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RESEARCH MEMORANDUM

LOW-SPEED CHORDWISE PRESSURE DISTRIBUTIONS NEAR THE MIDSPAN STATION OF THE SLOTTED FLAP AND AILERON OF A 1/4-SCALE MODEL OF THE BELL X-1 AIRPLANE WITH A 4-PERCENT-THICK, ASPECT-RATIO-4, UNSWEPT WING

By William C. Moseley, Jr., and Robert T. Taylor

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

WASHINGTON
March 9, 1954

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RESEARCH MEMORANDUM

LOW-SPEED CHORDWISE PRESSURE DISTRIBUTIONS NEAR THE MIDSPAN STATION OF THE SLOTTED FLAP AND AILERON OF A 1/4-SCALE MODEL OF THE BELL X-1 AIRPLANE WITH A 4-PERCENT-THICK, ASPECT-RATIO-4, UNSWEPT WING

By William C. Moseley, Jr., and Robert T. Taylor

SUMMARY

An investigation was made in the Langley 300 MPH 7- by 10-foot tunnel to determine the low-speed chordwise pressure distributions near the midspan station of the slotted flap and aileron of a 1/4-scale model of the Bell X-1 airplane with a wing of aspect ratio 4, taper ratio 0.5, 0° sweepback of the 40-percent-chord line, and a modified NACA 64A004 airfoil section.

The changes in pressure distributions due to deflecting the aileron and slotted flap, while generally similar to changes in pressure distributions obtained on thick wing sections in previous investigations, differed to an extent which will be significant in certain design problems. For the thin section of the present investigation, the pressures near the leading edge of the flap and aileron were larger than those of the thick airfoils.

Most of the change in shape of the pressure distribution caused by aileron deflection occurred directly in front of the aileron. The flap and aileron section hinge-moment coefficients agreed with previously published force data on the same wing. Deflecting the flap had little effect on the aileron hinge-moment parameters.

INTRODUCTION

As a part of a flight-research program designed to study aerodynamic characteristics in the transonic and low supersonic speed ranges, the National Advisory Committee for Aeronautics is modifying the wing of the Bell X-1 research airplane. The proposed wing is to have an aspect ratio

of 4, a taper ratio of 0.5, 0° sweepback of the 0.40-chord line and an NACA 64A004 airfoil section with a modified trailing edge.

A wind-tunnel investigation to determine the low-speed static stability and control characteristics of a 1/4-scale model of the Bell X-1 airplane modified to include the 4-percent-thick wing has been made by the NACA (ref. 1). In order to obtain some information on the aerodynamic loads at low speed, chordwise pressure measurements have been made on the wing at two spanwise locations. The spanwise stations were selected so that typical pressure data could be obtained over the slotted flap and aileron, and correspond to proposed orifice locations on the full-scale wing. This paper presents the results of the pressure-distribution measurements made in the Langley 300 MPH 7- by 10-foot tunnel.

SYMBOLS

S_P pressure coefficient, $\frac{H_o - p}{q} = 1 - P$

H_o total pressure in tunnel, lb/sq ft

P pressure coefficient, $\frac{p - p_o}{q}$

p local static pressure, lb/sq ft

p_o free-stream static pressure, lb/sq ft

q free-stream dynamic pressure, lb/sq ft

α angle of attack of fuselage center line, deg

$\bar{c}$ mean aerodynamic chord, $\frac{2}{S}\int_0^{b/2} c^2 dy$, ft

S wing area, sq ft

b wing span, ft

c local chord, ft

c_a chord of the aileron equal to 0.25c, ft

y spanwise distance from plane of symmetry, ft

c_f chord of the flap equal to 0.27c, ft

x chordwise coordinate, from wing leading edge parallel to plane of symmetry, ft

δ_f deflection of flap, deg

δ_a deflection of aileron, positive when down, deg

c_n wing section normal-force coefficient,
at station 0.512b/2:

$$\frac{1}{c}\int_0^{0.73c}\left(S_{P_u} - S_{P_l}\right)dx + \frac{\cos\delta_f}{c}\int_{0.73c}^{c}\left(S_{P_u} - S_{P_l}\right)dx$$

at station 0.805b/2:

$$\frac{1}{c}\int_0^{0.70c}\left(S_{P_u} - S_{P_l}\right)dx + \frac{\cos\delta_a}{c}\int_{0.70c}^{c}\left(S_{P_u} - S_{P_l}\right)dx$$

c_m wing section pitching-moment coefficient,
at station 0.512b/2:

$$\frac{1}{c^2}\int_0^{0.73c}\left(S_{P_u} - S_{P_l}\right)(0.25c - x)dx +$$

$$\frac{1}{c^2}\int_{0.73c}^{c}\left(S_{P_u} - S_{P_l}\right)\left(-0.498c\cos\delta_f + 0.748c - x\right)dx$$

at station 0.805b/2:

$$\frac{1}{c^2}\int_0^{0.70c}\left(S_{P_u} - S_{P_l}\right)(0.25c - x)dx +$$

$$\frac{1}{c^2}\int_{0.70c}^{c}\left(S_{P_u} - S_{P_l}\right)\left(-0.50c\cos\delta_a + 0.75c - x\right)dx$$

c_{n_f} flap section normal-force coefficient,

at station 0.512b/2: $\frac{1}{0.27c}\int_{0.73c}^{c}\left(S_{P_u} - S_{P_l}\right)dx$

c_{h_f} flap section hinge-moment coefficient,

at station 0.512b/2: $\frac{1}{(0.27c)^2}\int_{0.73c}^{c}\left(S_{P_u} - S_{P_l}\right)(0.748c - x)dx$

c_{n_a} aileron section normal-force coefficient,

at station 0.805b/2: $\frac{1}{0.30c}\int_{0.70c}^{c}\left(S_{P_u} - S_{P_l}\right)dx$

c_{h_a} aileron section hinge-moment coefficient,

at station 0.805b/2: $\frac{1}{(0.30c)^2}\int_{0.70c}^{c}\left(S_{P_u} - S_{P_l}\right)(0.75c - x)dx$

Subscripts:

u upper surface

l lower surface

Stability derivations are written in the following form where the subscripts outside the parentheses represent the factors held constant during the measurement of the parameters:

$c_{n_\alpha} = (\partial c_n/\partial\alpha)_\delta$

$c_{h_\alpha} = (\partial c_h/\partial\alpha)_\delta$

$c_{h_\delta} = (\partial c_h/\partial\delta)_\alpha$

MODEL, APPARATUS, AND TESTS

The model used during this investigation was a 1/4-scale model of the Bell X-1 airplane and was mounted on a single strut support in the Langley 300 MPH 7- by 10-foot tunnel (see fig. 1). The geometric characteristics and general dimensions of the model are given in figure 2. The wing

incorporated into the model for the present tests had an aspect ratio of 4, a taper ratio of 0.5, 0° sweepback of the 0.40-chord line, and a modified NACA 64A004 airfoil section. The airfoil section was modified rearward of the 0.70-chord line so that the wing would have a constant trailing-edge thickness of 1/16 inch for simplicity of construction. The wing was made of a steel core covered with wood and wrapped with glass cloth impregnated with paraplex. The flaps and ailerons were machined of aluminum.

The wing had a 0.25c aileron that extended from the 0.98b/2 station inboard to the 0.68b/2 station. The aileron had a round-nose overhang balance of $0.20c_a$ and a gap of 1/64 inch between the wing and the aileron nose. A 0.27c single-slotted flap extended inboard of the aileron to the juncture of the wing and fuselage. The flap was hinged externally at the 0.748c station, 0.022c below the chord plane. Details of the slotted flap and aileron are given in figure 3.

The pressure orifices were located on the upper and lower surfaces of the left semispan at the 0.512b/2 station and the 0.805b/2 station. These stations were selected to correspond to actual orifice locations proposed for the airplane and are believed to render typical chordwise pressure distributions across the slotted flap and aileron. Ordinates for the pressure orifices are given in figure 4 and chordwise locations are listed on all tables of data (tables I to XXVIII). An asterisk before stations in table indicates a lower-surface orifice.

All tests were made in the Langley 300 MPH 7- by 10-foot tunnel at a dynamic pressure of 25.86 pounds per square foot, which corresponds to a Mach number of about 0.13 and a Reynolds number of about 1.2×10^6 based on a mean aerodynamic chord of 1.48 feet. Flap deflections of 0°, 20°, 30°, 35°, 40°, and 45° and aileron deflections of 0°, ±6°, and ±12° were investigated through an angle-of-attack range from -6° to 24°.

PRESENTATION OF THE DATA

The results of the investigation are presented as tables of chordwise pressure coefficients for the two spanwise stations (tables I to XXVIII). Also presented are typical plots of chordwise pressure distributions and summary plots of the integrated coefficients (figs. 5 to 10). The section coefficients were obtained by a "step by step" integration process using an automatic digital computer and applying factors which weight the data according to orifice location. Comparisons were made between section coefficients obtained by the digital process and conventional mechanical integration and were found to agree within ±2 percent. The integrated coefficients are plotted against corrected angle of attack; all other plots and tabulated data are presented at nominal values of angle of attack.

RESULTS AND DISCUSSION

Section Pressure Distribution

Typical section pressure distribution for the slotted-flap station (0.512b/2) and for the aileron station (0.805b/2) are presented in figures 5 to 7. Since pressure data are not available on an NACA 64A004 airfoil section at low speed comparisons are made in this paper using pressure-coefficient data for a 6-percent-thick plain airfoil (ref. 2). A direct comparison was not available for a thin symmetrical airfoil equipped with a slotted flap or flap-type aileron. However, in references 3 and 4, pressure distributions are given for an NACA 23012 airfoil with a 0.2566c slotted flap and a 0.20c plain flap and an NACA 23021 airfoil with a 0.2566c slotted flap. Although the data of references 2 to 4 were taken from two-dimensional models and are therefore not directly comparable to the three-dimensional data of the present tests, it is felt that some generalities may be determined from a comparison.

A comparison of the pressure-coefficient diagrams at $\delta_f = 0^\circ$ and $\delta_a = 0^\circ$ (figs. 5(a) and 6(a)) with the pressure-coefficient data of reference 2 indicates a general agreement over the forward portion of the wing. As the slotted flap was deflected, a comparison of the pressure-coefficient data with the pressure-coefficient data of references 3 and 4 shows higher pressures near the leading edge of the slotted flap on the modified NACA 64A004 airfoil. Similar differences were noted when comparisons were made of the aileron pressure-coefficient data and the pressure-coefficient data for the plain flap of reference 3. In addition, the pressure-coefficient data of reference 3 showed a double-peak pressure region near the nose of the slotted flap which was not present on the slotted flap of the present investigation.

It should be pointed out that a 2° incidence existed between the wing and fuselage for the X-1 model (see fig. 2) and should be considered in the foregoing comparisons. In addition, the number and locations of the orifices over the slotted flap and aileron of the present wing were different from those used on the models of references 3 and 4.

General agreement between the present data and the data of reference 3 was obtained in that a rearward shift of loading occurred when a given lift coefficient was obtained at low angle of attack by deflection of the slotted flap. For example, comparison of the pressure diagrams for the slotted flap at $\delta_f = 0^\circ$, $\delta_f = 20^\circ$, and $\delta_f = 35^\circ$ at a low angle of attack indicates that increasing the flap deflection and decreasing the angle of attack to obtain constant section normal force reduced the magnitudes of the pressures near the leading edge of the wing and increased the pressures at the leading edge of the flap.

Comparison of the pressure diagrams for the slotted flap at $\delta_f = 0^\circ$, $\delta_f = 20^\circ$, and $\delta_f = 35^\circ$ (fig. 5) for the same angle of attack indicates that deflecting the flap increased the pressure coefficients over the entire upper surface of the airfoil at angles below about $\alpha = 3^\circ$ which is about the beginning of the stall with the flaps deflected. The pressure coefficients over the lower surface of the wing were decreased as the flaps were deflected.

Comparison of the pressure diagrams for the aileron at $\delta_a = 0^\circ$ and $\delta_a = +12^\circ$ (fig. 6) indicates that deflecting the aileron has little effect on the shape of the pressure distribution of the main portion of the wing, except immediately forward of the aileron. Increased pressure coefficients were obtained over the leading edge of the aileron as the aileron was deflected. These general trends were noted for both the $\delta_f = 0^\circ$ and $\delta_f = 35^\circ$ configurations (figs. 6 and 7).

Section Normal-Force and Pitching-Moment Coefficients

Slotted flap.- The effect of deflecting the slotted flap on the wing section pitching-moment coefficients and wing section normal-force coefficients is shown in figure 8. Also presented are the flap section hinge-moment coefficients and flap section normal-force coefficients. The data for the wing section pitching-moment coefficient for the flaps-retracted configuration show slight instability about the 0.25c point of the section with a stable break preceding the stall by about $0.20c_n$. As the flap was deflected, this stable break tended to decrease until at flap deflections of 35° and 40° the section was unstable throughout the normal-force-coefficient range. The data of reference 1 for wing-fuselage pitching-moment coefficient showed similar stability trends for $\delta_f = 0^\circ$ and $\delta_f = 35^\circ$. The section normal-force-curve slope c_{n_α} was about 0.080, which compared very favorably with the two-dimensional data of reference 5 which gives a value of $c_{n_\alpha} = 0.088$ for a Mach number of 0.3 for an NACA 64A004 airfoil section. The optimum flap deflection was 35° which substantiates the force data of reference 1. The flap had a large loss in effectiveness at $\delta_f = 45^\circ$ which was also noted in reference 1.

The flap section hinge-moment coefficient was generally linear with flap deflection up to optimum flap deflection ($\delta_f = 35^\circ$). The section hinge-moment parameters $c_{h_{f_\alpha}}$ and $c_{h_{f_\delta}}$ were -0.0042 and -0.0102, respectively. The data for flap section normal-force coefficient showed that the wing with deflected flap was generally stalled at angles of attack greater than $\alpha = 4^\circ$ to 6°.

Aileron.- The effect of deflecting the aileron on the section pitching-moment coefficients and section normal-force coefficients is presented in figures 9 and 10 for the flap deflected 0° and 35°. Also shown are the aileron section hinge-moment coefficients and aileron section normal-force coefficients.

The data of figures 9 and 10 for section pitching-moment coefficient show slight instability about the 0.25c point of the section with a stable break as the stall was approached for all aileron deflections. As noted previously, the data of reference 1 for wing-fuselage pitching moment show a similar stable break as the stall was approached.

The section normal-force coefficient at the aileron station generally varied nonlinearly with angle of attack, but section normal-force-curve slopes c_{n_α} of 0.070 for the flap-retracted configuration and of 0.080 for the flap deflected 35° were obtained at 0° angle of attack. Deflecting the flap increased the section normal-force coefficient at the aileron station.

The curves for aileron section hinge-moment coefficient were reasonably linear throughout the angle-of-attack and aileron-deflection ranges investigated for both flap conditions. The section hinge-moment parameters were generally the same as the values determined for the complete aileron from the strain-gage data of reference 1. The values of the hinge-moment parameters $c_{h_{a_\alpha}}$ and $c_{h_{a_\delta}}$ were only slightly changed by deflecting the flap 35°. They are -0.0020 and -0.0057 for the flap retracted and -0.0028 and -0.0060, respectively, for the flap deflected 35°.

CONCLUDING REMARKS

An investigation was made in the Langley 300 MPH 7- by 10-foot tunnel to determine the low-speed chordwise pressure distribution near the mid-span station of the slotted flap and aileron of a 1/4-scale model of the Bell X-1 airplane with a 4-percent-thick aspect-ratio-4, unswept wing.

The changes in pressure distributions due to deflecting the aileron and slotted flap, while generally similar to changes in pressure distributions obtained on thick wing sections in previous investigations, differed to an extent which will be significant in certain design problems. For the thin section of the present investigation, the pressures near the leading edge of the flap and aileron were larger than those of the thick airfoils.

Most of the change in shape of the pressure distribution caused by aileron deflection occurred directly in front of the aileron. The flap

and aileron section hinge-moment coefficients agreed with previously published force data on the same wing. Deflecting the flap had little effect on the aileron hinge-moment parameters.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., December 1, 1953.

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TABLE I.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.512b/2; $\delta_f = 0^\circ$; $\delta_a = 0^\circ$

Pressure Coefficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°	α=12°	α=15°	α=18°	α=21°
.0000	1.482	.472	.127	1.275	1.614	1.565	1.771	1.823	1.768	1.659
.0125	.301	.739	1.474	2.951	2.186	1.794	1.751	1.647	1.657	1.124
.0250	.577	.888	1.443	2.585	2.170	1.771	1.735	1.634	1.635	1.598
.0500	.741	.999	1.391	2.332	2.157	1.780	1.740	1.642	1.635	1.598
.1000	.842	1.063	1.344	1.714	2.157	1.799	1.768	1.665	1.635	1.614
.2000	.976	1.097	1.311	1.442	2.107	1.810	1.782	1.668	1.665	1.624
.3000	1.045	1.147	1.266	1.380	1.897	1.844	1.803	1.700	1.691	1.646
.4000	1.089	1.147	1.266	1.350	1.624	1.841	1.823	1.713	1.712	1.664
.5000	1.113	1.158	1.244	1.315	1.428	1.821	1.850	1.742	1.726	1.704
.6000	1.110	1.137	1.211	1.259	1.301	1.766	1.839	1.742	1.747	1.709
.7000	1.110	1.124	1.178	1.211	1.238	1.693	1.847	1.765	1.766	1.730
.7250	1.143	1.142	1.206	1.219	1.219	1.693	1.864	1.789	1.803	1.744
*.0125	2.513	1.654	.957	.430	.257	.111	.145	.100	.093	.050
*.0250	2.604	1.529	.984	.560	.419	.251	.285	.221	.175	.109
*.0500	2.226	1.460	1.089	.742	.586	.429	.456	.390	.324	.236
*.1000	1.641	1.352	1.095	.872	.705	.590	.615	.527	.465	.384
*.2000	1.488	1.320	1.153	.998	.867	.774	.793	.735	.658	.572
*.3000	1.428	1.291	1.178	1.057	.951	.874	.914	.862	.810	.726
*.4000	1.376	1.277	1.194	1.103	1.007	.952	.996	.957	.919	.840
*.5000	1.329	1.232	1.189	1.108	1.039	1.005	1.082	1.038	.998	.938
*.6000	1.285	1.216	1.172	1.111	1.057	1.033	1.134	1.118	1.086	1.026
*.7000	1.269	1.216	1.183	1.141	1.092	1.128	1.241	1.220	1.208	1.147
*.7250	1.231	1.203	1.189	1.141	1.121	1.209	1.345	1.320	1.317	1.269
.7300	1.198	1.174	1.206	1.211	1.198	1.571	1.762	1.718	1.715	1.696
.7350	1.182	1.174	1.206	1.211	1.198	1.557	1.746	1.702	1.707	1.667
.7400	1.179	1.169	1.208	1.211	1.198	1.587	1.782	1.731	1.736	1.707
.7500	1.187	1.174	1.214	1.211	1.195	1.587	1.762	1.721	1.726	1.693
.8000	1.045	1.063	1.139	1.151	1.166	1.587	1.806	1.755	1.744	1.709
.8500	1.061	1.089	1.125	1.141	1.158	1.548	1.790	1.755	1.744	1.709
.9000	1.078	1.073	1.112	1.116	1.142	1.512	1.790	1.755	1.744	1.709
*.7350	1.165	1.089	1.136	1.028	.914	.663	.758	.754	.743	.713
*.7400	1.198	1.124	1.089	.982	.935	.919	1.043	1.046	1.035	.983
*.7500	1.203	1.153	1.092	1.022	.994	1.008	1.128	1.128	1.115	1.079
*.8000	1.179	1.147	1.117	1.087	1.049	1.075	1.233	1.228	1.229	1.200
*.8500	1.160	1.121	1.112	1.065	1.052	1.120	1.276	1.275	1.282	1.253
*.9000	1.138	1.105	1.106	1.081	1.071	1.161	1.364	1.360	1.375	1.344
*.9500	1.110	1.105	1.095	1.081	1.097	1.220	1.460	1.462	1.476	1.460
*.9750	1.089	1.081	1.106	1.095	1.113	1.278	1.565	1.565	1.566	1.561
.9999	1.083	1.068	1.076	1.081	1.105	1.373	1.652	1.676	1.697	1.672

TABLE II.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.512b/2; $\delta_f = 20^\circ$; $\delta_a = 0^\circ$

