SE FIL
COPY

N62 11651

NASA TN D-1264

NASA TN D-1264

TECHNICAL NOTE

D-1264

PRESSURE DISTRIBUTIONS AT MACH NUMBER 2.05 ON A SERIES OF HIGHLY SWEPT ARROW WINGS EMPLOYING VARIOUS DEGREES OF TWIST AND CAMBER

By Harry W. Carlson

Langley Research Center
Langley Station, Hampton, Va.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

WASHINGTON May 1962

A

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

TECHNICAL NOTE D-1264

PRESSURE DISTRIBUTIONS AT MACH NUMBER 2.05 ON A SERIES OF HIGHLY SWEPT ARROW WINGS EMPLOYING VARIOUS DEGREES OF TWIST AND CAMBER

By Harry W. Carlson

SUMMARY

A series of three arrow wings employing various degrees of twist and camber were tested in the Langley 4- by 4-foot supersonic pressure tunnel. Pressure distributions were measured at a Mach number of 2.05 and at a Reynolds number of 4.4×10^6 based on the mean aerodynamic chord. The wings had a leading-edge sweep angle of 70° and an aspect ratio of 2.24, and were designed to produce a minimum drag (in comparison with that produced for other wings in the family) at lift coefficients of 0, 0.08, and 0.16.

The pressure tests substantiate previous force tests (NASA Technical Memorandum X-332) in indicating that the design methods chosen achieved the desired result of improving performance while avoiding transonic-flow phenomena and shock-induced flow separation. Failure to approach the full theoretical benefits of twist and camber may be due, at least partly, to the aeroelastic deflections believed to be present.

INTRODUCTION

Recent wind-tunnel tests of a series of twisted and cambered highly swept arrow wings (ref. 1) showed significant improvements in maximum lift-drag ratio for the warped wings as compared with the flat wing. The greatest improvement was shown for a wing designed for a lift coefficient approximately one-half that required for the maximum lift-drag ratio.

The purpose of these current tests is to examine, in detail, the pressure distributions on these wings and to compare them with the theoretical estimates. The primary objective is to ascertain the success of the design methods in producing the required loading distributions while avoiding flow separation and transonic-flow phenomena on the wing surfaces.

The three half-span wings tested in the Langley 4- by 4-foot supersonic pressure tunnel at a Mach number of 2.05 had a leading-edge sweep of 70° and an aspect ratio of 2.24. One of these wings was twisted and cambered for a design lift coefficient of 0.16, a second wing employed only one-half that twist and camber, and a third wing had no twist and camber.

SYMBOLS

L
1
8
5
3

b wing span

b' distance from wing trailing edge (or wing center line) to wing leading edge measured in y-direction

c local wing chord

C_A axial-force coefficient, $\frac{\text{Axial force}}{qS}$, $\frac{2}{S}\int_0^1 c_a c \, d\left(\frac{y}{b/2}\right)$

c_a chordwise section axial-force coefficient

$$\int_0^1 \left[\left(C_p \frac{dz}{dx}\right)_U - \left(C_p \frac{dz}{dx}\right)_L\right] d\left(\frac{x'}{c}\right)$$

C_D drag coefficient, $\frac{\text{Drag}}{qS}$

C_L lift coefficient, $\frac{\text{Lift}}{qS}$

c_n chordwise section normal-force coefficient,

$$\int_0^1 \left(C_{p,L} - C_{p,U}\right) d\left(\frac{x'}{c}\right)$$

$c_{n,b}$ spanwise section normal-force coefficient,

$$\int_0^1 \left(C_{p,L} - C_{p,U}\right) d\left(\frac{y'}{b'}\right)$$

C_p	pressure coefficient
$C_{p,o}$	pressure coefficient at zero lift for uncambered and untwisted wing
l	overall length of wing measured in streamwise direction
q	free-stream dynamic pressure
S	wing area, half-span model
x,y,z	Cartesian coordinate system with origin at wing apex, x-axis streamwise
x'	distance from wing leading edge measured in x-direction
y'	distance from wing trailing edge (or wing center line) measured in y-direction
α	angle of attack

Subscripts:

L	lower surface
U	upper surface

MODELS

Photographs of the three half-span models (designated wings 1, 2, and 3) mounted on the reflection plate are shown in figure 1. The wings had a 70° swept leading edge and an aspect ratio of 2.24. Each of the wings of this series was designed to produce a minimum drag (in comparison with that produced for other wings in the family) at a certain lift coefficient. These design lift coefficients were 0, 0.08, and 0.16 for wings 1, 2, and 3, respectively. (A design lift coefficient of 0 corresponds to a flat wing.)

Figure 2 presents a plan view of the wings showing orifice locations and some typical sections for each of the three wings. The tables included in this figure give the camber surface ordinates and the thickness distribution. Except for leading-edge and trailing-edge positions, the x' and y values listed correspond to orifice locations. Further details of the wing construction may be found in reference 1.

TESTS

The tests were conducted in the Langley 4- by 4-foot supersonic pressure tunnel with a free-stream Mach number of 2.05 and a Reynolds number of 4.4×10^6 based on the mean aerodynamic chord. The all-steel wings were attached to a reflection plate which was supported in a horizontal position by the permanent sting mounting system of the tunnel. During the tests, the wing and plate moved through an angle-of-attack range as a unit. As in the force tests of reference 1, transition strips were used on the wings. The strips, composed of a sparse distribution of No. 80 carborundum grains in a lacquer binder, were 1/8 of an inch wide and were located 1/4 inch behind the leading edge. Angle of attack was measured optically with the use of prisms recessed in the wing surface. The pressure distributions were registered on a bank of manometers using Alkazene as a fluid. It is estimated that the measured pressure coefficients have an accuracy of ±0.01.

L
1
8
5
3

RESULTS AND DISCUSSION

The pressure distributions measured on the wing surfaces are shown in table I and in figures 3, 4, and 5. Pressure coefficient has been plotted against chordwise location x'/c. Notice that the zero-pressure-coefficient line changes, and that for each angle of attack it is identified on the right margin by the angle and the corresponding test-point symbol.

A sample comparison of measured pressure distributions with linearized-theory results is shown in figure 6. The theoretical values used can be expressed by the following equation:

$$C_p = C_{p,o} \pm \frac{C_{L,design}}{2}\left[1.4 + 1.846\left(\frac{y}{b/2} - \frac{x}{l}\right)\right] \pm 0.28\,\frac{C_L - C_{L,design}}{\sqrt{1 - 7.56\left(\frac{y}{x}\right)^2}} \qquad (1)$$

The contribution of the thickness alone is represented by the first term, which was evaluated by using the line-source method of reference 2. Computations were performed on a digital computer by using the method described in reference 3. The second term accounts for the camber contribution, which is discussed in more detail in reference 1. The third term represents the effects of incidence on a flat-plate wing and was evaluated for the wings under consideration with the use of the equations of reference 4. In equation (1) the positive signs correspond to the lower surface and the negative signs correspond to the upper surface.

Data for the wing without twist and camber (fig. 6(a)) indicate that the correlation of theoretical and experimental thickness pressures ($\alpha = 0^{\circ}$) is generally good. For the maximum lift-drag ratio condition, which occurs at $\alpha \approx 4^{\circ}$, the agreement is good near the root but becomes poorer as the tip is approached. It is believed that aeroelastic deflections may cause some of this discrepancy. There is evidence of the presence of a vortex flow originating at the wing apex. The effect of the vortex is more pronounced at $\alpha = 8^{\circ}$, especially at the $\frac{y}{b/2} = 0.3$ station. The scrubbing effect of the vortex on the wing upper surface can be observed in oil-flow photographs. (The photographs are not presented in this paper because they were of poor quality and would therefore not reproduce satisfactorily.) Regions where the pressures are affected by the vortex correspond to the areas of separated flow described in reference 1.

L
1
8
5
3

For wing 2 with a design lift coefficient of 0.08, the maximum lift-drag ratio condition occurs at $\alpha \approx 2^{\circ}$, or a lift coefficient of about 0.15. In figure 6(b) it can be seen that for this condition the correlation of experiment and theory is again good at the root and only fair near the tip. Again aeroelastic deflections may play a part. There also is evidence of the vortex, particularly at the $\frac{y}{b/2} = 0.3$ station for $\alpha = 6^{\circ}$ (4° above design condition).

The third wing in the series shows discrepancies between measured data and theory similar to those for the first two wings. It does seem, however, that the design method employed and the restrictions imposed (as discussed in ref. 1) achieved the desired result of avoiding transonic-flow phenomena and shock-induced flow separation. The fact that reasonable agreement was obtained 4° above the design condition ($\alpha = 0^{\circ}$, $C_L = 0.16$) indicates that the design restrictions may have been more severe than necessary.

A comparison of theoretical and experimental axial-force distribution is shown in figure 7. The theory represented by the solid line results from an integration of the pressure-coefficient equation (eq. (1)). The dashed line represents linearized results with leading-edge suction forces included. The leading-edge thrust is given by the following relationship, which was determined by applying equations found in reference 4 to the present case:

$$\Delta c_a = -0.36\left(C_L - C_{L,\mathrm{design}}\right)^2 \frac{y}{c} \tag{2}$$

Since leading-edge suction is theoretically confined to the nose section of the airfoil (in this case a point), leading-edge-suction effects would not be expected to be apparent in pressure data. For those inboard stations where the data follow the leading-edge-suction curves, this tendency is believed to be caused not by actual leading-edge suction but by the vortex previously mentioned.

A comparison of theoretical and experimental section normal-force distribution is shown in figure 8. The experimental data follow fairly well the shape of these loading curves, but it can be seen that the integrated total normal force would be less experimentally. There is a noticeable loss of lift as the wing tip is approached, particularly at the higher lift coefficients.

L
1
8
5
3

An interesting aspect of the loadings on these wings at or near the maximum lift-drag ratio condition is treated in figure 9 where section normal-force coefficients are shown for both chordwise and spanwise sections. The near optimum combination of loadings on wing 3 does not, even in theory, produce a "full" spanwise loading curve, as might have been expected. Only when the chordwise distributions also are examined is an improved loading noted for the twisted and cambered wings.

Figure 10 shows a summary comparison of theory with integrated pressure data from the current tests and with force data from the tests reported in reference 1. There is reasonable agreement between the pressure data and the force data except for the axial-force coefficients of wing 3. This discrepancy may be due, in part, to difficulties in making a proper fairing of the pressure-distribution data. Particular difficulty is encountered in extrapolating section axial-force distribution to the wing root, where large values and steep slopes exist for the highly twisted and cambered wing. As noted before, it is now believed that the tendency of both pressure and force data to follow the leading-edge-suction theory is purely coincidence, and is actually caused by the presence of a vortex flow originating at the wing apex.

CONCLUSIONS

An experimental pressure-distribution investigation at a Mach number of 2.05 of a series of twisted and cambered arrow wings provides the following conclusions:

1. Measured pressure distributions for all wings agreed fairly well with the linearized theory estimates. Much of the discrepancies found may be due to aeroelastic deflections and the presence of a vortex flow originating at the wing apex.

2. Agreement between integrated pressure data and previously obtained force data is reasonably good except for axial-force coefficients for the wings with twist and camber. These discrepancies may be due to the fairing of the pressure data which becomes more critical with increased twist and camber.

3. The pressure tests substantiate previous force tests (NASA Technical Memorandum X-332) in indicating that the design methods chosen achieved the desired result of improving performance while avoiding transonic-flow phenomena and shock-induced flow separation. Failure to approach the full theoretical benefits of twist and camber may be due, at least partly, to the aeroelastic deflections believed to be present.

L
1
3
5
3

Langley Research Center,
National Aeronautics and Space Administration,
Langley Air Force Base, Va., March 6, 1962.

REFERENCES

1. Carlson, Harry W.: Aerodynamic Characteristics at Mach Number 2.05 of a Series of Highly Swept Arrow Wings Employing Various Degrees of Twist and Camber. NASA TM X-332, 1960.

2. Jones, Robert T.: Thin Oblique Airfoils at Supersonic Speed. NACA Rep. 851, 1946.

3. Carlson, Harry W.: Measurements of Flow Properties in the Vicinity of Three Wing-Fuselage Combinations at Mach Numbers of 1.61 and 2.01. NASA TM X-64, 1959.

4. Brown, Clinton E.: Theoretical Lift and Drag of Thin Triangular Wings at Supersonic Speeds. NACA Rep. 839, 1946.

TABLE I.- TABULATED PRESSURE COEFFICIENTS

(a) Wing 1

x'/c	C_p at $\frac{y}{b/2}$ of:										x'/c
	.1		.3		.5		.7		.9		
	Upper	Lower	Upper	Lower	Upper	Lower	Upper	Lower	Upper	Lower	
$\alpha = -2°$											
.025	.092	.012	.098	-.060	.094	-.134					.025
.050	.063	-.005	.091	-.020	.079	-.086	.073	-.134			.050
.100	.052	-.001	.054	-.001	.060	-.052	.062	-.123	.068	-.262	.100
.200	.035	-.010	.043	-.014	.038	-.040	.021	-.093	.035	-.222	.200
.300	.025	-.012	.027	-.030	.030	-.047	.022	-.072	.015	-.204	.300
.400	.007	-.033	.011	-.044	.010	-.058	.001	-.082	.001	-.180	.400
.500	.010	-.029	.005	-.042	-.010	-.065	-.021	-.093	-.017	-.168	.500
.600	-.002	-.038	-.004	-.050	-.021	-.075	-.036	-.103	-.033	-.165	.600
.700	-.016	-.055	-.026	-.063	-.039	-.079	-.049	-.108	-.030	-.154	.700
.800	-.023	-.059	-.020	-.060	-.047	-.094	-.049	-.112	-.046	-.130	.800
.900	-.034	-.068	-.031	-.072	-.054	-.097	-.062	-.114	-.055	-.117	.900
.975	-.031	-.060	-.041	-.063	-.060	-.099					.975
$\alpha = -1°$											
.025	.076	.033	.083	.003	.070	-.021					.025
.050	.048	.011	.078	.018	.054	-.010	.045	-.061			.050
.100	.037	.012	.040	.017	.041	-.005	.039	-.028	.017	-.157	.100
.200	.023	.000	.026	-.006	.021	-.013	.000	-.049	-.007	-.097	.200
.300	.017	-.005	.010	-.016	.008	-.029	.001	-.043	-.013	-.074	.300
.400	-.005	-.024	-.005	-.033	-.005	-.038	-.021	-.060	-.027	-.070	.400
.500	.000	-.021	-.007	-.031	-.024	-.052	-.026	-.076	-.039	-.094	.500
.600	-.010	-.029	-.017	-.038	-.031	-.061	-.048	-.085	-.061	-.110	.600
.700	-.028	-.046	-.038	-.054	-.050	-.068	-.060	-.091	-.069	-.102	.700
.800	-.033	-.052	-.031	-.048	-.056	-.083	-.071	-.092	-.082	-.108	.800
.900	-.042	-.060	-.041	-.060	-.066	-.085	-.072	-.103	-.089	-.106	.900
.975	-.040	-.054	-.049	-.058	-.073	-.092					.975
$\alpha = 0°$											
.025	.058	.058	.063	.063	.033	.033					.025
.050	.030	.030	.062	.062	.027	.027	.004	.004			.050
.100	.024	.024	.029	.029	.020	.020	.005	.005	-.013	-.013	.100
.200	.012	.012	.007	.007	.003	.003	-.013	-.013	-.025	-.025	.200
.300	.003	.003	-.002	-.002	-.010	-.010	-.020	-.020	-.031	-.031	.300
.400	-.012	-.012	-.016	-.016	-.020	-.020	-.040	-.041	-.045	-.045	.400
.500	-.010	-.010	-.018	-.019	-.039	-.039	-.059	-.059	-.066	-.066	.500
.600	-.023	-.023	-.029	-.029	-.044	-.044	-.065	-.065	-.078	-.078	.600
.700	-.037	-.037	-.045	-.045	-.058	-.059	-.071	-.071	-.071	-.071	.700
.800	-.041	-.041	-.038	-.038	-.067	-.066	-.080	-.080	-.081	-.081	.800
.900	-.052	-.051	-.048	-.048	-.077	-.077	-.093	-.093	-.082	-.082	.900
.975	-.045	-.045	-.053	-.053	-.083	-.083					.975
$\alpha = 1°$											
.025	.033	.076	.003	.083	-.021	.070					.025
.050	.011	.048	.018	.078	-.010	.054	-.061	.045			.050
.100	.012	.037	.017	.040	-.005	.041	-.028	.039	-.157	.017	.100
.200	.000	.023	-.006	.026	-.013	.021	-.049	.000	-.097	-.007	.200
.300	-.005	.017	-.016	.010	-.029	.008	-.043	.001	-.074	-.013	.300
.400	-.024	-.005	-.033	-.005	-.038	-.005	-.060	-.021	-.070	-.027	.400
.500	-.021	.000	-.031	-.007	-.052	-.024	-.076	-.026	-.095	-.039	.500
.600	-.029	-.010	-.038	-.017	-.061	-.031	-.085	-.048	-.110	-.061	.600
.700	-.046	-.028	-.054	-.038	-.068	-.050	-.091	-.060	-.102	-.069	.700
.800	-.052	-.033	-.048	-.031	-.083	-.056	-.092	-.071	-.108	-.082	.800
.900	-.060	-.042	-.060	-.041	-.085	-.066	-.103	-.072	-.106	-.090	.900
.975	-.054	-.040	-.058	-.053	-.092	-.073					.975

