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RESEARCH MEMORANDUM

LOW-SPEED PRESSURE-DISTRIBUTION INVESTIGATION OF A THIN-DELTA-WING—FUSELAGE MODEL HAVING DOUBLE SLOTTED FLAPS AND SPOILERS

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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LOW-SPEED PRESSURE-DISTRIBUTION INVESTIGATION OF A THIN-DELTA-WING—FUSELAGE MODEL HAVING DOUBLE SLOTTED FLAPS AND SPOILERS

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SUMMARY

An investigation was made in the Langley 300 MPH 7- by 10-foot tunnel to determine the chordwise and spanwise load distribution on a thin 60° delta-wing—fuselage model equipped with double slotted flaps having spoilers mounted on the flap. The wing had an aspect ratio of 2.31, taper ratio of 0, sweep of 60° at the leading edge, and NACA 65A003 airfoil sections parallel to the free stream.

Results of the investigation, without discussion, are presented in the form of lateral aerodynamic characteristics of the plain-wing and double-slotted-flap configurations and in the form of normal-force coefficients and pitching-moment coefficients of the wing, vane, flap, and spoiler; sample pressure plots; sample span-load distributions; tabulated pressure coefficients; and tabulated section normal-force coefficients and section pitching-moment coefficients.

INTRODUCTION

Interest is being shown in the use of delta wings for high-speed airplanes because this plan form has some desirable aerodynamic and structural characteristics. Results of previous investigations (for example, see refs. 1 and 2) indicate that, by employing double slotted flaps on a 60° delta wing, the angle of attack necessary to obtain a given lift coefficient is considerably reduced, thereby making the use of double slotted flaps desirable for the landing configuration. Reference 3 indicates that spoiler-type controls located on the flap of a 60° delta wing having double slotted flaps gave rolling-moment coefficients that varied fairly linearly with spoiler projection and were about the same magnitude for the flap-retracted or the flap-deflected condition. The present investigation was made in the Langley 300 MPH 7- by 10-foot

tunnel to determine the loads at several spanwise stations on a 60° delta-wing model having double slotted flaps with spoilers located on the flaps. This investigation is an extension of the investigation reported on in reference 4.

The present paper presents the results, without discussion, of the investigation in the form of lateral aerodynamic characteristics of the complete model with flaps extended and retracted, and in the form of normal-force and pitching-moment coefficients of the wing, vane, flap, and spoiler. Included are sample pressure plots and span-load distributions, tabulated pressure coefficients, and tabulated section normal-force and section pitching-moment coefficients.

SYMBOLS

b	wing span (based on theoretical tip), ft (fig. 1)
b_f	flap span (exposed span of one flap), ft
b_v	vane span (exposed span of one vane), ft
b_s	spoiler span (exposed span of spoiler), ft
c	chord, ft
$\bar{c}$	wing mean aerodynamic chord (based on theoretical tip), $\frac{2}{S}\int_0^{b/2} c_w^2\,dy$, ft (fig. 1)
c_{av}	average wing chord, ft
c_v	vane chord, ft
c_f	flap chord, ft
c_w	plain-wing chord, ft
c_s	spoiler chord, ft
c_F	fuselage length, ft
S	wing area (based on theoretical tip), sq ft (fig. 1)

S_v vane area, sq ft

S_f flap area, sq ft

S_s spoiler area, sq ft

C_L lift coefficient, $\frac{\text{Lift of model}}{q_o S}$

C_D drag coefficient, $\frac{\text{Drag of model}}{q_o S}$

ΔC_D Jet-boundary correction applied to drag coefficient

C_m pitching-moment coefficient of model referred to quarter mean aerodynamic chord, $\frac{\text{Pitching moment of model}}{q_o S \bar{c}}$

C_l rolling-moment coefficient due to spoilers

C_n yawing-moment coefficient due to spoilers

C_p pressure coefficient, $\frac{H_o - p}{q_o}$

h_s spoiler projection, ft

l_v distance from wing quarter chord to vane nose, measured parallel to vane chord, ft

l_f distance from wing quarter chord to flap nose measured parallel to flap chord, ft

l_s distance from wing quarter chord to spoiler hinge line measured parallel to spoiler chord, ft

x longitudinal distance, ft

x_v distance from vane nose to center of load on vane, ft

x_f distance from flap nose to center of load on flap, ft

x_s distance from spoiler hinge line to center of load on spoiler, ft

y lateral distance, ft

z vertical distance, ft

H_o free-stream total pressure, lb/sq ft

p local static pressure, lb/sq ft

q_o free-stream dynamic pressure, $\frac{\rho V_o^2}{2}$, lb/sq ft

ρ mass density of air, slugs/cu ft

V_o free-stream velocity, ft/sec

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

δ_v vane deflection (angle between vane chord line and wing chord line; positive direction, trailing edge down), deg (fig. 2)

δ_s spoiler deflection (angle between spoiler chord and wing chord line; positive direction, trailing edge down), deg

α angle of attack set in tunnel

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

α_c corrected angle of attack

$c_{n,v}$ vane section normal-force coefficient, based on vane chord

$c_{n,f}$ flap section normal-force coefficient, based on flap chord

$c_{n,s}$ spoiler section normal-force coefficient, based on spoiler chord

$c_{n,WF}$ section normal-force coefficient of wing forward of slot lip, based on plain-wing chord

$c_{n,w}$ wing section normal-force coefficient, based on plain-wing chord (chord force of vane, flap, and spoiler neglected),

$$c_{n,WF} + c_{n,v}\left(\frac{c_v}{c_w}\right)\cos\delta_v + c_{n,f}\left(\frac{c_f}{c_w}\right)\cos\delta_f + c_{n,s}\left(\frac{c_s}{c_w}\right)\cos\delta_s$$

$c_{m,v}$ vane section pitching-moment coefficient, based on vane chord (moments taken about vane nose)

$c_{m,f}$ flap section pitching-moment coefficient, based on flap chord (moments taken about flap nose)

$c_{m,s}$ spoiler section pitching-moment coefficient, based on spoiler chord (moments taken about spoiler hinge line)

$c_{m,WF}$ section pitching-moment coefficient of wing forward of slot lip, 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 vane, flap, and spoiler neglected),

$$c_{m,WF} - \frac{c_{n,v}(l_v + x_v)c_v}{c_w^2} - \frac{c_{n,f}(l_f + x_f)c_f}{c_w^2} - \frac{c_{n,s}(l_s + x_s)c_s}{c_w^2}$$

$C_{N,w}$ wing normal-force coefficient (chord force of vane, flap, and spoiler neglected), $\frac{\text{Wing normal force}}{q_o S}$

$C_{N,v}$ vane normal-force coefficient, $\frac{\text{Vane normal force}}{q_o S_v}$

$C_{N,f}$ flap normal-force coefficient, $\frac{\text{Flap normal force}}{q_o S_f}$

$C_{N,s}$ spoiler normal-force coefficient, $\frac{\text{Spoiler normal force}}{q_o S_s}$

$C_{m,w}$ pitching-moment coefficient of wing, referred to quarter mean aerodynamic chord (chord force of vane, flap, and spoiler neglected), $\frac{\text{Pitching moment of wing}}{q_o S \bar{c}}$

$C_{m,v}$ pitching-moment coefficient of vane, referred to vane nose, $\frac{\text{Pitching moment of vane}}{q_o S_v c_v}$

$C_{m,f}$ pitching-moment coefficient of flap, referred to flap nose, $\frac{\text{Pitching moment of flap}}{q_o S_f c_f}$

$C_{m,s}$ pitching-moment coefficient of spoiler, referred to spoiler hinge line, $\frac{\text{Pitching moment of spoiler}}{q_o S_s c_s}$

MODEL AND APPARATUS

The model was tested on the single-support-strut system in the Langley 300 MPH 7- by 10-foot tunnel. The geometric and physical characteristics of the wing-fuselage configuration are given in figure 1 and table I.

The wing of the model had a 60° apex angle, an aspect ratio of 2.31 (based on the theoretical tip), a taper ratio of 0, and an NACA 65A003 airfoil section parallel to the free stream.

The double-slotted-flap configuration used for this investigation is shown in figure 2. The general arrangement, that is, relation of flap to vane to wing, were obtained from preliminary explorative tests based on the information of the systematic investigations of references 2 and 5.

The flap which extended from the fuselage to $0.67b/2$ had a constant chord of 6.86 inches and exposed area equal to 12.78 percent of the total wing area. The flap leading edge was constructed to the ordinates given in table II. The vane had a constant chord of 1.768 inches and was constructed to the ordinates given in table III. The vane and flap were deflected as a unit about the pivot point shown in figure 2. The spoiler which extended from the fuselage to $0.67b/2$ had a constant chord of $0.10\bar{c}$ and was mounted with its hinge line perpendicular to the plane of symmetry. (See fig. 2.)

The wing, vane, flap, and spoiler were constructed with flush surface pressure orifices located on the right semispan at the 21-, 30-, 43-, and 55-percent-semispan stations. Orifices were also located on the wing at the 72-percent-semispan station and along the fuselage at the plane of symmetry.

TESTS

The tests were performed at a dynamic pressure of approximately 25 pounds per square foot which corresponds to a Mach number of approximately 0.13. Reynolds number based on the mean aerodynamic chord of the model was approximately 2.7×10^6. The tests were made through an angle-of-attack range of approximately -4° through the stall. Flap deflection for tests with the double-slotted-flap configuration was 60°. Force data were obtained on the plain-wing configuration with the spoiler hinged at the 70.0-, 89.4-, and 93.3-percent-root-chord location for several spoiler projections and on the double-slotted-flap configuration at the 89.4- and 93.3-percent-root-chord location for a spoiler projection of $-0.05\bar{c}$. Pressure distributions were obtained for the plain-wing

configuration and the double-slotted-flap configuration with the spoiler located at the 93.3-percent-root-chord location at spoiler deflections of -0.5-, -1.0-, -3.5-, -5.0-, and -10.0-percent $\bar{c}$.

CORRECTIONS

The following jet-boundary corrections applied to the data of this paper were obtained by the method outlined in reference 6:

$$\Delta\alpha = 1.028C_L$$

$$\Delta C_D = 0.0179C_L^2$$

The blockage correction as applied to the dynamic pressure was obtained by the method outlined in reference 7. The buoyancy correction due to the longitudinal static pressure gradient in the tunnel as applied to the data increased the drag coefficient by 0.001.

RESULTS

The results of this investigation are presented without discussion. Sample data figures are presented in order to give the reader a general idea of the chordwise and spanwise loadings to expect over a delta wing that is equipped with double slotted flaps when spoilers are used on the flap for lateral control.

The model aerodynamic characteristics in pitch for the plain-wing and double-slotted-flap configuration have been presented in reference 1. The pressure-distribution data of the plain-wing and double-slotted-flap configurations without spoilers have been presented in reference 4. The data in the form of figures and tables are as listed below.

The pressure coefficients are presented in tables IV to XIII.

The section data are presented in tables XIV and XV.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., November 23, 1956.

REFERENCES

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2. Riebe, John M., and MacLeod, Richard G.: Low-Speed Wind-Tunnel Investigation of a Thin 60° Delta Wing With Double Slotted, Single Slotted, Plain, and Split Flaps. NACA RM L52J29, 1953.

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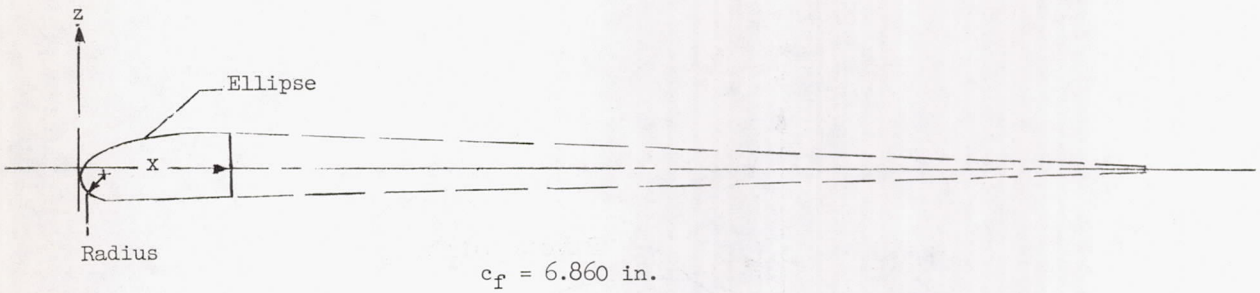
TABLE I.- PHYSICAL CHARACTERISTICS OF TEST MODEL

Wing:	
Section parallel to free stream	NACA 65A003
Span, ft	5.00
Aspect ratio (based on theoretical tip)	2.31
Leading-edge sweep, deg	60.00
Trailing-edge sweep, deg	0
Area (based on theoretical tip), sq ft	10.83
Mean aerodynamic chord, ft	2.89
Root chord, ft	4.33
Vane:	
Span, ft	3.33
Chord, ft	0.15
Chord, percent wing root chord	3.40
Chord, percent flap chord	25.77
Flap:	
Span, ft	3.33
Chord, ft	0.57
Chord, percent wing root chord	13.20
Exposed area, sq ft	1.38
Exposed area, percent wing area	12.78
Spoiler:	
Span, ft	3.33
Chord, ft	0.29
Chord, percent mean aerodynamic chord	10.0

TABLE II.- ORDINATES OF THE LEADING EDGE OF THE TRAILING-EDGE FLAP

[All dimensions in inches]

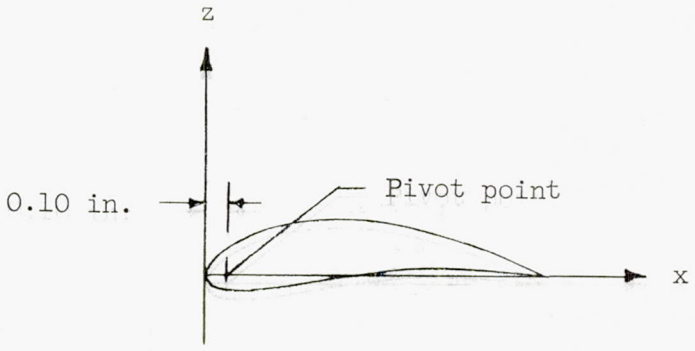


c_f = 6.860 in.

Station, in. x	Lower z	Upper z
0	-0.107	-0.107
.010	Radius	-.059
.030	↓	-.030
.050	↓	-.005
.075	↓	.022
.100	↓	.041
.143	-.245	------
.200	Straight taper	.096
.300	↓	.133
.400	↓	.161
.500	↓	.182
.600	↓	.198
.700	↓	.208
.800	↓	.215
.900	↓	.217
1.000	-.216	.216

TABLE III.- ORDINATES OF THE VANE

[All dimensions in inches]



c_v = 1.768 in.

Station, in. x	Lower z	Upper z
0	0	0
.022	-.047	.067
.044	-.060	.092
.088	-.072	.131
.133	-.079	.160
.177	-.077	.185
.265	-.072	.224
.354	-.053	.255
.530	-.025	.288
.707	.002	.294
.884	.032	.283
1.061	.053	.255
1.238	.057	.207
1.414	.053	.147
1.591	.032	.080
1.680	.019	.046
1.768	0	0

TABLE IV.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE

(a) Plain wing configuration with spoiler; $h_s = -0.005\bar{c}$; $\frac{y}{b/2} = 0$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Fuselage										
.0000	.003	.000	.000	.000	.000	.019	.067	.099	.108	.122
.0500	.742	.837	.883	.953	.978	1.034	1.064	1.051	1.081	1.095
.1000	.824	.928	.963	1.025	1.034	1.094	1.113	1.102	1.151	1.153
.1500	.934	1.000	1.015	1.075	1.083	1.125	1.143	1.135	1.157	1.156
.2000	.972	1.041	1.061	1.110	1.099	1.138	1.134	1.123	1.142	1.144
.2500	1.016	1.082	1.092	1.122	1.111	1.159	1.146	1.132	1.154	1.159
.3000	1.050	1.091	1.092	1.122	1.105	1.134	1.113	1.093	1.136	1.131
.3500	1.050	1.088	1.073	1.104	1.089	1.109	1.101	1.090	1.123	1.138
.4000	1.022	1.044	1.037	1.063	1.062	1.088	1.101	1.105	1.142	1.156
.4500	1.003	1.047	1.043	1.075	1.092	1.141	1.183	1.189	1.244	1.260
.5000	.997	1.047	1.055	1.119	1.148	1.228	1.284	1.303	1.349	1.413
.5500	.994	1.056	1.083	1.160	1.194	1.284	1.320	1.351	1.413	1.544
.6000	1.003	1.063	1.095	1.176	1.210	1.278	1.320	1.309	1.361	1.547
.6500	1.000	1.069	1.101	1.169	1.191	1.250	1.265	1.246	1.289	1.419
.7000	1.006	1.069	1.095	1.151	1.173	1.222	1.223	1.216	1.256	1.346
.7500	1.016	1.066	1.086	1.135	1.148	1.191	1.207	1.213	1.253	1.315
.8000	1.022	1.069	1.073	1.107	1.120	1.175	1.198	1.219	1.274	1.294
.8500	1.028	1.075	1.061	1.097	1.102	1.153	1.186	1.207	1.250	1.254
.9000	1.035	1.050	1.043	1.066	1.058	1.109	1.125	1.135	1.190	1.193
.9500	1.028	1.072	1.055	1.066	1.065	1.097	1.101	1.105	1.151	1.147
.9940	1.101	1.119	1.104	1.119	1.096	1.109	1.104	1.099	1.117	1.104

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Fuselage										
.0500	.840	.850	.782	.718	.633	.584	.509	.450	.434	.385
.1000	.918	.944	.865	.831	.744	.688	.607	.559	.548	.498
.1500	1.000	1.009	.932	.906	.815	.766	.683	.646	.620	.569
.2000	1.006	1.044	.978	.966	.883	.841	.744	.712	.687	.648
.2500	1.047	1.069	1.018	.994	.923	.894	.817	.766	.756	.697
.3000	1.069	1.082	1.040	1.028	.954	.922	.848	.802	.792	.737
.3500	1.054	1.072	1.040	1.035	.972	.944	.875	.826	.822	.765
.4000	.978	1.009	.969	.969	.926	.894	.817	.772	.765	.713
.4500	.966	.991	.948	.941	.870	.847	.756	.715	.702	.657
.5000	.950	.975	.920	.903	.824	.784	.701	.652	.633	.575
.5500	.912	.962	.899	.865	.784	.741	.643	.598	.578	.532
.6000	.918	.925	.850	.821	.738	.684	.607	.547	.542	.486
.7500	1.198	1.194	1.135	1.110	1.046	1.016	.939	.886	.889	.844
.9500	.991	1.025	.994	.997	.969	.966	.933	.901	.907	.887
.9940	.994	1.082	1.067	1.116	1.089	1.113	1.092	1.075	1.096	1.073

TABLE IV.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(b) Plain wing configuration with spoiler; $h_s = -0.005\bar{c}$; $\frac{y}{b/2} = 0.21$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0000	1.322	.827	1.238	2.592	4.490	6.828	9.400	10.402	8.604	4.103
.0125	.792	1.055	1.411	2.222	5.965	9.624	12.325	12.089	8.348	3.742
.0250	.830	1.055	1.317	1.826	3.058	5.113	8.000	9.138	7.559	3.742
.0500	.887	1.074	1.270	1.564	1.797	2.420	3.397	4.569	5.376	3.351
.0750	.902	1.058	1.232	1.462	1.665	2.003	2.578	3.153	4.117	3.218
.1000	.924	1.065	1.235	1.418	1.598	1.862	2.347	2.783	3.571	3.124
.1500	.947	1.062	1.210	1.370	1.491	1.687	2.056	2.314	2.808	2.927
.2000	.965	1.086	1.204	1.326	1.451	1.593	1.891	2.112	2.442	2.748
.2500	.978	1.080	1.198	1.307	1.386	1.524	1.797	1.951	2.155	2.497
.3000	.991	1.096	1.191	1.298	1.356	1.483	1.728	1.814	1.984	2.297
.3500	.988	1.083	1.185	1.257	1.316	1.439	1.597	1.708	1.849	2.121
.4000	.994	1.086	1.166	1.250	1.282	1.417	1.538	1.621	1.751	1.985
.4500	1.003	1.089	1.172	1.241	1.276	1.383	1.484	1.565	1.685	1.873
.5000	1.029	1.099	1.176	1.225	1.270	1.345	1.459	1.534	1.637	1.827
.5500	1.022	1.086	1.154	1.190	1.218	1.288	1.409	1.460	1.559	1.706
.6000	1.029	1.080	1.144	1.177	1.205	1.267	1.381	1.466	1.559	1.663
.6500	1.013	1.062	1.135	1.158	1.184	1.226	1.369	1.457	1.546	1.612
.7000	1.006	1.037	1.107	1.133	1.159	1.201	1.353	1.423	1.483	1.542
.8000	.997	1.028	1.091	1.098	1.110	1.182	1.266	1.295	1.338	1.430
.8500	.962	1.012	1.053	1.076	1.073	1.141	1.175	1.180	1.234	1.351
.9000	.924	.969	.997	1.013	1.024	1.063	1.063	1.068	1.136	1.270
.9500	1.085	1.123	1.138	1.181	1.178	1.210	1.234	1.258	1.259	1.245
Spoiler										
.2429	.852	1.031	1.031	1.066	1.058	1.097	1.052	1.030	1.084	1.156
.4852	.805	.994	1.018	1.047	1.037	1.066	1.018	1.003	1.039	1.113
.7367	.821	1.025	1.037	1.072	1.055	1.078	1.018	1.018	1.039	1.107
.9790	.959	1.132	1.141	1.163	1.123	1.172	1.110	1.114	1.154	1.199

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0125	1.205	.988	.806	.649	.586	.621	.697	.752	.704	.506
.0250	1.167	.984	.828	.652	.509	.426	.356	.354	.312	.258
.0500	1.114	.994	.872	.693	.537	.382	.284	.255	.230	.236
.0750	1.111	1.006	.890	.737	.577	.439	.322	.314	.284	.297
.1000	1.129	1.012	.909	.756	.607	.489	.391	.379	.341	.370
.1500	1.104	1.018	.941	.798	.653	.561	.472	.469	.432	.451
.2000	1.111	1.052	.972	.845	.705	.627	.544	.531	.495	.491
.2500	1.136	1.086	1.019	.896	.767	.683	.613	.596	.562	.551
.3000	1.133	1.099	1.053	.943	.810	.734	.659	.646	.606	.582
.3500	1.145	1.151	1.097	1.003	.865	.803	.716	.699	.666	.636
.4000	1.174	1.176	1.129	1.044	.914	.843	.772	.742	.700	.673
.4500	1.230	1.197	1.154	1.060	.942	.884	.800	.783	.741	.709
.5000	1.218	1.188	1.154	1.063	.966	.900	.828	.808	.770	.742
.5500	1.177	1.170	1.163	1.063	.994	.925	.822	.817	.786	.758
.6000	1.218	1.179	1.144	1.063	.988	.925	.866	.848	.808	.788
.6500	1.177	1.154	1.116	1.057	.978	.922	.863	.845	.805	.800
.7000	1.158	1.136	1.116	1.054	.978	.928	.884	.860	.833	.812
.8000	1.120	1.111	1.104	1.057	.988	.953	.922	.901	.877	.873
.8500	1.107	1.105	1.097	1.048	.997	.972	.944	.922	.905	.909
.9000	1.054	1.068	1.066	1.032	.975	.956	.938	.926	.915	.915
.9500	1.019	1.031	1.016	.978	.932	.915	.903	.895	.896	.927
Spoiler										
.2573	.632	.768	.788	.834	.867	.809	.784	.793	.825	.905
.4984	.673	.834	.840	.893	.904	.850	.838	.835	.876	.948
.7485	1.016	1.169	1.165	1.194	1.167	1.213	1.189	1.177	1.211	1.220
.9940	1.076	1.207	1.184	1.204	1.173	1.228	1.201	1.195	1.226	1.254

TABLE IV.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(c) Plain wing configuration with spoiler; $h_s = -0.005\bar{c}$; $\frac{y}{b/2} = 0.30$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.855	.898	1.571	2.418	3.125	4.088	4.553	4.544	4.064	3.106
.0125	.808	1.126	1.778	2.450	3.064	3.740	4.034	4.140	3.975	3.100
.0250	.855	1.111	1.781	2.545	3.174	3.891	4.150	4.134	3.975	3.100
.0500	.909	1.108	1.671	3.282	3.490	3.809	4.006	4.153	4.019	3.106
.0750	.928	1.092	1.301	2.763	4.542	5.126	4.634	4.476	4.076	3.115
.1000	.931	1.086	1.213	1.658	3.349	4.630	4.609	4.407	3.975	3.109
.1500	.947	1.089	1.198	1.206	1.748	2.862	3.878	3.904	3.710	3.042
.2000	.978	1.096	1.216	1.250	1.386	2.047	2.994	3.236	3.297	2.921
.2500	.956	1.083	1.194	1.260	1.307	1.699	2.472	2.739	2.953	2.797
.3000	.981	1.089	1.198	1.250	1.294	1.568	2.184	2.451	2.701	2.679
.3500	.994	1.089	1.188	1.225	1.267	1.464	1.934	2.177	2.439	2.515
.4000	.994	1.092	1.191	1.231	1.257	1.433	1.813	2.019	2.287	2.400
.4500	1.016	1.099	1.185	1.222	1.251	1.423	1.756	1.960	2.202	2.348
.5000	1.029	1.092	1.169	1.206	1.239	1.376	1.666	1.811	2.010	2.176
.5500	1.003	1.077	1.166	1.190	1.205	1.351	1.600	1.721	1.912	2.069
.6000	1.006	1.080	1.154	1.177	1.190	1.329	1.544	1.652	1.830	1.976
.6500	1.013	1.074	1.147	1.168	1.178	1.304	1.497	1.590	1.760	1.888
.7500	.991	1.031	1.091	1.101	1.107	1.210	1.369	1.488	1.609	1.727
.8000	.947	1.012	1.056	1.076	1.083	1.169	1.328	1.441	1.536	1.639
.8500	.937	.984	1.035	1.041	1.043	1.113	1.284	1.376	1.451	1.539
.9000	.849	.901	.956	.956	.988	1.009	1.150	1.239	1.284	1.403
.9500	1.066	1.099	1.138	1.196	1.236	1.304	1.363	1.388	1.417	1.430
Spoiler										
.2475	.871	1.025	1.024	1.060	1.058	1.106	1.189	1.249	1.292	1.352
.4909	.884	1.041	1.037	1.066	1.049	1.091	1.171	1.222	1.259	1.327
.7396	.896	1.060	1.046	1.097	1.055	1.091	1.156	1.177	1.214	1.263
.9796	1.019	1.213	1.202	1.216	1.170	1.241	1.262	1.273	1.307	1.352

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.249	.957	.796	.772	.874	1.078	1.259	1.336	1.385	1.285
.0250	1.278	1.000	.840	.725	.705	.771	.841	.873	.905	.870
.0500	1.224	1.015	.884	.737	.647	.627	.606	.596	.599	.594
.0750	1.180	1.040	.922	.779	.662	.596	.550	.531	.514	.515
.1000	1.183	1.049	.944	.807	.681	.589	.534	.519	.483	.485
.1500	1.177	1.080	.994	.855	.721	.646	.566	.544	.505	.503
.2000	1.193	1.105	1.035	.908	.773	.674	.594	.553	.517	.494
.2500	1.205	1.126	1.069	.946	.816	.724	.631	.596	.543	.527
.3000	1.224	1.148	1.097	.975	.843	.756	.678	.631	.596	.561
.3500	1.237	1.170	1.116	1.003	.880	.796	.716	.671	.631	.600
.4000	1.243	1.173	1.125	1.029	.914	.825	.753	.711	.663	.648
.4500	1.256	1.182	1.144	1.044	.932	.859	.788	.745	.710	.682
.5000	1.224	1.173	1.144	1.051	.951	.890	.809	.777	.735	.706
.5500	1.202	1.179	1.154	1.076	.969	.919	.850	.811	.776	.745
.6000	1.175	1.148	1.125	1.063	.966	.919	.859	.823	.792	.764
.6500	1.142	1.133	1.104	1.067	.975	.928	.872	.832	.817	.791
.7500	1.060	1.096	1.088	1.051	.972	.944	.903	.879	.858	.839
.8000	1.019	1.077	1.072	1.041	.972	.956	.913	.891	.883	.857
.8500	1.060	1.080	1.075	1.054	.991	.972	.941	.926	.924	.912
.9000	1.041	1.074	1.075	1.060	1.009	.991	.975	.966	.965	.967
.9500	1.079	1.096	1.100	1.092	1.040	1.035	1.050	1.044	1.054	1.073
Spoiler										
.2587	.629	.762	.785	.831	.855	.834	.854	.928	.979	1.058
.5024	.711	.850	.868	.906	.923	.900	.921	.970	1.021	1.098
.7531	1.060	1.220	1.196	1.210	1.160	1.244	1.256	1.267	1.295	1.358
.9940	1.079	1.235	1.205	1.226	1.160	1.272	1.274	1.285	1.337	1.388

TABLE IV.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(d) Plain wing configuration with spoiler; $h_s = -0.005\bar{c}$; $\frac{y}{b/2} = 0.43$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.508	.907	1.668	1.994	2.457	2.712	2.681	2.680	2.716	2.651
.0125	.776	1.099	1.649	1.984	2.423	2.687	2.763	2.780	2.789	2.663
.0250	.830	1.105	1.593	2.007	2.420	2.696	2.875	2.839	2.808	2.657
.0500	.871	1.086	1.539	2.073	2.490	2.793	2.822	2.811	2.836	2.651
.0750	.896	1.092	1.583	2.130	2.490	2.706	2.803	2.839	2.849	2.645
.1000	.915	1.083	1.511	2.336	2.447	2.627	2.788	2.870	2.868	2.657
.1500	.915	1.071	1.320	2.500	2.950	2.768	2.897	2.954	2.903	2.663
.2000	.947	1.077	1.201	1.988	3.187	3.201	3.003	2.976	2.874	2.648
.2500	.953	1.083	1.166	1.459	2.530	3.050	2.894	2.845	2.770	2.591
.3000	.984	1.102	1.179	1.260	2.021	2.840	2.816	2.802	2.748	2.563
.3500	1.000	1.108	1.160	1.158	1.592	2.433	2.594	2.597	2.612	2.476
.4000	.997	1.102	1.163	1.165	1.389	2.147	2.453	2.503	2.540	2.418
.4500	1.016	1.108	1.169	1.171	1.303	1.944	2.313	2.373	2.451	2.360
.5000	.965	1.086	1.135	1.152	1.199	1.712	2.116	2.190	2.303	2.260
.5500	.991	1.089	1.138	1.168	1.196	1.611	2.003	2.103	2.227	2.194
.6000	1.010	1.086	1.132	1.158	1.162	1.517	1.913	1.997	2.130	2.139
.7000	.984	1.049	1.088	1.114	1.116	1.376	1.728	1.826	1.981	2.009
.7500	.994	1.046	1.075	1.089	1.098	1.320	1.663	1.761	1.902	1.970
.8000	.981	1.025	1.031	1.051	1.067	1.254	1.578	1.690	1.824	1.918
.8500	.937	.984	.994	1.000	1.015	1.179	1.478	1.581	1.742	1.860
.9000	1.085	1.114	1.119	1.234	1.300	1.376	1.491	1.513	1.578	1.585
.9500	1.088	1.114	1.116	1.238	1.300	1.379	1.488	1.510	1.587	1.579
Spoiler										
.2443	.877	1.038	1.049	1.072	1.058	1.166	1.348	1.447	1.563	1.700
.4906	.868	1.013	1.037	1.069	1.049	1.134	1.290	1.387	1.506	1.636
.7370	.881	1.035	1.052	1.069	1.055	1.131	1.256	1.348	1.458	1.584
.9856	.962	1.198	1.218	1.245	1.213	1.250	1.332	1.378	1.440	1.557

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.587	1.062	.831	.842	.957	1.141	1.288	1.342	1.426	1.421
.0250	1.514	1.092	.893	.798	.788	.840	.906	.954	.997	.985
.0500	1.492	1.108	.937	.829	.730	.699	.719	.721	.748	.730
.0750	1.404	1.117	.959	.851	.733	.677	.666	.649	.659	.639
.1000	1.331	1.129	.981	.880	.745	.683	.641	.618	.625	.591
.1500	1.278	1.133	1.016	.899	.773	.705	.644	.609	.599	.564
.2000	1.265	1.148	1.035	.931	.800	.718	.666	.618	.615	.582
.2500	1.240	1.157	1.063	.984	.831	.749	.691	.649	.628	.594
.3000	1.237	1.163	1.066	.978	.846	.784	.722	.668	.637	.606
.3500	1.243	1.173	1.091	1.025	.892	.815	.750	.714	.681	.651
.4000	1.253	1.176	1.113	1.022	.911	.846	.778	.736	.716	.673
.4500	1.230	1.173	1.100	1.038	.917	.853	.800	.752	.735	.688
.5000	1.212	1.170	1.113	1.044	.929	.868	.828	.780	.767	.715
.5500	1.205	1.160	1.100	1.051	.942	.890	.838	.814	.792	.754
.6000	1.183	1.151	1.094	1.054	.951	.903	.859	.826	.820	.782
.7000	1.161	1.142	1.107	1.063	.978	.950	.916	.891	.887	.848
.7500	1.145	1.123	1.097	1.073	.988	.966	.947	.913	.915	.879
.8000	1.107	1.111	1.088	1.063	.985	.972	.966	.938	.959	.918
.8500	1.101	1.096	1.078	1.060	.997	.994	.975	.997	.997	.982
.9000	1.092	1.092	1.085	1.073	1.021	1.025	1.034	1.016	1.054	1.051
.9500	1.111	1.126	1.113	1.111	1.067	1.091	1.138	1.127	1.196	1.215
Spoiler										
.2587	.604	.712	.758	.771	.753	.919	1.067	1.159	1.313	1.465
.5039	.708	.837	.862	.890	.858	1.000	1.131	1.213	1.349	1.492
.7508	1.000	1.216	1.224	1.245	1.216	1.256	1.326	1.348	1.431	1.535
.9940	1.076	1.254	1.236	1.263	1.231	1.272	1.354	1.378	1.455	1.566

TABLE IV.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(e) Plain wing configuration with spoiler; $h_s = -0.005\bar{c}$; $\frac{y}{b/2} = 0.55$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0000	1.243	.950	1.649	1.795	2.073	2.204	2.188	2.199	2.284	2.239
.0125	.836	1.151	1.633	1.839	2.110	2.213	2.231	2.277	2.341	2.269
.0250	.827	1.102	1.640	1.817	2.076	2.179	2.253	2.277	2.338	2.263
.0500	.858	1.099	1.542	1.829	2.064	2.210	2.284	2.302	2.354	2.251
.0750	.902	1.096	1.433	1.861	2.070	2.245	2.266	2.314	2.360	2.251
.1000	.915	1.105	1.364	1.886	2.110	2.267	2.278	2.330	2.360	2.239
.1500	.937	1.102	1.314	1.905	2.082	2.169	2.303	2.339	2.354	2.215
.2000	.962	1.096	1.285	2.032	2.033	2.138	2.319	2.342	2.354	2.227
.2500	.972	1.105	1.273	2.234	2.073	2.138	2.356	2.357	2.369	2.227
.3000	.997	1.123	1.273	2.285	2.365	2.232	2.378	2.370	2.373	2.239
.3500	1.006	1.111	1.245	2.098	2.527	2.336	2.356	2.345	2.354	2.227
.4000	1.016	1.117	1.235	1.829	2.454	2.358	2.300	2.308	2.319	2.200
.4500	1.019	1.111	1.213	1.589	2.300	2.332	2.259	2.267	2.291	2.185
.5000	1.032	1.114	1.210	1.380	2.110	2.267	2.213	2.224	2.256	2.154
.6000	1.006	1.077	1.154	1.133	1.721	2.041	2.069	2.084	2.152	2.091
.6500	1.019	1.074	1.141	1.089	1.576	1.950	2.003	2.041	2.104	2.063
.7000	1.006	1.065	1.125	1.057	1.454	1.850	1.931	1.991	2.057	2.030
.7500	1.006	1.049	1.097	1.032	1.362	1.762	1.878	1.926	2.010	1.997
.8500	.915	.941	.969	.893	1.098	1.514	1.684	1.724	1.855	1.924
.9000	1.095	1.114	1.122	1.317	1.445	1.524	1.609	1.631	1.710	1.727
.9500	1.104	1.114	1.141	1.298	1.417	1.489	1.584	1.600	1.682	1.691
Spoiler										
.2443	.837	1.019	1.049	1.031	1.123	1.431	1.610	1.682	1.762	1.859
.4881	.859	1.016	1.055	1.028	1.089	1.375	1.549	1.610	1.717	1.823
.7416	.859	1.031	1.058	1.053	1.086	1.325	1.482	1.550	1.660	1.777
.9856	.991	1.204	1.221	1.263	1.281	1.409	1.506	1.571	1.666	1.792

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0125	1.735	1.136	.862	.915	1.077	1.251	1.394	1.463	1.600	1.600
.0250	1.704	1.148	.912	.848	.859	.934	1.003	1.031	1.126	1.130
.0500	1.691	1.160	.959	.861	.776	.781	.809	.801	.842	.833
.0750	1.521	1.139	.969	.864	.748	.721	.706	.699	.748	.718
.1000	1.461	1.154	.997	.880	.751	.715	.694	.671	.697	.658
.1500	1.369	1.151	1.016	.899	.770	.730	.706	.658	.669	.624
.2000	1.338	1.160	1.044	.931	.797	.749	.697	.668	.663	.624
.2500	1.328	1.170	1.050	.965	.831	.759	.713	.674	.675	.639
.3000	1.309	1.160	1.072	.978	.850	.790	.738	.686	.688	.661
.3500	1.290	1.167	1.085	1.010	.883	.821	.766	.724	.716	.676
.4000	1.284	1.167	1.097	1.022	.905	.846	.797	.742	.754	.703
.4500	1.253	1.160	1.091	1.022	.917	.862	.819	.773	.767	.718
.5000	1.246	1.167	1.097	1.035	.929	.884	.834	.798	.789	.742
.6000	1.221	1.148	1.100	1.060	.960	.931	.897	.845	.852	.797
.6500	1.212	1.154	1.107	1.067	.978	.944	.925	.888	.893	.851
.7000	1.183	1.129	1.091	1.057	.981	.966	.934	.910	.915	.888
.7500	1.174	1.117	1.100	1.063	1.003	.988	.972	.947	.947	.933
.8000	1.152	1.117	1.097	1.079	1.018	1.013	1.009	.981	1.003	.985
.8500	1.145	1.114	1.094	1.086	1.037	1.047	1.053	1.041	1.070	1.057
.9000	1.126	1.111	1.094	1.101	1.058	1.094	1.122	1.106	1.155	1.151
.9500	1.136	1.117	1.129	1.111	1.092	1.160	1.231	1.230	1.303	1.327
Spoiler										
.2584	.616	.752	.791	.743	.886	1.225	1.424	1.474	1.566	1.700
.5039	.777	.956	.994	.997	1.111	1.331	1.457	1.508	1.590	1.716
.7508	.988	1.220	1.224	1.270	1.324	1.425	1.494	1.541	1.623	1.743
.9940	.991	1.232	1.254	1.292	1.355	1.450	1.509	1.559	1.651	1.774

TABLE IV.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Concluded

(f) Plain wing configuration with spoiler; $h_s = -0.005\bar{c}$; $\frac{y}{b/2} = 0.72$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0000	1.585	.897	1.635	1.665	1.734	1.813	1.793	1.793	1.864	1.871
.0125	.764	1.063	1.610	1.633	1.688	1.772	1.738	1.766	1.846	1.865
.0250	.780	1.085	1.610	1.643	1.697	1.775	1.768	1.778	1.867	1.871
.0500	.833	1.100	1.644	1.652	1.697	1.784	1.790	1.823	1.883	1.887
.0750	.865	1.100	1.665	1.674	1.694	1.809	1.811	1.829	1.898	1.887
.1000	.855	1.082	1.546	1.643	1.666	1.797	1.802	1.820	1.883	1.881
.1500	.893	1.088	1.423	1.646	1.670	1.806	1.811	1.826	1.901	1.893
.2000	.940	1.100	1.362	1.668	1.691	1.831	1.832	1.850	1.907	1.881
.2500	.956	1.110	1.294	1.665	1.691	1.809	1.829	1.844	1.895	1.868
.3000	.959	1.113	1.251	1.655	1.676	1.797	1.826	1.838	1.886	1.862
.3500	.994	1.113	1.236	1.668	1.660	1.794	1.826	1.841	1.879	1.853
.4000	.994	1.122	1.221	1.655	1.645	1.781	1.817	1.826	1.864	1.841
.4500	1.006	1.119	1.196	1.683	1.629	1.763	1.808	1.808	1.855	1.838
.5000	1.016	1.125	1.184	1.749	1.648	1.759	1.805	1.796	1.849	1.832
.5500	1.003	1.107	1.159	1.765	1.645	1.744	1.784	1.790	1.828	1.813
.6000	.991	1.104	1.141	1.752	1.648	1.731	1.765	1.766	1.813	1.807
.6500	.994	1.094	1.132	1.737	1.657	1.713	1.750	1.748	1.795	1.804
.7000	1.006	1.088	1.126	1.749	1.666	1.722	1.738	1.733	1.786	1.792
.7500	1.019	1.091	1.107	1.715	1.666	1.713	1.729	1.721	1.771	1.777
.8000	1.013	1.085	1.098	1.665	1.666	1.700	1.701	1.700	1.756	1.777
.8500	1.016	1.063	1.080	1.602	1.639	1.678	1.677	1.670	1.726	1.761
.9000	.997	1.056	1.061	1.483	1.586	1.644	1.628	1.631	1.693	1.743

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0125	1.673	1.201	.813	.828	.892	1.009	1.076	1.129	1.202	1.260
.0250	1.676	1.191	.834	.793	.784	.869	.906	.937	.988	1.024
.0500	1.632	1.163	.877	.790	.734	.756	.753	.781	.816	.835
.0750	1.585	1.201	.917	.821	.734	.734	.710	.730	.744	.758
.1000	1.554	1.191	.923	.828	.738	.725	.695	.694	.711	.703
.1500	1.406	1.157	.948	.859	.756	.734	.671	.667	.666	.667
.2000	1.362	1.154	.960	.884	.784	.753	.695	.682	.687	.667
.2500	1.315	1.141	.975	.900	.809	.769	.704	.697	.693	.679
.3000	1.283	1.144	.994	.925	.827	.800	.738	.721	.717	.703
.3500	1.258	1.141	.997	.937	.855	.825	.756	.745	.741	.719
.4000	1.167	1.129	1.006	.962	.873	.844	.781	.757	.762	.740
.4500	1.154	1.129	1.012	.978	.892	.866	.802	.781	.783	.774
.5000	1.142	1.125	1.018	.988	.913	.878	.826	.802	.807	.777
.5500	1.101	1.125	1.024	.994	.923	.900	.845	.820	.810	.798
.6000	1.098	1.122	1.031	1.022	.947	.931	.866	.847	.846	.829
.6500	1.088	1.116	1.031	1.016	.966	.953	.896	.877	.873	.868
.7000	1.076	1.091	1.021	1.016	.981	.966	.918	.901	.913	.896
.7500	1.069	1.082	1.015	1.028	.997	1.006	.960	.943	.952	.942
.8000	1.041	1.069	1.012	1.041	1.028	1.031	1.003	.985	1.000	1.003
.8500	1.016	1.069	1.015	1.053	1.058	1.091	1.064	1.045	1.069	1.076
.9000	.997	1.056	1.012	1.078	1.108	1.147	1.125	1.123	1.154	1.180
.9500	.975	1.028	.994	1.113	1.185	1.250	1.223	1.225	1.262	1.300

TABLE V.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE

(a) Plain wing configuration with spoiler; $h_s = -0.010\bar{c}$; $\frac{y}{b/2} = 0$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Fuselage										
.0000	.000	.006	.000	.015	.000	.015	.064	.077	.107	.142
.0500	.701	.841	.896	.944	.969	1.043	1.073	1.092	1.088	1.091
.1000	.774	.925	.978	1.006	1.024	1.089	1.122	1.129	1.146	1.148
.1500	.901	1.006	1.038	1.059	1.070	1.117	1.143	1.156	1.159	1.154
.2000	.933	1.050	1.079	1.099	1.080	1.126	1.149	1.147	1.143	1.133
.2500	.981	1.087	1.110	1.111	1.101	1.151	1.162	1.165	1.153	1.148
.3000	1.035	1.109	1.110	1.105	1.098	1.129	1.134	1.132	1.125	1.142
.3500	1.035	1.084	1.101	1.096	1.073	1.102	1.119	1.126	1.119	1.148
.4000	.994	1.037	1.060	1.053	1.052	1.074	1.107	1.116	1.131	1.157
.4500	.994	1.037	1.060	1.068	1.067	1.129	1.183	1.202	1.241	1.278
.5000	.984	1.034	1.069	1.102	1.138	1.215	1.284	1.319	1.342	1.450
.5500	.971	1.044	1.098	1.152	1.184	1.265	1.345	1.374	1.412	1.607
.6000	.987	1.059	1.113	1.180	1.193	1.255	1.320	1.346	1.369	1.625
.6500	.994	1.068	1.120	1.152	1.165	1.228	1.262	1.273	1.296	1.477
.7000	.997	1.065	1.116	1.136	1.144	1.203	1.229	1.221	1.247	1.393
.7500	1.000	1.050	1.098	1.105	1.123	1.172	1.210	1.236	1.253	1.350
.8000	1.010	1.056	1.091	1.096	1.098	1.148	1.207	1.242	1.253	1.302
.8500	1.019	1.068	1.076	1.087	1.077	1.135	1.201	1.221	1.235	1.260
.9000	1.013	1.056	1.076	1.056	1.043	1.092	1.143	1.172	1.168	1.193
.9500	1.006	1.081	1.076	1.068	1.052	1.089	1.055	1.147	1.143	1.139
.9940	1.077	1.134	1.129	1.111	1.080	1.098	1.107	1.138	1.119	1.118

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Fuselage										
.0500	.793	.850	.802	.718	.626	.575	.500	.482	.430	.390
.1000	.854	.963	.899	.842	.745	.705	.628	.586	.543	.498
.1500	.949	1.019	.959	.913	.816	.782	.701	.672	.622	.571
.2000	.975	1.031	1.003	.941	.865	.837	.771	.739	.686	.634
.2500	.978	1.075	1.041	.994	.917	.880	.823	.794	.744	.698
.3000	1.038	1.081	1.060	1.006	.951	.932	.866	.834	.787	.737
.3500	1.016	1.068	1.050	1.006	.960	.935	.887	.883	.811	.752
.4000	.943	1.006	.994	.957	.911	.889	.835	.804	.759	.710
.4500	.914	.997	.978	.929	.871	.855	.774	.758	.710	.656
.5000	.901	.984	.947	.895	.828	.785	.717	.675	.631	.583
.5500	.854	.969	.931	.873	.785	.748	.668	.629	.582	.535
.6000	.870	.935	.887	.824	.739	.698	.622	.577	.540	.498
.7500	1.153	1.199	1.167	1.108	1.037	1.025	.951	.926	.896	.837
.9500	.962	1.015	1.019	.988	.957	.957	.945	.935	.906	.891
.9940	.959	1.068	1.101	1.096	1.083	1.111	1.098	1.104	1.088	1.072

TABLE V.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(b) Plain wing configuration with spoiler; $h_s = -0.010\bar{c}$; $\frac{y}{b/2} = 0.21$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.344	.830	1.275	2.743	4.551	6.963	9.340	10.225	8.031	4.222
.0125	.817	1.062	1.410	2.339	4.174	6.906	12.362	11.750	7.662	3.838
.0250	.870	1.062	1.315	1.906	3.055	5.236	7.954	9.078	7.061	3.814
.0500	.904	1.077	1.256	1.583	1.788	2.448	3.410	4.610	5.275	3.485
.0750	.920	1.059	1.222	1.470	1.653	2.029	2.578	3.177	4.107	3.296
.1000	.932	1.068	1.213	1.433	1.592	1.895	2.357	2.791	3.589	3.180
.1500	.950	1.068	1.185	1.364	1.487	1.708	2.047	2.309	2.793	2.934
.2000	.972	1.077	1.185	1.332	1.432	1.622	1.885	2.101	2.427	2.722
.2500	.978	1.084	1.182	1.295	1.377	1.546	1.786	1.960	2.134	2.479
.3000	.994	1.081	1.179	1.285	1.334	1.505	1.699	1.785	1.970	2.293
.3500	.991	1.087	1.154	1.257	1.307	1.467	1.581	1.687	1.823	2.108
.4000	.997	1.081	1.142	1.238	1.276	1.435	1.510	1.604	1.747	1.955
.4500	1.009	1.087	1.139	1.226	1.273	1.387	1.466	1.546	1.665	1.841
.5000	1.019	1.087	1.142	1.216	1.261	1.353	1.438	1.506	1.610	1.769
.5500	1.015	1.074	1.126	1.182	1.221	1.302	1.388	1.454	1.537	1.674
.6000	1.015	1.068	1.117	1.182	1.205	1.280	1.376	1.438	1.540	1.623
.6500	1.003	1.059	1.105	1.144	1.172	1.248	1.345	1.441	1.531	1.578
.7000	.994	1.028	1.074	1.122	1.153	1.210	1.348	1.420	1.470	1.509
.8000	.981	1.028	1.046	1.085	1.095	1.194	1.242	1.257	1.320	1.392
.8500	.957	.994	1.012	1.047	1.046	1.140	1.146	1.147	1.210	1.317
.9000	.882	.920	.938	.988	.985	1.035	1.013	1.015	1.098	1.234
.9500	1.241	1.285	1.290	1.301	1.257	1.375	1.345	1.365	1.378	1.392
Spoiler										
.2429	.745	.969	1.000	1.019	1.006	1.055	1.040	1.031	1.034	1.139
.4852	.720	.966	1.000	1.012	1.000	1.034	1.024	1.012	1.018	1.109
.7367	.752	1.025	1.079	1.074	1.043	1.092	1.079	1.061	1.064	1.142
.9796	.936	1.212	1.242	1.226	1.175	1.243	1.244	1.245	1.244	1.269

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.232	.991	.768	.627	.586	.622	.680	.727	.686	.509
.0250	1.173	.988	.805	.640	.506	.422	.357	.344	.317	.251
.0500	1.142	1.000	.839	.693	.531	.403	.276	.242	.232	.225
.0750	1.124	.994	.867	.718	.564	.435	.323	.307	.281	.302
.1000	1.127	1.019	.883	.743	.604	.486	.382	.368	.360	.374
.1500	1.115	1.019	.913	.787	.659	.572	.488	.460	.445	.440
.2000	1.127	1.043	.947	.828	.712	.622	.544	.521	.497	.482
.2500	1.164	1.084	.988	.890	.773	.683	.603	.586	.558	.530
.3000	1.167	1.105	1.015	.931	.813	.737	.652	.635	.604	.575
.3500	1.195	1.152	1.074	.991	.874	.800	.711	.690	.662	.623
.4000	1.223	1.183	1.096	1.028	.911	.845	.749	.736	.695	.653
.4500	1.241	1.195	1.120	1.050	.954	.883	.786	.764	.735	.704
.5000	1.235	1.192	1.126	1.063	.963	.908	.814	.794	.762	.725
.5500	1.214	1.161	1.136	1.063	.969	.933	.836	.804	.796	.749
.6000	1.214	1.176	1.111	1.063	.978	.943	.857	.828	.796	.775
.6500	1.183	1.149	1.096	1.053	.966	.933	.854	.828	.802	.775
.7000	1.164	1.136	1.089	1.053	.966	.943	.863	.846	.811	.805
.8000	1.136	1.105	1.077	1.056	1.006	.968	.895	.883	.866	.859
.8500	1.121	1.093	1.071	1.056	1.015	.994	.935	.917	.906	.889
.9000	1.090	1.068	1.046	1.035	.991	.981	.932	.926	.909	.904
.9500	1.093	1.081	1.055	1.044	1.000	1.006	.960	.963	.951	.961
Spoiler										
.2573	.981	1.277	1.324	1.285	1.236	1.323	1.363	1.389	1.387	1.396
.4994	1.067	1.324	1.330	1.297	1.236	1.338	1.378	1.395	1.396	1.405
.7485	1.080	1.324	1.327	1.297	1.236	1.345	1.378	1.395	1.396	1.402
.9940	1.131	1.361	1.349	1.310	1.242	1.369	1.415	1.429	1.436	1.429

TABLE V.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(c) Plain wing configuration with spoiler; $h_s = -0.010\bar{c}$; $\frac{y}{b/2} = 0.30$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.882	.895	1.574	2.442	3.134	4.162	4.504	4.456	3.912	3.135
.0125	.808	1.111	1.762	2.474	3.058	3.804	3.979	4.067	3.845	3.138
.0250	.864	1.102	1.811	2.571	3.168	3.950	4.097	4.067	3.848	3.138
.0500	.916	1.099	1.685	2.317	3.460	3.861	3.954	4.076	3.878	3.147
.0750	.935	1.087	1.268	2.890	4.521	5.178	4.578	4.358	3.915	3.156
.1000	.941	1.074	1.173	1.768	3.361	4.677	4.541	4.275	3.817	3.138
.1500	.954	1.077	1.157	1.210	1.754	2.937	3.839	3.828	3.576	3.066
.2000	.975	1.077	1.182	1.238	1.389	2.089	2.979	3.205	3.223	2.934
.2500	.969	1.074	1.157	1.245	1.307	1.737	2.454	2.714	2.906	2.796
.3000	.997	1.081	1.160	1.245	1.285	1.603	2.165	2.432	2.674	2.671
.3500	.997	1.074	1.148	1.220	1.261	1.508	1.913	2.159	2.409	2.500
.4000	1.003	1.074	1.148	1.223	1.248	1.464	1.792	2.012	2.250	2.374
.4500	1.019	1.090	1.148	1.216	1.254	1.441	1.736	1.938	2.177	2.323
.5000	1.009	1.084	1.145	1.207	1.227	1.400	1.634	1.785	1.979	2.135
.5500	1.012	1.068	1.117	1.179	1.199	1.368	1.575	1.711	1.900	2.048
.6000	1.009	1.077	1.111	1.169	1.190	1.343	1.525	1.638	1.808	1.955
.6500	1.009	1.059	1.105	1.154	1.172	1.314	1.472	1.573	1.720	1.856
.7500	.978	1.025	1.043	1.082	1.107	1.229	1.345	1.463	1.589	1.686
.8000	.957	1.003	1.021	1.050	1.070	1.168	1.301	1.405	1.515	1.596
.8500	.920	.963	.981	1.006	1.015	1.121	1.239	1.346	1.421	1.500
.9000	.805	.833	.870	.900	.954	.956	1.078	1.178	1.235	1.341
.9500	1.232	1.282	1.271	1.288	1.254	1.372	1.345	1.353	1.378	1.389
Spoiler										
.2475	.758	.953	.997	.994	.994	1.040	1.146	1.218	1.244	1.320
.4909	.806	1.006	1.032	1.031	1.018	1.077	1.177	1.233	1.235	1.320
.7396	.844	1.059	1.082	1.071	1.061	1.102	1.189	1.239	1.226	1.302
.9796	1.000	1.299	1.330	1.303	1.245	1.338	1.396	1.417	1.399	1.426

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.266	.957	.778	.774	.883	1.108	1.242	1.297	1.357	1.287
.0250	1.272	1.003	.815	.727	.715	.791	.823	.856	.884	.865
.0500	1.211	1.015	.855	.743	.659	.635	.593	.592	.592	.587
.0750	1.189	1.040	.895	.768	.662	.600	.531	.521	.509	.494
.1000	1.183	1.056	.917	.796	.687	.610	.531	.500	.485	.461
.1500	1.180	1.077	.969	.843	.733	.657	.556	.528	.494	.473
.2000	1.195	1.111	1.003	.900	.773	.686	.575	.549	.512	.476
.2500	1.217	1.136	1.046	.941	.816	.724	.612	.583	.543	.509
.3000	1.226	1.152	1.062	.975	.856	.765	.658	.616	.585	.539
.3500	1.235	1.170	1.077	1.003	.889	.810	.699	.672	.625	.578
.4000	1.238	1.186	1.092	1.022	.914	.845	.733	.696	.668	.617
.4500	1.251	1.192	1.114	1.047	.942	.886	.767	.733	.695	.656
.5000	1.232	1.183	1.117	1.060	.960	.905	.795	.767	.726	.683
.5500	1.214	1.183	1.126	1.075	.981	.930	.829	.797	.771	.722
.6000	1.186	1.152	1.105	1.063	.969	.933	.842	.819	.784	.740
.6500	1.158	1.139	1.108	1.066	.985	.956	.863	.843	.808	.772
.7500	1.105	1.108	1.086	1.066	.991	.975	.891	.874	.854	.829
.8000	1.074	1.093	1.074	1.053	.997	.978	.913	.896	.878	.862
.8500	1.090	1.096	1.077	1.063	1.006	1.010	.950	.932	.927	.907
.9000	1.087	1.099	1.089	1.078	1.031	1.035	.985	.972	.963	.958
.9500	1.121	1.127	1.133	1.122	1.073	1.111	1.065	1.046	1.058	1.069
Spoiler										
.2587	.997	1.296	1.321	1.288	1.236	1.329	1.366	1.380	1.390	1.384
.5024	1.026	1.305	1.324	1.294	1.230	1.332	1.378	1.395	1.390	1.399
.7531	1.080	1.318	1.330	1.288	1.239	1.329	1.375	1.399	1.390	1.396
.9940	1.073	1.339	1.340	1.310	1.254	1.369	1.399	1.400	1.418	1.444

TABLE V.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(d) Plain wing configuration with spoiler; $h_s = -0.010\bar{c}$; $\frac{y}{b/2} = 0.43$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.511	.876	1.657	2.010	2.460	2.765	2.649	2.644	2.659	2.641
.0125	.780	1.087	1.636	1.994	2.426	2.734	2.718	2.751	2.726	2.671
.0250	.827	1.096	1.574	2.022	2.429	2.750	2.830	2.800	2.738	2.659
.0500	.879	1.081	1.546	2.082	2.515	2.842	2.783	2.788	2.772	2.653
.0750	.898	1.077	1.592	2.144	2.497	2.756	2.755	2.812	2.781	2.656
.1000	.913	1.074	1.540	2.320	2.451	2.673	2.733	2.843	2.796	2.656
.1500	.938	1.074	1.345	2.568	2.920	2.794	2.864	2.917	2.817	2.659
.2000	.954	1.071	1.225	2.085	3.220	3.239	2.969	2.923	2.784	2.647
.2500	.954	1.074	1.163	1.527	2.567	3.096	2.861	2.785	2.698	2.587
.3000	.972	1.084	1.154	1.279	2.046	2.886	2.789	2.736	2.647	2.563
.3500	.975	1.087	1.139	1.154	1.604	2.473	2.550	2.539	2.515	2.482
.4000	.988	1.087	1.142	1.151	1.417	2.219	2.416	2.454	2.451	2.419
.4500	.997	1.077	1.136	1.154	1.303	1.997	2.292	2.325	2.378	2.362
.5000	.960	1.062	1.111	1.141	1.211	1.765	2.078	2.184	2.232	2.251
.5500	.991	1.068	1.117	1.144	1.196	1.645	1.975	2.082	2.165	2.183
.6000	.991	1.068	1.108	1.135	1.169	1.575	1.873	1.990	2.073	2.111
.7000	.966	1.022	1.071	1.097	1.113	1.410	1.690	1.813	1.921	2.000
.7500	.957	1.019	1.037	1.072	1.089	1.356	1.631	1.760	1.857	1.955
.8000	.935	.966	.994	1.028	1.046	1.280	1.544	1.662	1.781	1.901
.8500	.885	.923	.950	.972	.975	1.187	1.416	1.537	1.689	1.829
.9000	1.282	1.331	1.318	1.364	1.359	1.454	1.528	1.512	1.567	1.701
.9500	1.285	1.334	1.327	1.357	1.356	1.448	1.528	1.537	1.579	1.737
Spoiler										
.2443	.761	.972	1.010	1.006	.988	1.102	1.299	1.436	1.537	1.701
.4906	.780	.994	1.032	1.025	1.009	1.102	1.278	1.416	1.494	1.656
.7370	.819	1.053	1.082	1.068	1.046	1.142	1.284	1.376	1.467	1.616
.9856	.956	1.277	1.315	1.313	1.300	1.354	1.445	1.496	1.518	1.637