Pressure Coefficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°	α=12°	α=15°
.0000	.772	.074	1.141	1.891	2.027	1.778	1.918	1.946
.0125	.980	1.333	2.771	2.521	2.165	1.786	1.719	1.725
.0250	.785	1.286	2.391	2.483	2.144	1.770	1.700	1.696
.0500	.918	1.296	1.862	2.502	2.144	1.770	1.700	1.696
.1000	1.015	1.294	1.567	2.529	2.165	1.786	1.703	1.696
.2000	1.113	1.310	1.504	2.125	2.197	1.812	1.719	1.710
.3000	1.185	1.331	1.455	1.662	2.187	1.826	1.734	1.728
.4000	1.250	1.365	1.485	1.523	2.075	1.844	1.753	1.746
.5000	1.308	1.407	1.502	1.506	1.923	1.863	1.771	1.762
.6000	1.394	1.473	1.540	1.528	1.745	1.842	1.771	1.775
.7000	1.581	1.642	1.687	1.619	1.612	1.807	1.787	1.809
.7250	1.730	1.776	1.807	1.695	1.585	1.788	1.792	1.823
*.0125	1.623	1.020	.467	.215	.154	.147	.086	.068
*.0250	1.533	1.022	.617	.385	.282	.267	.204	.171
*.0500	1.442	1.080	.759	.560	.455	.421	.351	.305
*.1000	1.335	1.083	.852	.699	.601	.568	.506	.471
*.2000	1.255	1.083	.939	.815	.745	.730	.665	.626
*.3000	1.175	1.070	.966	.874	.811	.815	.765	.739
*.4000	1.105	1.033	.961	.882	.843	.858	.817	.794
*.5000	1.031	.983	.915	.861	.843	.853	.833	.815
*.6000	.929	.888	.868	.807	.787	.823	.812	.800
*.7000	.844	.796	.759	.707	.724	.781	.773	.773
*.7250	.897	.922	.906	.842	.806	.882	.883	.889
.7300	2.040	2.551	2.506	2.287	2.101	2.473	2.541	2.583
.7350	2.475	2.630	2.615	2.351	1.995	2.303	2.358	2.409
.7400	2.520	2.630	2.594	2.308	1.862	2.164	2.206	2.225
.7500	2.713	2.804	2.708	2.335	1.729	2.015	2.044	2.078
.8000	1.637	1.681	1.673	1.582	1.436	1.762	1.844	1.870
.8500	1.404	1.428	1.425	1.401	1.367	1.642	1.750	1.796
.9000	1.263	1.275	1.305	1.297	1.351	1.615	1.677	1.710
*.7350	.347	.042	.016	.003	.000	.000	.000	.000
*.7400	.283	.095	.082	.070	.096	.099	.097	.100
*.7500	.307	.237	.221	.229	.261	.261	.257	.258
*.8000	.587	.596	.581	.581	.604	.616	.642	.636
*.8500	.721	.727	.718	.699	.726	.794	.807	.821
*.9000	.828	.827	.830	.810	.862	.954	.969	.999
*.9500	.913	.912	.936	.947	1.000	1.146	1.169	1.202
*.9750	.988	1.001	1.018	1.017	1.128	1.306	1.344	1.373
.9999	1.071	1.075	1.119	1.146	1.303	1.578	1.656	1.683

TABLE III.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.512b/2; $\delta_f = 30^\circ$; $\delta_a = 0^\circ$

Pressure Coefficient, S_p

x/c	$\alpha=-6^\circ$	$\alpha=-3^\circ$	$\alpha=0^\circ$	$\alpha=3^\circ$	$\alpha=6^\circ$	$\alpha=9^\circ$
.0000	.104	.203	1.299	2.036	2.214	1.953
.0125	1.020	1.479	2.732	2.516	2.271	1.921
.0250	.903	1.420	2.489	2.484	2.263	1.891
.0500	1.006	1.420	2.435	2.484	2.260	1.883
.1000	1.157	1.404	1.987	2.527	2.241	1.813
.2000	1.217	1.404	1.585	2.414	2.276	1.840
.3000	1.288	1.404	1.561	2.057	2.276	1.861
.4000	1.362	1.471	1.596	1.728	2.209	1.880
.5000	1.480	1.554	1.666	1.631	2.109	1.883
.6000	1.595	1.656	1.733	1.658	1.949	1.872
.7000	1.904	1.939	1.985	1.806	1.793	1.832
.7250	2.150	2.159	2.187	1.898	1.760	1.832
*.0125	1.176	.827	.432	.200	.130	.108
*.0250	1.182	.877	.545	.340	.262	.229
*.0500	1.116	.947	.707	.518	.419	.387
*.1000	1.182	.982	.783	.618	.556	.519
*.2000	1.121	.982	.859	.751	.697	.678
*.3000	1.061	.976	.869	.786	.759	.742
*.4000	.979	.942	.864	.788	.783	.775
*.5000	.881	.869	.813	.761	.743	.753
*.6000	.755	.741	.716	.702	.686	.699
*.7000	.662	.645	.608	.567	.562	.573
*.7250	.812	.813	.788	.718	.680	.705
.7300	4.086	4.181	4.120	3.505	2.932	2.959
.7350	4.390	4.491	4.390	3.580	2.754	2.625
.7400	4.428	4.545	4.423	3.534	2.511	2.367
.7500	4.204	4.149	3.983	3.105	2.106	2.079
.8000	1.939	1.969	1.952	1.747	1.580	1.749
.8500	1.559	1.581	1.593	1.507	1.474	1.692
.9000	1.365	1.372	1.409	1.388	1.442	1.646
*.7350	.167	.174	.162	.108	.095	.073
*.7400	.000	.000	.000	.000	.000	.000
*.7500	.000	.029	.027	.038	.059	.062
*.8000	.328	.350	.356	.354	.386	.412
*.8500	.517	.524	.513	.513	.562	.592
*.9000	.648	.671	.670	.667	.718	.785
*.9500	.777	.819	.829	.832	.902	1.014
*.9750	.905	.931	.937	.942	1.094	1.200
.9999	1.072	1.126	1.134	1.161	1.377	1.611

TABLE IV.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.512b/2; $\delta_f = 35^\circ$; $\delta_a = 0^\circ$

Pressure Coefficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°	α=12°	α=15°
.0000	.066	.292	1.404	2.177	2.243	2.003	2.039	1.913
.0125	.931	1.495	2.647	2.623	2.265	1.932	1.778	1.803
.0250	.974	1.418	2.536	2.538	2.227	1.904	1.765	1.789
.0500	1.062	1.420	2.503	2.552	2.222	1.893	1.765	1.789
.1000	1.161	1.444	2.134	2.571	2.195	1.844	1.765	1.800
.2000	1.230	1.423	1.591	2.519	2.227	1.869	1.767	1.808
.3000	1.287	1.431	1.575	2.215	2.243	1.888	1.792	1.819
.4000	1.370	1.481	1.605	1.849	2.200	1.913	1.813	1.836
.5000	1.460	1.569	1.653	1.704	2.106	1.926	1.810	1.860
.6000	1.589	1.670	1.738	1.707	1.972	1.885	1.835	1.918
.7000	1.919	1.957	1.993	1.887	1.829	1.915	1.910	1.940
.7250	2.138	2.180	2.183	1.997	1.800	1.943	1.910	1.934
*.0125	1.189	.863	.434	.183	.132	.099	.091	.047
*.0250	1.178	.897	.538	.320	.253	.227	.167	.110
*.0500	1.136	.945	.690	.490	.409	.386	.307	.260
*.1000	1.136	.945	.760	.613	.541	.523	.457	.395
*.2000	1.084	.964	.842	.733	.678	.666	.603	.553
*.3000	1.038	.953	.861	.782	.726	.729	.683	.641
*.4000	.950	.900	.842	.782	.753	.759	.710	.688
*.5000	.832	.836	.785	.736	.710	.726	.694	.674
*.6000	.725	.693	.662	.667	.662	.677	.635	.619
*.7000	.607	.592	.565	.528	.506	.507	.479	.480
*.7250	.793	.810	.771	.725	.651	.600	.576	.578
.7300	5.130	5.132	5.023	4.447	3.255	2.099	1.985	2.050
.7350	5.427	5.384	5.202	4.420	2.948	1.899	1.738	1.808
.7400	5.284	5.222	5.028	4.220	2.556	1.850	1.705	1.759
.7500	4.697	4.514	4.287	3.539	2.109	1.751	1.679	1.740
.8000	1.979	1.981	2.012	1.920	1.619	1.718	1.679	1.740
.8500	1.639	1.675	1.708	1.679	1.544	1.718	1.689	1.740
.9000	1.537	1.590	1.613	1.537	1.496	1.729	1.708	1.754
*.7350	.343	.348	.320	.276	.194	.126	.108	.121
*.7400	.000	.019	.008	.000	.000	.003	.000	.000
*.7500	.000	.000	.011	.000	.022	.036	.038	.036
*.8000	.269	.297	.288	.284	.307	.343	.325	.345
*.8500	.469	.486	.472	.473	.471	.540	.511	.521
*.9000	.623	.653	.657	.635	.662	.732	.710	.723
*.9500	.815	.839	.847	.826	.858	.970	.968	.986
*.9750	.972	1.001	1.021	.971	1.054	1.211	1.181	1.211
.9999	1.287	1.357	1.385	1.264	1.431	1.688	1.679	1.726

TABLE V.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.512b/2; $\delta_f = 40^\circ$; $\delta_a = 0^\circ$

Pressure Coefficient, S_p

x/c	$\alpha=-6^\circ$	$\alpha=-3^\circ$	$\alpha=0^\circ$	$\alpha=3^\circ$	$\alpha=6^\circ$	$\alpha=9^\circ$	$\alpha=12^\circ$	$\alpha=15^\circ$
.0000	.044	.313	1.254	2.055	2.075	1.990	2.081	1.881
.0125	.857	1.534	2.478	2.543	2.086	1.901	1.815	1.760
.0250	.937	1.469	2.402	2.496	2.062	1.890	1.796	1.755
.0500	1.033	1.453	2.353	2.496	2.062	1.861	1.796	1.755
.1000	1.152	1.442	2.064	2.535	2.046	1.823	1.794	1.755
.2000	1.218	1.428	1.586	2.450	2.094	1.845	1.812	1.776
.3000	1.292	1.428	1.556	2.178	2.115	1.871	1.831	1.779
.4000	1.355	1.474	1.588	1.830	2.096	1.888	1.847	1.790
.5000	1.438	1.544	1.635	1.666	2.027	1.888	1.853	1.814
.6000	1.543	1.623	1.697	1.636	1.900	1.837	1.869	1.883
.7000	1.829	1.906	1.928	1.770	1.725	1.882	1.955	1.899
.7250	2.017	2.095	2.100	1.869	1.688	1.901	1.939	1.875
*.0125	1.141	.783	.468	.211	.133	.115	.070	.048
*.0250	1.138	.842	.555	.329	.244	.217	.156	.112
*.0500	1.091	.921	.721	.504	.400	.379	.311	.235
*.1000	1.146	.929	.753	.589	.538	.497	.430	.353
*.2000	1.080	.956	.832	.712	.657	.647	.585	.522
*.3000	1.019	.934	.857	.759	.708	.701	.658	.599
*.4000	.931	.891	.824	.762	.723	.720	.682	.637
*.5000	.818	.821	.759	.701	.660	.677	.650	.613
*.6000	.680	.664	.617	.575	.567	.615	.591	.562
*.7000	.570	.556	.520	.493	.421	.432	.419	.401
*.7250	.760	.778	.751	.674	.501	.502	.497	.476
.7300	5.797	5.951	5.859	5.042	1.728	1.716	1.735	1.712
.7350	5.527	5.724	5.628	4.707	1.797	1.708	1.708	1.699
.7400	5.157	5.368	5.280	4.379	1.813	1.700	1.708	1.691
.7500	3.664	3.834	3.824	3.252	1.688	1.694	1.697	1.691
.8000	1.912	1.966	1.912	1.869	1.609	1.694	1.718	1.699
.8500	1.846	1.860	1.825	1.817	1.598	1.694	1.729	1.699
.9000	1.873	1.863	1.871	1.852	1.598	1.708	1.740	1.720
*.7350	.501	.540	.503	.444	.172	.158	.156	.166
*.7400	.025	.065	.049	.047	.008	.011	.005	.011
*.7500	.000	.000	.000	.000	.008	.013	.005	.016
*.8000	.201	.238	.209	.230	.273	.279	.269	.265
*.8500	.397	.432	.405	.411	.437	.459	.443	.433
*.9000	.609	.632	.601	.603	.623	.658	.652	.642
*.9500	.851	.891	.865	.860	.877	.910	.913	.910
*.9750	1.085	1.126	1.102	1.080	1.084	1.133	1.155	1.150
.9999	1.659	1.725	1.727	1.677	1.571	1.678	1.702	1.693

TABLE VI.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.512b/2; $\delta_f = 45^\circ$; $\delta_a = 0^\circ$

Pressure Coefficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°	α=12°	α=15°
.0000	.184	.156	1.038	1.740	1.964	1.973	2.085	1.889
.0125	.834	1.245	2.367	2.359	2.023	1.836	1.822	1.806
.0250	.869	1.289	2.281	2.322	2.020	1.836	1.822	1.795
.0500	.983	1.305	2.125	2.322	2.004	1.836	1.822	1.795
.1000	1.080	1.337	1.697	2.357	1.993	1.796	1.822	1.819
.2000	1.156	1.329	1.469	2.221	2.023	1.830	1.830	1.814
.3000	1.204	1.332	1.426	1.881	2.047	1.844	1.860	1.832
.4000	1.247	1.353	1.431	1.567	2.020	1.870	1.868	1.835
.5000	1.288	1.364	1.404	1.439	1.977	1.878	1.868	1.862
.6000	1.307	1.356	1.364	1.386	1.871	1.812	1.871	1.937
.7000	1.269	1.278	1.307	1.314	1.716	1.870	1.979	1.963
.7250	1.258	1.245	1.248	1.269	1.682	1.870	1.952	1.929
*.0125	1.245	.958	.525	.269	.128	.098	.038	.035
*.0250	1.212	.979	.586	.396	.245	.201	.138	.104
*.0500	1.148	1.003	.737	.540	.400	.354	.285	.222
*.1000	1.161	.968	.751	.614	.517	.466	.426	.337
*.2000	1.094	.968	.834	.705	.634	.619	.546	.503
*.3000	1.015	.939	.829	.732	.677	.659	.627	.562
*.4000	.923	.882	.794	.718	.688	.674	.649	.594
*.5000	.805	.772	.726	.676	.629	.622	.603	.567
*.6000	.632	.624	.570	.535	.514	.495	.535	.524
*.7000	.478	.460	.417	.391	.376	.381	.356	.353
*.7250	.529	.525	.479	.450	.421	.444	.451	.428
.7300	2.538	2.230	2.109	1.737	1.642	1.695	1.713	1.723
.7350	2.079	2.096	2.069	1.764	1.642	1.666	1.708	1.699
.7400	1.895	2.004	1.996	1.753	1.642	1.666	1.719	1.699
.7500	1.847	1.945	1.921	1.716	1.642	1.666	1.713	1.707
.8000	1.812	1.886	1.856	1.708	1.642	1.680	1.735	1.723
.8500	1.809	1.835	1.770	1.689	1.642	1.682	1.732	1.736
.9000	1.858	1.859	1.786	1.721	1.663	1.703	1.740	1.736
*.7350	.397	.299	.280	.237	.243	.238	.236	.246
*.7400	.049	.043	.032	.021	.029	.034	.011	.027
*.7500	.000	.003	.000	.000	.000	.000	.000	.000
*.8000	.184	.215	.188	.189	.184	.190	.160	.169
*.8500	.386	.398	.382	.383	.365	.370	.345	.348
*.9000	.594	.627	.592	.551	.533	.553	.551	.546
*.9500	.888	.915	.885	.822	.794	.820	.828	.803
*.9750	1.134	1.159	1.127	1.072	1.029	1.058	1.062	1.065
.9999	1.742	1.773	1.732	1.665	1.623	1.656	1.702	1.691

TABLE VII.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.512b/2; $\delta_f = 35^\circ$; $\delta_a = -6^\circ$

Pressure Coefficient, S_p

x/c	$\alpha=-6^\circ$	$\alpha=-3^\circ$	$\alpha=0^\circ$	$\alpha=3^\circ$	$\alpha=6^\circ$	$\alpha=9^\circ$
.0000	.047	.291	1.171	2.043	2.278	1.956
.0125	.973	1.471	2.317	2.501	2.300	1.887
.0250	1.000	1.461	2.304	2.487	2.273	1.860
.0500	1.096	1.445	2.276	2.501	2.273	1.850
.1000	1.203	1.442	2.060	2.555	2.248	1.797
.2000	1.260	1.431	1.577	2.442	2.316	1.810
.3000	1.329	1.431	1.556	2.161	2.294	1.829
.4000	1.403	1.506	1.602	1.838	2.229	1.852
.5000	1.504	1.592	1.664	1.709	2.142	1.860
.6000	1.652	1.714	1.778	1.725	1.984	1.823
.7000	2.041	2.055	2.095	1.919	1.837	1.847
.7250	2.291	2.295	2.293	2.059	1.796	1.866
*.0125	1.181	.834	.493	.259	.153	.106
*.0250	1.173	.877	.591	.361	.256	.223
*.0500	1.137	.944	.726	.523	.411	.371
*.1000	1.170	.944	.756	.614	.570	.514
*.2000	1.110	.968	.840	.738	.692	.647
*.3000	1.047	.947	.848	.782	.747	.710
*.4000	.975	.909	.848	.782	.758	.742
*.5000	.871	.850	.770	.738	.709	.702
*.6000	.726	.697	.642	.658	.670	.663
*.7000	.617	.597	.564	.531	.523	.488
*.7250	.819	.829	.791	.741	.643	.572
.7300	5.592	5.531	5.431	4.692	3.393	2.139
.7350	5.943	5.835	5.686	4.703	3.090	1.958
.7400	5.773	5.665	5.509	4.501	2.643	1.836
.7500	5.165	4.933	4.705	3.695	2.172	1.701
.8000	2.104	2.074	2.054	1.895	1.624	1.648
.8500	1.641	1.641	1.640	1.614	1.553	1.643
.9000	1.408	1.431	1.409	1.399	1.485	1.675
*.7350	.381	.379	.363	.318	.210	.119
*.7400	.005	.032	.011	.013	.000	.000
*.7500	.000	.000	.000	.011	.000	.021
*.8000	.271	.266	.266	.270	.289	.313
*.8500	.441	.452	.428	.442	.469	.504
*.9000	.592	.608	.580	.571	.643	.689
*.9500	.762	.785	.759	.757	.847	.930
*.9750	.888	.893	.900	.906	1.041	1.155
.9999	1.132	1.178	1.149	1.186	1.433	1.617

TABLE VIII.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.512b/2; $\delta_f = 35^\circ$; $\delta_a = -12^\circ$

Pressure Coefficient, S_p

x/c	$\alpha=-6^\circ$	$\alpha=-3^\circ$	$\alpha=0^\circ$	$\alpha=3^\circ$	$\alpha=6^\circ$	$\alpha=9^\circ$
.0000	.122	.247	.867	2.068	2.232	1.929
.0125	.911	1.474	2.140	2.457	2.261	1.887
.0250	.940	1.463	2.073	2.420	2.251	1.863
.0500	1.030	1.453	1.996	2.444	2.248	1.855
.1000	1.147	1.453	1.757	2.500	2.214	1.768
.2000	1.211	1.436	1.535	2.425	2.251	1.789
.3000	1.288	1.453	1.525	2.095	2.235	1.805
.4000	1.362	1.514	1.578	1.772	2.172	1.810
.5000	1.466	1.619	1.659	1.655	2.082	1.807
.6000	1.620	1.787	1.774	1.684	1.939	1.776
.7000	1.997	2.109	2.092	1.889	1.780	1.786
.7250	2.241	2.359	2.322	2.007	1.759	1.813
*.0125	1.227	.861	.631	.200	.135	.122
*.0250	1.203	.915	.682	.314	.262	.239
*.0500	1.147	.987	.816	.490	.428	.398
*.1000	1.181	.979	.832	.597	.534	.525
*.2000	1.120	1.001	.899	.728	.688	.668
*.3000	1.049	.995	.899	.778	.746	.734
*.4000	.977	.952	.877	.776	.759	.755
*.5000	.871	.866	.811	.720	.730	.721
*.6000	.727	.732	.679	.634	.659	.665
*.7000	.627	.619	.583	.525	.503	.498
*.7250	.807	.839	.816	.720	.661	.599
.7300	5.491	5.749	5.628	4.645	3.510	2.218
.7350	5.814	6.053	5.885	4.602	3.259	2.019
.7400	5.666	5.880	5.698	4.392	2.806	1.889
.7500	5.108	5.149	4.938	3.651	2.288	1.712
.8000	2.076	2.195	2.129	1.876	1.632	1.640
.8500	1.651	1.695	1.688	1.588	1.521	1.640
.9000	1.466	1.480	1.404	1.412	1.439	1.654
*.7350	.393	.398	.415	.312	.225	.143
*.7400	.011	.038	.043	.024	.008	.011
*.7500	.000	.000	.016	.000	.019	.037
*.8000	.244	.288	.276	.256	.304	.339
*.8500	.435	.457	.468	.421	.473	.525
*.9000	.571	.613	.610	.570	.643	.710
*.9500	.735	.775	.776	.754	.841	.959
*.9750	.873	.923	.928	.890	1.021	1.171
.9999	1.120	1.186	1.188	1.146	1.365	1.635

TABLE IX.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.512b/2; $\delta_f = 35^\circ$; $\delta_a = 6^\circ$

