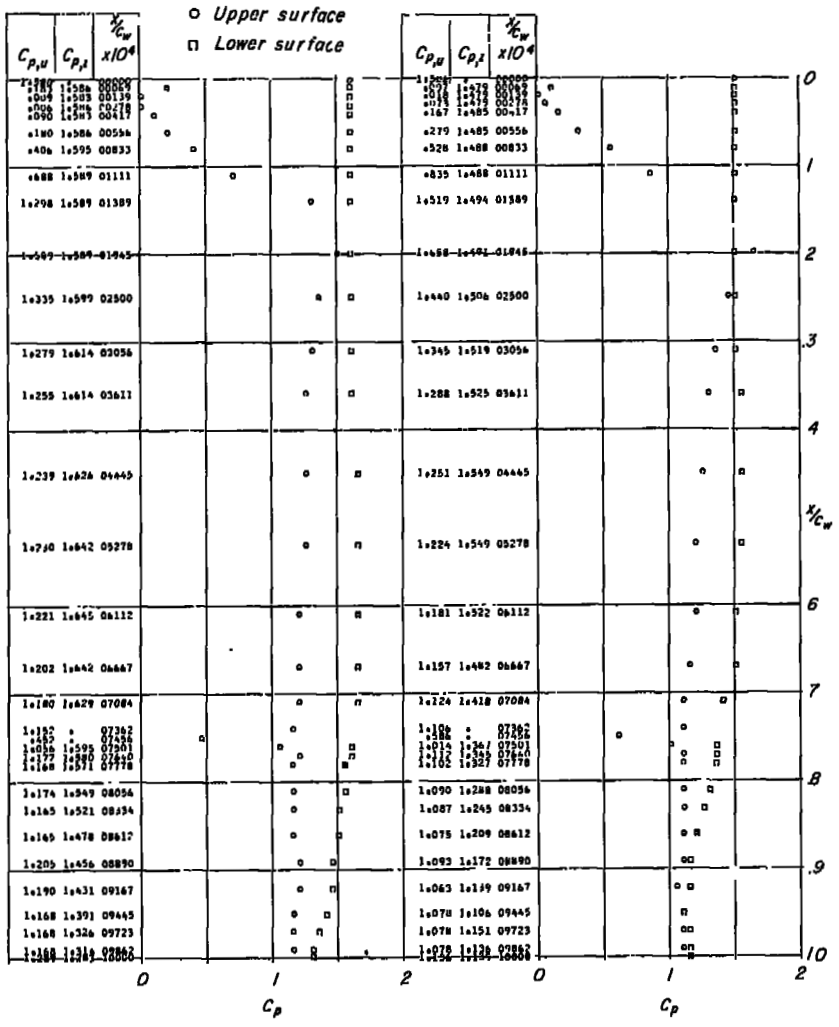
Figure 47.- Continued.



(i) $\alpha = 20^{\circ}$.

(j) $\alpha = 24^{\circ}$.

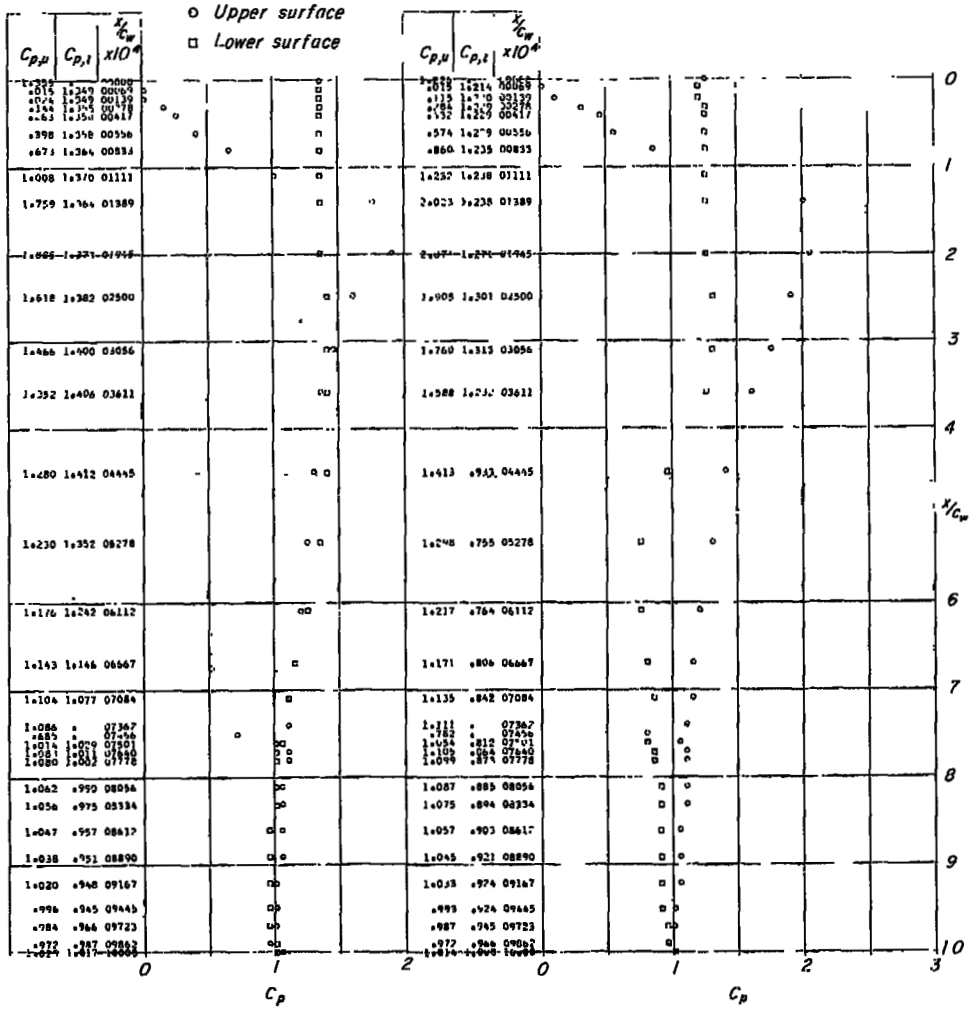
Figure 47.- Concluded.



(a) $\alpha = -4^{\circ}$.

(b) $\alpha = 0^{\circ}$.

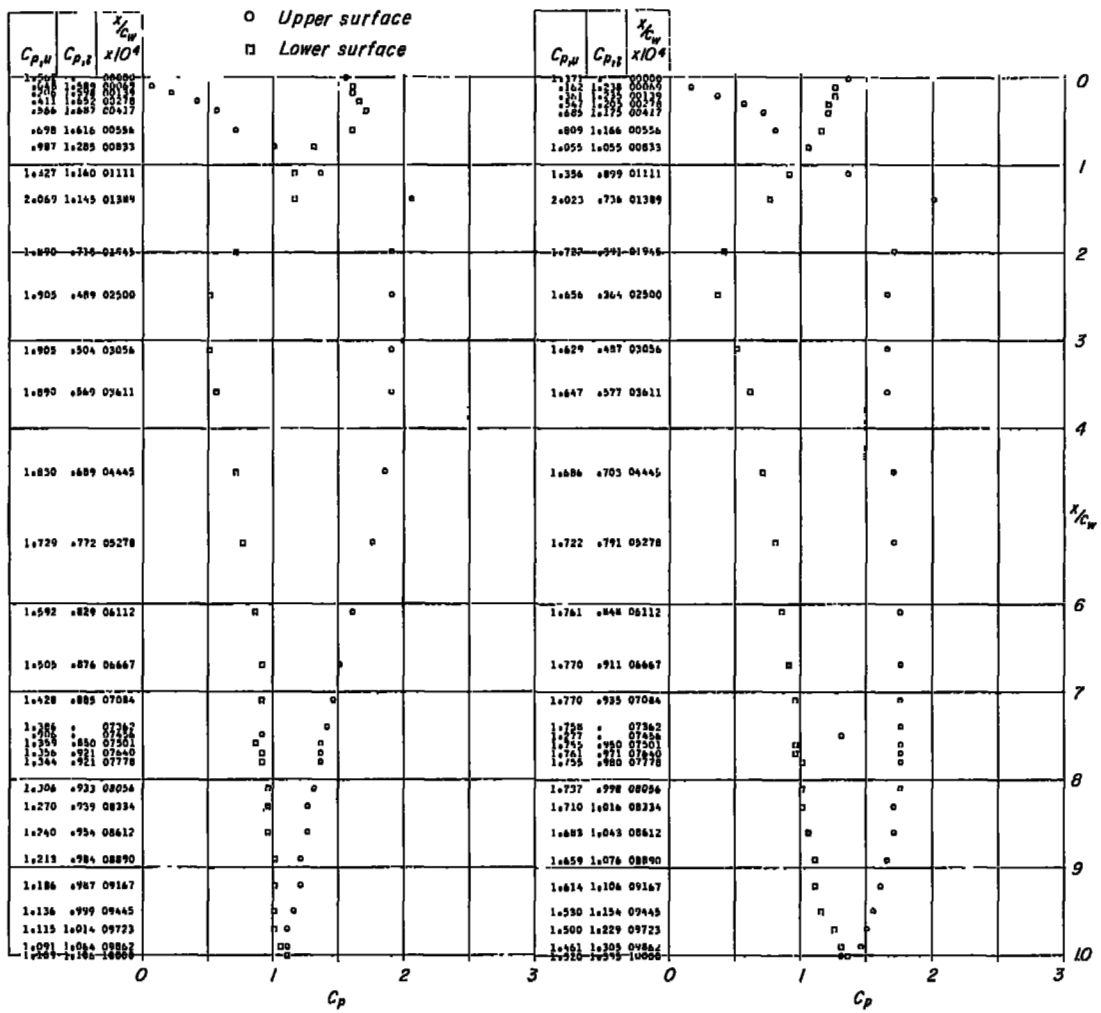
Figure 48.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 45^{\circ}$; $\delta_f = 0^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = 4^{\circ}$.

(d) $\alpha = 8^{\circ}$.