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.594	1.053	.818	.834	.963	1.162	1.255	1.337	1.378	1.419
.0250	1.523	1.093	.873	.793	.791	.870	.895	.929	.960	.988
.0500	1.498	1.102	.913	.821	.721	.730	.714	.712	.717	.719
.0750	1.387	1.111	.938	.846	.736	.699	.649	.659	.640	.635
.1000	1.322	1.115	.963	.862	.754	.699	.634	.620	.601	.590
.1500	1.272	1.136	.994	.903	.776	.721	.634	.610	.579	.548
.2000	1.241	1.142	1.012	.922	.800	.743	.652	.623	.585	.554
.2500	1.241	1.161	1.043	.959	.834	.778	.677	.647	.607	.569
.3000	1.226	1.155	1.055	.969	.853	.791	.696	.675	.619	.596
.3500	1.257	1.176	1.086	1.016	.889	.845	.733	.708	.659	.626
.4000	1.238	1.176	1.089	1.016	.908	.857	.767	.739	.695	.656
.4500	1.217	1.170	1.089	1.031	.920	.883	.777	.751	.713	.674
.5000	1.201	1.164	1.083	1.031	.935	.902	.811	.779	.732	.707
.5500	1.186	1.152	1.083	1.035	.945	.908	.826	.794	.759	.734
.6000	1.170	1.139	1.086	1.047	.954	.930	.851	.819	.784	.763
.7000	1.164	1.146	1.102	1.066	.994	.978	.904	.862	.851	.829
.7500	1.146	1.139	1.089	1.063	1.000	.991	.926	.917	.884	.874
.8000	1.127	1.115	1.077	1.063	1.009	1.010	.950	.942	.921	.910
.8500	1.121	1.111	1.077	1.060	1.018	1.029	.988	.969	.963	.958
.9000	1.099	1.121	1.089	1.082	1.049	1.070	1.034	1.037	1.028	1.033
.9500	1.152	1.161	1.133	1.154	1.123	1.153	1.159	1.169	1.171	1.225
Spoiler										
.2587	.946	1.293	1.333	1.328	1.322	1.360	1.430	1.456	1.479	1.571
.5039	1.022	1.336	1.355	1.341	1.322	1.369	1.448	1.496	1.515	1.637
.7508	.978	1.318	1.346	1.334	1.328	1.360	1.454	1.515	1.525	1.640
.9940	1.086	1.380	1.396	1.378	1.365	1.397	1.479	1.535	1.549	1.680

TABLE V.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(e) Plain wing configuration with spoiler; $h_s = -0.010\bar{c}$; $\frac{y}{b/2} = 0.55$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0000	1.229	.926	1.608	1.796	2.070	2.229	2.152	2.175	2.211	2.234
.0125	.833	1.127	1.602	1.834	2.095	2.245	2.202	2.245	2.262	2.251
.0250	.811	1.077	1.617	1.818	2.067	2.219	2.211	2.248	2.265	2.257
.0500	.845	1.090	1.540	1.825	2.073	2.245	2.236	2.279	2.272	2.231
.0750	.879	1.077	1.401	1.853	2.061	2.280	2.224	2.291	2.287	2.243
.1000	.904	1.084	1.333	1.884	2.104	2.286	2.239	2.300	2.287	2.231
.1500	.926	1.084	1.284	1.894	2.079	2.191	2.258	2.306	2.278	2.207
.2000	.938	1.084	1.262	2.000	2.030	2.169	2.274	2.309	2.281	2.225
.2500	.966	1.096	1.234	2.207	2.067	2.165	2.302	2.325	2.287	2.219
.3000	.981	1.105	1.241	2.307	2.355	2.267	2.330	2.316	2.287	2.243
.3500	.975	1.090	1.219	2.132	2.500	2.359	2.311	2.300	2.275	2.219
.4000	.988	1.096	1.207	1.862	2.435	2.375	2.264	2.260	2.238	2.177
.4500	.991	1.093	1.182	1.627	2.276	2.343	2.218	2.217	2.214	2.171
.5000	.994	1.093	1.170	1.426	2.101	2.276	2.180	2.171	2.183	2.138
.6000	.966	1.056	1.126	1.151	1.718	2.064	2.038	2.061	2.089	2.078
.6500	.991	1.059	1.111	1.088	1.573	1.962	1.969	2.003	2.040	2.048
.7000	.972	1.037	1.086	1.041	1.441	1.861	1.904	1.948	1.988	2.006
.7500	.963	1.006	1.055	1.009	1.362	1.778	1.848	1.889	1.933	1.988
.8500	.814	.861	.895	.818	1.043	1.505	1.612	1.665	1.784	1.898
.9000	1.294	1.350	1.315	1.423	1.583	1.632	1.711	1.718	1.805	1.946
.9500	1.294	1.331	1.296	1.401	1.540	1.591	1.662	1.684	1.756	1.895
Spoiler										
.2443	.704	.931	1.006	.957	1.037	1.323	1.592	1.695	1.732	1.834
.4881	.768	.994	1.041	.997	1.034	1.326	1.555	1.635	1.701	1.819
.7416	.787	1.044	1.079	1.050	1.040	1.295	1.518	1.615	1.662	1.801
.9856	.949	1.255	1.293	1.303	1.340	1.483	1.604	1.675	1.744	1.909

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0125	1.731	1.133	.858	.922	1.086	1.289	1.382	1.445	1.543	1.599
.0250	1.703	1.158	.901	.850	.850	.959	.994	1.021	1.088	1.126
.0500	1.687	1.161	.935	.859	.816	.797	.789	.797	.823	.832
.0750	1.511	1.146	.954	.843	.758	.737	.714	.702	.701	.698
.1000	1.443	1.155	.972	.868	.776	.721	.680	.669	.665	.653
.1500	1.362	1.155	.997	.900	.779	.730	.674	.641	.640	.614
.2000	1.328	1.161	1.018	.931	.804	.762	.677	.644	.643	.614
.2500	1.316	1.173	1.043	.959	.828	.775	.708	.672	.659	.617
.3000	1.307	1.164	1.055	.975	.853	.797	.733	.684	.668	.638
.3500	1.288	1.173	1.074	.991	.877	.816	.758	.724	.701	.662
.4000	1.276	1.173	1.089	1.009	.902	.851	.786	.739	.723	.680
.4500	1.260	1.173	1.089	1.022	.908	.883	.798	.761	.738	.704
.5000	1.245	1.173	1.092	1.031	.939	.892	.826	.770	.771	.746
.6000	1.217	1.164	1.102	1.047	.975	.937	.873	.840	.826	.805
.6500	1.201	1.161	1.105	1.056	.991	.965	.907	.880	.866	.841
.7000	1.176	1.139	1.086	1.056	.988	.987	.926	.911	.890	.871
.7500	1.173	1.136	1.086	1.078	1.021	1.016	.975	.942	.930	.919
.8000	1.164	1.146	1.092	1.085	1.018	1.045	1.009	.978	.979	.976
.8500	1.152	1.136	1.102	1.097	1.067	1.076	1.050	1.040	1.052	1.051
.9000	1.149	1.139	1.108	1.113	1.092	1.149	1.124	1.107	1.134	1.156
.9500	1.167	1.170	1.151	1.154	1.156	1.226	1.236	1.257	1.287	1.329
Spoiler										
.2584	.959	1.268	1.302	1.322	1.411	1.511	1.595	1.655	1.674	1.782
.5039	.927	1.252	1.302	1.325	1.402	1.514	1.601	1.675	1.723	1.864
.7508	.946	1.271	1.308	1.313	1.405	1.520	1.613	1.695	1.787	1.970
.9940	.930	1.274	1.327	1.347	1.469	1.585	1.640	1.715	1.784	1.982

TABLE V.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Concluded

(f) Plain wing configuration with spoiler; $h_s = -0.010\bar{c}$; $\frac{y}{b/2} = 0.72$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0000	1.542	.944	1.642	1.628	1.681	1.797	1.805	1.834	1.860	1.834
.0125	.733	1.037	1.604	1.601	1.650	1.751	1.753	1.800	1.835	1.834
.0250	.752	1.062	1.569	1.604	1.653	1.760	1.771	1.825	1.860	1.837
.0500	.796	1.078	1.566	1.610	1.650	1.760	1.808	1.849	1.875	1.846
.0750	.825	1.087	1.585	1.616	1.650	1.791	1.823	1.868	1.881	1.849
.1000	.806	1.068	1.494	1.598	1.616	1.769	1.808	1.852	1.872	1.849
.1500	.841	1.078	1.418	1.594	1.626	1.785	1.826	1.865	1.881	1.831
.2000	.905	1.103	1.371	1.616	1.641	1.800	1.854	1.886	1.890	1.828
.2500	.917	1.100	1.311	1.628	1.650	1.779	1.845	1.877	1.887	1.822
.3000	.914	1.096	1.258	1.607	1.626	1.766	1.842	1.868	1.869	1.813
.3500	.959	1.112	1.249	1.622	1.619	1.775	1.845	1.865	1.863	1.804
.4000	.965	1.109	1.217	1.619	1.601	1.754	1.829	1.849	1.854	1.797
.4500	.971	1.115	1.208	1.632	1.592	1.735	1.820	1.843	1.839	1.791
.5000	.965	1.112	1.198	1.687	1.610	1.739	1.820	1.834	1.823	1.779
.5500	.959	1.100	1.170	1.709	1.607	1.714	1.799	1.813	1.811	1.779
.6000	.930	1.093	1.154	1.690	1.604	1.702	1.775	1.797	1.796	1.773
.6500	.943	1.084	1.132	1.687	1.604	1.689	1.759	1.770	1.778	1.755
.7000	.965	1.075	1.126	1.694	1.613	1.692	1.756	1.767	1.771	1.767
.7500	.991	1.081	1.116	1.663	1.632	1.695	1.744	1.760	1.753	1.755
.8000	.981	1.068	1.104	1.628	1.632	1.686	1.717	1.736	1.741	1.755
.8500	1.003	1.056	1.094	1.548	1.583	1.646	1.677	1.693	1.701	1.734
.9000	.968	1.053	1.079	1.433	1.534	1.609	1.634	1.647	1.665	1.707

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0125	1.570	1.258	.874	.817	.862	.985	1.082	1.147	1.186	1.245
.0250	1.586	1.230	.896	.789	.779	.855	.915	.948	.979	1.033
.0500	1.542	1.206	.921	.783	.718	.735	.781	.791	.805	.834
.0750	1.459	1.234	.966	.820	.736	.729	.738	.745	.738	.758
.1000	1.459	1.215	.978	.830	.736	.729	.713	.705	.704	.710
.1500	1.411	1.190	.997	.854	.758	.729	.704	.690	.668	.671
.2000	1.389	1.171	1.003	.882	.782	.748	.717	.705	.677	.671
.2500	1.328	1.149	1.016	.898	.797	.778	.732	.721	.689	.692
.3000	1.280	1.153	1.028	.923	.828	.800	.762	.742	.717	.710
.3500	1.233	1.140	1.035	.944	.856	.822	.787	.767	.738	.728
.4000	1.121	1.137	1.044	.944	.865	.846	.799	.788	.750	.749
.4500	1.131	1.137	1.047	.969	.892	.877	.832	.804	.777	.767
.5000	1.108	1.140	1.057	.981	.908	.892	.848	.828	.796	.779
.5500	1.057	1.131	1.063	.988	.926	.908	.872	.853	.820	.798
.6000	1.038	1.140	1.076	1.009	.951	.942	.903	.880	.851	.837
.6500	1.067	1.131	1.072	1.012	.969	.960	.927	.905	.872	.864
.7000	1.051	1.109	1.057	1.012	.981	.975	.945	.935	.912	.897
.7500	1.057	1.090	1.060	1.025	.994	1.006	.979	.975	.951	.943
.8000	1.019	1.084	1.044	1.031	1.012	1.043	1.018	1.015	1.003	1.003
.8500	.994	1.078	1.054	1.050	1.037	1.089	1.079	1.080	1.070	1.063
.9000	.959	1.072	1.054	1.071	1.113	1.157	1.165	1.159	1.156	1.166
.9500	.936	1.047	1.025	1.093	1.172	1.249	1.262	1.270	1.262	1.299

TABLE VI.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE

(a) Plain wing configuration with spoiler; $h_s = -0.035\bar{c}$; $\frac{y}{b/2} = 0$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Fuselage										
.0000	.012	.000	.000	.000	.009	.021	.050	.085	.098	.149
.0500	.758	.833	.910	.941	1.003	1.028	1.065	1.079	1.082	1.084
.1000	.856	.926	.978	1.012	1.043	1.076	1.109	1.128	1.149	1.149
.1500	.932	.988	1.034	1.062	1.096	1.104	1.137	1.143	1.156	1.140
.2000	.978	1.037	1.077	1.084	1.127	1.104	1.140	1.137	1.137	1.128
.2500	1.024	1.080	1.105	1.105	1.133	1.134	1.146	1.146	1.134	1.140
.3000	1.049	1.092	1.099	1.102	1.133	1.116	1.131	1.128	1.119	1.122
.3500	1.052	1.080	1.087	1.087	1.108	1.110	1.109	1.113	1.116	1.131
.4000	1.018	1.037	1.040	1.050	1.068	1.070	1.100	1.116	1.131	1.143
.4500	1.003	1.031	1.050	1.059	1.105	1.125	1.171	1.219	1.229	1.257
.5000	.981	1.028	1.065	1.102	1.152	1.198	1.267	1.322	1.348	1.478
.5500	.981	1.043	1.081	1.133	1.211	1.250	1.336	1.362	1.412	1.719
.6000	.988	1.040	1.093	1.149	1.214	1.241	1.292	1.335	1.372	1.791
.6500	.978	1.040	1.096	1.142	1.180	1.207	1.249	1.252	1.274	1.561
.7000	.991	1.052	1.087	1.115	1.164	1.168	1.202	1.210	1.226	1.412
.7500	.991	1.046	1.068	1.093	1.136	1.146	1.193	1.207	1.226	1.331
.8000	.981	1.037	1.059	1.074	1.111	1.119	1.177	1.216	1.244	1.266
.8500	1.003	1.043	1.059	1.062	1.099	1.104	1.174	1.204	1.238	1.230
.9000	1.015	1.049	1.053	1.056	1.074	1.067	1.134	1.152	1.165	1.173
.9500	1.034	1.065	1.065	1.062	1.071	1.076	1.109	1.128	1.128	1.128
.9940	1.132	1.157	1.142	1.127	1.133	1.104	1.118	1.119	1.128	1.122

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Fuselage										
.0500	.853	.842	.783	.715	.669	.570	.509	.453	.412	.367
.1000	.929	.950	.882	.820	.768	.686	.624	.562	.527	.475
.1500	1.012	.994	.954	.898	.842	.747	.693	.641	.604	.552
.2000	1.037	1.037	.997	.947	.904	.820	.764	.708	.677	.615
.2500	1.061	1.062	1.025	.985	.947	.872	.811	.772	.735	.681
.3000	1.077	1.080	1.059	1.003	.978	.906	.848	.818	.774	.719
.3500	1.049	1.065	1.043	1.019	.997	.924	.873	.833	.802	.737
.4000	.994	1.009	.991	.960	.941	.878	.820	.790	.744	.696
.4500	.969	.984	.966	.929	.889	.826	.778	.733	.689	.627
.5000	.957	.975	.935	.885	.851	.768	.705	.654	.622	.558
.5500	.945	.963	.920	.864	.824	.729	.662	.605	.558	.507
.6000	.932	.938	.867	.814	.759	.680	.612	.565	.524	.481
.7500	1.199	1.200	1.158	1.118	1.090	1.000	.954	.915	.896	.836
.9500	1.006	1.040	1.015	1.009	.997	.957	.950	.930	.915	.890
.9940	.988	1.089	1.111	1.111	1.127	1.110	1.118	1.107	1.098	1.063

TABLE VI.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(b) Plain wing configuration with spoiler; $h_s = -0.035\bar{c}$; $\frac{y}{b/2} = 0.21$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.386	.848	1.300	2.781	4.450	6.591	9.094	9.816	6.619	4.099
.0125	.777	1.018	1.426	2.375	5.959	9.373	11.986	10.947	6.200	3.766
.0250	.821	1.021	1.316	1.891	3.000	4.902	7.729	8.807	5.852	3.682
.0500	.884	1.052	1.261	1.566	1.757	2.332	3.401	4.615	4.814	3.386
.0750	.890	1.031	1.221	1.456	1.629	1.948	2.552	3.189	4.007	3.184
.1000	.915	1.043	1.215	1.413	1.573	1.812	2.322	2.776	3.569	3.040
.1500	.934	1.046	1.181	1.344	1.463	1.643	2.021	2.274	2.848	2.772
.2000	.966	1.058	1.181	1.319	1.408	1.566	1.868	2.070	2.469	2.592
.2500	.975	1.058	1.159	1.291	1.346	1.502	1.767	1.918	2.149	2.389
.3000	.981	1.064	1.156	1.250	1.316	1.458	1.693	1.766	1.985	2.258
.3500	.975	1.055	1.141	1.219	1.276	1.406	1.570	1.660	1.842	2.124
.4000	.984	1.055	1.123	1.200	1.251	1.378	1.497	1.590	1.749	2.019
.4500	.991	1.061	1.119	1.194	1.236	1.348	1.441	1.529	1.665	1.922
.5000	.997	1.058	1.116	1.178	1.227	1.311	1.417	1.487	1.615	1.860
.5500	.988	1.034	1.083	1.147	1.185	1.246	1.365	1.417	1.556	1.751
.6000	.969	1.018	1.073	1.125	1.153	1.218	1.337	1.411	1.550	1.688
.6500	.953	1.003	1.046	1.088	1.126	1.163	1.310	1.398	1.525	1.632
.7000	.915	.970	1.015	1.047	1.083	1.129	1.291	1.356	1.469	1.567
.8000	.868	.903	.935	.975	.985	1.034	1.153	1.164	1.270	1.408
.8500	.781	.818	.856	.894	.920	.966	1.021	1.012	1.143	1.321
.9000	.655	.672	.718	.772	.825	.778	.834	.830	.978	1.190
.9500	1.505	1.544	1.543	1.653	1.647	1.735	1.797	1.809	1.848	1.841
Spoiler										
.2429	.653	.756	.786	.817	.864	.842	.826	.784	.857	1.045
.4852	.715	.849	.882	.898	.935	.909	.879	.809	.869	1.003
.7367	.850	1.000	1.053	1.065	1.090	1.052	1.003	.918	.970	1.048
.9796	1.303	1.457	1.486	1.520	1.542	1.540	1.556	1.511	1.540	1.504

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.235	.997	.736	.622	.574	.612	.687	.684	.621	.489
.0250	1.185	.988	.779	.625	.500	.428	.362	.334	.292	.243
.0500	1.135	.997	.816	.688	.524	.394	.282	.243	.224	.218
.0750	1.122	1.006	.846	.722	.561	.431	.331	.304	.283	.299
.1000	1.141	1.012	.871	.738	.598	.468	.399	.380	.363	.361
.1500	1.129	1.015	.896	.781	.656	.557	.488	.471	.441	.439
.2000	1.132	1.046	.932	.838	.708	.618	.549	.529	.506	.489
.2500	1.166	1.079	.981	.884	.764	.683	.607	.593	.556	.542
.3000	1.151	1.104	1.009	.925	.800	.732	.647	.635	.599	.586
.3500	1.160	1.143	1.064	.975	.868	.791	.715	.687	.658	.632
.4000	1.207	1.170	1.092	1.019	.917	.837	.754	.730	.705	.679
.4500	1.248	1.201	1.110	1.044	.954	.880	.804	.766	.742	.716
.5000	1.241	1.186	1.123	1.063	.975	.902	.831	.793	.773	.741
.5500	1.241	1.164	1.135	1.063	1.003	.951	.853	.812	.786	.773
.6000	1.238	1.183	1.123	1.078	1.000	.942	.871	.848	.820	.801
.6500	1.220	1.161	1.110	1.069	.994	.948	.871	.848	.826	.804
.7000	1.204	1.152	1.113	1.072	1.006	.954	.902	.872	.842	.838
.8000	1.188	1.158	1.119	1.103	1.034	1.003	.957	.927	.913	.900
.8500	1.198	1.158	1.132	1.122	1.061	1.040	1.006	.976	.960	.956
.9000	1.216	1.207	1.184	1.184	1.141	1.120	1.089	1.058	1.053	1.044
.9500	.561	.830	1.031	1.131	1.126	1.129	1.104	1.104	1.115	1.112
Spoiler										
.2573	1.380	1.552	1.604	1.675	1.734	1.756	1.854	1.867	1.872	1.830
.4984	1.438	1.555	1.579	1.635	1.666	1.698	1.780	1.794	1.796	1.734
.7485	1.481	1.605	1.638	1.681	1.706	1.738	1.814	1.827	1.829	1.731
.9940	1.518	1.617	1.659	1.709	1.768	1.811	1.923	1.930	1.948	1.946

TABLE VI.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(c) Plain wing configuration with spoiler; $h_s = -0.035\bar{c}$; $\frac{y}{b/2} = 0.30$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	2.006	.918	1.595	2.441	3.067	3.935	4.367	4.256	3.668	3.025
.0125	.793	1.079	1.764	2.450	3.006	3.594	3.883	3.906	3.625	3.028
.0250	.850	1.079	1.803	2.547	3.095	3.714	3.969	3.916	3.637	3.028
.0500	.900	1.070	1.656	3.263	3.417	3.619	3.852	3.955	3.650	3.028
.0750	.909	1.073	1.267	2.875	4.423	4.840	4.398	4.147	3.668	3.053
.1000	.915	1.052	1.172	1.750	3.266	4.382	4.331	4.034	3.578	3.034
.1500	.937	1.049	1.150	1.188	1.702	2.834	3.711	3.627	3.392	2.950
.2000	.975	1.061	1.169	1.225	1.359	2.022	2.917	3.095	3.100	2.825
.2500	.928	1.043	1.147	1.225	1.282	1.668	2.417	2.666	2.845	2.704
.3000	.966	1.049	1.144	1.225	1.267	1.542	2.153	2.399	2.634	2.592
.3500	.981	1.049	1.129	1.200	1.233	1.440	1.902	2.131	2.401	2.436
.4000	.978	1.049	1.132	1.191	1.230	1.397	1.767	1.982	2.243	2.336
.4500	.988	1.061	1.123	1.181	1.221	1.378	1.730	1.921	2.171	2.274
.5000	.988	1.043	1.104	1.159	1.190	1.323	1.613	1.754	1.985	2.118
.5500	.959	1.012	1.070	1.134	1.159	1.298	1.558	1.687	1.901	2.037
.6000	.956	1.003	1.055	1.113	1.126	1.268	1.484	1.620	1.817	1.956
.6500	.944	.994	1.037	1.078	1.113	1.231	1.432	1.529	1.736	1.863
.7500	.872	.906	.942	.981	.994	1.098	1.270	1.398	1.565	1.688
.8000	.784	.833	.874	.913	.932	1.025	1.184	1.322	1.469	1.595
.8500	.693	.736	.770	.813	.862	.889	1.067	1.186	1.323	1.467
.9000	.524	.556	.620	.647	.770	.640	.782	.945	1.084	1.271
.9500	1.505	1.526	1.543	1.644	1.656	1.729	1.791	1.803	1.845	1.844
Spoiler										
.2475	.598	.679	.721	.740	.783	.762	.876	.988	1.055	1.227
.4909	.733	.809	.861	.876	.904	.887	1.025	1.094	1.125	1.266
.7396	.889	.994	1.022	1.037	1.071	1.061	1.177	1.204	1.229	1.337
.9796	1.365	1.534	1.585	1.616	1.678	1.677	1.793	1.769	1.787	1.788

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.292	.979	.758	.766	.871	1.037	1.218	1.268	1.320	1.244
.0250	1.304	1.003	.797	.719	.715	.748	.816	.851	.873	.847
.0500	1.241	1.015	.853	.728	.659	.609	.595	.590	.587	.583
.0750	1.204	1.037	.883	.766	.659	.591	.543	.526	.506	.498
.1000	1.207	1.049	.914	.794	.678	.600	.531	.502	.488	.470
.1500	1.194	1.079	.969	.838	.724	.640	.564	.538	.503	.486
.2000	1.204	1.110	.997	.900	.770	.677	.589	.556	.509	.492
.2500	1.220	1.137	1.037	.934	.819	.714	.629	.593	.550	.523
.3000	1.248	1.152	1.061	.969	.850	.757	.669	.629	.593	.561
.3500	1.267	1.167	1.086	1.000	.889	.794	.708	.675	.637	.598
.4000	1.270	1.183	1.104	1.025	.926	.828	.745	.705	.677	.645
.4500	1.273	1.192	1.119	1.056	.948	.871	.791	.745	.711	.682
.5000	1.254	1.189	1.132	1.063	.966	.892	.813	.781	.742	.716
.5500	1.235	1.192	1.135	1.088	.997	.917	.840	.812	.792	.760
.6000	1.207	1.164	1.116	1.066	.994	.929	.865	.833	.804	.785
.6500	1.204	1.152	1.132	1.091	1.006	.954	.886	.866	.842	.810
.7500	1.132	1.134	1.123	1.103	1.021	.988	.932	.903	.898	.866
.8000	1.097	1.131	1.123	1.103	1.031	1.003	.951	.933	.926	.903
.8500	1.169	1.155	1.147	1.134	1.070	1.055	1.006	.991	.978	.966
.9000	1.188	1.186	1.184	1.178	1.116	1.098	1.067	1.064	1.050	1.025
.9500	1.301	1.265	1.279	1.284	1.239	1.243	1.215	1.210	1.211	1.215
Spoiler										
.2587	1.365	1.524	1.563	1.610	1.666	1.677	1.758	1.760	1.775	1.716
.5024	1.399	1.537	1.579	1.619	1.675	1.689	1.777	1.781	1.799	1.761
.7531	1.420	1.549	1.576	1.632	1.697	1.717	1.792	1.797	1.820	1.800
.9940	1.435	1.568	1.598	1.653	1.755	1.778	1.873	1.879	1.903	1.931

TABLE VI.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(d) Plain wing configuration with spoiler; $h_s = -0.035\bar{c}$; $\frac{y}{b/2} = 0.43$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.583	.876	1.635	1.969	2.374	2.594	2.546	2.551	2.590	2.567
.0125	.781	1.043	1.616	1.953	2.340	2.563	2.635	2.639	2.640	2.589
.0250	.828	1.058	1.561	1.988	2.340	2.575	2.723	2.687	2.662	2.585
.0500	.859	1.037	1.540	2.066	2.411	2.668	2.687	2.700	2.674	2.589
.0750	.906	1.049	1.567	2.094	2.395	2.575	2.674	2.718	2.674	2.585
.1000	.906	1.040	1.524	2.269	2.340	2.489	2.668	2.751	2.677	2.585
.1500	.903	1.034	1.343	2.569	2.766	2.600	2.754	2.788	2.693	2.592
.2000	.928	1.037	1.208	2.069	3.018	2.957	2.819	2.757	2.662	2.570
.2500	.947	1.034	1.144	1.506	2.466	2.837	2.717	2.648	2.600	2.514
.3000	.972	1.055	1.162	1.256	2.000	2.662	2.656	2.593	2.581	2.514
.3500	.969	1.061	1.141	1.128	1.595	2.299	2.438	2.408	2.466	2.423
.4000	.959	1.040	1.144	1.119	1.399	2.074	2.316	2.324	2.410	2.377
.4500	.978	1.040	1.123	1.100	1.282	1.902	2.193	2.234	2.326	2.318
.5000	.928	1.000	1.086	1.081	1.165	1.659	2.012	2.095	2.221	2.215
.5500	.937	1.006	1.080	1.088	1.132	1.551	1.920	2.019	2.152	2.174
.6000	.941	1.000	1.058	1.069	1.101	1.458	1.831	1.921	2.081	2.106
.7000	.884	.921	.978	.988	1.009	1.289	1.629	1.739	1.910	1.978
.7500	.843	.872	.917	.928	.969	1.209	1.546	1.651	1.836	1.934
.8000	.768	.784	.840	.828	.865	1.105	1.414	1.526	1.733	1.840
.8500	.649	.660	.699	.678	.699	.935	1.208	1.359	1.615	1.772
.9000	1.533	1.529	1.540	1.663	1.681	1.760	1.813	1.809	1.888	1.885
.9500	1.542	1.535	1.555	1.678	1.696	1.772	1.816	1.809	1.901	1.916
Spoiler										
.2443	.580	.673	.715	.697	.709	.793	1.019	1.271	1.460	1.669
.4906	.699	.821	.845	.839	.861	.903	1.121	1.341	1.500	1.666
.7370	.877	.991	1.019	1.022	1.050	1.061	1.236	1.435	1.561	1.692
.9856	1.273	1.481	1.533	1.560	1.663	1.662	1.764	1.794	1.829	1.875

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.715	1.091	.828	.838	.932	1.098	1.224	1.277	1.373	1.406
.0250	1.643	1.125	.883	.788	.758	.812	.874	.897	.969	.901
.0500	1.665	1.140	.942	.813	.715	.695	.690	.684	.721	.729
.0750	1.484	1.128	.966	.838	.712	.668	.647	.635	.658	.645
.1000	1.357	1.131	.981	.866	.730	.677	.635	.611	.621	.604
.1500	1.285	1.140	1.024	.900	.761	.695	.638	.599	.603	.573
.2000	1.276	1.146	1.046	.931	.797	.720	.662	.620	.627	.586
.2500	1.273	1.170	1.080	.969	.822	.742	.684	.641	.634	.598
.3000	1.273	1.167	1.083	.984	.846	.772	.712	.663	.665	.614
.3500	1.282	1.195	1.113	1.022	.896	.825	.758	.705	.702	.654
.4000	1.295	1.186	1.126	1.038	.917	.849	.782	.736	.730	.685
.4500	1.267	1.189	1.126	1.044	.929	.868	.800	.763	.758	.707
.5000	1.257	1.183	1.126	1.047	.932	.880	.822	.778	.786	.738
.5500	1.251	1.176	1.129	1.066	.954	.911	.850	.812	.828	.766
.6000	1.238	1.167	1.135	1.081	.975	.929	.880	.833	.851	.801
.7000	1.245	1.183	1.153	1.113	1.024	.988	.939	.906	.919	.872
.7500	1.238	1.186	1.165	1.125	1.043	1.015	.975	.942	.960	.922
.8000	1.226	1.180	1.169	1.138	1.061	1.040	1.015	.979	1.003	.950
.8500	1.232	1.195	1.187	1.169	1.098	1.080	1.064	1.021	1.056	1.019
.9000	1.260	1.228	1.233	1.225	1.159	1.163	1.156	1.116	1.143	1.131
.9500	1.367	1.353	1.346	1.356	1.300	1.320	1.328	1.298	1.373	1.352
Spoiler										
.2587	1.276	1.484	1.536	1.585	1.709	1.717	1.801	1.769	1.786	1.806
.5039	1.343	1.521	1.539	1.585	1.721	1.732	1.801	1.784	1.808	1.824
.7508	1.310	1.509	1.536	1.591	1.724	1.738	1.814	1.791	1.817	1.851
.9940	1.389	1.546	1.570	1.607	1.759	1.793	1.864	1.854	1.884	1.931

TABLE VI.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(e) Plain wing configuration with spoiler; $h_s = -0.035\bar{c}$; $\frac{y}{b/2} = 0.55$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0000	1.314	.891	1.849	1.744	1.963	2.065	2.082	2.101	2.180	2.221
.0125	.831	1.055	1.806	1.781	1.978	2.074	2.128	2.158	2.227	2.249
.0250	.813	1.031	1.840	1.763	1.951	2.062	2.135	2.161	2.233	2.249
.0500	.846	1.031	1.530	1.775	1.941	2.089	2.150	2.186	2.239	2.233
.0750	.868	1.043	1.325	1.794	1.944	2.123	2.156	2.198	2.230	2.240
.1000	.893	1.052	1.285	1.822	1.975	2.117	2.168	2.210	2.227	2.218
.1500	.919	1.049	1.227	1.825	1.944	2.055	2.193	2.210	2.218	2.212
.2000	.937	1.046	1.199	1.913	1.902	2.034	2.205	2.204	2.224	2.209
.2500	.931	1.043	1.181	2.138	1.917	2.025	2.217	2.219	2.233	2.212
.3000	.962	1.061	1.175	2.197	2.125	2.083	2.233	2.198	2.224	2.221
.3500	.962	1.049	1.165	2.050	2.245	2.135	2.211	2.174	2.211	2.209
.4000	.975	1.037	1.156	1.800	2.202	2.142	2.168	2.137	2.193	2.181
.4500	.975	1.037	1.144	1.578	2.092	2.111	2.138	2.101	2.165	2.159
.5000	.966	1.018	1.119	1.384	1.941	2.055	2.073	2.049	2.137	2.134
.6000	.925	.970	1.046	1.097	1.619	1.868	1.948	1.961	2.050	2.059
.6500	.915	.939	1.009	1.019	1.469	1.775	1.895	1.894	2.006	2.015
.7000	.865	.900	.963	.950	1.359	1.683	1.813	1.836	1.951	1.944
.7500	.806	.824	.908	.866	1.233	1.569	1.727	1.769	1.892	1.956
.8500	.558	.590	.681	.506	.819	1.305	1.420	1.450	1.696	1.832
.9000	1.555	1.553	1.552	1.697	1.714	1.791	1.834	1.809	1.910	1.931
.9500	1.539	1.547	1.546	1.659	1.681	1.739	1.785	1.775	1.864	1.860
Spoiler										
.2443	.561	.685	.743	.653	.762	1.107	1.404	1.511	1.643	1.779
.4881	.721	.839	.889	.793	.873	1.165	1.469	1.563	1.665	1.785
.7436	.868	1.006	1.043	.966	1.006	1.232	1.510	1.608	1.698	1.788
.9806	1.261	1.469	1.495	1.498	1.591	1.640	1.774	1.778	1.820	1.863

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0125	1.793	1.176	.859	.894	1.012	1.212	1.340	1.395	1.528	1.610
.0250	1.809	1.189	.905	.838	.825	.898	.969	.994	1.090	1.128
.0500	1.825	1.189	.966	.863	.761	.769	.788	.787	.845	.841
.0750	1.630	1.149	.963	.844	.745	.711	.699	.693	.730	.729
.1000	1.552	1.158	.988	.872	.742	.711	.684	.660	.690	.682
.1500	1.458	1.164	1.012	.906	.773	.723	.678	.641	.671	.626
.2000	1.433	1.173	1.043	.941	.800	.745	.705	.660	.674	.635
.2500	1.398	1.176	1.067	.969	.825	.763	.721	.672	.696	.642
.3000	1.379	1.183	1.083	.975	.856	.797	.745	.696	.714	.654
.3500	1.367	1.189	1.107	1.016	.889	.822	.776	.733	.745	.685
.4000	1.354	1.195	1.119	1.031	.908	.849	.800	.760	.767	.723
.4500	1.326	1.195	1.123	1.050	.929	.862	.816	.775	.795	.741
.5000	1.323	1.207	1.135	1.059	.951	.892	.850	.803	.823	.776
.6000	1.301	1.201	1.150	1.088	.991	.945	.892	.851	.888	.832
.6500	1.295	1.195	1.159	1.103	1.012	.978	.945	.897	.919	.882
.7000	1.273	1.180	1.141	1.113	1.034	1.000	.957	.921	.950	.906
.7500	1.279	1.198	1.165	1.138	1.055	1.034	1.015	.967	.997	.966
.8000	1.273	1.201	1.184	1.166	1.086	1.077	1.061	1.024	1.044	1.015
.8500	1.282	1.231	1.199	1.200	1.129	1.132	1.138	1.079	1.124	1.109
.9000	1.326	1.265	1.242	1.250	1.199	1.194	1.215	1.183	1.221	1.215
.9500	1.383	1.328	1.334	1.344	1.328	1.354	1.377	1.347	1.413	1.411
Spoiler										
.2584	1.322	1.509	1.560	1.585	1.712	1.720	1.798	1.769	1.796	1.806
.5039	1.285	1.512	1.563	1.585	1.706	1.723	1.798	1.769	1.790	1.815
.7508	1.303	1.531	1.582	1.591	1.715	1.726	1.795	1.766	1.802	1.830
.9940	1.319	1.546	1.594	1.616	1.728	1.753	1.820	1.803	1.842	1.872

TABLE VI.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Concluded

(f) Plain wing configuration with spoiler; $h_s = -0.035\bar{c}$; $\frac{y}{b/2} = 0.72$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0000	1.653	.926	1.529	1.520	1.619	1.659	1.718	1.760	1.787	1.785
.0125	.770	.954	1.495	1.492	1.585	1.616	1.683	1.733	1.784	1.788
.0250	.773	.994	1.483	1.492	1.591	1.625	1.702	1.751	1.790	1.791
.0500	.800	1.028	1.477	1.492	1.585	1.640	1.730	1.769	1.805	1.794
.0750	.831	1.031	1.502	1.511	1.601	1.659	1.742	1.781	1.805	1.785
.1000	.844	1.012	1.480	1.480	1.560	1.637	1.724	1.772	1.799	1.782
.1500	.871	1.025	1.427	1.483	1.563	1.646	1.746	1.788	1.802	1.773
.2000	.889	1.040	1.337	1.480	1.573	1.643	1.761	1.803	1.793	1.764
.2500	.911	1.037	1.260	1.477	1.567	1.646	1.755	1.791	1.781	1.755
.3000	.902	1.037	1.186	1.455	1.548	1.631	1.742	1.778	1.768	1.749
.3500	.935	1.052	1.170	1.464	1.542	1.628	1.733	1.772	1.762	1.746
.4000	.935	1.049	1.146	1.449	1.536	1.601	1.721	1.751	1.750	1.737
.4500	.942	1.037	1.121	1.452	1.517	1.585	1.705	1.736	1.744	1.731
.5000	.931	1.037	1.118	1.495	1.523	1.579	1.699	1.730	1.735	1.731
.5500	.945	1.025	1.090	1.495	1.511	1.558	1.674	1.702	1.717	1.713
.6000	.930	1.006	1.071	1.511	1.505	1.525	1.646	1.678	1.701	1.707
.6500	.929	.994	1.059	1.514	1.505	1.509	1.634	1.657	1.683	1.698
.7000	.918	1.006	1.062	1.529	1.508	1.512	1.624	1.648	1.671	1.695
.7500	.915	1.031	1.084	1.548	1.542	1.534	1.624	1.642	1.671	1.695
.8000	1.015	1.062	1.111	1.563	1.585	1.570	1.637	1.645	1.680	1.704
.8500	1.068	1.111	1.136	1.529	1.628	1.634	1.677	1.669	1.695	1.737
.9000	1.098	1.145	1.170	1.545	1.690	1.683	1.715	1.684	1.714	1.761

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0125	1.714	1.321	.848	.802	.858	.930	1.041	1.119	1.183	1.227
.0250	1.711	1.281	.879	.786	.786	.805	.879	.930	.985	1.009
.0500	1.699	1.234	.904	.786	.746	.723	.755	.778	.805	.824
.0750	1.672	1.256	.950	.817	.771	.710	.724	.730	.741	.752
.1000	1.690	1.237	.957	.842	.777	.710	.705	.699	.710	.707
.1500	1.690	1.222	.991	.867	.799	.713	.696	.687	.677	.666
.2000	1.681	1.200	1.003	.892	.833	.741	.714	.693	.683	.678
.2500	1.598	1.167	1.015	.910	.848	.765	.736	.708	.701	.687
.3000	1.466	1.170	1.028	.935	.882	.790	.767	.736	.726	.707
.3500	1.359	1.191	1.053	.966	.913	.817	.789	.763	.750	.728
.4000	1.248	1.167	1.056	.975	.929	.842	.814	.784	.771	.749
.4500	1.187	1.167	1.062	.997	.950	.875	.842	.803	.784	.770
.5000	1.156	1.170	1.071	1.015	.966	.887	.857	.827	.811	.800
.5500	1.129	1.167	1.081	1.031	.994	.912	.885	.851	.842	.812
.6000	1.138	1.176	1.102	1.053	1.022	.942	.916	.888	.866	.845
.6500	1.144	1.176	1.102	1.065	1.034	.963	.954	.912	.899	.878
.7000	1.123	1.154	1.093	1.068	1.050	.982	.969	.942	.927	.919
.7500	1.123	1.154	1.099	1.077	1.077	1.021	1.009	.985	.970	.961
.8000	1.113	1.142	1.096	1.105	1.111	1.055	1.059	1.043	1.031	1.018
.8500	1.107	1.148	1.111	1.133	1.155	1.119	1.124	1.097	1.098	1.090
.9000	1.101	1.148	1.121	1.164	1.217	1.183	1.211	1.186	1.186	1.188
.9500	1.089	1.136	1.111	1.232	1.319	1.290	1.314	1.292	1.311	1.301

TABLE VII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE

(a) Plain wing configuration with spoiler; $h_s = -0.050\bar{c}$; $\frac{y}{b/2} = 0$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Fuselage										
.0000	.003	.006	.000	.000	.006	.015	.061	.101	.097	.139
.0500	.740	.850	.891	.959	.985	1.027	1.064	1.056	1.079	1.088
.1000	.828	.922	.984	1.022	1.037	1.079	1.113	1.101	1.137	1.154
.1500	.919	1.006	1.041	1.081	1.080	1.110	1.147	1.119	1.149	1.151
.2000	.969	1.044	1.072	1.100	1.098	1.122	1.144	1.116	1.131	1.130
.2500	1.019	1.091	1.106	1.131	1.107	1.128	1.144	1.116	1.137	1.148
.3000	1.053	1.094	1.109	1.119	1.101	1.131	1.126	1.101	1.128	1.130
.3500	1.053	1.088	1.088	1.100	1.080	1.092	1.116	1.089	1.113	1.130
.4000	1.016	1.044	1.056	1.069	1.055	1.079	1.104	1.098	1.122	1.157
.4500	1.006	1.047	1.047	1.084	1.086	1.110	1.181	1.175	1.219	1.266
.5000	.984	1.041	1.063	1.119	1.132	1.190	1.276	1.279	1.338	1.462
.5500	.981	1.044	1.081	1.150	1.178	1.245	1.325	1.332	1.392	1.695
.6000	.984	1.044	1.091	1.169	1.184	1.245	1.300	1.291	1.359	1.758
.6500	.991	1.056	1.094	1.156	1.162	1.205	1.251	1.208	1.286	1.544
.7000	.991	1.044	1.081	1.141	1.135	1.171	1.193	1.160	1.210	1.423
.7500	.984	1.044	1.059	1.109	1.110	1.138	1.175	1.169	1.210	1.335
.8000	.991	1.031	1.050	1.088	1.092	1.119	1.169	1.172	1.231	1.287
.8500	1.003	1.053	1.050	1.084	1.080	1.116	1.159	1.178	1.237	1.239
.9000	1.006	1.044	1.059	1.078	1.055	1.076	1.132	1.124	1.173	1.181
.9500	1.044	1.075	1.072	1.078	1.055	1.070	1.113	1.101	1.131	1.142
.9940	1.154	1.166	1.147	1.147	1.113	1.128	1.123	1.101	1.131	1.127

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Fuselage										
.0500	.837	.859	.784	.713	.632	.584	.503	.457	.432	.375
.1000	.919	.931	.881	.831	.745	.703	.616	.570	.526	.480
.1500	.984	1.013	.953	.897	.816	.765	.696	.641	.608	.574
.2000	1.003	1.047	.997	.950	.877	.820	.764	.706	.675	.631
.2500	1.041	1.072	1.034	1.000	.929	.872	.813	.760	.733	.689
.3000	1.063	1.082	1.053	1.031	.963	.914	.862	.804	.772	.731
.3500	1.047	1.075	1.041	1.025	.966	.939	.886	.834	.806	.755
.4000	.991	1.022	.994	.975	.911	.887	.828	.777	.751	.698
.4500	.966	.988	.972	.944	.865	.835	.776	.721	.696	.634
.5000	.937	.978	.934	.900	.822	.774	.715	.647	.620	.583
.5500	.912	.969	.928	.872	.785	.743	.666	.599	.572	.532
.6000	.934	.928	.884	.825	.739	.688	.626	.564	.529	.489
.7500	1.210	1.216	1.159	1.144	1.064	1.021	.960	.893	.891	.858
.9500	1.000	1.038	1.025	1.025	1.000	.979	.963	.923	.927	.906
.9940	.997	1.100	1.116	1.131	1.104	1.113	1.126	1.101	1.110	1.085

TABLE VII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(b) Plain wing configuration with spoiler; $h_s = -0.050\bar{c}$; $\frac{y}{b/2} = 0.21$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0000	1.328	.861	1.217	2.635	4.337	6.566	8.944	9.582	6.640	3.603
.0125	.745	1.063	1.406	2.234	5.677	9.314	11.640	10.634	6.297	3.301
.0250	.785	1.060	1.291	1.808	2.957	4.834	7.574	8.515	5.916	3.322
.0500	.868	1.098	1.248	1.524	1.739	2.327	3.521	4.618	4.640	2.904
.0750	.871	1.076	1.214	1.432	1.622	1.950	2.561	3.189	3.846	2.791
.1000	.896	1.095	1.195	1.389	1.564	1.800	2.329	2.779	3.432	2.689
.1500	.926	1.086	1.186	1.315	1.454	1.651	2.019	2.289	2.816	2.495
.2000	.948	1.095	1.173	1.281	1.399	1.564	1.875	2.079	2.447	2.361
.2500	.954	1.098	1.158	1.247	1.343	1.501	1.778	1.924	2.135	2.221
.3000	.966	1.095	1.152	1.222	1.303	1.458	1.690	1.769	1.957	2.146
.3500	.963	1.092	1.133	1.191	1.276	1.402	1.564	1.654	1.819	2.054
.4000	.960	1.095	1.118	1.173	1.242	1.383	1.508	1.581	1.736	1.979
.4500	.966	1.079	1.118	1.154	1.230	1.346	1.458	1.514	1.659	1.925
.5000	.975	1.076	1.105	1.145	1.202	1.311	1.426	1.477	1.601	1.875
.5500	.951	1.054	1.059	1.111	1.159	1.252	1.351	1.414	1.546	1.776
.6000	.957	1.051	1.050	1.080	1.132	1.212	1.357	1.389	1.530	1.725
.6500	.908	1.000	1.034	1.037	1.092	1.165	1.320	1.371	1.503	1.660
.7000	.877	.972	.991	.997	1.055	1.115	1.263	1.338	1.463	1.543
.8000	.800	.886	.867	.892	.957	1.028	1.100	1.134	1.251	1.409
.8500	.662	.775	.768	.809	.871	.916	.941	.973	1.098	1.287
.9000	.558	.649	.641	.691	.797	.670	.768	.790	.929	1.197
.9500	1.564	1.715	1.659	1.713	1.742	1.857	1.868	1.894	1.941	1.854
Spoiler										
.2429	.533	.665	.678	.719	.779	.737	.708	.656	.724	.930
.4852	.624	.759	.791	.834	.850	.853	.779	.685	.717	.882
.7367	.774	.931	.981	1.031	1.012	1.009	.905	.780	.812	.903
.9796	1.245	1.417	1.450	1.531	1.540	1.560	1.515	1.421	1.471	1.447

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0125	1.187	1.019	.768	.608	.561	.604	.677	.684	.620	.448
.0250	1.141	1.019	.793	.626	.518	.442	.361	.334	.297	.265
.0500	1.095	1.038	.842	.667	.534	.414	.273	.240	.233	.227
.0750	1.095	1.038	.873	.697	.570	.452	.326	.322	.301	.287
.1000	1.119	1.051	.898	.741	.607	.505	.389	.371	.368	.367
.1500	1.104	1.048	.901	.784	.662	.579	.483	.465	.448	.442
.2000	1.092	1.089	.947	.821	.712	.639	.549	.526	.515	.496
.2500	1.113	1.120	1.000	.880	.779	.707	.618	.584	.567	.531
.3000	1.107	1.146	1.025	.920	.819	.754	.658	.635	.623	.582
.3500	1.098	1.187	1.081	.975	.883	.822	.718	.699	.681	.636
.4000	1.144	1.228	1.133	1.018	.920	.872	.774	.739	.724	.669
.4500	1.215	1.253	1.142	1.046	.963	.906	.828	.778	.764	.707
.5000	1.205	1.257	1.146	1.062	.988	.925	.837	.812	.782	.737
.5500	1.211	1.234	1.149	1.068	1.006	.928	.856	.827	.813	.752
.6000	1.236	1.241	1.146	1.068	1.009	.959	.897	.854	.834	.785
.6500	1.211	1.219	1.127	1.062	1.003	.972	.887	.851	.837	.791
.7000	1.199	1.212	1.133	1.074	1.027	.978	.912	.879	.871	.815
.8000	1.187	1.222	1.146	1.102	1.067	1.034	.969	.942	.935	.884
.8500	1.199	1.241	1.173	1.123	1.092	1.081	1.016	1.003	.994	.943
.9000	1.215	1.298	1.235	1.213	1.193	1.184	1.125	1.110	1.110	1.051
.9500	1.215	1.307	1.238	1.222	1.205	1.218	1.176	1.167	1.169	1.104
Spoiler										
.2573	1.411	1.627	1.666	1.766	1.806	1.881	1.960	1.890	1.958	1.927
.4984	1.486	1.636	1.659	1.747	1.773	1.838	1.911	1.860	1.909	1.879
.7485	1.473	1.621	1.650	1.744	1.760	1.838	1.898	1.840	1.882	1.837
.9940	1.520	1.633	1.634	1.756	1.791	1.847	1.929	1.890	1.940	1.952

TABLE VII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(c) Plain wing configuration with spoiler; $h_s = -0.050\bar{c}$; $\frac{y}{b/2} = 0.30$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.938	.924	1.548	2.364	3.033	3.906	4.276	4.168	3.637	2.657
.0125	.770	1.127	1.731	2.370	2.972	3.582	3.822	3.849	3.585	2.648
.0250	.831	1.130	1.737	2.463	3.079	3.710	3.900	3.864	3.595	2.654
.0500	.880	1.120	1.536	3.169	3.414	3.579	3.818	3.894	3.610	2.636
.0750	.892	1.108	1.254	2.697	4.389	4.782	4.273	4.046	3.622	2.648
.1000	.886	1.098	1.180	1.623	3.190	4.352	4.207	3.946	3.552	2.666
.1500	.920	1.095	1.149	1.145	1.675	2.847	3.618	3.557	3.337	2.591
.2000	.935	1.101	1.167	1.197	1.356	2.053	2.900	3.046	3.049	2.522
.2500	.899	1.092	1.152	1.204	1.276	1.695	2.386	2.633	2.794	2.466
.3000	.932	1.095	1.146	1.194	1.257	1.564	2.147	2.383	2.589	2.373
.3500	.935	1.089	1.130	1.173	1.224	1.458	1.906	2.128	2.359	2.266
.4000	.935	1.082	1.118	1.163	1.215	1.414	1.784	1.982	2.224	2.185
.4500	.960	1.079	1.118	1.151	1.208	1.386	1.731	1.921	2.135	2.140
.5000	.966	1.063	1.090	1.117	1.184	1.336	1.615	1.763	1.951	2.033
.5500	.911	1.038	1.087	1.096	1.144	1.305	1.533	1.678	1.874	1.940
.6000	.902	1.025	1.037	1.065	1.113	1.274	1.477	1.602	1.800	1.863
.6500	.889	1.003	1.012	1.037	1.083	1.230	1.408	1.520	1.705	1.791
.7500	.794	.893	.889	.917	.951	1.072	1.229	1.350	1.515	1.633
.8000	.684	.804	.805	.830	.874	.975	1.135	1.246	1.417	1.522
.8500	.574	.674	.697	.716	.791	.822	.975	1.085	1.251	1.409
.9000	.448	.535	.576	.586	.712	.579	.705	.851	1.034	1.230
.9500	1.549	1.703	1.644	1.703	1.736	1.844	1.856	1.891	1.908	1.809
Spoiler										
.2475	.464	.533	.584	.591	.662	.648	.754	.825	.921	1.118
.4909	.596	.680	.719	.731	.761	.777	.923	.952	1.012	1.181
.7396	.784	.897	.931	.944	.948	.966	1.119	1.095	1.131	1.263
.9796	1.326	1.527	1.563	1.641	1.647	1.716	1.803	1.727	1.769	1.767

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.236	1.013	.783	.756	.856	1.053	1.194	1.252	1.294	1.134
.0250	1.270	1.048	.824	.716	.715	.760	.806	.836	.862	.788
.0500	1.211	1.063	.867	.728	.647	.635	.589	.593	.586	.549
.0750	1.162	1.082	.904	.775	.669	.617	.533	.535	.528	.490
.1000	1.165	1.092	.938	.796	.696	.620	.536	.517	.494	.469
.1500	1.178	1.143	.988	.839	.733	.670	.571	.541	.524	.475
.2000	1.162	1.158	1.022	.895	.782	.704	.589	.565	.534	.487
.2500	1.175	1.196	1.059	.944	.828	.744	.649	.605	.567	.513
.3000	1.208	1.212	1.077	.969	.865	.779	.680	.638	.604	.558
.3500	1.230	1.234	1.108	1.003	.911	.825	.724	.684	.653	.600
.4000	1.245	1.238	1.124	1.025	.939	.860	.756	.724	.693	.633
.4500	1.257	1.253	1.149	1.052	.960	.891	.799	.766	.730	.669
.5000	1.227	1.263	1.146	1.065	.981	.922	.834	.803	.767	.698
.5500	1.193	1.260	1.167	1.086	1.009	.959	.872	.845	.804	.737
.6000	1.172	1.234	1.146	1.074	1.009	.956	.878	.851	.810	.744
.6500	1.159	1.228	1.139	1.083	1.024	.987	.909	.885	.856	.794
.7500	1.070	1.209	1.142	1.099	1.043	1.025	.950	.933	.917	.854
.8000	1.031	1.209	1.142	1.117	1.064	1.050	.984	.961	.948	.890
.8500	1.129	1.247	1.176	1.151	1.098	1.100	1.035	1.024	1.012	.955
.9000	1.156	1.291	1.223	1.191	1.165	1.159	1.110	1.100	1.089	1.033
.9500	1.316	1.393	1.328	1.305	1.300	1.311	1.263	1.271	1.267	1.206
Spoiler										
.2587	1.383	1.583	1.616	1.703	1.718	1.786	1.849	1.792	1.845	1.807
.5024	1.411	1.599	1.628	1.725	1.742	1.795	1.865	1.804	1.857	1.828
.7531	1.448	1.599	1.622	1.719	1.727	1.792	1.852	1.795	1.857	1.852
.9940	1.477	1.621	1.653	1.731	1.739	1.835	1.905	1.863	1.933	1.949

TABLE VII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(d) Plain wing configuration with spoiler; $h_s = -0.050\bar{c}$; $\frac{y}{b/2} = 0.43$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.530	.896	1.619	1.910	2.362	2.579	2.511	2.532	2.555	2.328
.0125	.758	1.105	1.598	1.898	2.319	2.551	2.586	2.627	2.610	2.349
.0250	.788	1.114	1.542	1.929	2.319	2.567	2.684	2.669	2.625	2.358
.0500	.825	1.101	1.511	1.984	2.386	2.660	2.646	2.675	2.638	2.367
.0750	.856	1.108	1.539	2.031	2.368	2.557	2.643	2.700	2.635	2.349
.1000	.856	1.095	1.474	2.210	2.316	2.492	2.637	2.715	2.656	2.364
.1500	.846	1.086	1.272	2.444	2.760	2.626	2.702	2.748	2.668	2.382
.2000	.877	1.089	1.173	1.960	2.950	2.947	2.768	2.730	2.631	2.358
.2500	.892	1.082	1.118	1.426	2.414	2.810	2.665	2.599	2.555	2.295
.3000	.920	1.086	1.136	1.197	2.003	2.629	2.583	2.563	2.546	2.281
.3500	.923	1.076	1.099	1.089	1.595	2.296	2.379	2.389	2.426	2.233
.4000	.902	1.070	1.096	1.074	1.389	2.078	2.260	2.298	2.365	2.188
.4500	.908	1.063	1.087	1.071	1.294	1.909	2.154	2.219	2.285	2.146
.5000	.831	1.016	1.056	1.052	1.172	1.670	1.963	2.067	2.171	2.066
.5500	.853	1.006	1.034	1.049	1.129	1.567	1.878	1.976	2.107	2.024
.6000	.865	.994	1.006	1.015	1.095	1.480	1.787	1.891	2.021	1.985
.7000	.776	.889	.904	.910	.994	1.283	1.589	1.708	1.862	1.863
.7500	.739	.823	.842	.836	.926	1.184	1.499	1.614	1.779	1.821
.8000	.641	.709	.709	.716	.794	1.047	1.339	1.480	1.656	1.761
.8500	.506	.554	.601	.559	.598	.869	1.104	1.274	1.521	1.701
.9000	1.580	1.671	1.632	1.691	1.742	1.829	1.840	1.867	1.902	1.797
.9500	1.586	1.693	1.635	1.713	1.748	1.841	1.859	1.885	1.923	1.833
Spoiler										
.2443	.433	.517	.550	.525	.463	.627	.880	1.071	1.304	1.559
.4906	.574	.677	.709	.700	.641	.740	.997	1.166	1.368	1.574
.7370	.765	.900	.931	.931	.862	.914	1.153	1.297	1.465	1.628
.9856	1.257	1.502	1.569	1.600	1.576	1.651	1.745	1.730	1.803	1.849

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.647	1.117	.824	.809	.942	1.103	1.185	1.274	1.356	1.295
.0250	1.561	1.168	.882	.781	.785	.832	.850	.894	.942	.904
.0500	1.586	1.171	.938	.812	.733	.723	.690	.693	.721	.693
.0750	1.414	1.174	.957	.821	.739	.692	.640	.641	.659	.618
.1000	1.276	1.177	.985	.855	.758	.692	.630	.620	.626	.582
.1500	1.242	1.190	1.015	.886	.788	.716	.627	.620	.607	.561
.2000	1.248	1.200	1.043	.913	.819	.732	.655	.635	.616	.573
.2500	1.218	1.215	1.077	.947	.862	.782	.683	.657	.638	.585
.3000	1.239	1.222	1.084	.963	.883	.804	.709	.672	.659	.588
.3500	1.233	1.238	1.105	1.009	.920	.841	.746	.724	.702	.645
.4000	1.267	1.241	1.130	1.028	.942	.866	.784	.748	.730	.663
.4500	1.224	1.241	1.127	1.040	.957	.894	.803	.769	.761	.696
.5000	1.224	1.234	1.139	1.037	.966	.910	.818	.793	.788	.716
.5500	1.221	1.231	1.133	1.055	.991	.928	.853	.833	.819	.752
.6000	1.205	1.228	1.136	1.065	1.009	.953	.878	.863	.843	.782
.7000	1.224	1.250	1.152	1.102	1.061	1.022	.962	.924	.923	.848
.7500	1.221	1.244	1.183	1.114	1.083	1.047	.984	.973	.963	.896
.8000	1.196	1.257	1.180	1.139	1.110	1.081	1.035	1.006	1.009	.940
.8500	1.218	1.272	1.211	1.170	1.150	1.131	1.078	1.067	1.077	1.009
.9000	1.248	1.320	1.263	1.228	1.202	1.215	1.166	1.161	1.165	1.107
.9500	1.374	1.462	1.399	1.373	1.429	1.405	1.373	1.353	1.389	1.319
Spoiler										
.2587	1.295	1.549	1.597	1.659	1.662	1.743	1.806	1.751	1.809	1.810
.5039	1.389	1.580	1.619	1.666	1.675	1.771	1.816	1.765	1.821	1.822
.7508	1.339	1.568	1.622	1.659	1.659	1.761	1.819	1.771	1.818	1.846
.9940	1.436	1.611	1.638	1.700	1.696	1.801	1.862	1.822	1.876	1.912

TABLE VII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(e) Plain wing configuration with spoiler; $h_s = -0.050\bar{c}$; $\frac{y}{b/2} = 0.55$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.267	.889	1.619	1.670	1.932	2.056	2.038	2.067	2.147	2.081
.0125	.800	1.082	1.594	1.700	1.966	2.075	2.085	2.137	2.184	2.104
.0250	.782	1.063	1.604	1.685	1.938	2.053	2.088	2.143	2.187	2.101
.0500	.791	1.067	1.437	1.694	1.926	2.103	2.110	2.171	2.193	2.090
.0750	.834	1.070	1.279	1.710	1.935	2.124	2.122	2.180	2.193	2.084
.1000	.853	1.076	1.238	1.731	1.954	2.112	2.132	2.189	2.190	2.075
.1500	.868	1.067	1.217	1.734	1.920	2.050	2.144	2.177	2.184	2.054
.2000	.880	1.073	1.198	1.830	1.886	2.040	2.144	2.171	2.178	2.060
.2500	.880	1.073	1.180	2.021	1.911	2.025	2.157	2.177	2.187	2.066
.3000	.920	1.073	1.164	2.058	2.095	2.075	2.166	2.164	2.184	2.066
.3500	.917	1.060	1.155	1.916	2.196	2.121	2.144	2.137	2.168	2.042
.4000	.911	1.051	1.130	1.694	2.153	2.100	2.100	2.101	2.144	2.030
.4500	.917	1.041	1.115	1.469	2.046	2.075	2.057	2.055	2.113	2.006
.5000	.905	1.022	1.087	1.293	1.923	2.022	2.022	2.034	2.082	1.994
.6000	.862	.953	1.009	1.018	1.598	1.838	1.890	1.912	1.990	1.934
.6500	.819	.908	.963	.938	1.469	1.744	1.828	1.864	1.938	1.898
.7000	.770	.842	.898	.864	1.343	1.648	1.749	1.781	1.865	1.812
.7500	.684	.756	.827	.762	1.193	1.533	1.655	1.696	1.822	1.845
.8500	.469	.541	.563	.420	.810	1.299	1.367	1.411	1.626	1.761
.9000	1.616	1.700	1.669	1.719	1.757	1.832	1.843	1.864	1.920	1.827
.9500	1.592	1.690	1.653	1.694	1.721	1.804	1.806	1.842	1.883	1.797
Spoiler										
.2443	.423	.495	.613	.494	.638	.988	1.291	1.359	1.511	1.689
.4881	.583	.696	.766	.656	.745	1.061	1.362	1.433	1.578	1.713
.7416	.759	.919	.969	.878	.871	1.156	1.435	1.501	1.632	1.731
.9856	1.267	1.499	1.538	1.534	1.481	1.627	1.767	1.768	1.830	1.864