L-1853

L-1853

TABLE I.- TABULATED PRESSURE COEFFICIENTS - Continued

(a) Wing 1 - Continued

x'/c	C_p at $\frac{y}{b/2}$ of :										x'/c
	.1		.3		.5		.7		.9		
	Upper	Lower	Upper	Lower	Upper	Lower	Upper	Lower	Upper	Lower	
$\alpha = 2^\circ$											
.025	.012	.092	-.060	.098	-.134	.094					.025
.050	-.005	.063	-.020	.091	-.086	.079	-.134	.073			.050
.100	-.001	.052	-.001	.054	-.052	.060	-.123	.062	-.262	.068	.100
.200	-.010	.035	-.014	.043	-.040	.038	-.093	.021	-.221	.035	.200
.300	-.012	.025	-.030	.027	-.047	.030	-.072	.021	-.204	.015	.300
.400	-.033	.007	-.044	.012	-.058	.010	-.082	.001	-.179	.001	.400
.500	-.029	.010	-.042	.005	-.065	-.010	-.093	-.022	-.168	-.017	.500
.600	-.038	-.002	-.050	-.004	-.075	-.021	-.103	-.036	-.165	-.033	.600
.700	-.055	-.016	-.063	-.026	-.079	-.039	-.108	-.049	-.154	-.030	.700
.800	-.059	-.023	-.060	-.020	-.094	-.047	-.111	-.049	-.130	-.046	.800
.900	-.068	-.034	-.072	-.031	-.097	-.054	-.114	-.062	-.117	-.055	.900
.975	-.060	-.031	-.063	-.041	-.099	-.060					.975
$\alpha = 3^\circ$											
.025	-.014	.107	-.113	.111	-.146	.113					.025
.050	-.023	.078	-.095	.101	-.134	.097	-.174	.097			.050
.100	-.013	.066	-.063	.073	-.103	.082	-.164	.082	-.280	.082	.100
.200	-.022	.067	-.024	.061	-.082	.055	-.142	.036	-.282	.049	.200
.300	-.022	.033	-.043	.040	-.073	.044	-.118	.043	-.277	.033	.300
.400	-.040	.019	-.058	.029	-.077	.024	-.115	.015	-.247	.018	.400
.500	-.039	.022	-.054	.020	-.082	.009	-.118	-.008	-.190	.002	.500
.600	-.048	.010	-.063	.009	-.088	-.010	-.120	-.023	-.143	-.012	.600
.700	-.065	-.006	-.072	-.011	-.092	-.020	-.121	-.026	-.140	-.010	.700
.800	-.069	-.012	-.072	-.009	-.103	-.036	-.125	-.034	-.126	-.021	.800
.900	-.077	-.024	-.084	-.020	-.108	-.042	-.129	-.047	-.119	-.031	.900
.975	-.068	-.023	-.068	-.031	-.105	-.046					.975
$\alpha = 4^\circ$											
.025	-.050	.122	-.143	.139	-.165	.133					.025
.050	-.045	.094	-.133	.125	-.154	.114	-.185	.113			.050
.100	-.029	.080	-.118	.092	-.143	.098	-.179	.099	-.154	.098	.100
.200	-.031	.063	-.050	.075	-.135	.074	-.165	.061	-.143	.071	.200
.300	-.034	.057	-.049	.055	-.123	.062	-.144	.060	-.139	.057	.300
.400	-.047	.029	-.067	.044	-.110	.042	-.149	.031	-.143	.041	.400
.500	-.050	.037	-.066	.032	-.106	.028	-.148	.005	-.152	.030	.500
.600	-.054	.023	-.070	.023	-.106	.000	-.146	.000	-.162	.010	.600
.700	-.072	.005	-.080	.005	-.106	-.003	-.144	-.010	-.173	.003	.700
.800	-.077	.000	-.079	.001	-.114	-.020	-.143	-.021	-.174	-.005	.800
.900	-.085	-.012	-.095	-.008	-.121	-.026	-.126	-.030	-.183	-.015	.900
.975	-.074	-.011	-.074	-.021	-.110	-.034					.975
$\alpha = 5^\circ$											
.025	-.093	.137	-.157	.141	-.175	.142					.025
.050	-.076	.107	-.154	.130	-.169	.123	-.191	.127			.050
.100	-.040	.095	-.156	.124	-.170	.113	-.188	.121	-.102	.108	.100
.200	-.039	.079	-.125	.092	-.170	.094	-.183	.080	-.082	.085	.200
.300	-.055	.057	-.082	.067	-.168	.077	-.178	.072	-.079	.071	.300
.400	-.054	.055	-.072	.061	-.165	.061	-.179	.041	-.083	.057	.400
.500	-.060	.048	-.070	.046	-.158	.039	-.179	.026	-.101	.042	.500
.600	-.064	.038	-.075	.034	-.153	.015	-.181	.014	-.121	.027	.600
.700	-.078	.016	-.086	.020	-.143	.008	-.179	.001	-.141	.021	.700
.800	-.086	.010	-.082	.010	-.142	-.008	-.159	-.005	-.149	.011	.800
.900	-.094	-.002	-.108	.002	-.138	-.012	-.143	-.017	-.153	.003	.900
.975	-.082	-.001	-.086	-.010	-.125	-.023					.975

TABLE I.- TABULATED PRESSURE COEFFICIENTS - Continued

(a) Wing 1 - Concluded

x'/c	C_p at $\frac{y}{b/2}$ of :										x'/c
	.1		.3		.5		.7		.9		
	Upper	Lower	Upper	Lower	Upper	Lower	Upper	Lower	Upper	Lower	
$\alpha = 6^\circ$											
.025	-.132	.152	-.172	.162	-.184	.155					.025
.050	-.119	.122	-.174	.152	-.185	.138	-.206	.146			.050
.100	-.044	.112	-.173	.125	-.188	.132	-.208	.134	-.101	.117	.100
.200	-.046	.095	-.175	.103	-.189	.108	-.212	.095	-.086	.097	.200
.300	-.063	.074	-.150	.098	-.191	.096	-.212	.085	-.075	.095	.300
.400	-.062	.057	-.114	.071	-.193	.081	-.216	.062	-.081	.078	.400
.500	-.068	.064	-.071	.062	-.194	.050	-.219	.044	-.097	.068	.500
.600	-.071	.057	-.072	.050	-.195	.039	-.221	.030	-.102	.054	.600
.700	-.086	.031	-.088	.032	-.193	.024	-.211	.020	-.122	.044	.700
.800	-.095	.024	-.084	.029	-.194	.005	-.181	.010	-.141	.032	.800
.900	-.100	.008	-.108	.018	-.189	.002	-.182	-.005	-.122	.021	.900
.975	-.089	.010	-.095	.000	-.173	-.010					.975
$\alpha = 7^\circ$											
.025	-.158	.166	-.191	.174	-.195	.179					.025
.050	-.136	.137	-.185	.165	-.197	.165	-.220	.157			.050
.100	-.081	.127	-.183	.143	-.202	.157	-.221	.147	-.112	.126	.100
.200	-.059	.110	-.197	.128	-.204	.123	-.225	.106	-.102	.113	.200
.300	-.072	.091	-.196	.103	-.206	.108	-.226	.103	-.092	.111	.300
.400	-.066	.068	-.178	.083	-.210	.097	-.228	.082	-.091	.093	.400
.500	-.077	.080	-.118	.089	-.215	.061	-.232	.057	-.094	.083	.500
.600	-.078	.065	-.062	.056	-.222	.056	-.234	.046	-.090	.072	.600
.700	-.093	.052	-.085	.048	-.226	.040	-.211	.036	-.108	.060	.700
.800	-.100	.037	-.086	.050	-.230	.021	-.200	.026	-.112	.046	.800
.900	-.106	.023	-.112	.030	-.231	.015	-.223	.014	-.116	.036	.900
.975	-.093	.024	-.100	.013	-.217	.001					.975
$\alpha = 8^\circ$											
.025	-.179	.181	-.205	.185	-.203	.175					.025
.050	-.158	.152	-.206	.180	-.206	.164	-.229	.163			.050
.100	-.141	.142	-.203	.156	-.210	.163	-.232	.154	-.133	.132	.100
.200	-.057	.124	-.218	.142	-.212	.162	-.237	.117	-.123	.123	.200
.300	-.074	.103	-.229	.121	-.217	.132	-.236	.120	-.113	.121	.300
.400	-.070	.081	-.220	.107	-.222	.103	-.237	.093	-.109	.102	.400
.500	-.082	.093	-.184	.092	-.229	.072	-.241	.071	-.108	.097	.500
.600	-.084	.079	-.083	.086	-.239	.065	-.242	.062	-.096	.085	.600
.700	-.100	.056	-.081	.057	-.246	.052	-.216	.046	-.110	.073	.700
.800	-.104	.057	-.084	.069	-.250	.037	-.210	.040	-.124	.062	.800
.900	-.113	.036	-.113	.043	-.253	.030	-.224	.029	-.118	.049	.900
.975	-.092	.037	-.106	.025	-.242	.014					.975

L-1853

L-1853

TABLE I.- TABULATED PRESSURE COEFFICIENTS - Continued

(b) Wing 2

x'/c	C_p at $\frac{y}{b/2}$ of :										x'/c
	.1		.3		.5		.7		.9		
	Upper	Lower	Upper	Lower	Upper	Lower	Upper	Lower	Upper	Lower	
$\alpha = -4^\circ$											
.025	.083	-.010	.133	-.092	.112	-.104					.025
.050	.059	.034	.096	-.102	.099	-.104	.121	-.127			.050
.100	.037	.005	.063	-.088	.081	-.103	.098	-.129	.117	-.214	.100
.200	.025	.000	.042	-.027	.041	-.103	.054	-.131	.092	-.164	.200
.300	.024	.005	.023	-.008	.025	-.105	.031	-.130	.069	-.138	.300
.400	.007	-.008	.001	-.027	.009	-.100	.011	-.134	.044	-.115	.400
.500	.002	-.020	-.002	-.028	-.016	-.071	.003	-.135	.009		.500
.600	.000	.009	-.008	-.042	-.030	-.058	-.019	-.141	.032	-.113	.600
.700	-.007	-.033	-.024	-.041	-.037	-.056	-.041	-.142	-.007	-.125	.700
.800	-.018	-.044	-.034	-.057	-.053	-.066	-.049	-.143	-.021	-.143	.800
.900	-.005	-.062	-.028	-.068	-.052	-.082	-.061	-.140	-.042	-.148	.900
.975	-.012	-.048		-.064	-.059	-.075					.975
$\alpha = -3^\circ$											
.025	.068	.028	.108	-.066	.092	-.078					.025
.050	.044	.052	.073	-.059	.076	-.079	.100	-.104			.050
.100	.021	.019	.049	-.013	.060	-.071	.079	-.105	.100	-.202	.100
.200	.029	.010	.025	-.011	.025	-.077	.033	-.104	.071	-.151	.200
.300	.009	.002	.005	.000	.005	-.052	.009	-.104	.046	-.141	.300
.400	-.006	.009	-.014	-.020	-.011	-.038	-.010	-.106	.023	-.127	.400
.500	-.010	-.021	-.015	-.018	-.034	-.029	-.022	-.104	-.008		.500
.600	-.012	-.010	-.025	-.032	-.041	-.041	-.038	-.102	-.010	-.102	.600
.700	-.019	-.028	-.038	-.033	-.050	-.056	-.056	-.097	-.028	-.104	.700
.800	-.029	-.038	-.046	-.050	-.063	-.062	-.072	-.097	-.041	-.115	.800
.900	-.018	-.056	-.038	-.061	-.065	-.081	-.075	-.091	-.061	-.122	.900
.975	-.022	-.044		-.061	-.073	-.068					.975
$\alpha = -2^\circ$											
.025	.052	.057	.094	-.032	.069	-.050					.025
.050	.028	.071	.061	.009	.053	-.043	.078	-.090			.050
.100	.008	.031	.026	.025	.034	-.030	.055	-.082	.080	-.152	.100
.200	-.001	.023	.009	.000	.001	-.013	.010	-.066	.049	-.124	.200
.300	.001	.013	-.010	.006	-.010	-.009	-.010	-.045	.026	-.114	.300
.400	-.018	.001	-.029	-.007	-.027	-.020	-.030	-.040	-.005	-.116	.400
.500	-.021	-.011	-.030	-.010	-.050	-.022	-.040	-.043	-.028		.500
.600	-.021	.003	-.038	-.022	-.065	-.033	-.062	-.054	-.041	-.108	.600
.700	-.030	-.019	-.050	-.025	-.061	-.049	-.071	-.061	-.055	-.106	.700
.800	-.040	-.029	-.057	-.043	-.080	-.054	-.082	-.071	-.067	-.103	.800
.900	-.028	-.048	-.049	-.053	-.081	-.074	-.099	-.071	-.082	-.110	.900
.975	-.031	-.038		-.050	-.085	-.060					.975
$\alpha = -1^\circ$											
.025	.035	.078	.075	.046	.040	-.010					.025
.050	.013	.089	.041	.039	.026	.021	.048	-.034			.050
.100	.005	.045	.008	.038	.010	.025	.033	.000	.049	-.080	.100
.200	-.012	.034	-.010	.012	-.020	.014	-.009	.002	.021	-.074	.200
.300	-.004	.029	-.027	.005	-.023	.006	-.039	-.004	.000	-.055	.300
.400	-.027	.013	-.043	.000	-.050	-.007	-.050	-.017	-.029	-.070	.400
.500	-.029	-.001	-.043	.000	-.066	-.008	-.062	-.029	-.053		.500
.600	-.031	.012	-.050	-.012	-.081	-.021	-.080	-.040	-.076	-.074	.600
.700	-.040	-.009	-.059	-.014	-.072	-.036	-.097	-.049	-.077	-.078	.700
.800	-.048	-.020	-.067	-.038	-.098	-.044	-.099	-.057	-.092	-.081	.800
.900	-.038	-.039	-.060	-.044	-.092	-.066	-.110	-.060	-.108	-.091	.900
.975	-.038	-.029		-.043	-.097	-.052					.975

TABLE I.- TABULATED PRESSURE COEFFICIENTS - Continued

(b) Wing 2 - Continued

x'/c	C_p at $\frac{y}{b/2}$ of :										x'/c
	.1		.3		.5		.7		.9		
	Upper	Lower	Upper	Lower	Upper	Lower	Upper	Lower	Upper	Lower	
$\alpha = 0°$											
.025	.017	.098	.043	.073	-.001	.066					.025
.050	-.001	.103	.013	.075	-.006	.068	.006	.043			.050
.100	-.013	.062	-.008	.056	-.015	.050	-.004	.036	.010	.000	.100
.200	-.021	.048	-.031	.027	-.040	.032	-.032	.038	-.009	-.004	.200
.300	-.011	.040	-.041	.035	-.046	.022	-.054	.019	-.031	-.006	.300
.400	-.032	.025	-.056	.010	-.063	.008	-.074	.002	-.052	-.034	.400
.500	-.038	.011	-.053	.014	-.077	.005	-.084	-.011	-.084		.500
.600	-.038	.021	-.061	-.002	-.092	-.009	-.102	-.027	-.090	-.049	.600
.700	-.048	.001	-.067	-.004	-.084	-.022	-.113	-.032	-.104	-.054	.700
.800	-.056	-.010	-.075	-.030	-.106	-.035	-.123	-.040	-.114	-.059	.800
.900	-.045	-.030	-.071	-.036	-.101	-.055	-.125	-.048	-.132	-.070	.900
.975	-.043	-.021		-.035	-.109	-.039					.975
$\alpha = 1°$											
.025	-.009	.113	-.023	.092	-.055	.092					.025
.050	-.025	.117	-.025	.092	-.057	.089	-.062	.081			.050
.100	-.026	.076	-.026	.070	-.058	.071	-.059	.060	-.064	.041	.100
.200	-.030	.061	-.047	.040	-.062	.052	-.064	.059	-.060	.031	.200
.300	-.031	.052	-.059	.050	-.069	.040	-.081	.038	-.072	.025	.300
.400	-.040	.036	-.068	.023	-.079	.022	-.084	.021	-.085	-.007	.400
.500	-.048	.022	-.067	.030	-.102	.021	-.104	.001	-.111		.500
.600	-.047	.031	-.072	.010	-.108	.002	-.123	-.009	-.112	-.028	.600
.700	-.055	.011	-.075	.008	-.105	-.010	-.136	-.014	-.135	-.030	.700
.800	-.066	-.001	-.083	-.020	-.116	-.028	-.143	-.032	-.141	-.031	.800
.900	-.057	-.021	-.083	-.026	-.116	.042	-.146	-.037	-.153	-.048	.900
.975	-.052	-.014		-.025	-.117	-.030					.975
$\alpha = 2°$											
.025	-.074	.129	-.084	.107	-.114	.107					.025
.050	-.041	.135	-.067	.104	-.119	.103	-.156	.103			.050
.100	-.039	.092	-.047	.082	-.098	.085	-.131	.080	-.216	.070	.100
.200	-.040	.073	-.059	.054	-.085	.066	-.100	.074	-.177	.059	.200
.300	-.045	.063	-.073	.062	-.089	.054	-.105	.053	-.150	.052	.300
.400	-.046	.049	-.079	.035	-.100	.038	-.117	.034	-.146	.018	.400
.500	-.059	.031	-.078	.042	-.114	.035	-.125	.021	-.150		.500
.600	-.055	.041	-.082	.021	-.120	.017	-.139	.009	-.136	-.007	.600
.700	-.064	.022	-.084	.020	-.119	.000	-.153	-.006	-.164	-.013	.700
.800	-.074	.009	-.094	-.008	-.126	-.017	-.160	-.020	-.164	-.021	.800
.900	-.066	-.010	-.094	-.016	-.129	-.029	-.169	-.025	-.172	-.037	.900
.975	-.062	-.005		-.016	-.122	-.020					.975
$\alpha = 3°$											
.025	-.105	.147	-.142	.124	-.190	.124					.025
.050	-.063	.155	-.122	.116	-.193	.118	-.258	.123			.050
.100	-.052	.106	-.088	.095	-.170	.102	-.219	.097	-.270	.089	.100
.200	-.050	.091	-.067	.071	-.107	.081	-.197	.084	-.266	.074	.200
.300	-.055	.074	-.088	.069	-.106	.070	-.185	.066	-.248	.066	.300
.400	-.053	.062	-.092	.049	-.114	.052	-.149	.050	-.238	.025	.400
.500	-.066	.044	-.090	.054	-.128	.046	-.141	.036	-.227		.500
.600	-.064	.054	-.093	.028	-.137	.032	-.152	.022	-.219	.007	.600
.700	-.071	.033	-.093	.029	-.131	.009	-.164	.005	-.231	.000	.700
.800	-.082	.020	-.103	.006	-.141	-.005	-.170	-.009	-.231	-.005	.800
.900	-.074	-.001	-.103	-.007	-.138	-.016	-.180	-.014	-.225	-.021	.900
.975	-.073	.005		-.007	-.125	-.009					.975