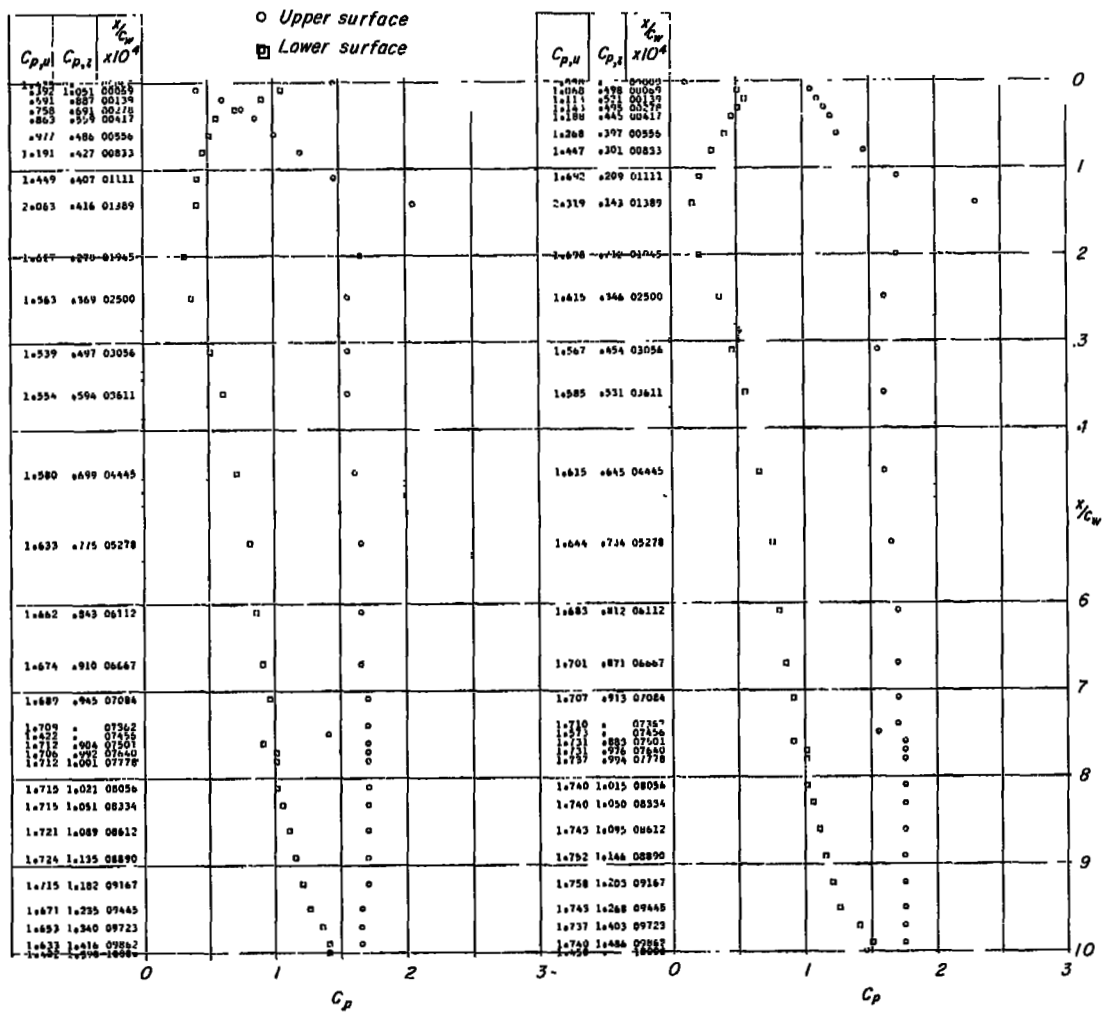
Figure 48.- Continued.



(e) $\alpha = 12^{\circ}$.

(f) $\alpha = 16^{\circ}$.

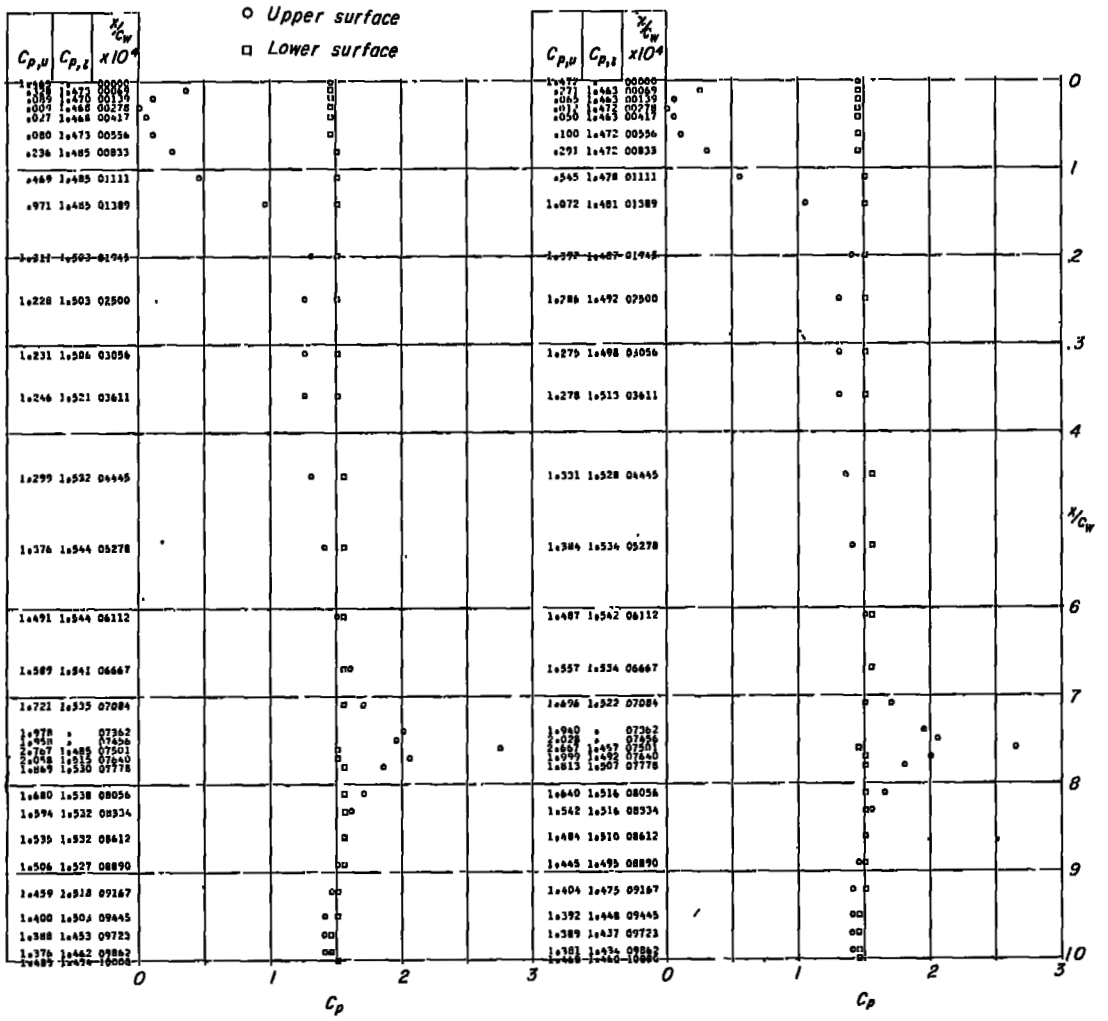
Figure 48.- Continued.



(g) $\alpha = 20^{\circ}$.

(h) $\alpha = 24^{\circ}$.

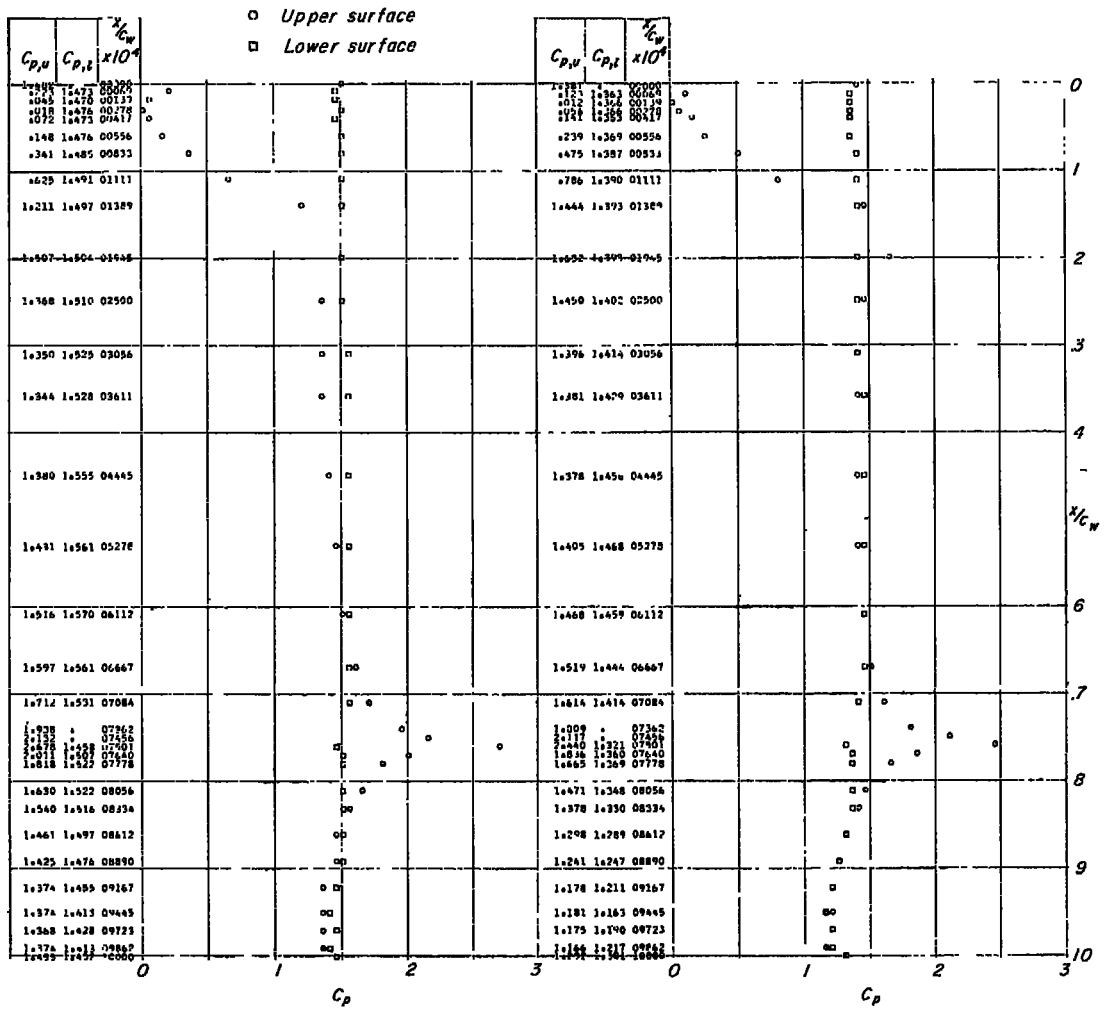
Figure 48.- Concluded.