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.751	1.234	.882	.864	1.012	1.199	1.317	1.374	1.491	1.498
.0250	1.757	1.238	.929	.818	.831	.910	.956	.991	1.058	1.075
.0500	1.776	1.234	.981	.839	.788	.763	.765	.775	.804	.812
.0750	1.586	1.206	.978	.833	.751	.716	.699	.687	.712	.698
.1000	1.512	1.209	1.003	.849	.770	.716	.674	.657	.675	.660
.1500	1.432	1.219	1.028	.883	.794	.735	.665	.644	.647	.630
.2000	1.395	1.234	1.059	.923	.819	.760	.693	.654	.653	.639
.2500	1.383	1.228	1.084	.950	.856	.779	.705	.672	.672	.636
.3000	1.356	1.234	1.090	.972	.886	.804	.730	.693	.693	.654
.3500	1.343	1.247	1.111	.984	.914	.844	.768	.733	.718	.681
.4000	1.325	1.253	1.133	1.006	.939	.857	.787	.757	.742	.704
.4500	1.303	1.247	1.127	1.018	.948	.875	.812	.778	.779	.734
.5000	1.294	1.266	1.136	1.043	.978	.900	.837	.806	.804	.752
.6000	1.288	1.260	1.161	1.074	1.027	.963	.922	.869	.877	.821
.6500	1.267	1.263	1.170	1.096	1.043	1.003	.956	.906	.908	.848
.7000	1.254	1.253	1.184	1.099	1.055	1.003	.975	.939	.929	.893
.7500	1.261	1.269	1.189	1.126	1.095	1.059	1.016	.982	.981	.928
.8000	1.267	1.275	1.204	1.151	1.119	1.103	1.069	1.037	1.043	.997
.8500	1.285	1.304	1.245	1.188	1.169	1.146	1.138	1.119	1.116	1.066
.9000	1.316	1.345	1.288	1.250	1.248	1.230	1.229	1.195	1.221	1.176
.9500	1.395	1.465	1.399	1.373	1.380	1.411	1.392	1.374	1.411	1.352
Spoiler										
.2584	1.357	1.593	1.638	1.672	1.659	1.734	1.800	1.745	1.809	1.816
.5039	1.314	1.586	1.650	1.675	1.656	1.749	1.803	1.765	1.833	1.822
.7508	1.339	1.608	1.653	1.672	1.600	1.749	1.810	1.783	1.854	1.843
.9940	1.342	1.618	1.678	1.694	1.678	1.764	1.840	1.807	1.888	1.903

TABLE VII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Concluded

(f) Plain wing configuration with spoiler; $h_s = -0.050\bar{c}$; $\frac{y}{b/2} = 0.72$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.731	1.031	1.531	1.481	1.546	1.633	1.699	1.682	1.751	1.770
.0125	.771	.900	1.475	1.459	1.509	1.596	1.665	1.659	1.754	1.776
.0250	.759	.959	1.444	1.459	1.506	1.596	1.684	1.667	1.751	1.764
.0500	.784	.978	1.425	1.456	1.509	1.605	1.696	1.691	1.760	1.767
.0750	.809	.994	1.441	1.463	1.509	1.621	1.711	1.700	1.763	1.770
.1000	.787	.978	1.403	1.438	1.491	1.599	1.699	1.691	1.757	1.764
.1500	.825	.991	1.322	1.434	1.487	1.612	1.718	1.703	1.757	1.755
.2000	.865	1.013	1.244	1.441	1.503	1.618	1.730	1.718	1.754	1.746
.2500	.878	1.019	1.169	1.428	1.497	1.605	1.727	1.700	1.754	1.743
.3000	.884	1.006	1.116	1.416	1.466	1.593	1.711	1.688	1.733	1.728
.3500	.903	1.016	1.116	1.419	1.466	1.590	1.696	1.682	1.730	1.731
.4000	.906	1.006	1.091	1.400	1.454	1.566	1.675	1.659	1.715	1.716
.4500	.900	1.003	1.078	1.400	1.435	1.538	1.662	1.641	1.699	1.713
.5000	.909	1.000	1.066	1.438	1.423	1.520	1.644	1.629	1.684	1.701
.5500	.893	.981	1.041	1.434	1.399	1.492	1.629	1.599	1.669	1.698
.6000	.884	.966	1.022	1.447	1.380	1.459	1.604	1.584	1.648	1.677
.6500	.890	.953	1.006	1.456	1.368	1.443	1.592	1.561	1.636	1.674
.7000	.919	.972	1.022	1.484	1.383	1.465	1.589	1.555	1.629	1.671
.7500	.975	1.013	1.059	1.531	1.445	1.526	1.607	1.549	1.623	1.680
.8000	1.047	1.075	1.134	1.606	1.567	1.621	1.647	1.587	1.645	1.692
.8500	1.138	1.160	1.194	1.616	1.675	1.728	1.699	1.620	1.684	1.734
.9000	1.198	1.229	1.256	1.691	1.718	1.758	1.757	1.662	1.708	1.761

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.743	1.411	.866	.800	.840	.920	1.034	1.056	1.155	1.202
.0250	1.743	1.361	.897	.788	.767	.804	.871	.884	.955	1.000
.0500	1.737	1.288	.931	.791	.724	.731	.758	.757	.784	.810
.0750	1.702	1.323	.975	.838	.748	.722	.727	.703	.724	.740
.1000	1.718	1.295	.991	.863	.758	.722	.712	.685	.699	.695
.1500	1.759	1.282	1.000	.878	.779	.728	.705	.662	.672	.668
.2000	1.809	1.248	1.016	.913	.816	.758	.733	.685	.681	.665
.2500	1.734	1.210	1.034	.928	.828	.786	.751	.703	.696	.689
.3000	1.574	1.194	1.050	.959	.862	.826	.770	.724	.720	.710
.3500	1.417	1.223	1.075	.994	.896	.850	.797	.745	.745	.737
.4000	1.248	1.198	1.078	1.000	.908	.862	.825	.774	.775	.755
.4500	1.169	1.191	1.091	1.031	.939	.890	.856	.798	.793	.779
.5000	1.141	1.191	1.106	1.047	.954	.917	.868	.819	.818	.792
.5500	1.116	1.194	1.109	1.063	.975	.945	.896	.840	.839	.813
.6000	1.129	1.191	1.119	1.081	1.003	.972	.929	.872	.866	.855
.6500	1.151	1.204	1.138	1.103	1.021	.997	.954	.899	.897	.882
.7000	1.147	1.179	1.116	1.103	1.037	1.015	.988	.923	.939	.918
.7500	1.157	1.191	1.128	1.119	1.070	1.052	1.034	.973	.979	.961
.8000	1.147	1.176	1.131	1.147	1.098	1.092	1.077	1.018	1.037	1.024
.8500	1.144	1.194	1.150	1.181	1.153	1.150	1.138	1.083	1.107	1.094
.9000	1.138	1.201	1.159	1.231	1.215	1.226	1.227	1.175	1.192	1.187
.9500	1.144	1.194	1.163	1.303	1.310	1.330	1.346	1.282	1.301	1.296

TABLE VIII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE

(a) Plain wing configuration with spoiler; $h_s = -0.100\bar{c}$; $\frac{y}{b/2} = 0$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Fuselage										
.0000	.009	.000	.009	.000	.000	.027	.046	.082	.111	**.134**
.0500	.799	.831	.908	.946	1.003	.994	1.043	1.073	1.066	**1.095**
.1000	.873	.920	.985	1.019	1.059	1.048	1.086	1.125	1.120	**1.149**
.1500	.957	.988	1.046	1.076	1.112	1.082	1.119	1.140	1.136	**1.168**
.2000	.991	1.028	1.073	1.105	1.109	1.094	1.116	1.128	1.123	**1.143**
.2500	1.040	1.068	1.098	1.118	1.127	1.100	1.123	1.140	1.130	**1.156**
.3000	1.058	1.074	1.098	1.108	1.112	1.091	1.095	1.116	1.111	**1.137**
.3500	1.068	1.052	1.083	1.102	1.103	1.066	1.095	1.107	1.090	**1.146**
.4000	1.021	1.018	1.052	1.048	1.062	1.054	1.083	1.097	1.105	**1.149**
.4500	.997	1.022	1.043	1.073	1.100	1.094	1.153	1.170	1.193	**1.256**
.5000	.984	1.012	1.061	1.105	1.143	1.154	1.251	1.280	1.313	**1.488**
.5500	.975	1.018	1.073	1.133	1.183	1.205	1.291	1.319	1.410	**1.741**
.6000	.978	1.022	1.089	1.130	1.180	1.202	1.270	1.298	1.386	**1.835**
.6500	.957	1.012	1.067	1.118	1.162	1.166	1.215	1.228	1.280	**1.634**
.7000	.954	.991	1.046	1.092	1.137	1.130	1.159	1.173	1.211	**1.479**
.7500	.957	.975	1.018	1.051	1.100	1.091	1.141	1.167	1.217	**1.390**
.8000	.957	.969	1.012	1.016	1.072	1.103	1.153	1.176	1.226	**1.305**
.8500	1.037	1.046	1.083	1.089	1.152	1.121	1.172	1.210	1.241	**1.262**
.9000	1.154	1.178	1.208	1.210	1.205	1.169	1.196	1.210	1.217	**1.223**
.9500	1.126	1.169	1.193	1.181	1.168	1.097	1.147	1.155	1.169	**1.177**
.9940	1.194	1.203	1.202	1.207	1.183	1.142	1.132	1.155	1.163	**1.165**

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Fuselage										
.0500	.886	.846	.791	.730	.652	.550	.475	.456	.419	**.402**
.1000	.972	.942	.899	.845	.752	.656	.583	.568	.521	**.500**
.1500	1.018	1.003	.948	.908	.829	.743	.684	.648	.605	**.592**
.2000	1.046	1.040	1.009	.962	.901	.807	.739	.711	.660	**.649**
.2500	1.062	1.052	1.040	1.006	.944	.861	.800	.772	.717	**.704**
.3000	1.074	1.074	1.064	1.035	.975	.894	.837	.812	.759	**.753**
.3500	1.058	1.052	1.064	1.038	.991	.903	.862	.839	.786	**.777**
.4000	1.003	1.012	.994	.981	.941	.855	.810	.787	.732	**.723**
.4500	.981	.975	.975	.943	.891	.810	.748	.727	.675	**.671**
.5000	.984	.960	.951	.914	.848	.752	.693	.666	.611	**.595**
.5500	1.009	.982	.929	.886	.814	.719	.653	.611	.569	**.555**
.6000	.978	.945	.892	.841	.764	.671	.613	.584	.527	**.512**
.7500	1.244	1.222	1.184	1.149	1.100	1.000	.951	.936	.886	**.887**
.9500	1.046	1.055	1.067	1.048	1.028	.976	.957	.967	.919	**.930**
.9940	.991	1.111	1.153	1.146	1.124	1.094	1.123	1.134	1.093	**1.104**

TABLE VIII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(b) Plain wing configuration with spoiler; $h_s = -0.100\bar{c}$; $\frac{y}{b/2} = 0.21$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.386	.830	1.256	2.489	4.163	6.359	**8.193**	8.675	4.766	**2.958**
.0125	.726	1.028	1.422	2.119	5.354	8.991	10.354	9.104	4.479	**2.710**
.0250	.760	1.043	1.313	1.749	2.935	4.716	7.217	7.712	4.305	**2.749**
.0500	.841	1.055	1.259	1.505	1.685	2.244	3.557	4.712	3.650	**2.544**
.0750	.857	1.037	1.203	1.407	1.577	1.888	2.554	3.354	3.275	**2.516**
.1000	.878	1.052	1.197	1.367	1.509	1.759	2.318	2.882	3.079	**2.462**
.1500	.900	1.046	1.172	1.290	1.413	1.613	1.979	2.336	2.711	**2.347**
.2000	.913	1.046	1.159	1.254	1.361	1.544	1.815	2.103	2.436	**2.278**
.2500	.928	1.043	1.146	1.223	1.302	1.469	1.721	1.932	2.165	**2.205**
.3000	.925	1.031	1.128	1.193	1.265	1.416	1.633	1.774	1.997	**2.139**
.3500	.913	1.018	1.088	1.156	1.231	1.369	1.515	1.662	1.823	**2.084**
.4000	.913	.997	1.075	1.125	1.182	1.347	1.445	1.587	1.735	**2.033**
.4500	.900	.991	1.053	1.098	1.154	1.316	1.385	1.516	1.640	**1.982**
.5000	.891	.960	1.028	1.055	1.114	1.259	1.357	1.454	1.589	**1.927**
.5500	.847	.910	.963	1.009	1.037	1.181	1.273	1.373	1.497	**1.843**
.6000	.813	.864	.916	.951	.981	1.131	1.224	1.336	1.467	**1.782**
.6500	.744	.787	.834	.862	.926	1.053	1.167	1.286	1.415	**1.710**
.7000	.657	.682	.731	.758	.889	.956	1.097	1.211	1.305	**1.607**
.8000	.483	.552	.578	.612	.805	.734	.839	.929	1.146	**1.529**
.8500	.455	.546	.584	.648	.775	.581	.733	.857	1.180	**1.465**
.9000	.533	.577	.625	.661	.497	.647	.676	.658	.680	**.846**
.9500	1.536	1.617	1.575	1.566	1.580	1.738	1.848	1.932	1.942	**1.979**
Spoiler										
.2429	.611	.637	.684	.711	.627	.595	.537	.611	.666	**.781**
.4852	.555	.597	.708	.784	.786	.550	.436	.508	.584	**.720**
.7367	.339	.412	.604	.721	.714	.535	.475	.407	.410	**.558**
.9796	.762	.751	.871	.953	.891	.885	.871	.748	.672	**.787**

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.190	.994	.778	.627	.565	.584	.633	.658	.521	**.405**
.0250	1.143	.997	.816	.651	.509	.419	.339	.326	.277	**.230**
.0500	1.084	1.003	.853	.691	.531	.416	.285	.242	.232	**.236**
.0750	1.106	1.009	.878	.722	.568	.438	.339	.311	.302	**.308**
.1000	1.125	1.018	.909	.752	.620	.488	.388	.373	.366	**.375**
.1500	1.106	1.034	.928	.795	.663	.563	.485	.463	.457	**.459**
.2000	1.096	1.058	.963	.844	.713	.638	.539	.544	.515	**.501**
.2500	1.112	1.096	1.016	.893	.772	.700	.615	.606	.564	**.556**
.3000	1.093	1.133	1.053	.930	.815	.753	.661	.655	.616	**.592**
.3500	1.096	1.179	1.106	.997	.867	.822	.730	.714	.665	**.662**
.4000	1.146	1.210	1.147	1.040	.920	.863	.773	.758	.707	**.689**
.4500	1.212	1.228	1.172	1.061	.966	.900	.806	.795	**.756**	**.734**
.5000	1.206	1.237	1.178	1.083	.988	.928	.842	.829	.777	**.755**
.5500	1.221	1.247	1.188	1.083	.997	1.063	.857	.848	.802	**.779**
.6000	1.255	1.228	1.188	1.092	1.018	.963	.885	.876	.826	**.792**
.6500	1.224	1.207	1.172	1.079	1.012	.978	.888	.876	.832	**.816**
.7000	1.202	1.210	1.181	1.095	1.046	.994	.909	.907	.863	**.834**
.8000	1.202	1.225	1.200	1.131	1.077	1.044	.979	.981	.927	**.906**
.8500	1.224	1.241	1.222	1.165	1.114	1.100	1.045	1.031	.985	**.955**
.9000	1.243	1.318	1.300	1.266	1.213	1.213	1.157	1.155	1.088	**1.057**
.9500	1.240	1.321	1.288	1.245	1.222	1.231	1.212	1.211	1.149	**1.088**
Spoiler										
.2573	1.620	1.603	1.534	1.546	1.596	1.622	1.767	1.836	1.816	**1.893**
.4984	1.645	1.603	1.537	1.562	1.609	1.625	1.773	1.836	1.810	**1.906**
.7485	1.645	1.609	1.543	1.565	1.615	1.622	1.757	1.815	1.795	**1.890**
.9940	1.663	1.628	1.555	1.591	1.640	1.640	1.767	1.827	1.804	**1.884**

TABLE VIII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(c) Plain wing configuration with spoiler; $h_s = -0.100\bar{c}$; $\frac{y}{b/2} = 0.30$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	2.078	.883	1.538	2.266	2.882	3.566	3.875	3.814	3.110	2.444
.0125	.741	1.083	1.722	2.278	2.845	3.306	3.527	3.609	3.104	2.423
.0250	.794	1.074	1.706	2.367	2.944	3.406	3.542	3.640	3.110	2.423
.0500	.847	1.065	1.556	3.027	3.287	3.316	3.530	3.653	3.125	2.387
.0750	.860	1.052	1.266	2.517	4.148	4.134	3.769	3.687	3.119	2.411
.1000	.850	1.052	1.172	1.523	3.015	3.963	3.648	3.566	3.073	2.447
.1500	.872	1.040	1.141	1.125	1.602	2.866	3.224	3.255	2.951	2.393
.2000	.891	1.037	1.141	1.165	1.318	2.103	2.682	2.867	2.772	2.362
.2500	.841	1.021	1.116	1.159	1.241	1.719	2.300	2.553	2.589	2.365
.3000	.869	1.009	1.103	1.138	1.216	1.569	2.094	2.345	2.448	2.284
.3500	.869	.991	1.069	1.107	1.173	1.450	1.863	2.093	2.262	2.220
.4000	.860	.966	1.047	1.089	1.142	1.388	1.727	1.975	2.140	2.154
.4500	.869	.950	1.025	1.058	1.117	1.344	1.667	1.895	2.067	2.103
.5000	.844	.913	.975	1.009	1.077	1.278	1.548	1.742	1.881	2.003
.5500	.773	.864	.913	.948	1.012	1.219	1.457	1.646	1.805	1.943
.6000	.713	.796	.844	.878	.960	1.163	1.388	1.556	1.707	1.888
.6500	.645	.707	.741	.774	.880	1.059	1.279	1.432	1.604	1.822
.7500	.427	.485	.575	.572	.738	.750	.967	1.146	1.345	1.656
.8000	.402	.497	.553	.605	.753	.581	.758	.919	1.186	1.532
.8500	.592	.707	.750	.875	1.102	.769	.564	.836	1.271	1.480
.9000	.190	.225	.334	.260	.133	.100	.339	.553	.774	1.103
.9500	1.529	1.611	1.572	1.547	1.602	1.684	1.766	1.851	1.814	1.819
Spoiler										
.2475	.420	.446	.518	.518	.298	.290	.371	.590	.825	1.034
.4909	.290	.317	.423	.410	.205	.211	.337	.547	.798	1.055
.7396	.105	.114	.218	.171	.062	.100	.460	.608	.774	1.043
.9796	.867	.840	.822	.806	.798	.776	1.119	1.164	1.208	1.378

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.293	.991	.791	.746	.827	.991	1.109	1.199	1.189	1.094
.0250	1.293	1.025	.838	.722	.685	.725	.751	.804	.811	.773
.0500	1.230	1.037	.881	.740	.642	.609	.564	.590	.573	.553
.0750	1.177	1.058	.913	.771	.657	.584	.530	.528	.506	.498
.1000	1.177	1.071	.944	.801	.673	.613	.524	.512	.491	.468
.1500	1.190	1.114	.997	.847	.728	.647	.561	.553	.512	.501
.2000	1.168	1.126	1.034	.902	.775	.688	.588	.568	.527	.508
.2500	1.190	1.170	1.078	.948	.824	.738	.639	.609	.561	.544
.3000	1.227	1.188	1.100	.976	.861	.775	.673	.658	.601	.577
.3500	1.249	1.207	1.144	1.012	.910	.819	.718	.699	.643	.616
.4000	1.262	1.222	1.159	1.034	.938	.863	.754	.745	.683	.653
.4500	1.277	1.241	1.178	1.064	.972	.894	.806	.780	.717	.695
.5000	1.255	1.237	1.181	1.083	.991	.922	.836	.814	.756	.731
.5500	1.212	1.253	1.203	1.107	1.018	.959	.870	.857	.793	.770
.6000	1.184	1.222	1.181	1.104	1.012	.966	.876	.870	.811	.795
.6500	1.174	1.222	1.197	1.119	1.034	.988	.909	.904	.851	.819
.7500	1.096	1.216	1.200	1.131	1.055	1.025	.961	.966	.906	.882
.8000	1.050	1.216	1.200	1.141	1.065	1.056	.991	.994	.939	.912
.8500	1.149	1.265	1.234	1.187	1.117	1.116	1.048	1.050	.994	.967
.9000	1.193	1.305	1.284	1.226	1.179	1.181	1.127	1.131	1.076	1.027
.9500	1.361	1.404	1.369	1.327	1.312	1.325	1.282	1.301	1.244	1.178
Spoiler										
.2587	1.614	1.585	1.521	1.537	1.572	1.559	1.662	1.705	1.678	1.726
.5024	1.626	1.591	1.534	1.549	1.590	1.571	1.699	1.757	1.732	1.808
.7531	1.620	1.591	1.527	1.556	1.596	1.589	1.724	1.775	1.756	1.839
.9940	1.654	1.634	1.546	1.591	1.609	1.631	1.757	1.809	1.804	1.890

TABLE VIII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(d) Plain wing configuration with spoiler; $h_s = -0.100\bar{c}$; $\frac{y}{b/2} = 0.43$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.607	.842	1.566	1.801	2.139	2.313	2.273	2.404	2.415	2.248
.0125	.716	1.006	1.541	1.783	2.102	2.309	2.369	2.485	2.436	2.260
.0250	.741	1.040	1.503	1.810	2.105	2.356	2.415	2.510	2.430	2.275
.0500	.776	1.040	1.469	1.856	2.142	2.394	2.427	2.531	2.421	2.272
.0750	.819	1.037	1.475	1.887	2.123	2.338	2.424	2.541	2.424	2.260
.1000	.801	1.028	1.409	2.037	2.065	2.300	2.421	2.541	2.424	2.272
.1500	.779	1.006	1.247	2.202	2.315	2.375	2.448	2.541	2.421	2.299
.2000	.810	1.012	1.147	1.804	2.438	2.500	2.421	2.494	2.400	2.272
.2500	.810	.984	1.084	1.346	2.145	2.366	2.321	2.385	2.339	2.226
.3000	.838	.978	1.075	1.122	1.907	2.281	2.254	2.333	2.332	2.229
.3500	.825	.947	1.025	1.009	1.605	2.053	2.115	2.205	2.238	2.181
.4000	.791	.917	.991	.972	1.438	1.913	2.030	2.149	2.183	2.166
.4500	.782	.883	.956	.951	1.296	1.784	1.939	2.062	2.122	2.127
.5000	.682	.821	.888	.896	1.154	1.600	1.806	1.932	2.034	2.069
.5500	.657	.768	.825	.847	1.068	1.481	1.715	1.876	1.976	2.021
.6000	.604	.691	.738	.761	.991	1.369	1.633	1.795	1.893	1.982
.7000	.386	.451	.519	.483	.663	1.028	1.373	1.513	1.677	1.867
.7500	.402	.441	.478	.477	.651	.931	1.185	1.311	1.558	1.825
.8000	.449	.472	.525	.572	1.000	1.031	.951	1.118	1.512	1.822
.8500	.570	.645	.644	.654	.447	.728	.885	1.186	1.518	1.740
.9000	1.573	1.595	1.575	1.538	1.543	1.634	1.654	1.696	1.692	1.677
.9500	1.592	1.611	1.594	1.569	1.562	1.666	1.688	1.730	1.729	1.749
Spoiler										
.2443	.481	.502	.475	.498	.367	.462	.659	.930	1.250	1.476
.4906	.407	.400	.350	.435	.304	.396	.610	.836	1.166	1.424
.7370	.204	.188	.153	.187	.118	.260	.699	.927	1.187	1.433
.9856	.926	.935	.902	.914	.795	.834	1.236	1.420	1.524	1.692

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.760	1.126	.828	.795	.870	1.022	1.127	1.218	1.287	1.293
.0250	1.679	1.157	.897	.783	.744	.788	.818	.879	.906	.924
.0500	1.776	1.160	.947	.813	.707	.694	.661	.693	.707	.713
.0750	1.545	1.142	.981	.838	.725	.678	.621	.634	.634	.646
.1000	1.333	1.151	1.003	.862	.750	.688	.624	.618	.613	.607
.1500	1.230	1.170	1.038	.905	.787	.719	.642	.627	.613	.598
.2000	1.262	1.179	1.069	.920	.821	.738	.661	.634	.619	.598
.2500	1.221	1.194	1.100	.966	.846	.784	.691	.671	.640	.640
.3000	1.258	1.194	1.113	.982	.876	.800	.712	.690	.665	.637
.3500	1.237	1.225	1.150	1.027	.917	.844	.770	.736	.701	.677
.4000	1.286	1.225	1.169	1.049	.947	.888	.794	.767	.738	.695
.4500	1.255	1.231	1.169	1.064	.966	.900	.818	.789	.744	.716
.5000	1.243	1.222	1.169	1.073	.978	.913	.839	.820	.781	.746
.5500	1.243	1.225	1.175	1.083	1.000	.947	.870	.851	.811	.782
.6000	1.240	1.231	1.175	1.095	1.015	.972	.900	.891	.848	.816
.7000	1.249	1.250	1.225	1.138	1.068	1.028	.954	.944	.912	.894
.7500	1.262	1.262	1.231	1.153	1.099	1.066	.997	.985	.954	.930
.8000	1.237	1.265	1.241	1.180	1.120	1.100	1.036	1.037	1.006	.967
.8500	1.268	1.299	1.269	1.214	1.154	1.134	1.094	1.084	1.058	1.033
.9000	1.296	1.345	1.328	1.269	1.216	1.228	1.176	1.180	1.146	1.124
.9500	1.424	1.466	1.453	1.404	1.355	1.384	1.354	1.360	1.329	1.311
Spoiler										
.2587	1.605	1.591	1.552	1.588	1.572	1.568	1.610	1.642	1.611	1.668
.5039	1.642	1.622	1.570	1.613	1.578	1.586	1.647	1.672	1.669	1.726
.7508	1.623	1.612	1.567	1.607	1.596	1.574	1.647	1.705	1.684	1.759
.9940	1.666	1.646	1.576	1.635	1.606	1.604	1.696	1.769	1.795	1.896

TABLE VIII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(e) Plain wing configuration with spoiler; $h_s = -0.100\bar{c}$; $\frac{y}{b/2} = 0.55$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.327	.858	1.519	1.520	1.676	1.825	1.860	1.972	2.028	2.081
.0125	.769	.960	1.463	1.538	1.691	1.853	1.906	2.010	2.055	2.091
.0250	.726	.954	1.447	1.517	1.673	1.847	1.906	2.013	2.055	2.078
.0500	.735	.960	1.294	1.517	1.663	1.900	1.924	2.025	2.037	2.075
.0750	.782	.978	1.188	1.523	1.676	1.878	1.939	2.044	2.049	2.066
.1000	.797	.966	1.156	1.529	1.682	1.884	1.951	2.047	2.031	2.054
.1500	.801	.963	1.119	1.505	1.642	1.856	1.939	2.022	2.006	2.048
.2000	.797	.947	1.088	1.520	1.614	1.831	1.927	2.000	2.000	2.048
.2500	.788	.929	1.059	1.630	1.626	1.831	1.915	1.982	2.000	2.048
.3000	.807	.907	1.028	1.642	1.676	1.816	1.897	1.966	2.025	2.042
.3500	.788	.883	.984	1.547	1.673	1.800	1.870	1.941	1.994	2.030
.4000	.760	.836	.941	1.401	1.623	1.750	1.830	1.913	1.985	2.018
.4500	.726	.793	.894	1.260	1.568	1.709	1.791	1.879	1.957	2.006
.5000	.676	.728	.825	1.153	1.497	1.641	1.730	1.851	1.927	1.985
.6000	.502	.534	.678	.875	1.210	1.425	1.582	1.696	1.808	1.909
.6500	.433	.466	.584	.716	1.108	1.359	1.482	1.587	1.735	1.870
.7000	.445	.481	.553	.673	1.099	1.344	1.351	1.395	1.399	1.384
.7500	.495	.531	.569	.731	1.315	1.544	1.270	1.364	1.640	1.846
.8500	.277	.256	.316	.263	.685	.997	1.015	1.236	1.546	1.782
.9000	1.663	1.673	1.684	1.602	1.555	1.663	1.636	1.680	1.701	1.713
.9500	1.651	1.679	1.688	1.602	1.565	1.656	1.624	1.671	1.659	1.704
Spoiler										
.2443	.444	.458	.478	.451	.721	.949	1.064	1.213	1.443	1.613
.4881	.309	.302	.331	.302	.696	.946	1.021	1.131	1.376	1.573
.7416	.127	.126	.172	.143	.568	.891	1.083	1.161	1.334	1.540
.9856	.883	.898	.889	.832	1.028	1.314	1.500	1.550	1.593	1.723

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.835	1.299	.894	.841	.920	1.072	1.194	1.317	1.424	1.492
.0250	1.875	1.262	.953	.820	.799	.844	.897	.969	1.040	1.075
.0500	1.981	1.247	1.000	.859	.768	.741	.733	.777	.799	.831
.0750	1.922	1.216	1.016	.850	.750	.709	.667	.693	.704	.725
.1000	1.875	1.213	1.044	.875	.772	.713	.664	.658	.662	.689
.1500	1.685	1.219	1.069	.911	.809	.722	.661	.668	.649	.650
.2000	1.508	1.219	1.097	.948	.830	.756	.679	.671	.665	.653
.2500	1.392	1.231	1.116	.979	.867	.794	.715	.705	.686	.671
.3000	1.333	1.237	1.138	.994	.901	.813	.730	.721	.704	.680
.3500	1.315	1.244	1.166	1.031	.935	.853	.773	.761	.735	.704
.4000	1.315	1.250	1.188	1.055	.954	.881	.800	.783	.762	.725
.4500	1.293	1.250	1.181	1.064	.975	.900	.818	.811	.784	.761
.5000	1.293	1.265	1.194	1.083	.997	.928	.848	.839	.817	.788
.6000	1.305	1.275	1.225	1.122	1.046	.988	.903	.913	.875	.846
.6500	1.308	1.281	1.238	1.150	1.083	1.016	.951	.947	.918	.882
.7000	1.299	1.281	1.234	1.153	1.089	1.053	.967	.978	.951	.915
.7500	1.305	1.296	1.263	1.193	1.117	1.088	1.015	1.019	1.000	.967
.8000	1.318	1.305	1.291	1.214	1.154	1.138	1.064	1.068	1.046	1.027
.8500	1.355	1.345	1.325	1.254	1.194	1.188	1.133	1.143	1.116	1.091
.9000	1.389	1.383	1.381	1.309	1.253	1.256	1.209	1.221	1.201	1.184
.9500	1.467	1.500	1.497	1.425	1.355	1.381	1.339	1.367	1.351	1.341
Spoiler										
.2584	1.666	1.674	1.632	1.670	1.600	1.556	1.592	1.605	1.590	1.643
.5039	1.676	1.686	1.656	1.686	1.618	1.589	1.616	1.648	1.614	1.665
.7508	1.691	1.689	1.662	1.699	1.621	1.604	1.635	1.654	1.636	1.692
.9940	1.697	1.708	1.681	1.721	1.668	1.680	1.730	1.760	1.762	1.845

TABLE VIII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Concluded

(f) Plain wing configuration with spoiler; $h_s = -0.100\bar{c}$; $\frac{y}{b/2} = 0.72$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.929	1.579	1.340	1.172	1.277	1.308	1.463	1.544	1.614	1.726
.0125	.833	.757	1.264	1.153	1.242	1.272	1.441	1.535	1.605	1.723
.0250	.756	.785	1.172	1.133	1.242	1.272	1.448	1.535	1.599	1.723
.0500	.728	.809	1.104	1.118	1.236	1.269	1.448	1.544	1.608	1.723
.0750	.744	.828	1.067	1.108	1.236	1.272	1.463	1.544	1.599	1.710
.1000	.741	.815	1.015	1.089	1.214	1.245	1.438	1.535	1.593	1.695
.1500	.753	.828	.981	1.054	1.205	1.242	1.445	1.538	1.587	1.689
.2000	.759	.828	.963	1.064	1.199	1.230	1.454	1.532	1.578	1.677
.2500	.762	.822	.917	1.035	1.174	1.199	1.429	1.517	1.566	1.659
.3000	.753	.791	.880	1.013	1.143	1.157	1.405	1.496	1.554	1.650
.3500	.750	.772	.862	.991	1.112	1.133	1.399	1.487	1.536	1.640
.4000	.731	.748	.819	.959	1.081	1.103	1.362	1.453	1.515	1.616
.4500	.707	.717	.800	.943	1.065	1.100	1.346	1.435	1.497	1.613
.5000	.682	.686	.800	.997	1.106	1.121	1.325	1.414	1.482	1.604
.5500	.713	.711	.840	1.080	1.159	1.169	1.305	1.377	1.467	1.598
.6000	.802	.803	.880	1.191	1.252	1.239	1.322	1.386	1.464	1.592
.6500	.950	.951	.978	1.454	1.454	1.429	1.392	1.429	1.485	1.604
.7000	1.197	1.206	1.199	1.403	1.454	1.387	1.451	1.511	1.527	1.628
.7500	1.657	1.646	1.558	1.330	1.370	1.372	1.420	1.465	1.521	1.634
.8000	1.342	1.326	1.395	1.556	1.596	1.526	1.564	1.581	1.584	1.674
.8500	1.429	1.369	1.457	1.689	1.587	1.514	1.549	1.566	1.581	1.674
.9000	1.626	1.569	1.607	1.721	1.615	1.526	1.558	1.572	1.578	1.662

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.944	1.828	.954	.813	.792	.795	.923	1.006	1.069	1.159
.0250	1.944	1.732	.978	.838	.761	.722	.804	.857	.895	.967
.0500	1.953	1.600	1.003	.870	.745	.674	.712	.745	.750	.793
.0750	1.978	1.609	1.052	.905	.783	.683	.693	.705	.711	.732
.1000	2.009	1.563	1.055	.927	.789	.692	.693	.696	.675	.701
.1500	2.089	1.505	1.073	.959	.829	.716	.684	.690	.663	.671
.2000	2.200	1.449	1.086	.987	.854	.749	.721	.708	.678	.683
.2500	2.200	1.360	1.101	1.003	.898	.773	.730	.736	.684	.704
.3000	2.098	1.308	1.116	1.041	.919	.819	.770	.760	.714	.729
.3500	1.947	1.283	1.132	1.048	.957	.837	.797	.787	.747	.753
.4000	1.660	1.274	1.144	1.080	.975	.855	.822	.812	.768	.781
.4500	1.401	1.249	1.150	1.095	1.003	.894	.859	.839	.789	.805
.5000	1.241	1.258	1.165	1.121	1.031	.915	.883	.866	.825	.832
.5500	1.176	1.249	1.181	1.140	1.059	.940	.905	.897	.843	.851
.6000	1.173	1.265	1.196	1.172	1.090	.973	.935	.927	.880	.887
.6500	1.173	1.265	1.205	1.184	1.115	1.006	.969	.967	.907	.921
.7000	1.182	1.252	1.205	1.194	1.118	1.024	1.000	.985	.937	.963
.7500	1.204	1.262	1.218	1.210	1.143	1.057	1.034	1.024	.982	1.009
.8000	1.222	1.268	1.239	1.248	1.186	1.088	1.086	1.070	1.027	1.061
.8500	1.256	1.302	1.279	1.283	1.236	1.342	1.129	1.131	1.108	1.125
.9000	1.324	1.338	1.322	1.346	1.308	1.206	1.208	1.204	1.181	1.220
.9500	1.389	1.403	1.395	1.435	1.376	1.281	1.288	1.295	1.274	1.332

TABLE IX.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE

(a) Double slotted flap configuration with spoiler; $h_s = -0.005\bar{c}$; $\frac{y}{b/2} = 0$

Upper surface

x/c	C_p for -									
	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Fuselage										
.0000	.015	.000	.009	.000	.009	.039	.050	.075	.103	.136
.0500	.790	.823	.873	.917	.979	1.015	1.035	1.045	1.062	1.077
.1000	.884	.911	.949	.991	1.037	1.068	1.074	1.104	1.127	1.145
.1500	.951	.972	1.009	1.042	1.095	1.095	1.106	1.128	1.127	1.148
.2000	1.000	1.021	1.039	1.068	1.104	1.113	1.112	1.122	1.121	1.136
.2500	1.043	1.064	1.072	1.095	1.125	1.122	1.121	1.137	1.142	1.142
.3000	1.073	1.076	1.082	1.095	1.116	1.089	1.103	1.110	1.115	1.113
.3500	1.067	1.064	1.072	1.074	1.107	1.095	1.092	1.104	1.115	1.127
.4000	1.040	1.031	1.045	1.048	1.052	1.077	1.036	1.093	1.115	1.136
.4500	1.043	1.049	1.060	1.089	1.122	1.127	1.183	1.209	1.233	1.249
.5000	1.046	1.064	1.103	1.158	1.210	1.240	1.290	1.334	1.387	1.533
.5500	1.070	1.098	1.154	1.220	1.287	1.305	1.366	1.406	1.540	1.867
.6000	1.104	1.141	1.199	1.268	1.308	1.329	1.369	1.373	1.572	1.923
.6500	1.156	1.193	1.239	1.292	1.339	1.320	1.328	1.316	1.454	1.781
.7000	1.198	1.238	1.278	1.298	1.351	1.326	1.322	1.298	1.404	1.701
.7500	1.262	1.281	1.308	1.321	1.372	1.353	1.363	1.373	1.434	1.627
.8000	1.302	1.324	1.338	1.354	1.403	1.392	1.419	1.448	1.484	1.583
.8500	1.308	1.312	1.308	1.336	1.384	1.400	1.443	1.493	1.525	1.536
.9000	1.232	1.226	1.217	1.244	1.287	1.303	1.354	1.403	1.463	1.468
.9500	1.171	1.150	1.148	1.161	1.223	1.237	1.286	1.328	1.395	1.414
.9940	1.210	1.187	1.166	1.173	1.210	1.246	1.292	1.337	1.404	1.417

Lower surface

x/c	C_p for -									
	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Fuselage										
.0500	.875	.820	.758	.696	.637	.540	.481	.445	.431	.379
.1000	.960	.908	.864	.806	.741	.647	.570	.552	.519	.491
.1500	1.003	.960	.903	.866	.814	.727	.655	.630	.596	.562
.2000	1.037	1.000	.946	.905	.863	.777	.726	.696	.652	.624
.2500	1.052	1.021	.982	.946	.910	.831	.767	.740	.705	.681
.3000	1.064	1.031	1.000	.967	.942	.863	.811	.788	.738	.710
.3500	1.049	1.018	.997	.976	.951	.875	.820	.809	.764	.740
.4000	.991	.957	.930	.905	.887	.816	.767	.749	.702	.675
.4500	.948	.924	.891	.854	.829	.760	.705	.681	.652	.621
.5000	.918	.881	.843	.795	.771	.674	.631	.597	.550	.539
.5500	.887	.850	.801	.756	.713	.623	.558	.534	.502	.485
.6000	.826	.777	.728	.667	.631	.549	.499	.481	.460	.429
.7500	.933	.887	.843	.818	.802	.745	.693	.666	.667	.654
.9500	1.079	1.083	1.103	1.140	1.174	1.148	1.145	1.179	1.224	1.259
.9940	1.046	1.135	1.163	1.193	1.250	1.273	1.307	1.349	1.392	1.397

TABLE IX.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(b) Double slotted flap configuration with spoiler; $h_s = -0.005\bar{c}$; $\frac{y}{b/2} = 0.21$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0000	.790	.971	1.978	3.352	5.441	7.695	10.098	7.703	3.558	2.698
.0125	.928	1.277	1.734	3.464	7.629	10.569	12.050	7.413	3.207	2.458
.0250	.947	1.223	1.571	2.602	3.719	6.222	8.863	6.778	3.224	2.483
.0500	.997	1.201	1.464	1.617	2.046	2.701	4.287	4.876	2.841	2.366
.0750	.997	1.185	1.406	1.545	1.842	2.162	2.979	3.743	2.738	2.376
.1000	1.025	1.191	1.389	1.530	1.765	2.012	2.665	3.256	2.676	2.357
.1500	1.047	1.201	1.364	1.476	1.642	1.790	2.244	2.606	2.555	2.300
.2000	1.072	1.217	1.364	1.458	1.592	1.737	2.061	2.290	2.431	2.275
.2500	1.104	1.239	1.357	1.434	1.534	1.677	1.942	2.079	2.283	2.243
.3000	1.110	1.252	1.367	1.431	1.512	1.614	1.860	1.928	2.171	2.218
.3500	1.144	1.255	1.367	1.416	1.481	1.581	1.732	1.800	2.065	2.190
.4000	1.172	1.284	1.367	1.416	1.466	1.566	1.674	1.728	1.980	2.174
.4500	1.207	1.309	1.389	1.425	1.475	1.545	1.622	1.679	1.929	2.158
.5000	1.245	1.344	1.430	1.437	1.491	1.527	1.616	1.655	1.882	2.142
.5500	1.276	1.373	1.433	1.443	1.484	1.491	1.585	1.597	1.805	2.098
.6000	1.314	1.398	1.458	1.455	1.500	1.503	1.595	1.600	1.767	2.079
.6500	1.370	1.449	1.489	1.497	1.521	1.515	1.634	1.621	1.711	2.041
.7000	1.430	1.526	1.546	1.545	1.571	1.557	1.620	1.647	1.708	1.994
.7615	1.621	1.678	1.705	1.693	1.725	1.728	1.762	1.710	1.741	1.978
.8179	1.803	1.863	1.853	1.849	1.898	1.964	1.951	1.896	1.876	2.003
.8325	1.912	1.956	1.953	1.937	2.012	2.087	2.082	2.032	1.953	2.060
.8362	1.941	1.997	1.970	1.958	2.043	2.126	2.143	2.087	1.997	2.079
.8398	1.897	1.949	1.925	1.919	2.012	2.081	2.092	2.055	1.985	2.063
.8434	1.923	2.064	2.028	2.006	2.114	2.210	2.253	2.200	2.077	2.117
Vane										
.0000	2.295	2.319	2.332	2.292	2.413	2.503	2.506	2.458	2.295	2.246
.0250	3.906	3.972	3.953	3.876	4.191	4.653	4.790	4.766	4.177	3.795
.0500	3.699	3.733	3.706	3.599	3.901	4.383	4.519	4.493	3.906	3.461
.1000	3.615	3.647	3.577	3.446	3.753	4.272	4.473	4.491	3.803	3.205
.1500	3.480	3.523	3.427	3.310	3.635	4.144	4.415	4.462	3.788	3.184
.2000	3.354	3.402	3.298	3.190	3.567	4.042	4.381	4.475	3.832	3.221
.3000	2.887	2.978	2.890	2.834	3.228	3.647	4.083	4.290	3.761	3.215
.4000	2.542	2.624	2.542	2.533	2.907	3.201	3.689	3.934	3.543	3.060
.5000	2.332	2.354	2.329	2.349	2.700	2.898	3.363	3.627	3.316	2.903
.6000	2.104	2.153	2.129	2.114	2.370	2.551	2.854	3.085	2.918	2.650
.7000	2.069	2.093	2.063	2.045	2.277	2.476	2.732	2.908	2.717	2.511
.8000	2.072	2.070	2.038	2.039	2.262	2.434	2.720	2.911	2.631	2.379
.9000	2.044	2.051	2.010	2.015	2.216	2.383	2.683	2.896	2.543	2.268
Flap										
.0000	1.013	1.465	1.502	1.361	1.345	1.383	1.540	1.621	1.549	1.489
.0125	2.176	2.249	2.160	2.033	2.166	2.353	2.610	2.774	2.379	2.136
.0250	2.367	2.421	2.373	2.328	2.515	2.728	3.037	3.247	2.738	2.527
.0500	2.423	2.500	2.455	2.443	2.648	2.965	3.195	3.366	3.039	2.862
.0750	2.285	2.338	2.276	2.265	2.450	2.644	2.909	3.067	2.841	2.723
.1000	2.104	2.144	2.066	2.075	2.216	2.365	2.601	2.728	2.611	2.546
.1500	1.809	1.815	1.737	1.735	1.845	1.982	2.140	2.296	2.248	2.224
.2000	1.746	1.752	1.712	1.702	1.787	1.940	2.122	2.279	2.186	2.161
.4000	1.806	1.812	1.771	1.768	1.830	1.940	2.110	2.276	2.201	2.152
.6000	1.731	1.726	1.646	1.560	1.595	1.665	1.829	1.858	1.847	1.912
.8000	1.442	1.421	1.364	1.358	1.373	1.392	1.403	1.299	1.289	1.413
Spoiler										
.2429	1.720	1.703	1.649	1.655	1.732	1.866	1.912	2.006	2.003	1.947
.4982	1.683	1.651	1.610	1.616	1.692	1.822	1.859	1.931	1.915	1.870
.7367	1.634	1.621	1.577	1.583	1.653	1.783	1.826	1.881	1.829	1.805
.9796	1.625	1.590	1.559	1.559	1.634	1.759	1.764	1.776	1.687	1.663

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0125	.997	.796	.649	.581	.583	.647	.726	.629	.443	.379
.0250	.969	.831	.674	.545	.463	.392	.339	.299	.251	.208
.0500	.956	.854	.712	.572	.457	.329	.256	.223	.230	.218
.0750	.950	.863	.727	.611	.491	.365	.293	.264	.292	.290
.1000	.969	.870	.752	.623	.522	.413	.348	.322	.354	.357
.1500	.956	.870	.762	.663	.565	.485	.424	.403	.407	.432
.2000	.959	.889	.787	.687	.611	.524	.494	.455	.463	.464
.2500	.966	.901	.821	.726	.654	.578	.531	.499	.502	.502
.3000	.950	.905	.821	.744	.679	.608	.561	.519	.525	.540
.3500	.962	.924	.853	.771	.719	.644	.601	.554	.561	.571
.4000	.956	.930	.859	.792	.741	.665	.616	.560	.578	.587
.4500	.928	.917	.856	.792	.744	.671	.628	.588	.599	.609
.5000	.862	.873	.828	.768	.734	.662	.610	.577	.590	.596
.5500	.803	.806	.777	.732	.697	.626	.585	.536	.558	.571
.6000	.730	.777	.737	.687	.660	.593	.567	.528	.537	.558
.6500	.696	.694	.655	.617	.602	.542	.509	.478	.493	.498
.7000	.677	.608	.564	.545	.512	.479	.457	.432	.448	.457
.7615	.583	.503	.433	.407	.392	.356	.335	.322	.325	.331
.8179	.492	.427	.398	.367	.361	.335	.311	.301	.307	.306
.8252	.470	.427	.389	.370	.364	.341	.323	.313	.319	.319
.8325	.448	.436	.414	.398	.398	.392	.381	.374	.375	.385
Vane										
.0250	1.310	1.306	1.276	1.175	1.142	1.084	1.049	1.000	.965	.940
.0500	.806	.771	.696	.563	.475	.395	.366	.351	.348	.338
.1000	.505	.459	.364	.220	.123	.087	.073	.064	.062	.063
.1500	.398	.360	.254	.123	.034	.027	.024	.012	.018	.013
.2000	.320	.293	.191	.081	.015	.012	.003	.012	.015	.016
.3000	.260	.229	.141	.048	.019	.036	.024	.014	.030	.019
.4000	.204	.185	.116	.045	.031	.042	.040	.035	.047	.041
.5000	.172	.166	.113	.060	.056	.069	.070	.067	.074	.057
.6000	.154	.159	.110	.081	.083	.096	.104	.096	.100	.088
.7000	.172	.175	.135	.136	.142	.156	.171	.168	.159	.155
.8000	.235	.245	.223	.232	.250	.287	.299	.307	.292	.265
.9200	.680	.694	.680	.705	.759	.838	.921	.971	.850	.773
Flap										
.0125	.480	.303	.176	.093	.083	.099	.119	.136	.121	.085
.0250	.169	.064	.016	.024	.012	.021	.027	.017	.032	.019
.0500	.107	.061	.034	.054	.049	.051	.046	.043	.062	.063
.0750	.094	.076	.047	.075	.074	.072	.067	.064	.086	.085
.1000	.110	.099	.091	.099	.105	.105	.104	.096	.109	.110
.1500	.138	.134	.122	.136	.139	.141	.134	.125	.142	.142
.4000	.342	.363	.339	.343	.352	.341	.329	.307	.322	.331
.6000	.825	.838	.790	.735	.685	.650	.613	.560	.569	.593
.8000	.630	.631	.605	.590	.574	.551	.518	.470	.481	.502
.9000	.715	.726	.712	.681	.685	.656	.625	.562	.563	.625
Spoiler										
.2573	1.826	1.792	1.707	1.693	1.744	1.866	1.918	1.994	2.036	2.015
.4984	1.796	1.755	1.680	1.673	1.726	1.848	1.885	1.973	2.003	1.985
.7485	1.579	1.547	1.498	1.500	1.555	1.653	1.649	1.633	1.528	1.515
.9940	1.619	1.590	1.553	1.574	1.625	1.756	1.785	1.797	1.637	1.598

TABLE IX.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(c) Double slotted flap configuration with spoiler; $h_s = -0.005\bar{c}$; $\frac{y}{b/2} = 0.30$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	.857	.743	2.242	2.918	3.541	4.183	4.262	3.938	2.982	2.277
.0125	1.021	1.254	2.341	2.945	3.480	3.916	3.947	3.844	2.955	2.265
.0250	1.052	1.480	2.449	3.040	3.607	4.069	3.988	3.856	2.982	2.274
.0500	1.070	1.379	3.084	3.942	3.936	4.021	3.944	3.888	2.904	2.253
.0750	1.079	1.266	1.932	4.028	5.254	5.302	4.384	3.956	2.925	2.250
.1000	1.095	1.229	1.282	2.509	4.042	4.802	4.320	3.850	2.934	2.271
.1500	1.104	1.211	1.288	1.439	2.145	3.245	3.685	3.505	2.791	2.244
.2000	1.128	1.211	1.344	1.378	1.613	2.371	3.026	3.118	2.695	2.241
.2500	1.159	1.235	1.368	1.396	1.474	1.928	2.571	2.800	2.603	2.220
.3000	1.180	1.260	1.381	1.409	1.438	1.787	2.320	2.590	2.489	2.220
.3500	1.201	1.269	1.368	1.390	1.417	1.629	2.055	2.298	2.361	2.178
.4000	1.238	1.297	1.393	1.424	1.444	1.599	1.941	2.180	2.272	2.152
.4500	1.278	1.336	1.412	1.448	1.459	1.611	1.877	2.092	2.185	2.140
.5000	1.317	1.352	1.449	1.491	1.474	1.581	1.784	1.974	2.069	2.107
.5500	1.339	1.398	1.464	1.470	1.486	1.575	1.740	1.903	2.006	2.059
.6000	1.396	1.437	1.505	1.516	1.511	1.608	1.734	1.859	1.952	2.030
.6500	1.494	1.523	1.570	1.570	1.577	1.623	1.720	1.823	1.898	2.003
.7521	1.707	1.761	1.780	1.787	1.788	1.787	1.831	1.894	1.890	1.917
.7934	2.079	2.070	2.081	2.089	2.078	2.108	2.084	2.121	1.961	1.961
.8099	2.296	2.272	2.272	2.290	2.266	2.332	2.282	2.269	2.042	1.982
.8141	2.393	2.370	2.378	2.378	2.362	2.398	2.349	2.357	2.081	1.988
.8182	2.451	2.425	2.427	2.430	2.420	2.467	2.414	2.410	2.119	1.994
.8224	2.628	2.587	2.560	2.586	2.568	2.596	2.559	2.540	2.230	2.027
Vane										
.0000	4.662	4.492	4.415	4.384	4.405	4.515	4.332	4.280	3.660	3.303
.0250	6.089	5.890	5.746	5.808	5.683	5.844	5.725	5.531	4.182	3.354
.0500	6.147	5.853	5.777	5.793	5.710	5.922	5.739	5.525	4.200	3.330
.1000	6.177	5.871	5.755	5.753	5.625	5.862	5.810	5.484	4.075	2.911
.1500	6.135	5.829	5.687	5.689	5.541	5.799	5.778	5.449	4.069	2.893
.2000	6.077	5.816	5.629	5.625	5.447	5.734	5.772	5.508	4.176	2.955
.3000	5.574	5.345	5.111	5.104	4.967	5.311	5.413	5.195	4.080	2.922
.4000	4.872	4.660	4.430	4.397	4.272	4.701	4.795	4.652	3.713	2.738
.5000	4.336	4.098	3.867	3.823	3.746	4.237	4.305	4.227	3.409	2.607
.6000	3.708	3.495	3.291	3.229	3.208	3.695	3.778	3.744	3.122	2.485
.7000	3.159	2.969	2.793	2.735	2.761	3.213	3.288	3.301	2.839	2.339
.8000	2.653	2.492	2.418	2.372	2.390	2.769	2.830	2.873	2.522	2.208
.9000	2.339	2.242	2.198	2.165	2.154	2.422	2.486	2.546	2.313	2.101
Flap										
.0000	1.531	1.511	1.505	1.396	1.314	1.410	1.490	1.510	1.400	1.280
.0125	2.198	2.180	2.142	2.086	2.018	2.177	2.277	2.310	2.176	1.997
.0250	2.683	2.654	2.567	2.500	2.477	2.716	2.839	2.900	2.743	2.521
.0500	2.650	2.630	2.536	2.488	2.501	2.778	2.941	3.059	2.928	2.661
.0750	2.583	2.556	2.455	2.412	2.453	2.757	2.932	3.065	2.934	2.643
.1000	2.470	2.397	2.303	2.284	2.338	2.632	2.793	2.915	2.782	2.515
.1500	2.000	1.972	1.929	1.945	2.042	2.272	2.428	2.549	2.451	2.253
.2000	1.796	1.725	1.709	1.723	1.791	1.967	2.052	2.145	2.057	1.958
.4000	1.659	1.645	1.641	1.689	1.737	1.790	1.871	1.935	1.898	1.863
.6000	1.592	1.605	1.616	1.621	1.635	1.720	1.783	1.786	1.760	1.738
.8000	1.470	1.486	1.474	1.403	1.372	1.359	1.341	1.348	1.328	1.309
Spoiler										
.2475	1.686	1.667	1.616	1.640	1.717	1.822	1.841	1.854	1.814	1.802
.4909	1.707	1.691	1.640	1.655	1.735	1.819	1.829	1.809	1.758	1.769
.7396	1.677	1.664	1.601	1.628	1.677	1.759	1.746	1.687	1.652	1.693
.9796	1.714	1.688	1.619	1.631	1.704	1.745	1.696	1.630	1.569	1.633

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	.979	.771	.734	.823	.979	1.153	1.259	1.301	1.176	1.048
.0250	1.000	.810	.712	.698	.737	.778	.810	.844	.806	.735
.0500	.994	.847	.724	.659	.616	.578	.554	.552	.567	.524
.0750	.997	.862	.746	.671	.607	.542	.490	.475	.484	.458
.1000	.994	.872	.768	.677	.607	.530	.461	.448	.454	.429
.1500	1.000	.887	.793	.707	.634	.542	.461	.448	.439	.417
.2000	1.000	.899	.814	.732	.646	.554	.472	.451	.442	.417
.2500	1.000	.902	.836	.750	.671	.578	.501	.469	.457	.440
.3000	.980	.905	.836	.756	.677	.590	.504	.481	.475	.449
.3500	.970	.899	.842	.768	.695	.614	.539	.502	.493	.482
.4000	.939	.884	.830	.765	.701	.608	.542	.519	.510	.488
.4500	.909	.850	.808	.756	.689	.614	.548	.525	.528	.503
.5000	.857	.816	.771	.726	.671	.608	.542	.516	.519	.488
.5500	.787	.768	.724	.683	.640	.578	.528	.516	.498	.482
.6000	.723	.685	.650	.610	.568	.530	.478	.463	.457	.449
.6500	.665	.605	.573	.546	.508	.479	.431	.425	.427	.417
.7521	.491	.437	.393	.375	.347	.320	.294	.295	.290	.280
.7934	.436	.379	.344	.317	.302	.272	.245	.251	.242	.226
.8017	.430	.367	.347	.317	.302	.272	.254	.248	.248	.235
.8099	.424	.367	.334	.320	.305	.302	.292	.286	.275	.265
Vane										
.0250	1.720	1.676	1.625	1.528	1.441	1.377	1.265	1.233	1.098	1.015
.0500	.756	.740	.749	.634	.526	.425	.367	.354	.313	.289
.1000	.287	.287	.310	.210	.121	.057	.047	.032	.036	.018
.1500	.149	.165	.189	.107	.036	.000	.000	.000	.000	.000
.2000	.076	.095	.118	.052	.024	.000	.000	.000	.000	.000
.3000	.037	.037	.050	.037	.006	.000	.000	.009	.000	.000
.4000	.030	.012	.025	.030	.033	.009	.017	.012	.030	.021
.5000	.043	.028	.019	.049	.048	.039	.038	.035	.036	.039
.6000	.079	.049	.050	.076	.066	.060	.070	.077	.075	.060
.7000	.140	.113	.111	.134	.142	.126	.131	.130	.128	.116
.8000	.268	.235	.226	.247	.245	.254	.271	.260	.236	.226
.9200	.811	.752	.743	.738	.722	.796	.816	.823	.767	.702
Flap										
.0125	.232	.220	.229	.210	.172	.171	.192	.192	.173	.155
.0250	.021	.012	.015	.030	.012	.003	.017	.012	.009	.003
.0500	.009	.000	.000	.012	.015	.000	.000	.000	.000	.000
.0750	.015	.003	.000	.018	.015	.000	.003	.000	.000	.000
.1000	.037	.009	.015	.040	.039	.024	.023	.012	.009	.018
.1500	.067	.052	.043	.058	.060	.033	.038	.027	.030	.033
.2000	.101	.076	.087	.101	.091	.063	.058	.062	.042	.054
.4000	.226	.214	.195	.223	.214	.159	.146	.139	.137	.134
.6000	.366	.352	.337	.345	.326	.281	.259	.242	.251	.244
.8000	.540	.538	.533	.512	.489	.467	.431	.410	.418	.417
.9000	.729	.725	.712	.704	.686	.665	.621	.599	.588	.595
Spoiler										
.2587	1.842	1.801	1.719	1.693	1.747	1.854	1.900	1.970	2.006	2.003
.5024	1.796	1.758	1.689	1.670	1.720	1.819	1.838	1.892	1.912	1.929
.7531	1.622	1.593	1.544	1.577	1.631	1.650	1.587	1.519	1.475	1.562
.9940	1.646	1.645	1.601	1.598	1.671	1.688	1.658	1.567	1.507	1.592

TABLE IX.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(d) Double slotted flap configuration with spoiler; $h_s = -0.005\bar{c}$; $\frac{y}{b/2} = 0.43$