Pressure Coefficient, S_p

x/c	$\alpha=-6^\circ$	$\alpha=-3^\circ$	$\alpha=0^\circ$	$\alpha=3^\circ$	$\alpha=6^\circ$	$\alpha=9^\circ$
.0000	7.370	.164	1.140	1.913	2.170	2.006
.0125	.923	1.384	2.307	2.466	2.218	1.935
.0250	.954	1.307	2.236	2.485	2.154	1.885
.0500	1.044	1.329	2.247	2.447	2.165	1.888
.1000	1.181	1.422	2.137	2.509	2.138	1.814
.2000	1.225	1.422	1.581	2.385	2.186	1.861
.3000	1.302	1.422	1.556	2.149	2.207	1.869
.4000	1.392	1.482	1.595	1.827	2.157	1.906
.5000	1.491	1.562	1.669	1.715	2.091	1.893
.6000	1.611	1.687	1.743	1.699	1.954	1.859
.7000	1.954	2.024	2.039	1.864	1.782	1.890
.7250	2.217	2.270	2.230	1.962	1.748	1.925
*.0125	1.000	.842	.488	.285	.127	.119
*.0250	1.028	.892	.575	.360	.227	.214
*.0500	1.000	.941	.715	.531	.396	.364
*.1000	1.077	.957	.740	.604	.512	.496
*.2000	1.041	.976	.830	.718	.649	.644
*.3000	.989	.941	.852	.753	.702	.710
*.4000	.923	.897	.817	.756	.723	.734
*.5000	.803	.831	.767	.699	.673	.700
*.6000	.693	.662	.644	.642	.628	.644
*.7000	.573	.585	.543	.507	.480	.494
*.7250	.759	.799	.773	.705	.605	.581
.7300	5.403	5.508	5.321	4.501	3.099	2.141
.7350	5.598	5.776	5.549	4.556	2.862	1.988
.7400	5.370	5.615	5.370	4.377	2.445	1.866
.7500	4.935	4.945	4.600	3.574	2.028	1.700
.8000	2.058	2.117	2.107	1.851	1.584	1.653
.8500	1.619	1.693	1.699	1.596	1.502	1.642
.9000	1.466	1.466	1.447	1.442	1.418	1.666
*.7350	.345	.361	.345	.295	.182	.132
*.7400	.000	.000	.005	.022	.000	.000
*.7500	.000	.000	.000	.008	.000	.034
*.8000	.238	.249	.247	.257	.280	.322
*.8500	.422	.440	.433	.442	.454	.502
*.9000	.567	.591	.592	.580	.626	.686
*.9500	.732	.774	.786	.762	.855	.929
*.9750	.874	.941	.915	.902	1.024	1.146
.9999	1.121	1.127	1.192	1.184	1.402	1.602

TABLE X.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.512b/2; $\delta_f = 35^\circ$; $\delta_a = 12^\circ$

Pressure Coefficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°
.0000	.000	.357	1.394	2.078	2.254	2.033
.0125	.981	1.469	2.551	2.471	2.235	1.913
.0250	.992	1.447	2.411	2.434	2.235	1.902
.0500	1.077	1.453	2.413	2.445	2.235	1.894
.1000	1.200	1.469	2.243	2.466	2.185	1.867
.2000	1.255	1.447	1.614	2.408	2.235	1.886
.3000	1.310	1.450	1.551	2.199	2.251	1.902
.4000	1.403	1.517	1.593	1.859	2.202	1.935
.5000	1.512	1.590	1.654	1.695	2.147	1.965
.6000	1.617	1.689	1.731	1.653	2.017	1.892
.7000	1.945	2.003	2.002	1.758	1.850	1.957
.7250	2.203	2.234	2.177	1.840	1.823	1.984
*.0125	1.025	.771	.419	.214	.128	.100
*.0250	1.058	.824	.499	.317	.251	.217
*.0500	1.022	.899	.658	.480	.414	.360
*.1000	1.107	.910	.709	.570	.534	.499
*.2000	1.041	.932	.802	.686	.676	.650
*.3000	1.003	.916	.818	.723	.719	.715
*.4000	.937	.881	.802	.723	.738	.748
*.5000	.811	.803	.746	.689	.698	.710
*.6000	.701	.669	.621	.628	.651	.645
*.7000	.592	.585	.542	.507	.491	.501
*.7250	.767	.797	.757	.678	.616	.607
.7300	5.327	5.378	5.119	4.134	3.025	2.062
.7350	5.551	5.641	5.302	4.081	2.839	1.897
.7400	5.365	5.459	5.103	3.897	2.496	1.816
.7500	4.842	4.771	4.333	3.147	2.060	1.696
.8000	1.992	2.030	1.981	1.753	1.638	1.664
.8500	1.589	1.632	1.622	1.531	1.542	1.694
.9000	1.502	1.450	1.436	1.412	1.515	1.694
*.7350	.318	.362	.340	.272	.202	.114
*.7400	.000	.019	.019	.032	.011	.011
*.7500	.000	.000	.000	.029	.019	.030
*.8000	.225	.266	.260	.251	.294	.325
*.8500	.427	.454	.433	.441	.480	.518
*.9000	.575	.604	.600	.591	.668	.710
*.9500	.748	.797	.781	.776	.883	.949
*.9750	.904	.932	.935	.927	1.074	1.171
.9999	1.167	1.214	1.211	1.222	1.472	1.642

TABLE XI.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.512b/2; $\delta_f = 0^\circ$; $\delta_a = 12^\circ$

Pressure Coefficient, S_p

x/c	$\alpha=-6^\circ$	$\alpha=-3^\circ$	$\alpha=0^\circ$	$\alpha=3^\circ$	$\alpha=6^\circ$	$\alpha=9^\circ$	$\alpha=12^\circ$	$\alpha=15^\circ$
.0000	1.343	.562	.273	1.138	1.550	1.634	1.704	1.868
.0125	.366	.705	1.426	2.389	2.207	1.906	1.725	1.647
.0250	.483	.797	1.354	2.302	2.183	1.909	1.699	1.663
.0500	.649	.919	1.354	2.255	2.183	1.893	1.704	1.658
.1000	.833	1.045	1.354	1.917	2.144	1.806	1.691	1.685
.2000	.949	1.098	1.311	1.423	2.075	1.855	1.704	1.685
.3000	.999	1.130	1.273	1.368	1.903	1.866	1.739	1.706
.4000	1.049	1.146	1.268	1.338	1.634	1.844	1.744	1.733
.5000	1.096	1.159	1.263	1.317	1.457	1.833	1.757	1.755
.6000	1.104	1.146	1.231	1.262	1.317	1.758	1.768	1.769
.7000	1.116	1.135	1.198	1.222	1.241	1.698	1.790	1.804
.7250	1.121	1.135	1.212	1.209	1.241	1.715	1.840	1.825
*.0125	2.320	1.742	1.075	.533	.301	.208	.126	.086
*.0250	2.389	1.600	1.022	.618	.396	.319	.233	.197
*.0500	1.829	1.449	1.097	.763	.568	.500	.417	.356
*.1000	1.532	1.299	1.043	.821	.665	.629	.546	.486
*.2000	1.426	1.254	1.097	.942	.832	.786	.730	.689
*.3000	1.354	1.238	1.137	1.003	.892	.886	.848	.826
*.4000	1.301	1.220	1.132	1.040	.956	.948	.923	.937
*.5000	1.249	1.183	1.124	1.045	.987	.991	1.000	1.015
*.6000	1.213	1.164	1.124	1.051	.998	1.042	1.054	1.075
*.7000	1.210	1.164	1.148	1.106	1.064	1.131	1.172	1.193
*.7250	1.185	1.162	1.166	1.183	1.151	1.304	1.372	1.426
.7300	1.182	1.177	1.233	1.286	1.315	1.995	2.306	2.371
.7350	1.132	1.156	1.198	1.167	1.167	1.515	1.709	1.766
.7400	1.107	1.140	1.196	1.188	1.191	1.553	1.763	1.820
.7500	1.154	1.162	1.198	1.193	1.209	1.547	1.731	1.793
.8000	1.027	1.067	1.153	1.180	1.172	1.542	1.712	1.739
.8500	1.030	1.064	1.129	1.146	1.154	1.482	1.709	1.744
.9000	1.060	1.053	1.113	1.125	1.117	1.466	1.709	1.747
*.7350	1.057	.942	.907	.705	.649	.667	.704	.710
*.7400	1.121	1.040	.952	.892	.876	.899	.923	.983
*.7500	1.116	1.074	1.022	.974	.950	.988	1.025	1.058
*.8000	1.116	1.096	1.094	1.035	1.003	1.075	1.142	1.177
*.8500	1.093	1.077	1.070	1.056	1.030	1.110	1.185	1.245
*.9000	1.068	1.072	1.065	1.038	1.035	1.148	1.249	1.320
*.9500	1.052	1.053	1.065	1.056	1.056	1.207	1.356	1.436
*.9750	1.043	1.064	1.078	1.048	1.082	1.285	1.447	1.536
.9999	1.049	1.051	1.067	1.069	1.093	1.366	1.565	1.679

TABLE XII.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.512b/2; $\delta_f = 0^\circ$; $\delta_a = 6^\circ$

Pressure Coefficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°	α=12°	α=15°
.0000	1.370	.587	.246	1.079	1.501	1.525	1.669	1.808
.0125	.362	.676	1.307	2.389	2.296	1.809	1.715	1.621
.0250	.463	.812	1.233	2.308	2.223	1.794	1.707	1.640
.0500	.630	.922	1.249	2.198	2.180	1.775	1.702	1.640
.1000	.833	1.027	1.280	1.802	2.156	1.741	1.672	1.640
.2000	.934	1.069	1.238	1.436	2.059	1.762	1.699	1.661
.3000	1.008	1.114	1.228	1.388	1.882	1.794	1.726	1.680
.4000	1.047	1.121	1.228	1.350	1.630	1.788	1.732	1.704
.5000	1.082	1.137	1.228	1.310	1.453	1.786	1.753	1.728
.6000	1.091	1.116	1.191	1.264	1.334	1.720	1.762	1.731
.7000	1.085	1.114	1.151	1.221	1.243	1.673	1.783	1.763
.7250	1.126	1.111	1.164	1.221	1.243	1.699	1.835	1.808
*.0125	2.296	1.713	1.196	.592	.352	.200	.130	.094
*.0250	2.436	1.582	1.114	.656	.454	.321	.260	.211
*.0500	1.978	1.478	1.156	.829	.620	.484	.444	.380
*.1000	1.545	1.305	1.096	.855	.701	.626	.558	.506
*.2000	1.447	1.268	1.114	.966	.843	.784	.756	.709
*.3000	1.373	1.242	1.135	1.038	.921	.871	.854	.843
*.4000	1.312	1.210	1.154	1.073	.983	.949	.949	.920
*.5000	1.260	1.192	1.140	1.081	1.012	.999	1.011	1.008
*.6000	1.233	1.163	1.130	1.076	1.036	1.036	1.062	1.099
*.7000	1.217	1.174	1.148	1.108	1.090	1.126	1.182	1.196
*.7250	1.189	1.161	1.156	1.173	1.152	1.294	1.382	1.420
.7300	1.178	1.163	1.191	1.275	1.308	1.994	2.282	2.338
.7350	1.134	1.145	1.175	1.178	1.184	1.518	1.699	1.731
.7400	1.104	1.137	1.170	1.200	1.195	1.554	1.762	1.790
.7500	1.151	1.148	1.175	1.200	1.203	1.520	1.718	1.763
.8000	1.025	1.058	1.106	1.173	1.173	1.539	1.696	1.709
.8500	1.025	1.069	1.106	1.146	1.168	1.496	1.696	1.709
.9000	1.041	1.069	1.104	1.130	1.136	1.465	1.696	1.733
*.7350	1.074	.998	.956	.721	.682	.658	.699	.725
*.7400	1.145	1.058	.977	.904	.881	.926	.954	.992
*.7500	1.115	1.087	1.038	.998	.969	.968	1.030	1.062
*.8000	1.115	1.090	1.085	1.065	1.034	1.084	1.149	1.185
*.8500	1.091	1.074	1.067	1.071	1.044	1.115	1.203	1.252
*.9000	1.058	1.072	1.072	1.068	1.071	1.141	1.274	1.329
*.9500	1.044	1.048	1.067	1.065	1.077	1.199	1.341	1.431
*.9750	1.058	1.069	1.064	1.076	1.101	1.262	1.447	1.546
.9999	1.038	1.038	1.048	1.081	1.112	1.362	1.583	1.659

TABLE XIII.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.512b/2; $\delta_f = 0^\circ$; $\delta_a = -6^\circ$

Pressure Coefficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°	α=12°	α=15°
.0000	1.471	.754	.238	1.010	1.444	1.479	1.599	1.816
.0125	.339	.950	1.101	2.461	2.204	1.776	1.676	1.672
.0250	.451	.757	1.159	2.258	2.138	1.776	1.676	1.632
.0500	.606	.872	1.188	2.023	2.080	1.762	1.676	1.632
.1000	.777	.992	1.198	1.556	2.096	1.738	1.658	1.643
.2000	.878	1.039	1.250	1.365	1.996	1.757	1.674	1.659
.3000	.948	1.073	1.203	1.308	1.749	1.778	1.698	1.686
.4000	.999	1.075	1.201	1.276	1.491	1.773	1.708	1.708
.5000	1.028	1.081	1.188	1.235	1.344	1.746	1.727	1.724
.6000	1.033	1.073	1.175	1.188	1.233	1.701	1.735	1.740
.7000	1.039	1.073	1.148	1.148	1.173	1.634	1.730	1.762
.7850	1.076	1.073	1.122	1.151	1.168	1.663	1.788	1.800
*.0125	2.336	1.793	1.138	.592	.326	.208	.152	.095
*.0250	2.470	1.650	1.250	.642	.439	.339	.269	.201
*.0500	2.283	1.582	1.167	.806	.618	.502	.440	.350
*.1000	1.730	1.336	1.214	.843	.707	.635	.568	.510
*.2000	1.428	1.302	1.117	.974	.849	.806	.765	.725
*.3000	1.370	1.268	1.169	1.031	.949	.913	.871	.839
*.4000	1.330	1.235	1.175	1.081	1.002	1.004	.967	.945
*.5000	1.274	1.216	1.185	1.083	1.034	1.031	1.031	1.021
*.6000	1.244	1.201	1.167	1.078	1.031	1.063	1.090	1.094
*.7000	1.210	1.185	1.148	1.101	1.076	1.153	1.197	1.208
*.7250	1.177	1.151	1.161	1.128	1.118	1.324	1.375	1.434
.7300	1.159	1.146	1.159	1.182	1.173	1.994	2.231	2.332
.7350	1.119	1.138	1.164	1.156	1.144	1.525	1.655	1.729
.7400	1.119	1.133	1.164	1.161	1.147	1.549	1.706	1.800
.7500	1.127	1.148	1.154	1.156	1.149	1.517	1.644	1.770
.8000	.991	1.028	1.151	1.117	1.128	1.501	1.668	1.708
.8500	.996	1.028	1.091	1.101	1.123	1.490	1.647	1.727
.9000	1.020	1.036	1.075	1.083	1.107	1.452	1.650	1.740
*.7350	1.119	1.065	.981	.895	.802	.676	.709	.738
*.7400	1.153	1.101	1.104	.934	.899	.908	.962	.991
*.7500	1.137	1.128	1.041	1.010	.976	.996	1.053	1.089
*.8000	1.132	1.114	1.065	1.060	1.041	1.084	1.159	1.203
*.8500	1.111	1.088	1.091	1.057	1.047	1.127	1.197	1.257
*.9000	1.076	1.070	1.083	1.062	1.049	1.161	1.261	1.333
*.9500	1.055	1.057	1.081	1.057	1.060	1.196	1.335	1.431
*.9750	1.039	1.034	1.070	1.067	1.070	1.276	1.428	1.523
.9999	1.028	1.031	1.101	1.062	1.073	1.367	1.548	1.656

TABLE XIV.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.512b/2; $\delta_f = 0^\circ$; $\delta_a = -12^\circ$

Pressure coefficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°	α=12°	α=15°
.0000	1.471	.945	.286	.874	1.436	1.442	1.639	1.863
.0125	.299	.528	1.156	2.265	2.283	1.815	1.677	1.648
.0250	.397	.684	1.123	2.084	2.222	1.802	1.698	1.646
.0500	.573	.829	1.188	1.844	2.164	1.802	1.696	1.646
.1000	.751	.956	1.250	1.546	2.208	1.745	1.690	1.662
.2000	.874	1.016	1.215	1.388	2.079	1.775	1.693	1.681
.3000	.929	1.061	1.204	1.330	1.799	1.786	1.723	1.703
.4000	.967	1.072	1.193	1.298	1.524	1.756	1.785	1.719
.5000	1.003	1.096	1.193	1.266	1.400	1.743	1.742	1.746
.6000	1.006	1.069	1.156	1.210	1.293	1.683	1.747	1.752
.7000	1.028	1.077	1.139	1.175	1.216	1.627	1.742	1.771
.7250	1.044	1.096	1.150	1.173	1.213	1.614	1.790	1.803
*.0125	2.244	1.882	1.350	.736	.413	.231	.149	.109
*.0250	2.419	1.769	1.247	.754	.501	.341	.289	.218
*.0500	2.299	1.698	1.291	.893	.677	.518	.446	.367
*.1000	1.814	1.394	1.161	.895	.754	.652	.591	.533
*.2000	1.447	1.336	1.202	1.007	.908	.822	.780	.729
*.3000	1.386	1.309	1.215	1.101	.996	.921	.899	.862
*.4000	1.332	1.294	1.229	1.114	1.056	1.002	.996	.968
*.5000	1.280	1.251	1.220	1.130	1.095	1.047	1.058	1.050
*.6000	1.252	1.225	1.199	1.119	1.100	1.074	1.123	1.107
*.7000	1.211	1.214	1.207	1.141	1.139	1.146	1.207	1.227
*.7250	1.184	1.183	1.199	1.165	1.180	1.329	1.418	1.450
.7300	1.156	1.175	1.196	1.189	1.227	1.952	2.263	2.345
.7350	1.104	1.148	1.193	1.189	1.205	1.490	1.685	1.752
.7400	1.080	1.133	1.193	1.175	1.205	1.514	1.728	1.806
.7500	1.112	1.154	1.196	1.175	1.216	1.504	1.655	1.768
.8000	.956	1.016	1.094	1.103	1.169	1.496	1.666	1.722
.8500	.973	1.022	1.085	1.106	1.155	1.458	1.669	1.724
.9000	.997	1.038	1.085	1.093	1.166	1.431	1.666	1.735
*.7350	1.126	1.109	1.161	.997	.880	.693	.724	.756
*.7400	1.170	1.151	1.110	.965	.957	.934	.988	1.006
*.7500	1.137	1.143	1.118	1.037	1.029	1.002	1.064	1.093
*.8000	1.137	1.119	1.131	1.095	1.106	1.101	1.180	1.219
*.8500	1.101	1.122	1.189	1.093	1.111	1.141	1.212	1.273
*.9000	1.060	1.093	1.115	1.093	1.111	1.173	1.283	1.349
*.9500	1.033	1.067	1.115	1.093	1.114	1.208	1.369	1.447
*.9750	1.049	1.080	1.104	1.087	1.133	1.270	1.445	1.545
.9999	1.017	1.043	1.069	1.071	1.133	1.351	1.550	1.659

TABLE XV.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.805b/2; $\delta_f = 0^\circ$; $\delta_a = 0^\circ$

Pressure Cofficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°	α=12°	α=15°	α=18°	α=21°
.0000	1.636	.212	.102	1.103	1.291	1.515	1.650	1.734	1.880	1.834
.0125	.424	.848	1.557	2.660	2.226	2.080	1.932	1.821	1.758	1.707
.0250	.580	.901	1.438	2.534	2.223	2.083	1.924	1.810	1.758	1.707
.0500	.752	.986	1.363	2.012	2.207	2.100	1.924	1.813	1.758	1.707
.1000	.875	1.065	1.341	1.598	2.242	2.100	1.927	1.810	1.736	1.707
.2000	.985	1.100	1.286	1.410	2.094	2.122	1.954	1.829	1.752	1.707
.3000	1.061	1.124	1.280	1.361	1.685	2.122	1.954	1.834	1.766	1.720
.4000	1.078	1.129	1.258	1.329	1.423	2.064	1.982	1.847	1.787	1.744
.5000	1.102	1.140	1.250	1.297	1.312	1.933	1.982	1.860	1.800	1.768
.6000	1.102	1.129	1.214	1.237	1.246	1.780	1.965	1.871	1.816	1.786
.6900	1.094	1.102	1.181	1.213	1.219	1.632	1.927	1.902	1.837	1.805
*.0125	2.697	1.452	.871	.385	.212	.175	.113	.105	.058	.042
*.0250	2.029	1.465	1.034	.597	.416	.329	.258	.232	.162	.138
*.0500	1.772	1.330	1.026	.716	.541	.476	.414	.369	.271	.233
*.1000	1.543	1.285	1.076	.855	.694	.685	.598	.543	.430	.400
*.2000	1.414	1.238	1.125	.979	.848	.863	.791	.751	.648	.599
*.3000	1.351	1.224	1.153	1.028	.928	.964	.906	.864	.791	.747
*.4000	1.313	1.214	1.172	1.081	.986	1.042	.996	.967	.892	.869
*.5000	1.275	1.185	1.159	1.081	1.018	1.075	1.049	1.030	.974	.943
*.6000	1.225	1.153	1.142	1.089	1.026	1.106	1.098	1.091	1.046	1.039
*.6900	1.154	1.097	1.106	1.065	1.034	1.131	1.147	1.159	1.139	1.132
.7000	1.127	1.079	1.092	1.095	1.087	1.315	1.416	1.418	1.386	1.389
.7063	1.080	1.094	1.112	1.146	1.166	1.635	1.878	1.850	1.816	1.807
.7125	1.140	1.161	1.258	1.275	1.246	1.585	1.880	1.852	1.827	1.807
.7250	1.119	1.153	1.208	1.235	1.214	1.557	1.867	1.855	1.821	1.807
.7500	1.105	1.132	1.181	1.192	1.193	1.537	1.845	1.855	1.819	1.807
.8000	1.078	1.110	1.142	1.170	1.169	1.482	1.795	1.839	1.811	1.807
.8500	1.080	1.094	1.120	1.149	1.140	1.412	1.735	1.808	1.811	1.807
.9000	1.061	1.068	1.114	1.122	1.121	1.359	1.658	1.787	1.800	1.807
.9500	1.072	1.071	1.092	1.103	1.092	1.312	1.589	1.744	1.779	1.805
.9750	1.061	1.057	1.092	1.079	1.081	1.284	1.554	1.713	1.771	1.802
*.7063	1.116	1.071	1.092	1.076	1.028	1.136	1.172	1.183	1.168	1.171
*.7125	1.127	1.063	1.070	.979	.906	.986	1.032	1.054	1.062	1.063
*.7250	1.138	1.071	1.076	1.001	.973	1.089	1.093	1.115	1.104	1.116
*.7500	1.127	1.076	1.078	1.044	.996	1.100	1.125	1.149	1.136	1.150
*.8000	1.110	1.065	1.087	1.049	1.018	1.120	1.136	1.178	1.187	1.208
*.8500	1.097	1.065	1.087	1.052	1.023	1.120	1.175	1.212	1.237	1.261
*.9000	1.086	1.060	1.081	1.081	1.034	1.153	1.213	1.294	1.306	1.352
*.9500	1.078	1.060	1.081	1.071	1.049	1.181	1.290	1.370	1.423	1.468
*.9750	1.067	1.060	1.078	1.054	1.041	1.198	1.337	1.449	1.503	1.574
.9999	1.017	1.060	1.078	1.098	1.092	1.256	1.452	1.541	1.641	1.717