(a) $\alpha = -12^{\circ}$.

(b) $\alpha = -10^{\circ}$.

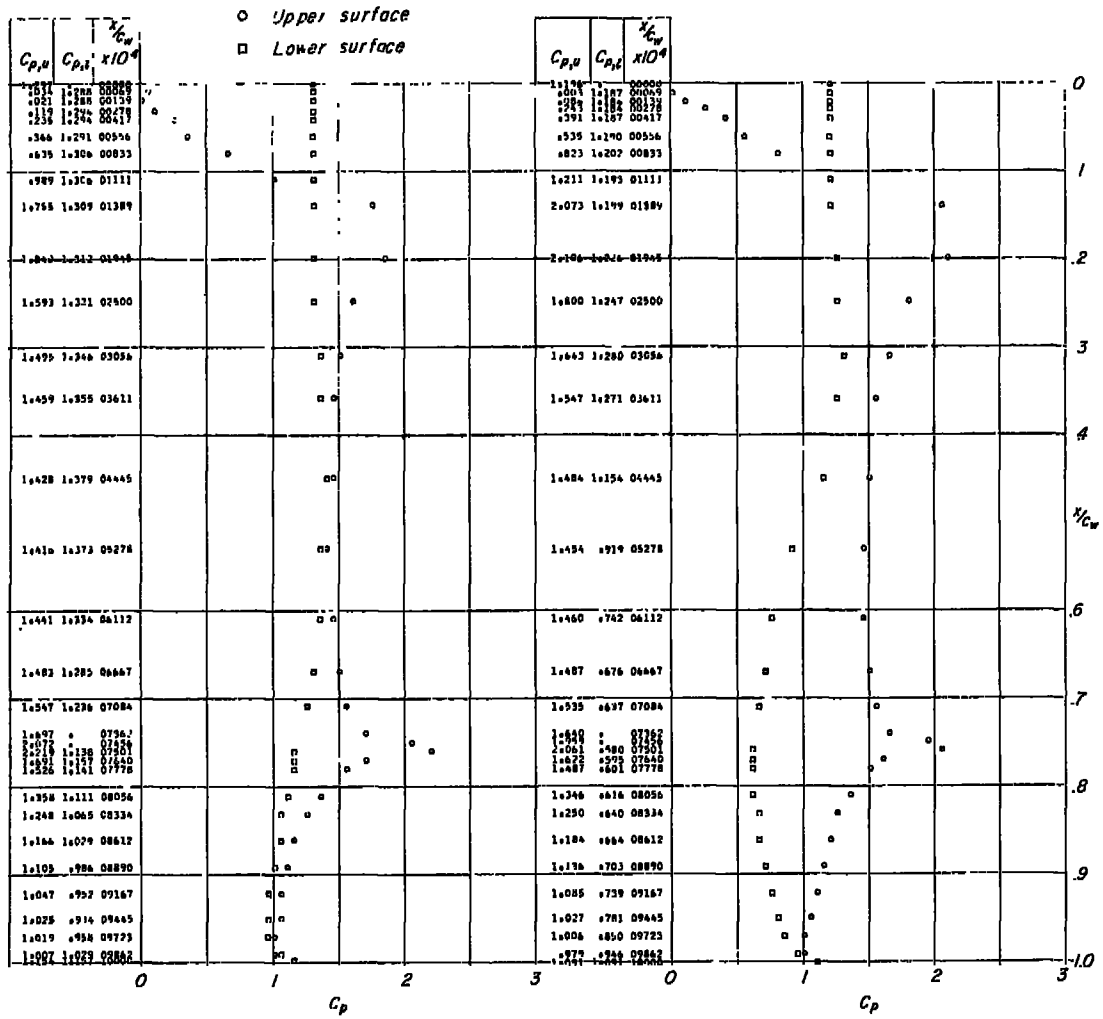
Figure 49.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 45^{\circ}$; $\delta_f = 15^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = -8^{\circ}$.

(d) $\alpha = -4^{\circ}$.

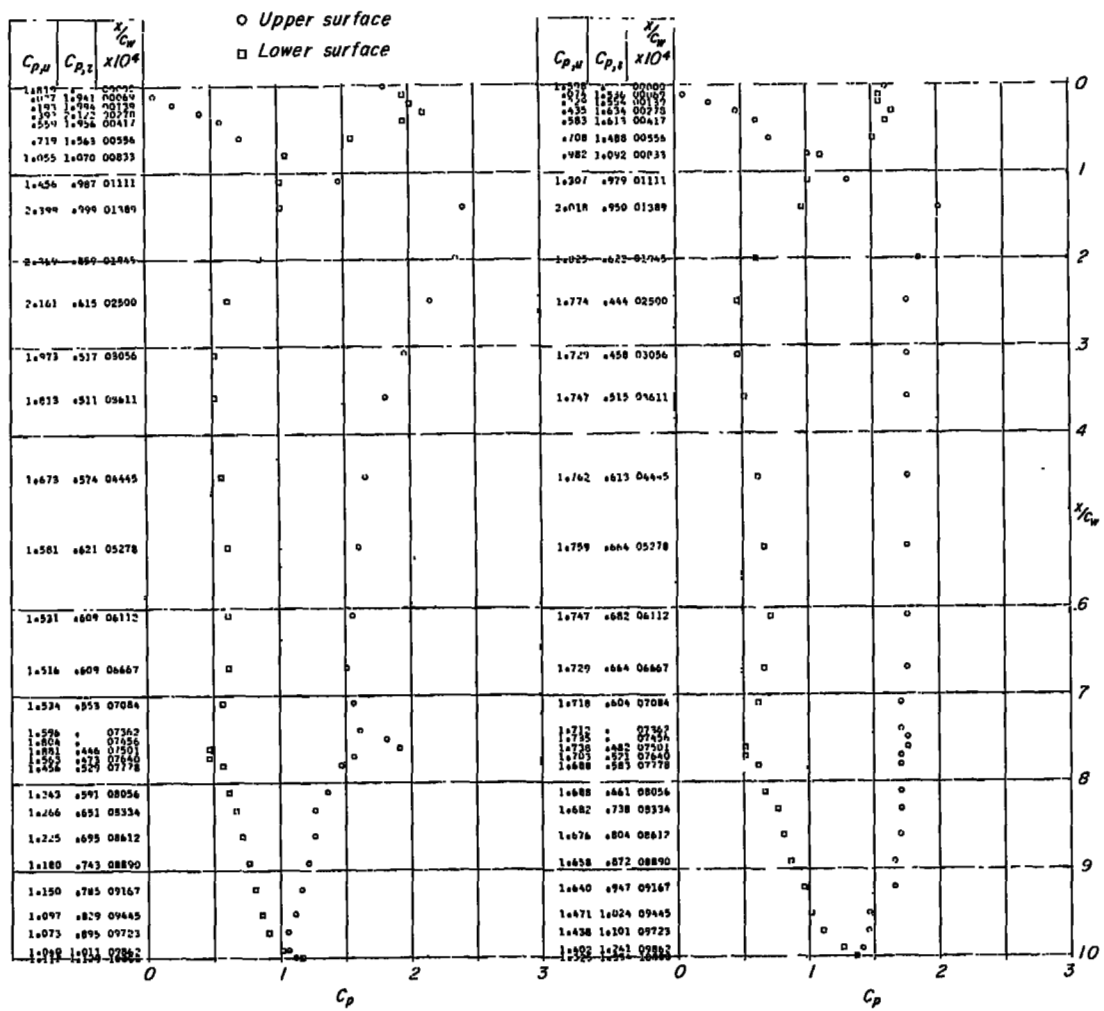
Figure 49.- Continued.



(e) $\alpha = 0^{\circ}$.

(f) $\alpha = 4^{\circ}$.

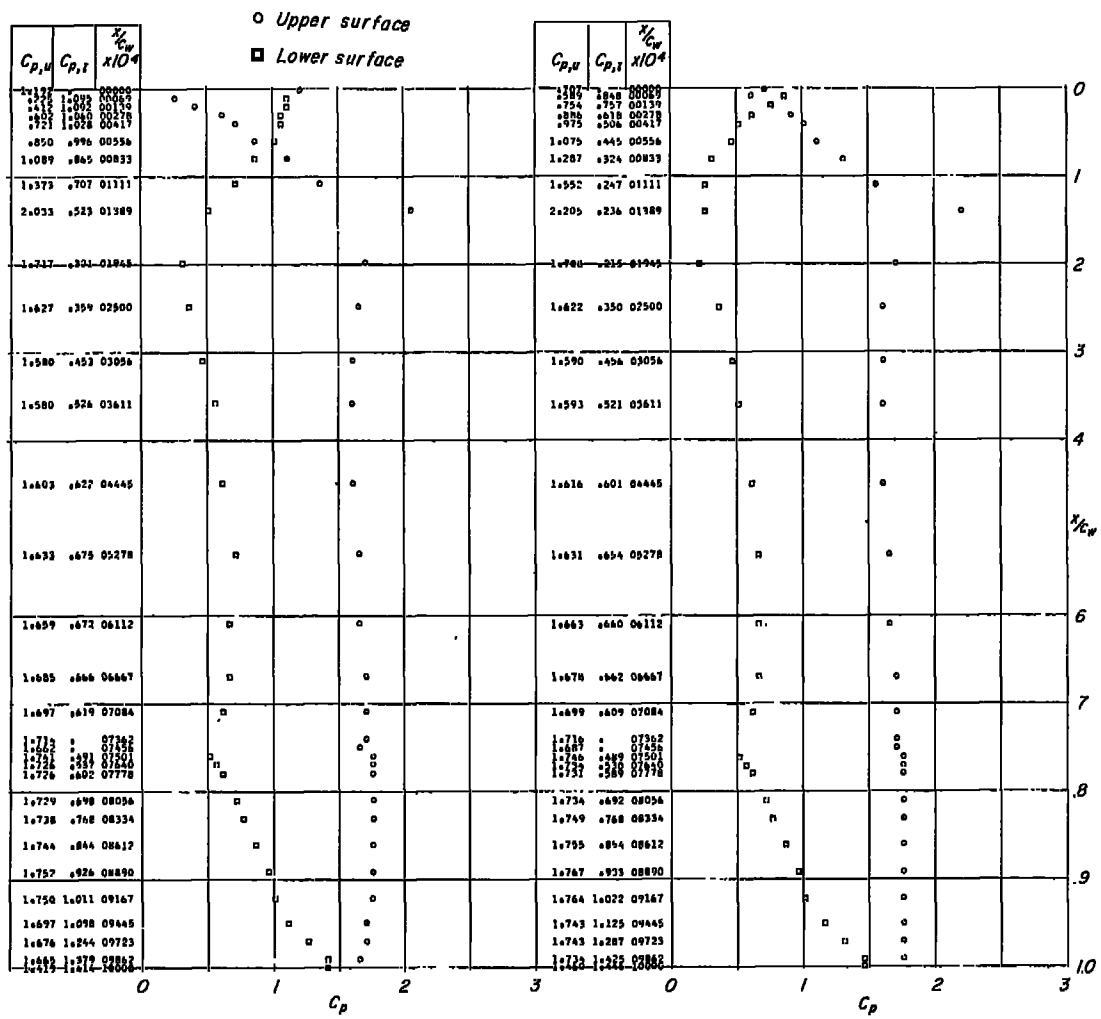
Figure 49.- Continued.