Upper surface

x/c	C_p for -									
	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	.846	1.863	2.147	2.527	2.981	2.883	2.720	2.519	2.390	2.291
.0125	1.078	1.822	2.119	2.509	2.953	2.892	2.860	2.589	2.407	2.316
.0250	1.122	1.758	2.188	2.551	2.956	2.949	2.909	2.589	2.407	2.297
.0500	1.066	1.726	2.223	2.629	3.089	2.997	2.909	2.635	2.422	2.347
.0750	1.100	1.714	2.511	2.696	3.046	2.895	2.918	2.638	2.413	2.322
.1000	1.122	1.542	2.872	2.925	2.987	2.859	2.921	2.638	2.413	2.309
.1500	1.122	1.290	2.423	3.696	3.706	3.078	3.012	2.644	2.413	2.322
.2000	1.169	1.303	1.605	2.949	3.999	3.335	2.994	2.603	2.395	2.303
.2500	1.182	1.319	1.276	1.964	3.200	3.189	2.872	2.523	2.372	2.281
.3000	1.226	1.370	1.314	1.545	2.571	3.000	2.768	2.458	2.345	2.281
.3500	1.251	1.385	1.345	1.343	2.080	2.665	2.598	2.351	2.289	2.259
.4000	1.292	1.411	1.395	1.343	1.839	2.452	2.488	2.296	2.248	2.250
.4500	1.329	1.459	1.452	1.358	1.728	2.278	2.384	2.224	2.207	2.224
.5000	1.348	1.487	1.470	1.386	1.605	2.066	2.214	2.116	2.136	2.212
.5500	1.439	1.557	1.555	1.461	1.611	1.976	2.146	2.052	2.077	2.180
.6000	1.524	1.640	1.633	1.536	1.632	1.907	2.076	2.000	2.041	2.155
.7000	1.662	1.921	1.915	1.834	1.821	1.865	1.994	1.919	1.974	2.101
.7500	2.100	2.344	2.351	2.262	2.188	2.057	2.110	1.983	1.977	2.108
.7720	2.383	2.701	2.690	2.608	2.534	2.272	2.262	2.070	2.012	2.120
.7750	2.464	2.784	2.784	2.714	2.626	2.344	2.299	2.102	2.033	2.123
.7800	2.590	2.924	2.925	2.834	2.750	2.428	2.357	2.134	2.047	2.133
Vane										
.0000	4.050	4.160	4.097	3.922	3.796	3.344	3.137	2.757	2.561	2.625
.0250	6.565	6.739	6.725	6.608	6.305	5.266	4.811	3.992	3.316	3.193
.0500	6.781	6.918	6.913	6.801	6.490	5.404	4.891	3.972	3.159	2.972
.1000	6.997	7.064	7.101	6.997	6.678	5.467	4.851	3.809	2.891	2.650
.1500	7.195	7.262	7.295	7.220	6.903	5.632	4.958	3.841	2.832	2.537
.2000	7.411	7.523	7.558	7.488	7.212	5.910	5.192	4.024	2.956	2.644
.3000	6.778	6.886	6.878	6.834	6.635	5.446	4.772	3.702	2.741	2.477
.4000	6.041	6.131	6.110	6.105	5.987	4.967	4.381	3.432	2.605	2.398
.5000	5.370	5.409	5.411	5.404	5.311	4.500	4.016	3.169	2.472	2.322
.6000	4.640	4.685	4.671	4.678	4.623	3.970	3.595	2.916	2.360	2.259
.7000	4.041	4.093	4.066	4.084	4.055	3.554	3.287	2.713	2.260	2.221
.8000	3.449	3.491	3.464	3.479	3.450	3.105	2.924	2.470	2.145	2.152
.9000	2.956	2.994	3.000	2.994	2.963	2.698	2.576	2.215	2.000	2.063
Flap										
.0000	2.110	2.401	2.533	2.485	2.410	2.114	2.009	1.748	1.587	1.647
.0125	2.564	2.902	2.944	2.919	2.901	2.575	2.464	2.168	2.000	2.070
.0250	3.135	3.542	3.552	3.533	3.555	3.207	3.076	2.719	2.525	2.619
.0500	3.195	3.561	3.530	3.533	3.592	3.299	3.208	2.835	2.602	2.663
.0750	3.079	3.462	3.430	3.425	3.490	3.272	3.208	2.838	2.584	2.612
.1000	3.091	3.293	3.245	3.256	3.311	3.126	3.095	2.737	2.484	2.515
.1500	2.414	2.535	2.480	2.488	2.601	2.578	2.607	2.348	2.174	2.227
.2000	2.166	2.150	2.147	2.157	2.169	2.150	2.183	1.995	1.894	1.988
.4000	1.520	1.624	1.599	1.623	1.697	1.751	1.863	1.771	1.805	1.931
.6000	1.633	1.755	1.665	1.479	1.481	1.482	1.601	1.603	1.620	1.716
.8000	1.122	1.331	1.323	1.319	1.407	1.338	1.387	1.403	1.540	1.716
Spoiler										
.2443	1.518	1.501	1.483	1.515	1.625	1.724	1.764	1.806	1.808	1.852
.4906	1.393	1.388	1.378	1.434	1.537	1.602	1.620	1.663	1.696	1.769
.7370	1.278	1.278	1.284	1.348	1.460	1.466	1.484	1.519	1.605	1.690
.9856	1.226	1.248	1.254	1.303	1.424	1.347	1.351	1.388	1.478	1.610

Lower surface

x/c	C_p for -									
	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	.947	.787	.812	.958	1.163	1.278	1.360	1.316	1.333	1.344
.0250	.972	.809	.762	.777	.852	.892	.948	.904	.923	.959
.0500	.972	.841	.762	.714	.722	.686	.701	.667	.690	.732
.0750	.981	.860	.771	.708	.673	.626	.622	.591	.608	.647
.1000	.972	.873	.793	.705	.663	.599	.573	.554	.572	.596
.1500	.969	.882	.796	.717	.657	.584	.549	.516	.531	.552
.2000	.953	.876	.806	.714	.663	.575	.537	.504	.522	.546
.2500	.941	.876	.803	.714	.679	.584	.546	.510	.519	.543
.3000	.903	.860	.799	.708	.663	.584	.543	.504	.516	.536
.3500	.881	.838	.790	.714	.670	.584	.552	.513	.519	.546
.4000	.840	.799	.756	.684	.640	.572	.546	.513	.519	.540
.4500	.796	.761	.721	.663	.617	.554	.515	.493	.502	.521
.5000	.734	.710	.674	.605	.586	.527	.494	.461	.472	.505
.5500	.690	.634	.583	.539	.537	.452	.454	.417	.422	.448
.6000	.608	.567	.549	.494	.500	.428	.424	.391	.413	.429
.7000	.414	.395	.395	.352	.358	.317	.311	.293	.289	.312
.7500	.348	.341	.345	.316	.306	.266	.256	.246	.257	.265
.7600	.332	.325	.335	.316	.302	.257	.259	.249	.242	.259
.7700	.345	.341	.323	.310	.327	.269	.268	.226	.280	.265
Vane										
.0250	.962	.994	1.063	.949	.873	.716	.649	.574	.572	.596
.0500	.420	.446	.552	.467	.401	.281	.250	.226	.212	.230
.1000	.138	.207	.317	.232	.179	.111	.079	.072	.080	.095
.1500	.050	.054	.141	.114	.068	.042	.027	.029	.018	.044
.2000	.000	.019	.097	.066	.043	.033	.034	.020	.044	.035
.3000	.019	.022	.075	.048	.031	.024	.067	.023	.062	.060
.4000	.016	.035	.063	.042	.040	.036	.052	.029	.068	.057
.5000	.069	.057	.069	.072	.093	.087	.085	.052	.089	.095
.6000	.088	.089	.094	.096	.105	.111	.110	.099	.112	.101
.7000	.147	.147	.141	.154	.167	.153	.152	.139	.136	.142
.8000	.314	.312	.307	.325	.327	.293	.293	.258	.248	.249
.9200	1.022	1.026	1.044	1.033	1.025	.919	.872	.762	.720	.773
Flap										
.0125	.226	.261	.351	.340	.315	.272	.256	.226	.201	.215
.0250	.022	.000	.060	.069	.089	.027	.052	.046	.071	.069
.0500	.003	.013	.041	.021	.049	.021	.049	.012	.047	.054
.0750	.003	.000	.041	.054	.037	.027	.061	.029	.053	.057
.1000	.009	.025	.047	.033	.049	.018	.043	.026	.077	.050
.1500	.031	.067	.078	.084	.071	.084	.046	.058	.074	.069
.2000	.069	.083	.075	.081	.108	.087	.091	.075	.083	.104
.4000	.191	.169	.172	.166	.170	.159	.152	.148	.171	.180
.6000	.298	.299	.279	.286	.284	.263	.244	.238	.257	.290
.8000	.467	.449	.448	.443	.447	.416	.399	.394	.443	.476
.9000	.574	.602	.574	.584	.599	.569	.546	.539	.608	.669
Spoiler										
.2587	1.869	1.844	1.785	1.678	1.698	1.758	1.791	1.854	1.876	1.906
.5039	1.753	1.731	1.665	1.601	1.640	1.676	1.714	1.770	1.814	1.855
.7508	1.192	1.211	1.193	1.253	1.357	1.294	1.286	1.334	1.437	1.556
.9940	1.195	1.217	1.233	1.286	1.384	1.297	1.301	1.349	1.475	1.592

TABLE IX.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(e) Double slotted flap configuration with spoiler; $h_s = -0.005\bar{c}$; $\frac{y}{b/2} = 0.55$

Upper surface

	C_p for -									
x/c	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.323	1.890	2.118	2.406	2.504	2.389	2.189	2.224	2.161	2.101
.0125	1.345	1.893	2.176	2.473	2.526	2.419	2.265	2.272	2.185	2.116
.0250	1.229	1.847	2.136	2.430	2.480	2.410	2.265	2.266	2.188	2.113
.0500	1.217	1.771	2.195	2.488	2.516	2.488	2.312	2.304	2.191	2.104
.0750	1.210	1.804	2.282	2.561	2.550	2.488	2.315	2.313	2.197	2.095
.1000	1.226	1.764	2.337	2.616	2.595	2.464	2.326	2.322	2.191	2.101
.1500	1.241	1.593	2.740	2.659	2.526	2.431	2.352	2.331	2.173	2.101
.2000	1.271	1.407	2.972	2.906	2.477	2.437	2.352	2.307	2.173	2.104
.2500	1.305	1.373	2.839	3.564	2.655	2.479	2.358	2.313	2.191	2.095
.3000	1.332	1.376	2.328	3.604	2.955	2.545	2.341	2.289	2.182	2.086
.3500	1.381	1.410	1.824	3.137	3.000	2.545	2.303	2.274	2.170	2.086
.4000	1.430	1.462	1.480	2.561	2.876	2.503	2.259	2.239	2.143	2.083
.4500	1.470	1.511	1.387	2.043	2.655	2.416	2.218	2.195	2.128	2.065
.5000	1.540	1.587	1.449	1.768	2.483	2.338	2.172	2.168	2.104	2.056
.6201	1.726	1.816	1.771	1.659	2.012	2.072	2.026	2.047	2.030	2.018
.6834	2.204	2.294	2.266	2.064	2.142	2.114	2.049	2.059	2.033	2.027
.7087	2.543	2.633	2.598	2.357	2.272	2.162	2.084	2.086	2.054	2.030
.7151	2.570	2.679	2.632	2.415	2.290	2.177	2.084	2.089	2.054	2.042
.7214	2.637	2.810	2.830	2.592	2.396	2.240	2.125	2.121	2.056	2.042
.7278	2.784	2.948	2.950	2.701	2.435	2.257	2.143	2.127	2.072	2.036
Vane										
.0000	3.354	3.339	3.214	2.845	2.489	2.287	2.169	2.162	2.075	2.042
.0250	6.543	6.526	6.508	5.808	4.643	3.946	3.571	3.522	3.140	2.943
.0500	6.683	6.627	6.687	5.970	4.707	3.940	3.513	3.422	2.970	2.774
.1000	7.153	7.058	7.192	6.482	4.888	3.898	3.381	3.251	2.749	2.556
.1500	7.205	7.098	7.273	6.577	4.828	3.814	3.288	3.154	2.615	2.464
.2000	7.229	7.113	7.334	6.720	4.906	3.832	3.291	3.142	2.576	2.417
.3000	6.897	6.777	7.009	6.485	4.698	3.662	3.139	2.994	2.451	2.286
.4000	6.156	6.055	6.276	5.872	4.248	3.323	2.871	2.758	2.298	2.178
.5000	5.583	5.486	5.721	5.403	3.948	3.117	2.731	2.634	2.245	2.131
.6000	4.857	4.777	4.957	4.714	3.489	2.826	2.539	2.475	2.167	2.092
.7000	4.205	4.141	4.285	4.110	3.133	2.611	2.376	2.331	2.107	2.053
.8000	3.668	3.627	3.734	3.583	2.846	2.440	2.245	2.221	2.033	2.018
.9000	3.144	3.101	3.189	3.061	2.544	2.222	2.084	2.065	1.949	1.976
Flap										
.0000	1.906	1.954	2.111	2.073	1.767	1.548	1.423	1.392	1.334	1.348
.0125	2.954	3.015	3.124	3.046	2.631	2.311	2.143	2.112	2.045	2.077
.0250	3.455	3.581	3.678	3.604	3.136	2.766	2.556	2.528	2.454	2.467
.0500	3.580	3.685	3.786	3.738	3.269	2.895	2.667	2.631	2.519	2.461
.0750	3.570	3.700	3.790	3.769	3.317	2.934	2.685	2.628	2.495	2.396
.1000	3.491	3.547	3.598	3.573	3.175	2.832	2.597	2.552	2.406	2.297
.1500	2.814	2.826	2.836	2.875	2.625	2.380	2.236	2.224	2.090	2.018
.2000	2.448	2.440	2.464	2.454	2.272	2.108	1.982	1.977	1.890	1.863
.4000	1.930	1.899	1.901	1.906	1.882	1.847	1.781	1.794	1.779	1.786
.6000	1.957	1.982	1.981	1.900	1.671	1.602	1.525	1.528	1.504	1.600
.8000	1.832	1.878	1.885	1.628	1.541	1.494	1.437	1.434	1.463	1.494
Spoiler										
.2443	1.814	1.786	1.737	1.723	1.845	1.822	1.805	1.815	1.805	1.840
.4881	1.631	1.602	1.541	1.565	1.695	1.712	1.696	1.722	1.738	1.772
.7416	1.421	1.410	1.350	1.372	1.540	1.596	1.587	1.621	1.661	1.716
.9856	1.268	1.272	1.217	1.226	1.381	1.472	1.475	1.510	1.578	1.654

Lower surface

	C_p for -									
x/c	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	.903	.786	.954	1.189	1.359	1.455	1.501	1.584	1.582	1.598
.0250	.951	.786	.817	.884	.946	1.012	1.046	1.094	1.107	1.137
.0500	.967	.823	.780	.750	.755	.757	.772	.817	.800	.833
.0750	.942	.804	.752	.713	.671	.665	.656	.673	.672	.696
.1000	.942	.810	.755	.701	.656	.629	.609	.611	.609	.631
.1500	.924	.807	.746	.677	.631	.587	.557	.558	.549	.571
.2000	.912	.807	.749	.680	.622	.566	.533	.537	.522	.542
.2500	.875	.792	.737	.671	.610	.566	.531	.534	.513	.518
.3000	.848	.774	.721	.662	.598	.566	.525	.516	.493	.500
.3500	.820	.755	.706	.646	.598	.542	.510	.504	.481	.497
.4000	.777	.716	.681	.610	.577	.530	.487	.493	.463	.479
.4500	.726	.664	.647	.585	.550	.503	.475	.463	.448	.464
.5000	.671	.621	.585	.555	.505	.470	.440	.437	.424	.426
.6201	.485	.446	.433	.412	.353	.344	.329	.313	.307	.330
.6834	.402	.379	.378	.329	.311	.281	.268	.263	.245	.268
.6961	.387	.376	.372	.338	.296	.275	.262	.257	.245	.250
.7087	.390	.373	.368	.329	.287	.272	1.245	.263	.239	.262
Vane										
.0250	1.192	1.131	1.167	1.012	.870	.787	.294	.699	.672	.673
.0500	.524	.477	.591	.476	.411	.359	.117	.292	.263	.277
.1000	.210	.180	.322	.244	.190	.150	.090	.121	.113	.107
.1500	.101	.080	.201	.159	.121	.090	.079	.077	.063	.080
.2000	.067	.070	.142	.107	.103	.081	.085	.071	.057	.071
.3000	.040	.055	.099	.070	.066	.102	.085	.077	.057	.074
.4000	.061	.080	.099	.091	.109	.069	.096	.083	.066	.089
.5000	.091	.107	.121	.101	.094	.090	.140	.100	.101	.107
.6000	.152	.147	.146	.131	.121	.132	.178	.130	.125	.128
.7000	.201	.214	.204	.195	.196	.171	.277	.159	.164	.170
.8000	.384	.370	.384	.366	.302	.302	.807	.298	.269	.286
.9200	1.149	1.122	1.164	1.107	.927	.838	.251	.805	.773	.804
Flap										
.0125	.271	.287	.372	.366	.284	.251	.073	.257	.227	.238
.0250	.034	.073	.084	.082	.088	.084	.073	.071	.054	.083
.0500	.040	.061	.068	.058	.057	.084	.082	.086	.051	.077
.0750	.040	.067	.068	.079	.054	.075	.079	.083	.075	.074
.1000	.076	.089	.093	.064	.066	.096	.093	.093	.060	.077
.1500	.116	.098	.127	.098	.097	.096	.125	.100	.096	.101
.2000	.134	.144	.142	.137	.115	.129	.204	.121	.137	.119
.4000	.271	.263	.251	.238	.218	.222	.294	.215	.200	.196
.6000	.390	.379	.365	.335	.332	.329	.455	.292	.293	.312
.8000	.540	.529	.502	.488	.483	.482	.598	.463	.469	.479
.9000	.656	.654	.638	.625	.622	.641	1.248	.608	.627	.655
Spoiler										
.2584	2.058	2.037	1.979	1.708	1.677	1.644	1.646	1.669	1.714	1.746
.5039	1.770	1.761	1.692	1.524	1.540	1.561	1.558	1.585	1.625	1.681
.7508	1.302	1.297	1.233	1.220	1.342	1.439	1.428	1.460	1.534	1.604
.9940	1.247	1.251	1.196	1.193	1.345	1.448	1.443	1.475	1.540	1.613

TABLE IX.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Concluded

(f) Double slotted flap configuration with spoiler; $h_s = -0.005\bar{c}$; $\frac{y}{b/2} = 0.72$

Upper surface

x/c	C_p for -									
	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0000	2.165	2.208	2.359	2.292	2.226	2.015	1.935	1.925	1.894	1.917
.0125	2.101	2.162	2.269	2.202	2.159	1.955	1.923	1.910	1.900	1.914
.0250	2.174	2.196	2.308	2.244	2.180	1.997	1.950	1.940	1.900	1.920
.0500	2.348	2.257	2.338	2.253	2.186	2.038	1.977	1.946	1.912	1.923
.0750	1.970	2.297	2.408	2.283	2.217	2.056	1.982	1.955	1.909	1.923
.1000	1.451	2.266	2.384	2.271	2.195	2.032	1.974	1.958	1.912	1.923
.1500	1.390	2.388	2.459	2.300	2.211	2.044	2.003	1.970	1.918	1.923
.2000	1.445	2.563	2.559	2.300	2.201	2.068	2.021	1.976	1.923	1.926
.2500	1.500	2.544	2.643	2.274	2.186	2.068	2.021	1.970	1.920	1.935
.3000	1.525	2.358	2.800	2.262	2.192	2.056	2.006	1.976	1.932	1.941
.3500	1.567	2.217	3.003	2.253	2.211	2.062	2.009	1.973	1.938	1.947
.4000	1.610	1.927	3.072	2.321	2.226	2.059	2.003	1.964	1.941	1.953
.4500	1.656	1.734	3.075	2.425	2.235	2.053	1.997	1.967	1.944	1.956
.5000	1.720	1.673	3.030	2.530	2.275	2.047	1.997	1.976	1.947	1.962
.5500	1.750	1.624	2.843	2.470	2.247	2.032	1.982	1.958	1.944	1.971
.6000	1.790	1.627	2.580	2.381	2.232	2.015	1.971	1.955	1.950	1.974
.6500	1.759	1.593	2.326	2.250	2.195	2.009	1.971	1.943	1.959	1.983
.7000	1.686	1.523	2.148	2.199	2.174	1.988	1.956	1.940	1.965	1.994
.7500	1.610	1.477	2.000	2.202	2.168	1.979	1.956	1.937	1.971	2.003
.8000	1.497	1.385	1.855	2.137	2.122	1.976	1.953	1.943	1.977	2.003
.8500	1.430	1.333	1.749	2.024	2.058	1.949	1.938	1.937	1.985	2.006
.9000	1.384	1.312	1.656	1.961	2.031	1.940	1.932	1.937	1.985	2.009

Lower surface

x/c	C_p for -									
	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0125	.802	.862	1.012	1.137	1.241	1.282	1.328	1.364	1.401	1.441
.0250	.799	.783	.864	.937	.994	1.047	1.080	1.119	1.162	1.201
.0500	.814	.758	.758	.780	.814	.846	.853	.896	.923	.950
.0750	.829	.771	.743	.741	.756	.754	.770	.794	.820	.843
.1000	.829	.771	.737	.714	.717	.715	.723	.740	.764	.781
.1500	.854	.783	.740	.702	.692	.679	.670	.678	.693	.710
.2000	.860	.807	.758	.717	.698	.674	.670	.675	.679	.684
.2500	.881	.832	.792	.738	.713	.682	.661	.666	.673	.681
.3000	.921	.875	.849	.786	.753	.706	.699	.698	.693	.710
.3500	1.009	.979	.949	.881	.835	.780	.755	.779	.761	.775
.4000	1.146	1.138	1.115	1.012	.954	.881	.861	.875	.858	.858
.4500	1.363	1.370	1.353	1.184	1.098	1.015	.982	.979	.976	.979
.5000	1.628	1.627	1.619	1.396	1.274	1.166	1.133	1.125	1.118	1.101
.5500	1.878	1.887	1.870	1.613	1.488	1.353	1.325	1.334	1.319	1.296
.6000	2.098	2.110	2.069	1.803	1.692	1.567	1.552	1.549	1.549	1.542
.6500	2.290	2.294	2.190	1.911	1.832	1.742	1.746	1.764	1.773	1.731
.7000	2.360	2.355	2.202	1.928	1.881	1.822	1.841	1.857	1.847	1.808
.7500	2.329	2.327	2.142	1.922	1.875	1.819	1.823	1.830	1.826	1.799
.8000	2.183	2.168	2.039	1.887	1.832	1.751	1.743	1.761	1.752	1.769
.8500	2.012	2.006	1.936	1.848	1.802	1.718	1.708	1.719	1.729	1.758
.9000	1.842	1.844	1.834	1.824	1.802	1.727	1.702	1.722	1.726	1.746
.9500	1.640	1.648	1.710	1.806	1.808	1.733	1.726	1.737	1.738	1.755

TABLE X.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE

(a) Double slotted flap configuration with spoiler; $h_s = -0.010\bar{c}$; $\frac{y}{b/2} = 0$

Upper surface

x/c	C_p for -									
	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Fuselage										
.0000	.000	.003	.000	.000	.000	.030	.053	.084	.105	.150
.0500	.585	.831	.890	.941	.973	1.015	1.018	1.072	1.047	1.075
.1000	.641	.923	.966	1.009	1.024	1.066	1.071	1.131	1.120	1.133
.1500	.771	.982	1.024	1.068	1.069	1.096	1.094	1.140	1.126	1.133
.2000	.814	1.031	1.067	1.100	1.091	1.111	1.100	1.140	1.111	1.127
.2500	.854	1.065	1.098	1.118	1.106	1.123	1.115	1.140	1.126	1.136
.3000	.930	1.080	1.101	1.115	1.100	1.114	1.092	1.119	1.102	1.118
.3500	.950	1.077	1.089	1.096	1.082	1.096	1.089	1.119	1.099	1.121
.4000	.930	1.043	1.055	1.084	1.069	1.084	1.092	1.125	1.105	1.133
.4500	.947	1.058	1.070	1.109	1.109	1.135	1.171	1.221	1.222	1.231
.5000	.957	1.083	1.110	1.165	1.193	1.246	1.289	1.349	1.386	1.522
.5500	.957	1.111	1.165	1.237	1.269	1.320	1.363	1.418	1.544	1.853
.6000	1.015	1.163	1.217	1.286	1.293	1.329	1.357	1.406	1.585	1.893
.6500	1.062	1.203	1.269	1.315	1.302	1.329	1.319	1.334	1.447	1.755
.7000	1.102	1.252	1.300	1.330	1.320	1.329	1.310	1.322	1.404	1.680
.7500	1.186	1.298	1.336	1.346	1.332	1.353	1.354	1.373	1.421	1.617
.8000	1.266	1.338	1.373	1.364	1.365	1.401	1.422	1.466	1.468	1.556
.8500	1.195	1.329	1.330	1.352	1.359	1.404	1.440	1.504	1.506	1.519
.9000	1.102	1.234	1.251	1.274	1.266	1.317	1.357	1.424	1.453	1.432
.9500	.975	1.175	1.168	1.190	1.190	1.234	1.286	1.361	1.374	1.398
.9940	1.046	1.203	1.190	1.190	1.193	1.240	1.292	1.352	1.380	1.392

Lower surface

x/c	C_p for -									
	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Fuselage										
.0500	.650	.809	.749	.695	.619	.563	.472	.445	.409	.386
.1000	.709	.911	.847	.813	.713	.656	.590	.552	.518	.499
.1500	.805	.960	.914	.872	.798	.743	.664	.639	.599	.574
.2000	.820	.997	.957	.925	.855	.805	.717	.701	.658	.628
.2500	.836	1.034	.991	.959	.894	.844	.773	.752	.702	.689
.3000	.910	1.031	1.012	.997	.921	.874	.808	.797	.737	.726
.3500	.882	1.022	1.003	.978	.927	.886	.817	.809	.751	.741
.4000	.799	.963	.936	.919	.867	.832	.767	.746	.702	.680
.4500	.749	.932	.893	.872	.819	.772	.699	.684	.643	.617
.5000	.703	.880	.841	.816	.749	.698	.622	.609	.567	.539
.5500	.632	.843	.798	.763	.695	.638	.555	.531	.494	.481
.6000	.628	.778	.731	.685	.616	.563	.499	.478	.444	.438
.7500	.774	.908	.859	.841	.788	.766	.690	.690	.658	.666
.9500	.901	1.098	1.122	1.156	1.157	1.168	1.153	1.185	1.216	1.242
.9940	.879	1.148	1.183	1.218	1.224	1.278	1.313	1.370	1.386	1.369

TABLE X.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(b) Double slotted flap configuration with spoiler; $h_s = -0.010\bar{c}$; $\frac{y}{b/2} = 0.21$

Upper surface

x/c	C_p for -								
	α = -4°	α = 0°	α = 4°	α = 8°	α = 12°	α = 16°	α = 20°	α = 22°	α = 24°	α = 26°
Wing										
.0000	.820	.988	1.938	3.581	5.423	7.439	9.589	8.334	3.482	2.616
.0125	.953	1.296	1.722	3.806	7.646	10.289	11.679	8.027	3.123	2.396
.0250	.988	1.245	1.556	2.824	3.733	5.952	8.345	7.307	3.168	2.405
.0500	1.019	1.211	1.456	1.678	2.027	2.648	4.087	5.204	2.889	2.315
.0750	1.029	1.201	1.400	1.596	1.814	2.131	2.859	3.891	2.835	2.294
.1000	1.038	1.208	1.378	1.576	1.736	1.967	2.568	3.342	2.799	2.277
.1500	1.060	1.211	1.353	1.514	1.607	1.782	2.201	2.626	2.704	2.226
.2000	1.098	1.220	1.356	1.473	1.553	1.707	2.030	2.322	2.575	2.196
.2500	1.117	1.233	1.338	1.473	1.508	1.645	1.910	2.103	2.368	2.149
.3000	1.133	1.261	1.356	1.453	1.477	1.600	1.826	1.959	2.207	2.113
.3500	1.164	1.271	1.353	1.432	1.447	1.570	1.709	1.832	2.075	2.089
.4000	1.174	1.293	1.356	1.432	1.438	1.552	1.637	1.755	1.967	2.056
.4500	1.212	1.315	1.372	1.432	1.438	1.543	1.613	1.711	1.877	2.033
.5000	1.256	1.365	1.406	1.453	1.456	1.516	1.592	1.676	1.829	2.012
.5500	1.290	1.381	1.425	1.473	1.444	1.484	1.559	1.623	1.769	1.976
.6000	1.331	1.422	1.450	1.494	1.465	1.484	1.565	1.640	1.728	1.949
.6500	1.388	1.466	1.488	1.514	1.489	1.504	1.601	1.655	1.686	1.922
.7000	1.451	1.522	1.531	1.596	1.550	1.543	1.643	1.670	1.686	1.881
.7815	1.634	1.686	1.678	1.698	1.685	1.713	1.733	1.729	1.725	1.869
.8179	1.824	1.846	1.831	1.842	1.829	1.922	1.919	1.900	1.853	1.902
.8325	1.934	1.947	1.919	1.944	1.925	2.042	2.051	2.027	1.943	1.940
.8362	1.969	1.978	1.956	1.964	1.973	2.084	2.099	2.083	1.997	1.955
.8398	1.928	1.931	1.909	1.944	1.928	2.030	2.051	2.044	1.970	1.949
.8434	2.016	2.044	2.016	2.046	2.024	2.155	2.198	2.183	2.078	1.994
Vane										
.0000	2.328	2.305	2.297	2.312	2.321	2.454	2.462	2.431	2.302	2.128
.0250	3.988	3.931	3.891	3.929	4.012	4.531	4.664	4.664	4.207	3.601
.0500	3.767	3.689	3.628	3.642	3.736	4.280	4.390	4.390	3.943	3.315
.1000	3.717	3.604	3.525	3.479	3.592	4.149	4.336	4.357	3.829	3.041
.1500	3.590	3.475	3.388	3.315	3.435	4.021	4.255	4.313	3.811	2.997
.2000	3.480	3.359	3.269	3.233	3.417	3.913	4.201	4.319	3.847	3.047
.3000	3.035	2.950	2.869	2.865	3.087	3.531	3.877	4.083	3.740	3.009
.4000	2.663	2.588	2.525	2.578	2.781	3.104	3.492	3.752	3.509	2.869
.5000	2.391	2.330	2.303	2.394	2.589	2.830	3.204	3.416	3.290	2.738
.6000	2.171	2.129	2.106	2.149	2.273	2.484	2.742	2.909	2.865	2.482
.7000	2.123	2.073	2.047	2.087	2.183	2.409	2.646	2.770	2.677	2.360
.8000	2.101	2.060	2.006	2.067	2.180	2.397	2.658	2.782	2.632	2.277
.9000	2.076	2.025	1.975	2.046	2.132	2.346	2.613	2.770	2.572	2.178
Flap										
.0000	1.208	1.525	1.528	1.412	1.267	1.337	1.456	1.590	1.518	1.375
.0125	2.205	2.233	2.141	2.005	2.078	2.316	2.541	2.679	2.398	2.045
.0250	2.391	2.396	2.341	2.415	2.417	2.692	2.976	3.130	2.829	2.422
.0500	2.489	2.485	2.428	2.497	2.568	2.827	3.099	3.266	3.024	2.696
.0750	2.335	2.318	2.253	2.292	2.372	2.609	2.844	2.971	2.787	2.544
.1000	2.130	2.120	2.050	2.087	2.162	2.352	2.556	2.637	2.548	2.363
.1500	1.820	1.799	1.725	1.698	1.808	1.964	2.126	2.213	2.186	2.065
.2000	1.767	1.745	1.694	1.698	1.730	1.901	2.063	2.171	2.129	2.006
.4000	1.814	1.805	1.750	1.821	1.778	1.916	2.087	2.186	2.159	2.042
.6000	1.622	1.626	1.597	1.678	1.634	1.687	1.652	1.596	1.593	1.708
.8000	1.628	1.629	1.600	1.678	1.637	1.681	1.658	1.605	1.590	1.708
Spoiler										
.2429	1.028	1.612	1.657	1.685	1.674	1.778	1.879	2.012	1.959	1.911
.4852	.966	1.560	1.621	1.651	1.652	1.740	1.841	1.955	1.871	1.833
.7367	.938	1.520	1.575	1.632	1.616	1.704	1.791	1.901	1.795	1.772
.9796	1.009	1.551	1.584	1.620	1.598	1.614	1.729	1.806	1.681	1.654

Lower surface

x/c	C_p for -									
	α = -4°	α = 0°	α = 4°	α = 8°	α = 12°	α = 16°	α = 20°	α = 22°	α = 24°	α = 26°
Wing										
.0125	1.013	.796	.656	.573	.571	.639	.718	.681	.437	.363
.0250	.988	.824	.672	.553	.450	.403	.345	.310	.231	.196
.0500	.981	.855	.722	.573	.435	.340	.258	.221	.216	.208
.0750	.972	.862	.738	.593	.471	.370	.300	.268	.278	.277
.1000	.975	.877	.756	.614	.498	.400	.354	.333	.335	.339
.1500	.959	.871	.769	.634	.544	.472	.432	.419	.419	.405
.2000	.965	.893	.794	.675	.586	.531	.477	.466	.458	.440
.2500	.978	.903	.813	.716	.637	.576	.529	.507	.491	.482
.3000	.965	.912	.825	.737	.655	.597	.562	.543	.524	.500
.3500	.981	.925	.859	.778	.700	.648	.598	.566	.554	.536
.4000	.975	.937	.866	.778	.715	.657	.616	.590	.566	.542
.4500	.953	.921	.866	.819	.721	.672	.631	.602	.581	.565
.5000	.893	.887	.834	.798	.700	.657	.622	.590	.578	.562
.5500	.833	.805	.788	.737	.637	.612	.580	.552	.557	.545
.6000	.764	.767	.741	.675	.631	.591	.568	.549	.527	.521
.6500	.710	.686	.669	.614	.559	.537	.514	.499	.488	.476
.7000	.685	.604	.572	.532	.502	.469	.453	.445	.437	.432
.7815	.593	.500	.441	.409	.372	.346	.333	.333	.320	.315
.8179	.492	.437	.400	.368	.336	.328	.324	.310	.287	.292
.8252	.473	.428	.394	.368	.333	.340	.327	.327	.305	.304
.8325	.461	.434	.413	.389	.387	.394	.393	.381	.362	.375
Vane										
.0250	1.325	1.289	1.278	1.187	1.096	1.063	1.027	1.003	.955	.899
.0500	.814	.764	.706	.553	.444	.394	.369	.357	.338	.312
.1000	.508	.453	.369	.184	.108	.084	.078	.071	.060	.060
.1500	.398	.352	.263	.082	.024	.024	.021	.024	.012	.009
.2000	.334	.302	.213	.041	.012	.006	.009	.024	.009	.009
.3000	.259	.230	.156	.020	.012	.024	.027	.030	.021	.015
.4000	.218	.198	.131	.020	.024	.036	.048	.047	.048	.033
.5000	.183	.164	.106	.041	.042	.054	.081	.074	.069	.054
.6000	.161	.154	.119	.061	.075	.090	.093	.109	.093	.086
.7000	.174	.176	.150	.102	.132	.158	.171	.177	.150	.137
.8000	.237	.242	.225	.246	.246	.269	.297	.313	.278	.253
.9200	.694	.698	.681	.675	.733	.818	.898	.956	.859	.729
Flap										
.0125	.483	.296	.188	.061	.078	.099	.120	.133	.105	.077
.0250	.164	.057	.019	.000	.006	.012	.027	.030	.015	.015
.0500	.101	.053	.041	.020	.042	.045	.051	.056	.057	.054
.0750	.104	.075	.063	.041	.060	.066	.069	.080	.069	.077
.1000	.117	.101	.088	.061	.099	.101	.105	.112	.108	.107
.1500	.145	.129	.134	.102	.132	.137	.135	.142	.135	.131
.4000	.372	.349	.353	.348	.342	.337	.327	.325	.314	.307
.6000	.808	.805	.788	.778	.712	.698	.667	.628	.620	.601
.8000	.644	.632	.625	.614	.574	.561	.532	.507	.491	.482
.9000	.770	.761	.744	.737	.688	.678	.655	.617	.590	.607
Spoiler										
.2573	.972	1.532	1.581	1.614	1.595	1.659	1.640	1.651	1.579	1.597
.4984	1.099	1.566	1.581	1.635	1.607	1.671	1.661	1.639	1.561	1.605
.7485	1.093	1.566	1.584	1.629	1.610	1.668	1.646	1.639	1.579	1.608
.9940	1.183	1.597	1.599	1.654	1.640	1.719	1.735	1.761	1.646	1.654

TABLE X.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(c) Double slotted flap configuration with spoiler; $h_s = -0.010\bar{c}$; $\frac{y}{b/2} = 0.30$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0000	.870	1.305	2.157	2.917	3.501	4.297	4.291	3.900	3.092	2.320
.0125	1.012	1.526	2.283	2.929	3.448	3.957	3.947	3.781	3.060	2.299
.0250	1.042	1.409	2.375	3.033	3.564	4.115	3.979	3.795	3.098	2.311
.0500	1.060	1.292	2.966	3.917	3.883	3.984	3.959	3.816	3.015	2.281
.0750	1.072	1.255	1.840	4.036	5.194	5.212	4.407	3.900	3.039	2.290
.1000	1.078	1.228	1.255	2.497	4.012	4.878	4.343	3.813	3.048	2.302
.1500	1.108	1.249	1.277	1.426	2.125	3.351	3.708	3.434	2.847	2.249
.2000	1.139	1.255	1.332	1.359	1.582	2.421	3.049	3.061	2.725	2.207
.2500	1.148	1.262	1.348	1.374	1.457	1.997	2.571	2.752	2.639	2.207
.3000	1.181	1.280	1.354	1.386	1.415	1.794	2.326	2.542	2.512	2.172
.3500	1.193	1.274	1.366	1.386	1.403	1.670	2.052	2.274	2.373	2.142
.4000	1.226	1.308	1.391	1.408	1.424	1.636	1.941	2.140	2.302	2.119
.4500	1.271	1.342	1.415	1.429	1.427	1.621	1.895	2.064	2.196	2.089
.5000	1.298	1.375	1.449	1.445	1.448	1.594	1.804	1.938	2.077	2.039
.5500	1.343	1.397	1.446	1.457	1.454	1.597	1.743	1.857	2.018	2.030
.6000	1.389	1.434	1.489	1.484	1.487	1.600	1.726	1.822	1.968	2.003
.6500	1.461	1.520	1.560	1.561	1.546	1.636	1.714	1.801	1.912	1.977
.7521	1.681	1.742	1.754	1.764	1.725	1.809	1.801	1.842	1.870	1.909
.7934	2.030	2.040	2.031	2.049	2.015	2.091	2.093	2.067	1.991	1.950
.8099	2.259	2.246	2.228	2.236	2.188	2.303	2.280	2.218	2.071	1.962
.8141	2.346	2.320	2.308	2.309	2.266	2.385	2.364	2.303	2.122	1.983
.8182	2.404	2.363	2.366	2.368	2.316	2.439	2.399	2.344	2.160	1.991
.8224	2.560	2.542	2.505	2.506	2.448	2.582	2.536	2.484	2.270	2.036
Vane										
.0000	4.581	4.452	4.317	4.297	4.224	4.475	4.303	4.192	3.740	3.267
.0250	6.009	5.772	5.566	5.649	5.457	5.863	5.795	5.451	4.409	3.498
.0500	6.006	5.742	5.594	5.634	5.427	5.863	5.772	5.416	4.362	3.296
.1000	6.030	5.732	5.563	5.585	5.307	5.836	5.769	5.393	4.258	2.918
.1500	5.982	5.683	5.508	5.518	5.230	5.790	5.731	5.358	4.243	2.906
.2000	5.940	5.686	5.459	5.468	5.161	5.775	5.731	5.419	4.367	2.989
.3000	5.422	5.194	4.954	4.929	4.663	5.366	5.334	5.093	4.243	2.962
.4000	4.738	4.523	4.283	4.251	4.015	4.751	4.731	4.559	3.856	2.764
.5000	4.208	3.963	3.726	3.683	3.528	4.260	4.268	4.119	3.533	2.607
.6000	3.593	3.379	3.160	3.110	3.027	3.718	3.737	3.644	3.216	2.480
.7000	3.072	2.874	2.686	2.647	2.609	3.215	3.253	3.218	2.903	2.332
.8000	2.575	2.419	2.332	2.303	2.287	2.745	2.804	2.798	2.568	2.166
.9000	2.274	2.185	2.123	2.101	2.048	2.409	2.452	2.478	2.349	2.065
Flap										
.0000	1.485	1.471	1.474	1.349	1.233	1.427	1.460	1.458	1.408	1.290
.0125	2.129	2.132	2.074	2.009	1.907	2.212	2.227	2.253	2.184	2.003
.0250	2.608	2.591	2.505	2.435	2.349	2.769	2.787	2.819	2.752	2.521
.0500	2.578	2.588	2.495	2.420	2.400	2.857	2.898	2.962	2.926	2.669
.0750	2.512	2.511	2.425	2.352	2.364	2.842	2.892	2.965	2.947	2.657
.1000	2.389	2.357	2.262	2.221	2.257	2.703	2.755	2.828	2.790	2.503
.1500	1.946	1.957	1.935	1.932	1.994	2.354	2.390	2.457	2.447	2.252
.2000	1.741	1.711	1.686	1.690	1.734	1.979	2.035	2.090	2.068	1.923
.4000	1.602	1.637	1.637	1.653	1.692	1.815	1.842	1.877	1.879	1.829
.6000	1.578	1.609	1.625	1.638	1.639	1.727	1.694	1.664	1.663	1.687
.8000	1.584	1.625	1.631	1.653	1.642	1.745	1.711	1.676	1.681	1.698
Spoiler										
.2475	1.056	1.597	1.621	1.670	1.665	1.748	1.805	1.857	1.769	1.761
.4909	1.090	1.637	1.642	1.691	1.665	1.760	1.800	1.830	1.743	1.746
.7396	1.077	1.600	1.602	1.654	1.628	1.707	1.723	1.734	1.655	1.672
.9796	1.043	1.612	1.630	1.682	1.671	1.748	1.767	1.767	1.678	1.706

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0125	.979	.772	.742	.816	.958	1.170	1.259	1.271	1.198	1.042
.0250	1.000	.818	.732	.702	.722	.785	.828	.831	.811	.737
.0500	.988	.858	.742	.662	.597	.582	.563	.551	.547	.521
.0750	.991	.880	.760	.666	.591	.536	.498	.478	.468	.456
.1000	.985	.889	.775	.681	.594	.524	.478	.446	.447	.432
.1500	.994	.923	.809	.712	.615	.539	.481	.446	.432	.426
.2000	.988	.914	.825	.727	.627	.548	.484	.452	.441	.429
.2500	.991	.926	.837	.751	.657	.573	.498	.469	.453	.444
.3000	.976	.923	.843	.761	.669	.591	.522	.484	.476	.459
.3500	.961	.917	.855	.764	.681	.609	.536	.501	.494	.479
.4000	.937	.895	.843	.758	.681	.609	.548	.519	.506	.488
.4500	.895	.868	.818	.758	.672	.615	.560	.522	.515	.500
.5000	.852	.831	.788	.727	.654	.597	.542	.519	.503	.500
.5500	.795	.778	.738	.684	.615	.582	.531	.496	.497	.488
.6000	.723	.695	.671	.626	.561	.533	.487	.455	.453	.462
.6500	.669	.622	.591	.555	.504	.476	.449	.414	.423	.414
.7521	.500	.437	.406	.380	.346	.324	.306	.289	.278	.284
.7934	.434	.385	.360	.322	.301	.276	.262	.251	.237	.249
.8017	.416	.382	.351	.322	.296	.276	.268	.251	.252	.237
.8099	.413	.372	.342	.322	.310	.306	.292	.274	.272	.278
Vane										
.0250	1.702	1.686	1.625	1.509	1.379	1.367	1.268	1.204	1.113	1.012
.0500	.741	.766	.760	.620	.501	.430	.385	.347	.323	.290
.1000	.277	.308	.329	.205	.113	.058	.047	.032	.030	.033
.1500	.142	.182	.197	.107	.027	.015	.012	.000	.000	.000
.2000	.078	.105	.126	.043	.000	.000	.000	.000	.000	.000
.3000	.030	.049	.062	.021	.000	.003	.026	.009	.000	.000
.4000	.030	.028	.040	.018	.018	.027	.029	.020	.012	.021
.5000	.039	.043	.040	.021	.036	.045	.061	.044	.036	.036
.6000	.075	.049	.049	.046	.060	.088	.079	.079	.071	.062
.7000	.142	.105	.105	.113	.110	.133	.137	.140	.121	.101
.8000	.256	.237	.215	.209	.221	.267	.268	.262	.237	.225
.9200	.792	.738	.705	.712	.687	.806	.810	.802	.758	.701
Flap										
.0125	.214	.203	.203	.156	.143	.188	.195	.201	.178	.148
.0250	.030	.012	.018	.000	.006	.018	.029	.017	.000	.009
.0500	.018	.006	.000	.000	.000	.000	.000	.000	.000	.000
.0750	.024	.018	.003	.000	.015	.015	.012	.003	.000	.000
.1000	.042	.031	.025	.018	.018	.015	.023	.015	.012	.015
.1500	.066	.040	.043	.049	.060	.052	.047	.032	.030	.030
.2000	.093	.095	.080	.067	.072	.064	.067	.055	.047	.062
.4000	.241	.215	.206	.196	.185	.179	.163	.140	.130	.139
.6000	.361	.360	.357	.319	.299	.294	.268	.257	.254	.252
.8000	.548	.545	.511	.512	.472	.448	.423	.417	.399	.420
.9000	.717	.720	.683	.693	.666	.661	.609	.600	.580	.601
Spoiler										
.2587	.994	1.548	1.584	1.626	1.604	1.671	1.667	1.660	1.585	1.611
.5024	1.028	1.560	1.581	1.629	1.610	1.692	1.687	1.687	1.597	1.623
.7531	1.099	1.579	1.587	1.638	1.616	1.692	1.693	1.687	1.597	1.620
.9940	1.124	1.612	1.627	1.667	1.637	1.743	1.761	1.755	1.646	1.640

TABLE X.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(d) Double slotted flap configuration with spoiler; $h_s = -0.010\bar{c}$; $\frac{y}{b/2} = 0.43$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	.868	1.856	2.163	2.640	2.907	2.886	2.709	2.623	2.479	2.178
.0125	1.123	1.821	2.150	2.619	2.883	2.895	2.823	2.699	2.488	2.202
.0250	1.139	1.771	2.200	2.660	2.889	2.937	2.877	2.717	2.482	2.190
.0500	1.120	1.745	2.253	2.722	3.027	3.006	2.883	2.749	2.476	2.226
.0750	1.142	1.745	2.525	2.803	2.967	2.904	2.871	2.761	2.476	2.208
.1000	1.158	1.573	2.875	2.988	2.913	2.839	2.892	2.776	2.473	2.196
.1500	1.174	1.302	2.419	3.929	3.562	3.101	2.976	2.770	2.473	2.208
.2000	1.202	1.308	1.622	3.131	3.892	3.382	2.970	2.720	2.461	2.199
.2500	1.221	1.327	1.303	2.087	3.150	3.215	2.847	2.646	2.431	2.178
.3000	1.253	1.371	1.334	1.617	2.541	2.982	2.742	2.584	2.398	2.172
.3500	1.278	1.378	1.359	1.432	2.057	2.639	2.556	2.451	2.338	2.149
.4000	1.309	1.415	1.416	1.392	1.829	2.406	2.444	2.381	2.284	2.131
.4500	1.363	1.453	1.466	1.392	1.697	2.260	2.330	2.301	2.225	2.104
.5000	1.404	1.475	1.513	1.412	1.574	2.018	2.174	2.192	2.153	2.086
.5500	1.480	1.554	1.572	1.494	1.568	1.943	2.102	2.133	2.111	2.056
.6000	1.571	1.635	1.653	1.596	1.586	1.872	2.018	2.071	2.048	2.021
.7000	1.802	1.906	1.941	1.883	1.760	1.842	1.928	1.991	1.973	1.979
.7500	2.237	2.330	2.372	2.291	2.102	2.069	2.075	2.050	1.985	1.985
.7700	2.565	2.683	2.719	2.701	2.414	2.292	2.237	2.156	2.021	1.997
.7750	2.647	2.774	2.813	2.763	2.501	2.379	2.285	2.177	2.036	1.988
.7800	2.798	2.893	2.941	2.824	2.631	2.466	2.348	2.215	2.039	2.000
Vane										
.0000	4.143	4.117	4.100	4.031	3.673	3.355	3.108	2.870	2.548	2.488
.0250	6.736	6.664	6.722	6.794	6.081	5.337	4.766	4.213	3.314	3.065
.0500	6.938	6.828	6.897	6.958	6.264	5.466	4.829	4.221	3.168	2.857
.1000	7.121	6.988	7.072	7.162	6.456	5.546	4.781	4.089	2.886	2.553
.1500	7.310	7.164	7.253	7.367	6.673	5.710	4.883	4.127	2.829	2.452
.2000	7.591	7.441	7.559	7.694	6.991	6.015	5.165	4.316	2.961	2.559
.3000	6.928	6.784	6.863	6.998	6.408	5.522	4.736	3.956	2.734	2.387
.4000	6.174	6.042	6.084	6.241	5.781	5.045	4.348	3.667	2.596	2.309
.5000	5.433	5.318	5.363	5.423	5.114	4.522	3.955	3.375	2.467	2.226
.6000	4.723	4.611	4.647	4.788	4.456	3.997	3.562	3.095	2.350	2.170
.7000	4.111	4.026	4.047	4.154	3.913	3.567	3.237	2.864	2.260	2.119
.8000	3.489	3.428	3.447	3.520	3.330	3.107	2.883	2.611	2.138	2.042
.9000	3.004	2.956	2.997	3.049	2.853	2.692	2.535	2.331	1.997	1.958
Flap										
.0000	2.284	2.365	2.538	2.537	2.300	2.134	2.012	1.797	1.578	1.553
.0125	2.786	2.853	2.953	2.967	2.760	2.597	2.450	2.218	1.979	1.952
.0250	3.401	3.497	3.569	3.622	3.387	3.227	3.057	2.782	2.518	2.467
.0500	3.442	3.500	3.525	3.622	3.429	3.316	3.174	2.921	2.599	2.509
.0750	3.335	3.400	3.434	3.499	3.342	3.269	3.177	2.915	2.575	2.455
.1000	3.246	3.239	3.247	3.336	3.159	3.113	3.039	2.817	2.464	2.354
.1500	2.511	2.469	2.472	2.517	2.468	2.543	2.559	2.428	2.159	2.086
.2000	2.161	2.101	2.109	2.149	2.090	2.104	2.135	2.065	1.886	1.875
.4000	1.565	1.560	1.584	1.637	1.598	1.690	1.778	1.797	1.778	1.803
.6000	1.300	1.333	1.366	1.473	1.477	1.442	1.420	1.466	1.590	1.690
.8000	1.246	1.333	1.366	1.473	1.483	1.445	1.426	1.460	1.572	1.667
Spoiler										
.2443	.898	1.415	1.477	1.539	1.559	1.638	1.717	1.797	1.787	1.787
.4906	.842	1.329	1.391	1.461	1.498	1.566	1.623	1.675	1.684	1.715
.7370	.811	1.268	1.327	1.402	1.444	1.476	1.525	1.579	1.623	1.660
.9856	.777	1.338	1.410	1.492	1.535	1.539	1.502	1.537	1.585	1.640

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	.975	.780	.828	.962	1.141	1.257	1.345	1.366	1.353	1.280
.0250	.994	.805	.772	.778	.823	.881	.919	.947	.931	.911
.0500	1.000	.837	.775	.716	.685	.710	.691	.687	.683	.684
.0750	1.000	.855	.791	.716	.652	.624	.613	.605	.602	.607
.1000	1.000	.871	.797	.716	.643	.597	.571	.561	.548	.559
.1500	.988	.884	.822	.737	.637	.588	.547	.528	.509	.521
.2000	.975	.877	.816	.737	.646	.582	.538	.516	.506	.503
.2500	.956	.877	.822	.737	.652	.579	.535	.516	.497	.500
.3000	.937	.859	.803	.716	.643	.579	.544	.510	.488	.494
.3500	.905	.840	.797	.716	.640	.579	.547	.516	.494	.506
.4000	.868	.808	.772	.675	.625	.579	.529	.513	.494	.509
.4500	.823	.767	.734	.655	.607	.558	.520	.502	.482	.491
.5000	.767	.708	.691	.634	.574	.531	.495	.484	.449	.470
.5500	.713	.604	.603	.573	.471	.469	.444	.422	.386	.411
.6000	.625	.576	.556	.471	.477	.451	.420	.410	.389	.405
.7000	.426	.403	.409	.368	.333	.310	.306	.286	.266	.280
.7500	.363	.340	.353	.348	.297	.272	.261	.248	.219	.241
.7600	.338	.327	.350	.348	.288	.266	.252	.242	.213	.232
.7700	.363	.346	.350	.368	.336	.304	.282	.268	.246	.247
Vane										
.0250	.984	1.000	1.072	.941	.838	.722	.652	.596	.554	.562
.0500	.426	.459	.563	.450	.378	.287	.246	.227	.204	.185
.1000	.164	.198	.316	.225	.168	.110	.090	.074	.051	.068
.1500	.050	.075	.159	.082	.090	.048	.033	.047	.033	.048
.2000	.016	.022	.116	.041	.054	.039	.057	.038	.015	.024
.3000	.041	.019	.072	.020	.054	.054	.024	.050	.045	.080
.4000	.050	.035	.059	.020	.036	.036	.036	.047	.048	.071
.5000	.047	.038	.069	.041	.090	.063	.099	.068	.045	.071
.6000	.088	.075	.103	.061	.096	.090	.090	.091	.066	.083
.7000	.158	.135	.153	.123	.150	.149	.135	.145	.108	.131
.8000	.325	.299	.316	.266	.306	.296	.279	.257	.201	.235
.9200	1.032	1.000	1.031	1.044	.985	.916	.865	.799	.692	.723
Flap										
.0125	.249	.258	.344	.286	.288	.260	.252	.215	.177	.185
.0250	.000	.041	.056	.000	.060	.021	.036	.024	.027	.042
.0500	.000	.000	.031	.000	.012	.000	.015	.012	.003	.054
.0750	.000	.022	.069	.000	.027	.042	.027	.009	.051	.048
.1000	.009	.000	.053	.000	.048	.030	.033	.024	.063	.086
.1500	.044	.044	.075	.020	.051	.072	.033	.024	.051	.057
.2000	.082	.063	.094	.041	.072	.101	.072	.071	.051	.086
.4000	.208	.170	.188	.143	.174	.155	.156	.139	.129	.149
.6000	.325	.292	.309	.246	.282	.266	.258	.248	.246	.259
.8000	.508	.459	.472	.450	.474	.430	.417	.413	.419	.449
.9000	.640	.616	.631	.614	.619	.582	.580	.569	.596	.640
Spoiler										
.2587	.746	1.286	1.367	1.439	1.492	1.482	1.440	1.463	1.535	1.576
.5039	.827	1.326	1.385	1.467	1.498	1.494	1.440	1.487	1.538	1.602
.7508	.786	1.329	1.385	1.477	1.511	1.491	1.446	1.481	1.550	1.591
.9940	.923	1.403	1.456	1.533	1.571	1.557	1.484	1.537	1.573	1.631

TABLE X.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(e) Double slotted flap configuration with spoiler; $h_S = -0.010\bar{c}$; $\frac{y}{b/2} = 0.55$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.292	1.908	2.071	2.389	2.442	2.366	2.180	2.178	2.190	2.086
.0125	1.340	1.917	2.117	2.447	2.481	2.406	2.253	2.230	2.210	2.080
.0250	1.235	1.877	2.089	2.414	2.430	2.397	2.256	2.227	2.216	2.089
.0500	1.220	1.840	2.145	2.475	2.463	2.497	2.297	2.262	2.222	2.092
.0750	1.208	1.877	2.215	2.539	2.495	2.469	2.297	2.259	2.216	2.077
.1000	1.229	1.794	2.274	2.592	2.534	2.445	2.317	2.265	2.210	2.077
.1500	1.247	1.594	2.652	2.635	2.463	2.436	2.349	2.274	2.222	2.077
.2000	1.268	1.428	2.862	2.895	2.427	2.430	2.347	2.262	2.210	2.071
.2500	1.307	1.403	2.708	3.545	2.579	2.479	2.355	2.262	2.219	2.077
.3000	1.325	1.409	2.215	3.588	2.872	2.506	2.341	2.245	2.204	2.074
.3500	1.370	1.443	1.717	3.131	2.907	2.518	2.294	2.227	2.193	2.068
.4000	1.428	1.489	1.434	2.536	2.794	2.460	2.259	2.186	2.175	2.051
.4500	1.470	1.539	1.366	2.027	2.591	2.376	2.207	2.163	2.148	2.054
.5000	1.548	1.606	1.437	1.751	2.430	2.303	2.151	2.125	2.110	2.042
.6201	1.726	1.834	1.760	1.647	2.000	2.079	2.008	2.011	2.024	2.012
.6834	2.193	2.308	2.252	2.052	2.095	2.115	2.032	2.023	2.033	2.018
.7087	2.518	2.643	2.588	2.346	2.209	2.145	2.058	2.046	2.045	2.027
.7151	2.551	2.689	2.628	2.392	2.221	2.151	2.049	2.052	2.045	2.021
.7214	2.629	2.822	2.806	2.576	2.310	2.200	2.093	2.081	2.071	2.033
.7278	2.750	2.951	2.939	2.671	2.352	2.206	2.096	2.084	2.071	2.039
Vane										
.0000	3.331	3.354	3.203	2.837	2.424	2.273	2.166	2.131	2.092	2.012
.0250	6.434	6.523	6.459	5.751	4.480	3.857	3.600	3.440	3.196	2.906
.0500	6.575	6.640	6.622	5.922	4.537	3.854	3.545	3.326	3.048	2.737
.1000	7.018	7.059	7.126	6.438	4.692	3.788	3.399	3.174	2.832	2.533
.1500	7.072	7.108	7.212	6.536	4.651	3.700	3.326	3.075	2.707	2.429
.2000	7.102	7.099	7.246	6.612	4.675	3.709	3.320	3.052	2.651	2.388
.3000	6.750	6.748	6.917	6.431	4.507	3.545	3.169	2.927	2.548	2.290
.4000	6.024	6.022	6.163	5.827	4.075	3.227	2.892	2.682	2.382	2.178
.5000	5.473	5.450	5.612	5.358	3.779	3.024	2.755	2.580	2.314	2.130
.6000	4.741	4.739	4.859	4.665	3.352	2.769	2.551	2.425	2.210	2.086
.7000	4.105	4.123	4.194	4.070	3.021	2.554	2.384	2.291	2.130	2.048
.8000	3.578	3.597	3.649	3.545	2.740	2.388	2.242	2.180	2.074	2.015
.9000	3.060	3.080	3.095	3.030	2.454	2.194	2.081	2.038	1.977	1.968
Flap										
.0000	1.879	1.945	2.037	2.046	1.695	1.473	1.385	1.358	1.335	1.349
.0125	2.802	3.000	3.052	3.027	2.528	2.233	2.105	2.081	2.051	2.071
.0250	3.416	3.563	3.603	3.570	3.006	2.663	2.501	2.475	2.447	2.450
.0500	3.506	3.662	3.702	3.705	3.134	2.788	2.618	2.577	2.518	2.462
.0750	3.500	3.665	3.705	3.720	3.188	2.830	2.638	2.589	2.500	2.409
.1000	3.392	3.486	3.520	3.542	3.045	2.736	2.551	2.507	2.412	2.314
.1500	2.711	2.763	2.726	2.822	2.543	2.336	2.198	2.186	2.098	2.027
.2000	2.364	2.406	2.394	2.432	2.203	2.073	1.956	1.941	1.914	1.867
.4000	1.822	1.852	1.812	1.843	1.809	1.803	1.740	1.752	1.769	1.781
.6000	1.416	1.452	1.385	1.408	1.451	1.497	1.437	1.487	1.574	1.681
.8000	1.398	1.458	1.394	1.402	1.469	1.497	1.443	1.472	1.571	1.684
Spoiler										
.2443	1.022	1.665	1.709	1.751	1.776	1.799	1.767	1.806	1.751	1.770
.4881	.978	1.539	1.572	1.620	1.680	1.713	1.690	1.737	1.705	1.721
.7416	.861	1.391	1.416	1.477	1.547	1.599	1.584	1.651	1.658	1.672
.9856	.820	1.363	1.349	1.402	1.486	1.527	1.519	1.576	1.626	1.651