L-1853

L-1853

TABLE I.- TABULATED PRESSURE COEFFICIENTS - Continued

(b) Wing 2 - Concluded

x'/c	C_p at $\frac{y}{b/2}$ of :										x'/c
	.1		.3		.5		.7		.9		
	Upper	Lower	Upper	Lower	Upper	Lower	Upper	Lower	Upper	Lower	
$\alpha = 4^\circ$											
.025	-.130	.164	-.196	.137	-.233	.142					.025
.050	-.091	.174	-.191	.132	-.225	.133	-.290	.138			.050
.100	-.063	.123	-.127	.106	-.209	.114	-.265	.108	-.277	.102	.100
.200	-.061	.103	-.078	.082	-.158	.095	-.247	.097	-.288	.087	.200
.300	-.063	.091	-.092	.081	-.138	.082	-.236	.082	-.278	.081	.300
.400	-.062	.075	-.103	.063	-.137	.069	-.235	.067	-.261	.041	.400
.500	-.073	.063	-.102	.070	-.144	.061	-.229	.053	-.254		.500
.600	-.074	.070	-.103	.043	-.150	.045	-.215	.033	-.238	.026	.600
.700	-.079	.046	-.104	.040	-.142	.021	-.197	.021	-.242	.020	.700
.800	-.089	.030	-.113	.023	-.150	.009	-.192	.002	-.225	.010	.800
.900	-.085	.010	-.112	.006	-.145	-.005	-.175	.002	-.215	-.008	.900
.975	-.082	.018		.003	-.123	.000					.975
$\alpha = 5^\circ$											
.025	-.148	.181	-.234	.152	-.221	.154					.025
.050	-.113	.192	-.216	.143	-.226	.147	-.288	.145			.050
.100	-.080	.135	-.169	.120	-.219	.128	-.277	.121	-.262	.116	.100
.200	-.074	.119	-.121	.096	-.190	.106	-.265	.110	-.280	.106	.200
.300	-.076	.103	-.118	.096	-.174	.098	-.240	.097	-.276	.102	.300
.400	-.073	.091	-.128	.074	-.166	.082	-.237	.082	-.265	.060	.400
.500	-.082	.075	-.122	.082	-.165	.073	-.228	.062	-.254		.500
.600	-.083	.082	-.121	.058	-.166	.060	-.222	.046	-.237	.040	.600
.700	-.087	.060	-.121	.050	-.153	.032	-.213	.031	-.235	.030	.700
.800	-.096	.035	-.128	.039	-.164	.022	-.151	.017	-.228	.024	.800
.900	-.093	.021	-.129	.016	-.155	.007	-.123	.021	-.226	.007	.900
.975	-.099	.029		.014	-.131	.011					.975
$\alpha = 6^\circ$											
.025	-.172	.198	-.225	.165	-.230	.167					.025
.050	-.141	.205	-.236	.153	-.231	.158	-.270	.155			.050
.100	-.126	.154	-.200	.136	-.224	.144	-.274	.133	-.147	.129	.100
.200	-.101	.136	-.194	.110	-.201	.118	-.267	.128	-.134	.115	.200
.300	-.082	.119	-.182	.108	-.183	.114	-.252	.113	-.122	.111	.300
.400	-.083	.105	-.176	.092	-.176	.094	-.246	.094	-.117	.071	.400
.500	-.086	.088	-.154	.098	-.181	.090	-.244	.079	-.137		.500
.600	-.090	.098	-.134	.071	-.178	.069	-.240	.062	-.131	.056	.600
.700	-.092	.072	-.118	.064	-.170	.046	-.204	.044	-.158	.047	.700
.800	-.102	.046	-.121	.053	-.175	.039	-.182	.036	-.174	.042	.800
.900	-.099	.032	-.121	.030	-.168	.020	-.192	.034	-.184	.029	.900
.975	-.092	.040		.029	-.130	.024					.975

TABLE I.- TABULATED PRESSURE COEFFICIENTS - Continued

(c) Wing 3

x'/c	C_p at $\frac{y}{b/2}$ of :										x'/c
	.1		.3		.5		.7		.9		
	Upper	Lower	Upper	Lower	Upper	Lower	Upper	Lower	Upper	Lower	
$\alpha = -6^\circ$											
.025	.062	-.008	.136	-.081	.162	-.102					.025
.050	.034	-.004	.096	-.096	.122	-.106	.149	-.135			.050
.100	.023	.010	.060	-.108	.115	-.107	.131	-.138	.155	-.176	.100
.200	.003	.025	.008	-.083	.059	-.109	.083	-.137	.122	-.174	.200
.300	-.005	.004	.007	.005	.008	-.125	.043	-.144		-.166	.300
.400	-.015	-.012	-.029	.002	-.004	-.140	.022	-.149	.058	-.145	.400
.500	-.001	.033	-.041	-.014	-.019	-.131	-.001	-.158		-.124	.500
.600	.008	-.023	-.038	-.032	-.024	-.102	-.022	-.163	.033	-.112	.600
.700	-.012	-.024	-.045	-.028	-.048	-.053	-.041	-.169	.002	-.111	.700
.800	-.029	-.041	-.041	-.048	-.053	-.023	-.052	-.175	-.016	-.116	.800
.900	-.009	-.061	-.035	-.067	-.061	-.040	-.069	-.157	-.066	-.132	.900
.975	-.011	-.057	-.023	-.071	-.071	-.055					.975
$\alpha = -5^\circ$											
.025	.045	.022	.111	-.058	.146	-.073					.025
.050	.020	.023	.069	-.081	.100	-.076	.136	-.112			.050
.100	.009	.032	.050	-.089	.103	-.081	.113	-.117	.141	-.161	.100
.200	-.010	.041	-.009	.032	.036	-.086	.070	-.115	.109	-.166	.200
.300	-.015	.010	-.011	-.006	-.005	-.104	.025	-.125		-.162	.300
.400	-.016	-.002	-.029	.004	-.029	-.107	-.003	-.133	.039	-.149	.400
.500	-.011	.039	-.050	-.010	-.042	-.067	-.028	-.140		-.126	.500
.600	-.012	-.014	-.049	-.025	-.049	-.021	-.040	-.143	.010	-.112	.600
.700	-.021	-.016	-.060	-.022	-.061	-.021	-.059	-.146	-.018	-.111	.700
.800	-.040	-.034	-.055	-.044	-.066	-.030	-.069	-.148	-.034	-.120	.800
.900	-.020	-.056	-.050	-.062	-.075	-.045	-.071	-.137	-.078	-.138	.900
.975	-.024	-.052	-.035	-.066	-.088	-.058					.975
$\alpha = -4^\circ$											
.025	.028	.054	.102	-.048	.132	-.043					.025
.050	.005	.052	.061	-.071	.082	-.049	.119	-.092			.050
.100	-.005	.050	.011	-.013	.070	-.058	.098	-.093	.125	-.144	.100
.200	-.021	.052	-.026	.032	.014	-.065	.045	-.094	.093	-.148	.200
.300	-.029	.021	-.026	.002	-.026	-.062	.007	-.102		-.150	.300
.400	-.038	.005	-.052	.009	-.049	-.052	-.028	-.106	.014	-.147	.400
.500	-.022	.058	-.063	-.003	-.063	-.007	-.052	-.107		-.126	.500
.600	-.024	.000	-.062	-.017	-.067	-.006	-.070	-.113	-.021	-.111	.600
.700	-.029	-.008	-.070	-.012	-.072	-.025	-.076	-.113	-.041	-.109	.700
.800	-.038	-.028	-.066	-.038	-.086	-.030	-.085	-.101	-.058	-.119	.800
.900	-.031	-.050	-.061	-.054	-.090	-.045	-.106	-.085	-.097	-.141	.900
.975	-.034	-.045	-.045	-.058	-.100	-.054					.975
$\alpha = -3^\circ$											
.025	.012	.075	.070	-.029	.109	-.010					.025
.050	-.006	.077	.032	-.014	.061	-.017	.098	-.065			.050
.100	-.018	.066	-.010	.032	.042	-.018	.073	-.064	.109	-.120	.100
.200	-.031	.064	-.046	.042	-.024	-.019	.022	-.063	.075	-.134	.200
.300	-.038	.039	-.045	.009	-.046	-.004	-.019	-.074		-.137	.300
.400	-.046	.018	-.061	.019	-.060	-.002	-.046	-.077	-.008	-.140	.400
.500	-.032	.068	-.079	.003	-.083	-.001	-.069	-.070		-.132	.500
.600	-.034	.005	-.074	-.009	-.086	-.005	-.093	-.054	-.039	-.117	.600
.700	-.046	.000	-.081	-.004	-.088	-.022	-.109	-.050	-.068	-.109	.700
.800	-.056	-.021	-.079	-.030	-.104	-.026	-.114	-.051	-.078	-.118	.800
.900	-.041	-.042	-.073	-.047	-.104	-.040	-.122	-.041	-.117	-.117	.900
.975	-.043	-.040	-.058	-.051	-.113	-.042					.975

L-1853

L-1853

TABLE I.- TABULATED PRESSURE COEFFICIENTS - Continued

(c) Wing 3 - Continued

x'/c	C_p at $\frac{y}{b/2}$ of:										x'/c
	.1		.3		.5		.7		.9		
	Upper	Lower	Upper	Lower	Upper	Lower	Upper	Lower	Upper	Lower	
$\alpha = -2^\circ$											
.025	-.002	.096	.050	.032	.082	.014					.025
.050	-.025	.102	.013	.045	.035	.017	.075	-.050			.050
.100	-.032	.093	-.038	.044	.015	.027	.051	-.029	.090	-.119	.100
.200	-.046	.077	-.064	.055	-.039	.031	-.004	-.004	.055	-.122	.200
.300	-.050	.042	-.063	.019	-.068	.029	-.042	-.005		-.122	.300
.400	-.054	.027	-.073	.031	-.079	.012	-.070	-.011	-.031	-.123	.400
.500	-.043	.079	-.093	.012	-.101	.008	-.091	-.021		-.118	.500
.600	-.044	.021	-.087	.001	-.109	.003	-.104	-.023	-.080	-.119	.600
.700	-.057	.010	-.091	.004	-.107	-.012	-.124	-.030	-.095	-.109	.700
.800	-.064	-.011	-.091	-.022	-.113	-.016	-.134	-.040	-.108	-.091	.800
.900	-.052	-.034	-.090	-.039	-.114	-.031	-.150	-.038	-.139	-.114	.900
.975	-.054	-.031	-.065	-.042	-.125	-.035					.975
$\alpha = -1^\circ$											
.025	-.014	.124	.023	.074	.048	.064					.025
.050	-.052	.120	-.008	.070	.009	.067	.045	.010			.050
.100	-.045	.105	-.057	.061	-.010	.065	.029	.035	.070	-.080	.100
.200	-.057	.092	-.081	.067	-.064	.049	-.036	.044	.036	-.079	.200
.300	-.060	.054	-.082	.032	-.094	.041	-.067	.023		-.078	.300
.400	-.063	.037	-.091	.040	-.099	.026	-.093	.007	-.052	-.081	.400
.500	-.053	.093	-.110	.023	-.112	.021	-.111	-.010		-.064	.500
.600	-.055	.030	-.098	.010	-.119	.011	-.126	-.009	-.097	-.061	.600
.700	-.067	.020	-.097	.014	-.123	-.003	-.139	-.016	-.117	-.030	.700
.800	-.080	-.004	-.100	-.015	-.122	-.010	-.147	-.032	-.131	-.003	.800
.900	-.061	-.027	-.101	-.030	-.132	-.021	-.164	-.031	-.157	-.027	.900
.975	-.061	-.022	-.067	-.034	-.135	-.029					.975
$\alpha = 0^\circ$											
.025	-.028	.154	-.022	.098	-.008	.086					.025
.050	-.072	.136	-.039	.080	-.031	.093	-.001	.062			.050
.100	-.059	.123	-.072	.072	-.043	.085	-.004	.065	.036	-.006	.100
.200	-.071	.105	-.010	.081	-.089	.066	-.069	.071	.006	.001	.200
.300	-.063	.069	-.100	.049	-.116	.058	-.096	.040		.000	.300
.400	-.073	.052	-.103	.050	-.124	.008	-.124	.021	-.080	-.010	.400
.500	-.061	.106	-.123	.037	-.126	.032	-.141	.006		.003	.500
.600	-.065	.040	-.103	.023	-.137	.024	-.151	.005	-.127	-.011	.600
.700	-.078	.030	-.105	.026	-.146	.003	-.164	-.008	-.147	.005	.700
.800	-.092	.002	-.108	-.031	-.136	-.001	-.170	-.023	-.158	.026	.800
.900	-.077	-.017	-.110	-.019	-.143	-.008	-.178	-.019	-.179	-.014	.900
.975	-.072	-.013	-.063	-.024	-.141	-.020					.975
$\alpha = 1^\circ$											
.025	-.054	.169	-.108	.108	-.081	.112					.025
.050	-.096	.155	-.089	.085	-.072	.106	-.062	.091			.050
.100	-.070	.139	-.083	.077	-.082	.102	-.051	.086	-.013	.046	.100
.200	-.081	.116	-.122	.091	-.115	.081	-.096	.091	-.034	.033	.200
.300	-.069	.081	-.117	.056	-.141	.070	-.123	.055		.029	.300
.400	-.081	.064	-.117	.061	-.148	.051	-.147	.038	-.106	.010	.400
.500	-.071	.122	-.133	.049	-.147	.044	-.164	.021		.012	.500
.600	-.073	.050	-.112	.031	-.148	.038	-.176	.018	-.158	.002	.600
.700	-.088	.040	-.113	.034	-.156	.015	-.185	.003	-.171	.022	.700
.800	-.101	.008	-.106	.008	-.152	.009	-.188	-.010	-.183	.030	.800
.900	-.090	-.008	-.105	-.010	-.151	.001	-.199	-.003	-.194	.000	.900
.975	-.080	-.005	-.032	-.017	-.139	-.009					.975

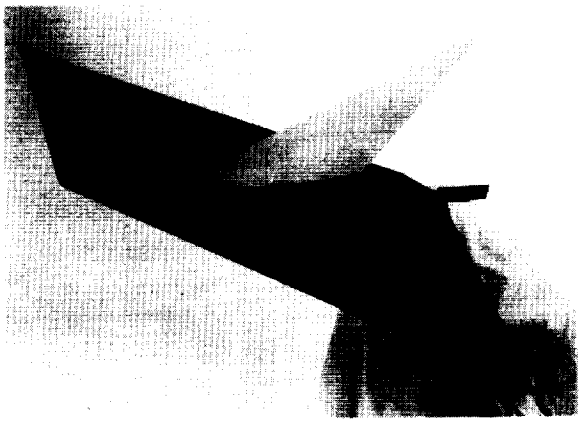
TABLE I.- TABULATED PRESSURE COEFFICIENTS - Concluded

(c) Wing 3 - Concluded

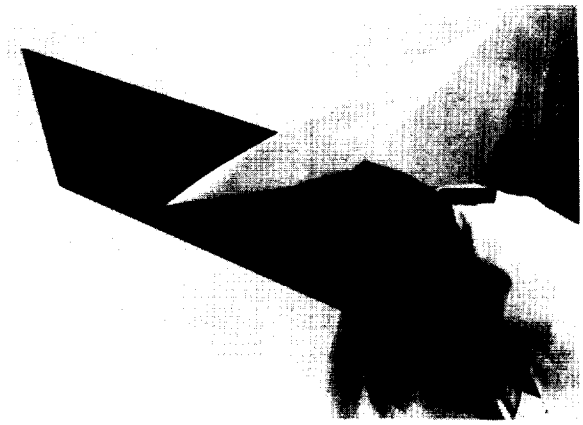
x'/c	C_p at $\frac{y}{b/2}$ of :										x'/c
	.1		.3		.5		.7		.9		
	Upper	Lower	Upper	Lower	Upper	Lower	Upper	Lower	Upper	Lower	
$\alpha = 2^\circ$											
.025	-.062	.186	-.179	.120	-.177	.128					.025
.050	-.109	.176	-.157	.099	-.174	.115	-.183	.108			.050
.100	-.083	.157	-.113	.085	-.134	.115	-.166	.105	-.152	.067	.100
.200	-.088	.134	-.131	.098	-.135	.098	-.143	.105	-.110	.054	.200
.300	-.074	.099	-.135	.071	-.162	.086	-.152	.070		.044	.300
.400	-.087	.079	-.130	.070	-.168	.064	-.163	.051	-.144	.031	.400
.500	-.079	.138	-.140	.061	-.170	.057	-.179	.034		.026	.500
.600	-.081	.062	-.120	.041	-.164	.050	-.193	.028	-.185	.021	.600
.700	-.095	.052	-.125	.043	-.170	.024	-.205	.013	-.198	.038	.700
.800	-.107	.019	-.127	.019	-.169	.023	-.213	.003	-.208	.059	.800
.900	-.098	.003	-.124	-.002	-.162	.013	-.220	.008	-.204	.015	.900
.975	-.089	.000	-.054	-.005	-.129	.001					.975
$\alpha = 3^\circ$											
.025	-.110	.209	-.246	.134	-.245	.142					.025
.050	-.124	.197	-.210	.112	-.232	.129	-.240	.125			.050
.100	-.103	.174	-.196	.097	-.199	.128	-.210	.114	-.240	.081	.100
.200	-.097	.153	-.137	.112	-.197	.110	-.207	.115	-.208	.071	.200
.300	-.090	.113	-.143	.082	-.200	.090	-.213	.085		.061	.300
.400	-.095	.093	-.142	.081	-.180	.076	-.219	.065	-.194	.044	.400
.500	-.090	.155	-.147	.074	-.182	.070	-.228	.043		.036	.500
.600	-.089	.074	-.132	.055	-.174	.062	-.235	.040	-.221	.031	.600
.700	-.105	.063	-.133	.054	-.177	.035	-.251	.030	-.232	.049	.700
.800	-.114	.029	-.134	.032	-.177	.033	-.253	.013	-.226	.070	.800
.900	-.108	.013	-.132	.009	-.167	.025	-.230	.021	-.219	.027	.900
.975	-.100	.018	-.063	.004	-.125	.010					.975
$\alpha = 4^\circ$											
.025	-.136	.232	-.281	.144	-.281	.155					.025
.050	-.149	.218	-.261	.115	-.278	.142	-.280	.134			.050
.100	-.127	.193	-.236	.108	-.256	.139	-.252	.123	-.278	.094	.100
.200	-.105	.173	-.170	.125	-.238	.122	-.242	.132	-.275	.085	.200
.300	-.102	.132	-.145	.092	-.246	.110	-.243	.199		.073	.300
.400	-.102	.108	-.153	.094	-.241	.090	-.225	.074	-.254	.053	.400
.500	-.101	.173	-.155	.087	-.216	.083	-.259	.056		.049	.500
.600	-.101	.091	-.139	.067	-.196	.073	-.265	.052	-.251	.042	.600
.700	-.114	.076	-.140	.064	-.199	.046	-.269	.039	-.250	.066	.700
.800	-.125	.040	-.141	.045	-.198	.047	-.271	.026	-.240	.092	.800
.900	-.120	.024	-.136	.019	-.180	.035	-.195	.032	-.243	.041	.900
.975	-.107	.030	-.068	.013	-.137	.022					.975