(g) $\alpha = 8^{\circ}$.

(h) $\alpha = 12^{\circ}$.

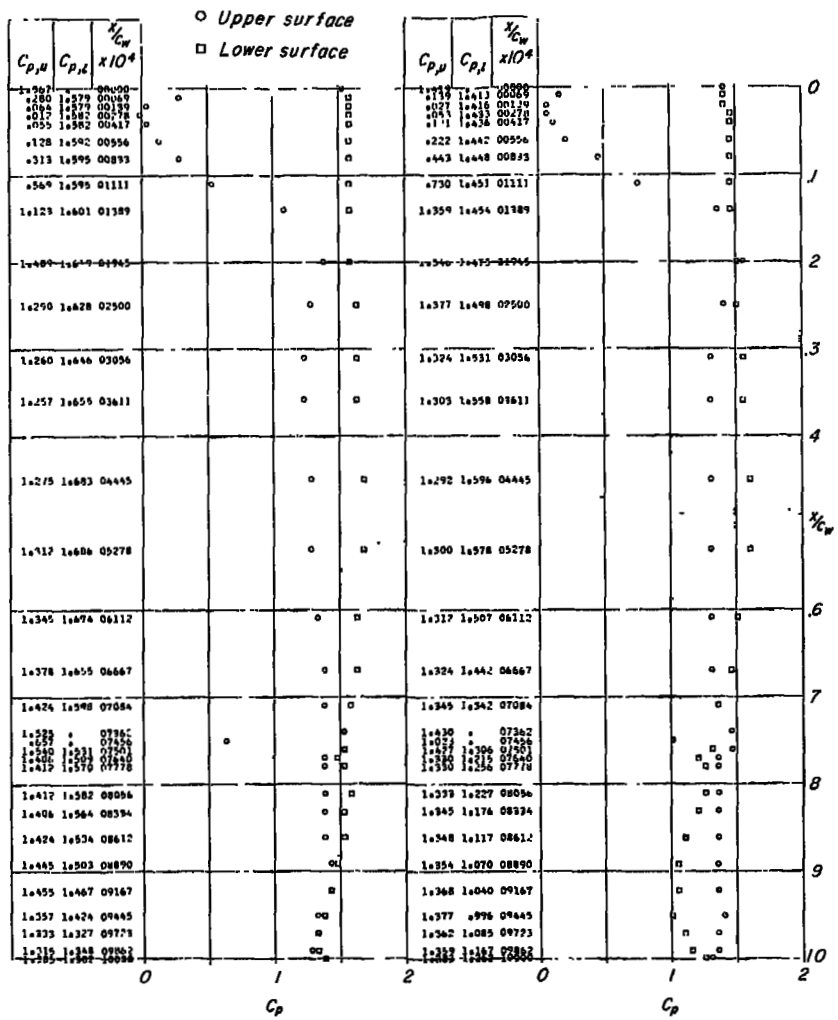
Figure 49.- Continued.



(i) $\alpha = 16^{\circ}$.

(j) $\alpha = 20^{\circ}$.

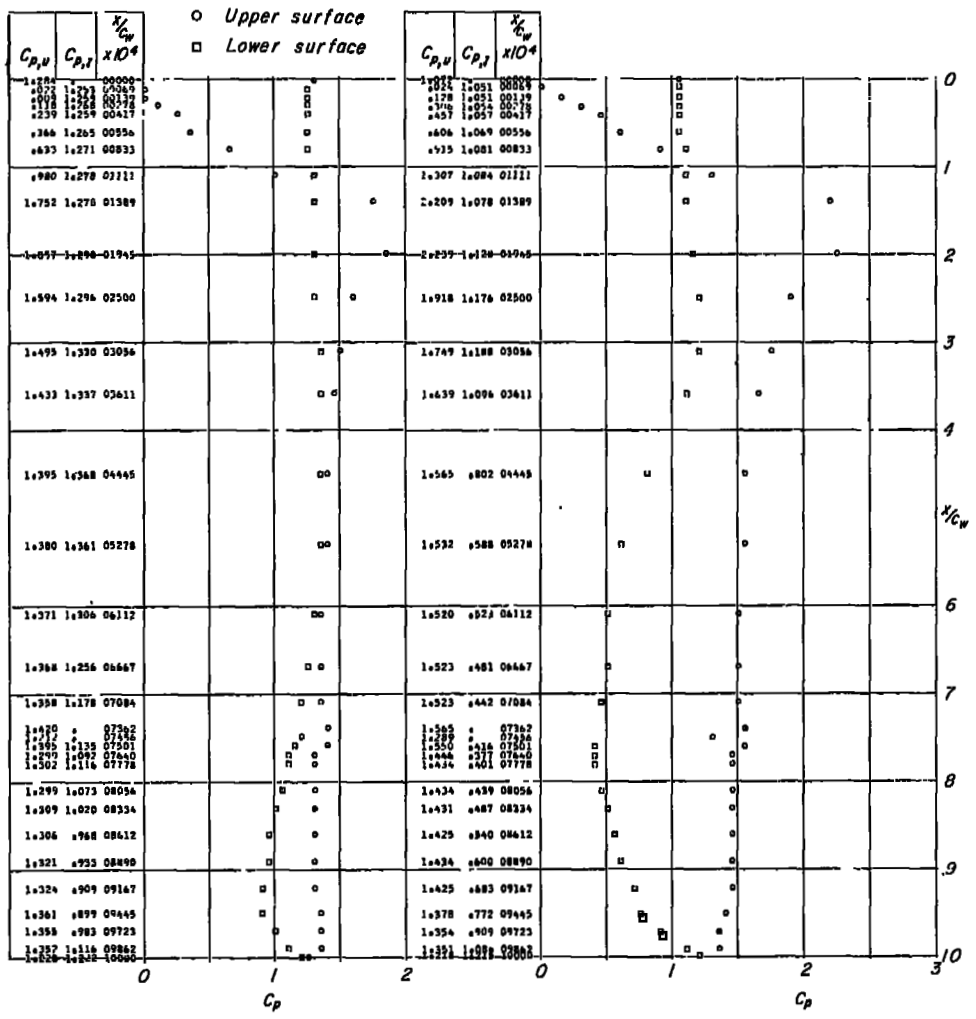
Figure 49.- Concluded.



(a) $\alpha = -8^{\circ}$.

(b) $\alpha = -4^{\circ}$.

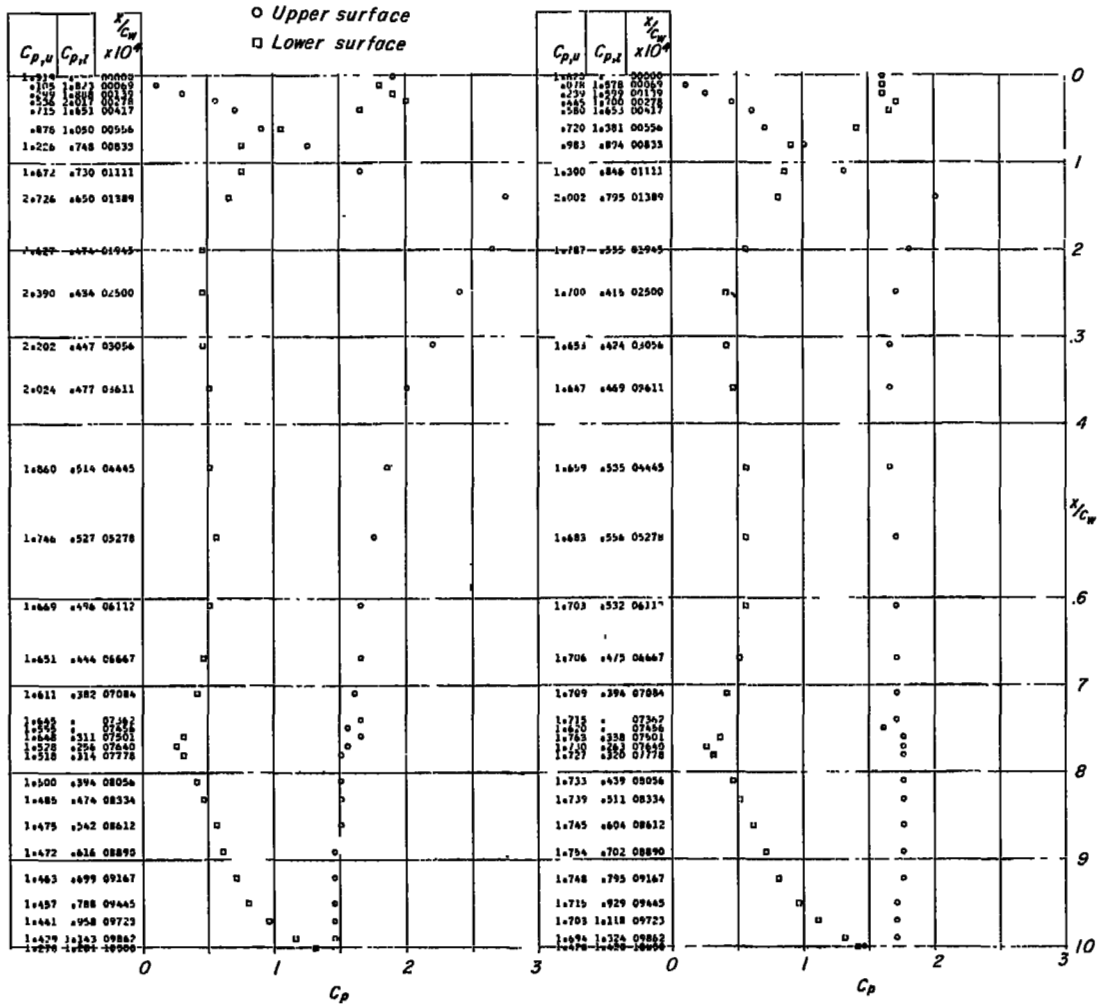
Figure 50.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 45^{\circ}$; $\delta_f = 30^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = 0^{\circ}$.