TABLE XVI.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.805b/2; $\delta_f = 20^\circ$; $\delta_a = 0^\circ$

Pressure Coefficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°	α=12°	α=15°
.0000	.635	.047	.830	1.385	1.819	1.620	1.645	1.836
.0125	.857	1.262	2.326	2.545	2.402	1.993	1.826	1.778
.0250	.782	1.223	1.878	2.545	2.413	1.993	1.824	1.778
.0500	.897	1.202	1.613	2.574	2.426	2.009	1.824	1.778
.1000	1.017	1.244	1.512	2.445	2.445	2.009	1.824	1.778
.2000	1.095	1.244	1.409	1.730	2.455	2.025	1.829	1.778
.3000	1.143	1.257	1.373	1.461	2.210	2.039	1.847	1.778
.4000	1.164	1.257	1.354	1.423	1.788	2.028	1.847	1.791
.5000	1.188	1.257	1.332	1.396	1.514	1.975	1.865	1.791
.6000	1.180	1.233	1.297	1.356	1.391	1.863	1.876	1.799
.6900	1.148	1.209	1.250	1.315	1.335	1.746	1.894	1.820
*.0125	1.717	1.049	.486	.258	.149	.117	.115	.074
*.0250	1.501	1.125	.693	.447	.314	.277	.231	.192
*.0500	1.420	1.099	.775	.584	.466	.416	.375	.310
*.1000	1.338	1.112	.895	.761	.644	.608	.555	.494
*.2000	1.260	1.128	.988	.885	.806	.784	.749	.679
*.3000	1.220	1.117	1.037	.955	.891	.879	.859	.810
*.4000	1.188	1.117	1.067	1.009	.958	.949	.943	.907
*.5000	1.153	1.104	1.065	1.009	.979	.991	1.009	.978
*.6000	1.113	1.078	1.065	1.033	1.005	1.039	1.056	1.052
*.6900	1.073	1.062	1.043	1.033	1.032	1.082	1.121	1.128
.7000	1.089	1.094	1.111	1.132	1.138	1.338	1.402	1.383
.7063	1.132	1.144	1.212	1.256	1.303	1.743	1.860	1.807
.7125	1.228	1.304	1.357	1.375	1.386	1.698	1.860	1.809
.7250	1.210	1.238	1.297	1.323	1.335	1.674	1.860	1.812
.7500	1.193	1.217	1.264	1.307	1.319	1.652	1.860	1.812
.8000	1.161	1.175	1.239	1.267	1.287	1.578	1.839	1.815
.8500	1.129	1.149	1.198	1.240	1.261	1.511	1.816	1.815
.9000	1.113	1.133	1.166	1.202	1.229	1.466	1.782	1.809
.9500	1.089	1.107	1.144	1.176	1.197	1.407	1.740	1.796
.9750	1.076	1.091	1.127	1.165	1.181	1.375	1.706	1.786
*.7063	1.060	1.046	1.051	1.038	1.053	1.093	1.148	1.147
*.7125	1.025	.993	.939	.928	.918	.970	1.045	1.047
*.7250	1.044	1.014	.994	.993	1.003	1.053	1.100	1.118
*.7500	1.055	1.028	1.026	1.014	1.024	1.066	1.119	1.155
*.8000	1.052	1.033	1.029	1.030	1.037	1.077	1.163	1.181
*.8500	1.057	1.038	1.029	1.046	1.053	1.114	1.221	1.247
*.9000	1.057	1.038	1.054	1.063	1.061	1.151	1.271	1.331
*.9500	1.057	1.038	1.067	1.079	1.096	1.191	1.378	1.447
*.9750	1.057	1.043	1.073	1.111	1.101	1.261	1.457	1.546
.9999	1.057	1.096	1.130	1.157	1.186	1.348	1.603	1.702

TABLE XVII.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.805b/2; $\delta_f = 30^\circ$; $\delta_a = 0^\circ$

Pressure Coefficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°
.0000	.120	.059	.980	1.598	1.985	1.716
.0125	.771	1.415	2.662	2.646	2.533	2.085
.0250	.845	1.316	2.346	2.622	2.533	2.082
.0500	.908	1.260	1.787	2.554	2.533	2.120
.1000	1.086	1.321	1.612	2.635	2.543	2.023
.2000	1.157	1.300	1.490	2.012	2.581	2.023
.3000	1.192	1.300	1.453	1.544	2.354	2.055
.4000	1.225	1.300	1.426	1.455	1.917	2.034
.5000	1.242	1.311	1.412	1.447	1.601	1.999
.6000	1.236	1.287	1.377	1.407	1.458	1.926
.6900	1.209	1.255	1.342	1.377	1.399	1.840
*.0125	1.346	.901	.427	.216	.108	.140
*.0250	1.061	.968	.689	.456	.302	.285
*.0500	1.212	.995	.743	.543	.421	.420
*.1000	1.225	1.033	.848	.691	.608	.594
*.2000	1.179	1.059	.953	.832	.761	.767
*.3000	1.162	1.059	.994	.921	.859	.869
*.4000	1.138	1.065	1.021	.956	.929	.939
*.5000	1.105	1.059	1.021	.980	.945	.987
*.6000	1.072	1.043	1.021	1.004	.983	1.028
*.6900	1.088	1.033	1.042	1.015	1.010	1.071
.7000	1.064	1.091	1.145	1.150	1.150	1.350
.7063	1.132	1.174	1.285	1.328	1.350	1.759
.7125	1.264	1.391	1.461	1.474	1.463	1.727
.7250	1.212	1.313	1.388	1.404	1.396	1.697
.7500	1.209	1.276	1.339	1.382	1.380	1.695
.8000	1.138	1.233	1.304	1.334	1.342	1.636
.8500	1.113	1.201	1.274	1.291	1.296	1.568
.9000	1.121	1.174	1.242	1.258	1.269	1.539
.9500	1.105	1.150	1.199	1.226	1.231	1.490
.9750	1.099	1.134	1.183	1.215	1.218	1.463
*.7063	1.001	1.027	1.040	1.034	1.007	1.087
*.7125	.903	.942	.886	.869	.842	.923
*.7250	.957	.952	.959	.959	.956	1.020
*.7500	.974	.995	.994	.994	.988	1.054
*.8000	.974	1.011	1.015	1.029	1.002	1.079
*.8500	.996	1.033	1.064	1.053	1.048	1.119
*.9000	1.012	1.054	1.075	1.077	1.069	1.176
*.9500	1.017	1.065	1.085	1.112	1.104	1.248
*.9750	1.031	1.070	1.107	1.137	1.129	1.291
.9999	.831	1.059	1.183	1.183	1.193	1.350

TABLE XVIII.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.805b/2; $\delta_f = 35^\circ$; $\delta_a = 0^\circ$

Pressure Coefficient, Sp

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°	α=12°	α=15°
.0000	.124	.101	1.021	1.731	2.020	1.737	1.732	1.956
.0125	.851	1.397	2.682	2.721	2.510	2.074	1.859	1.850
.0250	.917	1.312	2.435	2.705	2.510	2.077	1.859	1.850
.0500	.977	1.243	1.917	2.653	2.521	2.110	1.872	1.863
.1000	1.103	1.338	1.613	2.716	2.521	2.041	1.837	1.852
.2000	1.161	1.314	1.493	2.147	2.550	2.066	1.835	1.858
.3000	1.208	1.314	1.463	1.573	2.365	2.082	1.835	1.858
.4000	1.235	1.314	1.428	1.469	1.939	2.077	1.835	1.863
.5000	1.235	1.314	1.417	1.444	1.622	2.047	1.856	1.863
.6000	1.235	1.296	1.374	1.414	1.474	1.959	1.859	1.869
.6900	1.213	1.264	1.336	1.368	1.401	1.858	1.872	1.877
*.0125	1.340	.908	.418	.183	.132	.118	.086	.071
*.0250	1.128	.993	.690	.429	.309	.274	.237	.186
*.0500	1.227	1.004	.722	.522	.433	.411	.360	.304
*.1000	1.208	1.017	.834	.686	.611	.581	.525	.460
*.2000	1.164	1.051	.937	.834	.783	.767	.716	.663
*.3000	1.142	1.059	.988	.916	.858	.866	.823	.800
*.4000	1.125	1.062	1.015	.955	.923	.943	.901	.885
*.5000	1.090	1.057	1.015	.976	.955	.995	.958	.948
*.6000	1.071	1.054	1.029	1.001	.987	1.033	1.017	1.030
*.6900	1.032	1.035	1.043	1.020	1.017	1.085	1.095	1.110
.7000	1.087	1.099	1.143	1.157	1.170	1.373	1.391	1.414
.7063	1.169	1.192	1.287	1.324	1.383	1.808	1.829	1.866
.7125	1.296	1.370	1.434	1.452	1.474	1.770	1.829	1.863
.7250	1.268	1.304	1.379	1.398	1.418	1.748	1.829	1.860
.7500	1.230	1.274	1.333	1.370	1.391	1.734	1.829	1.860
.8000	1.180	1.235	1.309	1.343	1.369	1.663	1.800	1.855
.8500	1.153	1.203	1.271	1.310	1.342	1.600	1.770	1.847
.9000	1.147	1.187	1.254	1.285	1.305	1.551	1.749	1.841
.9500	1.131	1.163	1.216	1.255	1.278	1.504	1.727	1.825
.9750	1.128	1.152	1.200	1.239	1.259	1.474	1.705	1.822
*.7063	1.081	1.038	1.032	1.023	1.036	1.104	1.114	1.129
*.7125	.955	.937	.888	.878	.888	.943	.976	1.003
*.7250	.988	.982	.980	.974	.982	1.030	1.071	1.085
*.7500	1.005	1.017	1.005	1.009	1.003	1.069	1.087	1.112
*.8000	1.010	1.027	1.032	1.037	1.038	1.101	1.132	1.181
*.8500	1.043	1.046	1.053	1.064	1.073	1.140	1.194	1.249
*.9000	1.043	1.062	1.075	1.083	1.100	1.195	1.278	1.337
*.9500	1.071	1.081	1.108	1.138	1.143	1.260	1.383	1.480
*.9750	1.090	1.099	1.135	1.157	1.170	1.326	1.466	1.559
.9999	.961	1.073	1.151	1.220	1.245	1.397	1.558	1.710

TABLE XIX.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.805b/2; $\delta_f = 40^\circ$; $\delta_a = 0^\circ$

Pressure Coefficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°	α=12°	α=15°
.0000	.105	.089	.881	1.633	1.900	1.724	1.783	1.918
.0125	.793	1.436	2.524	2.674	2.372	2.070	1.853	1.814
.0250	.865	1.337	2.263	2.647	2.372	2.070	1.853	1.814
.0500	.915	1.277	1.776	2.543	2.372	2.100	1.874	1.814
.1000	1.074	1.334	1.583	2.652	2.377	2.000	1.847	1.819
.2000	1.146	1.315	1.480	2.104	2.419	2.000	1.858	1.827
.3000	1.185	1.312	1.420	1.570	2.295	2.027	1.858	1.827
.4000	1.198	1.310	1.404	1.452	1.953	2.006	1.861	1.827
.5000	1.220	1.304	1.387	1.425	1.664	1.979	1.880	1.827
.6000	1.209	1.277	1.344	1.397	1.476	1.922	1.880	1.827
.6900	1.179	1.242	1.308	1.351	1.378	1.828	1.896	1.846
*.0125	1.331	.875	.468	.219	.140	.121	.083	.070
*.0250	1.083	.961	.724	.477	.310	.274	.226	.182
*.0500	1.209	.986	.745	.537	.421	.400	.349	.278
*.1000	1.207	1.013	.830	.674	.596	.567	.507	.449
*.2000	1.160	1.053	.930	.825	.761	.733	.703	.639
*.3000	1.141	1.061	.974	.904	.843	.838	.816	.762
*.4000	1.119	1.050	1.012	.964	.909	.905	.908	.861
*.5000	1.091	1.050	1.012	.975	.946	.959	.956	.920
*.6000	1.066	1.034	1.039	.997	.975	1.002	1.036	1.000
*.6900	1.036	1.083	1.039	1.028	1.012	1.058	1.098	1.075
.7000	1.058	1.158	1.121	1.143	1.158	1.313	1.385	1.370
.7063	1.119	1.312	1.238	1.299	1.381	1.718	1.842	1.816
.7125	1.201	1.256	1.376	1.408	1.423	1.692	1.845	1.819
.7250	1.185	1.245	1.325	1.370	1.381	1.667	1.842	1.819
.7500	1.182	1.215	1.300	1.345	1.370	1.662	1.837	1.819
.8000	1.135	1.196	1.273	1.321	1.341	1.608	1.802	1.814
.8500	1.113	1.188	1.251	1.296	1.317	1.552	1.783	1.806
.9000	1.132	1.175	1.240	1.277	1.285	1.517	1.756	1.798
.9500	1.124	1.164	1.219	1.255	1.261	1.485	1.735	1.792
.9750	1.119	1.023	1.213	1.255	1.243	1.461	1.724	1.782
*.7063	1.019	.961	1.028	1.036	1.020	1.071	1.122	1.121
*.7125	.978	.991	.928	.915	.917	.961	1.047	1.043
*.7250	1.003	1.018	.996	1.006	.991	1.028	1.090	1.070
*.7500	1.014	1.034	1.039	1.030	1.010	1.063	1.117	1.105
*.8000	1.017	1.045	1.047	1.047	1.039	1.077	1.163	1.156
*.8500	1.036	1.061	1.072	1.074	1.063	1.136	1.219	1.225
*.9000	1.069	1.085	1.099	1.099	1.094	1.184	1.313	1.316
*.9500	1.083	1.104	1.126	1.143	1.153	1.262	1.423	1.450
*.9750	1.091	1.069	1.148	1.178	1.174	1.305	1.520	1.552
.9999	.895	1.067	1.153	1.228	1.248	1.372	1.584	1.685

TABLE XX.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.805b/2; $\delta_f = 45^\circ$; $\delta_a = 0^\circ$

Pressure Coefficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°	α=12°	α=15°
.0000	.235	.070	.794	1.335	1.751	1.674	1.833	1.969
.0125	.724	1.259	2.383	2.450	2.305	1.984	1.876	1.843
.0250	.810	1.208	2.001	2.415	2.295	1.984	1.876	1.843
.0500	.880	1.178	1.568	2.290	2.295	2.008	1.887	1.843
.1000	1.048	1.259	1.490	2.423	2.295	1.952	1.871	1.851
.2000	1.102	1.259	1.418	1.886	2.329	1.978	1.881	1.867
.3000	1.150	1.259	1.356	1.474	2.241	2.002	1.892	1.867
.4000	1.183	1.259	1.340	1.399	1.956	2.000	1.898	1.870
.5000	1.196	1.270	1.318	1.375	1.692	1.984	1.903	1.886
.6000	1.193	1.248	1.275	1.335	1.500	1.926	1.911	1.878
.6900	1.169	1.219	1.240	1.301	1.410	1.859	1.909	1.878
*.0125	1.415	1.022	.506	.263	.139	.116	.084	.059
*.0250	1.139	1.065	.745	.511	.333	.280	.204	.158
*.0500	1.253	1.065	.772	.575	.429	.402	.329	.278
*.1000	1.247	1.060	.834	.681	.586	.550	.494	.439
*.2000	1.188	1.079	.909	.830	.760	.735	.687	.626
*.3000	1.158	1.073	.963	.883	.845	.823	.795	.754
*.4000	1.137	1.081	.995	.931	.909	.891	.891	.851
*.5000	1.110	1.079	.998	.958	.935	.936	.948	.915
*.6000	1.094	1.079	1.017	.979	.975	.992	1.013	1.003
*.6900	1.069	1.065	1.030	1.000	1.005	1.050	1.094	1.097
.7000	1.064	1.087	1.089	1.107	1.154	1.328	1.358	1.372
.7063	1.121	1.151	1.197	1.253	1.378	1.767	1.854	1.856
.7125	1.172	1.270	1.313	1.357	1.412	1.735	1.862	1.854
.7250	1.166	1.245	1.272	1.317	1.375	1.709	1.849	1.851
.7500	1.177	1.240	1.245	1.293	1.372	1.709	1.843	1.851
.8000	1.156	1.205	1.245	1.293	1.338	1.648	1.803	1.830
.8500	1.137	1.202	1.221	1.269	1.317	1.592	1.781	1.835
.9000	1.145	1.184	1.205	1.250	1.287	1.553	1.748	1.816
.9500	1.145	1.184	1.205	1.234	1.287	1.513	1.748	1.814
.9750	1.150	1.184	1.192	1.234	1.282	1.497	1.721	1.811
*.7063	1.048	1.060	1.044	1.037	1.034	1.077	1.116	1.107
*.7125	1.048	1.011	.952	.952	.962	1.032	1.089	1.086
*.7250	1.053	1.046	1.006	1.005	1.007	1.069	1.105	1.105
*.7500	1.053	1.065	1.028	1.024	1.031	1.079	1.119	1.129
*.8000	1.056	1.071	1.044	1.045	1.079	1.100	1.170	1.177
*.8500	1.075	1.084	1.071	1.069	1.077	1.151	1.235	1.255
*.9000	1.085	1.103	1.087	1.101	1.122	1.209	1.306	1.351
*.9500	1.085	1.130	1.114	1.133	1.173	1.285	1.425	1.461
*.9750	1.107	1.151	1.135	1.154	1.205	1.341	1.512	1.578
.9999	.961	1.081	1.173	1.205	1.258	1.394	1.610	1.707

TABLE XXI.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.805b/2; $\delta_f = 35^\circ$; $\delta_a = -6^\circ$

Pressure Coefficient, S_p

x/c	$\alpha = -6^\circ$	$\alpha = -3^\circ$	$\alpha = 0^\circ$	$\alpha = 3^\circ$	$\alpha = 6^\circ$	$\alpha = 9^\circ$
.0000	.132	.070	.780	1.590	2.052	1.699
.0125	.836	1.401	2.417	2.668	2.570	2.051
.0250	.904	1.302	2.141	2.633	2.567	2.051
.0500	.964	1.259	1.724	2.547	2.567	2.080
.1000	1.107	1.321	1.545	2.625	2.586	2.009
.2000	1.170	1.297	1.453	1.989	2.624	2.022
.3000	1.195	1.297	1.401	1.520	2.354	2.041
.4000	1.203	1.272	1.371	1.434	1.839	2.030
.5000	1.203	1.272	1.331	1.399	1.542	1.993
.6000	1.167	1.224	1.268	1.348	1.395	1.868
.6900	1.118	1.170	1.220	1.288	1.322	1.760
*.0125	1.362	.925	.515	.243	.139	.138
*.0250	1.167	1.006	.775	.501	.341	.292
*.0500	1.255	1.006	.775	.558	.439	.421
*.1000	1.238	1.030	.846	.695	.619	.596
*.2000	1.206	1.065	.943	.841	.793	.769
*.3000	1.178	1.076	.997	.919	.875	.875
*.4000	1.178	1.092	1.041	.981	.956	.949
*.5000	1.156	1.092	1.070	1.021	1.003	1.004
*.6000	1.162	1.122	1.095	1.073	1.065	1.063
*.6900	1.118	1.103	1.089	1.073	1.065	1.108
.7000	1.038	1.084	1.084	1.105	1.090	1.222
.7063	1.085	1.108	1.176	1.256	1.294	1.685
.7125	1.060	1.038	1.089	1.164	1.191	1.598
.7250	1.047	1.009	1.062	1.175	1.196	1.566
.7500	.970	1.038	1.108	1.186	1.218	1.571
.8000	1.036	1.087	1.138	1.218	1.232	1.516
.8500	1.066	1.108	1.154	1.215	1.237	1.447
.9000	1.093	1.127	1.165	1.215	1.240	1.407
.9500	1.093	1.127	1.165	1.202	1.234	1.359
.9750	1.123	1.141	1.187	1.213	1.234	1.354
*.7063	.847	1.065	1.033	.932	.910	.567
*.7125	1.677	1.603	1.618	1.649	1.643	1.717
*.7250	1.359	1.453	1.428	1.420	1.436	1.436
*.7500	1.260	1.227	1.203	1.221	1.215	1.240
*.8000	1.173	1.178	1.173	1.178	1.180	1.182
*.8500	1.151	1.157	1.160	1.170	1.183	1.187
*.9000	1.134	1.146	1.149	1.153	1.172	1.208
*.9500	1.134	1.135	1.157	1.175	1.185	1.224
*.9750	1.118	1.138	1.160	1.180	1.207	1.264
.9999	.970	1.098	1.149	1.191	1.245	1.288