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	.922	.818	.923	1.193	1.355	1.460	1.490	1.551	1.601	1.589
.0250	.961	.815	.803	.880	.943	1.009	1.035	1.073	1.127	1.119
.0500	.964	.849	.754	.751	.746	.758	.767	.793	.799	.834
.0750	.949	.834	.738	.699	.672	.667	.662	.673	.686	.707
.1000	.949	.843	.738	.693	.645	.624	.612	.624	.613	.630
.1500	.940	.840	.738	.684	.627	.588	.560	.568	.556	.571
.2000	.916	.828	.738	.681	.627	.582	.533	.542	.533	.544
.2500	.892	.825	.729	.681	.600	.573	.522	.528	.509	.530
.3000	.855	.800	.705	.672	.603	.558	.516	.516	.509	.518
.3500	.828	.772	.695	.659	.591	.554	.507	.516	.488	.506
.4000	.777	.732	.671	.629	.561	.533	.493	.490	.468	.491
.4500	.723	.695	.628	.589	.546	.509	.458	.481	.453	.470
.5000	.681	.646	.585	.546	.507	.479	.437	.440	.423	.444
.6201	.470	.465	.428	.405	.370	.333	.315	.326	.305	.328
.6834	.407	.382	.366	.325	.310	.297	.271	.268	.260	.266
.6961	.398	.385	.363	.337	.299	.279	.251	.262	.254	.257
.7087	.404	.394	.357	.334	.299	.282	.254	.274	.257	.269
Vane										
.0250	1.217	1.166	1.157	1.009	.866	.791	.714	.705	.672	.669
.0500	.524	.505	.588	.500	.427	.361	.289	.283	.275	.275
.1000	.217	.188	.298	.242	.206	.170	.114	.108	.104	.121
.1500	.111	.086	.197	.150	.143	.082	.070	.105	.056	.077
.2000	.066	.062	.135	.117	.096	.091	.064	.090	.047	.077
.3000	.078	.086	.077	.077	.066	.091	.079	.085	.059	.065
.4000	.087	.080	.077	.080	.093	.082	.090	.093	.080	.077
.5000	.090	.114	.092	.117	.104	.091	.131	.099	.092	.109
.6000	.139	.163	.135	.153	.137	.133	.122	.140	.127	.142
.7000	.190	.222	.191	.199	.191	.176	.163	.189	.151	.180
.8000	.377	.397	.360	.353	.304	.297	.277	.292	.246	.290
.9200	1.111	1.151	1.126	1.107	.919	.833	.784	.790	.772	.784
Flap										
.0125	.271	.289	.329	.365	.272	.248	.236	.254	.222	.243
.0250	.039	.058	.052	.080	.075	.082	.073	.105	.050	.083
.0500	.042	.046	.055	.046	.060	.061	.055	.082	.044	.089
.0750	.054	.065	.071	.055	.090	.076	.064	.082	.059	.083
.1000	.096	.080	.086	.083	.084	.109	.076	.111	.065	.092
.1500	.108	.105	.102	.095	.096	.103	.102	.117	.086	.101
.2000	.133	.142	.123	.117	.122	.127	.111	.117	.109	.139
.4000	.280	.265	.246	.248	.239	.221	.192	.210	.189	.195
.6000	.395	.394	.351	.356	.352	.333	.289	.312	.299	.305
.8000	.545	.538	.505	.506	.498	.494	.452	.466	.456	.479
.9000	.678	.698	.640	.653	.648	.651	.600	.618	.621	.660
Spoiler										
.2584	.802	1.317	1.309	1.358	1.426	1.473	1.469	1.519	1.561	1.591
.5039	.755	1.311	1.327	1.371	1.429	1.479	1.469	1.522	1.556	1.594
.7508	.793	1.323	1.330	1.371	1.426	1.482	1.472	1.531	1.573	1.608
.9940	.811	1.388	1.379	1.420	1.477	1.521	1.510	1.561	1.591	1.631

TABLE X.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Concluded

(f) Double slotted flap configuration with spoiler; $h_s = -0.010\bar{c}$; $\frac{y}{b/2} = 0.72$

Upper surface

x/c	C_p for -									
	α = -4°	α = 0°	α = 4°	α = 8°	α = 12°	α = 16°	α = 20°	α = 22°	α = 24°	α = 26°
Wing										
.0000	1.811	2.237	2.434	2.383	2.166	2.042	1.929	1.934	1.889	1.893
.0125	1.697	2.188	2.333	2.274	2.100	1.976	1.909	1.937	1.883	1.899
.0250	1.669	2.225	2.370	2.314	2.112	2.024	1.938	1.952	1.895	1.902
.0500	1.656	2.265	2.413	2.330	2.133	2.063	1.959	1.970	1.898	1.911
.0750	1.437	2.320	2.480	2.367	2.154	2.084	1.977	1.976	1.898	1.899
.1000	1.136	2.314	2.459	2.355	2.142	2.063	1.971	1.976	1.904	1.902
.1500	1.111	2.455	2.532	2.389	2.148	2.072	1.994	1.991	1.901	1.911
.2000	1.207	2.659	2.624	2.386	2.133	2.093	2.009	2.000	1.909	1.917
.2500	1.248	2.683	2.712	2.364	2.124	2.090	2.006	2.000	1.909	1.919
.3000	1.254	2.532	2.920	2.352	2.133	2.090	2.000	1.997	1.918	1.919
.3500	1.356	2.342	3.131	2.330	2.154	2.090	2.003	2.000	1.924	1.928
.4000	1.353	2.037	3.226	2.414	2.169	2.084	1.997	1.994	1.927	1.934
.4500	1.402	1.775	3.238	2.536	2.175	2.075	1.991	1.997	1.933	1.934
.5000	1.424	1.699	3.220	2.654	2.200	2.084	1.991	1.988	1.936	1.945
.5500	1.446	1.625	3.012	2.576	2.184	2.069	1.979	1.985	1.930	1.942
.6000	1.390	1.631	2.780	2.492	2.169	2.051	1.959	1.976	1.936	1.942
.6500	1.393	1.572	2.480	2.324	2.130	2.036	1.953	1.982	1.953	1.963
.7000	1.396	1.517	2.254	2.258	2.118	2.018	1.947	1.970	1.950	1.968
.7500	1.347	1.471	2.098	2.246	2.100	2.006	1.947	1.976	1.959	1.968
.8000	1.260	1.375	1.942	2.193	2.066	1.994	1.947	1.964	1.962	1.974
.8500	1.257	1.317	1.832	2.065	2.015	1.955	1.929	1.964	1.965	1.977
.9000	1.139	1.298	1.737	1.990	1.991	1.958	1.923	1.964	1.962	1.974

Lower surface

x/c	C_p for -									
	α = -4°	α = 0°	α = 4°	α = 8°	α = 12°	α = 16°	α = 20°	α = 22°	α = 24°	α = 26°
Wing										
.0125	.598	.874	1.058	1.165	1.224	1.281	1.310	1.385	1.389	1.427
.0250	.622	.788	.887	.947	.988	1.030	1.074	1.143	1.158	1.190
.0500	.628	.745	.777	.788	.807	.835	.850	.904	.918	.937
.0750	.585	.757	.755	.751	.749	.757	.767	.815	.822	.839
.1000	.591	.757	.740	.726	.716	.713	.714	.746	.763	.775
.1500	.610	.763	.746	.716	.683	.659	.667	.696	.673	.700
.2000	.653	.800	.771	.735	.686	.671	.661	.684	.673	.683
.2500	.706	.831	.804	.760	.704	.680	.667	.687	.678	.677
.3000	.771	.883	.862	.801	.740	.716	.693	.710	.702	.703
.3500	.854	.985	.969	.903	.825	.796	.752	.785	.769	.764
.4000	.926	1.132	1.135	1.047	.940	.904	.853	.881	.866	.847
.4500	1.130	1.372	1.367	1.234	1.085	1.027	.976	1.003	.974	.965
.5000	1.350	1.643	1.645	1.445	1.257	1.195	1.130	1.143	1.105	1.092
.5500	1.464	1.883	1.905	1.679	1.447	1.386	1.322	1.352	1.304	1.285
.6000	1.601	2.105	2.101	1.891	1.656	1.581	1.540	1.567	1.526	1.510
.6500	1.830	2.308	2.223	1.978	1.779	1.763	1.735	1.794	1.757	1.712
.7000	1.932	2.360	2.229	1.990	1.840	1.844	1.826	1.895	1.822	1.772
.7500	1.941	2.314	2.168	1.987	1.837	1.838	1.805	1.860	1.798	1.772
.8000	1.752	2.154	2.064	1.944	1.801	1.769	1.729	1.773	1.734	1.738
.8500	1.563	2.006	1.969	1.919	1.761	1.728	1.699	1.743	1.711	1.726
.9000	1.378	1.840	1.868	1.891	1.749	1.737	1.693	1.734	1.702	1.712
.9500	1.235	1.646	1.743	1.860	1.770	1.737	1.711	1.758	1.719	1.721

TABLE XI.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE

(a) Double slotted flap configuration with spoiler; $h_s = -0.035\bar{c}$; $\frac{y}{b/2} = 0$

Upper surface

x/c	C_p for -									
	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Fuselage										
.0000	.006	.018	.009	.024	.015	.050	.076	.082	.124	.149
.0500	.768	.846	.900	.946	.976	1.009	1.058	1.067	1.072	1.111
.1000	.843	.931	.967	1.000	1.030	1.058	1.102	1.116	1.124	1.172
.1500	.941	.991	1.030	1.054	1.071	1.087	1.116	1.140	1.142	1.160
.2000	.994	1.030	1.063	1.084	1.098	1.108	1.125	1.140	1.124	1.151
.2500	1.038	1.069	1.091	1.102	1.113	1.119	1.137	1.151	1.142	1.163
.3000	1.069	1.087	1.103	1.093	1.101	1.105	1.116	1.125	1.107	1.154
.3500	1.078	1.072	1.082	1.090	1.098	1.087	1.096	1.116	1.098	1.151
.4000	1.038	1.036	1.048	1.060	1.071	1.079	1.096	1.116	1.116	1.160
.4500	1.041	1.048	1.069	1.099	1.113	1.137	1.189	1.218	1.220	1.288
.5000	1.047	1.063	1.103	1.144	1.190	1.224	1.306	1.350	1.376	1.609
.5500	1.053	1.096	1.160	1.219	1.262	1.303	1.361	1.414	1.532	2.035
.6000	1.110	1.139	1.214	1.257	1.301	1.309	1.361	1.402	1.575	2.023
.6500	1.144	1.190	1.245	1.293	1.301	1.326	1.318	1.344	1.451	1.901
.7000	1.188	1.232	1.275	1.311	1.327	1.320	1.315	1.318	1.402	1.819
.7500	1.251	1.286	1.311	1.338	1.342	1.353	1.347	1.370	1.410	1.732
.8000	1.307	1.331	1.336	1.353	1.375	1.399	1.417	1.452	1.457	1.629
.8500	1.317	1.313	1.329	1.335	1.360	1.390	1.440	1.492	1.494	1.539
.9000	1.229	1.232	1.220	1.240	1.274	1.309	1.361	1.402	1.428	1.452
.9500	1.154	1.163	1.157	1.162	1.202	1.236	1.303	1.335	1.370	1.411
.9940	1.216	1.199	1.184	1.195	1.211	1.256	1.320	1.353	1.399	1.431

Lower surface

x/c	C_p for -									
	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Fuselage										
.0500	.853	.810	.743	.695	.622	.548	.496	.469	.419	.394
.1000	.922	.907	.849	.796	.720	.665	.586	.571	.523	.510
.1500	.994	.964	.915	.868	.792	.735	.665	.641	.590	.577
.2000	1.016	.997	.952	.913	.851	.793	.732	.703	.653	.644
.2500	1.044	1.015	.991	.949	.884	.845	.778	.755	.711	.682
.3000	1.050	1.039	1.000	.976	.923	.877	.816	.793	.746	.723
.3500	1.028	1.021	.997	.979	.926	.892	.834	.816	.763	.740
.4000	.969	.961	.933	.916	.872	.834	.778	.761	.717	.685
.4500	.931	.925	.897	.880	.815	.775	.711	.705	.644	.615
.5000	.897	.892	.846	.823	.759	.705	.638	.621	.575	.548
.5500	.856	.852	.798	.766	.693	.633	.577	.551	.520	.490
.6000	.809	.783	.725	.686	.625	.568	.525	.501	.465	.440
.7500	.903	.898	.864	.832	.792	.752	.714	.703	.673	.673
.9500	1.060	1.078	1.106	1.135	1.146	1.143	1.163	1.183	1.205	1.271
.9940	1.028	1.111	1.184	1.222	1.247	1.274	1.318	1.355	1.361	1.411

TABLE XI.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(b) Double slotted flap configuration with spoiler; $h_s = -0.035\bar{c}$; $\frac{y}{b/2} = 0.21$

Upper surface

	C_p for -									
x/c	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0000	.807	.902	1.778	3.259	5.255	**7.575**	9.658	9.092	3.889	2.542
.0125	.945	1.220	1.643	3.281	7.382	**10.444**	**11.841**	9.065	3.497	2.313
.0250	.952	1.159	1.502	2.500	3.620	6.136	8.390	7.971	3.509	2.345
.0500	1.010	1.156	1.423	1.616	1.994	2.676	4.003	5.198	3.150	2.258
.0750	1.026	1.147	1.373	1.534	1.801	2.157	2.814	3.758	3.015	2.261
.1000	1.042	1.165	1.361	1.506	1.724	1.982	2.531	3.227	2.916	2.244
.1500	1.067	1.165	1.348	1.464	1.614	1.786	2.174	2.552	2.719	2.189
.2000	1.100	1.190	1.354	1.439	1.558	1.715	2.003	2.280	2.560	2.139
.2500	1.125	1.199	1.348	1.421	1.504	1.656	1.894	2.077	2.341	2.105
.3000	1.151	1.220	1.354	1.409	1.489	1.605	1.808	1.923	2.210	2.073
.3500	1.164	1.229	1.354	1.403	1.448	1.564	1.696	1.811	2.066	2.050
.4000	1.190	1.242	1.351	1.396	1.445	1.558	1.625	1.729	1.964	2.035
.4500	1.235	1.269	1.367	1.399	1.436	1.537	1.593	1.676	1.880	2.015
.5000	1.283	1.303	1.398	1.418	1.454	1.513	1.587	1.640	1.826	2.000
.5500	1.299	1.330	1.414	1.421	1.451	1.489	1.534	1.611	1.754	1.966
.6000	1.337	1.367	1.436	1.439	1.469	1.489	1.552	1.617	1.713	1.937
.6500	1.389	1.410	1.464	1.464	1.492	1.507	1.587	1.652	1.686	1.913
.7000	1.456	1.459	1.514	1.521	1.534	1.543	1.649	1.679	1.680	1.896
.7815	1.646	1.605	1.658	1.656	1.664	1.685	1.717	1.714	1.719	1.867
.8179	1.807	1.758	1.806	1.781	1.819	1.905	1.888	1.888	1.862	1.902
.8325	1.923	1.853	1.875	1.863	1.902	2.006	2.006	2.006	1.970	1.928
.8362	1.929	1.887	1.909	1.890	1.934	2.047	2.053	2.059	2.009	1.948
.8398	1.903	1.838	1.859	1.848	1.890	2.000	2.003	2.024	1.982	1.934
.8434	1.897	1.942	1.969	1.951	1.982	2.121	2.154	2.165	2.090	2.000
Vane										
.0000	2.334	2.214	2.251	2.232	2.288	2.415	2.395	2.404	2.314	2.099
.0250	3.958	3.752	3.762	3.720	3.911	4.418	4.514	4.578	4.260	3.514
.0500	3.774	3.529	3.502	3.427	3.655	4.160	4.245	4.295	3.935	3.203
.1000	3.665	3.434	3.411	3.290	3.486	4.038	4.183	4.266	3.889	2.928
.1500	3.533	3.336	3.282	3.168	3.382	3.905	4.106	4.207	3.871	2.882
.2000	3.379	3.220	3.157	3.079	3.305	3.813	4.053	4.189	3.901	2.922
.3000	2.884	2.823	2.765	2.717	2.979	3.430	3.749	3.944	3.772	2.899
.4000	2.566	2.480	2.439	2.433	2.691	3.023	3.375	3.602	3.521	2.766
.5000	2.373	2.238	2.226	2.256	2.513	2.724	3.059	3.263	3.293	2.618
.6000	2.151	2.043	2.047	2.031	2.199	2.391	2.614	2.785	2.850	2.406
.7000	2.115	1.988	1.981	1.970	2.133	2.320	2.525	2.661	2.671	2.299
.8000	2.128	1.969	1.950	1.939	2.127	2.305	2.543	2.676	2.614	2.189
.9000	2.090	1.942	1.912	1.915	2.086	2.255	2.505	2.655	2.572	2.105
Flap										
.0000	.881	1.287	1.448	1.339	1.231	1.303	1.404	1.505	1.491	1.328
.0125	2.212	2.128	2.094	1.951	2.038	2.219	2.451	2.575	2.416	1.995
.0250	2.392	2.275	2.276	2.229	2.359	2.590	2.862	3.015	2.838	2.357
.0500	2.447	2.358	2.351	2.336	2.480	2.724	2.988	3.151	3.018	2.632
.0750	2.305	2.193	2.195	2.165	2.291	2.516	2.738	2.873	2.766	2.470
.1000	2.109	2.006	1.994	1.976	2.080	2.249	2.451	2.558	2.503	2.287
.1500	1.800	1.709	1.702	1.674	1.748	1.890	2.050	2.154	2.144	1.992
.2000	1.736	1.648	1.646	1.616	1.670	1.813	1.950	2.071	2.090	1.948
.4000	1.775	1.712	1.709	1.668	1.718	1.822	1.953	2.080	2.111	1.937
.6000	1.653	1.584	1.618	1.646	1.647	1.727	1.735	1.758	1.763	1.815
.8000	1.730	1.651	1.680	1.692	1.697	1.759	1.752	1.776	1.832	1.867
Spoiler										
.2429	1.483	1.617	1.577	1.602	1.625	1.734	1.851	1.921	1.896	1.880
.4852	1.417	1.596	1.562	1.584	1.601	1.702	1.810	1.869	1.841	1.831
.7367	1.383	1.575	1.547	1.566	1.583	1.667	1.775	1.822	1.803	1.819
.9796	1.470	1.602	1.580	1.599	1.601	1.679	1.752	1.784	1.783	1.857

Lower surface

	C_p for -									
x/c	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0125	1.016	.786	.665	.585	.579	.650	.708	.690	.461	.371
.0250	.997	.810	.693	.549	.460	.389	.342	.316	.243	.206
.0500	.968	.847	.734	.598	.454	.338	.260	.218	.210	.220
.0750	.974	.853	.752	.616	.484	.368	.298	.260	.272	.278
.1000	.993	.862	.765	.640	.522	.421	.348	.333	.338	.342
.1500	.971	.856	.774	.659	.561	.487	.428	.410	.416	.415
.2000	.981	.872	.803	.701	.602	.537	.481	.469	.452	.452
.2500	.993	.893	.821	.729	.647	.582	.519	.507	.491	.490
.3000	.977	.896	.834	.753	.674	.614	.558	.540	.518	.513
.3500	.987	.914	.862	.787	.709	.653	.584	.572	.548	.542
.4000	.981	.917	.878	.802	.727	.674	.614	.599	.575	.565
.4500	.958	.908	.868	.802	.727	.688	.625	.599	.593	.588
.5000	.874	.859	.840	.784	.715	.679	.617	.599	.581	.574
.5500	.810	.789	.790	.750	.679	.629	.572	.555	.557	.548
.6000	.746	.752	.746	.695	.644	.614	.569	.543	.530	.533
.6500	.707	.676	.668	.634	.584	.561	.519	.490	.488	.484
.7000	.675	.599	.574	.543	.519	.495	.454	.437	.434	.435
.7815	.585	.511	.445	.409	.383	.371	.336	.336	.323	.333
.8179	.505	.434	.404	.366	.347	.338	.319	.307	.293	.301
.8252	.476	.425	.401	.372	.353	.344	.325	.325	.317	.319
.8325	.469	.431	.414	.396	.386	.404	.384	.378	.365	.383
Vane										
.0250	1.321	1.257	1.276	1.177	1.095	1.065	1.012	1.000	.964	.904
.0500	.820	.755	.712	.585	.460	.404	.360	.354	.335	.325
.1000	.511	.462	.379	.241	.128	.089	.065	.074	.060	**.072**
.1500	.405	.352	.270	.146	.045	.033	.030	.030	.006	.023
.2000	.334	.300	.232	.104	.027	.012	.009	.006	.000	.014
.3000	.260	.229	.188	.055	.021	.036	.032	.030	.009	.023
.4000	.215	.208	.141	.052	.033	.042	.038	.047	.036	.052
.5000	.174	.177	.129	.064	.050	.065	.065	.068	.054	.067
.6000	.154	.159	.132	.070	.074	.101	.089	.097	.090	.096
.7000	.177	.187	.154	.128	.145	.166	.159	.168	.162	.154
.8000	.231	.235	.219	.223	.234	.279	.283	.304	.278	.255
.9200	.678	.664	.652	.662	.712	.801	.861	.920	.844	.736
Flap										
.0125	.476	.303	.182	.091	.065	.101	.112	.124	.102	.084
.0250	.154	.064	.019	.021	.015	.030	.024	.024	.015	.032
.0500	.084	.064	.028	.049	.050	.065	.053	.053	.051	.072
.0750	.090	.086	.053	.067	.080	.083	.080	.077	.072	.093
.1000	.106	.107	.085	.104	.101	.119	.106	.106	.099	.122
.1500	.132	.144	.138	.143	.142	.148	.142	.148	.135	.148
.4000	.367	.382	.364	.369	.341	.356	.330	.325	.323	.333
.6000	.225	.309	.351	.375	.389	.412	.419	.416	.413	.406
.8000	.666	.651	.646	.628	.599	.584	.543	.531	.530	.533
.9000	.797	.749	.740	.732	.709	.679	.649	.634	.644	.655
Spoiler										
.2573	1.464	1.599	1.583	1.611	1.613	1.679	1.714	1.746	1.748	1.825
.4984	1.574	1.629	1.601	1.632	1.634	1.691	1.734	1.766	1.786	1.863
.7485	1.608	1.648	1.619	1.653	1.655	1.708	1.737	1.787	1.795	1.877
.9940	1.687	1.690	1.649	1.683	1.696	1.746	1.766	1.804	1.821	1.892

TABLE XI.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(c) Double slotted flap configuration with spoiler; $h_s = -0.035\bar{c}$; $\frac{y}{b/2} = 0.30$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0000	.873	1.399	2.250	2.863	3.520	4.240	4.317	3.960	2.900	2.368
.0125	1.016	1.594	2.348	2.890	3.465	3.934	3.991	3.874	2.897	2.371
.0250	1.038	1.498	2.475	2.991	3.592	4.084	4.011	3.883	2.900	2.374
.0500	1.061	1.325	2.083	3.894	3.928	3.976	4.005	3.895	2.891	2.374
.0750	1.077	1.260	1.901	3.897	5.213	5.168	4.422	3.951	2.888	2.368
.1000	1.086	1.238	1.271	2.406	4.003	4.844	4.343	3.842	2.882	2.377
.1500	1.108	1.248	1.290	1.421	2.132	3.359	3.757	3.496	2.826	2.354
.2000	1.127	1.260	1.330	1.348	1.610	2.455	3.104	3.121	2.746	2.325
.2500	1.153	1.280	1.349	1.372	1.468	1.988	2.621	2.813	2.634	2.290
.3000	1.172	1.285	1.364	1.378	1.435	1.814	2.358	2.561	2.542	2.244
.3500	1.198	1.285	1.361	1.375	1.420	1.671	2.081	2.317	2.385	2.200
.4000	1.233	1.307	1.392	1.403	1.438	1.626	1.968	2.185	2.290	2.151
.4500	1.268	1.341	1.416	1.412	1.456	1.620	1.906	2.094	2.210	2.113
.5000	1.306	1.368	1.441	1.427	1.468	1.605	1.819	1.956	2.074	2.058
.5500	1.344	1.387	1.438	1.436	1.477	1.593	1.755	1.895	2.006	2.024
.6000	1.389	1.446	1.481	1.470	1.508	1.599	1.737	1.839	1.953	1.983
.6500	1.471	1.511	1.555	1.534	1.556	1.620	1.723	1.807	1.894	1.945
.7521	1.659	1.737	1.750	1.741	1.751	1.772	1.799	1.889	1.843	1.905
.7934	2.038	2.015	2.018	1.997	2.009	2.048	2.078	2.091	1.935	1.937
.8099	2.249	2.214	2.197	2.189	2.195	2.248	2.256	2.258	2.015	1.963
.8141	2.344	2.297	2.277	2.265	2.270	2.329	2.341	2.323	2.057	1.977
.8102	2.395	2.353	2.336	2.314	2.321	2.395	2.384	2.376	2.071	1.997
.8274	2.538	2.505	2.469	2.457	2.459	2.512	2.521	2.511	2.160	2.029
Vane										
.0000	4.641	4.390	4.296	4.241	4.237	4.383	4.285	4.218	3.628	3.351
.0250	5.988	5.616	5.567	5.525	5.489	5.739	5.734	5.438	4.190	3.461
.0500	6.010	5.607	5.515	5.510	5.402	5.689	5.696	5.391	4.137	3.453
.1000	6.020	5.598	5.471	5.458	5.312	5.638	5.687	5.353	3.968	3.070
.1500	5.956	5.536	5.407	5.385	5.219	5.590	5.632	5.312	3.935	3.050
.2000	5.870	5.489	5.330	5.305	5.153	5.545	5.629	5.367	4.015	3.105
.3000	5.341	5.019	4.827	4.793	4.658	5.135	5.230	5.062	3.900	3.038
.4000	4.669	4.381	4.175	4.125	4.003	4.533	4.632	4.514	3.554	2.841
.5000	4.163	3.827	3.641	3.592	3.492	4.075	4.157	4.089	3.314	2.708
.6000	3.561	3.269	3.083	3.009	2.988	3.545	3.623	3.602	3.024	2.554
.7000	3.045	2.780	2.620	2.570	2.577	3.078	3.157	3.179	2.767	2.397
.8000	2.567	2.368	2.296	2.229	2.252	2.638	2.711	2.751	2.486	2.226
.9000	2.255	2.130	2.089	2.046	2.018	2.317	2.384	2.449	2.267	2.096
Flap										
.0000	1.452	1.449	1.478	1.351	1.249	1.386	1.452	1.496	1.332	1.293
.0125	2.083	2.081	2.046	1.979	1.928	2.105	2.201	2.270	2.080	1.992
.0250	2.538	2.517	2.435	2.384	2.342	2.635	2.728	2.830	2.619	2.516
.0500	2.494	2.505	2.416	2.363	2.384	2.698	2.825	2.980	2.793	2.655
.0750	2.427	2.427	2.336	2.284	2.336	2.674	2.810	2.971	2.805	2.644
.1000	2.344	2.279	2.194	2.143	2.225	2.542	2.670	2.813	2.663	2.499
.1500	1.879	1.895	1.870	1.863	1.967	2.204	2.297	2.440	2.332	2.244
.2000	1.729	1.663	1.648	1.637	1.718	1.886	1.968	2.036	1.977	1.919
.4000	1.494	1.548	1.558	1.595	1.658	1.686	1.711	1.766	1.687	1.728
.6000	1.663	1.681	1.673	1.683	1.700	1.754	1.732	1.783	1.767	1.815
.8000	1.704	1.721	1.713	1.720	1.742	1.763	1.737	1.798	1.778	1.823
Spoiler										
.2475	1.411	1.533	1.517	1.554	1.574	1.627	1.664	1.694	1.659	1.729
.4009	1.477	1.572	1.550	1.590	1.613	1.679	1.726	1.755	1.728	1.790
.7396	1.445	1.539	1.517	1.560	1.571	1.632	1.673	1.711	1.688	1.755
.9796	1.586	1.732	1.692	1.695	1.711	1.764	1.813	1.860	1.864	1.924

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0125	1.000	.774	.741	.814	.970	1.168	1.286	1.308	1.178	1.070
.0250	1.022	.817	.719	.701	.733	.787	.840	.854	.802	.745
.0500	1.010	.858	.738	.671	.619	.584	.571	.575	.547	.519
.0750	1.013	.879	.762	.674	.604	.551	.490	.493	.468	.452
.1000	1.013	.885	.781	.686	.607	.533	.472	.458	.435	.423
.1500	1.013	.910	.812	.720	.634	.548	.469	.458	.429	.423
.2000	1.013	.910	.830	.735	.649	.566	.484	.458	.435	.417
.2500	1.013	.929	.846	.753	.673	.587	.496	.475	.450	.435
.3000	1.006	.929	.846	.762	.682	.596	.525	.499	.459	.452
.3500	.997	.923	.852	.777	.703	.623	.548	.516	.473	.470
.4000	.959	.904	.839	.771	.703	.623	.551	.531	.497	.487
.4500	.920	.879	.827	.759	.697	.635	.560	.543	.503	.493
.5000	.860	.836	.787	.738	.679	.617	.548	.537	.503	.490
.5500	.812	.786	.744	.689	.646	.593	.536	.519	.488	.481
.6000	.736	.706	.663	.634	.589	.539	.493	.481	.450	.444
.6500	.675	.622	.583	.558	.526	.482	.455	.446	.417	.415
.7521	.503	.440	.398	.381	.360	.338	.326	.305	.287	.287
.7934	.446	.393	.349	.326	.312	.284	.257	.264	.240	.235
.8017	.433	.390	.361	.329	.300	.284	.268	.276	.243	.244
.8099	.427	.378	.346	.335	.324	.305	.297	.299	.272	.267
Vane										
.0250	1.752	1.678	1.639	1.515	1.414	1.365	1.274	1.229	1.092	1.032
.0500	.771	.768	.768	.643	.526	.428	.376	.352	.320	.301
.1000	.290	.328	.336	.223	.138	.060	.035	.044	.021	.020
.1500	.150	.195	.219	.128	.048	.006	.000	.009	.000	.000
.2000	.083	.124	.142	.064	.018	.000	.000	.000	.000	.000
.3000	.045	.059	.068	.024	.015	.015	.020	.018	.003	.000
.4000	.035	.031	.037	.024	.015	.024	.029	.023	.012	.012
.5000	.045	.043	.034	.034	.057	.051	.052	.053	.030	.035
.6000	.076	.065	.052	.067	.072	.072	.082	.079	.065	.061
.7000	.127	.115	.102	.110	.120	.129	.140	.132	.115	.104
.8000	.248	.248	.213	.223	.231	.260	.277	.270	.234	.226
.9200	.787	.731	.710	.701	.706	.775	.807	.810	.740	.702
Flap										
.0125	.210	.211	.219	.180	.156	.159	.189	.194	.154	.154
.0250	.035	.025	.009	.021	.012	.012	.026	.018	.000	.006
.0500	.006	.009	.000	.015	.009	.000	.000	.003	.000	.000
.0750	.016	.012	.006	.024	.030	.015	.012	.012	.000	.000
.1000	.035	.043	.019	.037	.027	.033	.035	.018	.009	.012
.1500	.057	.059	.062	.055	.060	.060	.052	.038	.018	.035
.2000	.102	.111	.089	.095	.090	.084	.085	.059	.047	.049
.4000	.248	.211	.219	.201	.210	.177	.163	.153	.136	.139
.6000	.389	.381	.370	.345	.333	.311	.292	.276	.254	.244
.8000	.567	.567	.531	.527	.520	.485	.440	.437	.426	.415
.9000	.742	.762	.728	.710	.694	.674	.624	.636	.627	.603
Spoiler										
.2587	1.477	1.599	1.565	1.587	1.604	1.679	1.723	1.749	1.757	1.813
.5024	1.530	1.617	1.586	1.614	1.619	1.691	1.726	1.764	1.757	1.822
.7531	1.605	1.654	1.616	1.638	1.646	1.702	1.732	1.764	1.772	1.834
.9940	1.662	1.720	1.698	1.725	1.741	1.743	1.781	1.819	1.815	1.877

TABLE XI.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(d) Double slotted flap configuration with spoiler; $h_s = -0.035\bar{c}$; $\frac{y}{b/2} = 0.43$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	.871	1.768	2.091	2.506	2.896	2.881	2.685	2.658	2.518	2.145
.0125	1.106	1.706	2.057	2.482	2.857	2.878	2.805	2.744	2.542	2.177
.0250	1.154	1.660	2.110	2.537	2.869	2.928	2.862	2.752	2.536	2.148
.0500	1.087	1.560	2.129	2.616	2.982	3.009	2.856	2.805	2.548	2.195
.0750	1.122	1.544	2.405	2.686	2.946	2.881	2.853	2.811	2.536	2.168
.1000	1.164	1.446	2.684	2.933	2.899	2.836	2.867	2.823	2.536	2.163
.1500	1.161	1.269	2.182	3.641	3.563	3.074	2.968	2.835	2.548	2.183
.2000	1.209	1.278	1.517	2.869	3.857	3.341	2.968	2.782	2.527	2.154
.2500	1.220	1.284	1.257	1.927	3.095	3.184	2.838	2.687	2.497	2.151
.3000	1.267	1.327	1.307	1.534	2.492	2.970	2.738	2.608	2.464	2.125
.3500	1.299	1.339	1.336	1.360	2.015	2.644	2.546	2.478	2.389	2.110
.4000	1.305	1.358	1.389	1.335	1.792	2.418	2.428	2.404	2.344	2.108
.4500	1.386	1.416	1.433	1.363	1.673	2.258	2.322	2.333	2.278	2.079
.5000	1.382	1.416	1.464	1.393	1.584	2.038	2.159	2.207	2.204	2.070
.5500	1.472	1.505	1.539	1.451	1.570	1.955	2.083	2.151	2.147	2.044
.6000	1.569	1.572	1.618	1.537	1.584	1.890	2.003	2.068	2.090	2.015
.7000	1.611	1.777	1.872	1.826	1.739	1.840	1.932	1.991	2.015	1.986
.7500	2.032	2.183	2.273	2.220	2.080	2.050	2.059	2.062	2.039	1.980
.7 9	2.279	2.480	2.599	2.555	2.388	2.255	2.224	2.186	2.084	2.003
.7750	2.350	2.556	2.684	2.659	2.477	2.341	2.272	2.224	2.093	2.000
.7800	2.472	2.682	2.812	2.778	2.587	2.418	2.339	2.277	2.114	2.009
Vane										
.0000	4.032	3.899	3.978	3.848	3.605	3.302	3.086	2.932	2.638	2.476
.0250	6.411	6.186	6.392	6.351	5.943	5.210	4.770	4.381	3.470	3.018
.0500	6.642	6.336	6.549	6.513	6.133	5.332	4.847	4.398	3.338	2.789
.1000	6.851	6.480	6.706	6.702	6.320	5.400	4.809	4.278	3.066	2.484
.1500	7.028	6.636	6.881	6.872	6.524	5.548	4.909	4.328	3.015	2.380
.2000	7.218	6.853	7.132	7.165	6.818	5.839	5.186	4.552	3.156	2.502
.3000	6.543	6.229	6.446	6.488	6.240	5.355	4.752	4.174	2.919	2.322
.4000	5.790	5.520	5.696	5.763	5.599	4.875	4.354	3.853	2.751	2.247
.5000	5.131	4.853	5.000	5.061	4.961	4.373	3.935	3.519	2.605	2.168
.6000	4.388	4.180	4.292	4.372	4.293	3.863	3.528	3.201	2.476	2.128
.7000	3.803	3.639	3.728	3.799	3.744	3.445	3.204	2.947	2.374	2.081
.8000	3.247	3.092	3.163	3.220	3.184	2.991	2.847	2.664	2.231	2.015
.9000	2.800	2.676	2.765	2.787	2.747	2.599	2.496	2.369	2.069	1.931
Flap										
.0000	1.910	2.095	2.317	2.360	2.228	2.059	1.959	1.856	1.644	1.542
.0125	2.347	2.553	2.718	2.753	2.673	2.504	2.404	2.286	2.069	1.957
.0250	2.855	3.104	3.270	3.323	3.261	3.118	3.006	2.862	2.614	2.476
.0500	2.852	3.064	3.201	3.278	3.258	3.175	3.098	2.971	2.692	2.508
.0750	2.694	2.936	3.057	3.144	3.139	3.098	3.062	2.953	2.656	2.455
.1000	2.723	2.819	2.890	2.970	2.964	2.943	2.921	2.832	2.554	2.351
.1500	2.070	2.113	2.151	2.201	2.255	2.380	2.431	2.404	2.231	2.073
.2000	1.910	1.801	1.843	1.884	1.920	1.955	2.000	1.997	1.916	1.829
.4000	1.151	1.211	1.251	1.293	1.335	1.427	1.505	1.578	1.710	1.751
.6000	1.630	1.703	1.721	1.710	1.697	1.739	1.738	1.758	1.802	1.815
.8000	1.492	1.685	1.746	1.726	1.709	1.739	1.752	1.770	1.817	1.815
Spoiler										
.2443	1.125	1.295	1.296	1.353	1.423	1.481	1.557	1.612	1.694	1.834
.4906	1.210	1.352	1.369	1.428	1.491	1.548	1.629	1.667	1.708	1.799
.7370	1.292	1.407	1.393	1.458	1.512	1.548	1.612	1.644	1.682	1.778
.9856	1.618	1.789	1.749	1.760	1.747	1.761	1.801	1.825	1.826	1.883

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.010	.774	.799	.936	1.130	1.258	1.316	1.378	1.386	1.249
.0250	1.022	.813	.762	.774	.834	.875	.906	.941	.961	.890
.0500	1.029	.844	.765	.707	.697	.682	.684	.693	.704	.678
.0750	1.035	.862	.777	.720	.656	.632	.602	.611	.611	.594
.1000	1.035	.875	.796	.720	.650	.599	.561	.563	.569	.545
.1500	1.016	.884	.806	.726	.650	.593	.534	.522	.524	.502
.2000	1.003	.884	.806	.735	.659	.593	.531	.519	.530	.502
.2500	.987	.887	.812	.741	.662	.593	.531	.522	.518	.496
.3000	.965	.865	.796	.732	.656	.590	.531	.516	.509	.490
.3500	.930	.847	.790	.729	.659	.593	.534	.516	.521	.499
.4000	.894	.807	.771	.701	.644	.587	.534	.516	.518	.496
.4500	.842	.771	.724	.680	.620	.576	.525	.502	.509	.478
.5000	.791	.725	.674	.625	.584	.540	.496	.487	.485	.452
.5500	.694	.633	.614	.561	.534	.490	.431	.422	.449	.391
.6000	.649	.578	.549	.509	.493	.445	.407	.410	.407	.388
.7000	.447	.413	.392	.372	.353	.320	.298	.280	.296	.278
.7500	.376	.358	.351	.329	.306	.270	.257	.242	.254	.232
.7600	.367	.339	.335	.323	.312	.270	.248	.242	.251	.226
.7700	.395	.370	.348	.332	.306	.276	.277	.280	.263	.232
Vane										
.0250	1.026	1.003	1.069	.970	.852	.736	.646	.605	.590	.542
.0500	.469	.474	.564	.476	.401	.306	.245	.224	.228	.194
.1000	.199	.223	.310	.268	.187	.113	.086	.062	.078	.055
.1500	.090	.086	.179	.128	.086	.033	.047	.032	.054	.009
.2000	.042	.061	.103	.088	.033	.039	.050	.027	.024	.009
.3000	.026	.028	.056	.064	.000	.036	.035	.018	.042	.038
.4000	.058	.055	.053	.043	.027	.036	.047	.035	.066	.052
.5000	.064	.070	.041	.073	.077	.071	.068	.074	.081	.061
.6000	.100	.095	.072	.088	.107	.104	.103	.094	.099	.078
.7000	.167	.144	.138	.149	.169	.169	.148	.130	.129	.116
.8000	.315	.303	.266	.308	.317	.288	.283	.251	.249	.226
.9200	1.003	.945	.950	.976	.961	.902	.844	.805	.746	.710
Flap										
.0125	.215	.229	.285	.311	.285	.258	.242	.215	.198	.165
.0250	.035	.021	.019	.040	.036	.036	.030	.027	.054	.055
.0500	.039	.012	.016	.021	.018	.009	.009	.000	.024	.038
.0750	.061	.052	.022	.015	.045	.021	.030	.015	.045	.038
.1000	.051	.061	.028	.018	.047	.045	.035	.015	.039	.046
.1500	.051	.083	.056	.076	.039	.059	.041	.068	.087	.046
.2000	.084	.095	.066	.076	.089	.095	.077	.089	.081	.067
.4000	.225	.217	.191	.204	.187	.181	.165	.156	.159	.157
.6000	.360	.358	.329	.329	.312	.285	.277	.268	.272	.258
.8000	.617	.566	.536	.531	.510	.481	.466	.445	.473	.461
.9000	.752	.755	.737	.732	.703	.682	.661	.628	.677	.664
Spoiler										
.2587	1.499	1.678	1.652	1.647	1.655	1.676	1.708	1.720	1.751	1.816
.5039	1.615	1.705	1.665	1.677	1.675	1.697	1.717	1.749	1.757	1.831
.7508	1.602	1.741	1.701	1.695	1.696	1.691	1.729	1.755	1.760	1.836
.9940	1.834	1.873	1.804	1.778	1.747	1.758	1.813	1.839	1.838	1.898

TABLE XI.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(e) Double slotted flap configuration with spoiler; $h_s = -0.035\bar{c}$; $\frac{y}{b/2} = 0.55$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.252	1.895	2.095	2.357	2.486	2.356	2.186	2.214	2.181	2.087
.0125	1.328	1.904	2.166	2.412	2.514	2.386	2.259	2.267	2.196	2.093
.0250	1.229	1.901	2.129	2.378	2.471	2.386	2.247	2.270	2.193	2.105
.0500	1.207	1.929	2.200	2.439	2.505	2.476	2.294	2.302	2.193	2.096
.0750	1.210	1.932	2.265	2.503	2.532	2.452	2.309	2.311	2.190	2.093
.1000	1.220	1.867	2.327	2.552	2.568	2.428	2.338	2.305	2.190	2.081
.1500	1.249	1.678	2.722	2.604	2.495	2.416	2.367	2.305	2.196	2.084
.2000	1.255	1.468	2.947	2.884	2.441	2.416	2.358	2.285	2.196	2.081
.2500	1.303	1.399	2.830	3.476	2.622	2.455	2.373	2.299	2.196	2.084
.3000	1.325	1.409	2.308	3.488	2.913	2.479	2.349	2.270	2.190	2.079
.3500	1.376	1.437	1.802	3.034	2.946	2.491	2.309	2.244	2.181	2.076
.4000	1.414	1.480	1.463	2.427	2.790	2.428	2.256	2.220	2.151	2.050
.4500	1.452	1.517	1.379	1.964	2.595	2.359	2.210	2.185	2.119	2.047
.5000	1.542	1.588	1.444	1.710	2.429	2.284	2.151	2.159	2.089	2.029
.6201	1.643	1.808	1.725	1.622	1.982	2.069	2.003	2.059	2.012	2.000
.6834	2.083	2.248	2.194	2.012	2.114	2.099	2.035	2.085	2.006	1.995
.7087	2.389	2.557	2.518	2.293	2.240	2.138	2.067	2.094	2.015	2.009
.7151	2.411	2.594	2.543	2.336	2.252	2.138	2.064	2.112	2.015	2.006
.7214	2.456	2.740	2.728	2.518	2.351	2.210	2.099	2.135	2.030	2.009
.7278	2.586	2.842	2.842	2.613	2.405	2.219	2.110	2.144	2.036	2.015
Vane										
.0000	3.322	3.282	3.154	2.793	2.471	2.284	2.198	2.182	2.065	2.035
.0250	6.185	6.211	6.237	5.604	4.544	3.853	3.594	3.511	3.083	2.960
.0500	6.338	6.303	6.400	5.753	4.601	3.841	3.536	3.420	2.929	2.792
.1000	6.752	6.694	6.873	6.226	4.781	3.781	3.399	3.250	2.693	2.569
.1500	6.800	6.709	6.940	6.308	4.712	3.692	3.329	3.144	2.565	2.464
.2000	6.797	6.700	6.968	6.379	4.733	3.689	3.326	3.127	2.512	2.412
.3000	6.440	6.362	6.623	6.189	4.592	3.542	3.154	2.986	2.385	2.308
.4000	5.679	5.613	5.870	5.567	4.147	3.213	2.886	2.760	2.258	2.180
.5000	5.128	5.059	5.333	5.104	3.850	3.015	2.734	2.625	2.199	2.134
.6000	4.408	4.353	4.573	4.421	3.396	2.754	2.545	2.467	2.130	2.081
.7000	3.781	3.752	3.922	3.845	3.054	2.542	2.373	2.329	2.065	2.032
.8000	3.287	3.257	3.388	3.339	2.772	2.368	2.236	2.211	2.009	1.997
.9000	2.835	2.799	2.910	2.857	2.465	2.180	2.073	2.059	1.932	1.942
Flap										
.0000	1.736	1.799	1.960	1.957	1.739	1.518	1.388	1.405	1.320	1.331
.0125	2.615	2.746	2.873	2.857	2.574	2.251	2.110	2.129	2.027	2.044
.0250	3.045	3.226	3.358	3.354	3.042	2.680	2.498	2.528	2.400	2.435
.0500	3.058	3.269	3.379	3.415	3.144	2.790	2.597	2.628	2.474	2.441
.0750	2.981	3.207	3.321	3.384	3.162	2.829	2.618	2.622	2.438	2.389
.1000	2.886	3.028	3.123	3.195	3.024	2.716	2.533	2.540	2.341	2.296
.1500	2.169	2.220	2.247	2.412	2.447	2.284	2.163	2.188	2.054	2.009
.2000	1.936	2.012	2.055	2.125	2.120	2.012	1.912	1.942	1.855	1.841
.4000	1.319	1.362	1.358	1.433	1.550	1.617	1.589	1.654	1.681	1.725
.6000	1.831	1.873	1.830	1.778	1.757	1.743	1.694	1.739	1.749	1.783
.8000	1.809	1.895	1.864	1.790	1.757	1.751	1.711	1.772	1.772	1.792
Spoiler										
.2443	1.141	1.280	1.266	1.368	1.524	1.577	1.638	1.673	1.688	1.784
.4881	1.273	1.410	1.384	1.479	1.622	1.650	1.676	1.708	1.717	1.799
.7416	1.361	1.533	1.471	1.551	1.628	1.629	1.656	1.685	1.699	1.784
.9856	1.759	1.937	1.888	1.820	1.768	1.737	1.769	1.784	1.798	1.866

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	.920	.824	.944	1.159	1.345	1.455	1.510	1.566	1.592	1.580
.0250	.965	.824	.821	.869	.931	1.009	1.038	1.079	1.101	1.107
.0500	.981	.842	.772	.744	.763	.769	.770	.798	.808	.803
.0750	.956	.830	.759	.698	.670	.674	.659	.660	.681	.661
.1000	.952	.842	.747	.695	.649	.629	.612	.613	.621	.606
.1500	.943	.842	.759	.680	.634	.593	.560	.563	.556	.557
.2000	.908	.830	.747	.677	.625	.569	.542	.537	.527	.528
.2500	.882	.817	.747	.671	.625	.572	.528	.525	.515	.510
.3000	.841	.793	.728	.662	.619	.566	.519	.513	.509	.499
.3500	.825	.783	.713	.640	.604	.563	.516	.499	.500	.481
.4000	.780	.731	.701	.628	.571	.542	.501	.484	.485	.470
.4500	.720	.700	.654	.588	.538	.518	.475	.472	.462	.444
.5000	.656	.635	.596	.549	.520	.464	.437	.440	.435	.415
.6201	.455	.458	.451	.412	.384	.347	.326	.323	.311	.299
.6834	.395	.390	.376	.335	.318	.302	.265	.267	.260	.249
.6961	.385	.378	.380	.329	.315	.281	.257	.258	.263	.249
.7087	.382	.378	.370	.326	.306	.275	.268	.264	.252	.246
Vane										
.0250	1.319	1.180	1.182	1.018	.883	.799	.723	.701	.675	.652
.0500	.659	.533	.605	.503	.420	.365	.306	.296	.266	.261
.1000	.325	.235	.318	.268	.207	.150	.125	.111	.109	.099
.1500	.188	.111	.201	.159	.129	.087	.099	.076	.065	.067
.2000	.099	.077	.142	.116	.084	.069	.064	.076	.053	.058
.3000	.041	.068	.093	.079	.069	.063	.061	.076	.059	.064
.4000	.067	.068	.080	.073	.084	.087	.096	.109	.083	.084
.5000	.038	.099	.105	.088	.096	.099	.105	.103	.104	.104
.6000	.089	.146	.130	.128	.138	.123	.157	.141	.121	.125
.7000	.166	.204	.207	.183	.180	.183	.169	.173	.157	.168
.8000	.306	.350	.358	.351	.315	.296	.268	.276	.263	.255
.9200	1.032	1.053	1.086	1.049	.916	.832	.802	.795	.781	.791
Flap										
.0125	.255	.260	.346	.332	.291	.249	.251	.243	.234	.229
.0250	.048	.074	.074	.079	.069	.069	.073	.079	.053	.058
.0500	.064	.056	.065	.058	.075	.072	.055	.067	.047	.046
.0750	.064	.074	.065	.091	.081	.069	.070	.076	.056	.046
.1000	.076	.087	.086	.085	.081	.075	.087	.079	.071	.064
.1500	.073	.121	.123	.088	.093	.117	.108	.103	.077	.104
.2000	.102	.142	.142	.128	.138	.135	.122	.114	.112	.110
.4000	.264	.288	.268	.250	.231	.210	.204	.217	.198	.191
.6000	.430	.421	.407	.363	.351	.338	.324	.314	.308	.296
.8000	.618	.638	.602	.561	.544	.521	.498	.487	.485	.481
.9000	.806	.830	.796	.759	.745	.716	.668	.680	.666	.673
Spoiler										
.2584	1.655	1.807	1.770	1.737	1.699	1.679	1.702	1.711	1.728	1.807
.5039	1.643	1.828	1.791	1.748	1.705	1.685	1.717	1.729	1.740	1.819
.7508	1.668	1.867	1.828	1.760	1.702	1.697	1.717	1.732	1.746	1.822
.9940	1.850	2.090	2.094	1.934	1.777	1.743	1.764	1.793	1.792	1.866

TABLE XI.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Concluded

(f) Double slotted flap configuration with spoiler; $h_s = -0.035\bar{c}$; $\frac{y}{b/2} = 0.72$

Upper surface

	C_p for -									
x/c	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0000	2.141	2.196	2.378	2.246	2.122	1.982	1.930	1.930	1.879	1.953
.0125	2.063	2.184	2.305	2.192	2.077	1.953	1.936	1.947	1.893	1.953
.0250	2.057	2.199	2.323	2.204	2.083	1.968	1.947	1.959	1.899	1.965
.0500	2.025	2.247	2.365	2.216	2.092	2.011	1.971	1.971	1.902	1.968
.0750	1.643	2.301	2.417	2.240	2.113	2.026	1.979	1.973	1.905	1.959
.1000	1.379	2.349	2.414	2.225	2.092	2.000	1.973	1.976	1.922	1.956
.1500	1.389	2.518	2.483	2.263	2.107	2.011	2.000	2.003	1.910	1.968
.2000	1.439	2.663	2.565	2.248	2.101	2.043	2.017	2.006	1.913	1.971
.2500	1.464	2.620	2.668	2.231	2.092	2.046	2.017	2.006	1.913	1.976
.3000	1.486	2.428	2.852	2.234	2.107	2.038	2.011	1.994	1.922	1.979
.3500	1.514	2.154	3.051	2.216	2.125	2.035	2.006	1.994	1.910	1.982
.4000	1.571	1.886	3.145	2.272	2.125	2.032	1.994	1.988	1.928	1.991
.4500	1.599	1.687	3.157	2.347	2.119	2.023	1.988	1.991	1.928	2.000
.5000	1.662	1.626	3.130	2.443	2.137	2.017	1.985	1.991	1.936	2.003
.5500	1.690	1.575	2.927	2.389	2.119	2.006	1.973	1.979	1.936	2.006
.6000	1.693	1.572	2.680	2.332	2.101	1.988	1.965	1.968	1.936	2.008
.6500	1.665	1.536	2.405	2.207	2.074	1.971	1.962	1.973	1.954	2.029
.7000	1.602	1.467	2.184	2.150	2.053	1.962	1.947	1.968	1.954	2.032
.7500	1.530	1.413	2.000	2.108	2.027	1.947	1.936	1.959	1.936	2.038
.8000	1.442	1.337	1.885	2.066	2.006	1.941	1.938	1.968	1.954	2.043
.8500	1.383	1.292	1.758	1.964	1.961	1.912	1.918	1.956	1.954	2.041
.9000	1.339	1.277	1.683	1.916	1.925	1.901	1.921	1.953	1.962	2.038

Lower surface

	C_p for -									
x/c	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0125	.749	.858	1.021	1.114	1.193	1.248	1.318	1.364	1.387	1.478
.0250	.777	.786	.861	.925	.970	1.035	1.084	1.134	1.144	1.230
.0500	.790	.750	.761	.784	.798	.837	.872	.901	.919	.977
.0750	.784	.753	.740	.740	.738	.767	.781	.807	.818	.860
.1000	.784	.756	.728	.719	.711	.723	.735	.746	.757	.793
.1500	.796	.759	.728	.707	.676	.676	.673	.691	.694	.726
.2000	.818	.783	.746	.719	.682	.676	.682	.676	.682	.705
.2500	.850	.807	.773	.740	.696	.688	.676	.685	.676	.697
.3000	.878	.840	.807	.772	.726	.708	.705	.708	.694	.717
.3500	.941	.925	.906	.850	.798	.778	.770	.764	.760	.775
.4000	1.063	1.051	1.051	.976	.914	.880	.866	.872	.855	.872
.4500	1.248	1.247	1.245	1.135	1.042	.997	.991	.994	.959	.974
.5000	1.464	1.452	1.462	1.305	1.199	1.154	1.125	1.137	1.107	1.108
.5500	1.636	1.660	1.659	1.515	1.381	1.332	1.315	1.332	1.298	1.300
.6000	1.627	1.843	1.837	1.728	1.592	1.527	1.527	1.548	1.520	1.539
.6500	1.975	1.988	1.949	1.820	1.720	1.688	1.720	1.764	1.728	1.729
.7000	2.028	2.036	1.961	1.862	1.780	1.784	1.822	1.857	1.809	1.784
.7500	1.988	1.994	1.918	1.862	1.803	1.796	1.804	1.842	1.798	1.801
.8000	1.859	1.876	1.831	1.823	1.756	1.737	1.740	1.764	1.740	1.775
.8500	1.724	1.762	1.767	1.790	1.726	1.702	1.711	1.734	1.720	1.781
.9000	1.608	1.648	1.701	1.760	1.708	1.685	1.699	1.723	1.711	1.775
.9500	1.473	1.521	1.625	1.731	1.717	1.702	1.708	1.740	1.734	1.781

TABLE XII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE

(a) Double slotted flap configuration with spoiler; $h_s = -0.050\bar{c}$; $\frac{y}{b/2} = 0$

Upper surface

x/c	C_p for -									
	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Fuselage										
.0000	.000	.006	.000	.000	.015	.018	.041	.076	.116	.152
.0500	.703	.828	.890	.948	.988	1.009	1.029	1.062	1.078	1.092
.1000	.780	.910	.966	1.006	1.039	1.050	1.076	1.115	1.130	1.146
.1500	.907	.979	1.021	1.064	1.084	1.092	1.099	1.126	1.133	1.155
.2000	.941	1.009	1.055	1.097	1.0	.103	1.099	1.126	1.127	1.135
.2500	.985	1.051	1.092	1.104	1.114	1.112	1.117	1.132	1.130	1.155
.3000	1.040	1.072	1.089	1.113	1.114	1.097	1.099	1.112	1.118	1.132
.3500	1.043	1.075	1.086	1.097	1.102	1.092	1.088	1.100	1.107	1.137
.4000	1.015	1.039	1.055	1.070	1.075	1.062	1.088	1.109	1.116	1.146
.4500	1.025	1.045	1.067	1.097	1.120	1.115	1.158	1.205	1.225	1.281
.5000	1.025	1.057	1.107	1.164	1.198	1.230	1.287	1.326	1.381	1.590
.5500	1.056	1.099	1.171	1.231	1.270	1.298	1.357	1.399	1.514	1.960
.6000	1.077	1.142	1.214	1.277	1.294	1.313	1.354	1.376	1.520	1.980
.6500	1.130	1.181	1.251	1.289	1.306	1.304	1.325	1.299	1.399	1.851
.7000	1.180	1.229	1.281	1.310	1.327	1.316	1.301	1.305	1.367	1.745
.7500	1.248	1.277	1.315	1.325	1.342	1.345	1.339	1.343	1.410	1.662
.8000	1.310	1.328	1.346	1.356	1.366	1.384	1.398	1.420	1.459	1.573
.8500	1.285	1.304	1.321	1.344	1.363	1.375	1.430	1.478	1.506	1.513
.9000	1.201	1.223	1.232	1.259	1.273	1.295	1.342	1.399	1.436	1.435
.9500	1.130	1.157	1.165	1.186	1.210	1.233	1.263	1.335	1.384	1.381
.9940	1.176	1.193	1.187	1.198	1.216	1.242	1.284	1.343	1.396	1.398

Lower surface

x/c	C_p for -									
	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Fuselage										
.0500	.786	.810	.752	.708	.640	.587	.474	.455	.419	.393
.1000	.854	.898	.856	.796	.739	.661	.579	.546	.523	.496
.1500	.941	.952	.914	.885	.805	.729	.661	.622	.595	.564
.2000	.966	.997	.963	.921	.871	.785	.722	.689	.659	.616
.2500	.997	1.015	.991	.961	.910	.826	.772	.742	.711	.676
.3000	1.022	1.027	1.018	.991	.937	.870	.813	.774	.754	.719
.3500	.997	1.018	1.000	.985	.949	.882	.827	.801	.769	.736
.4000	.932	.955	.930	.921	.886	.820	.760	.733	.723	.685
.4500	.885	.913	.908	.879	.832	.767	.708	.672	.653	.622
.5000	.845	.876	.856	.809	.766	.696	.635	.607	.578	.544
.5500	.802	.837	.820	.769	.712	.631	.570	.534	.506	.484
.6000	.768	.768	.743	.702	.634	.566	.509	.478	.457	.441
.7500	.892	.898	.865	.848	.799	.752	.702	.680	.676	.662
.9500	1.006	1.057	1.092	1.140	1.153	1.130	1.140	1.161	1.202	1.235
.9940	.981	1.093	1.171	1.225	1.255	1.277	1.301	1.343	1.364	1.375

TABLE XII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(b) Double slotted flap configuration with spoiler; $h_s = -0.050\bar{c}$; $\frac{y}{b/2} = 0.21$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	.842	.926	2.053	3.412	5.273	7.553	9.895	8.449	3.891	2.683
.0125	.954	1.241	1.762	3.552	7.395	10.377	11.553	8.228	3.506	2.415
.0250	.984	1.188	1.585	2.683	3.637	6.111	8.367	7.422	3.515	2.450
.0500	1.009	1.170	1.458	1.619	1.982	2.658	4.041	5.177	3.089	2.344
.0750	1.018	1.163	1.396	1.543	1.774	2.139	2.806	3.829	2.947	2.344
.1000	1.043	1.170	1.375	1.525	1.705	1.962	2.528	3.277	2.838	2.320
.1500	1.055	1.182	1.341	1.464	1.580	1.782	2.157	2.584	2.651	2.276
.2000	1.083	1.197	1.337	1.445	1.536	1.713	1.992	2.283	2.471	2.215
.2500	1.105	1.213	1.337	1.424	1.482	1.649	1.884	2.074	2.311	2.147
.3000	1.126	1.228	1.341	1.409	1.452	1.601	1.783	1.929	2.166	2.091
.3500	1.139	1.241	1.337	1.387	1.426	1.565	1.667	1.808	2.033	2.056
.4000	1.176	1.256	1.334	1.403	1.417	1.550	1.609	1.741	1.950	2.026
.4500	1.204	1.284	1.353	1.406	1.426	1.535	1.574	1.679	1.876	2.009
.5000	1.247	1.318	1.390	1.415	1.437	1.517	1.554	1.646	1.820	1.973
.5500	1.275	1.345	1.393	1.421	1.431	1.471	1.519	1.587	1.749	1.947
.6000	1.312	1.370	1.421	1.439	1.434	1.492	1.531	1.623	1.704	1.920
.6500	1.355	1.416	1.455	1.464	1.467	1.507	1.574	1.640	1.672	1.888
.7000	1.423	1.469	1.495	1.500	1.515	1.535	1.615	1.664	1.672	1.859
.7815	1.605	1.636	1.632	1.640	1.652	1.704	1.676	1.693	1.716	1.850
.8179	1.784	1.790	1.768	1.784	1.789	1.897	1.841	1.844	1.829	1.891
.8325	1.904	1.886	1.861	1.866	1.887	1.994	1.948	1.959	1.935	1.929
.8362	1.941	1.916	1.885	1.909	1.922	2.039	2.000	2.018	1.980	1.950
.8398	1.901	1.876	1.836	1.866	1.884	1.991	1.948	1.977	1.947	1.935
.8434	2.015	1.994	1.929	1.951	1.973	2.115	2.084	2.109	2.054	1.985
Vane										
.0000	2.305	2.268	2.242	2.256	2.292	2.426	2.351	2.369	2.287	2.144
.0250	3.941	3.833	3.731	3.762	3.907	4.402	4.404	4.455	4.163	3.612
.0500	3.719	3.592	3.477	3.482	3.634	4.142	4.157	4.180	3.891	3.350
.1000	3.675	3.527	3.362	3.342	3.482	4.024	4.082	4.133	3.790	3.073
.1500	3.567	3.407	3.220	3.201	3.381	3.897	3.995	4.068	3.761	3.023
.2000	3.481	3.302	3.111	3.104	3.300	3.800	3.946	4.059	3.788	3.065
.3000	3.077	2.922	2.728	2.768	2.994	3.396	3.635	3.808	3.648	3.020
.4000	2.685	2.561	2.409	2.470	2.684	2.991	3.279	3.475	3.421	2.873
.5000	2.398	2.315	2.195	2.272	2.482	2.674	2.969	3.168	3.187	2.747
.6000	2.166	2.083	2.003	2.031	2.253	2.371	2.569	2.726	2.779	2.494
.7000	2.120	2.037	1.944	1.973	2.107	2.323	2.493	2.614	2.607	2.368
.8000	2.095	2.015	1.926	1.957	2.101	2.296	2.522	2.628	2.560	2.297
.9000	2.065	1.972	1.885	1.927	2.053	2.242	2.458	2.596	2.500	2.197
Flap										
.0000	1.605	1.639	1.536	1.348	1.241	1.308	1.357	1.460	1.429	1.323
.0125	2.163	2.160	2.034	1.945	2.000	2.193	2.386	2.499	2.341	2.047
.0250	2.339	2.305	2.217	2.214	2.324	2.556	2.783	2.915	2.752	2.423
.0500	2.413	2.370	2.294	2.317	2.446	2.692	2.911	3.053	2.918	2.667
.0750	2.253	2.206	2.133	2.143	2.265	2.468	2.647	2.779	2.672	2.517
.1000	2.034	1.997	1.935	1.957	2.047	2.214	2.368	2.481	2.418	2.341
.1500	1.741	1.734	1.694	1.686	1.732	1.876	1.989	2.100	2.062	2.006
.2000	1.691	1.666	1.610	1.613	1.661	1.804	1.934	2.047	2.012	1.944
.4000	1.713	1.707	1.672	1.671	1.684	1.813	1.939	2.092	2.039	1.912
.6000	1.682	1.645	1.628	1.653	1.622	1.725	1.884	1.717	1.746	1.812
.8000	1.768	1.719	1.700	1.726	1.696	1.758	1.728	1.779	1.832	1.888
Spoiler										
.2429	1.313	1.575	1.615	1.623	1.682	1.758	1.842	1.924	1.887	1.862
.4852	1.214	1.512	1.575	1.587	1.649	1.711	1.789	1.842	1.815	1.773
.7367	1.201	1.509	1.572	1.584	1.631	1.682	1.757	1.804	1.786	1.776
.9796	1.375	1.620	1.657	1.666	1.688	1.714	1.754	1.801	1.815	1.854