(d) $\alpha = 4^{\circ}$.

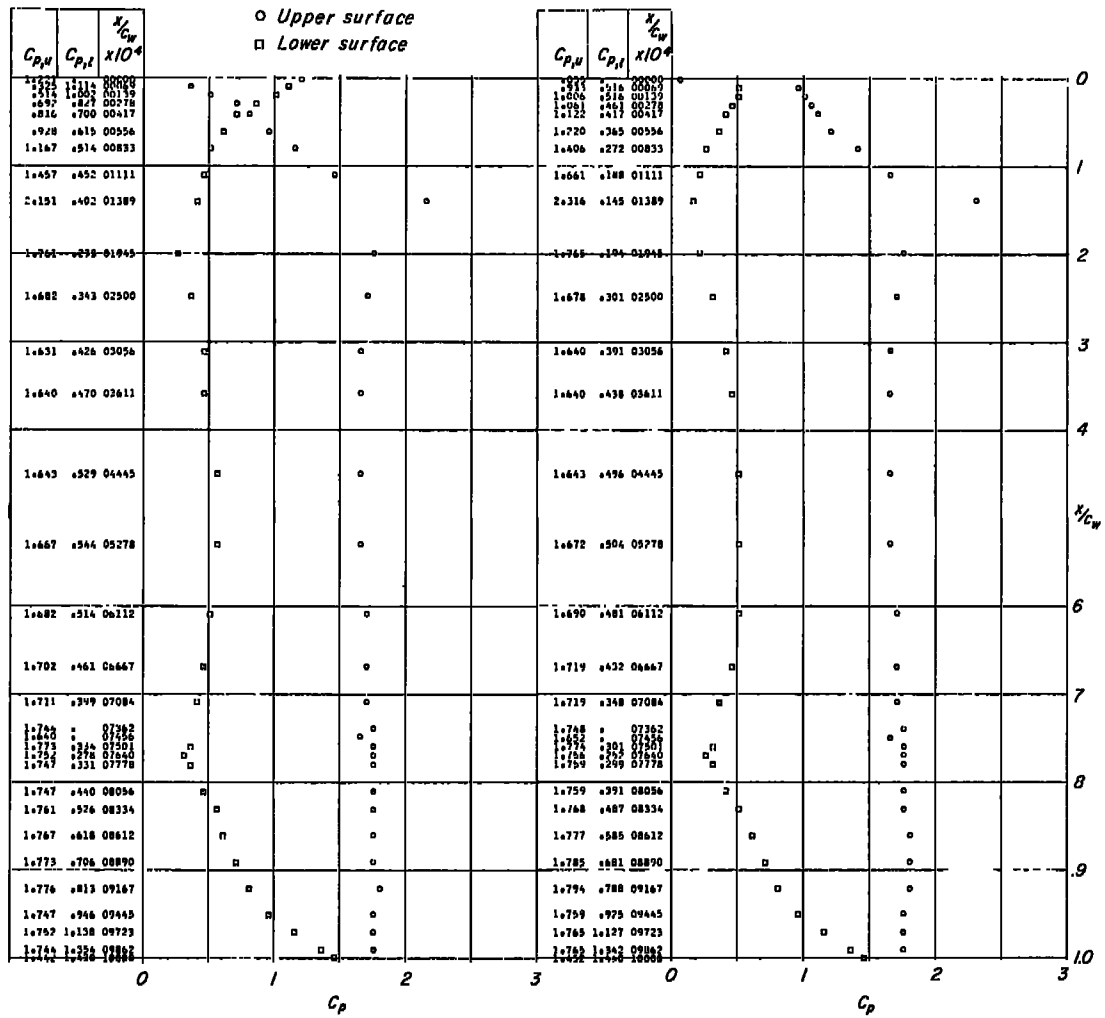
Figure 50.- Continued.



(e) $\alpha = 8^{\circ}$.

(f) $\alpha = 12^{\circ}$.

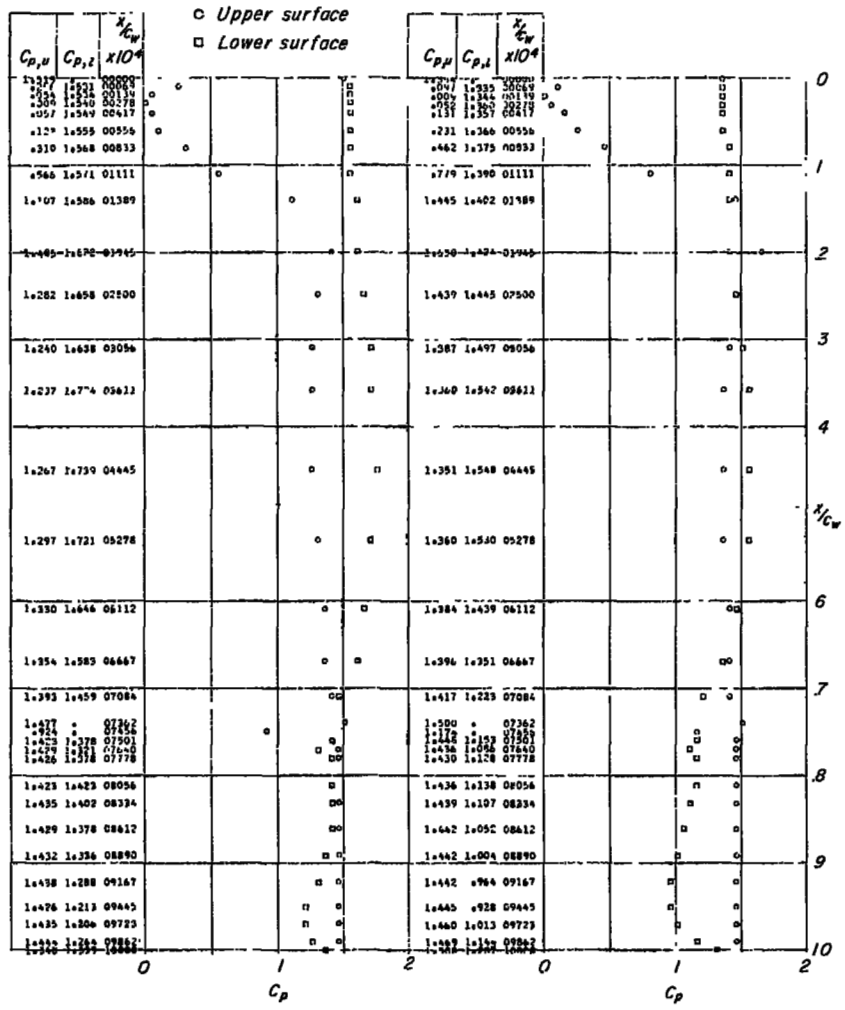
Figure 50.- Continued.



(g) $\alpha = 16^{\circ}$.

(h) $\alpha = 20^{\circ}$.

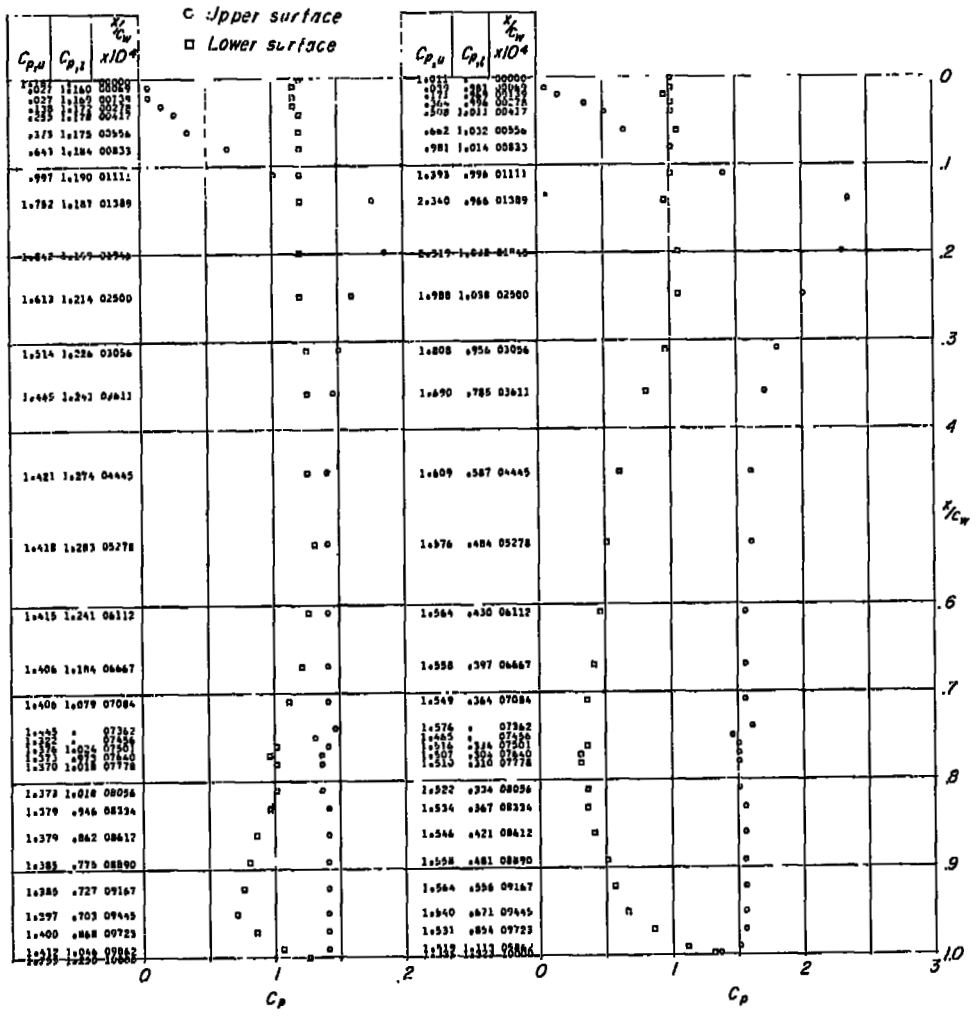
Figure 50.- Concluded.