TABLE XXII.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.805b/2; $\delta_f = 35^\circ$; $\delta_a = -12^\circ$

Pressure Coefficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°
.0000	.297	.070	.532	1.623	1.968	1.707
.0125	.706	1.375	2.081	2.630	2.583	2.109
.0250	.802	1.291	1.757	2.628	2.544	2.107
.0500	.876	1.235	1.461	2.598	2.544	2.149
.1000	1.030	1.305	1.469	2.553	2.560	2.022
.2000	1.099	1.289	1.399	1.818	2.568	2.035
.3000	1.120	1.280	1.340	1.434	2.185	2.064
.4000	1.115	1.251	1.300	1.362	1.624	2.014
.5000	1.110	1.229	1.263	1.301	1.354	1.919
.6000	1.057	1.143	1.177	1.223	1.227	1.754
.6900	.956	1.044	1.073	1.114	1.111	1.585
*.0125	1.487	.968	.663	.211	.127	.138
*.0250	1.197	1.073	.885	.469	.344	.299
*.0500	1.290	1.057	.872	.536	.452	.429
*.1000	1.258	1.084	.912	.698	.614	.599
*.2000	1.216	1.119	.995	.834	.780	.774
*.3000	1.211	1.141	1.049	.933	.889	.880
*.4000	1.189	1.170	1.089	1.007	.968	.965
*.5000	1.203	1.181	1.115	1.061	1.042	1.039
*.6000	1.232	1.221	1.182	1.122	1.116	1.124
*.6900	1.099	1.178	1.137	1.087	1.069	1.116
.7000	.953	1.184	1.158	1.090	1.079	1.182
.7063	.903	1.038	1.062	1.090	1.071	1.548
.7125	.913	.995	1.041	1.090	1.090	1.458
.7250	.648	.734	.803	.965	.973	1.386
.7500	.664	.780	.843	.991	.973	1.386
.8000	.850	.936	.963	1.050	1.042	1.333
.8500	.921	1.009	1.041	1.093	1.100	1.277
.9000	.998	1.068	1.070	1.119	1.116	1.261
.9500	1.027	1.111	1.105	1.135	1.132	1.230
.9750	1.057	1.124	1.132	1.151	1.151	1.219
*.7063	1.537	1.501	1.420	1.167	1.143	1.087
*.7125	2.719	2.803	2.704	2.297	2.264	2.380
*.7250	1.784	1.883	1.851	2.212	2.232	2.303
*.7500	1.545	1.614	1.576	1.543	1.534	1.450
*.8000	1.346	1.383	1.362	1.327	1.299	1.314
*.8500	1.274	1.310	1.308	1.274	1.280	1.277
*.9000	1.216	1.270	1.252	1.250	1.248	1 259
*.9500	1.184	1.227	1.220	1.221	1.238	1.251
*.9750	1.147	1.192	1.198	1.221	1.211	1.238
.9999	.943	1.087	1.148	1.199	1.222	1.238

TABLE XXIII.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.805b/2; $\delta_f = 35^\circ$; $\delta_a = 6^\circ$

Pressure Coefficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°
.0000	.000	.060	.784	1.499	1.988	1.740
.0125	.847	1.302	2.373	2.569	2.437	2.035
.0250	.899	1.247	2.230	2.553	2.437	2.046
.0500	.932	1.187	1.874	2.434	2.437	2.078
.1000	1.104	1.318	1.633	2.585	2.458	1.985
.2000	1.184	1.318	1.491	2.184	2.484	1.977
.3000	1.208	1.324	1.449	1.612	2.373	2.004
.4000	1.238	1.335	1.441	1.488	2.017	2.001
.5000	1.274	1.357	1.441	1.474	1.703	1.975
.6000	1.271	1.362	1.411	1.463	1.529	1.919
.6900	1.217	1.272	1.343	1.360	1.418	1.885
*.0125	1.184	.960	.488	.263	.121	.145
*.0250	.940	.913	.718	.493	.319	.267
*.0500	1.104	1.004	.748	.547	.399	.404
*.1000	1.151	1.039	.822	.659	.560	.554
*.2000	1.123	1.039	.904	.786	.715	.718
*.3000	1.091	1.045	.923	.856	.792	.821
*.4000	1.052	1.039	.956	.894	.840	.876
*.5000	1.022	1.012	.964	.905	.853	.908
*.6000	.970	.974	.945	.911	.866	.935
*.6900	.932	.944	.915	.905	.879	1.001
.7000	1.288	1.182	1.203	1.287	1.428	2.194
.7063	1.178	1.121	1.159	1.238	1.307	1.761
.7125	1.926	2.046	2.055	2.008	1.851	1.837
.7250	1.521	1.625	1.674	1.683	1.624	1.766
.7500	1.397	1.474	1.537	1.556	1.539	1.740
.8000	1.249	1.346	1.411	1.442	1.444	1.682
.8500	1.175	1.269	1.356	1.363	1.370	1.597
.9000	1.178	1.223	1.293	1.322	1.336	1.589
.9500	1.129	1.168	1.236	1.274	1.278	1.544
.9750	1.110	1.171	1.206	1.252	1.243	1.513
*.7063	.937	.930	.940	.921	.921	1.146
*.7125	.564	.853	.915	.894	.797	.314
*.7250	.663	.711	.718	.596	.544	.618
*.7500	.745	.738	.729	.718	.723	.789
*.8000	.830	.859	.866	.864	.837	.927
*.8500	.893	.924	.926	.924	.908	1.016
*.9000	.934	.976	.992	.986	.969	1.090
*.9500	.967	1.034	1.069	1.054	1.016	1.191
*.9750	1.014	1.067	1.071	1.084	1.096	1.280
.9999	.806	.949	1.069	1.144	1.185	1.360

TABLE XXIV.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.805b/2; $\delta_f = 35^\circ$; $\delta_a \doteq 12^\circ$

Pressure Coefficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°
.0000	.000	.150	.948	1.658	2.019	1.724
.0125	.926	1.525	2.506	2.577	2.480	2.024
.0250	.981	1.383	2.416	2.574	2.488	2.019
.0500	1.014	1.289	2.135	2.529	2.504	2.065
.1000	1.151	1.391	1.731	2.622	2.488	1.986
.2000	1.217	1.364	1.497	2.318	2.510	1.986
.3000	1.249	1.369	1.471	1.708	2.423	2.014
.4000	1.307	1.385	1.466	1.502	2.169	2.022
.5000	1.337	1.407	1.474	1.497	1.878	2.005
.6000	1.354	1.434	1.471	1.494	1.698	1.967
.6900	1.299	1.240	1.304	1.336	1.523	1.970
*.0125	1.189	.832	.393	.195	.117	.117
*.0250	.964	.886	.624	.428	.313	.266
*.0500	1.107	.937	.666	.507	.411	.401
*.1000	1.154	.961	.775	.628	.575	.556
*.2000	1.115	.993	.871	.771	.728	.737
*.3000	1.077	.996	.903	.837	.809	.816
*.4000	1.038	.999	.913	.855	.858	.864
*.5000	.992	.945	.897	.866	.864	.897
*.6000	.910	.905	.863	.845	.864	.908
*.6900	.880	.806	.839	.787	.853	.973
.7000	1.447	1.198	1.235	1.296	1.717	2.680
.7063	1.447	1.573	1.646	1.671	1.747	2.062
.7125	2.756	2.841	2.719	2.606	2.297	1.883
.7250	1.965	2.030	2.007	1.993	1.905	1.837
.7500	1.677	1.737	1.747	1.745	1.730	1.808
.8000	1.425	1.498	1.521	1.552	1.616	1.780
.8500	1.293	1.377	1.404	1.441	1.512	1.748
.9000	1.238	1.302	1.338	1.368	1.447	1.715
.9500	1.175	1.230	1.256	1.304	1.376	1.683
.9750	1.134	1.187	1.221	1.262	1.338	1.683
*.7063	.882	.848	.879	.850	.924	1.203
*.7125	.740	.795	.852	.824	.793	.274
*.7250	.471	.470	.547	.729	.324	.336
*.7500	.559	.553	.515	.457	.531	.626
*.8000	.718	.736	.720	.723	.747	.818
*.8500	.822	.848	.836	.818	.845	.935
*.9000	.880	.905	.913	.903	.935	1.068
*.9500	.951	.991	.998	.985	1.044	1.211
*.9750	.997	1.042	1.046	1.059	1.117	1.341
.9999	.858	1.010	1.112	1.159	1.245	1.480

TABLE XXV.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.805b/2; $\delta_f = 0^\circ$; $\delta_a = 12^\circ$

Pressure Coefficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°	α=12°	α=15°
.0000	1.335	.367	.206	.898	1.257	1.428	1.546	1.747
.0125	.389	.810	1.517	2.376	2.228	2.030	1.824	1.809
.0250	.524	.884	1.383	2.310	2.233	2.020	1.824	1.809
.0500	.641	.927	1.287	2.059	2.186	2.049	1.819	1.804
.1000	.863	1.080	1.351	1.753	2.202	2.009	1.811	1.809
.2000	.991	1.135	1.305	1.410	2.146	2.014	1.814	1.814
.3000	1.052	1.167	1.308	1.370	1.824	2.003	1.824	1.831
.4000	1.113	1.180	1.308	1.357	1.552	1.960	1.848	1.841
.5000	1.157	1.212	1.313	1.352	1.412	1.852	1.848	1.858
.6000	1.188	1.233	1.311	1.341	1.365	1.733	1.848	1.879
.6900	1.146	1.172	1.142	1.185	1.257	1.647	1.883	1.887
*.0125	2.345	1.510	.856	.388	.232	.157	.102	.092
*.0250	1.496	1.354	1.046	.678	.491	.389	.273	.227
*.0500	1.515	1.291	.976	.686	.533	.443	.393	.340
*.1000	1.432	1.222	1.006	.792	.660	.610	.572	.508
*.2000	1.326	1.183	1.067	.903	.803	.780	.736	.697
*.3000	1.260	1.156	1.067	.942	.866	.842	.816	.815
*.4000	1.207	1.125	1.067	.969	.906	.907	.893	.894
*.5000	1.124	1.077	1.027	.945	.908	.929	.936	.929
*.6000	1.041	.995	.974	.900	.876	.910	.931	.950
*.6900	.910	.882	.848	.837	.816	.913	.968	.999
.7000	1.160	1.262	1.078	1.096	1.236	2.030	2.664	2.757
.7063	1.246	1.338	1.447	1.463	1.436	1.696	1.953	2.047
.7125	2.609	2.669	2.678	2.474	2.059	1.755	1.736	1.814
.7250	1.759	1.861	1.894	1.785	1.621	1.634	1.723	1.806
.7500	1.490	1.581	1.586	1.555	1.473	1.574	1.741	1.809
.8000	1.274	1.336	1.370	1.365	1.352	1.507	1.731	1.833
.8500	1.157	1.217	1.279	1.272	1.270	1.469	1.736	1.841
.9000	1.124	1.154	1.182	1.188	1.199	1.426	1.728	1.841
.9500	1.063	1.096	1.124	1.114	1.151	1.391	1.691	1.820
.9750	1.041	1.067	1.097	1.090	1.130	1.388	1.675	1.793
*.7063	.913	.916	.853	.876	.882	1.053	1.231	1.274
*.7125	.869	.853	.829	.876	.863	.648	.233	.213
*.7250	.669	.594	.551	.541	.705	.362	.340	.362
*.7500	.708	.647	.589	.528	.462	.545	.621	.643
*.8000	.785	.784	.770	.726	.713	.767	.819	.859
*.8500	.866	.858	.856	.818	.810	.875	.923	1.002
*.9000	.907	.908	.910	.876	.882	.967	1.062	1.134
*.9500	.935	.942	.960	.940	.961	1.080	1.204	1.288
*.9750	.963	.977	.984	.972	.993	1.139	1.297	1.420
.9999	.788	.937	1.019	1.064	1.093	1.250	1.439	1.628

TABLE XXVI.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.805b/2; $\delta_f = 0^\circ$; $\delta_a = 6^\circ$

Pressure Coefficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°	α=12°	α=15°
.0000	1.480	.356	.153	.904	1.238	1.389	1.537	1.680
.0125	.353	.783	1.331	2.434	2.290	1.954	1.875	1.798
.0250	.499	.838	1.251	2.362	2.290	1.957	1.875	1.790
.0500	.625	.912	1.199	1.956	2.183	1.951	1.883	1.803
.1000	.855	1.045	1.280	1.633	2.199	1.941	1.856	1.779
.2000	.992	1.095	1.251	1.434	2.108	1.957	1.856	1.800
.3000	1.041	1.127	1.233	1.383	1.756	1.951	1.883	1.811
.4000	1.077	1.145	1.220	1.348	1.493	1.909	1.900	1.822
.5000	1.123	1.176	1.233	1.345	1.377	1.828	1.894	1.846
.6000	1.143	1.163	1.225	1.305	1.329	1.702	1.889	1.854
.6900	1.104	1.119	1.159	1.235	1.262	1.596	1.913	1.886
*.0125	2.504	1.520	.964	.455	.260	.174	.138	.072
*.0250	1.639	1.420	1.140	.772	.545	.384	.295	.238
*.0500	1.589	1.313	1.045	.756	.580	.450	.404	.353
*.1000	1.485	1.237	1.045	.842	.687	.621	.566	.527
*.2000	1.354	1.205	1.072	.950	.857	.781	.756	.709
*.3000	1.302	1.176	1.080	.998	.913	.868	.870	.837
*.4000	1.244	1.148	1.093	1.036	.959	.926	.946	.910
*.5000	1.197	1.103	1.051	1.030	.975	.957	.976	.966
*.6000	1.112	1.048	1.024	1.003	.967	.963	1.008	1.014
*.6900	1.019	.990	.966	.947	.932	.981	1.070	1.086
.7000	1.060	1.127	1.072	1.127	1.181	1.686	2.209	2.247
.7063	1.047	1.121	1.032	1.089	1.157	1.544	1.840	1.856
.7125	1.797	1.871	1.898	1.870	1.683	1.581	1.813	1.824
.7250	1.403	1.472	1.500	1.525	1.447	1.536	1.797	1.827
.7500	1.269	1.320	1.365	1.391	1.361	1.512	1.791	1.838
.8000	1.145	1.192	1.246	1.283	1.286	1.452	1.753	1.832
.8500	1.091	1.129	1.170	1.216	1.230	1.404	1.721	1.819
.9000	1.091	1.087	1.119	1.170	1.179	1.365	1.694	1.811
.9500	1.052	1.058	1.088	1.116	1.144	1.328	1.656	1.779
.9750	1.060	1.043	1.067	1.106	1.128	1.304	1.612	1.752
*.7063	1.006	.980	.977	.974	.956	1.031	1.217	1.228
*.7125	.901	.823	.966	.974	.937	.634	.301	.270
*.7250	.841	.776	.803	.812	.666	.605	.686	.714
*.7500	.882	.836	.800	.759	.744	.789	.854	.877
*.8000	.932	.917	.898	.890	.883	.899	.986	1.019
*.8500	.959	.954	.940	.939	.942	.968	1.079	1.110
*.9000	.984	.975	.961	.971	.975	1.036	1.138	1.190
*.9500	.986	.990	.982	1.020	1.010	1.073	1.228	1.319
*.9750	.986	1.011	.998	1.036	1.034	1.134	1.320	1.436
.9999	.797	.948	1.008	1.079	1.109	1.199	1.412	1.525

TABLE XXVII.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

STATION 0.805b/2; $\delta_f = 0^\circ$; $\delta_a = -6^\circ$

Pressure Coefficient, S_p

x/c	$\alpha=-6^\circ$	$\alpha=-3^\circ$	$\alpha=0^\circ$	$\alpha=3^\circ$	$\alpha=6^\circ$	$\alpha=9^\circ$	$\alpha=12^\circ$	$\alpha=15^\circ$
.0000	1.490	.518	.151	.843	1.141	1.359	1.492	1.708
.0125	.339	.689	1.208	2.404	2.214	1.968	1.868	1.800
.0250	.459	.788	1.159	2.085	2.170	1.970	1.860	1.795
.0500	.609	.872	1.128	1.597	2.067	1.978	1.863	1.800
.1000	.793	.997	1.206	1.451	2.167	1.973	1.860	1.795
.2000	.905	1.039	1.182	1.331	1.915	2.011	1.866	1.805
.3000	.964	1.067	1.169	1.282	1.483	1.986	1.876	1.822
.4000	.988	1.062	1.148	1.219	1.299	1.904	1.876	1.841
.5000	1.004	1.049	1.128	1.182	1.220	1.749	1.879	1.843
.6000	.983	1.015	1.067	1.114	1.136	1.583	1.831	1.852
.6900	.951	.976	1.028	1.054	1.076	1.434	1.818	1.944
*.0125	2.539	1.657	1.070	.478	.268	.168	.125	.079
*.0250	2.125	1.657	1.282	.799	.537	.379	.301	.236
*.0500	2.051	1.433	1.112	.762	.576	.481	.410	.345
*.1000	1.551	1.302	1.091	.851	.715	.646	.586	.529
*.2000	1.372	1.263	1.135	.963	.863	.828	.773	.725
*.3000	1.340	1.248	1.151	1.021	.939	.921	.882	.872
*.4000	1.322	1.240	1.175	1.073	1.013	1.001	.981	.986
*.5000	1.292	1.281	1.164	1.099	1.055	1.060	1.042	1.062
*.6000	1.255	1.216	1.169	1.120	1.070	1.097	1.111	1.148
*.6900	1.191	1.133	1.133	1.078	1.047	1.095	1.149	1.197
.7000	1.113	1.070	1.117	1.078	1.036	1.140	1.218	1.311
.7063	.929	.976	1.013	1.041	1.078	1.391	1.634	1.778
.7125	.886	.968	.840	.885	.989	1.327	1.618	1.778
.7250	.700	.887	.835	.890	.986	1.306	1.615	1.808
.7500	.774	.788	.864	.942	1.020	1.300	1.620	1.822
.8000	.870	.916	.950	.984	1.036	1.258	1.556	1.805
.8500	.902	.955	.976	1.021	1.041	1.204	1.482	1.773
.9000	.943	.989	.992	1.015	1.049	1.178	1.428	1.743
.9500	.975	.994	1.007	1.013	1.041	1.151	1.375	1.708
.9750	.993	1.013	1.034	1.026	1.060	1.129	1.333	1.662
*.7063	1.057	1.023	1.114	1.081	1.034	.756	.528	.475
*.7125	1.717	1.762	1.743	1.631	1.631	1.781	1.903	2.058
*.7250	1.410	1.425	1.456	1.527	1.518	1.535	1.522	1.512
*.7500	1.311	1.308	1.302	1.240	1.205	1.247	1.314	1.393
*.8000	1.193	1.211	1.201	1.154	1.112	1.164	1.245	1.338
*.8500	1.159	1.154	1.138	1.109	1.089	1.135	1.205	1.330
*.9000	1.121	1.107	1.091	1.078	1.078	1.129	1.197	1.338
*.9500	1.089	1.075	1.057	1.062	1.060	1.113	1.215	1.415
*.9750	1.063	1.057	1.049	1.044	1.052	1.113	1.229	1.463
.9999	.870	.997	1.036	1.052	1.060	1.105	1.226	1.450

TABLE XXVIII.- PRESSURE DATA OVER A MODIFIED NACA 64A004 AIRFOIL SECTION AT SEVERAL ANGLES OF ATTACK.