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.021	.809	.638	.582	.574	.625	.702	.673	.459	.379
.0250	.997	.830	.659	.555	.440	.390	.333	.310	.249	.206
.0500	.984	.867	.697	.579	.440	.338	.249	.221	.222	.209
.0750	.978	.873	.715	.604	.464	.369	.287	.260	.275	.279
.1000	.978	.867	.740	.631	.497	.411	.342	.327	.337	.335
.1500	.960	.886	.752	.652	.551	.489	.426	.413	.417	.412
.2000	.969	.886	.786	.695	.592	.538	.473	.472	.450	.450
.2500	.975	.901	.808	.732	.643	.580	.519	.507	.497	.482
.3000	.972	.904	.817	.744	.658	.607	.548	.540	.524	.506
.3500	.981	.929	.845	.781	.693	.656	.580	.566	.553	.535
.4000	.981	.932	.861	.796	.720	.683	.606	.590	.571	.565
.4500	.957	.923	.858	.802	.726	.692	.617	.608	.592	.582
.5000	.910	.889	.824	.777	.714	.671	.609	.596	.580	.579
.5500	.836	.821	.768	.729	.673	.637	.574	.555	.544	.559
.6000	.772	.778	.740	.692	.643	.607	.548	.543	.536	.535
.6500	.719	.704	.659	.622	.571	.553	.502	.507	.479	.488
.7000	.682	.614	.573	.549	.503	.483	.452	.457	.438	.444
.7815	.596	.512	.446	.415	.375	.359	.328	.333	.325	.335
.8179	.497	.432	.402	.381	.339	.326	.313	.310	.296	.306
.8252	.488	.435	.396	.372	.348	.344	.325	.325	.317	.321
.8325	.472	.438	.412	.396	.381	.387	.377	.384	.373	.374
Vane										
.0250	1.318	1.281	1.251	1.189	1.092	1.066	.989	.988	.953	.906
.0500	.815	.765	.703	.445	.461	.396	.351	.366	.340	.324
.1000	.518	.469	.368	.256	.113	.079	.067	.077	.056	.059
.1500	.420	.364	.266	.143	.030	.012	.012	.027	.012	.015
.2000	.349	.309	.217	.101	.012	.006	.000	.006	.000	.006
.3000	.281	.247	.170	.061	.012	.018	.026	.024	.015	.021
.4000	.235	.198	.133	.055	.024	.030	.035	.053	.041	.044
.5000	.207	.176	.127	.067	.045	.066	.067	.074	.065	.062
.6000	.188	.173	.124	.076	.071	.094	.104	.115	.098	.097
.7000	.201	.182	.149	.128	.131	.151	.159	.180	.157	.150
.8000	.268	.247	.217	.220	.235	.263	.281	.304	.275	.256
.9200	.697	.670	.641	.671	.705	.779	.847	.891	.834	.744
Flap										
.0125	.494	.299	.180	.088	.068	.076	.104	.115	.101	.076
.0250	.179	.068	.015	.018	.006	.009	.023	.021	.021	.024
.0500	.108	.065	.034	.043	.042	.048	.052	.065	.056	.062
.0750	.111	.077	.059	.070	.062	.079	.075	.083	.077	.079
.1000	.127	.105	.093	.116	.092	.103	.099	.112	.115	.112
.1500	.151	.148	.124	.137	.125	.148	.130	.162	.145	.144
.4000	.386	.380	.368	.372	.348	.356	.316	.333	.325	.326
.6000	.546	.531	.511	.512	.476	.471	.415	.431	.417	.426
.8000	.685	.654	.635	.646	.583	.586	.533	.543	.539	.538
.9000	.762	.765	.743	.735	.699	.698	.638	.631	.654	.650
Spoiler										
.2573	1.307	1.581	1.609	1.605	1.625	1.652	1.678	1.725	1.751	1.788
.4984	1.474	1.642	1.664	1.663	1.679	1.699	1.731	1.792	1.803	1.845
.7485	1.492	1.666	1.691	1.693	1.712	1.723	1.740	1.792	1.844	1.894
.9940	1.731	1.837	1.838	1.839	1.826	1.805	1.807	1.851	1.890	1.934

TABLE XII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(c) Double slotted flap configuration with spoiler; $h_s = -0.050\bar{c}$; $\frac{y}{b/2} = 0.30$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	.872	1.325	2.145	2.812	3.399	4.231	4.406	3.883	2.909	2.268
.0125	.988	1.528	2.265	2.852	3.369	3.934	4.054	3.798	2.897	2.277
.0250	1.031	1.406	2.372	2.943	3.483	4.066	4.086	3.813	2.909	2.277
.0500	1.044	1.300	2.935	3.852	3.871	3.979	4.045	3.854	2.888	2.282
.0750	1.060	1.263	1.825	3.812	5.105	5.180	4.492	3.894	2.891	2.280
.1000	1.063	1.231	1.252	2.338	3.853	4.835	4.424	3.787	2.897	2.288
.1500	1.097	1.244	1.277	1.401	2.093	3.347	3.790	3.460	2.805	2.268
.2000	1.116	1.259	1.326	1.344	1.574	2.422	3.104	3.102	2.714	2.253
.2500	1.141	1.263	1.338	1.362	1.435	1.988	2.654	2.784	2.617	2.236
.3000	1.166	1.288	1.354	1.371	1.417	1.796	2.373	2.562	2.522	2.212
.3500	1.188	1.294	1.357	1.371	1.390	1.665	2.101	2.291	2.387	2.178
.4000	1.216	1.319	1.363	1.389	1.408	1.623	1.985	2.157	2.301	2.160
.4500	1.260	1.350	1.394	1.404	1.429	1.605	1.917	2.075	2.204	2.137
.5000	1.295	1.381	1.425	1.423	1.444	1.581	1.820	1.944	2.080	2.090
.5500	1.326	1.400	1.428	1.429	1.447	1.572	1.775	1.877	2.015	2.055
.6000	1.373	1.456	1.480	1.471	1.477	1.593	1.746	1.831	1.962	2.029
.6500	1.452	1.525	1.548	1.529	1.532	1.617	1.716	1.801	1.900	1.991
.7521	1.662	1.738	1.739	1.736	1.697	1.748	1.835	1.848	1.876	1.906
.7934	2.025	2.041	2.009	2.019	1.988	2.051	2.113	2.061	1.953	1.944
.8099	2.251	2.238	2.203	2.195	2.165	2.237	2.284	2.224	2.027	1.968
.8141	2.348	2.331	2.289	2.283	2.252	2.338	2.379	2.291	2.062	1.982
.8182	2.401	2.388	2.339	2.329	2.324	2.383	2.426	2.332	2.092	1.997
.8224	2.555	2.534	2.502	2.478	2.429	2.503	2.568	2.454	2.174	2.023
Vane										
.0000	4.665	4.503	4.335	4.286	4.225	4.395	4.370	4.174	3.640	3.338
.0250	6.025	5.753	5.594	5.606	5.444	5.742	5.832	5.367	4.204	3.408
.0500	6.044	5.753	5.566	5.581	5.417	5.701	5.823	5.358	4.151	3.384
.1000	6.054	5.750	5.542	5.533	5.324	5.653	5.841	5.305	3.968	2.965
.1500	6.000	5.688	5.480	5.466	5.237	5.596	5.782	5.253	3.950	2.930
.2000	5.944	5.672	5.446	5.411	5.174	5.554	5.791	5.291	4.018	2.973
.3000	5.405	5.191	4.926	4.888	4.673	5.144	5.397	4.973	3.909	2.921
.4000	4.724	4.506	4.274	4.216	4.018	4.548	4.782	4.442	3.567	2.743
.5000	4.210	3.950	3.708	3.654	3.501	4.096	4.285	4.034	3.310	2.632
.6000	3.580	3.372	3.139	3.073	2.991	3.551	3.734	3.559	3.036	2.498
.7000	3.057	2.863	2.662	2.602	2.574	3.087	3.249	3.139	2.776	2.355
.8000	2.586	2.422	2.302	2.262	2.249	2.641	2.758	2.728	2.484	2.210
.9000	2.273	2.169	2.099	2.067	2.021	2.311	2.420	2.419	2.260	2.078
Flap										
.0000	1.461	1.456	1.474	1.353	1.237	1.359	1.477	1.458	1.360	1.271
.0125	2.094	2.116	2.074	1.991	1.901	2.096	2.249	2.204	2.097	1.973
.0250	2.549	2.563	2.483	2.411	2.324	2.623	2.799	2.758	2.643	2.475
.0500	2.511	2.544	2.452	2.389	2.354	2.686	2.891	2.889	2.811	2.618
.0750	2.436	2.456	2.363	2.307	2.303	2.656	2.870	2.874	2.814	2.597
.1000	2.339	2.306	2.215	2.174	2.189	2.518	2.713	2.720	2.664	2.454
.1500	1.834	1.869	1.849	1.845	1.925	2.144	2.317	2.344	2.339	2.195
.2000	1.727	1.656	1.622	1.642	1.703	1.847	1.968	1.979	1.962	1.909
.4000	1.357	1.428	1.455	1.520	1.580	1.554	1.633	1.603	1.605	1.664
.6000	1.671	1.672	1.649	1.645	1.622	1.695	1.731	1.732	1.785	1.813
.8000	1.737	1.750	1.708	1.705	1.682	1.737	1.767	1.746	1.811	1.828
Spoiler										
.2475	1.142	1.376	1.462	1.511	1.565	1.528	1.532	1.534	1.514	1.599
.4909	1.248	1.467	1.526	1.563	1.628	1.625	1.652	1.654	1.636	1.699
.7396	1.263	1.461	1.514	1.541	1.595	1.602	1.626	1.651	1.644	1.693
.9796	1.480	1.774	1.820	1.818	1.829	1.838	1.848	1.886	1.887	1.894

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.013	.788	.723	.800	.946	1.156	1.293	1.286	1.183	1.052
.0250	1.025	.828	.717	.684	.715	.778	.834	.840	.805	.740
.0500	1.013	.869	.732	.657	.607	.581	.574	.557	.555	.525
.0750	1.013	.884	.754	.660	.601	.539	.494	.475	.481	.458
.1000	1.016	.888	.763	.681	.595	.536	.473	.449	.451	.429
.1500	1.013	.919	.803	.699	.628	.542	.479	.452	.445	.429
.2000	1.016	.925	.812	.727	.637	.551	.485	.458	.445	.426
.2500	1.019	.934	.831	.742	.667	.578	.509	.481	.454	.443
.3000	1.000	.925	.837	.754	.679	.590	.524	.493	.469	.461
.3500	.988	.931	.840	.760	.688	.608	.539	.510	.493	.481
.4000	.956	.906	.828	.760	.694	.614	.553	.525	.504	.493
.4500	.922	.884	.809	.754	.688	.620	.562	.528	.516	.507
.5000	.865	.841	.785	.724	.667	.602	.547	.519	.510	.504
.5500	.812	.788	.732	.687	.637	.581	.539	.510	.496	.493
.6000	.740	.706	.658	.614	.577	.536	.491	.472	.475	.458
.6500	.665	.628	.578	.553	.517	.482	.447	.431	.428	.423
.7521	.508	.450	.391	.371	.348	.332	.311	.303	.298	.294
.7934	.451	.394	.348	.322	.303	.275	.263	.254	.257	.254
.8017	.436	.384	.348	.322	.300	.278	.269	.262	.257	.254
.8099	.433	.375	.342	.319	.306	.299	.302	.297	.292	.283
Vane										
.0250	1.765	1.709	1.634	1.526	1.417	1.368	1.290	1.224	1.106	1.035
.0500	.787	.769	.760	.635	.526	.422	.379	.359	.322	.306
.1000	.288	.309	.326	.219	.123	.066	.044	.044	.035	.041
.1500	.160	.175	.209	.112	.042	.006	.000	.000	.000	.000
.2000	.088	.109	.123	.055	.015	.000	.000	.003	.000	.000
.3000	.031	.044	.058	.018	.000	.006	.018	.017	.000	.015
.4000	.031	.022	.037	.012	.024	.024	.024	.038	.021	.026
.5000	.038	.025	.028	.024	.027	.036	.050	.052	.044	.044
.6000	.075	.047	.034	.040	.063	.060	.077	.076	.071	.067
.7000	.132	.106	.077	.088	.114	.129	.130	.134	.124	.120
.8000	.251	.225	.209	.213	.216	.251	.266	.265	.239	.236
.9200	.777	.744	.714	.705	.688	.769	.817	.790	.743	.708
Flap										
.0125	.194	.191	.203	.167	.153	.153	.186	.192	.165	.154
.0250	.019	.016	.006	.000	.006	.006	.018	.017	.018	.009
.0500	.016	.000	.000	.000	.003	.000	.000	.000	.000	.009
.0750	.025	.006	.000	.006	.018	.003	.006	.012	.000	.006
.1000	.034	.025	.003	.015	.018	.015	.021	.023	.006	.017
.1500	.069	.063	.034	.036	.054	.042	.030	.032	.032	.041
.2000	.110	.094	.080	.073	.069	.072	.065	.061	.056	.058
.4000	.245	.231	.209	.201	.174	.174	.166	.163	.153	.143
.6000	.395	.378	.348	.334	.315	.299	.287	.268	.268	.277
.8000	.580	.566	.535	.514	.505	.485	.456	.440	.434	.431
.9000	.756	.766	.735	.708	.691	.662	.642	.624	.634	.627
Spoiler										
.2587	1.337	1.578	1.618	1.608	1.637	1.673	1.713	1.748	1.748	1.771
.5024	1.409	1.623	1.660	1.663	1.685	1.705	1.731	1.766	1.774	1.799
.7531	1.498	1.684	1.712	1.721	1.730	1.726	1.740	1.774	1.774	1.802
.9940	1.591	1.780	1.816	1.818	1.829	1.808	1.801	1.848	1.838	1.857

TABLE XII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(d) Double slotted flap configuration with spoiler; $h_s = -0.050\bar{c}$; $\frac{y}{b/2} = 0.43$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	.805	1.771	2.130	2.540	2.848	2.888	2.632	2.593	2.459	2.182
.0125	1.074	1.703	2.111	2.522	2.821	2.894	2.742	2.685	2.486	2.200
.0250	1.089	1.666	2.164	2.555	2.821	2.936	2.803	2.690	2.480	2.185
.0500	1.096	1.568	2.223	2.662	2.943	3.003	2.789	2.735	2.494	2.220
.0750	1.102	1.546	2.492	2.711	2.908	2.900	2.792	2.761	2.480	2.197
.1000	1.117	1.432	2.861	2.900	2.833	2.852	2.824	2.749	2.494	2.194
.1500	1.129	1.262	2.498	3.711	3.470	3.060	2.905	2.749	2.491	2.203
.2000	1.170	1.287	1.666	2.994	3.774	3.341	2.899	2.705	2.477	2.191
.2500	1.191	1.290	1.263	2.000	3.074	3.187	2.780	2.617	2.438	2.173
.3000	1.213	1.330	1.291	1.564	2.512	2.970	2.676	2.552	2.400	2.168
.3500	1.241	1.333	1.319	1.357	2.039	2.649	2.499	2.440	2.341	2.144
.4000	1.271	1.370	1.365	1.339	1.798	2.441	2.395	2.366	2.284	2.126
.4500	1.315	1.407	1.418	1.351	1.678	2.272	2.284	2.295	2.234	2.097
.5000	1.358	1.423	1.443	1.375	1.545	2.048	2.131	2.177	2.166	2.065
.5500	1.416	1.494	1.514	1.442	1.553	1.970	2.052	2.121	2.116	2.041
.6000	1.497	1.571	1.585	1.509	1.559	1.885	1.983	2.044	2.062	2.018
.7000	1.771	1.815	1.836	1.793	1.720	1.849	1.876	1.962	1.977	1.962
.7500	2.160	2.194	2.220	2.183	2.042	2.057	1.997	2.038	1.994	1.965
.7700	2.481	2.512	2.523	2.518	2.348	2.269	2.157	2.139	2.048	1.979
.7750	2.568	2.586	2.622	2.607	2.428	2.341	2.197	2.159	2.059	1.973
.7800	2.691	2.719	2.743	2.732	2.542	2.417	2.255	2.207	2.068	1.991
Vane										
.0000	3.916	3.885	3.858	3.802	3.541	3.308	3.009	2.856	2.601	2.488
.0250	6.184	6.116	6.173	6.244	5.800	5.172	4.656	4.204	3.421	3.073
.0500	6.351	6.252	6.328	6.412	5.976	5.287	4.737	4.213	3.296	2.862
.1000	6.521	6.382	6.477	6.580	6.163	5.356	4.711	4.080	3.039	2.570
.1500	6.660	6.521	6.625	6.754	6.339	5.498	4.795	4.121	2.986	2.470
.2000	6.811	6.663	6.768	6.924	6.553	5.710	4.972	4.254	3.077	2.529
.3000	6.237	6.098	6.180	6.360	6.065	5.308	4.609	3.956	2.885	2.394
.4000	5.509	5.391	5.443	5.638	5.467	4.828	4.218	3.652	2.728	2.297
.5000	4.808	4.712	4.768	4.949	4.803	4.323	3.835	3.345	2.565	2.229
.6000	4.126	4.049	4.074	4.259	4.175	3.806	3.421	3.053	2.447	2.159
.7000	3.561	3.503	3.502	3.692	3.640	3.390	3.108	2.823	2.346	2.109
.8000	3.000	2.959	2.978	3.125	3.077	2.942	2.751	2.561	2.190	2.035
.9000	2.605	2.586	2.629	2.723	2.649	2.574	2.412	2.272	2.033	1.935
Flap										
.0000	2.052	2.068	2.248	2.265	2.149	2.027	1.873	1.773	1.598	1.535
.0125	2.524	2.531	2.610	2.668	2.577	2.468	2.299	2.189	2.012	1.938
.0250	3.067	3.067	3.102	3.208	3.155	3.057	2.853	2.749	2.530	2.450
.0500	3.024	2.993	3.009	3.131	3.137	3.115	2.937	2.847	2.622	2.491
.0750	2.873	2.858	2.861	2.991	3.030	3.027	2.899	2.826	2.595	2.444
.1000	2.716	2.685	2.697	2.833	2.842	2.861	2.760	2.708	2.483	2.335
.1500	1.997	1.972	1.978	2.076	2.143	2.296	2.276	2.295	2.166	2.070
.2000	1.657	1.657	1.672	1.762	1.795	1.876	1.890	1.909	1.858	1.841
.4000	.997	1.049	1.084	1.131	1.167	1.287	1.316	1.437	1.574	1.673
.6000	1.787	1.765	1.737	1.732	1.690	1.758	1.696	1.743	1.778	1.785
.8000	1.836	1.818	1.783	1.771	1.726	1.770	1.710	1.752	1.781	1.794
Spoiler										
.2443	.811	1.048	1.113	1.155	1.240	1.298	1.368	1.490	1.592	1.713
.4906	.938	1.169	1.229	1.271	1.375	1.440	1.503	1.596	1.653	1.733
.7370	1.074	1.307	1.364	1.404	1.486	1.528	1.561	1.634	1.682	1.716
.9856	1.433	1.792	1.847	1.809	1.811	1.791	1.798	1.845	1.838	1.891

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	.988	.784	.814	.951	1.104	1.260	1.310	1.354	1.340	1.268
.0250	.997	.815	.752	.781	.809	.882	.902	.926	.935	.903
.0500	.994	.849	.762	.717	.679	.695	.667	.684	.695	.671
.0750	.994	.867	.774	.717	.649	.637	.591	.602	.615	.594
.1000	.984	.876	.780	.713	.637	.607	.551	.561	.556	.544
.1500	.975	.889	.786	.726	.643	.598	.531	.525	.533	.506
.2000	.963	.892	.796	.729	.637	.604	.522	.519	.512	.497
.2500	.950	.880	.796	.741	.643	.607	.531	.510	.515	.494
.3000	.926	.864	.783	.729	.643	.604	.516	.519	.518	.488
.3500	.895	.842	.777	.723	.649	.610	.533	.522	.518	.491
.4000	.855	.818	.759	.713	.637	.604	.522	.510	.512	.491
.4500	.802	.762	.718	.674	.616	.589	.507	.502	.494	.479
.5000	.741	.713	.672	.634	.574	.544	.478	.487	.465	.456
.5500	.716	.663	.582	.573	.482	.498	.444	.437	.429	.424
.6000	.599	.602	.545	.524	.467	.456	.420	.413	.405	.382
.7000	.420	.410	.396	.387	.333	.338	.293	.295	.284	.274
.7500	.358	.352	.353	.320	.286	.293	.255	.248	.240	.226
.7600	.349	.346	.337	.314	.283	.278	.252	.242	.240	.250
.7700	.370	.373	.375	.332	.295	.320	.284	.274	.257	.229
Vane										
.0250	.963	.994	1.053	.967	.833	.737	.641	.602	.568	.541
.0500	.451	.475	.551	.485	.381	.299	.241	.230	.222	.185
.1000	.191	.213	.337	.259	.176	.133	.081	.065	.074	.053
.1500	.056	.089	.183	.134	.077	.042	.020	.027	.024	.015
.2000	.031	.062	.121	.085	.042	.033	.020	.015	.030	.026
.3000	.028	.028	.068	.055	.024	.039	.029	.050	.062	.035
.4000	.006	.037	.053	.037	.033	.039	.035	.053	.071	.038
.5000	.043	.043	.046	.061	.077	.073	.072	.074	.083	.053
.6000	.065	.083	.068	.098	.104	.100	.093	.089	.095	.082
.7000	.136	.139	.118	.143	.149	.163	.133	.139	.136	.121
.8000	.275	.293	.276	.302	.295	.287	.267	.251	.231	.221
.9200	.913	.907	.910	.939	.923	.894	.826	.776	.728	.709
Flap										
.0125	.191	.210	.291	.296	.265	.248	.220	.212	.186	.159
.0250	.000	.006	.050	.064	.021	.033	.026	.009	.027	.015
.0500	.000	.012	.025	.015	.030	.006	.029	.012	.038	.018
.0750	.000	.025	.040	.018	.033	.033	.023	.015	.056	.047
.1000	.006	.037	.031	.030	.042	.027	.032	.030	.056	.056
.1500	.049	.065	.065	.061	.048	.033	.043	.024	.074	.038
.2000	.083	.105	.087	.076	.077	.085	.064	.059	.086	.074
.4000	.235	.219	.192	.195	.185	.175	.159	.150	.163	.144
.6000	.376	.383	.337	.338	.318	.308	.273	.271	.287	.259
.8000	.614	.602	.548	.546	.509	.511	.461	.460	.482	.447
.9000	.827	.809	.762	.747	.711	.713	.664	.655	.672	.665
Spoiler										
.2587	1.331	1.657	1.712	1.687	1.709	1.699	1.716	1.772	1.754	1.785
.5039	1.471	1.723	1.768	1.733	1.727	1.717	1.731	1.772	1.763	1.788
.7508	1.424	1.738	1.786	1.748	1.742	1.735	1.731	1.780	1.780	1.802
.9940	1.656	1.879	1.899	1.818	1.796	1.779	1.772	1.824	1.821	1.848

TABLE XII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(e) Double slotted flap configuration with spoiler; $h_s = -0.050\bar{c}$; $\frac{y}{b/2} = 0.55$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	1.198	1.916	2.065	2.332	2.456	2.347	2.237	2.183	2.198	2.096
.0125	1.320	1.919	2.120	2.386	2.474	2.383	2.302	2.242	2.213	2.102
.0250	1.226	1.928	2.089	2.341	2.432	2.371	2.296	2.230	2.221	2.099
.0500	1.213	1.878	2.142	2.405	2.453	2.455	2.344	2.262	2.218	2.096
.0750	1.204	1.819	2.200	2.475	2.492	2.446	2.349	2.280	2.204	2.096
.1000	1.216	1.728	2.268	2.529	2.529	2.425	2.367	2.277	2.198	2.096
.1500	1.245	1.559	2.631	2.587	2.468	2.419	2.397	2.282	2.198	2.093
.2000	1.260	1.431	2.843	2.870	2.432	2.425	2.391	2.277	2.201	2.096
.2500	1.298	1.416	2.695	3.456	2.601	2.455	2.415	2.277	2.204	2.093
.3000	1.323	1.419	2.203	3.438	2.877	2.485	2.388	2.265	2.195	2.087
.3500	1.357	1.450	1.735	2.985	2.895	2.497	2.352	2.233	2.177	2.081
.4000	1.404	1.494	1.434	2.396	2.736	2.431	2.290	2.189	2.151	2.075
.4500	1.458	1.531	1.366	1.924	2.544	2.359	2.252	2.160	2.142	2.058
.5000	1.527	1.609	1.431	1.681	2.366	2.272	2.196	2.137	2.106	2.055
.6201	1.662	1.803	1.723	1.593	1.913	2.033	2.059	2.014	2.027	2.003
.6834	2.075	2.225	2.175	1.982	2.057	2.066	2.065	2.026	2.021	1.997
.7087	2.370	2.519	2.480	2.253	2.174	2.099	2.101	2.049	2.036	2.006
.7151	2.389	2.556	2.511	2.295	2.195	2.099	2.101	2.052	2.038	2.000
.7214	2.448	2.694	2.686	2.475	2.300	2.156	2.133	2.078	2.044	2.006
.7278	2.564	2.797	2.772	2.563	2.342	2.162	2.145	2.078	2.050	2.011
Vane										
.0000	3.264	3.250	3.105	2.757	2.432	2.272	2.234	2.151	2.083	2.041
.0250	6.047	6.103	6.102	5.490	4.459	3.820	3.672	3.443	3.127	2.967
.0500	6.176	6.188	6.231	5.627	4.517	3.817	3.613	3.323	2.971	2.804
.1000	6.565	6.538	6.674	6.083	4.682	3.748	3.474	3.180	2.746	2.571
.1500	6.598	6.538	6.702	6.156	4.625	3.677	3.397	3.087	2.617	2.466
.2000	6.596	6.516	6.717	6.211	4.631	3.674	3.358	3.067	2.572	2.408
.3000	6.210	6.156	6.394	6.016	4.486	3.503	3.225	2.912	2.443	2.285
.4000	5.455	5.413	5.625	5.405	4.048	3.186	2.950	2.682	2.301	2.175
.5000	4.919	4.881	5.080	4.934	3.754	3.000	2.787	2.559	2.236	2.131
.6000	4.188	4.172	4.335	4.265	3.309	2.734	2.592	2.422	2.168	2.070
.7000	3.593	3.581	3.708	3.688	2.979	2.527	2.409	2.280	2.092	2.035
.8000	3.113	3.097	3.209	3.189	2.703	2.362	2.284	2.166	2.033	2.000
.9000	2.690	2.666	2.745	2.733	2.408	2.168	2.107	2.023	1.950	1.950
Flap										
.0000	1.671	1.688	1.825	1.851	1.673	1.446	1.406	1.355	1.333	1.335
.0125	2.514	2.613	2.705	2.727	2.480	2.195	2.136	2.067	2.041	2.052
.0250	2.903	3.075	3.160	3.189	2.922	2.593	2.536	2.482	2.434	2.437
.0500	2.887	3.069	3.148	3.219	3.012	2.701	2.636	2.551	2.484	2.434
.0750	2.803	2.984	3.052	3.162	3.015	2.722	2.639	2.536	2.449	2.373
.1000	2.684	2.816	2.859	2.976	2.871	2.626	2.560	2.457	2.366	2.285
.1500	1.953	1.994	2.034	2.201	2.312	2.201	2.172	2.113	2.062	1.994
.2000	1.762	1.819	1.831	1.936	1.982	1.922	1.912	1.871	1.850	1.822
.4000	1.097	1.138	1.129	1.213	1.366	1.467	1.533	1.539	1.652	1.697
.6000	1.850	1.859	1.840	1.778	1.721	1.707	1.716	1.714	1.764	1.778
.8000	1.812	1.894	1.868	1.797	1.733	1.734	1.737	1.740	1.785	1.801
Spoiler										
.2443	.824	1.039	1.079	1.192	1.408	1.499	1.547	1.613	1.650	1.705
.4881	.978	1.211	1.251	1.392	1.568	1.617	1.649	1.695	1.717	1.762
.7416	1.087	1.361	1.413	1.511	1.622	1.640	1.661	1.698	1.705	1.725
.9856	1.480	1.822	1.908	1.833	1.799	1.770	1.778	1.830	1.809	1.831

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	.947	.806	.929	1.140	1.333	1.446	1.515	1.554	1.617	1.586
.0250	.984	.797	.803	.860	.943	1.006	1.042	1.076	1.112	1.122
.0500	.991	.834	.772	.751	.757	.766	.784	.793	.811	.845
.0750	.969	.831	.754	.696	.673	.665	.675	.676	.673	.682
.1000	.966	.838	.751	.690	.664	.629	.615	.624	.622	.624
.1500	.947	.834	.735	.678	.637	.602	.574	.571	.555	.563
.2000	.928	.834	.735	.681	.631	.587	.559	.548	.522	.522
.2500	.903	.822	.748	.678	.631	.575	.544	.531	.513	.516
.3000	.872	.794	.723	.660	.613	.572	.533	.516	.507	.507
.3500	.837	.769	.705	.648	.604	.563	.524	.507	.493	.496
.4000	.787	.731	.677	.623	.589	.545	.506	.490	.484	.472
.4500	.730	.681	.631	.578	.565	.515	.482	.475	.454	.455
.5000	.668	.631	.582	.541	.517	.476	.456	.443	.431	.426
.6201	.458	.431	.425	.392	.378	.353	.343	.326	.313	.309
.6834	.414	.375	.357	.337	.327	.314	.287	.274	.254	.257
.6961	.398	.369	.366	.325	.318	.293	.272	.259	.251	.257
.7087	.392	.363	.342	.328	.306	.287	.272	.259	.251	.259
Vane										
.0250	1.323	1.156	1.157	1.003	.892	.814	.734	.703	.679	.670
.0500	.677	.488	.597	.508	.438	.365	.325	.294	.289	.274
.1000	.348	.194	.314	.261	.231	.174	.124	.120	.106	.096
.1500	.210	.091	.215	.170	.135	.102	.074	.096	.068	.067
.2000	.150	.053	.145	.112	.090	.096	.065	.085	.059	.058
.3000	.069	.044	.095	.082	.081	.096	.071	.096	.068	.073
.4000	.056	.075	.086	.082	.099	.084	.089	.093	.094	.082
.5000	.050	.075	.095	.076	.120	.132	.101	.114	.100	.105
.6000	.088	.125	.108	.128	.132	.132	.127	.143	.124	.125
.7000	.138	.184	.169	.164	.174	.174	.175	.204	.156	.172
.8000	.288	.331	.326	.319	.324	.287	.284	.292	.283	.286
.9200	.959	1.003	1.012	1.009	.889	.841	.808	.799	.788	.802
Flap										
.0125	.219	.225	.286	.301	.282	.240	.240	.251	.230	.245
.0250	.000	.028	.065	.058	.072	.075	.074	.087	.068	.070
.0500	.019	.028	.046	.046	.078	.057	.077	.079	.047	.058
.0750	.060	.066	.062	.064	.087	.066	.077	.096	.065	.070
.1000	.069	.078	.080	.088	.084	.084	.077	.082	.080	.090
.1500	.082	.094	.098	.100	.108	.102	.095	.108	.086	.093
.2000	.113	.134	.126	.128	.135	.120	.124	.128	.118	.128
.4000	.273	.288	.265	.252	.249	.234	.228	.224	.215	.210
.6000	.420	.431	.391	.383	.366	.341	.331	.335	.322	.306
.8000	.633	.650	.612	.581	.553	.527	.512	.507	.493	.501
.9000	.825	.850	.809	.772	.745	.713	.698	.691	.687	.688
Spoiler										
.2584	1.437	1.759	1.804	1.751	1.733	1.705	1.711	1.757	1.748	1.771
.5039	1.387	1.753	1.826	1.763	1.733	1.714	1.719	1.769	1.754	1.782
.7508	1.421	1.771	1.838	1.772	1.745	1.729	1.728	1.780	1.766	1.793
.9940	1.474	1.883	2.000	1.867	1.787	1.758	1.778	1.807	1.803	1.822

TABLE XII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Concluded

(f) Double slotted flap configuration with spoiler; $h_s = -0.050\bar{c}$; $\frac{y}{b/2} = 0.72$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	2.050	2.184	2.373	2.250	2.129	1.988	1.944	1.912	1.881	1.922
.0125	1.923	2.169	2.300	2.180	2.075	1.947	1.924	1.924	1.896	1.928
.0250	1.873	2.190	2.336	2.195	2.096	1.974	1.947	1.936	1.902	1.925
.0500	1.762	2.253	2.376	2.210	2.102	2.012	1.974	1.945	1.907	1.925
.0750	1.517	2.301	2.422	2.228	2.120	2.030	1.985	1.956	1.905	1.928
.1000	1.203	2.286	2.419	2.219	2.123	2.012	1.982	1.962	1.907	1.928
.1500	1.307	2.416	2.498	2.244	2.117	2.018	2.000	1.980	1.916	1.934
.2000	1.372	2.566	2.578	2.234	2.105	2.041	2.018	1.974	1.925	1.940
.2500	1.390	2.497	2.685	2.216	2.105	2.036	2.018	1.965	1.922	1.940
.3000	1.409	2.274	2.856	2.207	2.117	2.041	2.006	1.974	1.933	1.948
.3500	1.468	2.051	3.049	2.192	2.123	2.038	2.009	1.968	1.922	1.954
.4000	1.486	1.789	3.107	2.253	2.126	2.030	2.000	1.965	1.936	1.951
.4500	1.533	1.617	3.107	2.323	2.111	2.030	1.988	1.965	1.936	1.957
.5000	1.570	1.575	3.061	2.408	2.141	2.024	1.991	1.971	1.945	1.963
.5500	1.604	1.548	2.878	2.362	2.123	2.006	1.980	1.965	1.942	1.965
.6000	1.563	1.566	2.618	2.292	2.111	1.988	1.965	1.959	1.942	1.974
.6500	1.554	1.533	2.352	2.171	2.072	1.974	1.959	1.962	1.954	1.985
.7000	1.505	1.458	2.156	2.125	2.051	1.956	1.953	1.953	1.962	1.991
.7500	1.455	1.401	1.988	2.095	2.024	1.941	1.939	1.942	1.948	1.991
.8000	1.375	1.325	1.868	2.061	2.003	1.929	1.953	1.953	1.965	1.997
.8500	1.331	1.274	1.749	1.981	1.961	1.912	1.930	1.930	1.971	1.994
.9000	1.266	1.262	1.657	1.906	1.925	1.897	1.924	1.948	1.971	1.997

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	.721	.822	1.012	1.113	1.204	1.242	1.298	1.361	1.387	1.441
.0250	.734	.756	.847	.921	.982	1.015	1.070	1.129	1.156	1.200
.0500	.737	.723	.755	.775	.799	.826	.866	.895	.928	.957
.0750	.715	.732	.734	.736	.748	.746	.772	.798	.824	.851
.1000	.712	.738	.731	.714	.724	.708	.725	.748	.769	.788
.1500	.724	.744	.737	.696	.691	.661	.667	.686	.699	.711
.2000	.755	.759	.746	.714	.691	.667	.664	.680	.682	.690
.2500	.793	.780	.771	.730	.706	.684	.673	.683	.679	.690
.3000	.830	.810	.807	.766	.736	.702	.690	.713	.705	.716
.3500	.901	.889	.902	.842	.814	.779	.766	.777	.775	.771
.4000	.978	.997	1.031	.964	.925	.879	.863	.871	.864	.860
.4500	1.155	1.172	1.208	1.125	1.060	1.003	.971	.988	.977	.966
.5000	1.337	1.364	1.398	1.292	1.207	1.159	1.126	1.132	1.113	1.100
.5500	1.458	1.542	1.599	1.487	1.402	1.342	1.319	1.329	1.301	1.272
.6000	1.399	1.696	1.768	1.687	1.613	1.552	1.523	1.546	1.523	1.518
.6500	1.780	1.867	1.899	1.788	1.745	1.708	1.737	1.769	1.740	1.716
.7000	1.836	1.925	1.939	1.839	1.805	1.794	1.822	1.854	1.815	1.768
.7500	1.833	1.901	1.911	1.845	1.817	1.794	1.801	1.830	1.800	1.773
.8000	1.709	1.786	1.829	1.812	1.781	1.735	1.734	1.766	1.746	1.745
.8500	1.573	1.681	1.752	1.772	1.757	1.699	1.699	1.736	1.722	1.745
.9000	1.437	1.572	1.676	1.742	1.736	1.696	1.702	1.719	1.720	1.733
.9500	1.334	1.443	1.605	1.730	1.739	1.711	1.713	1.736	1.731	1.745

TABLE XIII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE

(a) Double slotted flap configuration with spoiler; $h_s = -0.100\bar{c}$; $\frac{y}{b/2} = 0$

Upper surface

x/c	C_p for -									
	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Fuselage										
.0000	.000	.006	.000	.000	.022	.039	.059	.077	.111	.142
.0500	.748	.840	.884	.947	.997	1.015	1.033	1.050	1.060	1.080
.1000	.852	.929	.967	1.009	1.055	1.063	1.083	1.094	1.117	1.136
.1500	.937	.991	1.028	1.072	1.089	1.090	1.113	1.115	1.132	1.142
.2000	.984	1.040	1.058	1.106	1.108	1.101	1.110	1.115	1.117	1.118
.2500	1.032	1.080	1.092	1.131	1.123	1.122	1.130	1.124	1.126	1.136
.3000	1.057	1.086	1.107	1.131	1.117	1.084	1.092	1.103	1.111	1.118
.3500	1.063	1.083	1.088	1.115	1.105	1.090	1.086	1.092	1.108	1.130
.4000	1.032	1.043	1.058	1.072	1.065	1.078	1.083	1.094	1.102	1.133
.4500	1.038	1.046	1.070	1.115	1.129	1.125	1.169	1.192	1.222	1.260
.5000	1.044	1.065	1.107	1.165	1.209	1.233	1.287	1.313	1.398	1.534
.5500	1.063	1.117	1.162	1.233	1.268	1.298	1.349	1.387	1.578	1.876
.6000	1.095	1.142	1.204	1.283	1.305	1.307	1.337	1.363	1.632	1.929
.6500	1.120	1.182	1.241	1.295	1.308	1.295	1.287	1.310	1.476	1.770
.7000	1.164	1.218	1.271	1.317	1.317	1.307	1.302	1.286	1.437	1.696
.7500	1.221	1.262	1.308	1.329	1.326	1.331	1.317	1.336	1.425	1.649
.8000	1.265	1.308	1.329	1.339	1.357	1.352	1.382	1.422	1.476	1.575
.8500	1.294	1.305	1.314	1.336	1.354	1.367	1.414	1.457	1.491	1.519
.9000	1.224	1.231	1.238	1.258	1.277	1.298	1.343	1.389	1.428	1.440
.9500	1.224	1.194	1.180	1.193	1.218	1.224	1.281	1.333	1.365	1.387
.9940	1.221	1.212	1.192	1.199	1.215	1.245	1.296	1.322	1.377	1.387

Lower surface

x/c	C_p for -									
	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Fuselage										
.0500	.855	.815	.765	.705	.634	.546	.485	.440	.404	.386
.1000	.928	.914	.863	.804	.742	.645	.589	.543	.512	.484
.1500	1.013	.978	.924	.888	.812	.719	.675	.628	.590	.561
.2000	1.022	1.000	.967	.941	.855	.788	.737	.693	.647	.625
.2500	1.044	1.040	1.003	.969	.914	.842	.772	.743	.704	.681
.3000	1.054	1.052	1.015	.981	.935	.875	.814	.779	.746	.714
.3500	1.035	1.025	1.015	.994	.945	.893	.840	.805	.763	.740
.4000	.972	.972	.951	.935	.895	.830	.790	.738	.710	.687
.4500	.943	.942	.912	.895	.840	.779	.716	.673	.653	.611
.5000	.909	.902	.866	.836	.778	.701	.642	.608	.569	.537
.5500	.871	.868	.826	.786	.720	.642	.577	.546	.509	.496
.6000	.823	.806	.747	.705	.649	.564	.512	.490	.455	.448
.7500	.934	.911	.884	.860	.822	.764	.713	.696	.674	.667
.9500	1.155	1.151	1.156	1.196	1.194	1.176	1.192	1.198	1.231	1.245
.9940	1.088	1.175	1.183	1.249	1.268	1.290	1.329	1.348	1.386	1.372

TABLE XIII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(b) Double slotted flap configuration with spoiler; $h_s = -0.100\bar{c}$; $\frac{y}{b/2} = 0.21$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0000	.876	.908	1.842	3.168	5.397	7.416	9.440	7.392	3.347	2.504
.0125	.879	1.252	1.648	3.046	7.582	10.266	11.121	7.078	3.033	2.291
.0250	.918	1.200	1.502	2.412	3.659	5.871	8.392	6.507	3.050	2.312
.0500	.957	1.160	1.404	1.637	2.006	2.593	4.220	4.835	2.780	2.198
.0750	.970	1.145	1.347	1.525	1.815	2.104	2.940	3.752	2.687	2.192
.1000	.988	1.160	1.319	1.509	1.748	1.951	2.604	3.279	2.619	2.175
.1500	1.009	1.157	1.304	1.460	1.603	1.774	2.184	2.616	2.499	2.137
.2000	1.042	1.175	1.301	1.436	1.560	1.694	2.006	2.293	2.393	2.122
.2500	1.057	1.185	1.295	1.415	1.498	1.627	1.884	2.068	2.258	2.105
.3000	1.088	1.197	1.301	1.396	1.468	1.581	1.803	1.938	2.147	2.081
.3500	1.094	1.209	1.292	1.378	1.443	1.541	1.678	1.799	2.036	2.067
.4000	1.118	1.218	1.292	1.384	1.425	1.523	1.625	1.728	1.959	2.049
.4500	1.139	1.237	1.304	1.387	1.428	1.505	1.565	1.675	1.892	2.043
.5000	1.173	1.268	1.332	1.406	1.434	1.480	1.539	1.648	1.854	2.026
.5500	1.188	1.277	1.322	1.387	1.431	1.440	1.500	1.592	1.777	1.997
.6000	1.212	1.298	1.341	1.406	1.434	1.456	1.500	1.577	1.728	1.965
.6500	1.239	1.311	1.356	1.403	1.443	1.453	1.509	1.604	1.684	1.947
.7000	1.288	1.369	1.383	1.448	1.471	1.480	1.559	1.633	1.666	1.906
.7815	1.385	1.458	1.468	1.540	1.572	1.593	1.622	1.678	1.678	1.880
.8179	1.512	1.582	1.587	1.665	1.689	1.774	1.750	1.799	1.760	1.880
.8325	1.606	1.655	1.660	1.744	1.779	1.871	1.839	1.876	1.818	1.901
.8362	1.630	1.683	1.681	1.771	1.812	1.896	1.866	1.906	1.848	1.903
.8398	1.597	1.652	1.642	1.726	1.779	1.847	1.824	1.870	1.821	1.895
.8434	1.639	1.748	1.730	1.814	1.868	1.957	1.943	1.971	1.892	1.906
Vane										
.0000	2.054	2.068	2.061	2.186	2.231	2.349	2.268	2.275	2.138	2.070
.0250	3.275	3.329	3.326	3.503	3.625	4.079	4.038	4.033	3.678	3.329
.0500	3.100	3.129	3.131	3.269	3.354	3.829	3.797	3.788	3.420	3.069
.1000	3.036	3.065	3.040	3.140	3.206	3.694	3.675	3.645	3.238	2.711
.1500	2.933	2.957	2.937	3.034	3.092	3.587	3.571	3.557	3.182	2.635
.2000	2.842	2.874	2.836	2.939	3.025	3.486	3.479	3.492	3.171	2.653
.3000	2.476	2.542	2.499	2.601	2.729	3.119	3.161	3.213	3.018	2.583
.4000	2.206	2.243	2.219	2.332	2.474	2.737	2.780	2.888	2.798	2.449
.5000	2.009	2.031	2.040	2.171	2.308	2.483	2.518	2.639	2.610	2.355
.6000	1.760	1.819	1.848	1.951	2.043	2.186	2.202	2.296	2.305	2.163
.7000	1.709	1.779	1.797	1.878	1.951	2.113	2.095	2.169	2.147	2.073
.8000	1.712	1.779	1.794	1.854	1.939	2.079	2.074	2.154	2.088	2.014
.9000	1.706	1.763	1.775	1.826	1.908	2.031	2.039	2.133	2.018	1.924
Flap										
.0000	.912	1.391	1.496	1.424	1.172	1.171	1.155	1.207	1.156	1.128
.0125	1.770	1.942	1.988	1.918	1.852	1.979	1.982	2.039	1.901	1.834
.0250	1.918	2.043	2.088	2.116	2.163	2.318	2.321	2.388	2.253	2.163
.0500	1.960	2.080	2.137	2.201	2.283	2.422	2.449	2.524	2.426	2.414
.0750	1.821	1.939	1.982	2.055	2.108	2.229	2.232	2.314	2.270	2.297
.1000	1.645	1.723	1.763	1.848	1.911	1.945	1.985	2.059	2.065	2.163
.1500	1.327	1.354	1.401	1.503	1.548	1.563	1.604	1.669	1.728	1.859
.2000	1.257	1.286	1.344	1.448	1.495	1.535	1.556	1.619	1.628	1.766
.4000	1.109	1.172	1.289	1.390	1.462	1.517	1.437	1.447	1.308	1.393
.6000	1.530	1.557	1.584	1.674	1.714	1.777	1.774	1.811	1.789	1.834
.8000	1.585	1.612	1.634	1.747	1.815	1.899	1.881	1.900	1.842	1.845
Spoiler										
.2429	1.000	1.154	1.186	1.182	1.120	1.140	1.104	1.080	1.111	1.177
.4852	.997	1.145	1.162	1.242	1.366	1.376	1.406	1.407	1.440	1.505
.7367	.988	1.172	1.177	1.263	1.363	1.409	1.441	1.446	1.500	1.584
.9796	1.189	1.360	1.369	1.485	1.591	1.636	1.657	1.670	1.707	1.743

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0125	1.039	.800	.635	.592	.572	.627	.664	.633	.431	.361
.0250	1.009	.825	.654	.564	.449	.398	.310	.311	.243	.210
.0500	.988	.849	.705	.601	.443	.339	.229	.237	.232	.222
.0750	.976	.855	.724	.625	.480	.373	.271	.284	.279	.286
.1000	.985	.874	.742	.649	.511	.416	.321	.340	.340	.350
.1500	.970	.874	.754	.674	.563	.486	.411	.432	.416	.417
.2000	.970	.889	.775	.698	.597	.541	.455	.476	.452	.452
.2500	.991	.908	.803	.738	.652	.593	.512	.524	.499	.487
.3000	.970	.911	.818	.759	.671	.624	.545	.553	.531	.522
.3500	.985	.929	.839	.790	.714	.661	.580	.589	.557	.545
.4000	.976	.932	.845	.808	.735	.685	.601	.610	.578	.571
.4500	.951	.923	.845	.814	.745	.700	.619	.624	.592	.583
.5000	.888	.880	.815	.787	.726	.685	.601	.610	.592	.583
.5500	.833	.862	.751	.771	.677	.648	.589	.615	.595	.560
.6000	.761	.769	.720	.707	.652	.627	.554	.574	.546	.536
.6500	.706	.680	.635	.631	.591	.566	.500	.518	.499	.490
.7000	.676	.591	.553	.549	.514	.495	.440	.473	.449	.440
.7815	.570	.492	.426	.418	.385	.367	.324	.352	.334	.324
.8179	.482	.431	.389	.366	.351	.346	.289	.314	.305	.300
.8252	.470	.422	.395	.378	.354	.349	.304	.340	.317	.312
.8325	.455	.418	.398	.399	.385	.391	.351	.388	.372	.361
Vane										
.0250	1.230	1.212	1.210	1.207	1.132	1.079	.982	.994	.924	.892
.0500	.776	.742	.733	.646	.514	.419	.345	.355	.337	.324
.1000	.506	.474	.432	.323	.178	.095	.048	.083	.076	.070
.1500	.400	.375	.334	.223	.077	.024	.000	.018	.021	.026
.2000	.342	.338	.292	.174	.037	.015	.000	.018	.009	.012
.3000	.276	.262	.231	.119	.025	.012	.000	.030	.032	.020
.4000	.242	.240	.201	.104	.034	.037	.015	.047	.041	.041
.5000	.188	.206	.182	.113	.049	.055	.039	.074	.067	.061
.6000	.185	.185	.176	.116	.071	.083	.060	.101	.091	.087
.7000	.197	.200	.188	.134	.126	.138	.131	.160	.141	.134
.8000	.242	.246	.237	.210	.222	.248	.220	.260	.232	.224
.9200	.585	.603	.611	.619	.662	.713	.696	.743	.686	.665
Flap										
.0125	.418	.342	.292	.125	.055	.061	.042	.083	.065	.061
.0250	.179	.086	.043	.018	.012	.018	.000	.027	.032	.032
.0500	.118	.062	.030	.046	.046	.055	.039	.074	.067	.073
.0750	.118	.080	.049	.064	.074	.080	.065	.092	.091	.090
.1000	.127	.105	.085	.098	.111	.110	.086	.124	.117	.120
.1500	.155	.148	.122	.131	.151	.150	.131	.160	.150	.146
.4000	.388	.391	.371	.381	.375	.370	.330	.358	.340	.326
.6000	.570	.551	.538	.540	.535	.514	.458	.479	.458	.443
.8000	.703	.692	.669	.674	.652	.639	.577	.601	.569	.551
.9000	.800	.818	.787	.793	.785	.761	.723	.707	.683	.656
Spoiler										
.2573	1.407	1.557	1.595	1.687	1.739	1.800	1.826	1.826	1.817	1.823
.4984	1.505	1.615	1.640	1.727	1.782	1.845	1.870	1.873	1.868	1.853
.7485	1.524	1.628	1.662	1.747	1.788	1.842	1.876	1.888	1.868	1.861
.9940	1.587	1.662	1.689	1.768	1.809	1.878	1.917	1.926	1.898	1.891

TABLE XIII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(c) Double slotted flap configuration with spoiler; $h_s = -0.100\bar{c}$; $\frac{y}{b/2} = 0.30$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	.920	1.179	2.015	2.836	3.468	4.141	4.235	3.740	2.699	2.129
.0125	.963	1.407	2.165	2.854	3.414	3.871	3.939	3.686	2.687	2.123
.0250	1.000	1.308	2.265	2.939	3.547	4.012	3.982	3.701	2.693	2.129
.0500	1.037	1.250	2.701	3.817	3.843	3.933	3.954	3.710	2.651	2.120
.0750	1.031	1.207	1.650	3.854	5.151	5.101	4.332	3.719	2.663	2.123
.1000	1.053	1.197	1.210	2.412	3.958	4.758	4.263	3.630	2.681	2.132
.1500	1.071	1.213	1.262	1.378	2.091	3.333	3.643	3.322	2.598	2.135
.2000	1.102	1.219	1.299	1.314	1.565	2.428	3.045	3.012	2.527	2.135
.2500	1.118	1.222	1.308	1.342	1.441	1.975	2.598	2.743	2.480	2.123
.3000	1.146	1.234	1.320	1.332	1.396	1.804	2.371	2.522	2.397	2.129
.3500	1.146	1.241	1.308	1.339	1.378	1.657	2.091	2.310	2.296	2.100
.4000	1.167	1.259	1.323	1.360	1.402	1.615	1.970	2.176	2.225	2.088
.4500	1.207	1.284	1.345	1.369	1.408	1.605	1.900	2.075	2.145	2.088
.5000	1.229	1.305	1.363	1.378	1.417	1.578	1.813	1.955	2.033	2.065
.5500	1.257	1.315	1.360	1.378	1.414	1.560	1.743	1.890	1.988	2.018
.6000	1.285	1.349	1.390	1.396	1.435	1.569	1.722	1.827	1.932	1.994
.6500	1.322	1.392	1.430	1.439	1.480	1.578	1.707	1.788	1.885	1.968
.7521	1.421	1.497	1.518	1.564	1.619	1.676	1.725	1.824	1.846	1.856
.7934	1.628	1.703	1.710	1.762	1.825	1.917	1.952	1.979	1.882	1.897
.8099	1.740	1.805	1.817	1.872	1.964	2.082	2.100	2.087	1.935	1.900
.8141	1.789	1.864	1.869	1.933	2.036	2.156	2.166	2.146	1.959	1.915
.8182	1.814	1.898	1.903	1.964	2.072	2.193	2.196	2.188	1.985	1.912
.8224	1.904	1.990	2.000	2.089	2.193	2.294	2.317	2.295	2.048	1.920
Vane										
.0000	3.480	3.586	3.467	3.586	3.819	4.067	4.024	3.916	3.432	3.206
.0250	4.127	4.243	4.189	4.509	4.825	5.177	5.241	4.928	3.823	3.235
.0500	4.056	4.203	4.147	4.421	4.764	5.140	5.187	4.857	3.734	3.150
.1000	3.941	4.092	4.031	4.327	4.661	5.006	5.127	4.755	3.512	2.676
.1500	3.817	3.981	3.915	4.208	4.577	4.929	5.066	4.695	3.468	2.653
.2000	3.647	3.833	3.762	4.073	4.477	4.856	5.015	4.692	3.521	2.688
.3000	3.059	3.287	3.223	3.506	3.982	4.428	4.640	4.391	3.409	2.629
.4000	2.396	2.657	2.598	2.860	3.365	3.847	4.048	3.854	3.101	2.467
.5000	1.910	2.179	2.104	2.320	2.885	3.413	3.565	3.424	2.864	2.373
.6000	1.672	1.852	1.811	1.951	2.408	2.899	3.045	2.979	2.604	2.259
.7000	1.610	1.722	1.720	1.799	2.072	2.465	2.607	2.606	2.382	2.135
.8000	1.598	1.676	1.686	1.744	1.846	2.104	2.229	2.251	2.122	2.023
.9000	1.610	1.660	1.680	1.720	1.728	1.896	1.985	2.006	1.941	1.909
Flap										
.0000	1.046	1.120	1.305	1.214	1.088	1.092	1.145	1.170	1.145	1.141
.0125	1.536	1.648	1.717	1.741	1.701	1.728	1.794	1.848	1.805	1.794
.0250	1.855	1.981	2.040	2.082	2.075	2.125	2.226	2.301	2.264	2.244
.0500	1.749	1.870	1.924	1.970	2.012	2.079	2.184	2.322	2.355	2.329
.0750	1.542	1.703	1.787	1.826	1.876	1.939	2.060	2.218	2.290	2.276
.1000	1.433	1.537	1.601	1.619	1.746	1.826	1.933	2.063	2.142	2.153
.1500	1.279	1.302	1.329	1.351	1.384	1.324	1.390	1.564	1.755	1.879
.2000	1.235	1.210	1.262	1.238	1.214	1.113	1.205	1.328	1.491	1.670
.4000	1.050	.972	.979	.948	.776	.633	.634	.710	.814	1.000
.6000	1.539	1.605	1.613	1.695	1.734	1.823	1.843	1.866	1.832	1.779
.8000	1.557	1.629	1.650	1.744	1.770	1.865	1.885	1.901	1.858	1.809
Spoiler										
.2475	.984	.972	.985	.980	.858	.648	.589	.555	.728	.953
.4909	1.000	.972	1.015	.960	.846	.612	.654	.714	.991	1.257
.7396	.909	.935	.957	.919	.892	.678	.755	.844	1.120	1.325
.9796	1.215	1.351	1.363	1.424	1.529	1.561	1.645	1.682	1.728	1.779

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.022	.805	.729	.790	.940	1.138	1.227	1.257	1.124	.997
.0250	1.025	.852	.713	.683	.710	.777	.804	.824	.772	.706
.0500	1.006	.895	.750	.649	.601	.587	.559	.552	.536	.506
.0750	1.009	.907	.768	.665	.589	.544	.495	.481	.473	.447
.1000	1.009	.917	.774	.668	.598	.538	.471	.454	.438	.424
.1500	1.006	.935	.808	.704	.622	.557	.483	.448	.441	.421
.2000	1.006	.938	.829	.726	.631	.572	.501	.466	.441	.424
.2500	1.006	.944	.842	.744	.650	.596	.514	.481	.459	.444
.3000	.994	.950	.848	.753	.671	.612	.538	.498	.479	.450
.3500	.978	.954	.854	.765	.686	.624	.559	.522	.491	.471
.4000	.947	.923	.838	.759	.686	.636	.574	.531	.509	.488
.4500	.920	.895	.826	.747	.683	.642	.580	.543	.521	.497
.5000	.851	.849	.787	.720	.665	.624	.574	.543	.515	.482
.5500	.793	.799	.735	.683	.634	.605	.553	.525	.503	.476
.6000	.712	.710	.659	.622	.574	.563	.511	.484	.473	.450
.6500	.644	.639	.579	.555	.511	.508	.462	.439	.429	.409
.7521	.477	.444	.399	.363	.338	.336	.314	.299	.290	.294
.7934	.412	.395	.360	.314	.296	.281	.269	.248	.246	.244
.8017	.409	.395	.360	.314	.284	.291	.272	.248	.240	.247
.8099	.396	.376	.360	.317	.296	.294	.296	.287	.278	.265
Vane										
.0250	1.517	1.531	1.494	1.421	1.356	1.333	1.242	1.188	1.071	1.009
.0500	.746	.753	.787	.671	.538	.453	.381	.337	.305	.294
.1000	.319	.339	.415	.299	.157	.080	.042	.030	.030	.035
.1500	.183	.213	.290	.177	.060	.012	.000	.000	.000	.000
.2000	.115	.136	.210	.116	.018	.006	.000	.000	.000	.000
.3000	.059	.077	.140	.067	.003	.012	.000	.000	.000	.003
.4000	.034	.040	.091	.027	.003	.021	.018	.009	.015	.026
.5000	.031	.037	.061	.030	.021	.043	.033	.024	.036	.047
.6000	.046	.052	.049	.037	.033	.064	.054	.048	.059	.068
.7000	.090	.086	.079	.073	.088	.119	.109	.099	.118	.112
.8000	.189	.191	.155	.180	.190	.226	.224	.212	.213	.218
.9200	.563	.596	.576	.610	.604	.648	.665	.660	.639	.653
Flap										
.0125	.124	.142	.204	.149	.103	.116	.112	.122	.112	.135
.0250	.009	.009	.012	.003	.000	.018	.006	.000	.024	.006
.0500	.012	.012	.003	.006	.000	.012	.006	.000	.000	.003
.0750	.028	.025	.006	.009	.000	.018	.012	.000	.003	.006
.1000	.037	.040	.015	.018	.012	.024	.015	.000	.012	.012
.1500	.077	.071	.055	.052	.042	.061	.042	.027	.036	.041
.2000	.105	.111	.088	.079	.079	.089	.069	.051	.062	.059
.4000	.254	.259	.238	.226	.202	.202	.172	.155	.166	.141
.6000	.406	.389	.384	.357	.338	.346	.311	.290	.284	.271
.8000	.598	.593	.561	.549	.529	.511	.486	.475	.473	.429
.9000	.771	.784	.756	.747	.728	.716	.686	.687	.686	.632
Spoiler										
.2587	1.404	1.572	1.598	1.727	1.779	1.833	1.861	1.847	1.835	1.820
.5024	1.461	1.600	1.622	1.747	1.819	1.866	1.888	1.891	1.862	1.847
.7531	1.524	1.628	1.656	1.788	1.843	1.895	1.929	1.920	1.880	1.864
.9940	1.568	1.689	1.714	1.828	1.895	1.955	1.997	1.982	1.943	1.929