(a) $\alpha = -8^{\circ}$.

(b) $\alpha = -4^{\circ}$.

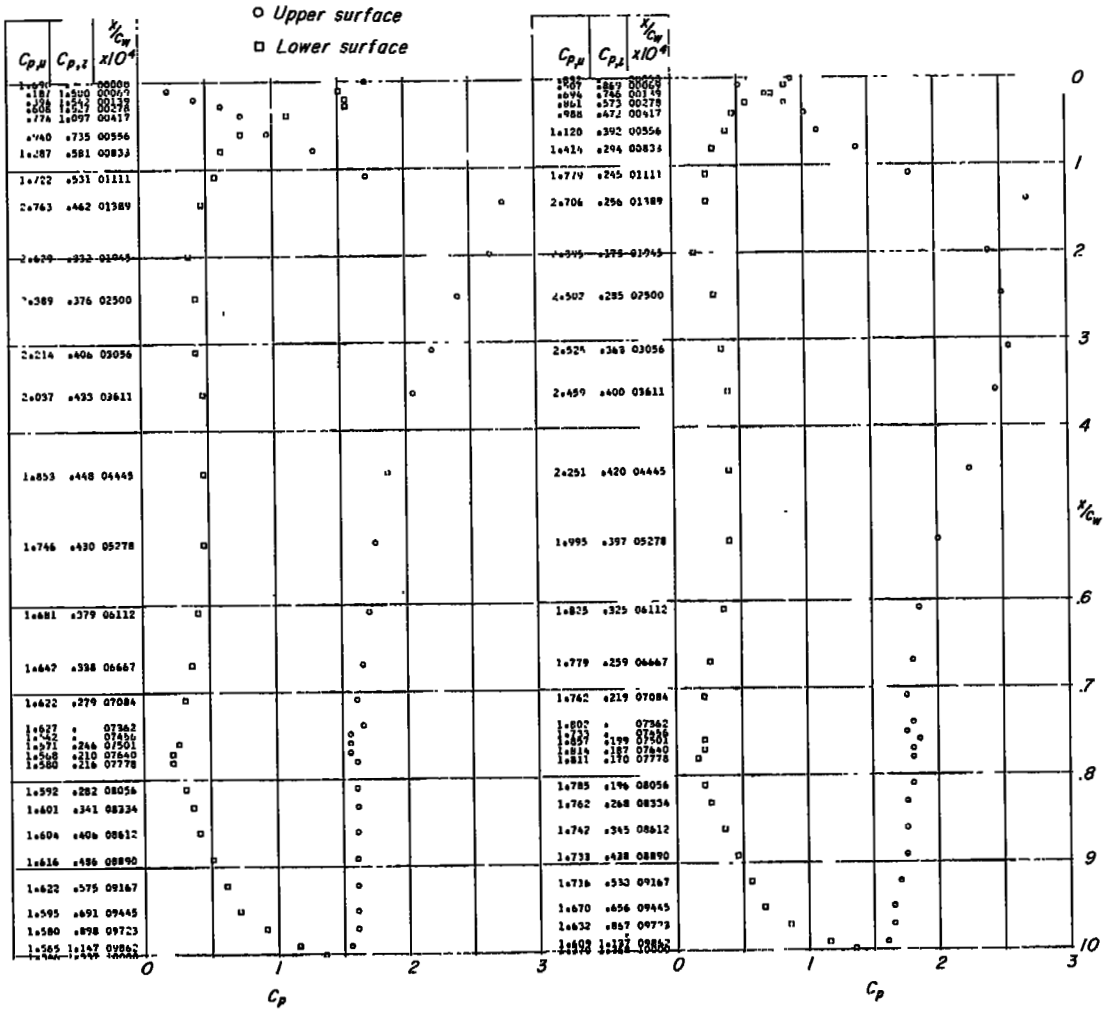
Figure 51.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 45^{\circ}$; $\delta_f = 45^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = 0^{\circ}$.

(d) $\alpha = 4^{\circ}$.

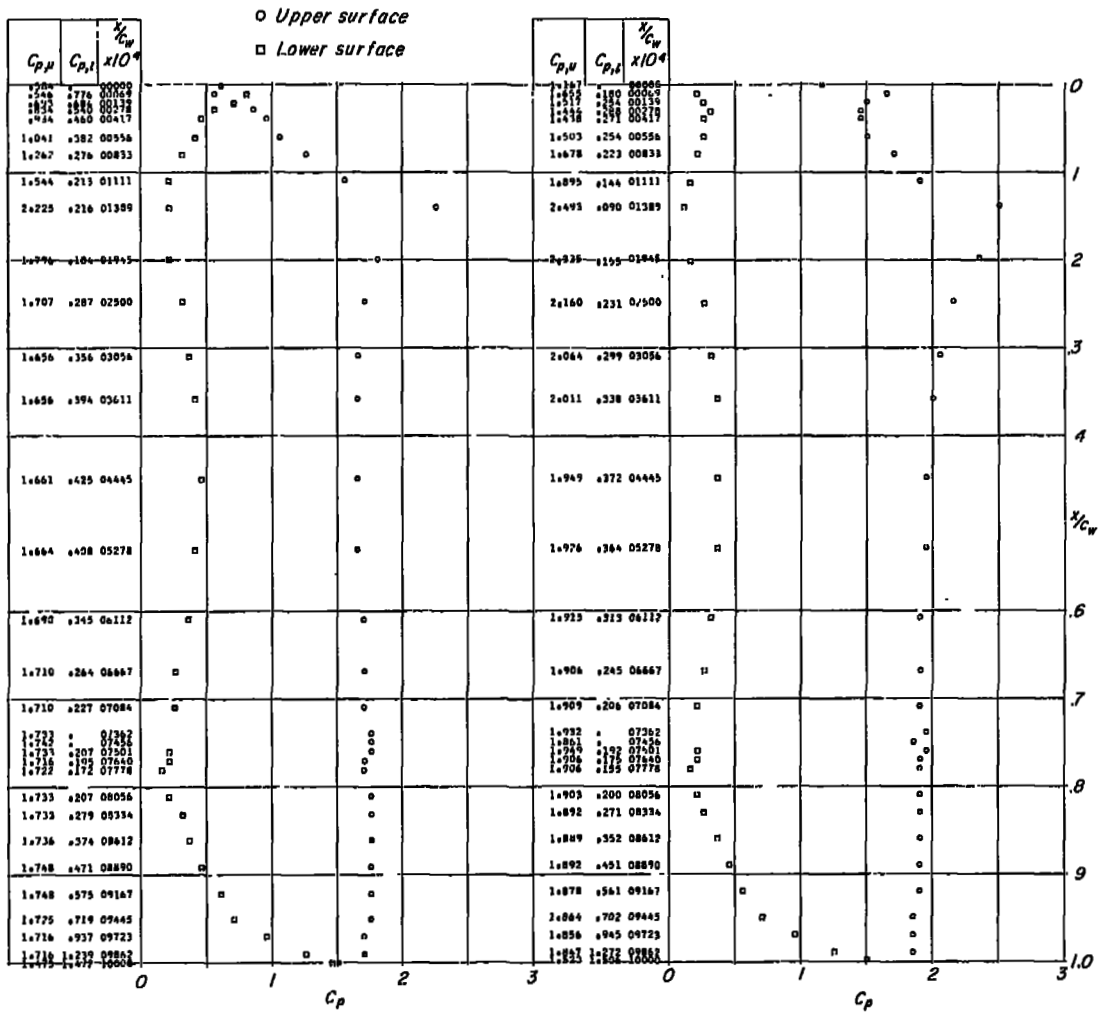
Figure 51.- Continued.



(e) $\alpha = 8^{\circ}$.

(f) $\alpha = 12^{\circ}$.

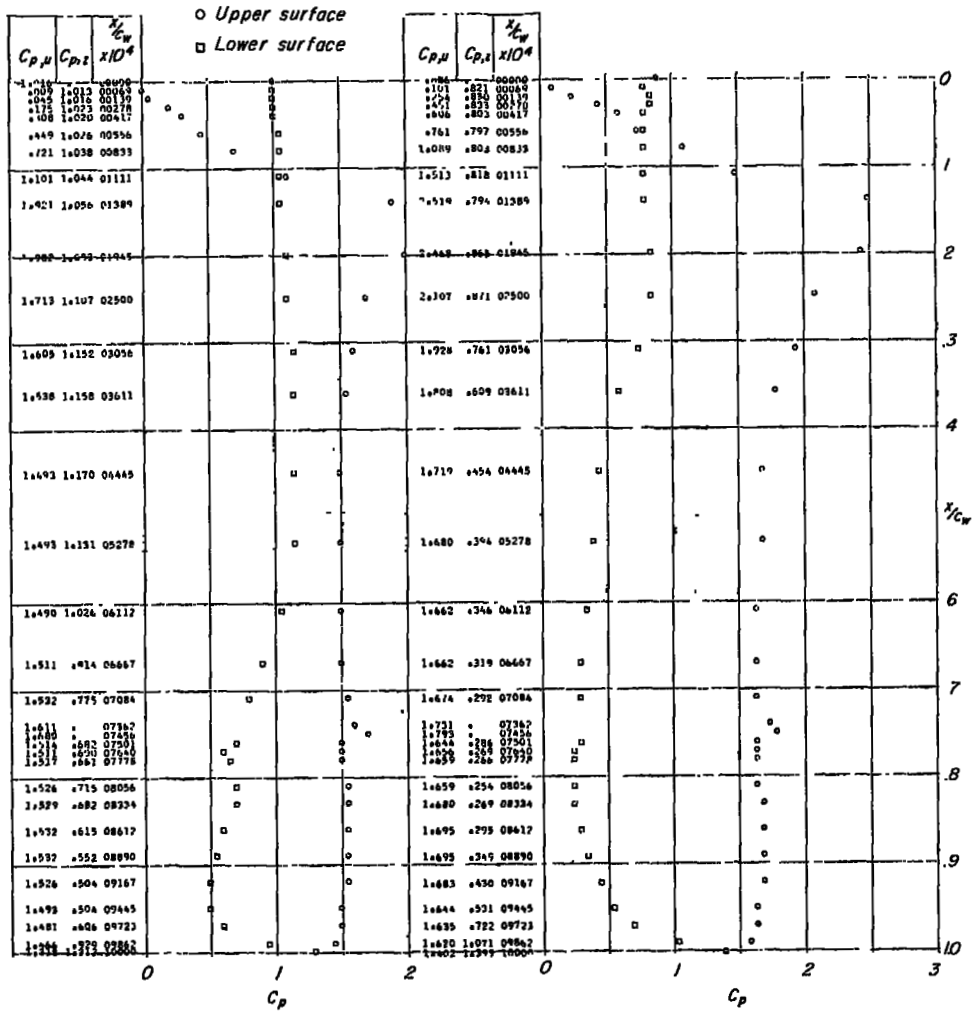
Figure 51.- Continued.