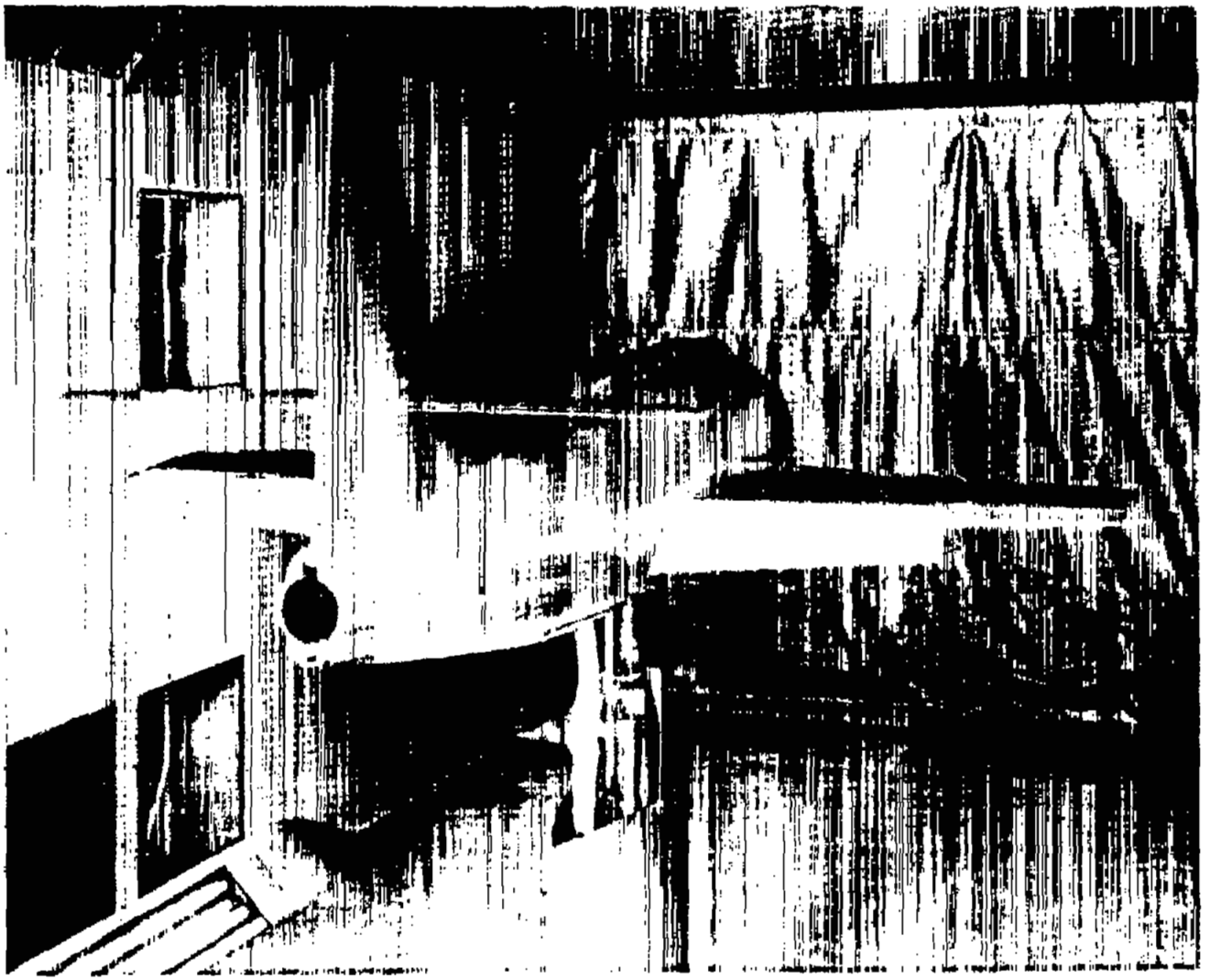
STATION 0.805b/2; $\delta_f = 0^\circ$; $\delta_a = -12^\circ$

Pressure Coefficient, S_p

x/c	α=-6°	α=-3°	α=0°	α=3°	α=6°	α=9°	α=12°	α=15°
.0000	1.428	.700	.173	.685	1.125	1.305	1.496	1.711
.0125	.266	.612	1.139	2.156	2.296	2.003	1.895	1.798
.0250	.422	.708	1.134	1.836	2.236	2.014	1.895	1.798
.0500	.543	.789	1.102	1.484	2.131	2.008	1.895	1.803
.1000	.762	.950	1.183	1.431	2.261	2.008	1.895	1.798
.2000	.882	1.011	1.169	1.309	1.955	2.014	1.906	1.812
.3000	.932	1.035	1.150	1.255	1.504	1.976	1.928	1.822
.4000	.954	1.030	1.129	1.197	1.328	1.853	1.931	1.831
.5000	.954	1.006	1.085	1.141	1.227	1.654	1.917	1.850
.6000	.899	.956	1.007	1.037	1.130	1.439	1.833	1.847
.6900	.838	.853	.923	.962	1.081	1.289	1.814	1.961
*.0125	2.419	1.777	1.177	.600	.327	.196	.124	.073
*.0250	2.066	1.737	1.377	.951	.668	.435	.324	.231
*.0500	2.208	1.547	1.199	.861	.657	.505	.427	.351
*.1000	1.682	1.352	1.156	.909	.767	.677	.608	.530
*.2000	1.414	1.291	1.188	1.005	.941	.865	.799	.753
*.3000	1.370	1.288	1.207	1.087	1.029	.956	.929	.889
*.4000	1.351	1.288	1.229	1.133	1.103	1.055	1.042	1.015
*.5000	1.343	1.288	1.242	1.173	1.147	1.109	1.107	1.099
*.6000	1.337	1.291	1.264	1.213	1.207	1.181	1.193	1.219
*.6900	1.173	1.133	1.188	1.149	1.163	1.138	1.183	1.224
.7000	1.047	1.032	1.220	1.218	1.191	1.181	1.264	1.229
.7063	.866	.874	.956	.978	1.070	1.251	1.593	1.771
.7125	.806	.834	.848	.879	.993	1.187	1.558	1.760
.7250	.658	.568	.583	.698	.850	1.138	1.544	1.782
.7500	.493	.583	.659	.760	.916	1.138	1.544	1.798
.8000	.699	.755	.807	.853	.957	1.112	1.482	1.801
.8500	.775	.840	.886	.919	1.004	1.069	1.391	1.760
.9000	.858	.898	.940	.970	1.048	1.066	1.334	1.724
.9500	.915	.942	.972	.983	1.053	1.058	1.269	1.667
.9750	.954	.972	1.002	1.010	1.059	1.050	1.242	1.618
*.7063	1.611	1.626	1.550	1.439	1.403	1.028	1.139	1.336
*.7125	2.513	2.703	2.808	2.734	2.695	2.459	2.587	2.772
*.7250	1.726	1.798	1.898	1.818	1.922	2.344	2.408	2.551
*.7500	1.521	1.542	1.609	1.543	1.579	1.482	1.523	1.635
*.8000	1.310	1.320	1.355	1.325	1.372	1.326	1.374	1.474
*.8500	1.228	1.222	1.253	1.234	1.268	1.243	1.310	1.409
*.9000	1.145	1.151	1.166	1.159	1.205	1.187	1.258	1.404
*.9500	1.096	1.082	1.123	1.098	1.141	1.138	1.215	1.423
*.9750	1.058	1.056	1.085	1.079	1.119	1.112	1.210	1.444
.9999	.838	.950	1.018	1.061	1.100	1.106	1.177	1.439

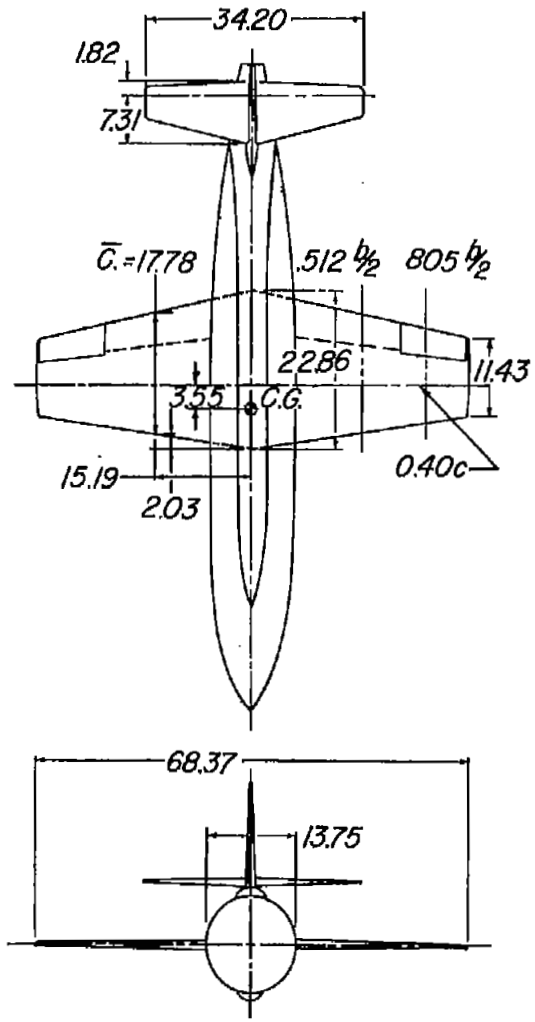
L-78613

Figure 1.- Photographs of 1/4-scale model of Bell X-1 airplane in the Langley 300 MPH 7- by 10-foot tunnel.



L-78616

Figure 1.- Concluded.



Wing	
Area, total	8.125 sq ft
Area, aileron	0.476 sq ft
Area, slotted flap	1.104 sq ft
Span	5.69 ft
Mean aerodynamic chord	1.48 ft
Aspect ratio	4
Airfoil section	Modified NACA 64A004
Horizontal tail	
Area, total	1.625 sq ft
Area, elevator	0.325 sq ft
Airfoil section	NACA 65-008
Vertical tail	
Area, total	1.600 sq ft
Area, rudder	0.325 sq ft
Airfoil section	NACA 65-008

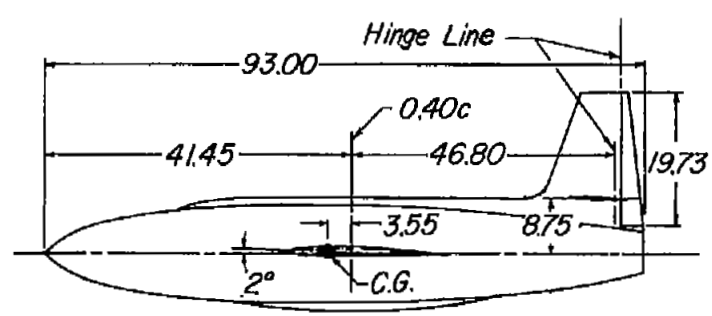
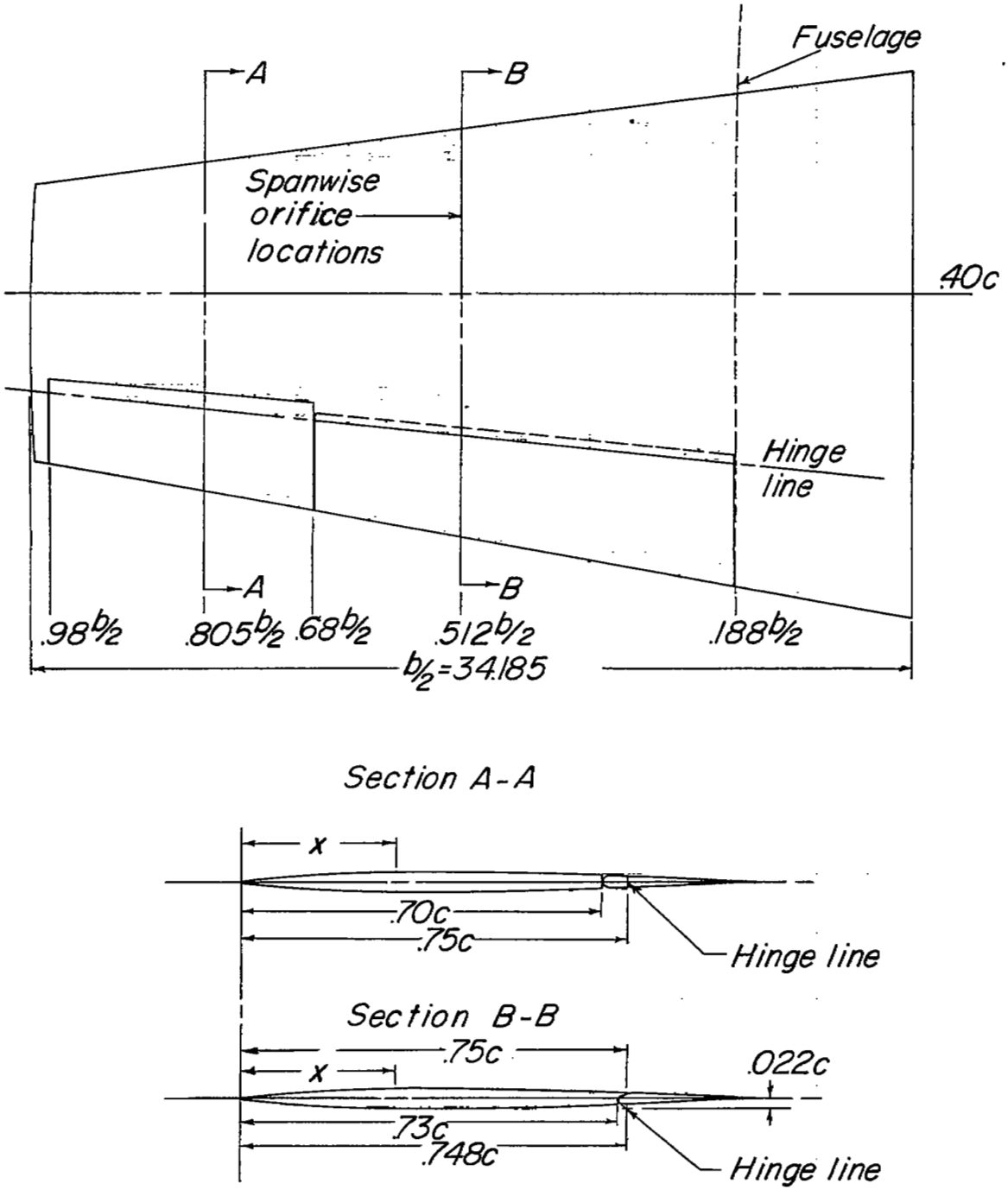
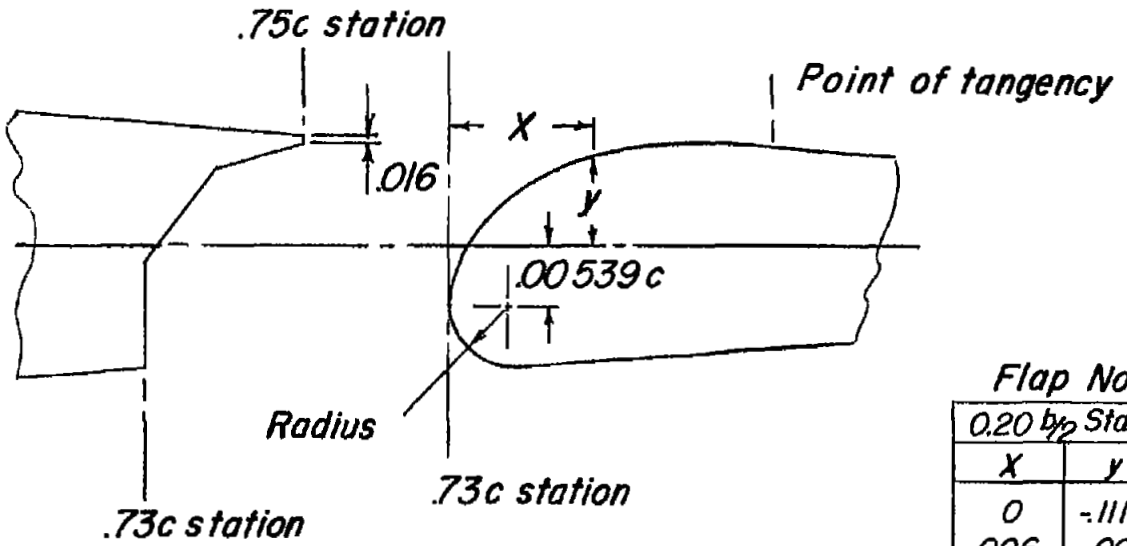


Figure 2.- Three-view drawing of the 1/4-scale model of the Bell X-1 airplane with thin, aspect-ratio-4 wing. (Dimensions are in inches.)



(a) Spanwise location of pressure orifices and general dimensions of the slotted flap and aileron.

Figure 3.- Aileron and slotted flap details.

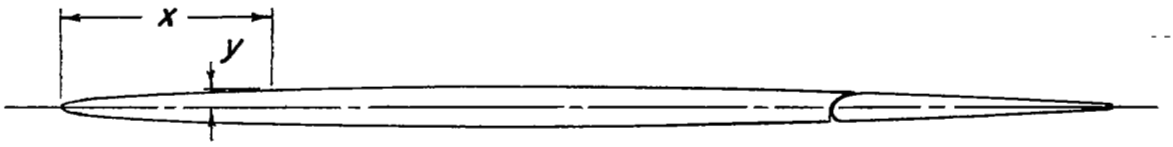


Flap Nose Ordinates

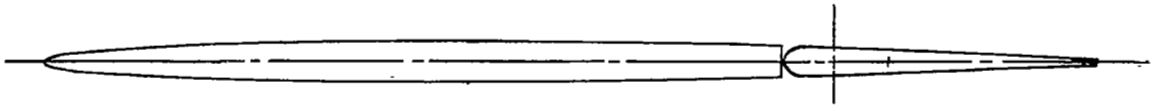
0.20 b/2 Sta.		0.68 b/2 Sta.	
X	y	X	y
0	-.111	0	-.081
.006	-.061	.002	-.056
.026	-.011	.010	-.031
.061	.039	.040	.019
.114	.089	.096	.069
.194	.139	.196	.119
.251	.164	.261	.138
.329	.189	.298	.146
.373	.199	.336	.152
.448	.211	.398	.157
.548	.216	.436	.158
.578	.214	.486	.157
5.563	.031	4.074	.031

(b) Flap nose shape.

Figure 3.- Concluded.



Flap Station (.512 b/2)



Aileron Station (.805 b/2)

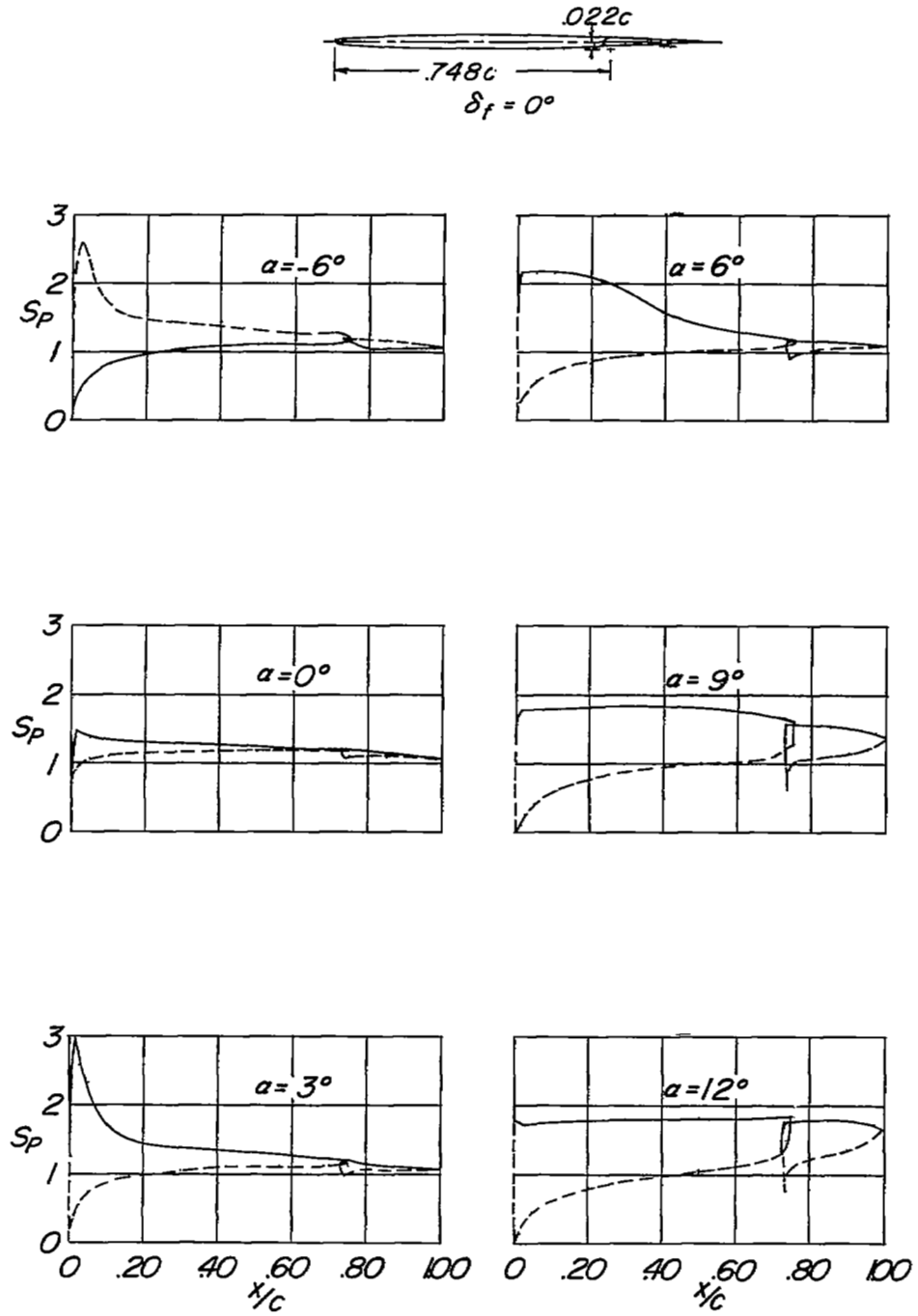
Station .512b/2

x	y
0	0
.0125	±.00493
.0250	±.00678
.0500	±.00932
.1000	±.01287
.2000	±.01702
.3000	±.01929
.4000	±.01999
.5000	±.01889
.6000	±.01634
.7000	±.01282
.7250	±.01190
.7300	-.00540
.7350	+.00394
.7350	-.01125
.7400	+.00705
.7400	-.01136
.7500	+.00982
.7500	-.01099
.8000	±.00916
.8500	±.00733
.9000	±.00550
.9500	±.00367
.9750	±.00275
1.0000	0