TABLE XIII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(d) Double slotted flap configuration with spoiler; $h_s = -0.100\bar{c}$; $\frac{y}{b/2} = 0.43$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0000	.764	1.782	2.009	2.439	2.874	2.813	2.580	2.512	2.352	2.108
.0125	.967	1.742	1.982	2.418	2.846	2.823	2.690	2.577	2.382	2.134
.0250	1.006	1.655	2.040	2.470	2.837	2.865	2.729	2.586	2.364	2.110
.0500	1.009	1.600	2.070	2.540	2.966	2.945	2.738	2.601	2.379	2.151
.0750	1.036	1.514	2.307	2.619	2.911	2.841	2.756	2.613	2.370	2.137
.1000	1.061	1.382	2.617	2.890	2.837	2.795	2.765	2.613	2.364	2.131
.1500	1.073	1.268	2.259	3.552	3.440	2.997	2.815	2.619	2.376	2.134
.2000	1.112	1.268	1.511	2.750	3.757	3.220	2.771	2.574	2.352	2.122
.2500	1.115	1.274	1.192	1.851	3.108	3.067	2.649	2.500	2.317	2.116
.3000	1.157	1.311	1.216	1.470	2.563	2.871	2.565	2.450	2.291	2.105
.3500	1.167	1.308	1.249	1.311	2.092	2.566	2.417	2.349	2.235	2.087
.4000	1.194	1.323	1.295	1.296	1.843	2.370	2.318	2.275	2.194	2.087
.4500	1.224	1.351	1.322	1.305	1.705	2.211	2.229	2.210	2.150	2.044
.5000	1.230	1.348	1.335	1.311	1.563	2.006	2.083	2.110	2.088	2.035
.5500	1.282	1.400	1.383	1.363	1.551	1.923	2.024	2.062	2.038	2.020
.6000	1.330	1.428	1.420	1.409	1.545	1.838	1.952	1.997	1.989	1.994
.7000	1.342	1.539	1.514	1.558	1.612	1.764	1.863	1.926	1.912	1.938
.7500	1.545	1.711	1.690	1.750	1.809	1.905	1.934	1.956	1.918	1.933
.7700	1.645	1.859	1.818	1.912	1.994	2.061	2.042	2.039	1.956	1.944
.7750	1.673	1.880	1.857	1.951	2.046	2.110	2.071	2.057	1.959	1.956
.7800	1.727	1.935	1.915	2.012	2.114	2.174	2.113	2.080	1.977	1.956
Vane										
.0000	2.830	2.895	2.788	2.903	3.019	3.076	2.914	2.790	2.552	2.310
.0250	3.751	3.874	3.791	4.119	4.462	4.498	4.181	3.870	3.226	3.014
.0500	3.778	3.865	3.800	4.159	4.523	4.547	4.205	3.829	3.068	2.787
.1000	3.763	3.815	3.760	4.141	4.557	4.529	4.101	3.663	2.801	2.463
.1500	3.672	3.745	3.669	4.080	4.539	4.581	4.128	3.651	2.731	2.364
.2000	3.621	3.708	3.612	4.040	4.575	4.715	4.262	3.773	2.839	2.463
.3000	2.927	3.062	2.900	3.293	3.908	4.211	3.845	3.435	2.622	2.300
.4000	2.300	2.462	2.277	2.604	3.145	3.679	3.461	3.128	2.475	2.221
.5000	1.818	1.966	1.827	2.058	2.474	3.150	3.062	2.844	2.341	2.186
.6000	1.597	1.680	1.629	1.781	2.025	2.630	2.693	2.568	2.217	2.105
.7000	1.539	1.591	1.575	1.704	1.871	2.269	2.381	2.341	2.112	2.055
.8000	1.533	1.575	1.566	1.698	1.840	1.994	2.098	2.107	1.992	1.985
.9000	1.536	1.575	1.575	1.701	1.840	1.826	1.878	1.888	1.854	1.909
Flap										
.0000	1.115	1.280	1.429	1.473	1.523	1.443	1.405	1.420	1.417	1.487
.0125	1.376	1.594	1.620	1.707	1.874	1.801	1.759	1.778	1.789	1.869
.0250	1.645	1.960	1.943	2.012	2.271	2.190	2.167	2.216	2.247	2.355
.0500	1.563	1.843	1.845	1.896	2.092	2.003	2.089	2.207	2.282	2.382
.0750	1.433	1.708	1.708	1.756	1.895	1.823	1.934	2.104	2.206	2.306
.1000	1.394	1.575	1.569	1.610	1.631	1.618	1.803	1.968	2.112	2.224
.1500	1.215	1.283	1.325	1.360	1.317	1.187	1.327	1.556	1.804	1.956
.2000	1.191	1.200	1.246	1.268	1.206	1.012	1.125	1.305	1.560	1.793
.4000	.945	.948	.994	1.021	.951	.661	.631	.725	.909	1.116
.6000	1.406	1.514	1.499	1.573	1.640	1.688	1.723	1.749	1.736	1.764
.8000	1.324	1.514	1.502	1.567	1.649	1.694	1.723	1.746	1.736	1.789
Spoiler										
.2443	.928	.954	1.015	1.044	.969	.690	.651	.655	.904	1.092
.4906	.921	.951	1.040	1.075	.966	.818	.861	.926	1.281	1.475
.7370	.871	.951	1.015	1.062	.957	.887	.994	1.100	1.410	1.566
.9856	1.129	1.329	1.366	1.435	1.375	1.510	1.613	1.664	1.784	1.829

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 28^\circ$
Wing										
.0125	1.067	.803	.787	.909	1.108	1.214	1.280	1.311	1.288	1.242
.0250	1.042	.840	.742	.759	.818	.853	.881	.908	.895	.898
.0500	1.030	.877	.754	.720	.698	.673	.667	.678	.675	.685
.0750	1.027	.895	.763	.707	.658	.621	.595	.592	.590	.600
.1000	1.024	.898	.784	.707	.649	.602	.559	.553	.551	.568
.1500	1.006	.911	.793	.729	.649	.590	.530	.524	.519	.516
.2000	.988	.914	.793	.732	.655	.590	.533	.536	.502	.510
.2500	.973	.902	.790	.735	.658	.593	.539	.536	.499	.504
.3000	.939	.889	.778	.726	.665	.599	.539	.533	.504	.501
.3500	.912	.865	.772	.726	.665	.605	.545	.541	.510	.516
.4000	.870	.831	.748	.707	.649	.593	.533	.527	.504	.507
.4500	.818	.791	.708	.671	.622	.584	.524	.512	.484	.493
.5000	.761	.732	.657	.619	.585	.535	.494	.494	.463	.475
.5500	.703	.634	.578	.607	.551	.492	.443	.456	.440	.446
.6000	.615	.591	.526	.509	.489	.450	.417	.423	.396	.399
.7000	.412	.418	.380	.360	.351	.315	.292	.293	.276	.292
.7500	.361	.354	.340	.311	.295	.260	.244	.249	.243	.239
.7600	.358	.338	.334	.305	.283	.260	.244	.246	.235	.233
.7700	.370	.440	.334	.329	.305	.294	.244	.281	.246	.257
Vane										
.0250	.976	.957	.970	.924	.812	.743	.649	.624	.569	.574
.0500	.558	.529	.590	.518	.412	.315	.288	.240	.214	.222
.1000	.321	.305	.398	.323	.215	.131	.089	.080	.070	.076
.1500	.182	.163	.264	.195	.098	.003	.024	.044	.044	.026
.2000	.127	.102	.204	.137	.052	.006	.000	.038	.026	.000
.3000	.058	.055	.134	.076	.062	.015	.030	.047	.053	.041
.4000	.052	.040	.082	.043	.049	.021	.039	.056	.053	.050
.5000	.048	.055	.058	.034	.043	.043	.060	.071	.059	.070
.6000	.055	.077	.058	.049	.058	.055	.077	.086	.079	.090
.7000	.094	.117	.082	.082	.098	.101	.113	.118	.117	.128
.8000	.179	.212	.152	.171	.218	.205	.214	.210	.208	.236
.9200	.570	.612	.556	.598	.680	.657	.643	.663	.657	.714
Flap										
.0125	.127	.145	.204	.189	.163	.128	.155	.157	.155	.178
.0250	.085	.025	.012	.037	.018	.000	.009	.033	.041	.035
.0500	.036	.052	.015	.018	.009	.003	.003	.044	.021	.041
.0750	.036	.080	.058	.030	.043	.021	.012	.062	.038	.050
.1000	.058	.092	.040	.034	.037	.018	.012	.047	.041	.044
.1500	.073	.092	.064	.043	.049	.034	.048	.056	.044	.064
.2000	.127	.132	.103	.095	.074	.070	.071	.080	.065	.087
.4000	.258	.265	.222	.216	.191	.162	.173	.169	.141	.160
.6000	.379	.406	.340	.341	.323	.300	.280	.284	.264	.271
.8000	.612	.603	.556	.552	.532	.502	.491	.482	.455	.469
.9000	.758	.775	.730	.744	.729	.709	.684	.689	.660	.668
Spoiler										
.2587	.303	.492	.506	.584	.622	.666	.701	.708	.740	.735
.5039	.385	.529	.540	.606	.643	.692	.722	.738	.775	.782
.7508	.353	.514	.543	.618	.649	.687	.734	.749	.784	.788
.9940	.464	.572	.595	.668	.699	.740	.790	.800	.847	.853

TABLE XIII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Continued

(e) Double slotted flap configuration with spoiler; $h_s = -0.100\bar{c}$; $\frac{y}{b/2} = 0.55$

Upper surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0000	1.006	1.882	1.985	2.265	2.390	2.300	2.166	2.152	2.157	2.032
.0125	1.198	1.867	2.018	2.326	2.417	2.330	2.236	2.194	2.163	2.035
.0250	1.136	1.827	1.982	2.299	2.393	2.333	2.233	2.200	2.175	2.023
.0500	1.130	1.676	2.018	2.357	2.414	2.410	2.284	2.224	2.178	2.026
.0750	1.124	1.531	2.101	2.412	2.423	2.391	2.281	2.230	2.180	2.023
.1000	1.142	1.450	2.156	2.454	2.459	2.382	2.293	2.221	2.145	2.026
.1500	1.164	1.386	2.497	2.497	2.381	2.379	2.311	2.221	2.145	2.026
.2000	1.192	1.345	2.650	2.741	2.350	2.382	2.290	2.212	2.142	2.032
.2500	1.214	1.352	2.479	3.308	2.483	2.401	2.302	2.215	2.145	2.026
.3000	1.226	1.364	2.021	3.339	2.719	2.425	2.284	2.206	2.145	2.029
.3500	1.257	1.383	1.573	2.936	2.743	2.404	2.248	2.179	2.119	2.023
.4000	1.282	1.401	1.305	2.366	2.610	2.358	2.196	2.137	2.095	2.023
.4500	1.313	1.416	1.265	1.921	2.453	2.290	2.157	2.125	2.092	2.003
.5000	1.353	1.454	1.305	1.656	2.320	2.214	2.109	2.101	2.068	2.000
.6201	1.396	1.512	1.467	1.454	1.894	1.988	1.973	1.994	2.009	1.953
.6834	1.632	1.734	1.704	1.680	1.955	1.997	1.982	2.006	2.000	1.941
.7087	1.759	1.861	1.835	1.832	2.018	2.027	2.006	2.009	2.003	1.941
.7151	1.752	1.864	1.835	1.845	2.033	2.027	1.997	2.018	2.003	1.941
.7214	1.786	1.910	1.909	1.939	2.072	2.067	2.033	2.036	2.018	1.944
.7278	1.836	1.957	1.942	1.991	2.094	2.079	2.033	2.033	2.021	1.950
Vane										
.0000	2.269	2.358	2.238	2.247	2.278	2.223	2.154	2.131	2.054	1.994
.0250	3.749	3.858	3.872	4.083	3.988	3.670	3.447	3.313	3.021	2.870
.0500	3.762	3.833	3.900	4.144	4.027	3.654	3.362	3.188	2.850	2.700
.1000	3.864	3.885	4.025	4.360	4.075	3.572	3.208	3.003	2.616	2.470
.1500	3.752	3.759	3.915	4.327	4.000	3.480	3.103	2.901	2.489	2.359
.2000	3.604	3.604	3.769	4.259	3.958	3.443	3.072	2.866	2.438	2.306
.3000	3.059	3.095	3.201	3.833	3.704	3.272	2.915	2.722	2.326	2.194
.4000	2.384	2.438	2.464	3.107	3.208	2.933	2.649	2.510	2.201	2.088
.5000	1.954	1.984	1.988	2.534	2.828	2.703	2.492	2.388	2.139	2.047
.6000	1.700	1.694	1.710	2.028	2.393	2.431	2.284	2.236	2.074	2.012
.7000	1.644	1.614	1.640	1.866	2.124	2.211	2.124	2.107	2.015	1.970
.8000	1.625	1.614	1.631	1.832	1.958	2.055	1.994	2.012	1.956	1.944
.9000	1.628	1.614	1.631	1.829	1.834	1.878	1.837	1.881	1.891	1.906
Flap										
.0000	1.303	1.133	1.156	1.287	1.275	1.214	1.187	1.230	1.278	1.288
.0125	1.622	1.660	1.646	1.842	1.912	1.871	1.840	1.910	1.965	1.982
.0250	1.765	1.889	1.866	2.073	2.217	2.196	2.172	2.254	2.332	2.359
.0500	1.653	1.784	1.765	1.918	2.091	2.159	2.184	2.287	2.358	2.350
.0750	1.539	1.657	1.653	1.775	1.939	2.061	2.100	2.227	2.308	2.288
.1000	1.446	1.524	1.521	1.595	1.788	1.927	1.991	2.122	2.213	2.209
.1500	1.254	1.281	1.284	1.342	1.414	1.480	1.586	1.749	1.914	1.965
.2000	1.180	1.200	1.220	1.247	1.296	1.278	1.375	1.528	1.710	1.759
.4000	.969	.972	1.012	1.043	.930	.761	.795	.884	1.042	1.129
.6000	1.505	1.531	1.525	1.546	1.589	1.633	1.659	1.713	1.740	1.706
.8000	1.495	1.543	1.528	1.567	1.598	1.639	1.662	1.722	1.758	1.715
Spoiler										
.2443	.801	.877	.890	.975	.957	.731	.772	.823	1.003	1.118
.4881	.820	.880	.884	.950	1.083	1.087	1.186	1.224	1.452	1.575
.7416	.618	.815	.774	.836	1.077	1.224	1.320	1.369	1.566	1.652
.9856	1.025	1.225	1.165	1.224	1.446	1.588	1.651	1.693	1.781	1.808

Lower surface

x/c	C_p for - $\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0125	.972	.805	.890	1.095	1.290	1.407	1.447	1.516	1.568	1.856
.0250	1.006	.821	.787	.832	.915	.966	1.009	1.063	1.104	1.106
.0500	1.003	.852	.762	.729	.740	.752	.758	.779	.814	.815
.0750	.972	.839	.738	.677	.659	.654	.643	.669	.689	.691
.1000	.966	.849	.747	.674	.640	.630	.607	.618	.627	.629
.1500	.950	.842	.738	.665	.625	.596	.559	.564	.571	.559
.2000	.920	.833	.738	.662	.613	.584	.553	.549	.550	.538
.2500	.895	.818	.738	.652	.613	.587	.538	.537	.533	.523
.3000	.858	.786	.707	.646	.598	.575	.532	.531	.521	.506
.3500	.820	.765	.683	.625	.595	.563	.520	.519	.509	.506
.4000	.771	.728	.665	.604	.568	.544	.505	.501	.494	.494
.4500	.703	.667	.604	.561	.526	.511	.480	.481	.468	.471
.5000	.632	.608	.555	.515	.489	.483	.453	.448	.438	.424
.6201	.421	.407	.387	.354	.353	.342	.314	.328	.325	.324
.6834	.399	.346	.341	.308	.293	.300	.263	.266	.260	.265
.6961	.396	.343	.329	.299	.281	.272	.251	.260	.263	.259
.7087	.387	.336	.335	.305	.278	.275	.260	.263	.263	.262
Vane										
.0250	1.124	1.018	1.009	.930	.870	.826	.743	.722	.686	.659
.0500	.721	.549	.598	.524	.435	.379	.317	.316	.293	.279
.1000	.498	.293	.393	.308	.233	.180	.124	.119	.127	.118
.1500	.390	.182	.293	.223	.145	.107	.073	.084	.077	.071
.2000	.328	.117	.213	.162	.097	.080	.051	.066	.071	.071
.3000	.245	.062	.152	.107	.069	.061	.051	.072	.086	.076
.4000	.192	.046	.098	.082	.060	.067	.063	.090	.095	.088
.5000	.127	.062	.085	.076	.063	.076	.088	.093	.107	.112
.6000	.108	.083	.079	.076	.094	.110	.115	.143	.130	.147
.7000	.099	.123	.101	.113	.133	.150	.136	.155	.166	.176
.8000	.173	.219	.180	.226	.236	.251	.254	.263	.275	.288
.9200	.604	.670	.576	.720	.719	.725	.698	.734	.758	.794
Flap										
.0125	.272	.176	.189	.207	.193	.187	.175	.209	.216	.235
.0250	.043	.022	.043	.040	.045	.049	.042	.072	.074	.082
.0500	.046	.037	.043	.034	.036	.046	.066	.048	.065	.065
.0750	.034	.056	.037	.046	.054	.055	.054	.066	.071	.076
.1000	.056	.074	.061	.058	.054	.073	.066	.078	.077	.079
.1500	.099	.102	.085	.082	.085	.092	.085	.093	.115	.106
.2000	.136	.130	.119	.113	.106	.122	.118	.119	.124	.129
.4000	.297	.278	.268	.241	.236	.239	.208	.224	.222	.218
.6000	.433	.432	.396	.366	.350	.358	.335	.337	.325	.326
.8000	.632	.630	.585	.564	.544	.557	.520	.531	.512	.497
.9000	.808	.799	.765	.744	.734	.734	.695	.713	.704	.671
Spoiler										
.2584	1.372	1.511	1.512	1.572	1.585	1.618	1.651	1.658	1.707	1.717
.5039	1.350	1.520	1.525	1.572	1.588	1.618	1.663	1.676	1.722	1.723
.7508	1.372	1.526	1.531	1.590	1.591	1.624	1.666	1.684	1.743	1.741
.9940	1.401	1.579	1.567	1.628	1.637	1.666	1.707	1.729	1.799	1.791

TABLE XIII.- PRESSURE COEFFICIENT C_p AT THE SIX SPANWISE STATIONS THROUGH THE ANGLE-OF-ATTACK RANGE - Concluded

(f) Double slotted flap configuration with spoiler; $h_s = -0.100\bar{c}$; $\frac{y}{b/2} = 0.72$

Upper surface

x/c	C_p for -									
	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0000	2.079	2.105	2.223	2.192	2.062	1.928	1.846	1.829	1.862	1.897
.0125	1.877	2.077	2.150	2.111	2.003	1.872	1.838	1.838	1.862	1.900
.0250	1.578	2.099	2.174	2.131	2.019	1.907	1.855	1.841	1.865	1.897
.0500	1.432	2.145	2.207	2.172	2.028	1.937	1.870	1.853	1.874	1.897
.0750	1.382	2.194	2.259	2.192	2.046	1.955	1.895	1.864	1.874	1.891
.1000	1.303	2.200	2.244	2.192	2.040	1.931	1.876	1.864	1.874	1.891
.1500	1.306	2.354	2.317	2.212	2.040	1.946	1.897	1.876	1.874	1.900
.2000	1.335	2.514	2.378	2.212	2.037	1.967	1.914	1.876	1.880	1.903
.2500	1.353	2.422	2.473	2.172	2.046	1.967	1.906	1.873	1.880	1.903
.3000	1.357	2.105	2.643	2.172	2.049	1.961	1.897	1.870	1.889	1.918
.3500	1.388	1.779	2.817	2.151	2.046	1.949	1.888	1.873	1.889	1.920
.4000	1.404	1.502	2.894	2.232	2.040	1.949	1.885	1.864	1.895	1.920
.4500	1.423	1.378	2.900	2.313	2.043	1.943	1.876	1.870	1.898	1.923
.5000	1.454	1.391	2.860	2.414	2.062	1.940	1.873	1.861	1.898	1.923
.5500	1.467	1.378	2.665	2.353	2.040	1.928	1.861	1.864	1.898	1.912
.6000	1.454	1.409	2.397	2.293	2.028	1.898	1.858	1.856	1.892	1.920
.6500	1.436	1.391	2.131	2.151	1.982	1.892	1.855	1.859	1.898	1.935
.7000	1.410	1.360	1.961	2.071	1.966	1.875	1.849	1.850	1.910	1.953
.7500	1.376	1.357	1.802	2.030	1.948	1.863	1.846	1.853	1.907	1.956
.8000	1.331	1.305	1.671	1.990	1.914	1.857	1.843	1.856	1.910	1.977
.8500	1.312	1.292	1.546	1.949	1.877	1.830	1.829	1.864	1.943	1.994
.9000	1.297	1.305	1.436	1.848	1.859	1.836	1.840	1.867	1.949	2.009

Lower surface

x/c	C_p for -									
	$\alpha = -4^\circ$	$\alpha = 0^\circ$	$\alpha = 4^\circ$	$\alpha = 8^\circ$	$\alpha = 12^\circ$	$\alpha = 16^\circ$	$\alpha = 20^\circ$	$\alpha = 22^\circ$	$\alpha = 24^\circ$	$\alpha = 26^\circ$
Wing										
.0125	.760	.812	.954	1.061	1.169	1.209	1.263	1.292	1.365	1.419
.0250	.782	.751	.814	.879	.957	.991	1.047	1.074	1.129	1.177
.0500	.792	.723	.726	.778	.788	.803	.837	.858	.904	.947
.0750	.776	.729	.698	.717	.726	.731	.755	.773	.799	.835
.1000	.773	.726	.695	.697	.695	.693	.707	.720	.746	.770
.1500	.773	.729	.692	.677	.665	.651	.654	.664	.683	.699
.2000	.782	.732	.704	.677	.668	.648	.654	.658	.662	.690
.2500	.786	.732	.698	.697	.671	.657	.660	.661	.674	.690
.3000	.792	.751	.713	.717	.695	.678	.681	.690	.698	.711
.3500	.823	.797	.774	.759	.760	.734	.737	.743	.763	.767
.4000	.909	.868	.851	.859	.840	.821	.823	.829	.844	.870
.4500	1.003	.966	.960	.980	.945	.931	.935	.932	.955	.976
.5000	1.107	1.071	1.070	1.081	1.046	1.042	1.056	1.062	1.084	1.100
.5500	1.183	1.172	1.180	1.222	1.191	1.179	1.213	1.218	1.260	1.283
.6000	1.253	1.265	1.271	1.323	1.326	1.340	1.385	1.410	1.470	1.505
.6500	1.328	1.357	1.384	1.444	1.468	1.513	1.571	1.623	1.683	1.723
.7000	1.363	1.397	1.451	1.525	1.572	1.609	1.666	1.705	1.760	1.800
.7500	1.366	1.415	1.494	1.566	1.585	1.609	1.648	1.687	1.748	1.791
.8000	1.335	1.397	1.476	1.505	1.517	1.534	1.577	1.608	1.680	1.732
.8500	1.309	1.382	1.494	1.545	1.477	1.495	1.530	1.561	1.632	1.684
.9000	1.316	1.397	1.515	1.545	1.477	1.475	1.506	1.534	1.608	1.658
.9500	1.335	1.403	1.508	1.586	1.529	1.519	1.548	1.569	1.644	1.684

TABLE XIV.- INTEGRATED SECTION DATA FOR THE PLAIN WING CONFIGURATION WITH SPOILERS

(a) $h_s = -0.005\bar{c}$

$\frac{y}{b/2}$	α, deg	$c_{n,w}$	$c_{m,w}$	$c_{n,s}$	$c_{m,s}$
0.21	-4	-.1504	.0206	.0536	.0168
0.21	0	-.0285	.0128	.0946	-.0040
0.21	4	.1112	.0024	.0997	-.0116
0.21	8	.2968	-.0069	.0913	-.0092
0.21	12	.5405	.0003	.0715	-.0040
0.21	16	.8248	.0058	.1221	-.0164
0.21	20	1.1683	.0130	.0922	.0006
0.21	22	1.3329	.0127	.0819	.0006
0.21	24	1.4502	-.0174	.0869	.0009
0.21	26	1.3531	-.0965	.1008	-.0114
0.30	-4	-.2057	.0329	.0664	.0100
0.30	0	-.0643	.0306	.0835	.0015
0.30	4	.0959	.0251	.0768	.0002
0.30	8	.3206	.0336	.0801	-.0043
0.30	12	.6042	.0358	.0700	-.0044
0.30	16	.9516	.0232	.0887	.0000
0.30	20	1.3263	-.0328	.1425	-.0236
0.30	22	1.4708	-.0675	.1419	-.0258
0.30	24	1.5753	-.1102	.1367	-.0239
0.30	26	1.4951	-.1664	.1238	-.0184
0.43	-4	-.2332	.0132	.1006	-.0050
0.43	0	-.0471	.0078	.1180	-.0070
0.43	4	.1324	.0100	.1116	-.0106
0.43	8	.4037	.0109	.1152	-.0112
0.43	12	.7708	-.0122	.1232	-.0156
0 .3	16	1.1398	-.0836	.0945	-.0100
(43	20	1.3795	-.1673	.1259	-.0248
(43	22	1.4624	-.1967	.1516	-.0422
0 43	24	1.5166	-.2248	.1370	-.0398
0 43	26	1.4793	-.2426	.1340	-.0418
0.55	-4	-.2979	.0306	.0743	-.0037
0.55	0	-.0758	.0243	.0683	.0099
0.55	4	.1289	.0189	.0700	.0069
0.55	8	.5023	.0028	.0626	.0169
0.55	12	.9042	-.0973	.0183	.0369
0.55	16	1.0851	-.1641	.0589	.0030
0.55	20	1.1754	-.1872	.0887	-.0217
0.55	22	1.2311	-.2029	.1065	-.0302
0.55	24	1.2634	-.2180	.1154	-.0378
0.55	26	1.2222	-.2198	.0965	-.0329
0.72	-4	-.2353	-.0083		
0.72	0	-.0244	-.0034		
0.72	4	.2458	-.0067		
0.72	8	.6912	-.1405		
0.72	12	.7332	-.1410		
0.72	16	.8281	-.1527		
0.72	20	.8879	-.1667		
0.72	22	.9031	-.1706		
0.72	24	.9469	-.1796		
0.72	26	.9504	-.1840		

(b) $h_s = -0.010\bar{c}$

$\frac{y}{b/2}$	α, deg	$c_{n,w}$	$c_{m,w}$	$c_{n,s}$	$c_{m,s}$
0.21	-4	-.1940	.0435	-.2783	.1414
0.21	0	-.0581	.0319	-.2944	.1368
0.21	4	.0877	.0211	-.2755	.1199
0.21	8	.2752	.0140	-.2338	.1033
0.21	12	.5112	.0205	-.1992	.0873
0.21	16	.8111	.0280	-.2525	.1164
0.21	20	1.1321	.0356	-.3025	.1405
0.21	22	1.2909	.0374	-.3330	.1542
0.21	24	1.3838	.0011	-.3306	.1541
0.21	26	1.3296	-.0750	-.2532	.1206
0.30	-4	-.2384	.0543	-.2129	.0967
0.30	0	-.1076	.0527	-.2694	.1117
0.30	4	.0520	.0521	-.2536	.1036
0.30	8	.2915	.0589	-.2265	.0915
0.30	12	.5611	.0595	-.1866	.0755
0.30	16	.9251	.0513	-.2285	.0949
0.30	20	1.2753	-.0053	-.1760	.0729
0.30	22	1.4127	-.0434	-.1365	.0573
0.30	24	1.5022	-.0881	-.1344	.0604
0.30	26	1.4756	-.1446	-.0677	.0324
0.43	-4	-.2552	.0280	-.1834	.0864
0.43	0	-.0898	.0308	-.2810	.1242
0.43	4	.1103	.0284	-.2741	.1190
0.43	8	.3798	.0369	-.2705	.1163
0.43	12	.7227	.0226	-.2782	.1198
0.43	16	1.1200	-.0615	-.2202	.0947
0.43	20	1.3316	-.1451	-.1363	.0645
0.43	22	1.4089	-.1761	-.0659	.0409
0.43	24	1.4667	-.2117	-.0017	.0126
0.43	26	1.4934	-.2469	.0420	-.0019
0.55	-4	-.3460	.0489	-.1702	.0635
0.55	0	-.1317	.0491	-.2466	.0982
0.55	4	.0797	.0431	-.2347	.0979
0.55	8	.4620	.0323	-.2865	.1182
0.55	12	.8435	-.0650	-.3364	.1532
0.55	16	1.0572	-.1413	-.1847	.0894
0.55	20	1.1406	-.1762	-.0406	.0276
0.55	22	1.1927	-.1896	-.0200	.0217
0.55	24	1.2133	-.2053	-.0191	.0261
0.55	26	1.2256	-.2222	-.0414	.0409
0.72	-4	-.2412	-.0067		
0.72	0	-.0503	.0013		
0.72	4	.2060	-.0021		
0.72	8	.6480	-.1298		
0.72	12	.6927	-.1308		
0.72	16	.7992	-.1441		
0.72	20	.8740	-.1622		
0.72	22	.9057	-.1676		
0.72	24	.9319	-.1734		
0.72	26	.9096	-.1766		

TABLE XIV.- INTEGRATED SECTION DATA FOR THE PLAIN WING CONFIGURATION WITH SPOILERS - Continued

(c) $h_s = -0.035\bar{c}$

$\frac{y}{b/2}$	α, deg	$c_{n,w}$	$c_{m,w}$	$c_{n,s}$	$c_{m,s}$
0.21	-4	-.2301	.0521	-.6279	.2771
0.21	0	-.1162	.0547	-.6355	.2681
0.21	4	.0264	.0577	-.6384	.2665
0.21	8	.1956	.0556	-.6731	.2819
0.21	12	.4229	.0610	-.6812	.2867
0.21	16	.6868	.0723	-.7350	.3151
0.21	20	1.0372	.0747	-.8485	.3711
0.21	22	1.1876	.0759	-.9209	.4095
0.21	24	1.2801	.0174	-.8676	.3873
0.21	26	1.2859	-.0660	-.6964	.3203
0.30	-4	-.3569	.1099	-.5872	.2379
0.30	0	-.2127	.1057	-.6316	.2503
0.30	4	-.0438	.1044	-.6242	.2455
0.30	8	.1918	.1113	-.6549	.2614
0.30	12	.4573	.1069	-.6797	.2763
0.30	16	.7812	.0997	-.7100	.2910
0.30	20	1.1572	.0383	-.6731	.2732
0.30	22	1.2909	-.0059	-.6138	.2566
0.30	24	1.3997	-.0638	-.5921	.2518
0.30	26	1.3925	-.1265	-.4353	.1949
0.43	-4	-.3951	.0845	-.5361	.2180
0.43	0	-.2138	.0845	-.6068	.2419
0.43	4	-.0082	.0834	-.6054	.2383
0.43	8	.2615	.0918	-.6573	.2598
0.43	12	.5931	.0711	-.7588	.3063
0.43	16	.9441	-.0104	-.7286	.3025
0.43	20	1.1863	-.0944	-.6109	.2528
0.43	22	1.2879	-.1411	-.3876	.1592
0.43	24	1.3621	-.1878	-.2694	.1131
0.43	26	1.4132	-.2267	-.1348	.0634
0.55	-4	-.5228	.1175	-.5353	.2071
0.55	0	-.2797	.1132	-.6100	.2434
0.55	4	-.0618	.1075	-.6141	.2484
0.55	8	.3069	.0978	-.7095	.2888
0.55	12	.6404	.0066	-.7500	.3129
0.55	16	.8574	-.0786	-.5012	.2142
0.55	20	.9988	-.1228	-.3031	.1245
0.55	22	1.0544	-.1455	-.1883	.0744
0.55	24	1.1121	-.1758	-.1155	.0470
0.55	26	1.1704	-.2009	-.0284	.0138
0.72	-4	-.3671	.0061		
0.72	0	-.1319	.0154		
0.72	4	.1486	.0098		
0.72	8	.4954	-.0967		
0.72	12	.5808	-.1075		
0.72	16	.6831	-.1229		
0.72	20	.7833	-.1425		
0.72	22	.8258	-.1503		
0.72	24	.8465	-.1581		
0.72	26	.8603	-.1690		

(d) $h_s = -0.050\bar{c}$

$\frac{y}{b/2}$	α, deg	$c_{n,w}$	$c_{m,w}$	$c_{n,s}$	$c_{m,s}$
0.21	-4	-.2888	.0987	-.7428	.3243
0.21	0	-.1788	.0918	-.7679	.3230
0.21	4	-.0306	.0817	-.7660	.3160
0.21	8	.1437	.0752	-.8149	.3384
0.21	12	.3817	.0745	-.8224	.3478
0.21	16	.6461	.0876	-.9058	.3829
0.21	20	1.0030	.0907	-1.0400	.4541
0.21	22	1.1539	.0865	-1.0713	.4779
0.21	24	1.2275	.0322	-1.0749	.4811
0.21	26	1.1906	-.0721	-.9112	.4214
0.30	-4	-.3697	.1200	-.7245	.3014
0.30	0	-.2660	.1310	-.7980	.3213
0.30	4	-.0916	.1203	-.7853	.3160
0.30	8	.1203	.1295	-.8564	.3472
0.30	12	.4084	.1259	-.8335	.3425
0.30	16	.7371	.1182	-.8884	.3646
0.30	20	1.1007	.0566	-.8279	.3331
0.30	22	1.2319	.0131	-.7552	.3119
0.30	24	1.3369	-.0439	-.7482	.3166
0.30	26	1.2360	-.1141	-.5828	.2547
0.43	-4	-.4346	.1048	-.6828	.2836
0.43	0	-.2656	.1107	-.7813	.3144
0.43	4	-.0690	.1065	-.7890	.3156
0.43	8	.1960	.1123	-.8483	.3389
0.43	12	.5263	.0958	-.9088	.3667
0.43	16	.8986	.0083	-.8938	.3736
0.43	20	1.1313	-.0768	-.7223	.2977
0.43	22	1.2314	-.1221	-.5268	.2161
0.43	24	1.3081	-.1674	-.3939	.1622
0.43	26	1.2690	-.2016	-.2148	.0940
0.55	-4	-.5662	.1358	-.6785	.2680
0.55	0	-.3544	.1441	-.8107	.3238
0.55	4	-.1276	.1272	-.7839	.3199
0.55	8	.2226	.1221	-.9040	.3696
0.55	12	.5869	.0227	-.8087	.3385
0.55	16	.8212	-.0641	-.6101	.2602
0.55	20	.9410	-.1044	-.3988	.1658
0.55	22	1.0095	-.1291	-.2985	.1228
0.55	24	1.0692	-.1579	-.2347	.0987
0.55	26	1.0559	-.1782	-.1060	.0468
0.72	-4	-.4121	.0037		
0.72	0	-.1855	.0188		
0.72	4	.0898	.0138		
0.72	8	.4506	-.0950		
0.72	12	.5287	-.1004		
0.72	16	.6401	-.1185		
0.72	20	.7498	-.1378		
0.72	22	.7627	-.1371		
0.72	24	.8122	-.1500		
0.72	26	.8442	-.1656		

TABLE XIV.- INTEGRATED SECTION DATA FOR THE PLAIN WING CONFIGURATION WITH SPOILERS - Concluded

(e) $h_s = -0.100\bar{c}$

$\frac{y}{b/2}$	α, deg	$c_{n,w}$	$c_{m,w}$	$c_{n,s}$	$c_{m,s}$
0.21	-4	-.3366	.1057	-1.0865	.5536
0.21	0	-.2417	.1068	-1.0201	.5177
0.21	4	-.1004	.0991	-.8430	.4165
0.21	8	.0799	.0842	-.7946	.3860
0.21	12	.3274	.0787	-.8843	.4252
0.21	16	.6031	.0873	-1.0149	.4943
0.21	20	.9556	.0726	-1.2219	.5932
0.21	22	1.1394	.0573	-1.2751	.6390
0.21	24	1.1593	-.0284	-1.2244	.6245
0.21	26	1.2192	-.1362	-1.1827	.6029
0.30	-4	-.4522	.1350	-1.2492	.6183
0.30	0	-.3417	.1475	-1.2057	.6012
0.30	4	-.1741	.1398	-1.0644	.5327
0.30	8	.0487	.1389	-1.1048	.5592
0.30	12	.3559	.1235	-1.2999	.6367
0.30	16	.6592	.1240	-1.2872	.6311
0.30	20	.9673	.0622	-1.2107	.5641
0.30	22	1.1348	.0133	-1.0849	.5171
0.30	24	1.2268	-.0670	-.8685	.4259
0.30	26	1.2097	-.1482	-.7070	.3487
0.43	-4	-.5344	.1130	-1.1704	.5793
0.43	0	-.3507	.1202	-1.1515	.5726
0.43	4	-.1639	.1202	-1.1395	.5652
0.43	8	.0841	.1196	-1.1415	.5679
0.43	12	.4654	.0645	-1.2344	.6097
0.43	16	.7800	-.0065	-1.1310	.5573
0.43	20	.9521	-.0597	-.9023	.4214
0.43	22	1.0825	-.0986	-.7070	.3352
0.43	24	1.1769	-.1569	-.4150	.2094
0.43	26	1.2262	-.2057	-.2510	.1335
0.55	-4	-.6972	.1393	-1.2871	.6394
0.55	0	-.4712	.1531	-1.2883	.6405
0.55	4	-.2697	.1437	-1.2301	.6133
0.55	8	.0925	.1071	-1.2981	.6494
0.55	12	.4003	.0134	-.8996	.4460
0.55	16	.6310	-.0427	-.6083	.2960
0.55	20	.7327	-.0595	-.5086	.2377
0.55	22	.8196	-.0836	-.4217	.2056
0.55	24	.9156	-.1265	-.2095	.1144
0.55	26	1.0040	-.1627	-.0862	.0550
0.72	-4	-.5809	-.0336		
0.72	0	-.3485	-.0114		
0.72	4	-.0670	-.0090		
0.72	8	.1192	-.0499		
0.72	12	.2673	-.0589		
0.72	16	.3607	-.0716		
0.72	20	.5173	-.0894		
0.72	22	.5838	-.1005		
0.72	24	.6635	-.1184		
0.72	26	.7452	-.1360		

TABLE XV.- INTEGRATED SECTION DATA FOR THE DOUBLE SLOTTED FLAP CONFIGURATION WITH SPOILERS

(a) $h_s = -0.005\bar{c}$

$\frac{y}{b/2}$	α, deg	$c_{n,w}$	$c_{m,w}$	$c_{n,v}$	$c_{m,v}$	$c_{n,f}$	$c_{m,f}$	$c_{n,s}$	$c_{m,s}$
0.21	-4	.5357	-.2699	2.2330	-.9652	1.2170	-.4799	-.0547	.0111
0.21	0	.6811	-.2816	2.2812	-.9786	1.2241	-.4742	-.0423	.0083
0.21	4	.8312	-.2830	2.2851	-.9778	1.2048	-.4658	-.0200	-.0020
0.21	8	.9772	-.2724	2.3015	-.9814	1.1998	-.4676	-.0111	-.0031
0.21	12	1.2117	-.2652	2.6108	-1.1090	1.2818	-.4999	.0113	-.0135
0.21	16	1.4687	-.2583	2.8901	-1.2035	1.3962	-.5461	.0248	-.0206
0.21	20	1.7985	-.2582	3.2080	-1.3563	1.5545	-.6074	.0306	-.0254
0.21	22	1.8144	-.2874	3.3732	-1.4513	1.6536	-.6352	.0499	-.0371
0.21	24	1.6495	-.3412	2.9937	-1.3063	1.5703	-.6164	.0428	-.0457
0.21	26	1.6744	-.3876	2.6177	-1.1574	1.5588	-.6288	.0226	-.0400
0.30	-4	.7212	-.3641	4.0150	-1.5954	1.3751	-.5461	-.0475	-.0028
0.30	0	.8461	-.3632	3.8283	-1.5225	1.3838	-.5567	-.0336	-.0066
0.30	4	1.0415	-.3488	3.6699	-1.4540	1.3788	-.5601	-.0256	-.0045
0.30	8	1.2397	-.3239	3.6559	-1.4312	1.3310	-.5302	-.0013	-.0110
0.30	12	1.4449	-.3073	3.6106	-1.4187	1.3502	-.5320	.0134	-.0155
0.30	16	1.7442	-.3263	3.9773	-1.6010	1.4624	-.5651	.0320	-.0305
0.30	20	1.9489	-.3610	4.0247	-1.6302	1.5225	-.5836	.0336	-.0368
0.30	22	2.0192	-.3886	3.9243	-1.6155	1.5663	-.5956	.0061	-.0325
0.30	24	1.8173	-.3926	3.1353	-1.3403	1.4963	-.5675	-.0345	-.0217
0.30	26	1.6296	-.3915	2.4111	-1.0739	1.4135	-.5410	-.0569	-.0073
0.43	-4	.8998	-.4138	4.9685	-2.0027	1.3670	-.4569	-.2012	.0527
0.43	0	1.1475	-.4437	5.0297	-2.0283	1.5191	-.5330	-.1983	.0533
0.43	4	1.3359	-.4187	4.9931	-2.0165	1.5084	-.5344	-.1638	.0399
0.43	8	1.5538	-.3981	4.9858	-2.0157	1.5166	-.5393	-.0805	.0147
0.43	12	1.8738	-.4123	4.8640	-1.9796	1.5920	-.5855	-.0254	-.0042
0.43	16	1.9661	-.4367	4.1078	-1.7035	1.5870	-.5953	.0121	-.0229
0.43	20	1.9605	-.4483	3.6850	-1.5472	1.6559	-.6436	.0163	-.0256
0.43	22	1.7689	-.4167	2.9497	-1.2558	1.5617	-.6270	.0004	-.0187
0.43	24	1.6640	-.4051	2.2966	-1.0076	1.5233	-.6326	-.0176	-.0091
0.43	26	1.6556	-.4264	2.1596	-.9687	1.6210	-.6877	-.0097	-.0102
0.55	-4	1.1603	-.5517	5.0218	-2.0514	1.8061	-.7200	-.1040	.0155
0.55	0	1.3413	-.5471	4.9493	-2.0198	1.8290	-.7338	-.1109	.0181
0.55	4	1.6827	-.5349	5.0699	-2.0828	1.8570	-.7469	-.1045	.0160
0.55	8	1.9694	-.5393	4.7364	-1.9722	1.8548	-.7325	.0512	-.0345
0.55	12	1.9315	-.5238	3.5293	-1.4850	1.7485	-.7078	.1498	-.0688
0.55	16	1.7765	-.4769	2.8320	-1.2046	1.6262	-.6680	.1424	-.0610
0.55	20	1.6225	-.4176	2.5168	-1.0864	1.3700	-.5084	.1337	-.0594
0.55	22	1.6386	-.4465	2.4146	-1.0434	1.5368	-.6405	.1298	-.0592
0.55	24	1.5573	-.4280	2.0758	-.9151	1.4890	-.6251	.0958	-.0469
0.55	26	1.4952	-.4232	1.9677	-.8807	1.4923	-.6336	.0886	-.0422
0.72	-4	.1044	.1359						
0.72	0	.3511	.1698						
0.72	4	.9565	-.0176						
0.72	8	.8967	-.0697						
0.72	12	.8697	-.0789						
0.72	16	.7708	-.0657						
0.72	20	.7405	-.0633						
0.72	22	.7069	-.0587						
0.72	24	.6977	-.0681						
0.72	26	.7126	-.0756						

TABLE XV.- INTEGRATED SECTION DATA FOR THE DOUBLE SLOTTED FLAP CONFIGURATION WITH SPOILERS - Continued

(b) $h_S = -0.010\bar{c}$

$\frac{y}{b/2}$	α, deg	$c_{n,w}$	$c_{m,w}$	$c_{n,v}$	$c_{m,v}$	$c_{n,f}$	$c_{m,f}$	$c_{n,s}$	$c_{m,s}$
0.21	-4	.5396	-.2726	2.3037	-.9915	1.2367	-.4880	-.0726	.0617
0.21	0	.6920	-.2869	2.2519	-.9662	1.2933	-.5272	.0105	.0094
0.21	4	.8184	-.2840	2.2464	-.9609	1.2700	-.5237	.0333	-.0065
0.21	8	1.0255	-.2839	2.3595	-1.0091	1.3379	-.5619	.0261	-.0031
0.21	12	1.2070	-.2658	2.5069	-1.0650	1.3510	-.5673	.0359	-.0064
0.21	16	1.4505	-.2645	2.8190	-1.1782	1.4759	-.6142	.0551	-.0094
0.21	20	1.7494	-.2623	3.0797	-1.3044	1.6271	-.6790	.1676	-.0641
0.21	22	1.8587	-.2854	3.2142	-1.3741	1.7271	-.7264	.2822	-.1168
0.21	24	1.6801	-.3471	2.9916	-1.3037	1.6936	-.7206	.2740	-.1071
0.21	26	1.6119	-.3778	2.4780	-1.0979	1.6456	-.7279	.2118	-.0786
0.30	-4	.7122	-.3609	3.9093	-1.5505	1.4087	-.5813	.0184	.0044
0.30	0	.8424	-.3631	3.7110	-1.4739	1.4494	-.6107	.0418	-.0162
0.30	4	1.0127	-.3488	3.5368	-1.4023	1.4506	-.6203	.0330	-.0134
0.30	8	1.2314	-.3291	3.5525	-1.3928	1.4686	-.6342	.0376	-.0154
0.30	12	1.4334	-.3108	3.4217	-1.3458	1.4975	-.6498	.0438	-.0174
0.30	16	1.7856	-.3465	3.9782	-1.5982	1.6834	-.7169	.0504	-.0175
0.30	20	1.9621	-.3757	3.9781	-1.6078	1.7236	-.7354	.0877	-.0300
0.30	22	2.0001	-.4005	3.8362	-1.5744	1.7611	-.7460	.1224	-.0417
0.30	24	1.8795	-.4189	3.2543	-1.3842	1.7651	-.7539	.1234	-.0454
0.30	26	1.6377	-.4108	2.4156	-1.0689	1.6880	-.7384	.1082	-.0434
0.43	-4	.9466	-.4378	5.0566	-2.0360	1.4775	-.5101	.0476	.0008
0.43	0	1.1436	-.4403	4.9628	-2.0018	1.5206	-.5428	.0253	.0095
0.43	4	1.3424	-.4217	4.9701	-2.0069	1.5237	-.5464	.0222	.0077
0.43	8	1.6307	-.4156	5.1164	-2.0672	1.6422	-.6115	.0121	.0121
0.43	12	1.8345	-.4034	4.6951	-1.9110	1.5736	-.5922	.0039	.0118
0.43	16	1.9612	-.4358	4.1501	-1.7166	1.6162	-.6206	.0697	-.0149
0.43	20	1.9340	-.4449	3.6485	-1.5307	1.6731	-.6629	.1693	-.0580
0.43	22	1.8617	-.4406	3.1372	-1.3331	1.6726	-.6870	.1942	-.0638
0.43	24	1.7227	-.4200	2.3210	-1.0217	1.6264	-.6986	.1488	-.0498
0.43	26	1.5878	-.4141	2.0769	-.9292	1.6268	-.7128	.1244	-.0420
0.55	-4	1.1260	-.5247	4.9090	-2.0045	1.6443	-.6100	.1541	-.0557
0.55	0	1.3269	-.5291	4.9129	-1.9984	1.7219	-.6463	.1982	-.0609
0.55	4	1.6278	-.5130	4.9998	-2.0474	1.7240	-.6463	.2254	-.0693
0.55	8	1.9387	-.5215	4.6851	-1.9498	1.7464	-.6558	.2303	-.0765
0.55	12	1.8679	-.5023	3.3770	-1.4208	1.6603	-.6640	.2224	-.0780
0.55	16	1.7551	-.4708	2.7523	-1.1735	1.6076	-.6666	.2080	-.0730
0.55	20	1.6542	-.4492	2.5143	-1.0803	1.5488	-.6501	.1935	-.0690
0.55	22	1.6014	-.4423	2.3515	-1.0187	1.5436	-.6543	.1909	-.0698
0.55	24	1.5821	-.4413	2.1393	-.9419	1.5645	-.6747	.1327	-.0512
0.55	26	1.4912	-.4353	1.9586	-.8789	1.5771	-.6978	.1163	-.0420
0.72	-4	.1266	.0899						
0.72	0	.3916	.1711						
0.72	4	1.0448	-.0392						
0.72	8	.9284	-.0669						
0.72	12	.8369	-.0763						
0.72	16	.7864	-.0668						
0.72	20	.7398	-.0645						
0.72	22	.7145	-.0604						
0.72	24	.6947	-.0691						
0.72	26	.7072	-.0747						

TABLE XV.- INTEGRATED SECTION DATA FOR THE DOUBLE SLOTTED FLAP CONFIGURATION WITH SPOILERS - Continued

(c) $h_s = -0.035\bar{c}$

$\frac{y}{b/2}$	α, deg	$c_{n,w}$	$c_{m,w}$	$c_{n,v}$	$c_{m,v}$	$c_{n,f}$	$c_{m,f}$	$c_{n,s}$	$c_{m,s}$
0.21	-4	.5505	-.2808	2.2689	-.9856	1.4254	-.5630	-.1153	.0853
0.21	0	.6524	-.2764	2.1449	-.9219	1.3884	-.5571	-.0310	.0269
0.21	4	.8019	-.2826	2.1572	-.9260	1.4010	-.5655	-.0385	.0267
0.21	8	.9493	-.2726	2.1964	-.9393	1.3854	-.5632	-.0476	.0325
0.21	12	1.1800	-.2620	2.4254	-1.0339	1.4412	-.5831	-.0340	.0281
0.21	16	1.4413	-.2560	2.7165	-1.1302	1.5661	-.6389	.0041	.0106
0.21	20	1.7350	-.2571	2.9624	-1.2520	1.7026	-.6921	.0768	-.0238
0.21	22	1.8602	-.2741	3.1024	-1.3208	1.7936	-.7265	.0958	-.0283
0.21	24	1.7059	-.3409	3.0118	-1.3074	1.7919	-.7323	.0649	-.0134
0.21	26	1.5724	-.3697	2.3728	-1.0497	1.6697	-.6992	-.0063	.0149
0.30	-4	.6875	-.3531	3.8689	-1.5358	1.3621	-.5644	-.0898	.0510
0.30	0	.8285	-.3509	3.5851	-1.4232	1.3889	-.5859	-.0626	.0295
0.30	4	1.0088	-.3388	3.4561	-1.3693	1.3944	-.5961	-.0518	.0261
0.30	8	1.1946	-.3181	3.4408	-1.3471	1.4127	-.6122	-.0411	.0226
0.30	12	1.4249	-.3063	3.3857	-1.3246	1.4623	-.6336	-.0350	.0196
0.30	16	1.7507	-.3285	3.8170	-1.5308	1.5746	-.6708	-.0361	.0147
0.30	20	1.9609	-.3631	3.8859	-1.5619	1.6201	-.6856	-.0314	.0101
0.30	22	2.0102	-.3938	3.7934	-1.5534	1.7011	-.7193	-.0289	.0084
0.30	24	1.8357	-.4034	3.0472	-1.3056	1.6428	-.6997	-.0570	.0188
0.30	26	1.6699	-.4063	2.4936	-1.1014	1.6532	-.7236	-.0516	.0175
0.43	-4	.8342	-.3787	4.7447	-1.8962	1.1797	-.3976	-.3399	.1560
0.43	0	1.0131	-.3893	4.5118	-1.8071	1.2923	-.4712	-.3197	.1387
0.43	4	1.2463	-.3925	4.6551	-1.8698	1.3847	-.5212	-.2857	.1217
0.43	8	1.4954	-.3756	4.6940	-1.8852	1.4244	-.5394	-.2289	.0945
0.43	12	1.7777	-.3825	4.5501	-1.8413	1.4675	-.5693	-.1750	.0708
0.43	16	1.9291	-.4212	4.0118	-1.6543	1.5335	-.6110	-.1424	.0561
0.43	20	1.9152	-.4329	3.6388	-1.5201	1.5839	-.6426	-.1066	.0428
0.43	22	1.8794	-.4366	3.2789	-1.3852	1.6118	-.6664	-.0879	.0382
0.43	24	1.7470	-.4212	2.4295	-1.0618	1.6089	-.6876	-.0529	.0248
0.43	26	1.5737	-.4119	2.0441	-.9159	1.6070	-.7064	-.0166	.0144
0.55	-4	1.0285	-.4621	4.6290	-1.8820	1.3346	-.4985	-.3674	.1460
0.55	0	1.2367	-.4591	4.5696	-1.8433	1.3791	-.5127	-.3982	.1651
0.55	4	1.5531	-.4466	4.7380	-1.9251	1.3963	-.5112	-.3996	.1728
0.55	8	1.8381	-.4701	4.4783	-1.8537	1.4897	-.5606	-.2678	.1090
0.55	12	1.8577	-.4875	3.4318	-1.4419	1.5580	-.6215	-.1036	.0361
0.55	16	1.7245	-.4579	2.7439	-1.1677	1.5380	-.6396	-.0631	.0239
0.55	20	1.6415	-.4394	2.5019	-1.0733	1.4927	-.6328	-.0476	.0195
0.55	22	1.6327	-.4484	2.4065	-1.0404	1.5483	-.6652	-.0312	.0145
0.55	24	1.5491	-.4299	2.0325	-.8977	1.5289	-.6690	-.0308	.0129
0.55	26	1.5005	-.4313	1.9724	-.8789	1.5504	-.6847	-.0222	.0106
0.72	-4	.2116	.0775						
0.72	0	.4878	.1222						
0.72	4	1.1126	-.0750						
0.72	8	.8889	-.0732						
0.72	12	.8208	-.0730						
0.72	16	.7578	-.0644						
0.72	20	.7352	-.0632						
0.72	22	.7246	-.0630						
0.72	24	.6957	-.0672						
0.72	26	.7401	-.0820						

TABLE XV.- INTEGRATED SECTION DATA FOR THE DOUBLE SLOTTED FLAP CONFIGURATION WITH SPOILERS - Continued

(d) $h_s = -0.050\bar{c}$

$\frac{y}{b/2}$	α, deg	$c_{n,w}$	$c_{m,w}$	$c_{n,v}$	$c_{m,v}$	$c_{n,f}$	$c_{m,f}$	$c_{n,s}$	$c_{m,s}$
0.21	-4	.5178	-.2646	2.2831	-.9814	1.2484	-.5078	-.1788	.1269
0.21	0	.6466	-.2733	2.2018	-.9456	1.2691	-.5270	-.1005	.0707
0.21	4	.8012	-.2732	2.1308	-.9134	1.2677	-.5351	-.0719	.0537
0.21	8	.9584	-.2671	2.2222	-.9459	1.2817	-.5474	-.0605	.0483
0.21	12	1.1686	-.2581	2.4243	-1.0301	1.3374	-.5713	-.0237	.0307
0.21	16	1.4396	-.2570	2.7090	-1.1286	1.4664	-.6312	.0204	.0096
0.21	20	1.7139	-.2512	2.8935	-1.2254	1.5817	-.6772	.0724	-.0151
0.21	22	1.8270	-.2715	3.0104	-1.2853	1.6605	-.7094	.0823	-.0159
0.21	24	1.6719	-.3330	2.9226	-1.2683	1.6326	-.7023	.0295	.0088
0.21	26	1.5884	-.3644	2.4851	-1.1002	1.5747	-.6909	-.0293	.0346
0.30	-4	.6686	-.3445	3.9065	-1.5517	1.2992	-.5282	-.1872	.0911
0.30	0	.8209	-.3482	3.7057	-1.4710	1.3293	-.5472	-.1696	.0742
0.30	4	.9934	-.3347	3.5219	-1.3940	1.3393	-.5579	-.1392	.0623
0.30	8	1.1885	-.3167	3.5129	-1.3761	1.3725	-.5809	-.1055	.0515
0.30	12	1.3913	-.2987	3.4022	-1.3335	1.4069	-.5981	-.0741	.0360
0.30	16	1.7388	-.3205	3.8301	-1.5369	1.5010	-.6297	-.0990	.0375
0.30	20	1.9793	-.3629	3.9977	-1.6069	1.5890	-.6616	-.1069	.0358
0.30	22	1.9765	-.3783	3.7441	-1.5340	1.5776	-.6541	-.1305	.0448
0.30	24	1.8183	-.3967	3.0462	-1.3037	1.5938	-.6723	-.1441	.0490
0.30	26	1.6369	-.4018	2.4154	-1.0718	1.5943	-.6922	-.1087	.0379
0.43	-4	.8389	-.3830	4.5013	-1.7916	1.2235	-.4478	-.4360	.1896
0.43	0	.9861	-.3722	4.4007	-1.7544	1.2035	-.4361	-.4762	.1962
0.43	4	1.2328	-.3558	4.4320	-1.7730	1.2348	-.4534	-.4622	.1880
0.43	8	1.4691	-.3486	4.5868	-1.8408	1.3038	-.4840	-.3941	.1546
0.43	12	1.7400	-.3628	4.4166	-1.7869	1.3456	-.5112	-.3193	.1182
0.43	16	1.9030	-.4046	3.9608	-1.6321	1.4388	-.5639	-.2642	.0961
0.43	20	1.8514	-.4079	3.5357	-1.4749	1.4423	-.5749	-.2218	.0783
0.43	22	1.8147	-.4148	3.1110	-1.3167	1.5067	-.6180	-.1796	.0637
0.43	24	1.7016	-.4064	2.3963	-1.0468	1.5210	-.6461	-.1078	.0394
0.43	26	1.5783	-.4051	2.0876	-.9313	1.5645	-.6821	-.0602	.0264
0.55	-4	.9807	-.4378	4.4422	-1.7999	1.2333	-.4669	-.4083	.1513
0.55	0	1.1854	-.4311	4.4263	-1.7742	1.2367	-.4532	-.5077	.1964
0.55	4	1.4767	-.4150	4.5459	-1.8386	1.2507	-.4549	-.5248	.2072
0.55	8	1.7774	-.4398	4.3381	-1.7884	1.3484	-.5055	-.3654	.1342
0.55	12	1.7876	-.4566	3.3426	-1.4040	1.4240	-.5620	-.1884	.0627
0.55	16	1.6932	-.4388	2.7134	-1.1563	1.4437	-.5974	-.1195	.0401
0.55	20	1.6580	-.4389	2.5544	-1.0955	1.4671	-.6170	-.0934	.0314
0.55	22	1.5855	-.4274	2.3391	-1.0101	1.4512	-.6180	-.0878	.0305
0.55	24	1.5550	-.4287	2.0646	-.9097	1.5083	-.6569	-.0591	.0213
0.55	26	1.4939	-.4258	1.9636	-.8743	1.5188	-.6693	-.0443	.0179
0.72	-4	.2411	.0540						
0.72	0	.5075	.1022						
0.72	4	1.1134	-.0739						
0.72	8	.8843	-.0733						
0.72	12	.8070	-.0666						
0.72	16	.7559	-.0613						
0.72	20	.7428	-.0644						
0.72	22	.7070	-.0598						
0.72	24	.6971	-.0686						
0.72	26	.7209	-.0776						

TABLE XV.- INTEGRATED SECTION DATA FOR THE DOUBLE SLOTTED FLAP CONFIGURATION WITH SPOILERS - Concluded

(e) $h_s = -0.100\bar{c}$

$\frac{y}{b/2}$	α, deg	$c_{n,w}$	$c_{m,w}$	$c_{n,v}$	$c_{m,v}$	$c_{n,f}$	$c_{m,f}$	$c_{n,s}$	$c_{m,s}$
0.21	-4	.3757	-.2748	1.8304	-.7898	.9835	-.3737	-.4599	.2365
0.21	0	.5378	-.3035	1.8807	-.8141	1.0384	-.4005	-.4169	.2052
0.21	4	.6947	-.3138	1.8972	-.8235	1.1183	-.4349	-.4309	.2143
0.21	8	.8631	-.3275	2.0573	-.8875	1.2052	-.4784	-.4624	.2165
0.21	12	1.1104	-.3214	2.2091	-.9445	1.2661	-.5179	-.4632	.1964
0.21	16	1.3335	-.3161	2.4757	-1.0296	1.3443	-.5570	-.4963	.2102
0.21	20	1.6361	-.3031	2.5078	-1.0439	1.3698	-.5689	-.5208	.2161
0.21	22	1.6723	-.3284	2.5404	-1.0675	1.3848	-.5744	-.5308	.2187
0.21	24	1.5311	-.3964	2.4127	-1.0357	1.3494	-.5654	-.4798	.1896
0.21	26	1.5173	-.4526	2.1552	-.9523	1.4148	-.6022	-.4097	.1539
0.30	-4	.4767	-.2368	2.1901	-.8537	1.0470	-.4100	-.4704	.2426
0.30	0	.6010	-.2403	2.3371	-.9156	1.0808	-.4186	-.5930	.2860
0.30	4	.7878	-.2360	2.2821	-.9058	1.1246	-.4385	-.5964	.2874
0.30	8	1.0212	-.2195	2.5035	-.9732	1.1793	-.4578	-.7399	.3562
0.30	12	1.2638	-.2133	2.8687	-1.1093	1.1826	-.4531	-.8631	.3997
0.30	16	1.5567	-.2175	3.2343	-1.2667	1.1932	-.4478	-1.0990	.5034
0.30	20	1.7746	-.2580	3.3859	-1.3349	1.2564	-.4764	-1.1103	.4943
0.30	22	1.8058	-.2950	3.2504	-1.3062	1.3327	-.5091	-1.0717	.4635
0.30	24	1.6341	-.3292	2.6471	-1.1239	1.3629	-.5404	-.8387	.3457
0.30	26	1.5082	-.3523	2.1909	-.9718	1.4228	-.5855	-.6176	.2469
0.43	-4	.5089	-.2346	2.1052	-.8113	.9034	-.3290	-.4159	.2115
0.43	0	.7268	-.2464	2.1860	-.8390	1.0070	-.3783	-.5122	.2398
0.43	4	.9293	-.2328	2.0980	-.8214	1.0602	-.4107	-.4648	.2188
0.43	8	1.1960	-.2306	2.3723	-.9169	1.1138	-.4372	-.4983	.2316
0.43	12	1.5570	-.2612	3.7214	-1.0356	1.1628	-.4513	-.6220	.2934
0.43	16	1.6970	-.3024	3.0305	-1.1857	1.1127	-.4360	-.8030	.3467
0.43	20	1.6613	-.3200	2.8959	-1.1724	1.1602	-.4598	-.8046	.3322
0.43	22	1.6159	-.3311	2.6678	-1.1077	1.2344	-.4908	-.7653	.3046
0.43	24	1.5548	-.3486	2.1917	-.9553	1.3407	-.5550	-.5289	.1917
0.43	26	1.4788	-.3660	2.0144	-.9008	1.4324	-.6035	-.3686	.1232
0.55	-4	.6270	-.2639	2.0947	-.8315	.9611	-.3515	-.5796	.2892
0.55	0	.8220	-.2686	2.1801	-.8392	1.0052	-.3701	-.6157	.2958
0.55	4	1.1127	-.2509	2.2169	-.8636	1.0320	-.3814	-.6298	.3097
0.55	8	1.5114	-.3011	2.5922	-1.0084	1.0961	-.4108	-.6153	.3055
0.55	12	1.5999	-.3621	2.6541	-1.0866	1.1299	-.4465	-.5090	.2222
0.55	16	1.5274	-.3698	2.4865	-1.0392	1.1198	-.4494	-.3666	.2073
0.55	20	1.4809	-.3527	2.2984	-.9747	1.1820	-.4849	-.5386	.1899
0.55	22	1.4568	-.3633	2.1926	-.9443	1.2591	-.5199	-.5007	.1731
0.55	24	1.4430	-.3781	1.9661	-.8703	1.3513	-.5727	-.3706	.1142
0.55	26	1.3867	-.3798	1.8820	-.8402	1.3639	-.5843	-.2752	.0718
0.72	-4	.3176	-.0038						
0.72	0	.6097	.0338						
0.72	4	1.1351	-.1028						
0.72	8	1.0125	-.1287						
0.72	12	.8783	-.1105						
0.72	16	.7929	-.0929						
0.72	20	.7270	-.0797						
0.72	22	.7043	-.0769						
0.72	24	.6998	-.0773						
0.72	26	.7033	-.0796						

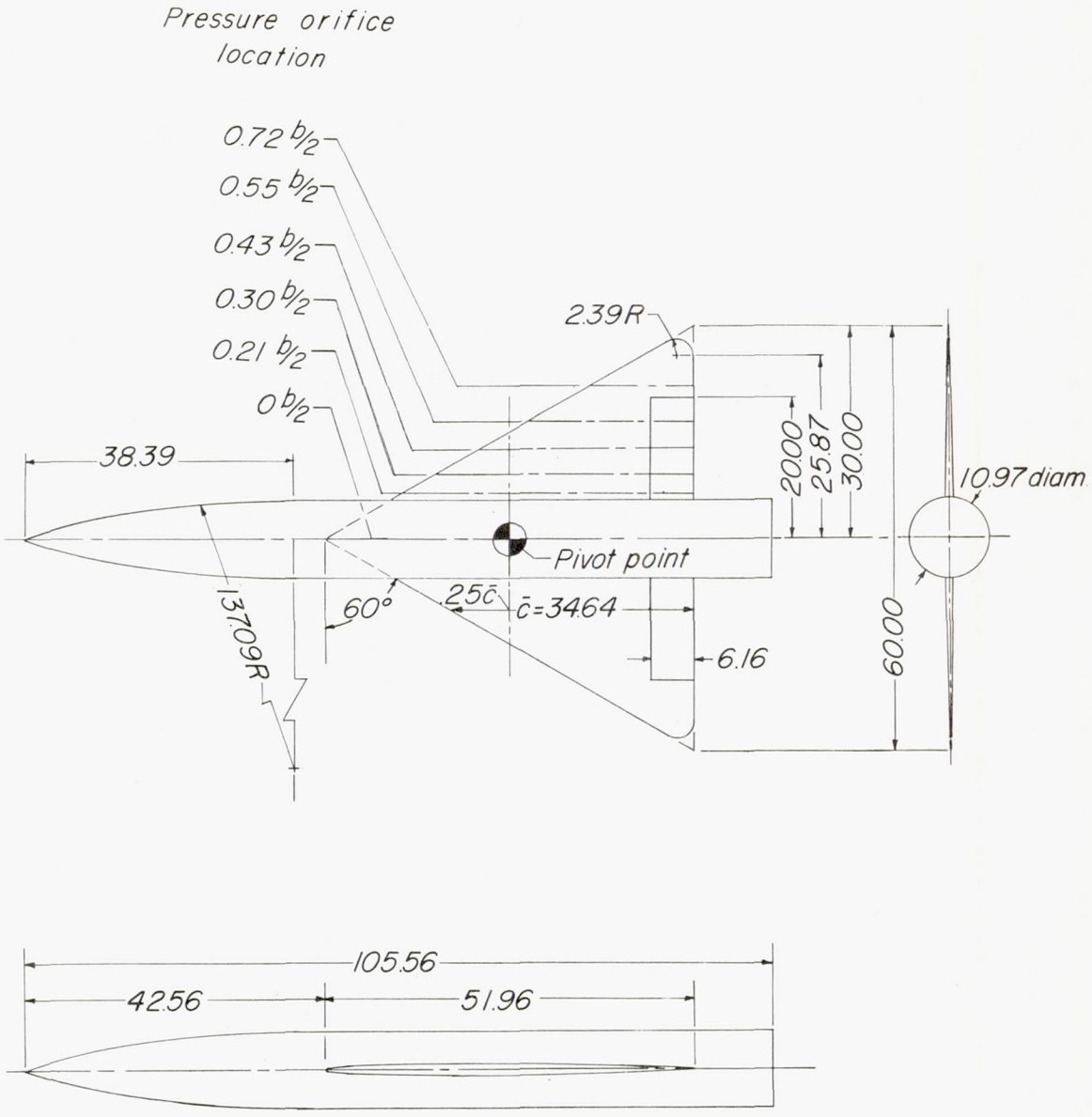


Figure 1.- General arrangement of 60° delta-wing model (all dimensions are in inches unless otherwise noted).