L-1853

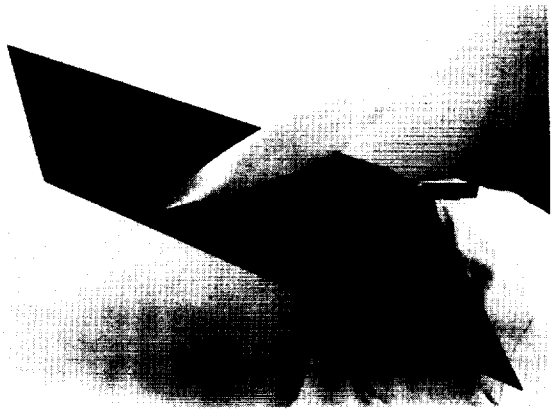
L-1853



(a) Wing 1.
$C_{L,design} = 0.$



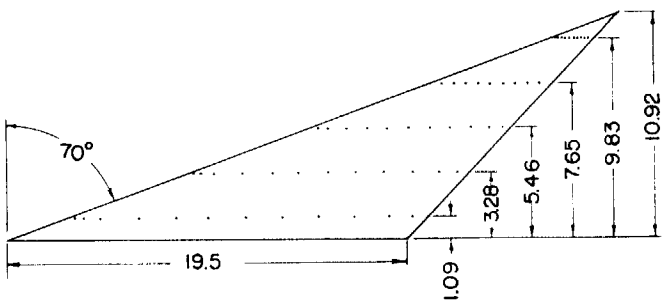
(b) Wing 2.
$C_{L,design} = 0.08.$



(c) Wing 3.
$C_{L,design} = 0.16.$

L-62-43

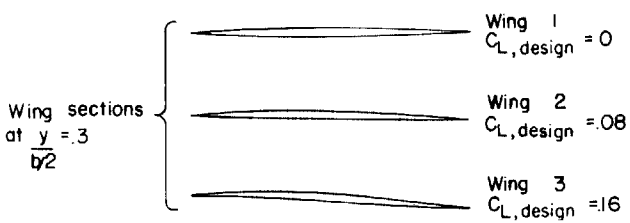
Figure 1.- Photographs of the models mounted on the reflection plate.



Wing layout showing orifice locations

Camber surface ordinates, $\frac{z}{C_{L,design}}$

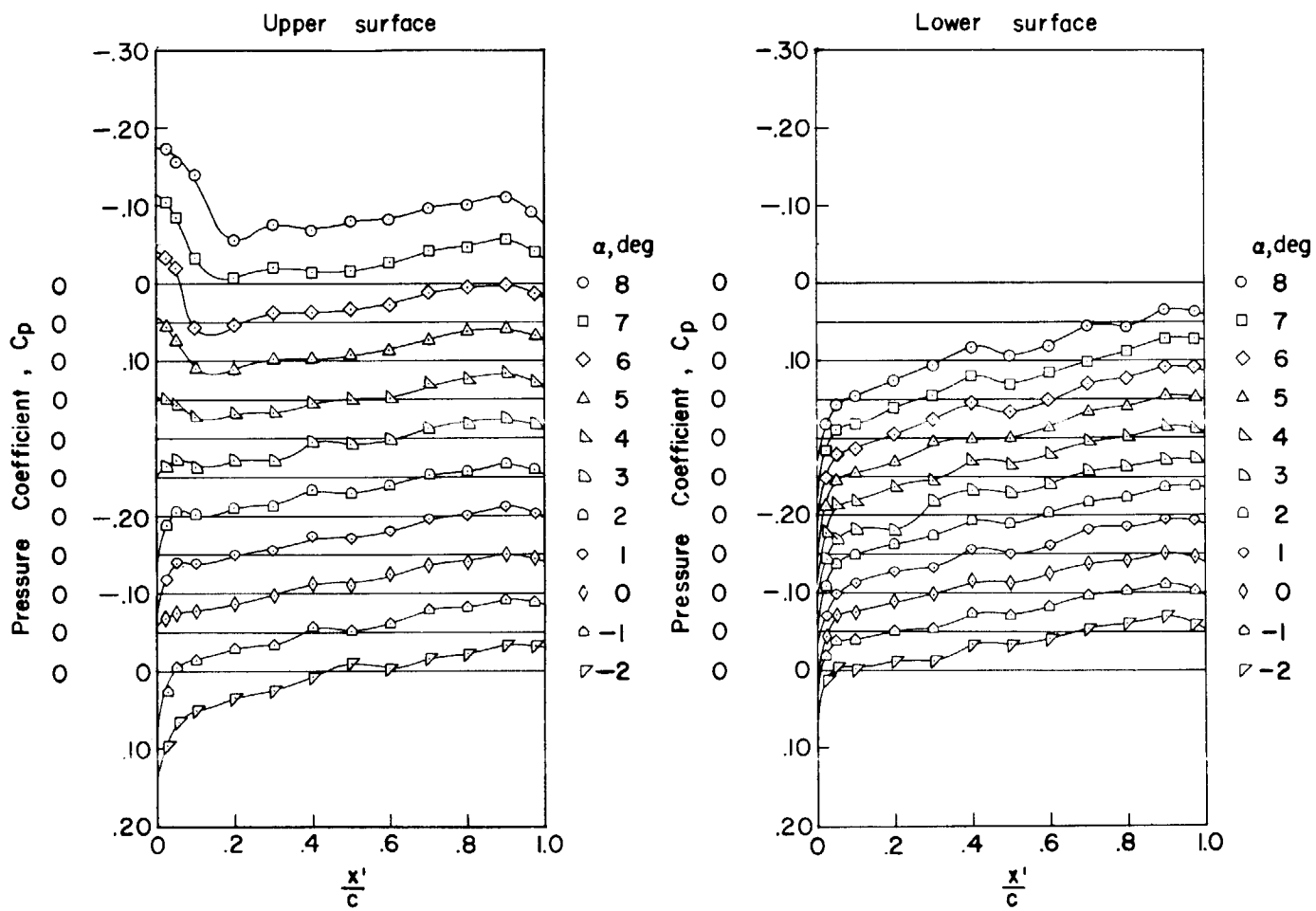
$\frac{y}{b/2}$	x'/c												
	0	.025	.050	.100	.200	.300	.400	.500	.600	.700	.800	.900	1.000
.1	4.70	4.78	4.78	4.60	3.70	2.62	1.29	-.12	-1.60	-3.00	-4.30	-5.63	-6.89
.3	.51	.67	.77	.87	.69	.33	-.12	-.66	-1.60	-1.95	-2.65	-3.40	-4.05
.5	.84	.96	1.65	1.20	1.37	1.35	1.20	1.00	.75	.45	.12	-.24	-.64
.7	1.14	1.26	1.34	1.50	1.70	1.80	1.86	1.85	1.80	1.71	1.62	1.50	1.26
.9	1.44	1.49	1.52	1.60	1.72	1.83	1.94	2.01	2.07	2.13	2.18	2.21	2.24



Wing thickness, t (3% circular-arc sections)

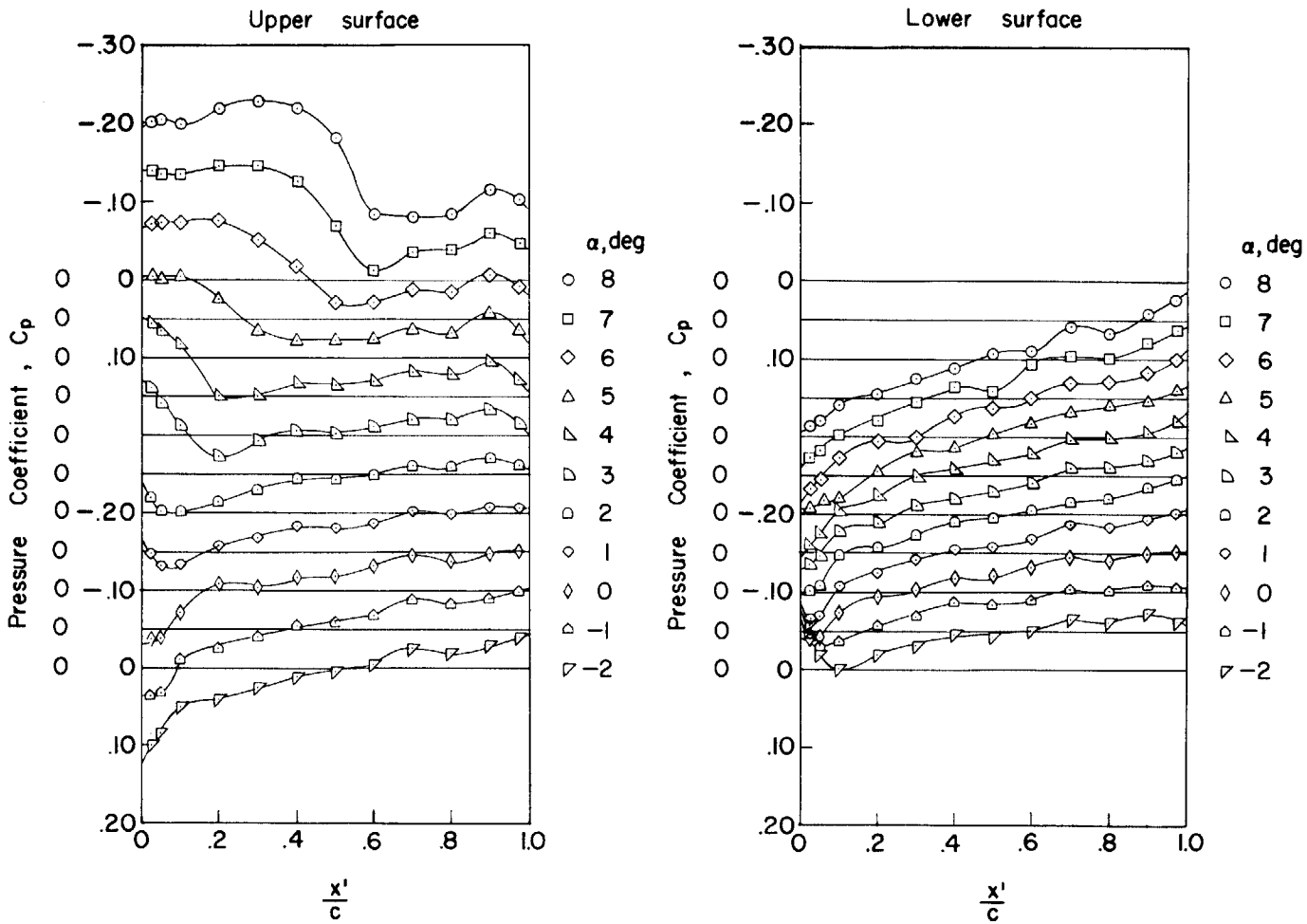
$\frac{y}{b/2}$	x'/c												
	0	.025	.050	.100	.200	.300	.400	.500	.600	.700	.800	.900	1.000
.1	0	.053	.095	.190	.338	.440	.517	.528	.517	.440	.338	.190	0
.3	0	.041	.077	.147	.262	.342	.393	.410	.393	.342	.262	.147	0
.5	0	.029	.055	.105	.187	.244	.281	.292	.281	.244	.187	.105	0
.7	0	.017	.033	.063	.112	.146	.168	.175	.168	.146	.112	.063	0
.9	0	.006	.011	.021	.037	.049	.056	.058	.056	.049	.037	.021	0

Figure 2.- Planform and typical sections for the series of wings. All dimensions in inches.



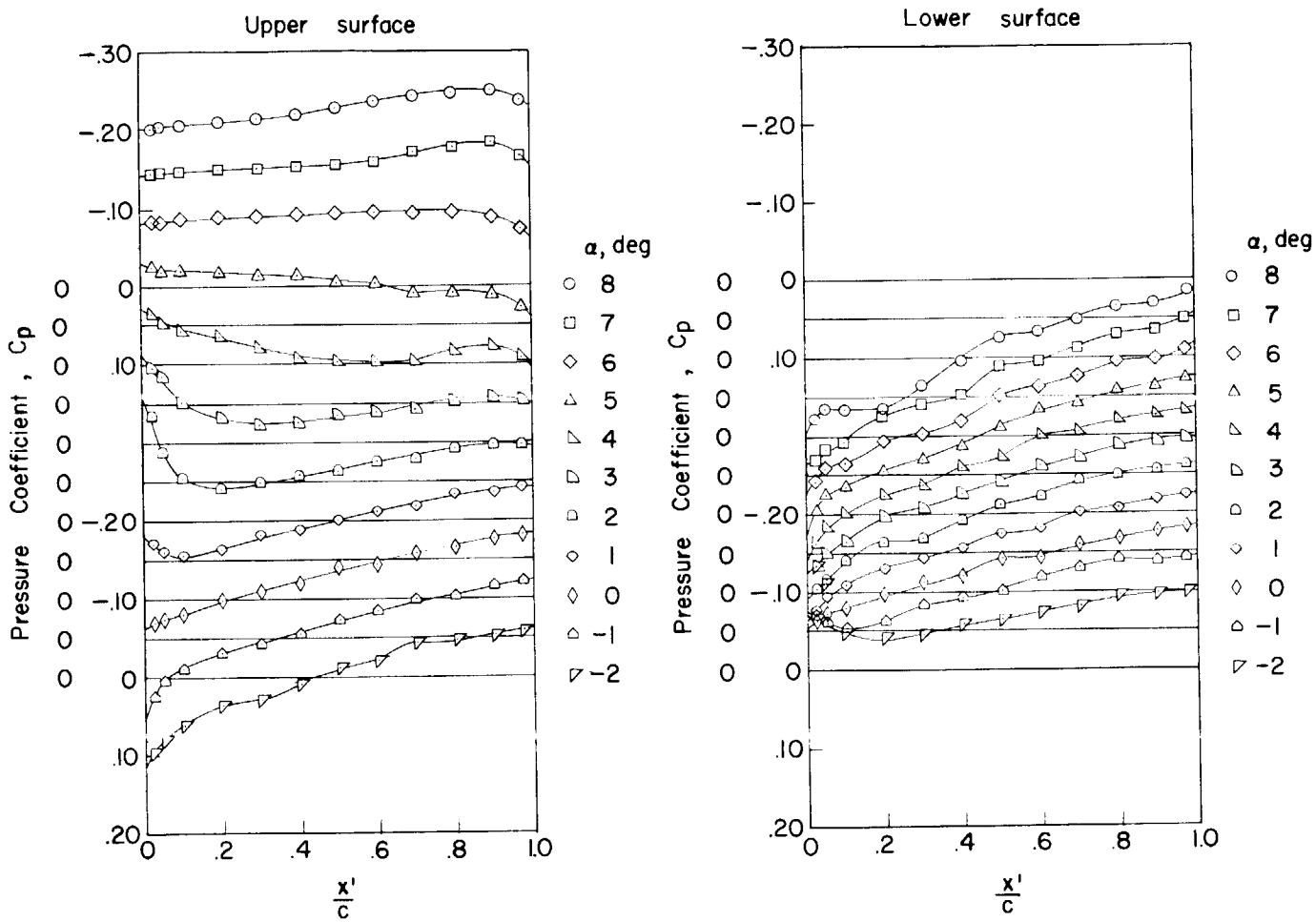
(a) $\frac{y}{b/2} = 0.1.$

Figure 3.- Pressure distributions measured on the surface of wing 1.