Station .805b/2

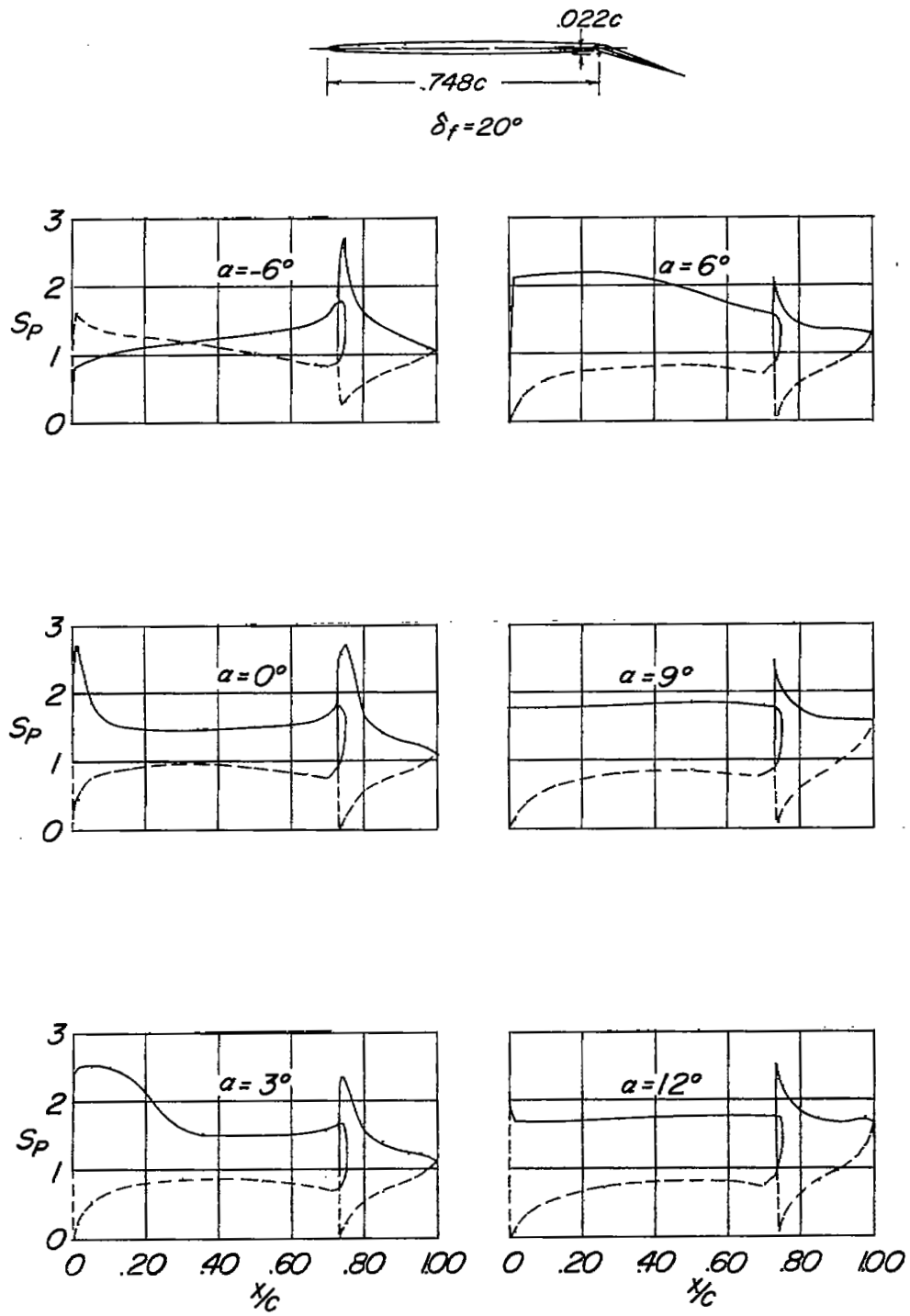
x	y
0	0
.0125	±.00493
.0250	±.00678
.0500	±.00932
.1000	±.01287
.2000	±.01702
.3000	±.01929
.4000	±.01999
.5000	±.01889
.6000	±.01634
.6900	±.01320
.7000	0
.7063	±.00880
.7125	±.01241
.7250	±.01197
.7500	±.01109
.8000	±.00933
.8500	±.00757
.9000	±.00581
.9500	±.00405
.9750	±.00317
1.0000	0

Figure 4.- Wing orifice locations.



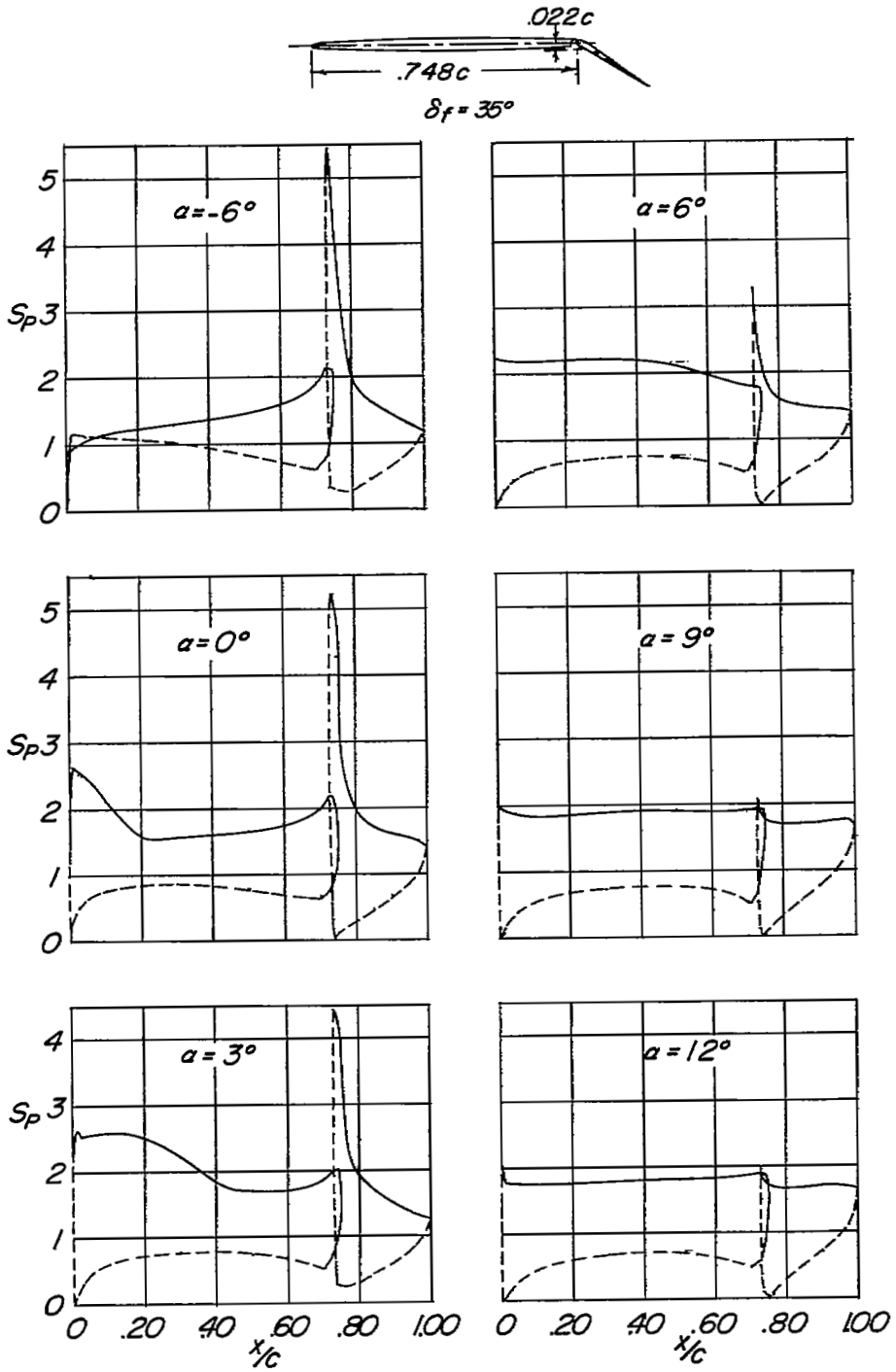
(a) $\delta_f = 0^\circ$; $\delta_a = 0^\circ$.

Figure 5.- Typical pressure distributions over the NACA 64A004, aspect-ratio-4 wing mounted on a 1/4-scale model of the Bell X-1 airplane at several angles of attack and flap deflections. Station 0.512b/2.



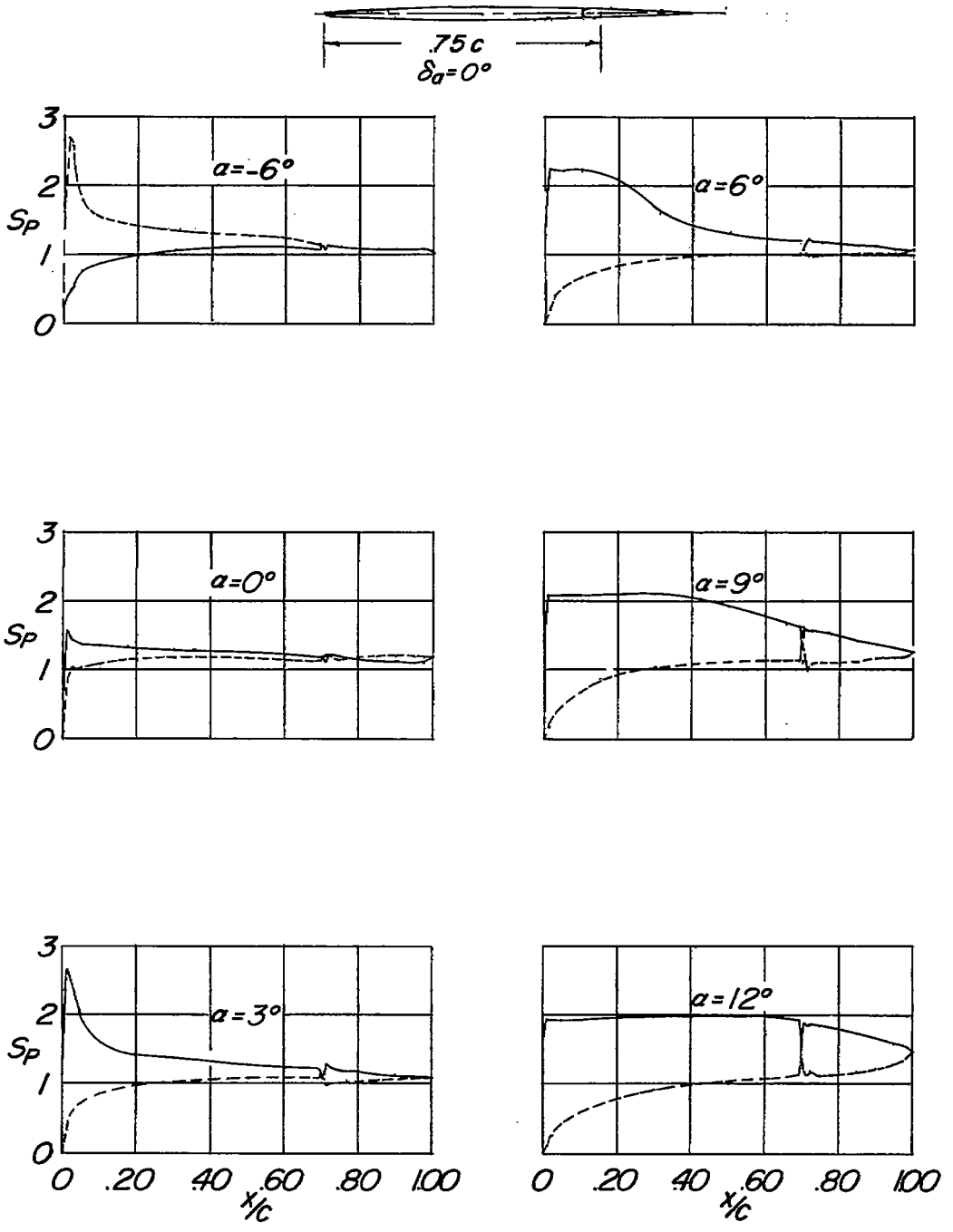
(b) $\delta_f = 20°$; $\delta_a = 0°$.

Figure 5.- Continued.



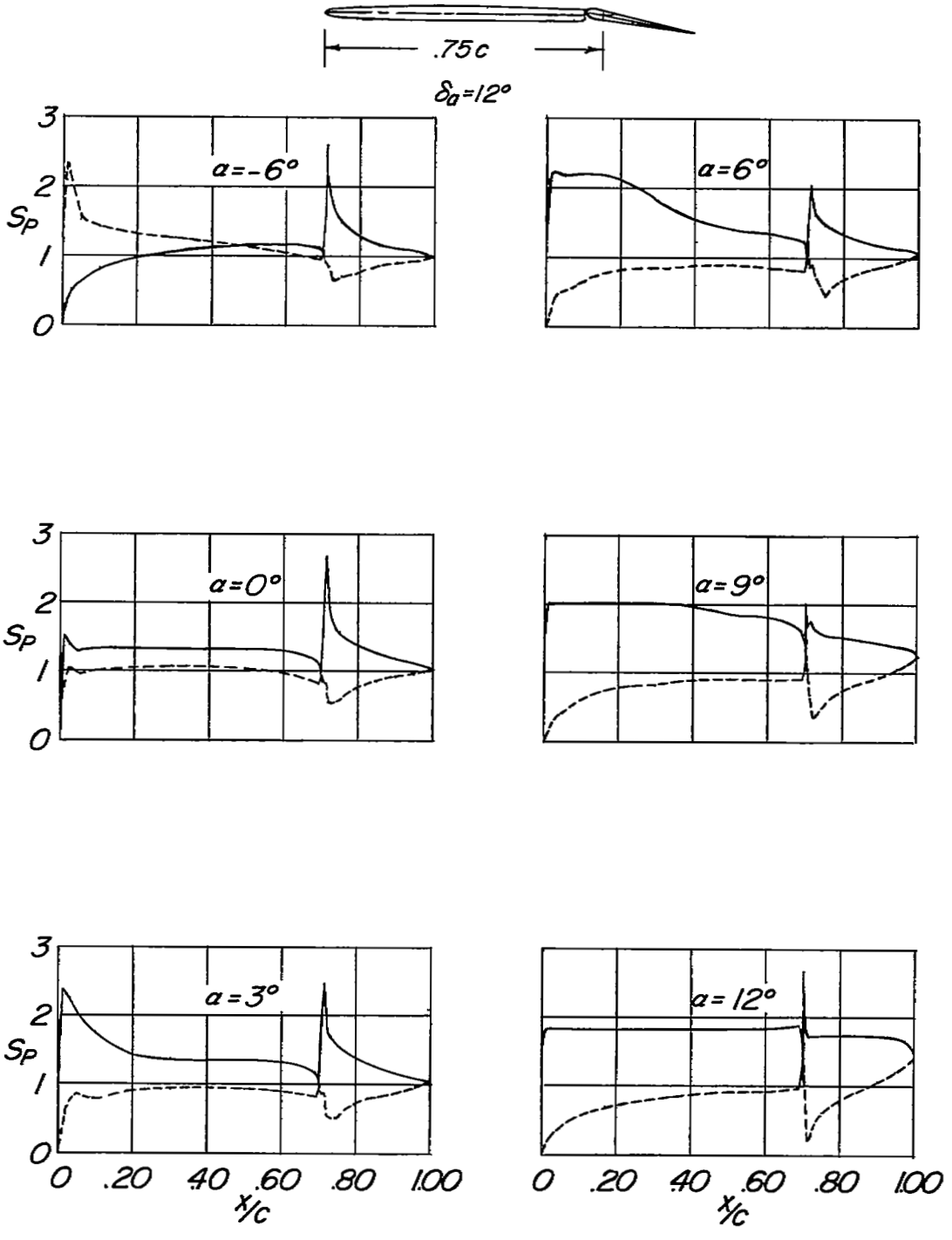
(c) $\delta_f = 35°$; $\delta_a = 0°$.

Figure 5.- Concluded.



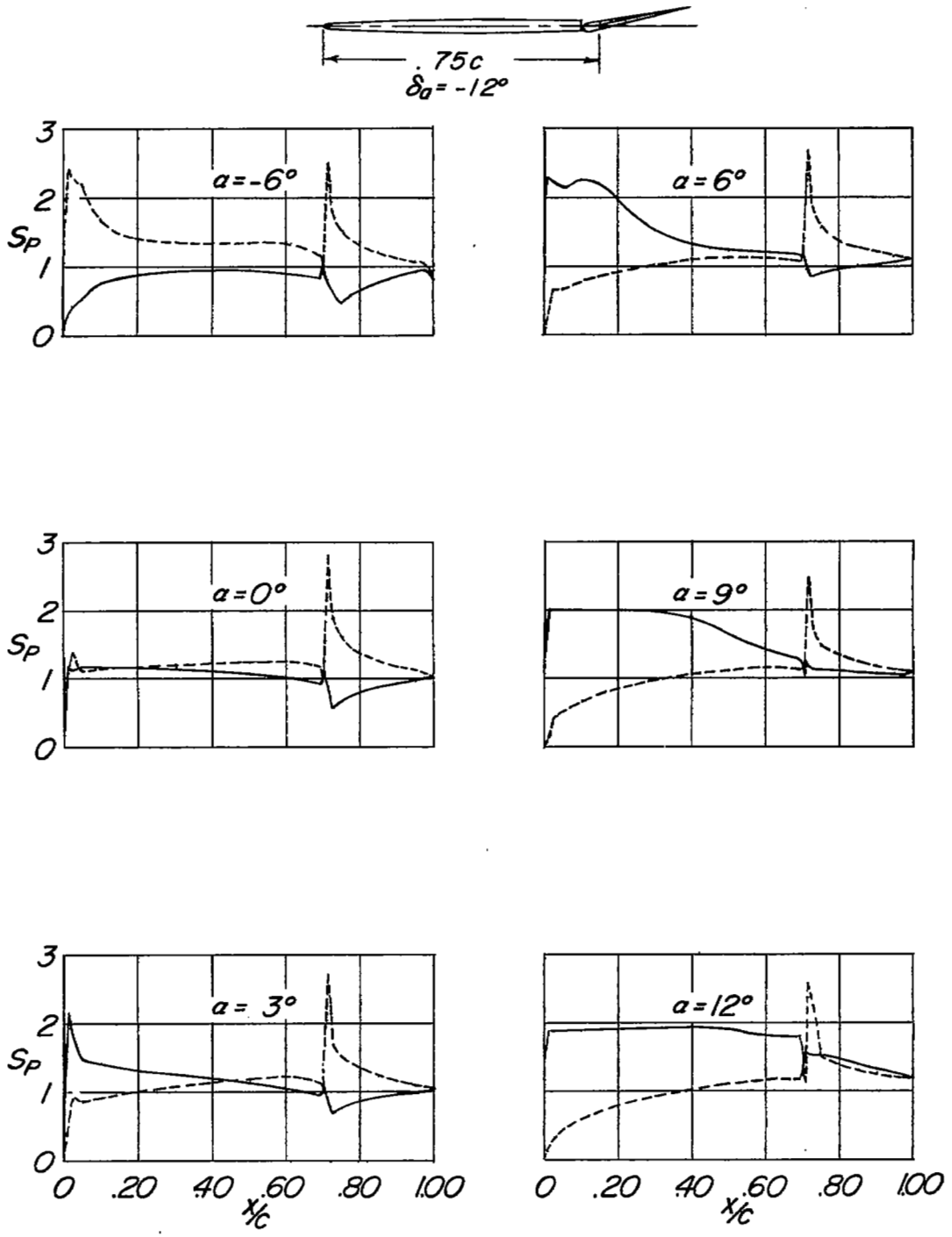
(a) $\delta_f = 0°$; $\delta_a = 0°$.

Figure 6.- Typical pressure distributions over the NACA 64A004, aspect-ratio-4 wing mounted on a 1/4-scale model of the Bell X-1 airplane at several angles of attack and aileron deflections. Station 0.805b/2.



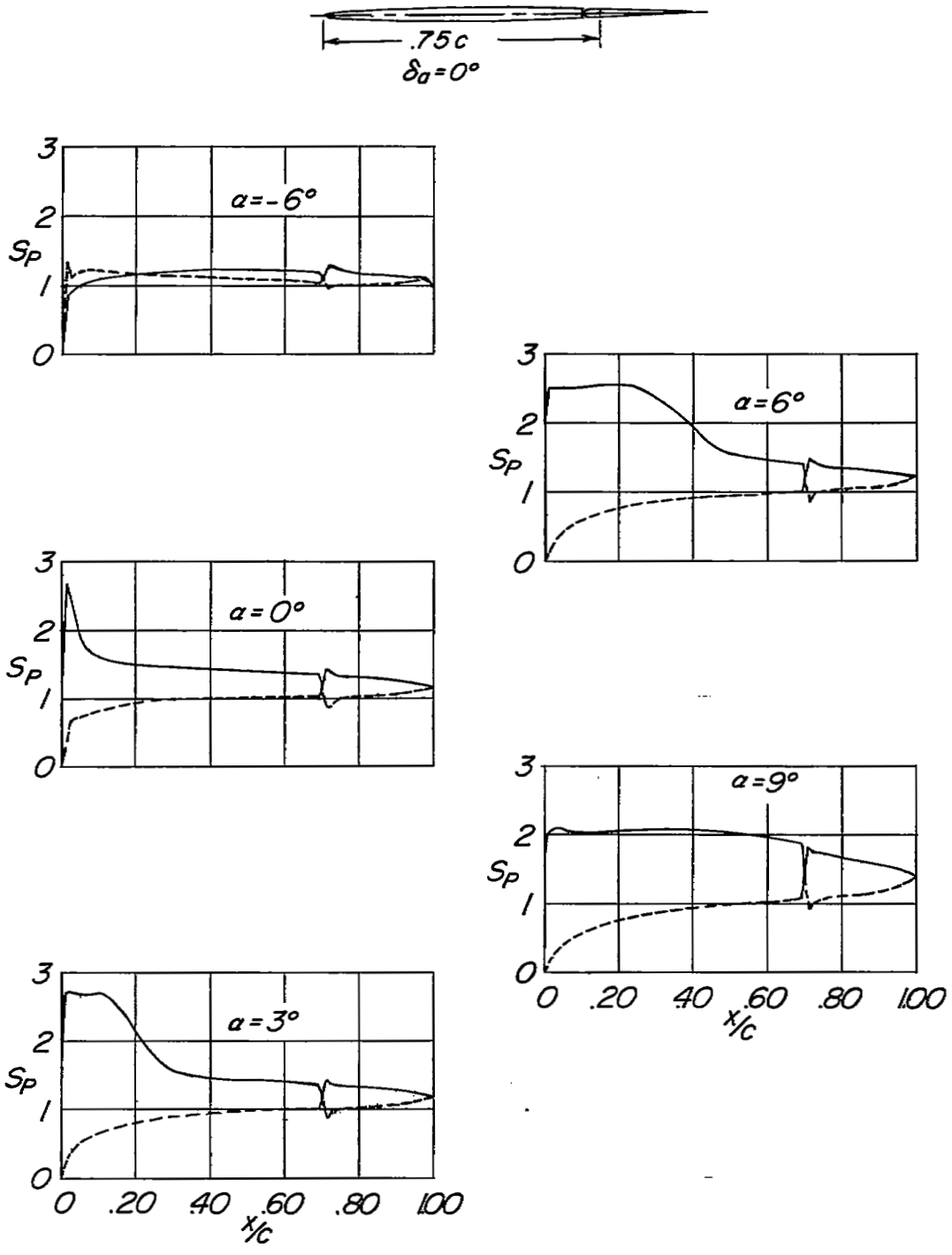
(b) $\delta_f = 0^\circ$; $\delta_a = +12^\circ$.