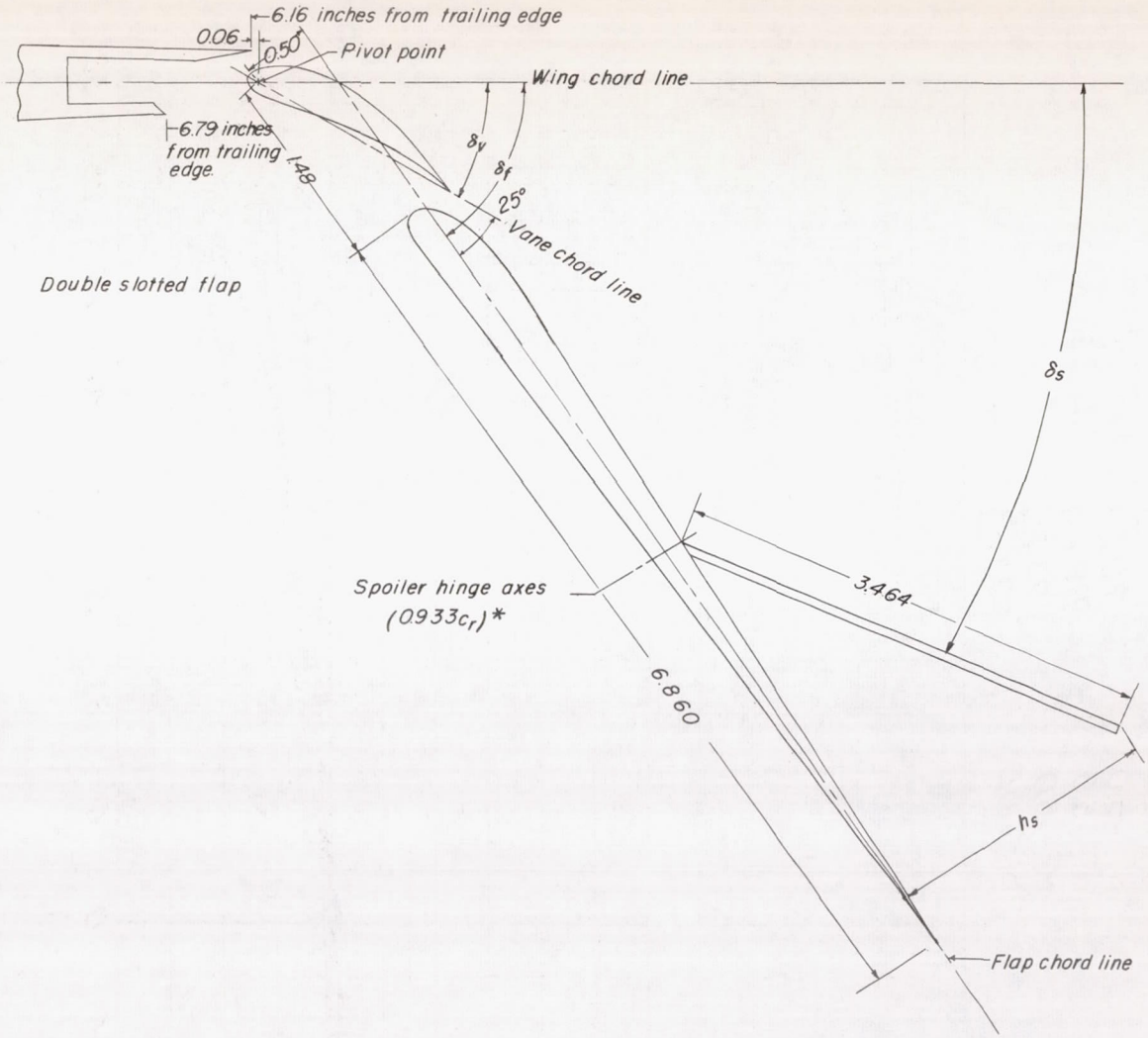


Figure 2.- Details of double slotted flap with spoiler. All dimensions are in inches unless otherwise noted. (* indicates spoiler axes for pressure-distribution test. For force tests spoiler hinge axis at 0.700, 0.894, and $0.933c_r$ on plain-wing configuration and at 0.894 and $0.933c_r$ on double-slotted-flap configuration.)

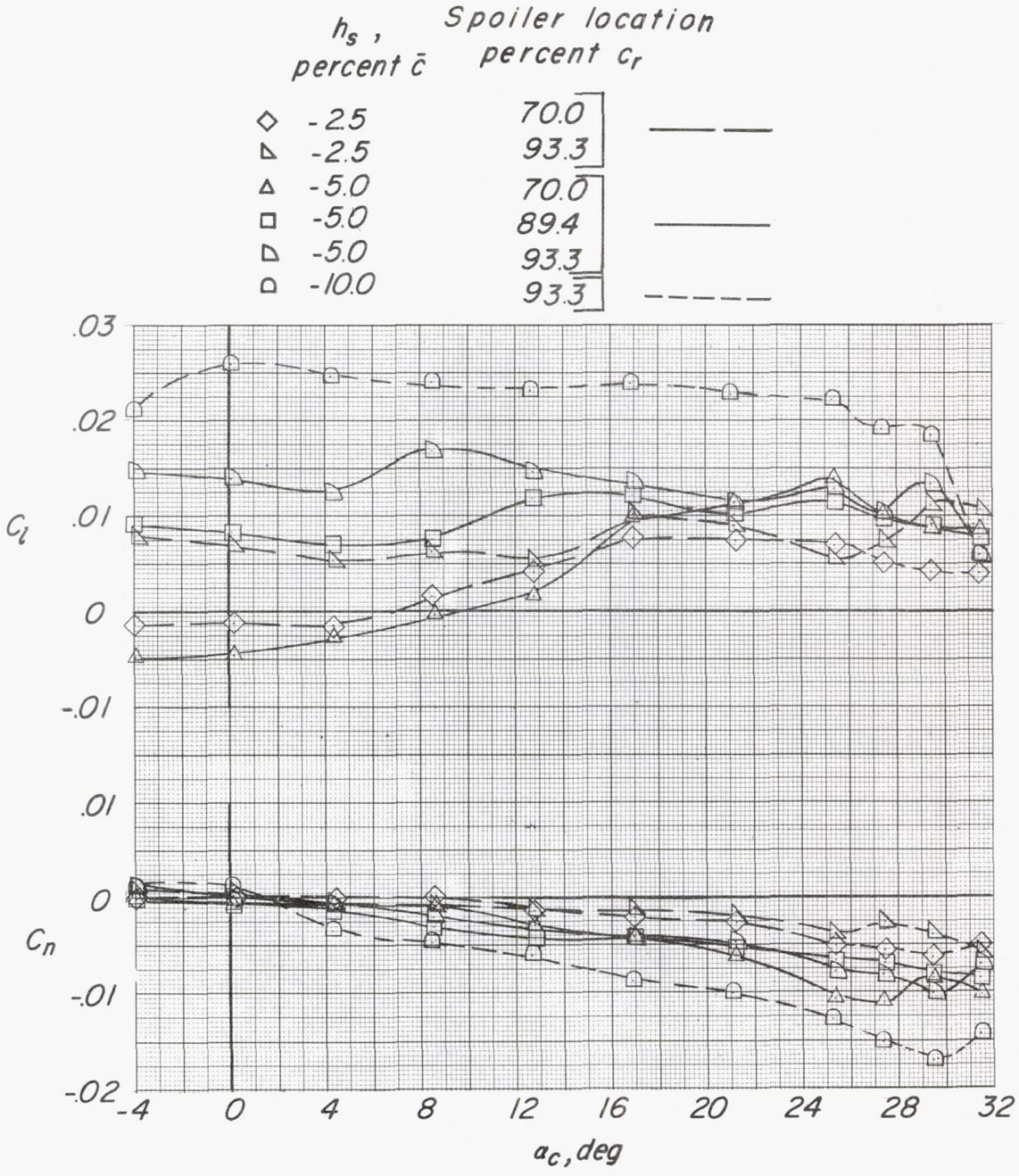


Figure 3.- Variation of rolling-moment coefficient and yawing-moment coefficient with angle of attack for plain-wing configuration having spoilers at various chordwise positions and at several projections.

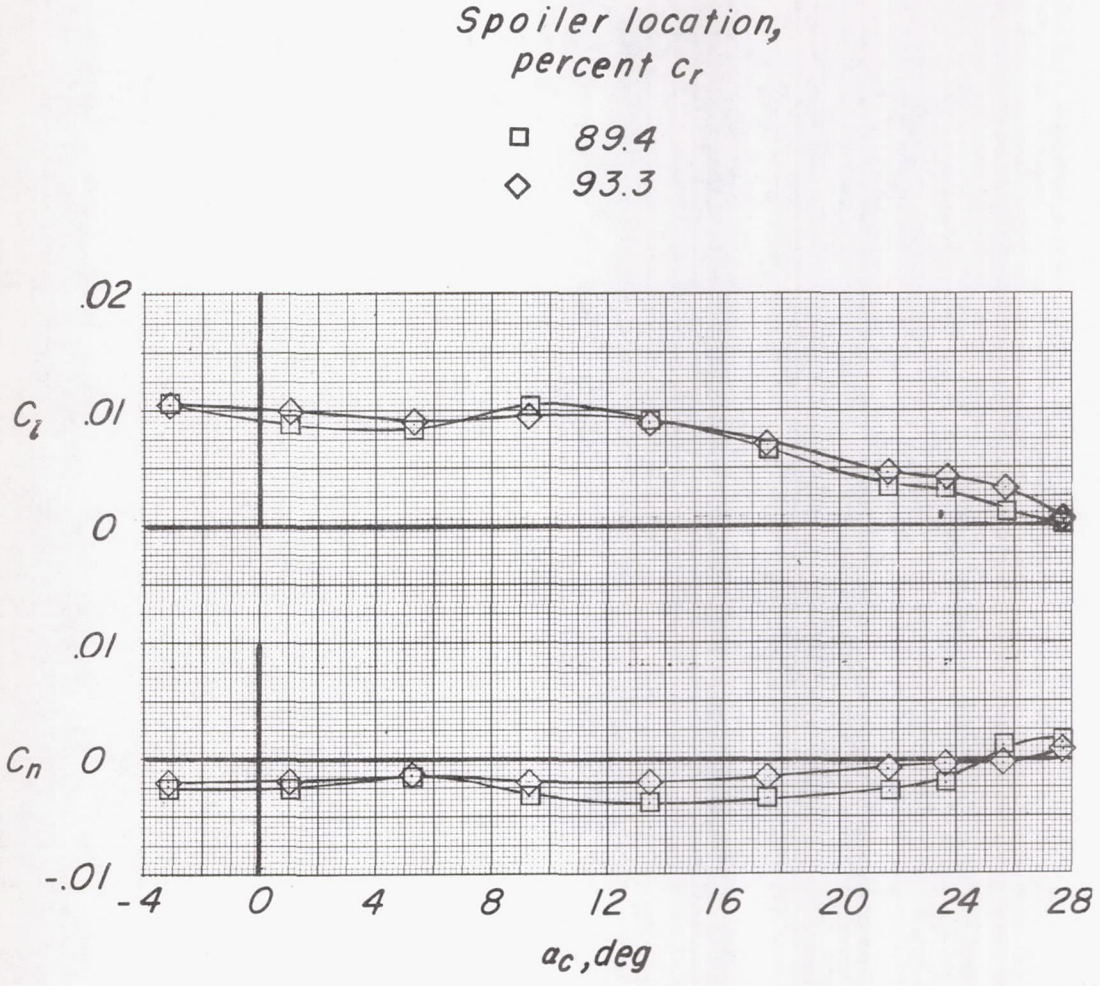
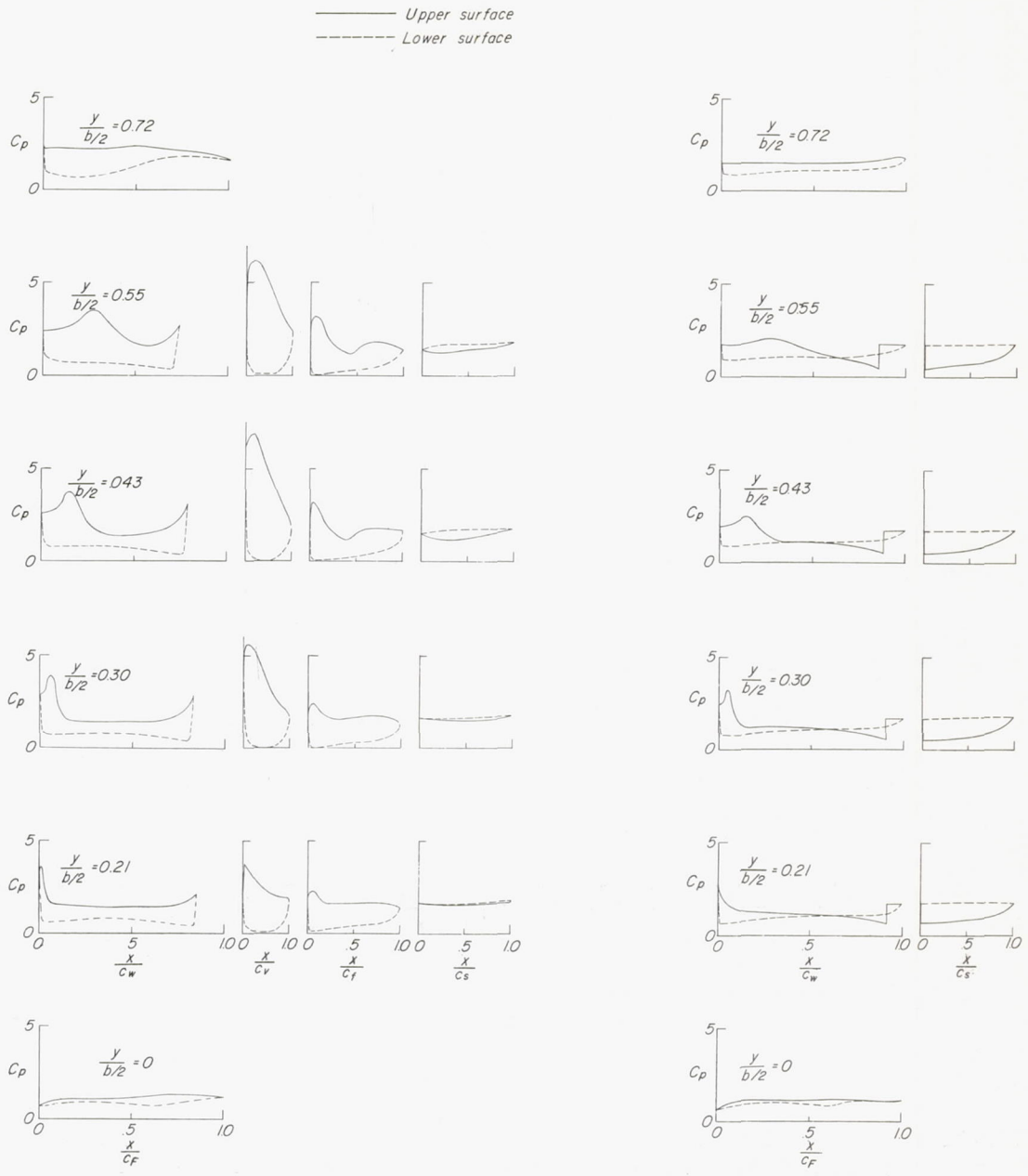


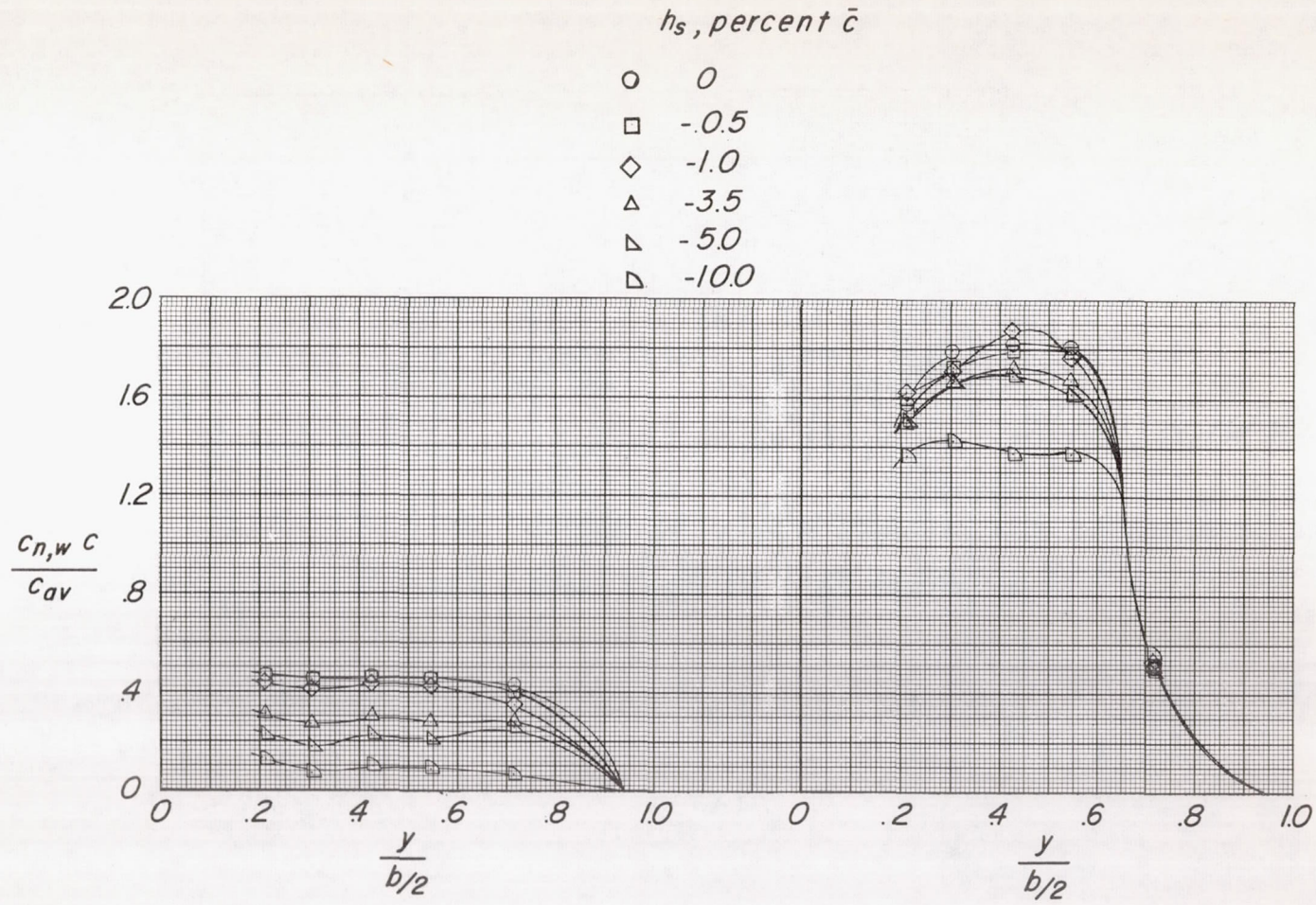
Figure 4.- Variation of rolling-moment coefficient and yawing-moment coefficient with angle of attack for the double-slotted-flap configuration having spoilers at two chordwise positions; $\delta_f = 60^\circ$, $h_s = -0.05\bar{c}$.



(a) Double-slotted-flap configuration.

(b) Plain-wing configuration.

Figure 5.- Chordwise load distribution at several spanwise positions for plain-wing configuration and double-slotted-flap configuration at $\alpha = 8^\circ$ and $h_s = -0.05\bar{c}$. (Note: x/c_F, x/c_w, x/c_v, x/c_f, and x/c_s are not the same scale.)



(a) Plain-wing configuration.

(b) Double-slotted-flap configuration; $\delta_f = 60°$.

Figure 6.- Span-load distribution over plain-wing and double-slotted-flap configuration at several spoiler projections; $\alpha = 8°$.

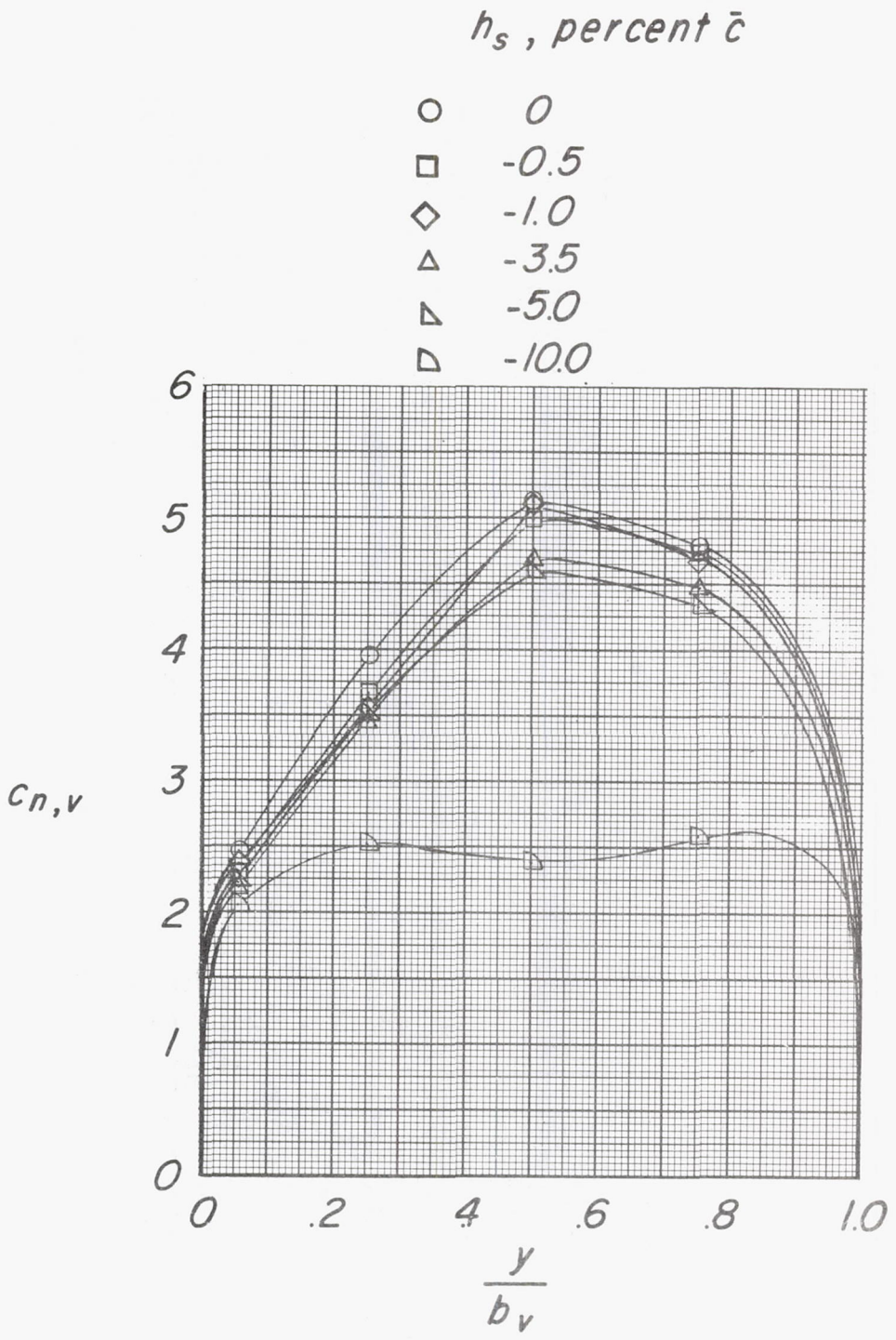


Figure 7.- Span-load distribution over vane of double-slotted-flap configuration at several spoiler projections; $\delta_f = 60^\circ$; $\alpha = 8^\circ$.

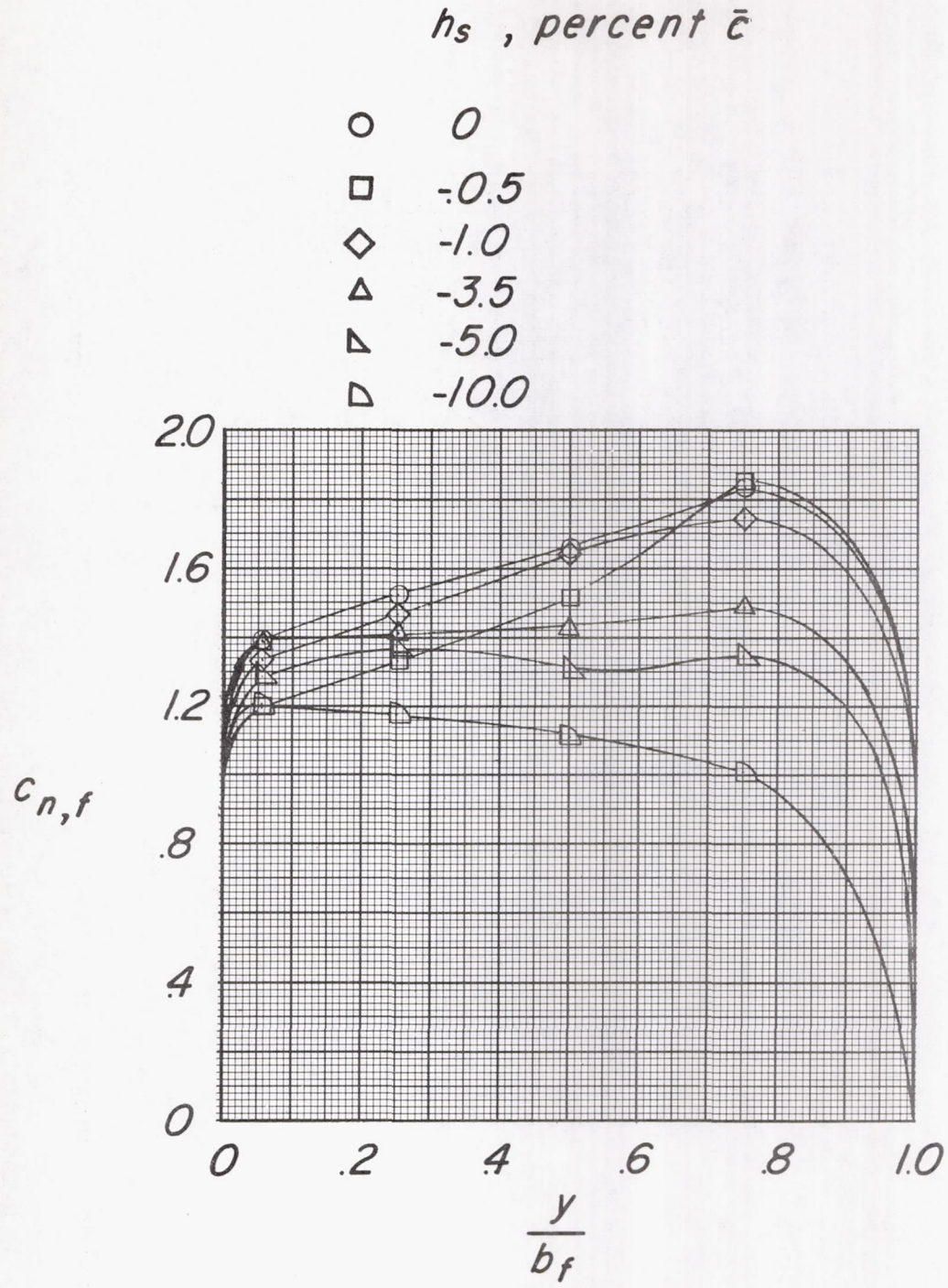
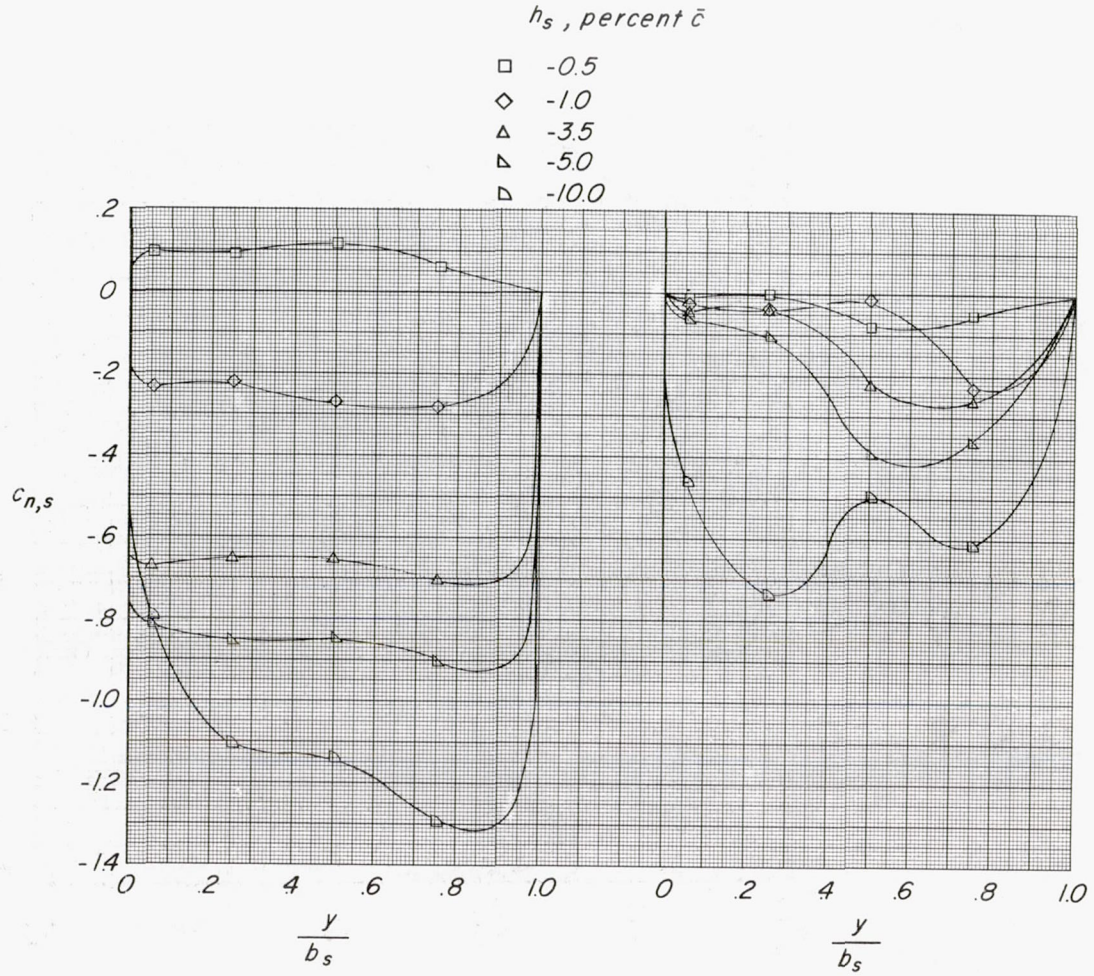


Figure 8.- Span-load distribution over flap of the double-slotted-flap configuration at several spoiler projections; $\alpha = 8^{\circ}$.



(a) Plain-wing configuration.

(b) Double-slotted-flap configuration; $\delta_f = 60^\circ$.

Figure 9.- Span-load distribution over spoiler for plain-wing and double-slotted-flap configurations at several spoiler projections; $\alpha = 8^\circ$.

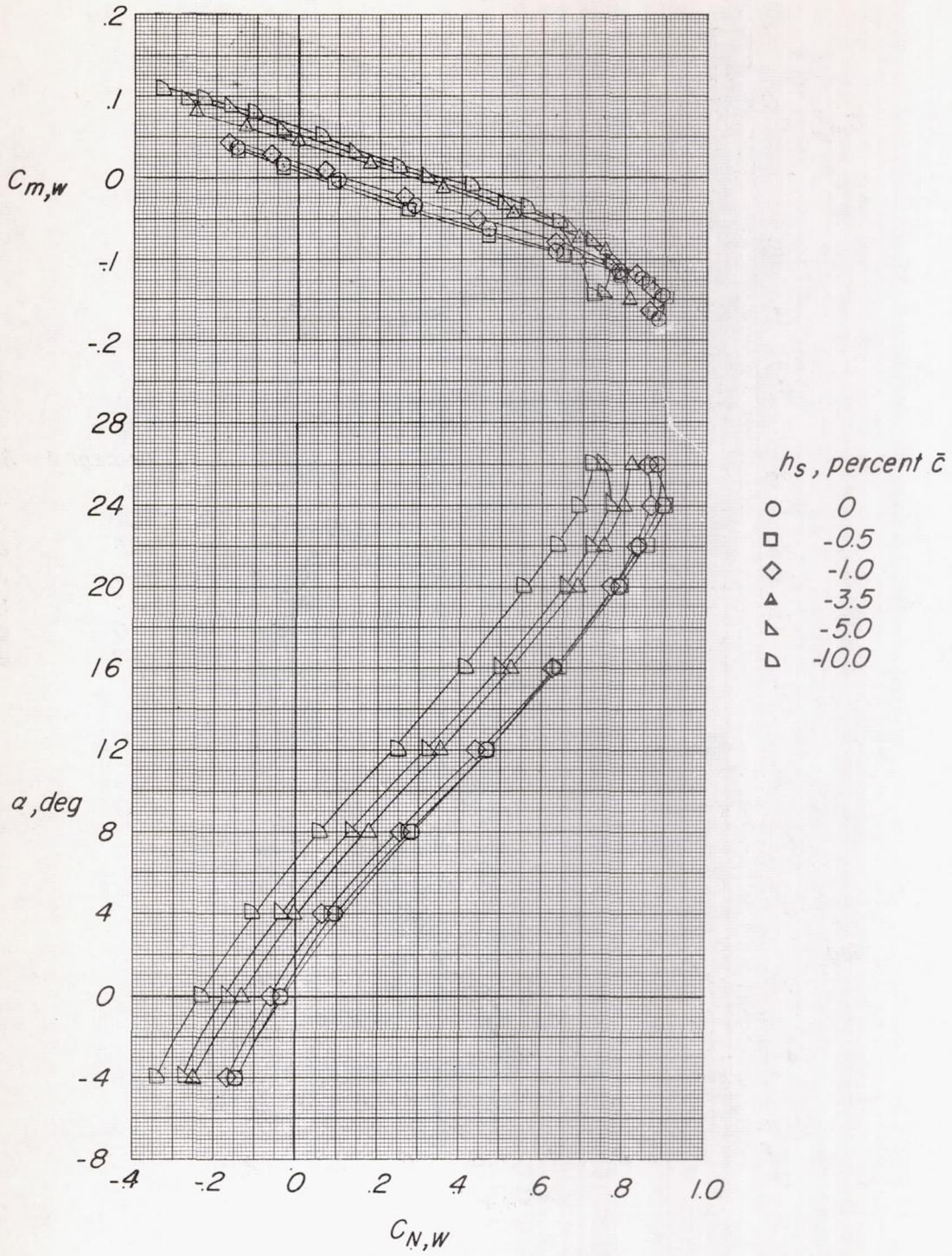


Figure 10.- Effect of spoiler projection on wing normal-force coefficients and pitching-moment coefficients of plain-wing configuration.

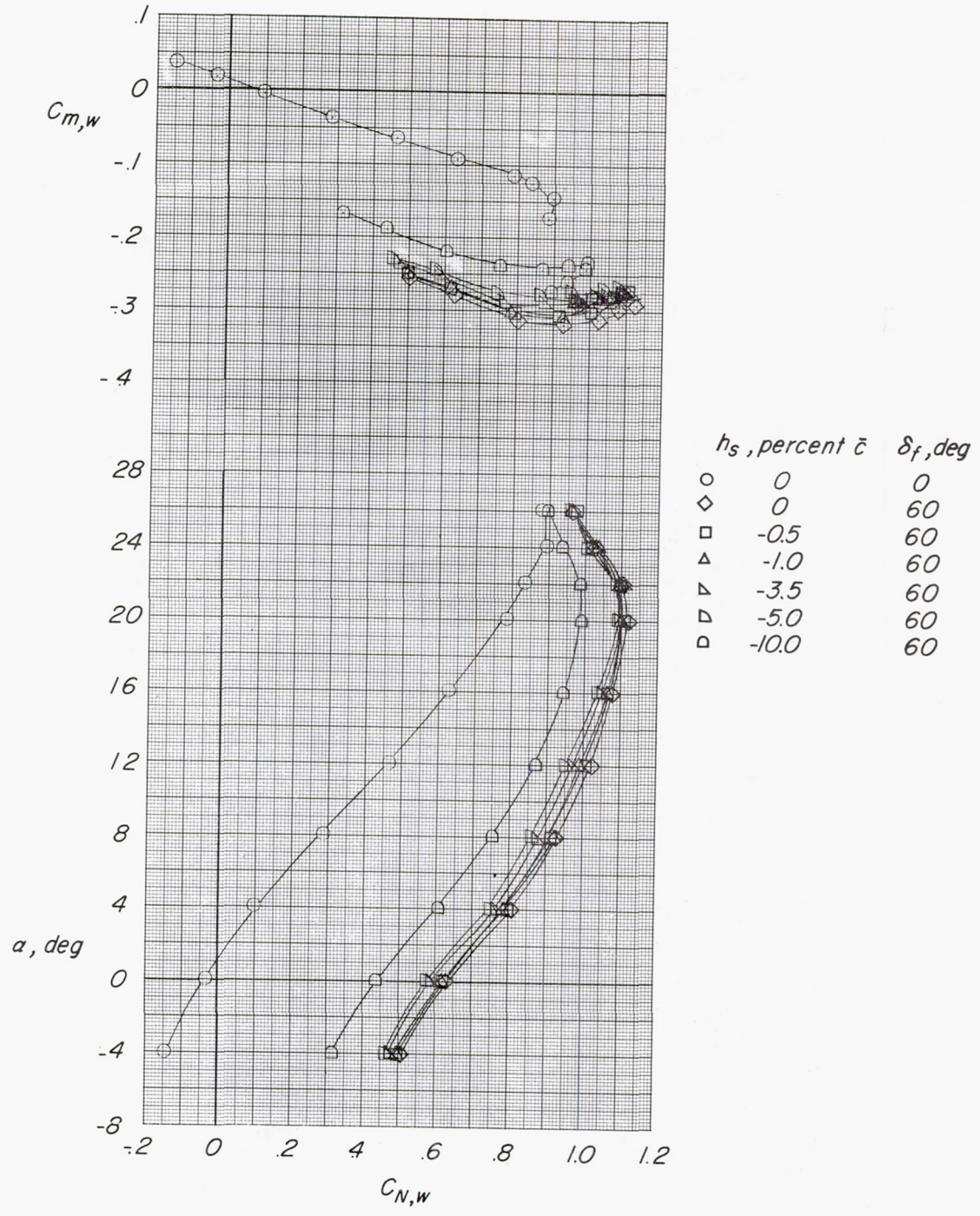


Figure 11.- Effect of spoiler projection on wing normal-force coefficients and pitching-moment coefficients of double-slotted-flap configuration; $\delta_f = 60^\circ$.

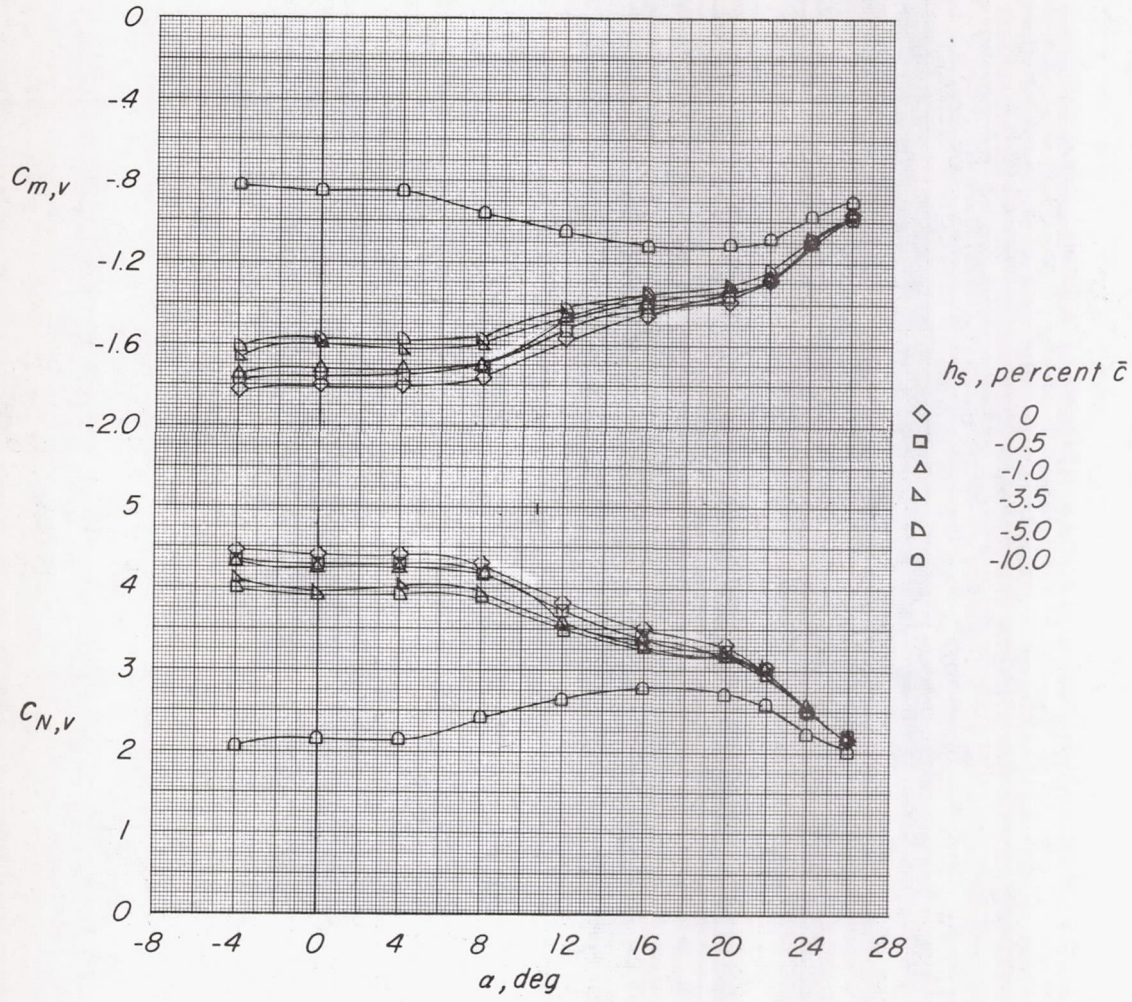


Figure 12.- Effect of spoiler projection on vane normal-force coefficients and pitching-moment coefficients of double-slotted-flap configuration; $\delta_f = 60^\circ$.

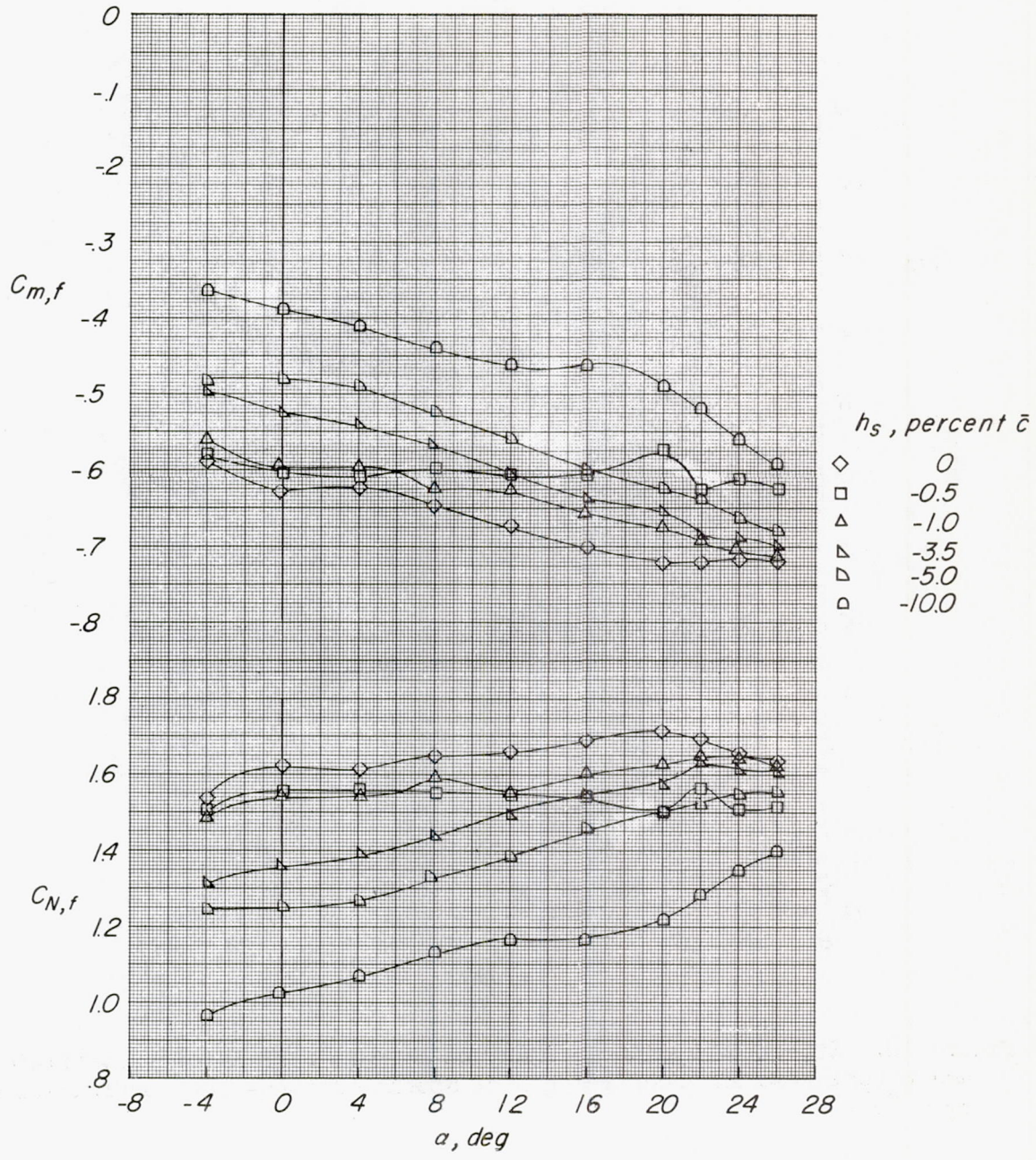


Figure 13.- Effect of spoiler projection on flap normal-force coefficients and pitching-moment coefficients of double-slotted-flap configuration; $\delta_f = 60^\circ$.

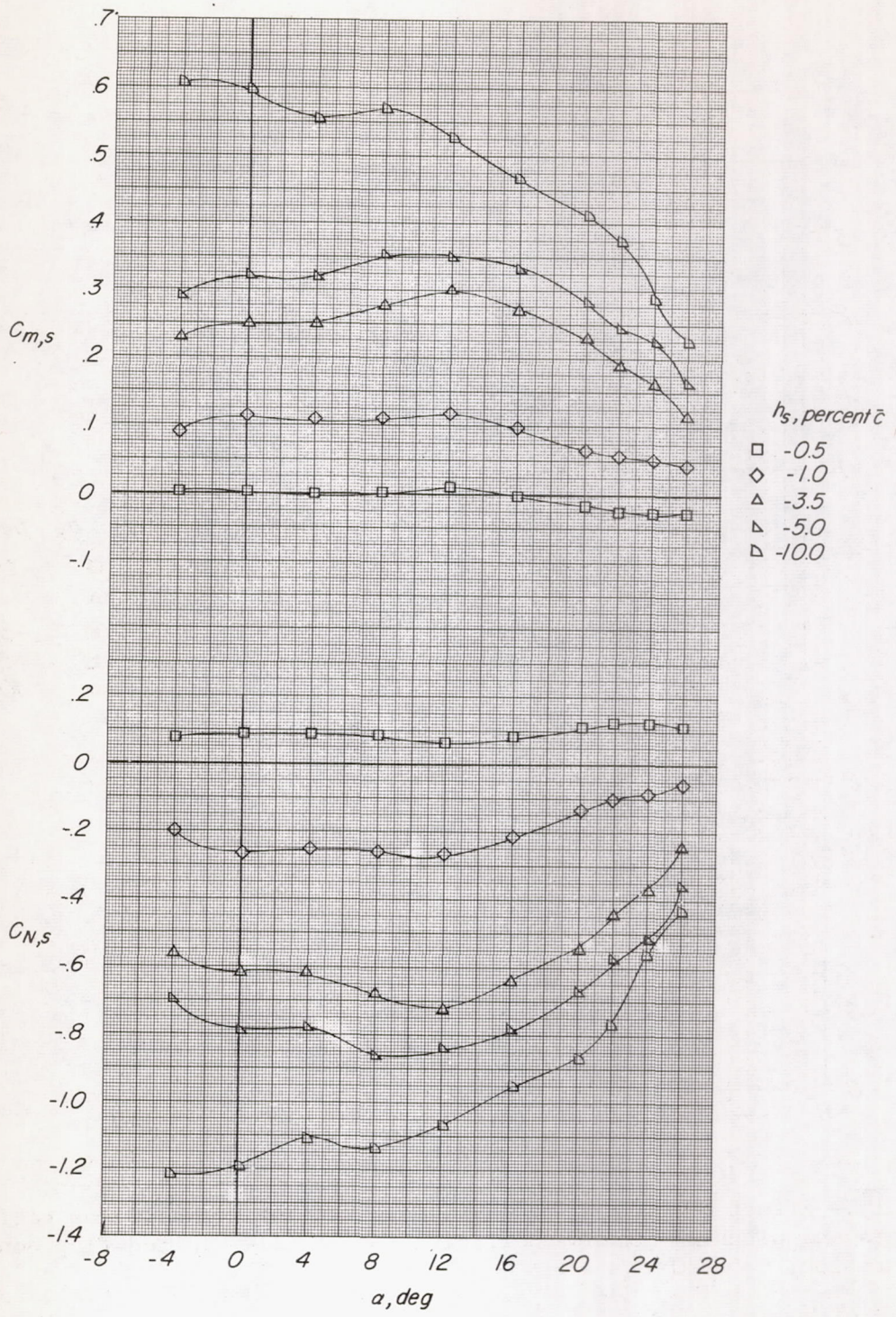


Figure 14.- Effect of spoiler projection on spoiler normal-force coefficients and pitching-moment coefficients of plain-wing configuration.

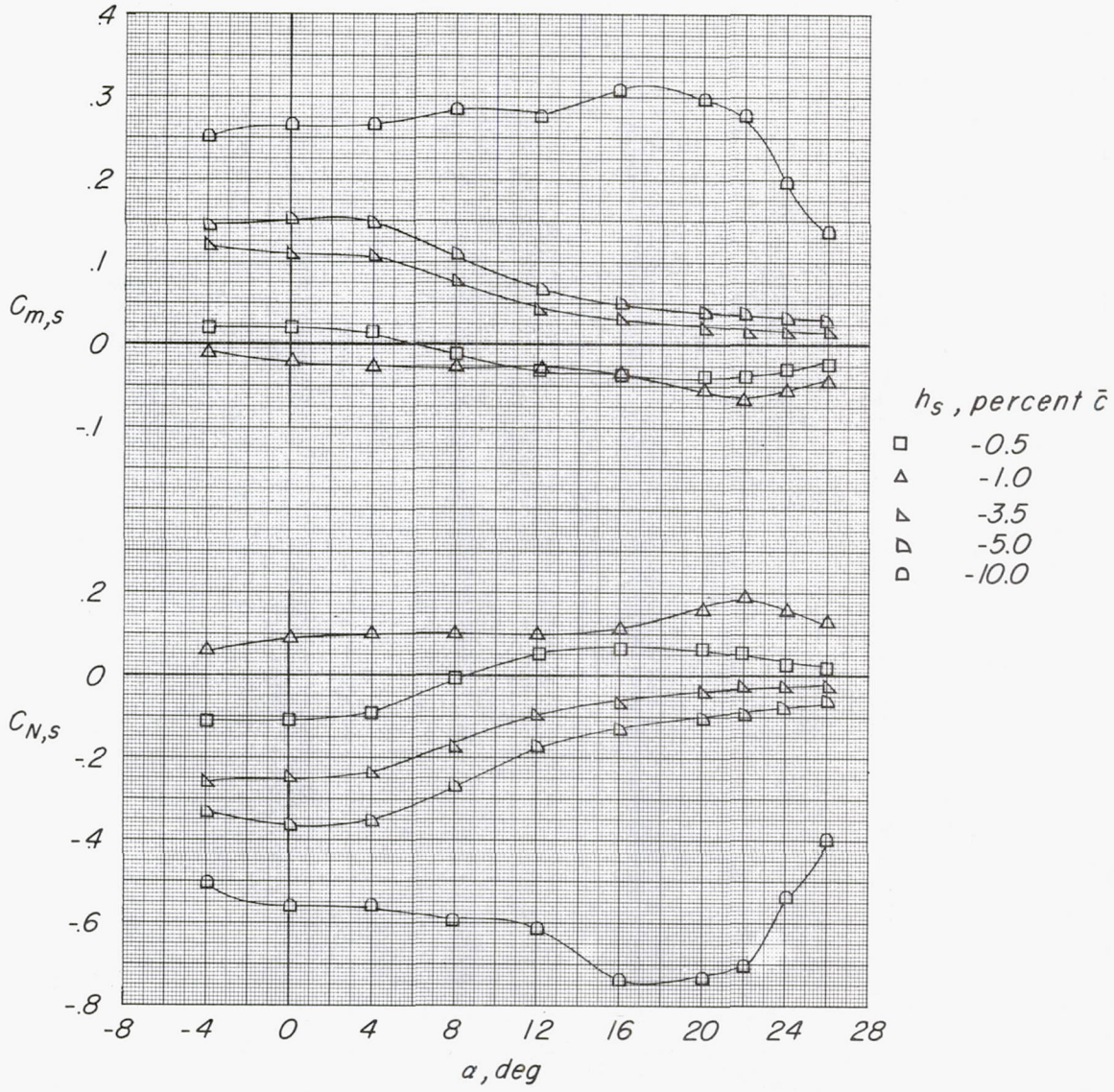


Figure 15.- Effect of spoiler projection on spoiler normal-force coefficients and pitching-moment coefficients of double-slotted-flap configuration; $\delta_f = 60^\circ$.