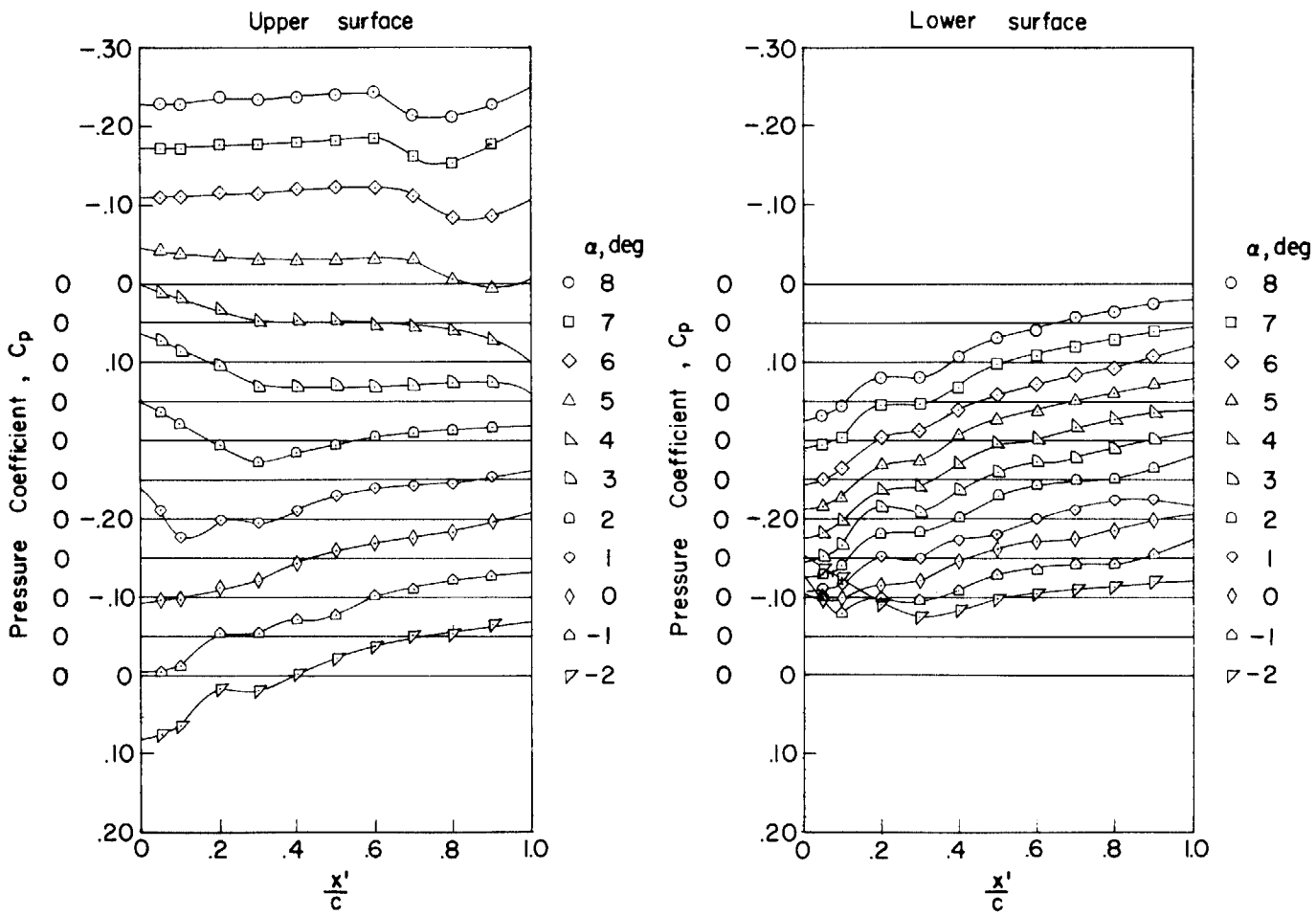
(b) $\frac{y}{b/2} = 0.3$.

Figure 3.- Continued.



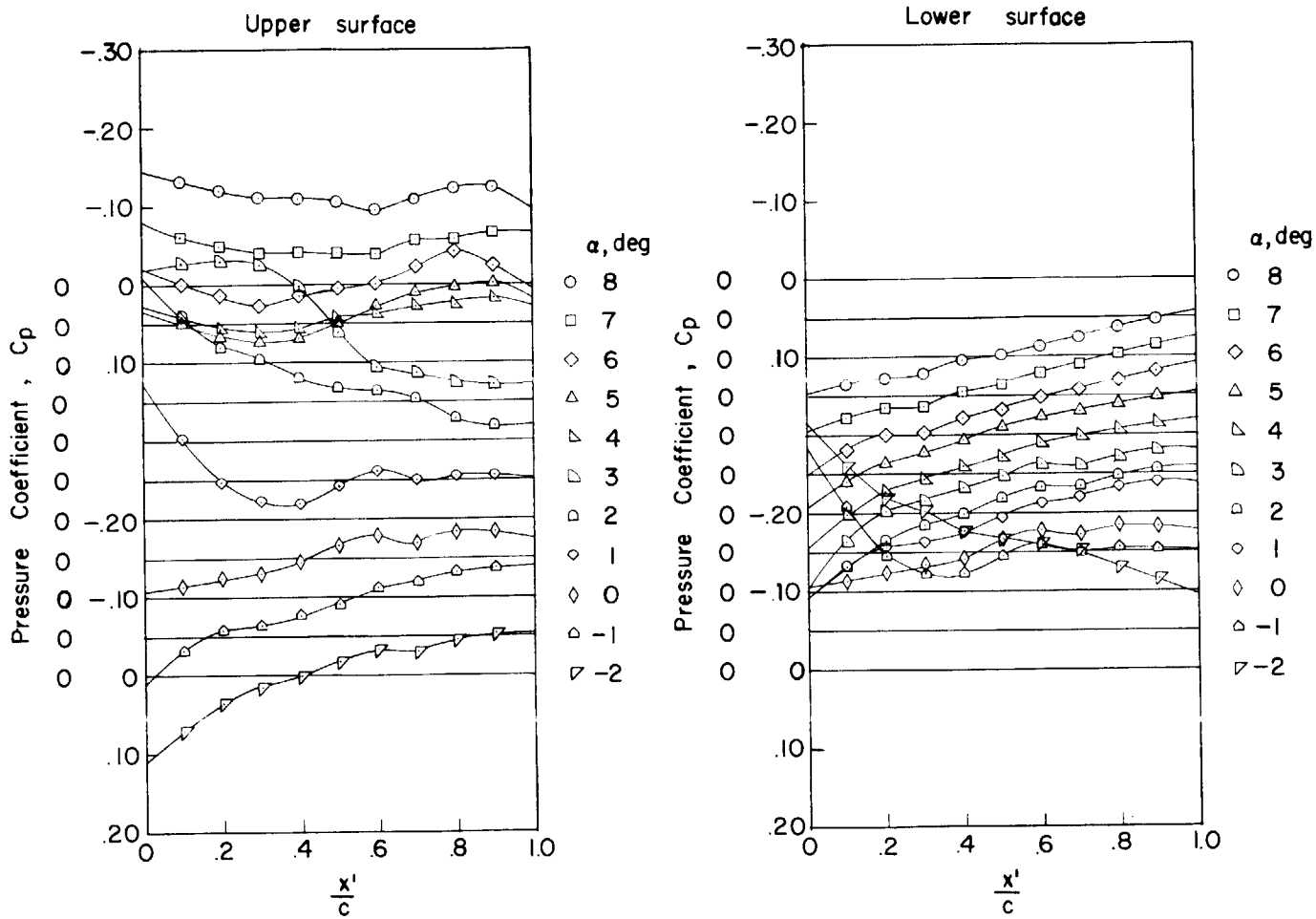
(c) $\frac{y}{b/2} = 0.5$.

Figure 3.- Continued.



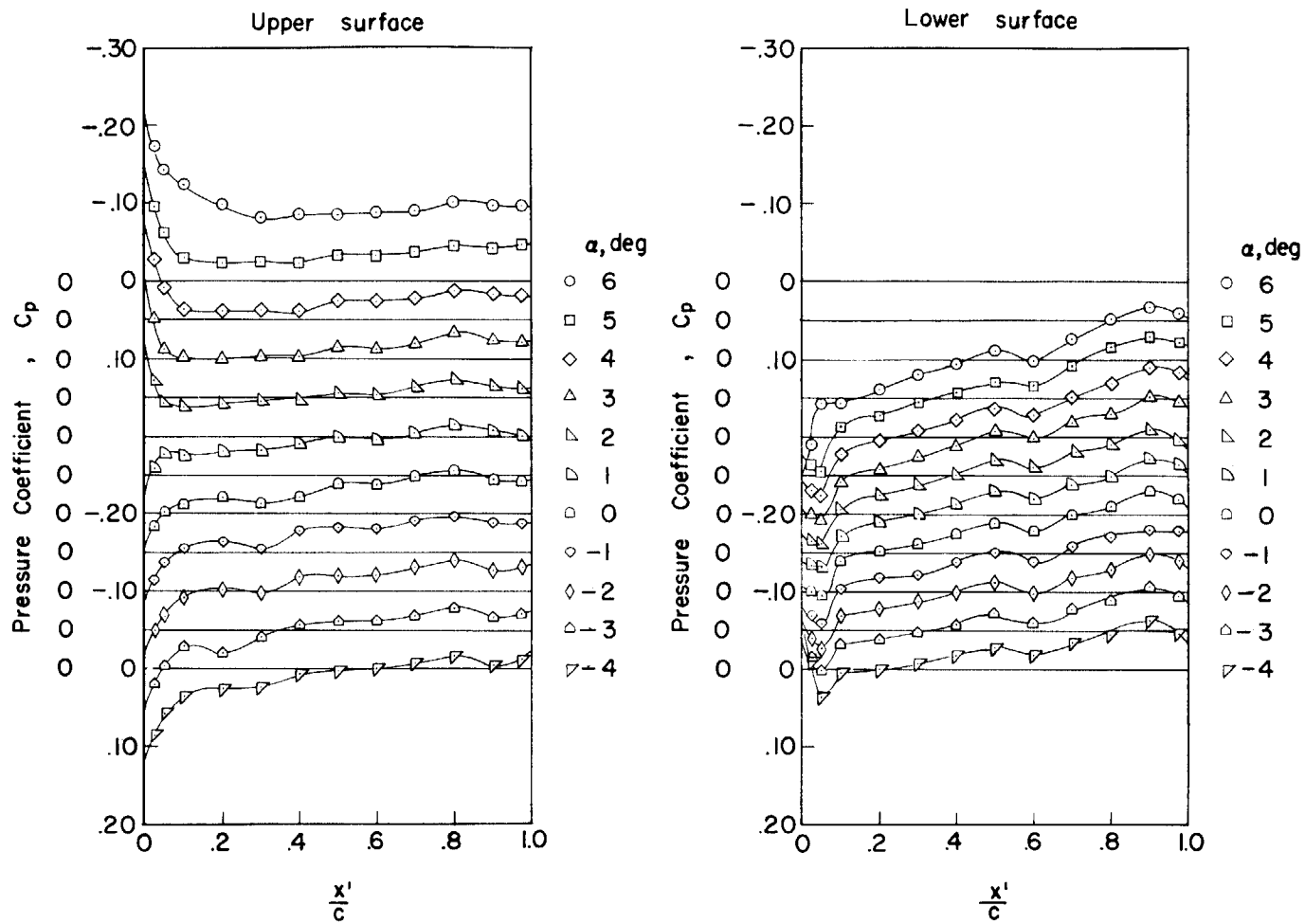
(d) $\frac{y}{b/2} = 0.7.$

Figure 3.- Continued.



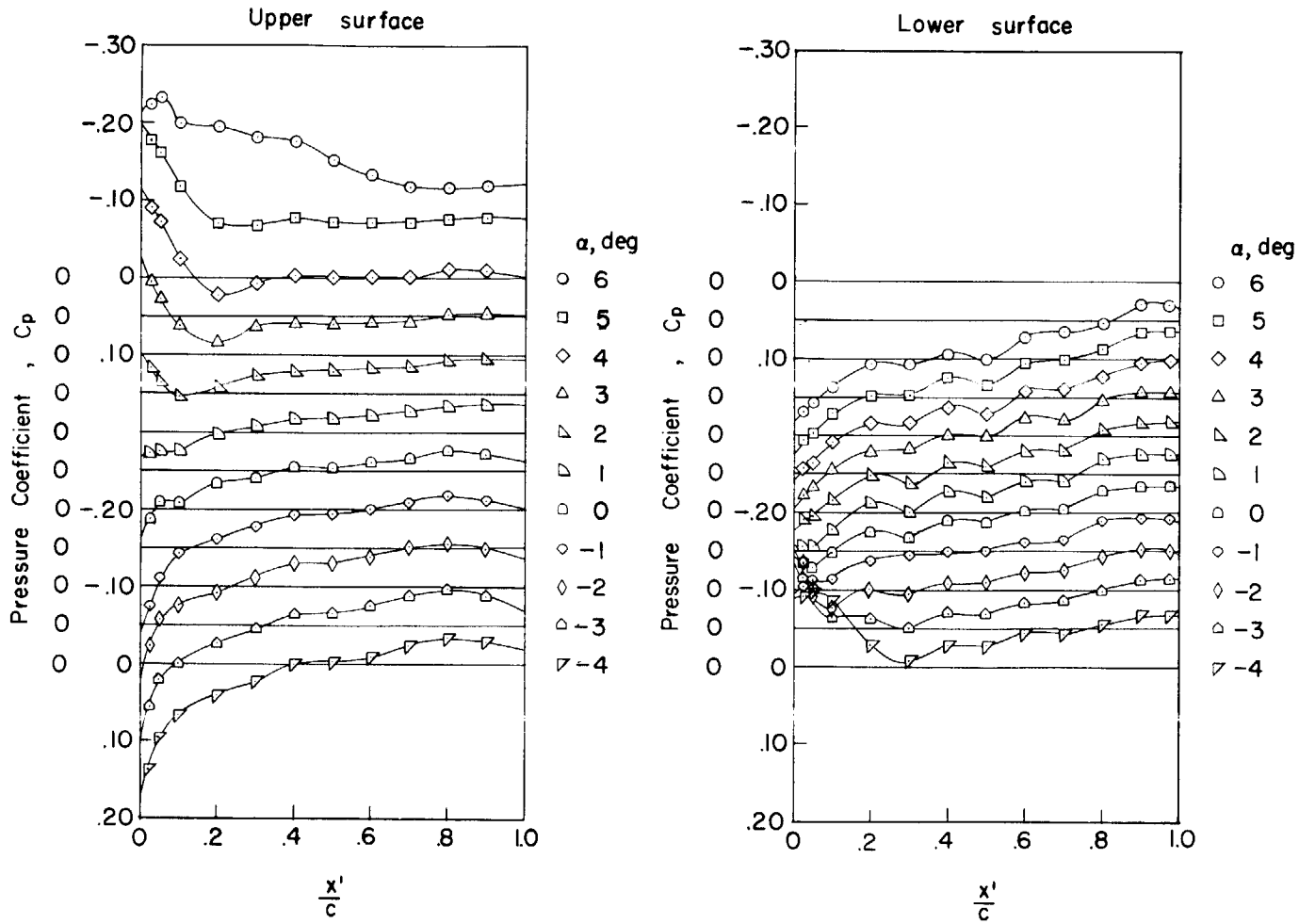
(e) $\frac{y}{b/2} = 0.9$.

Figure 3.- Concluded.



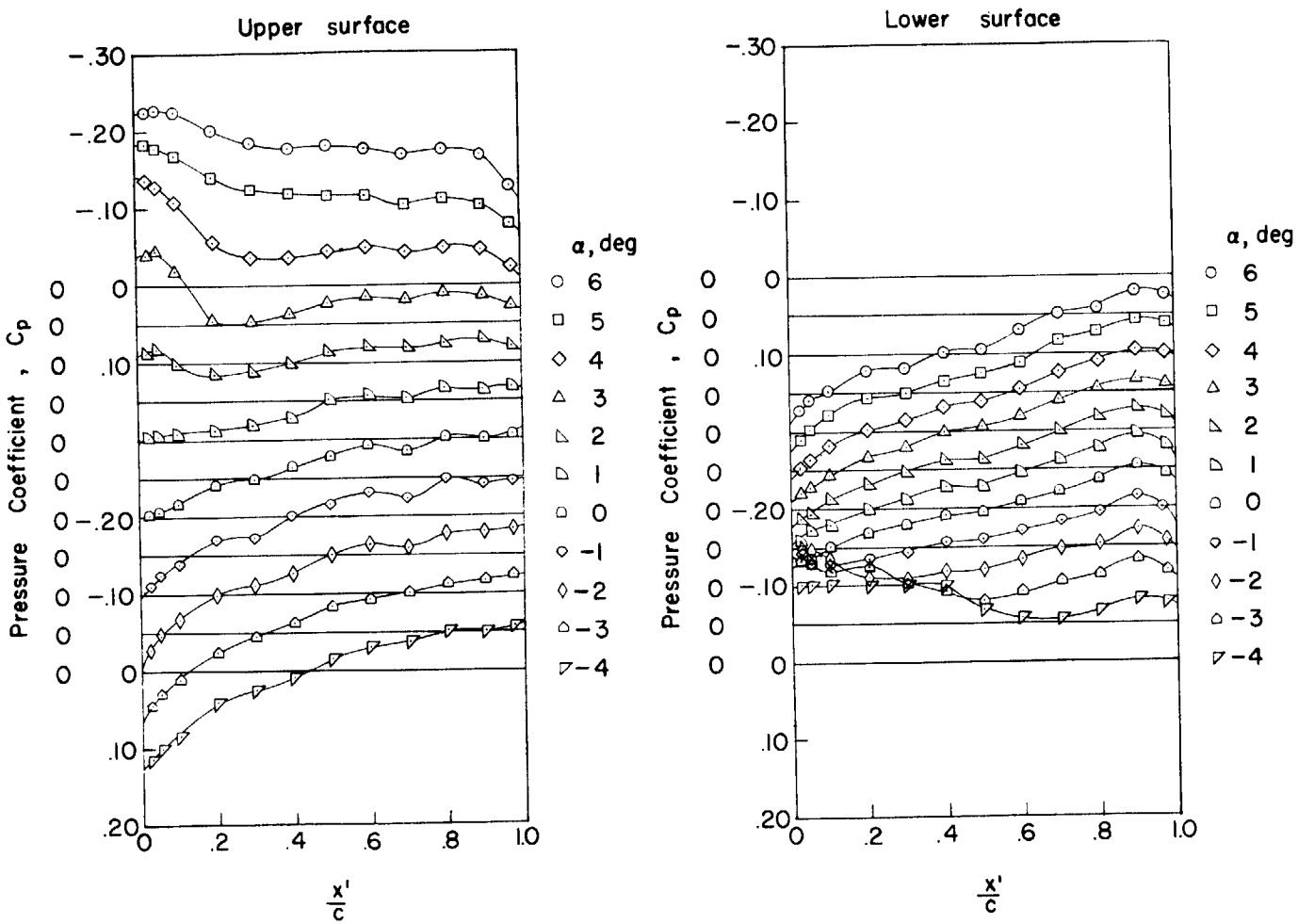
(a) $\frac{y}{b/2} = 0.1.$

Figure 4.- Pressure distributions measured on the surface of wing 2.