(g) $\alpha = 16^{\circ}$.

(h) $\alpha = 20^{\circ}$.

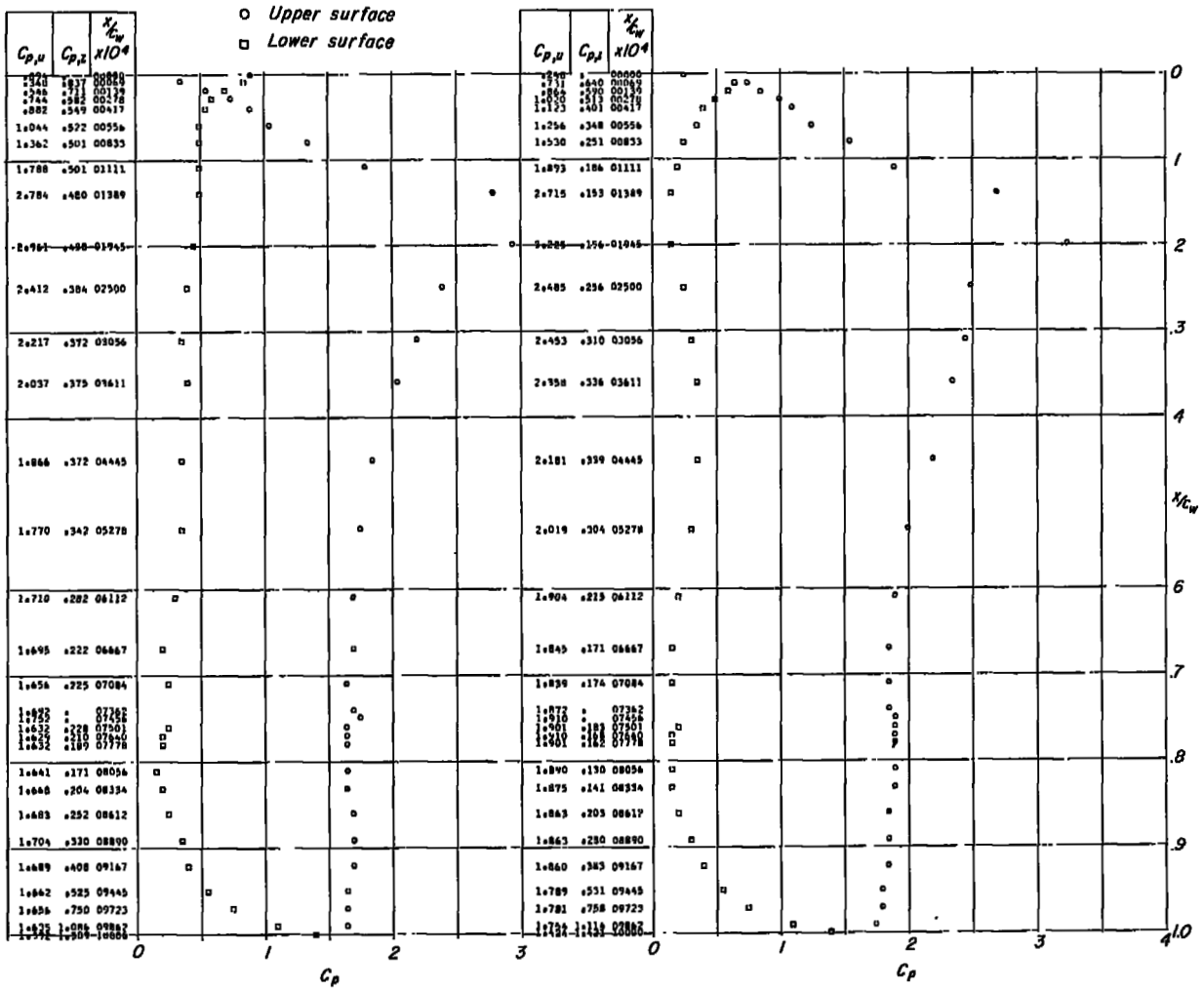
Figure 51.- Concluded.



(a) $\alpha = 0^{\circ}$.

(b) $\alpha = 4^{\circ}$.

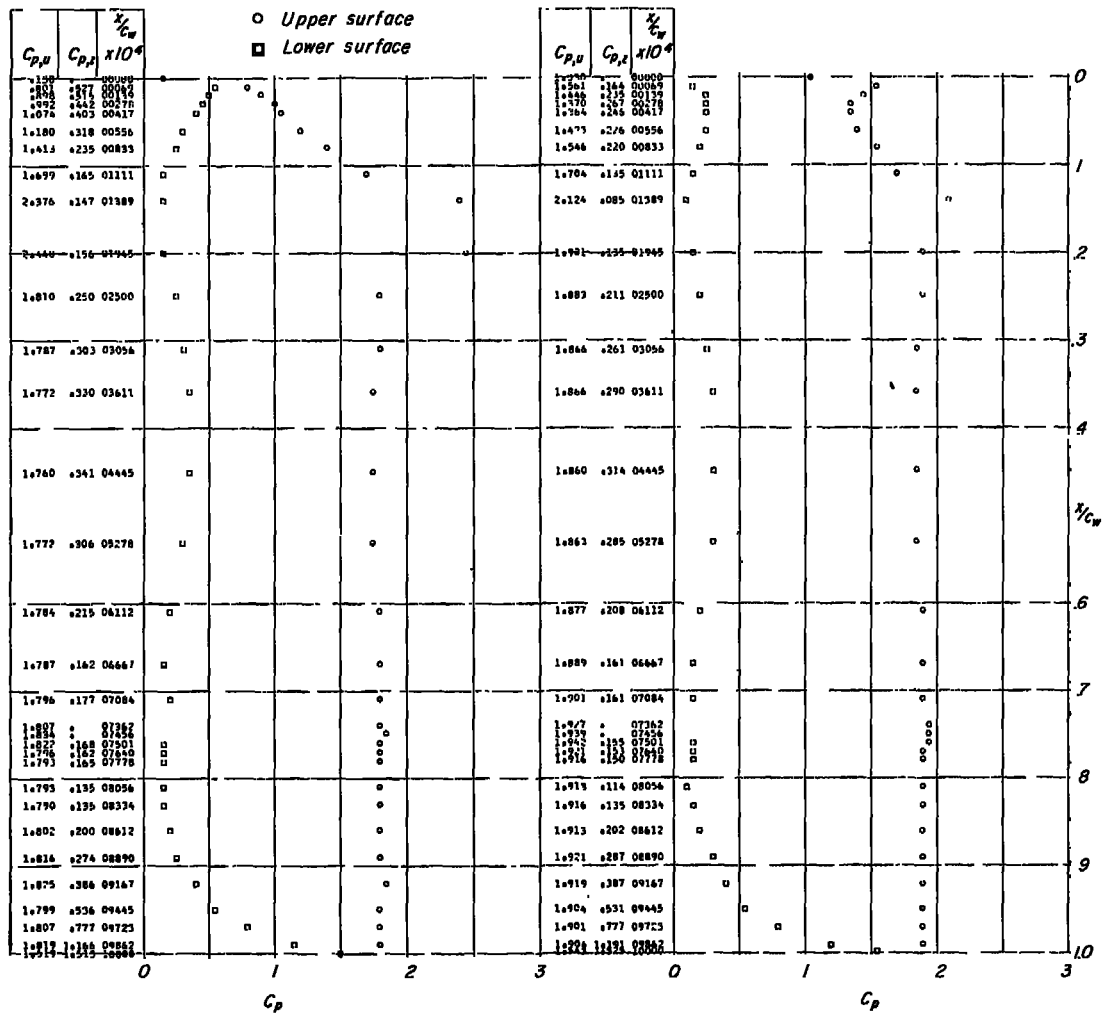
Figure 52.- Chordwise pressure distribution over model. $c_f = 0.25c_w$; $\delta_N = 45^{\circ}$; $\delta_f = 60^{\circ}$; $q \approx 25$ lb/sq ft. (Tabulated data of points plotted are to left of plot.)



(c) $\alpha = 8^{\circ}$.

(d) $\alpha = 12^{\circ}$.