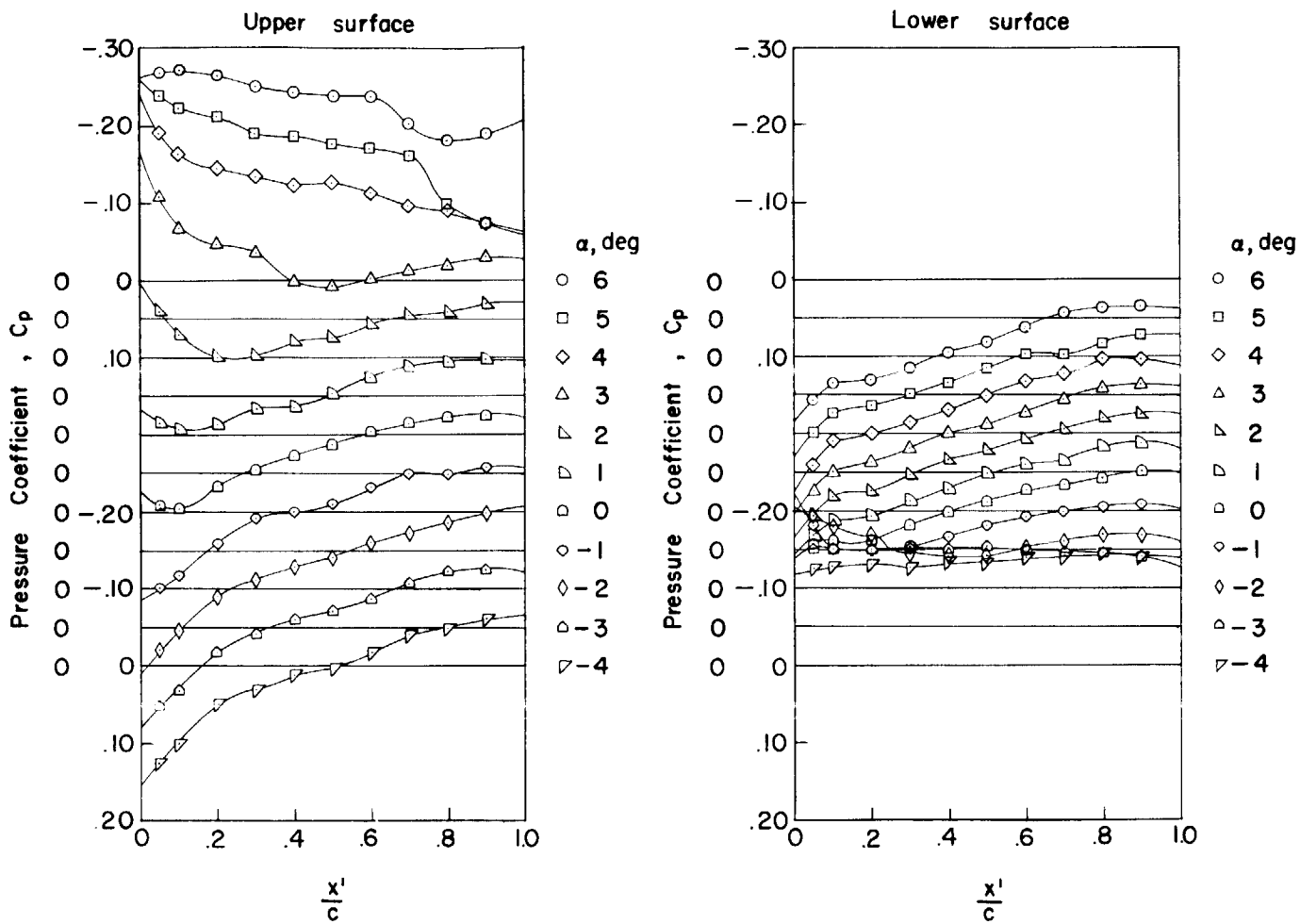
(b) $\frac{y}{b/2} = 0.3$.

Figure 4.- Continued.



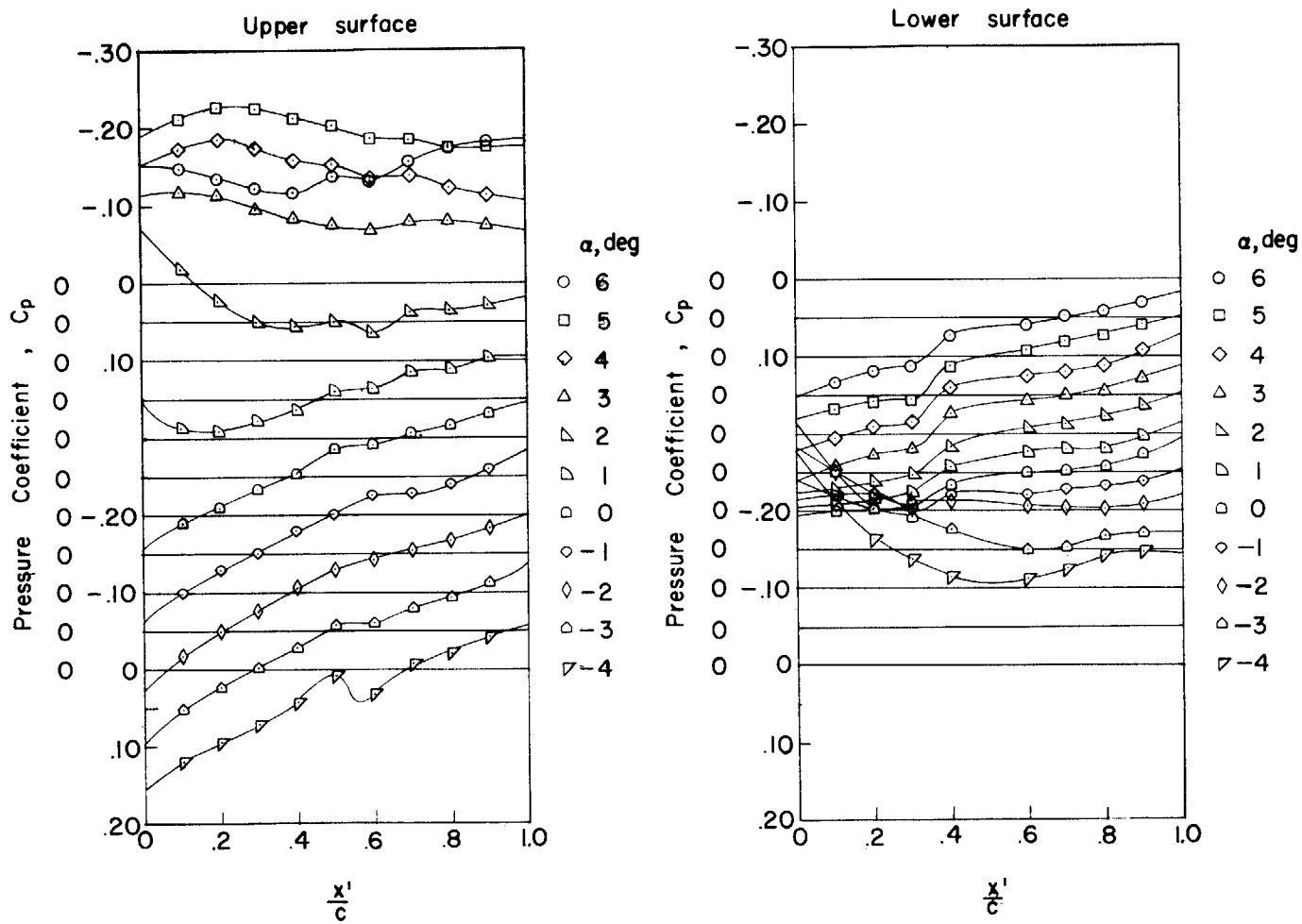
(c) $\frac{y}{b/2} = 0.5$.

Figure 4.- Continued.



(d) $\frac{y}{b/2} = 0.7.$

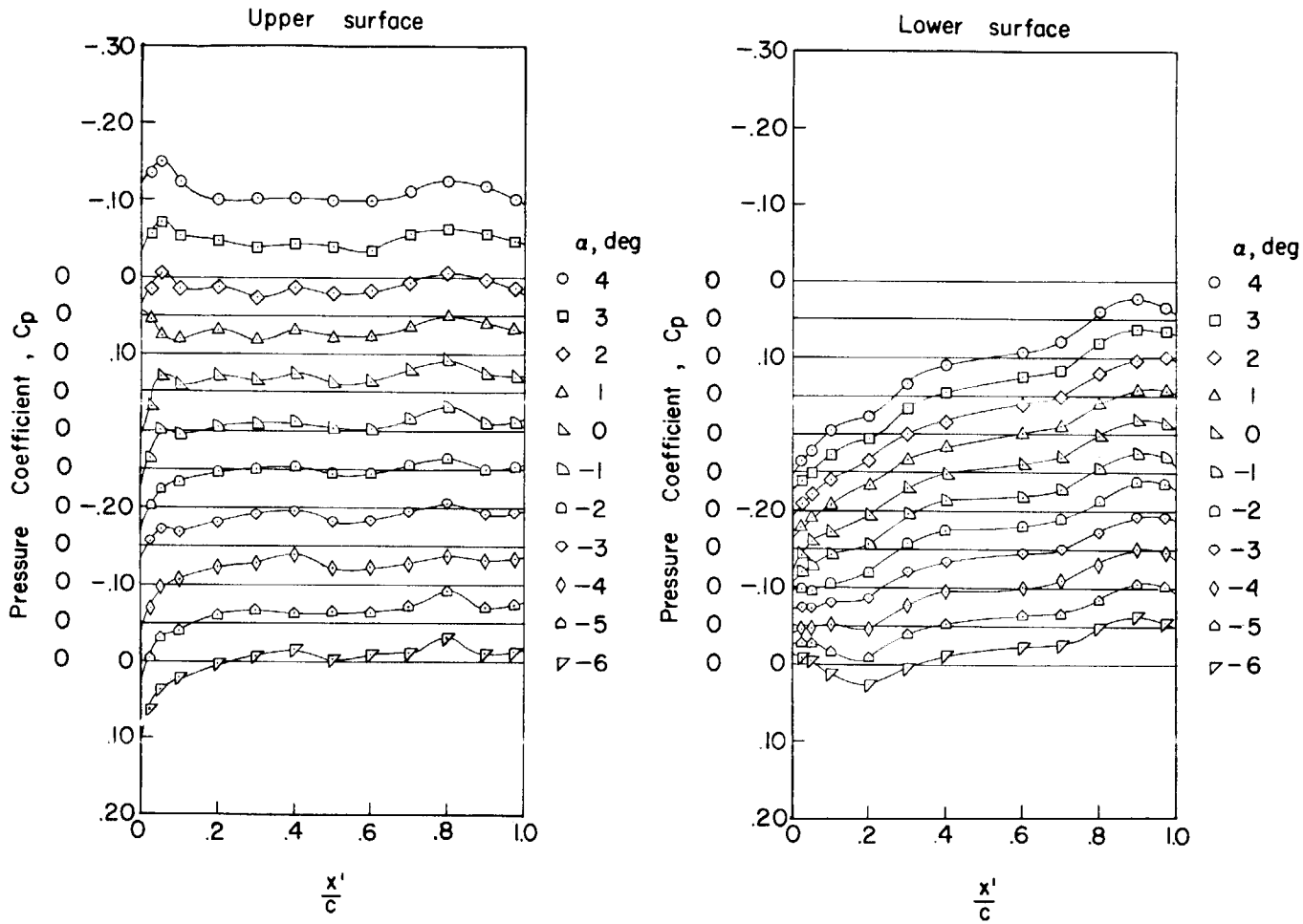
Figure 4.- Continued.



(e) $\frac{y}{b/2} = 0.9$.

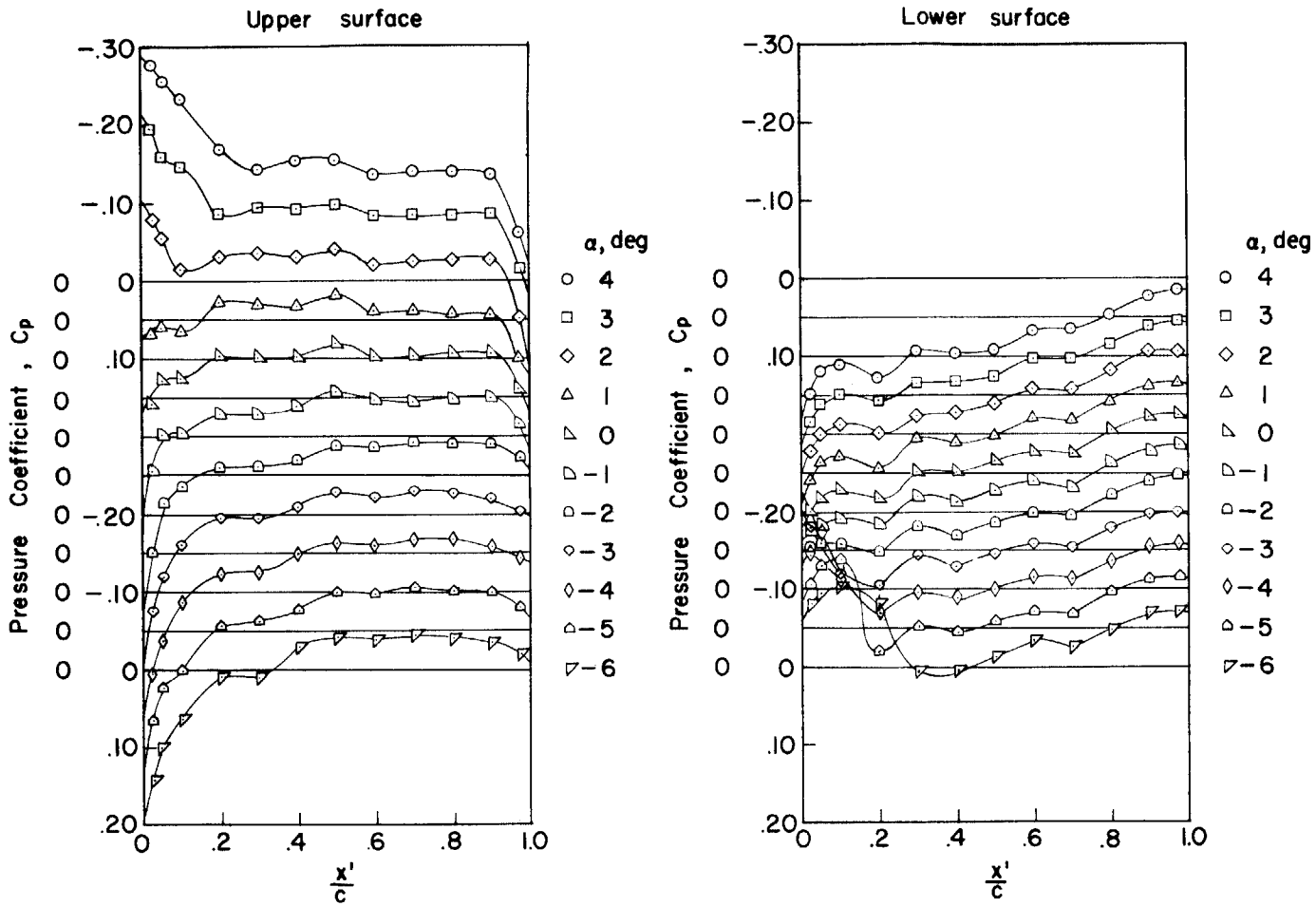
Figure 4.- Concluded.

L-1853



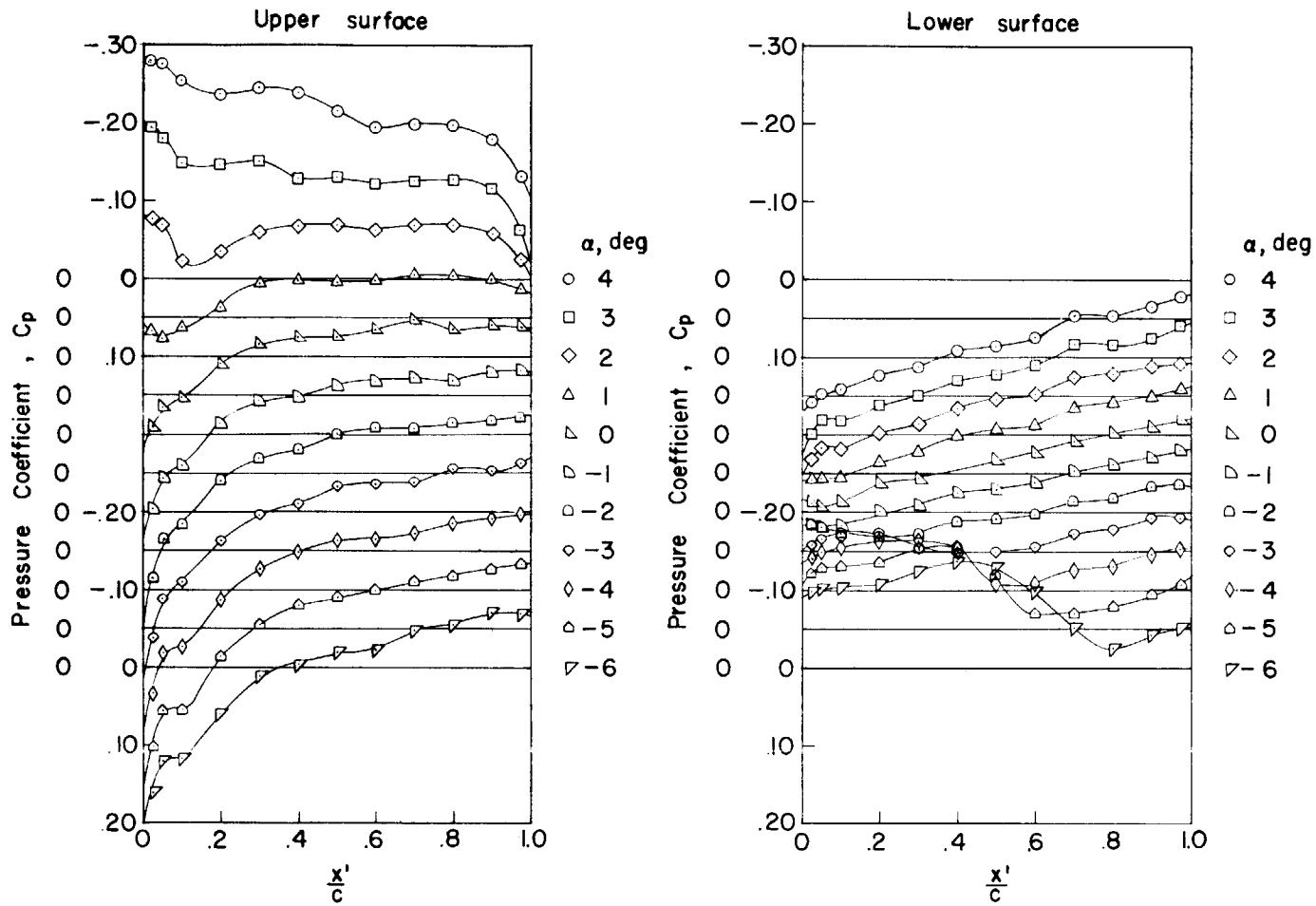
(a) $\frac{y}{b/2} = 0.1$.

Figure 5.- Pressure distributions measured on the surface of wing 3.



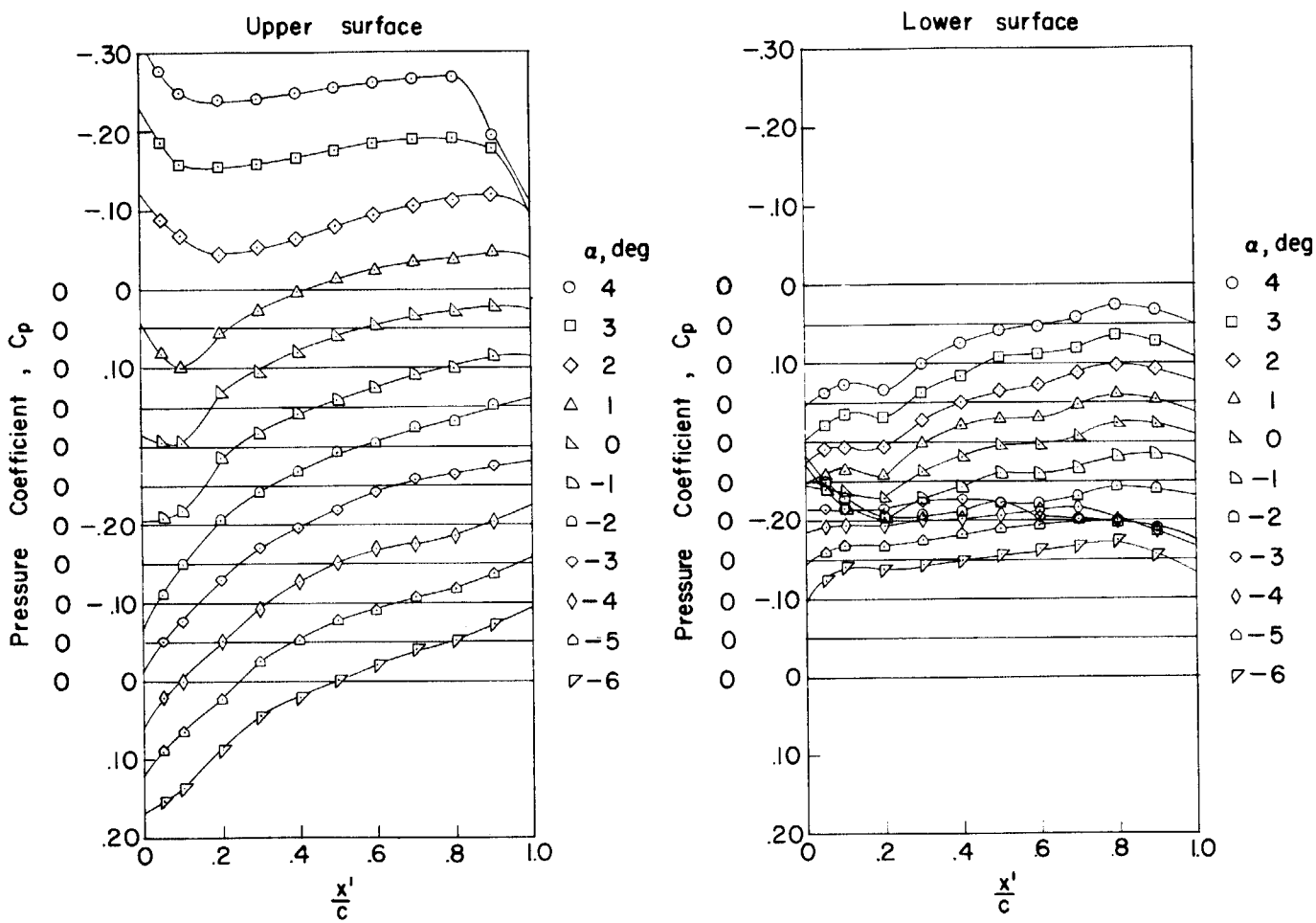
(b) $\frac{y}{b/2} = 0.3.$

Figure 5.- Continued.



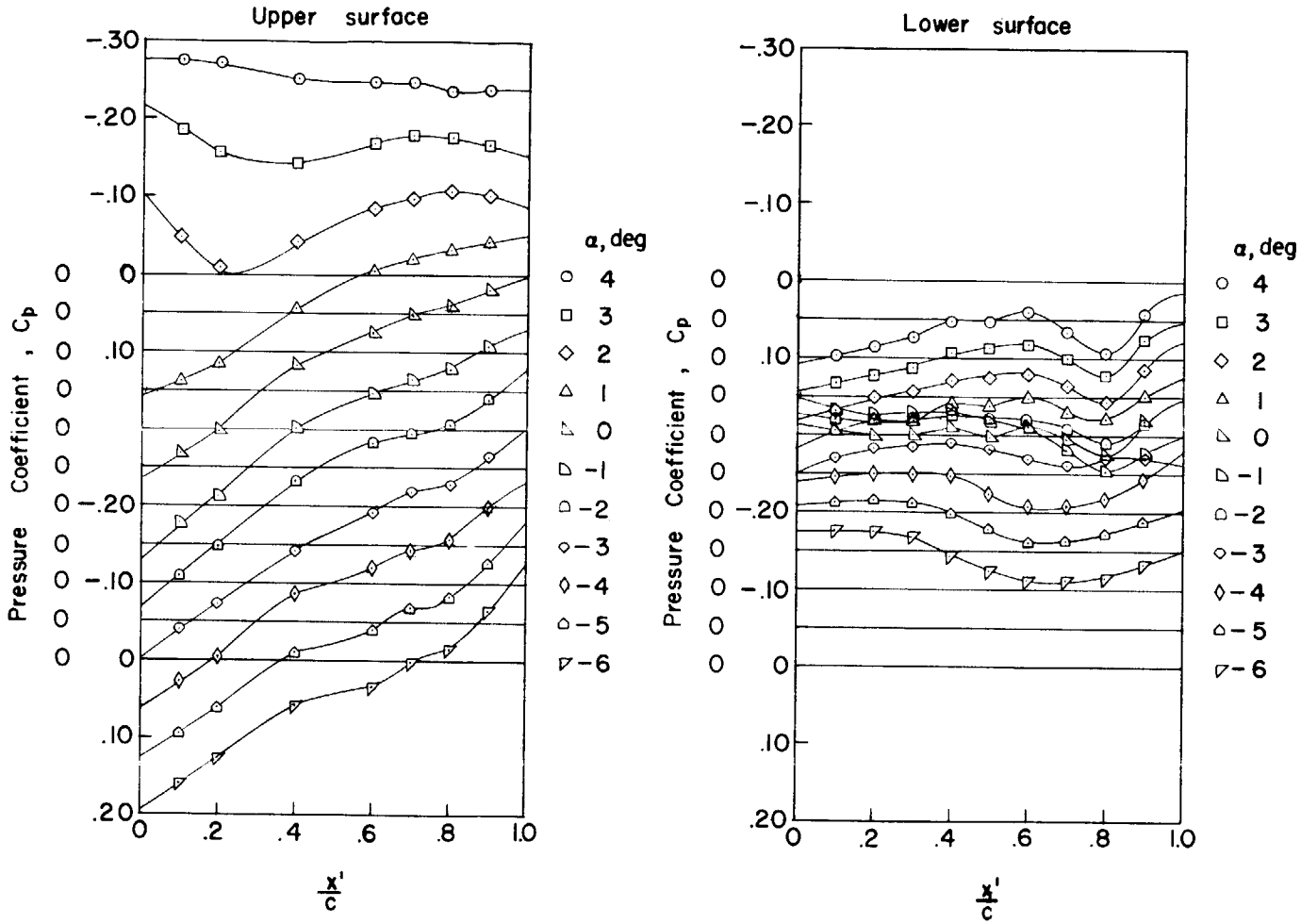
(c) $\frac{y}{b/2} = 0.5$.

Figure 5.- Continued.



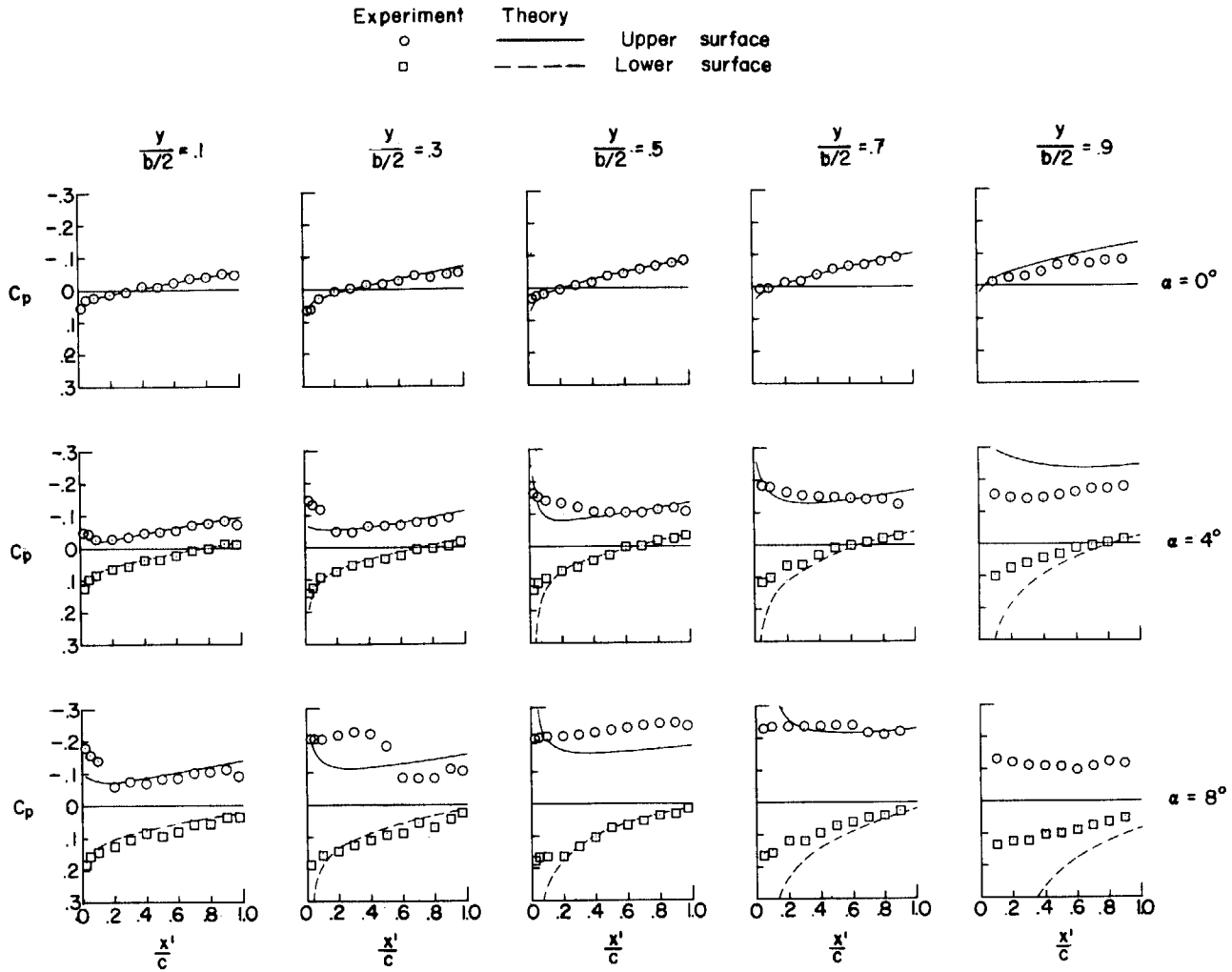
(d) $\frac{y}{b/2} = 0.7.$

Figure 5.- Continued.



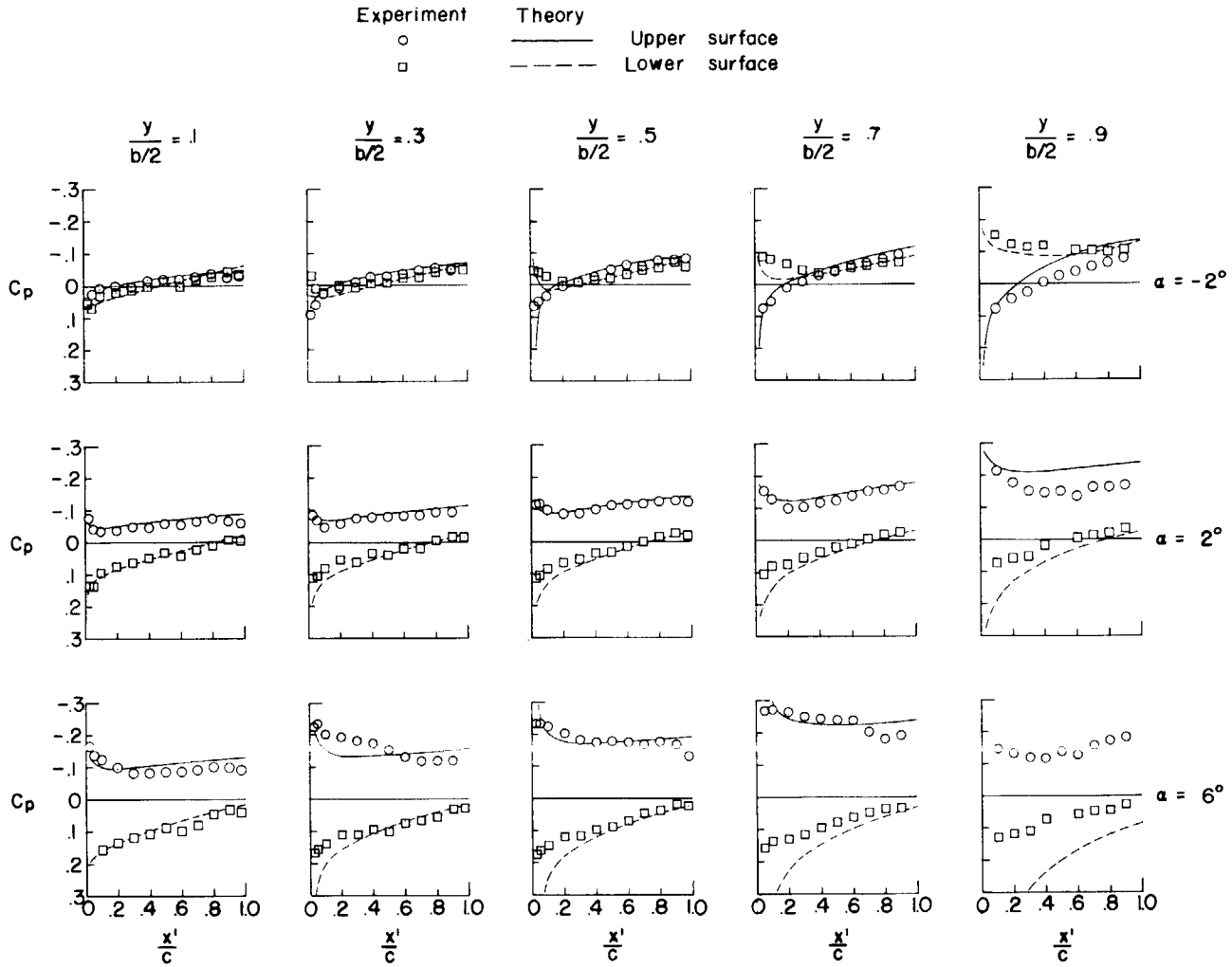
(e) $\frac{y}{b/2} = 0.9$.

Figure 5.- Concluded.