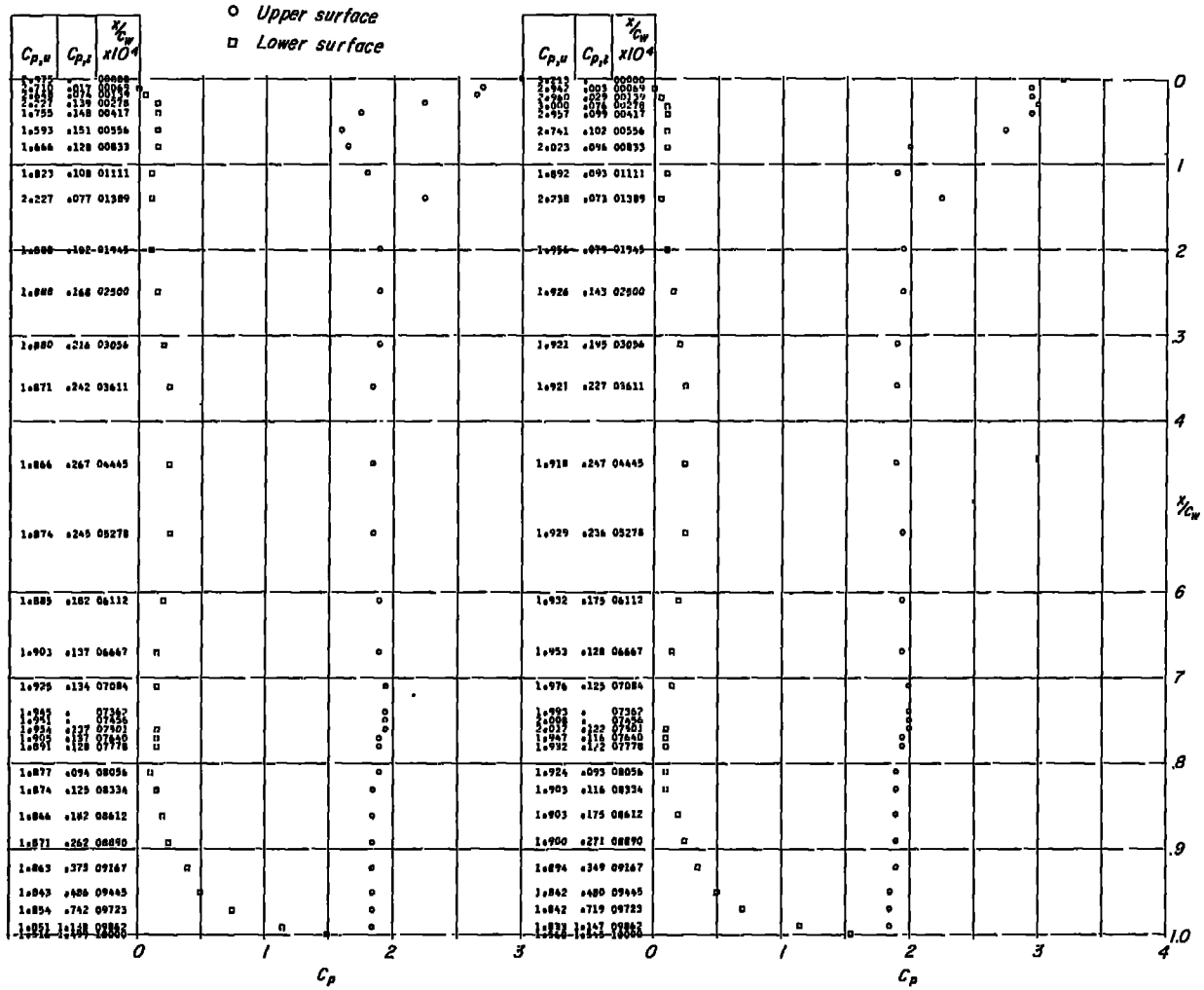
Figure 52.- Continued.



(e) $\alpha = 16^{\circ}$.

(f) $\alpha = 20^{\circ}$.

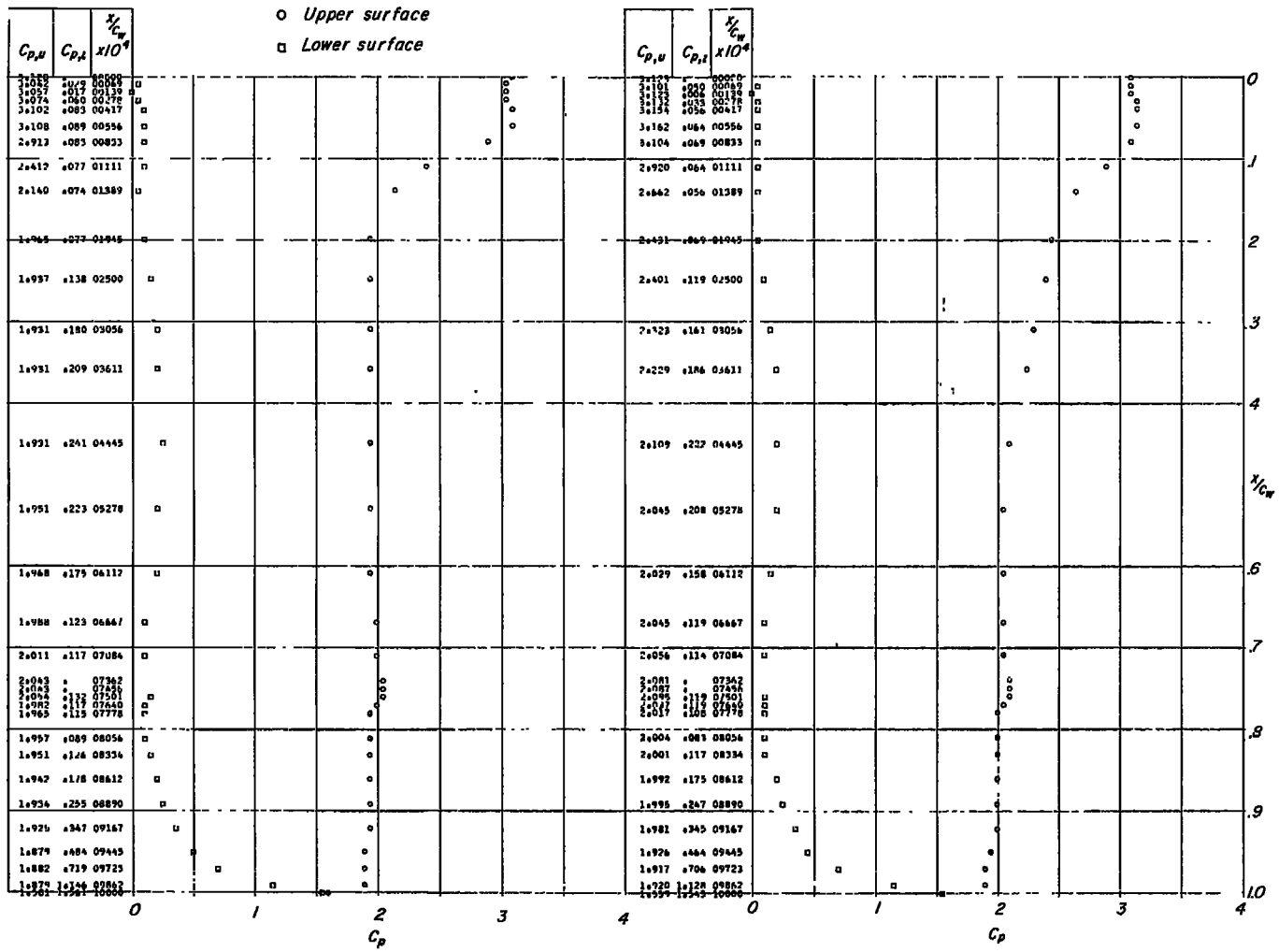
Figure 52.- Continued.



(g) $\alpha = 24^{\circ}$.

(h) $\alpha = 26^{\circ}$.

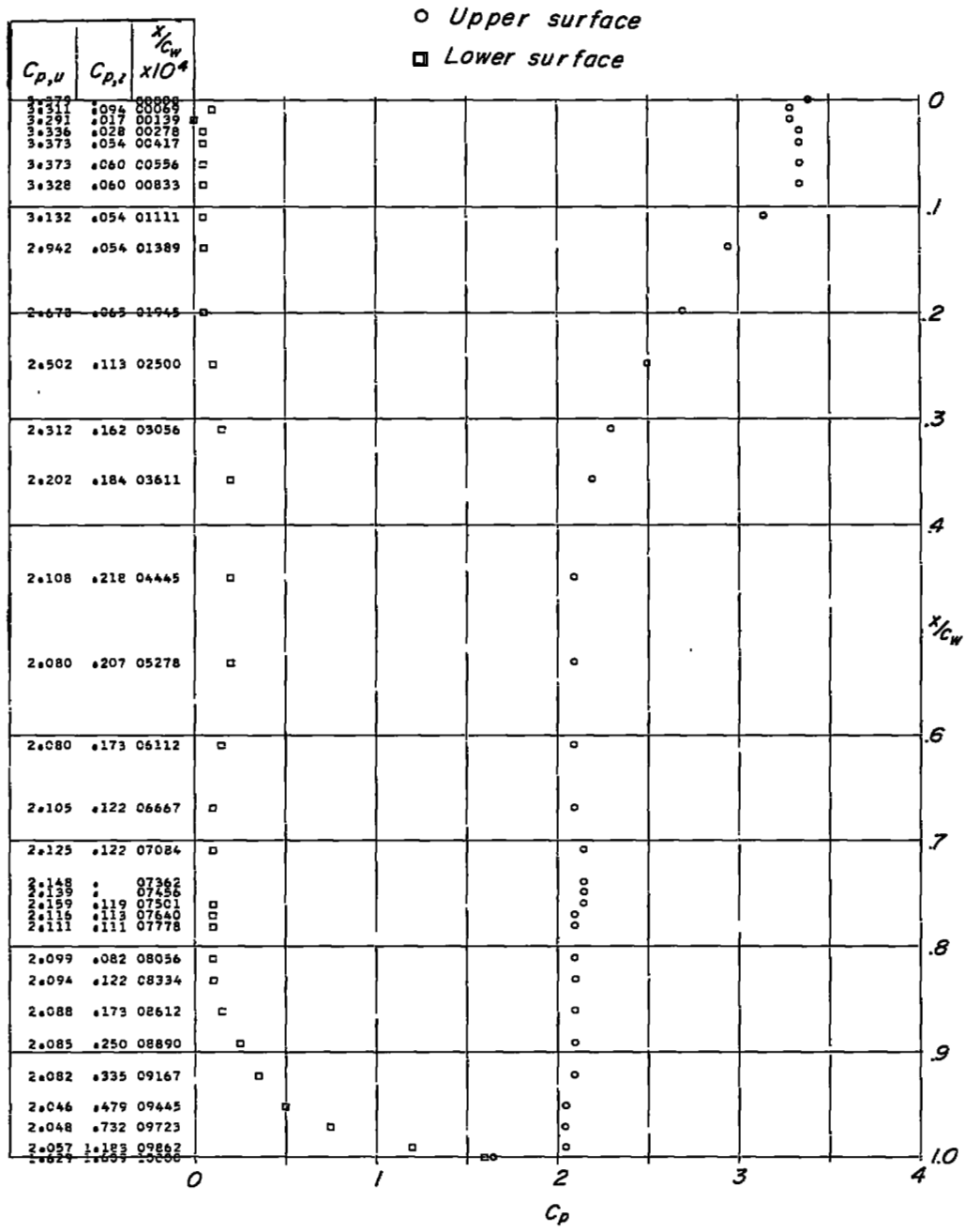
Figure 52.- Continued.



(i) $\alpha = 28^{\circ}$.

(j) $\alpha = 30^{\circ}$.

Figure 52.- Continued.



(k) $\alpha = 32^{\circ}$.

Figure 52.- Concluded.

LANGLEY RESEARCH CENTER
3 1176 00501 2530