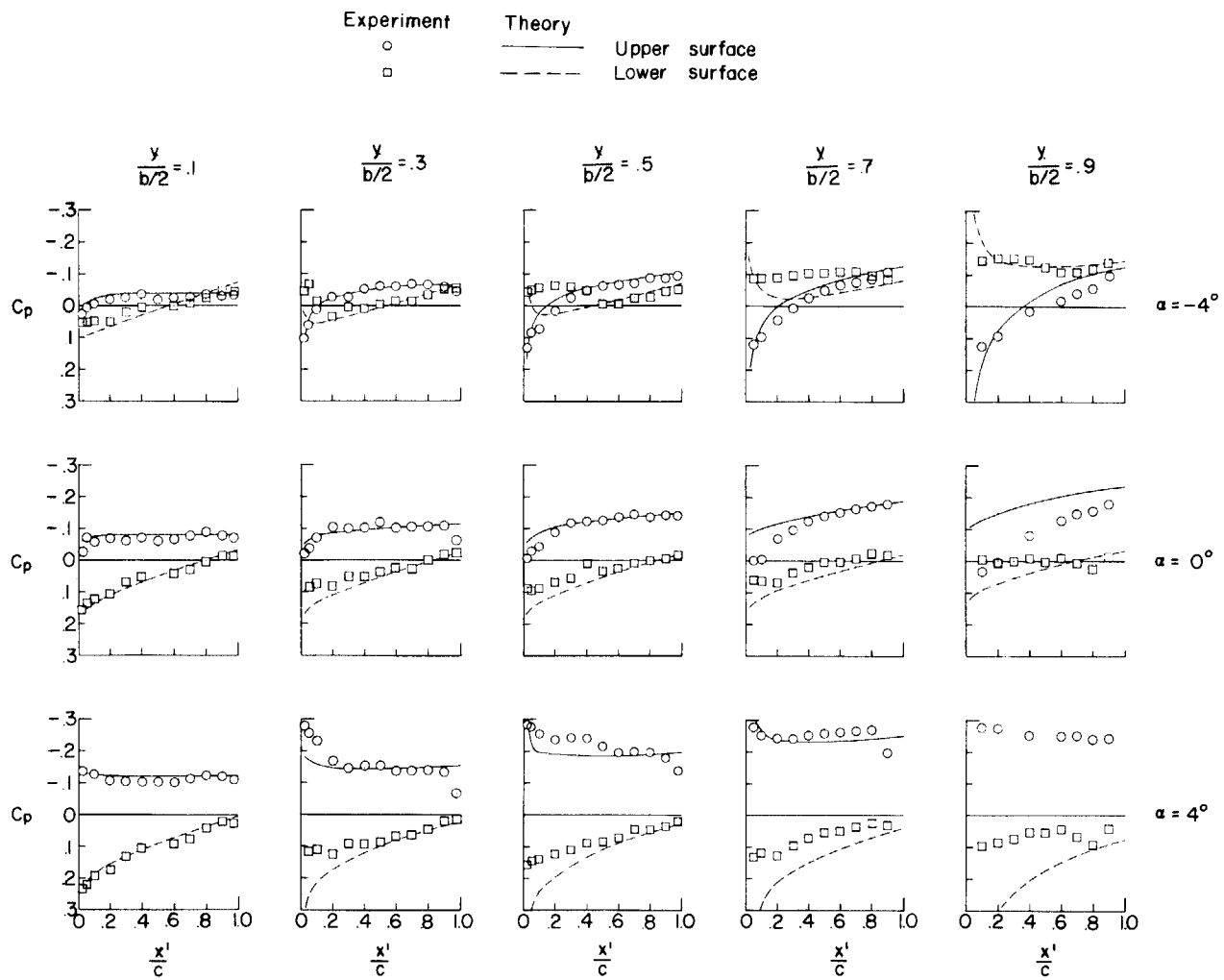
(a) Wing 1.

Figure 6.- Comparison of theoretical and experimental pressure distributions.



(b) Wing 2.

Figure 6.- Continued.



(c) Wing 3.

Figure 6.- Concluded.

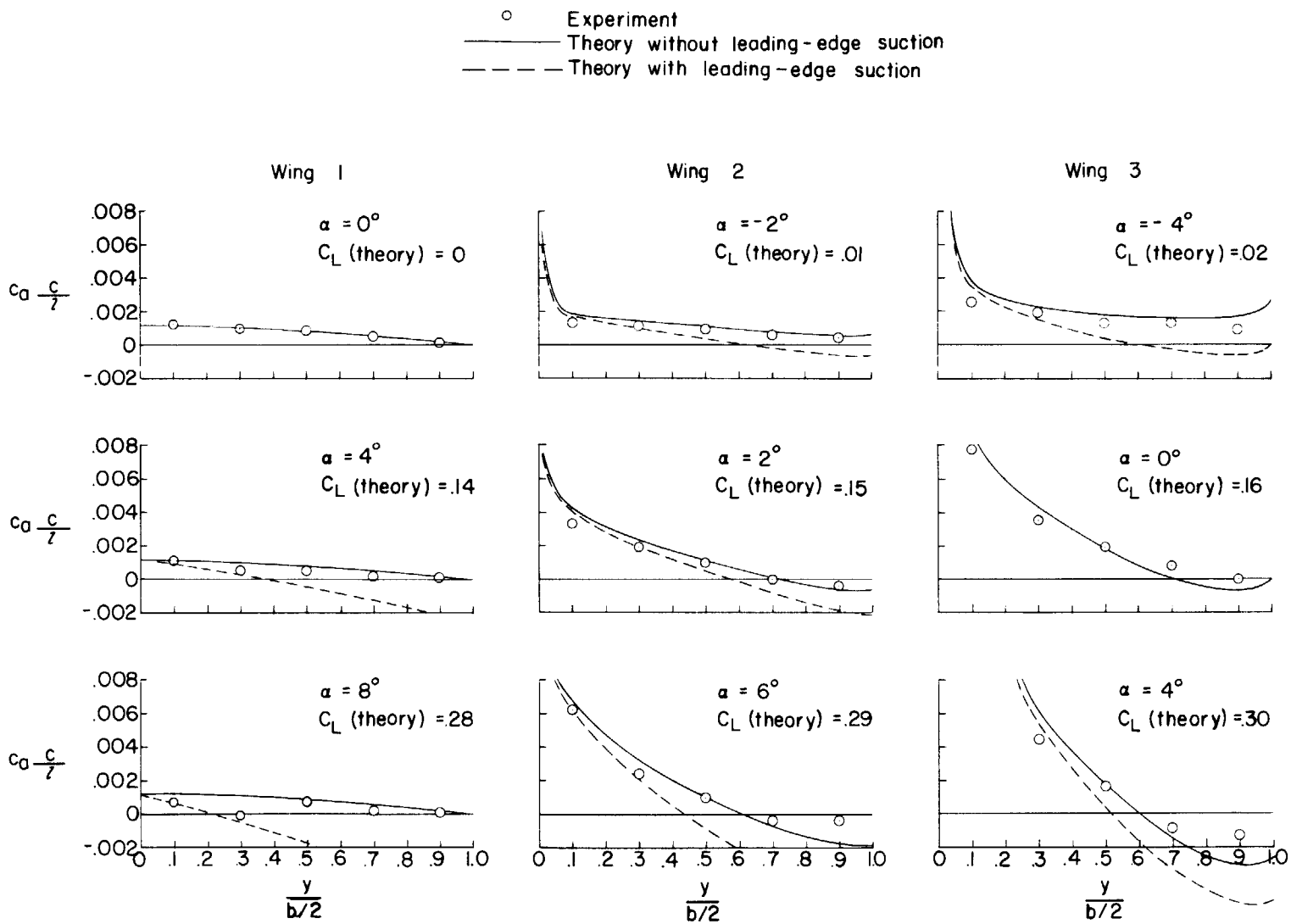


Figure 7.- Comparison of theoretical and experimental spanwise axial-force distributions.

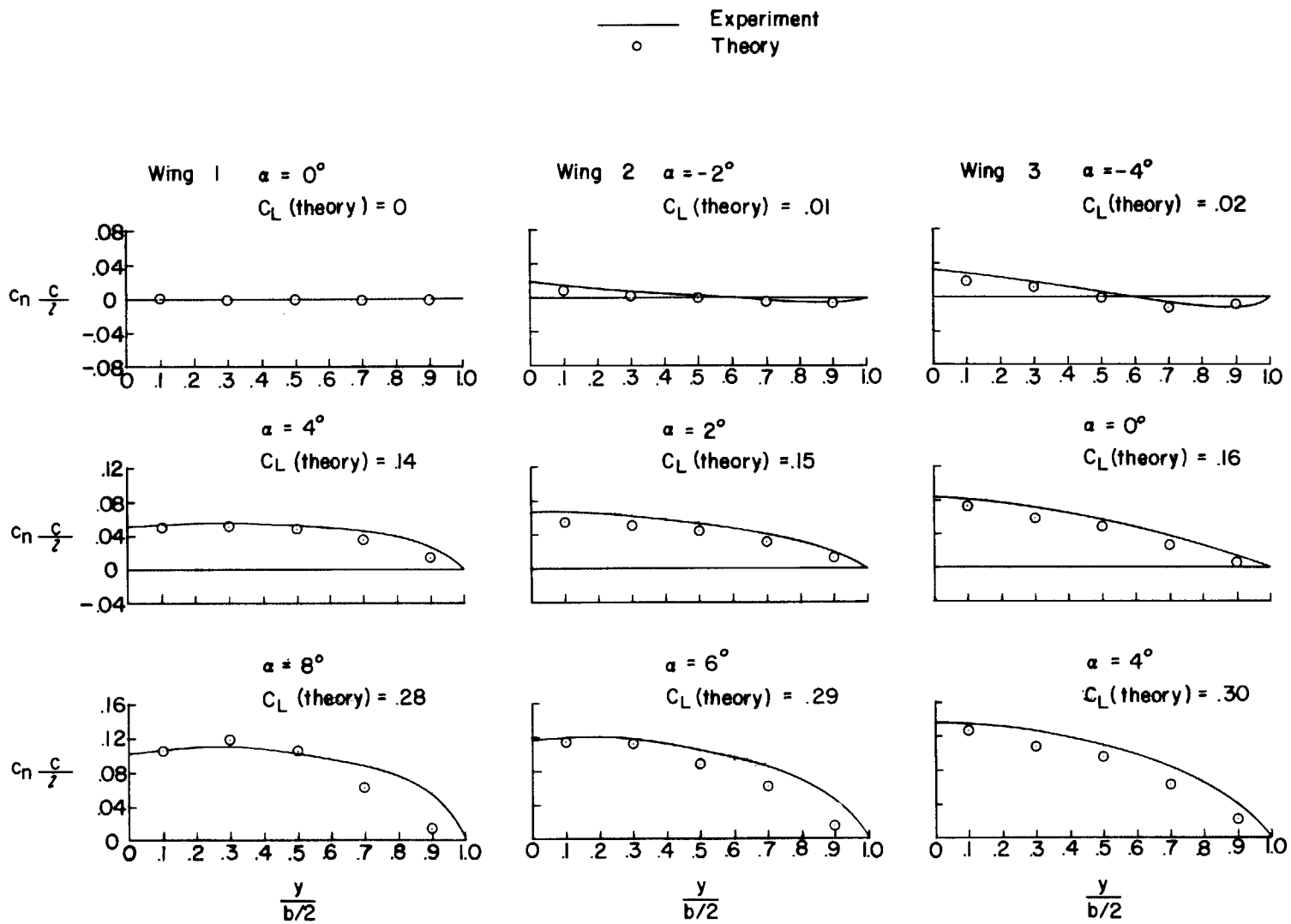


Figure 8.- Comparison of theoretical and experimental spanwise normal-force distributions.

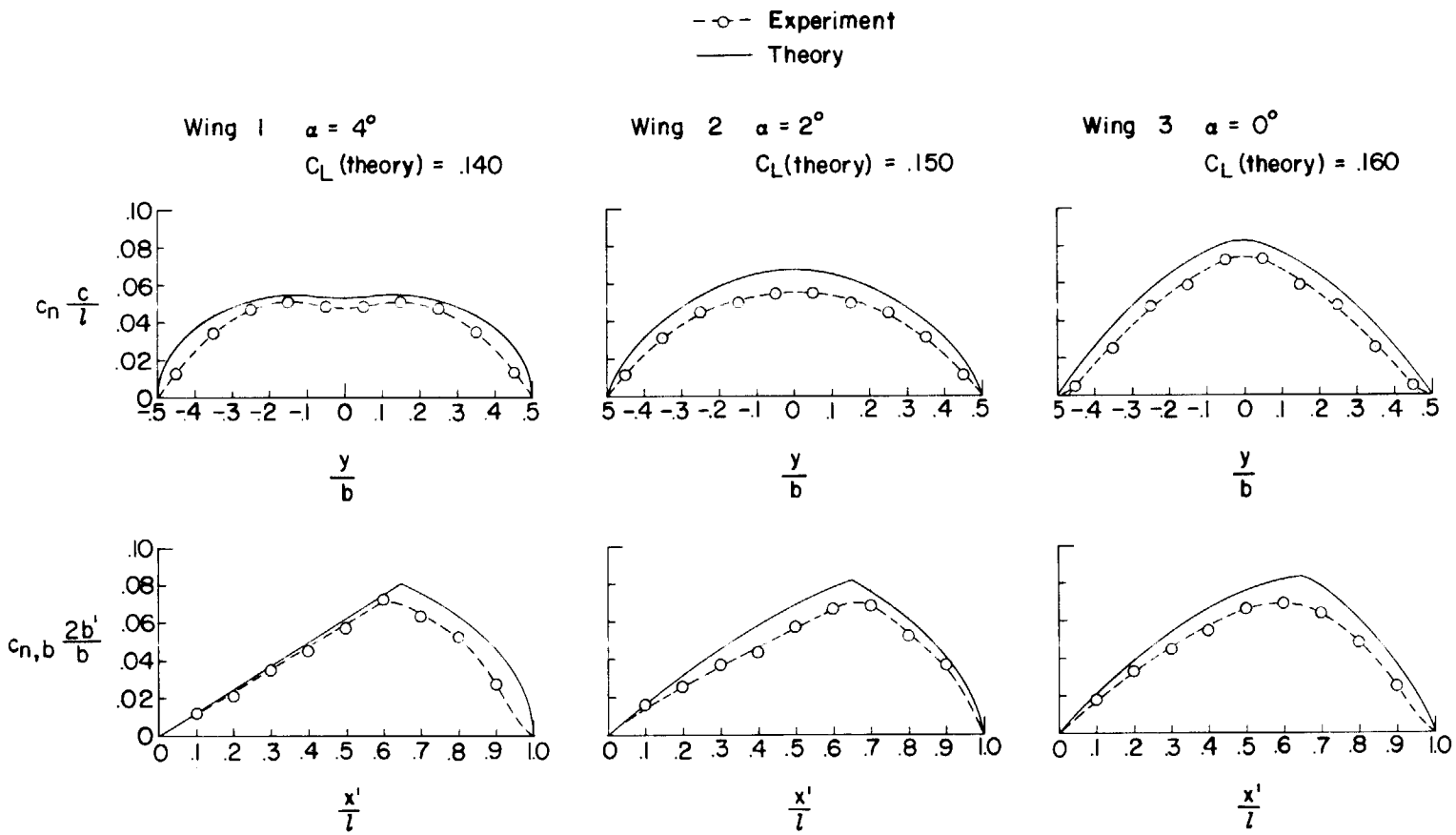


Figure 9.- Comparison of theoretical and experimental normal-force distributions on the three wings for lift coefficients near the maximum lift-drag ratio condition.

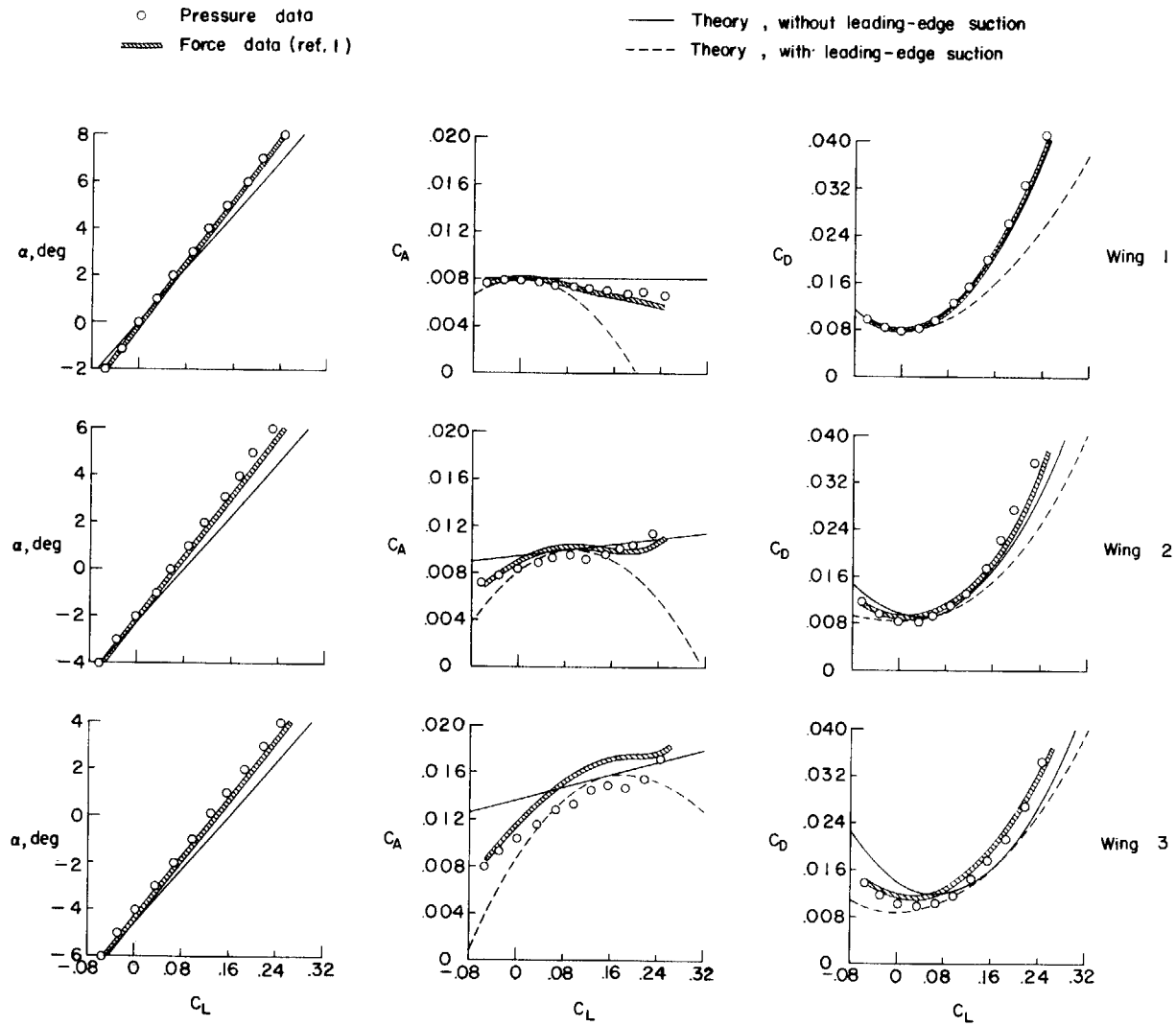


Figure 10.- Comparison of theoretical and experimental aerodynamic characteristics.

L-1853