Figure 6.- Continued.



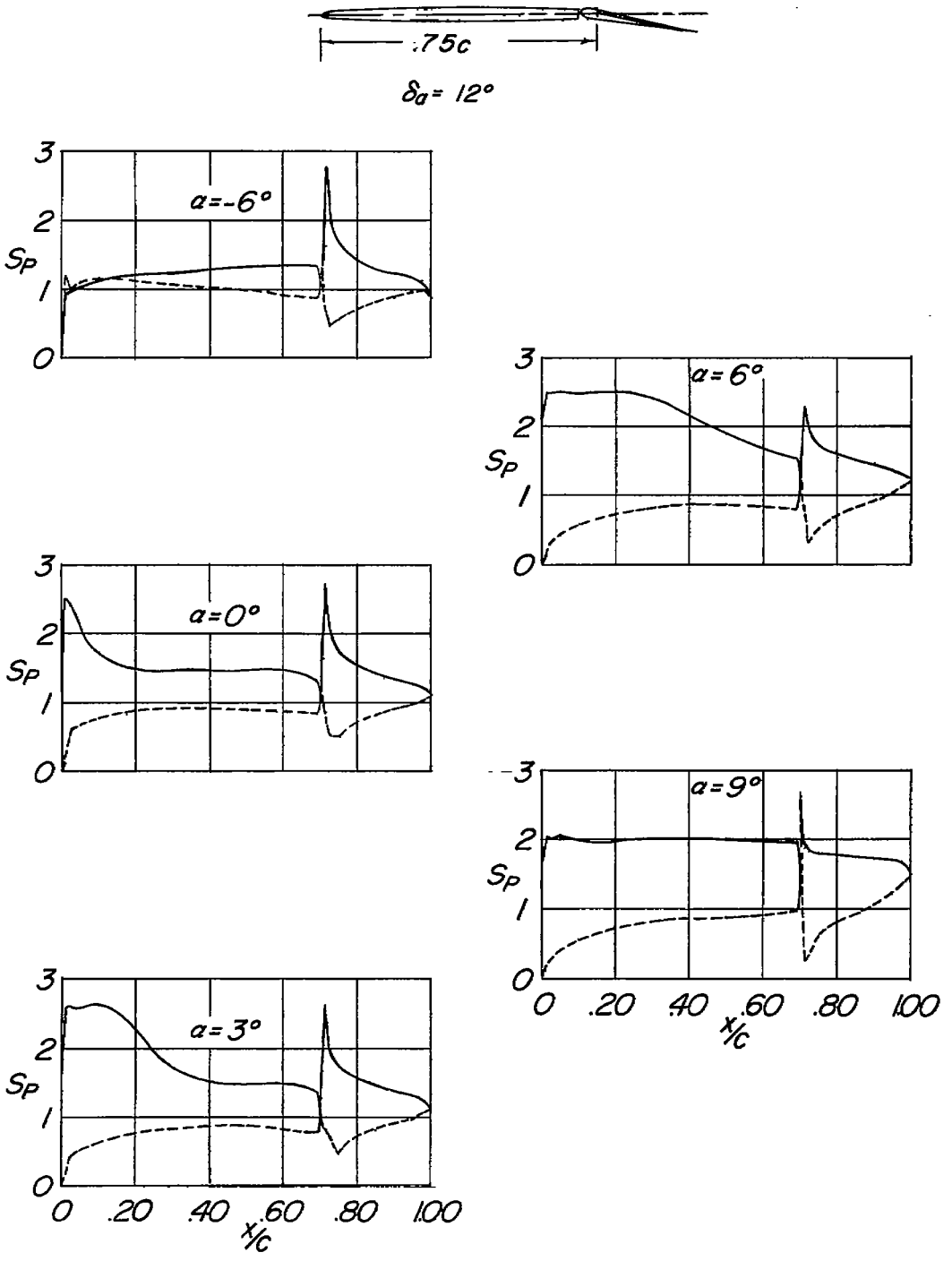
(c) $\delta_f = 0^\circ$; $\delta_a = -12^\circ$.

Figure 6.- Concluded.



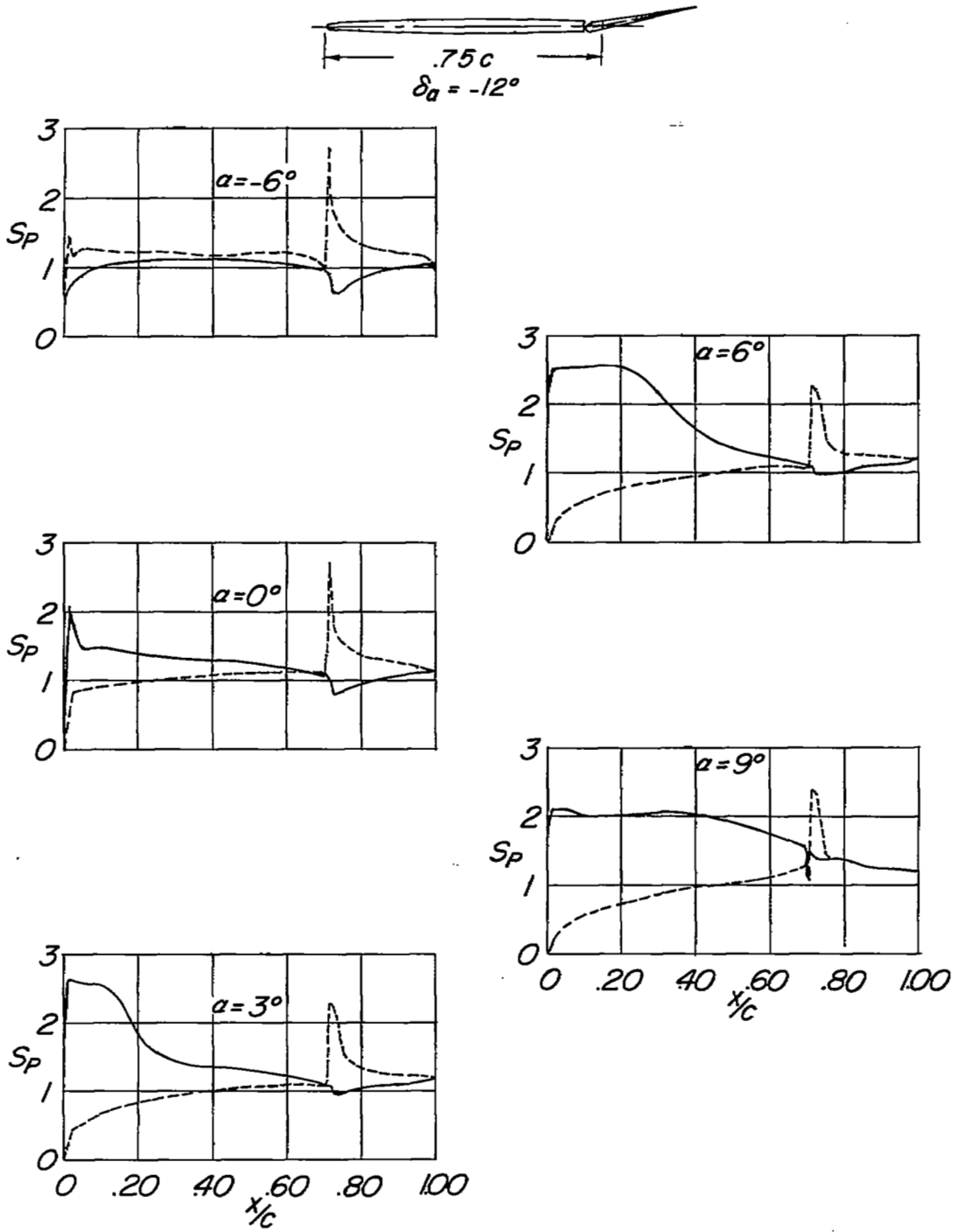
(a) $\delta_f = 35°$; $\delta_a = 0°$.

Figure 7.- Typical pressure distributions over the NACA 64A004, aspect-ratio-4 wing mounted on a 1/4-scale model of the Bell X-1 airplane at several angles of attack and flap and aileron deflections. Station 0.805b/2. [redacted]



(b) $\delta_f = 35^\circ$; $\delta_a = +12^\circ$.

Figure 7.- Continued.



(c) $\delta_f = 35°$; $\delta_a = -12°$.

Figure 7.- Concluded.

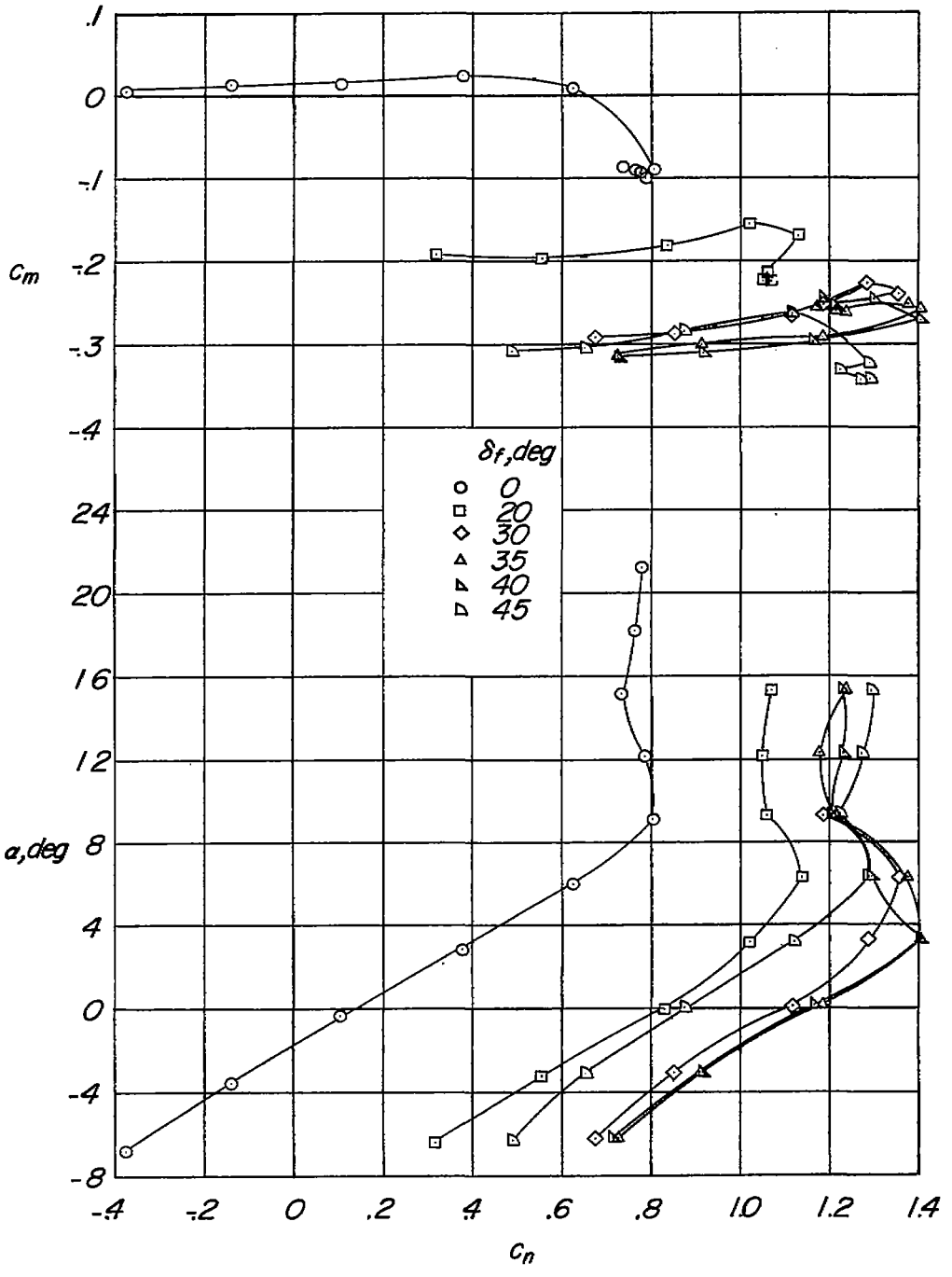


Figure 8.- Effect of flap deflection on wing section normal-force and section pitching-moment coefficients and flap section normal-force and flap section hinge-moment coefficients as determined from pressure-coefficient data on a 4-percent-thick, aspect-ratio-4 wing of a 1/4-scale model of the Bell X-1 airplane. Station 0.512b/2; $\delta_a = 0^\circ$.

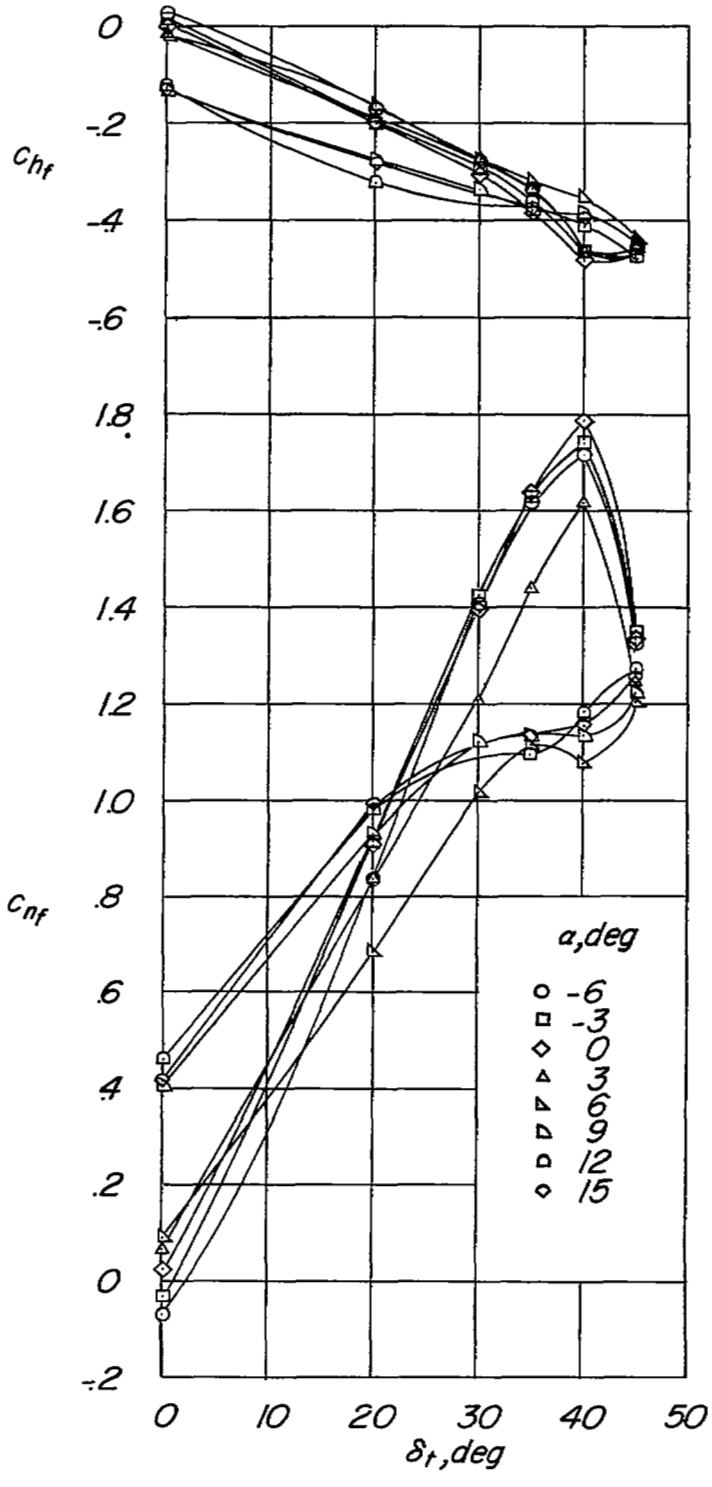


Figure 8.- Concluded.

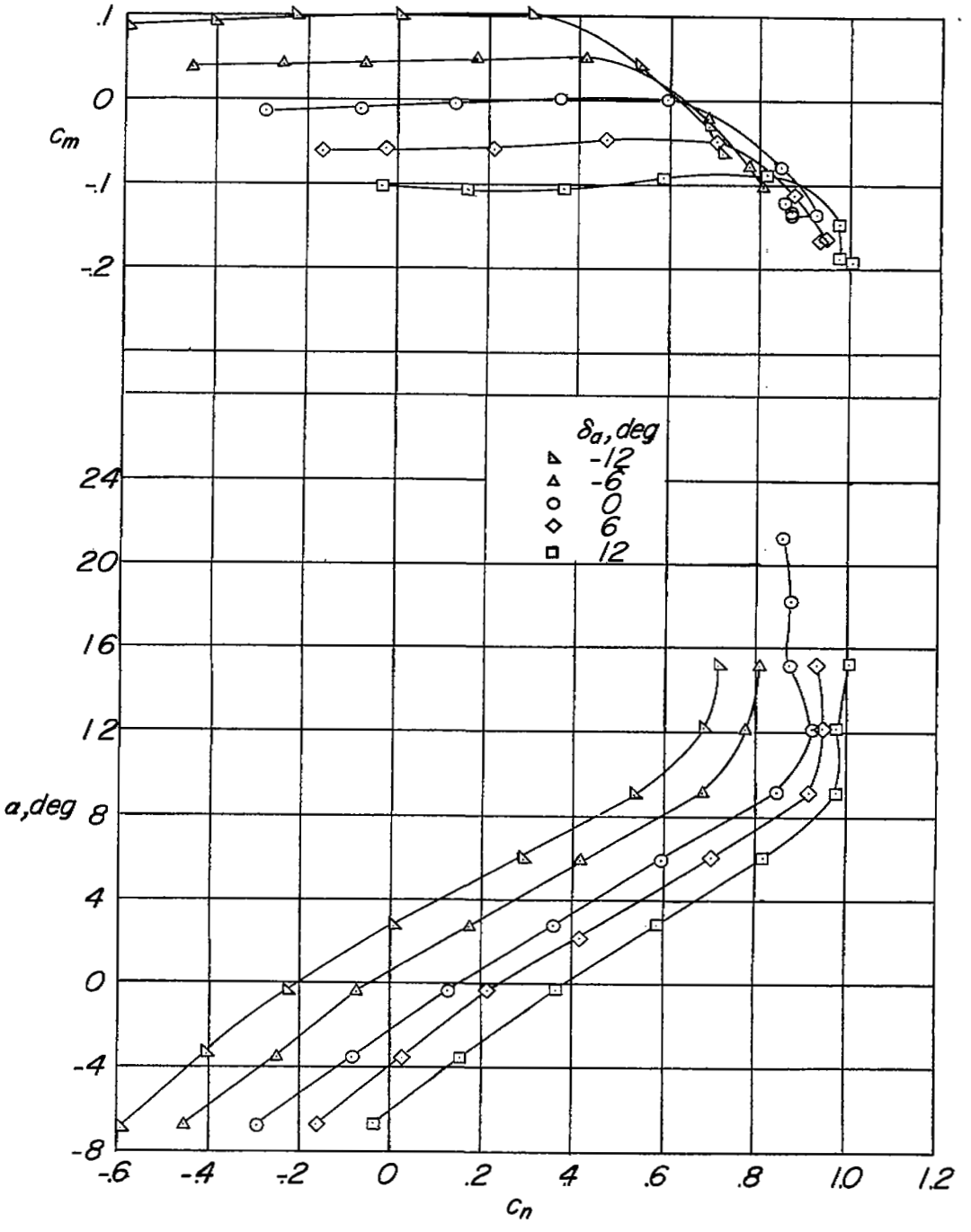


Figure 9.- Effect of aileron deflection on wing section normal-force and section pitching-moment coefficients and flap section normal-force and flap section hinge-moment coefficients as determined from pressure-coefficient data on a 4-percent-thick, aspect-ratio-4 wing of a 1/4-scale model of the Bell X-1 airplane. Station 0.805b/2; $\delta_f = 0^\circ$.

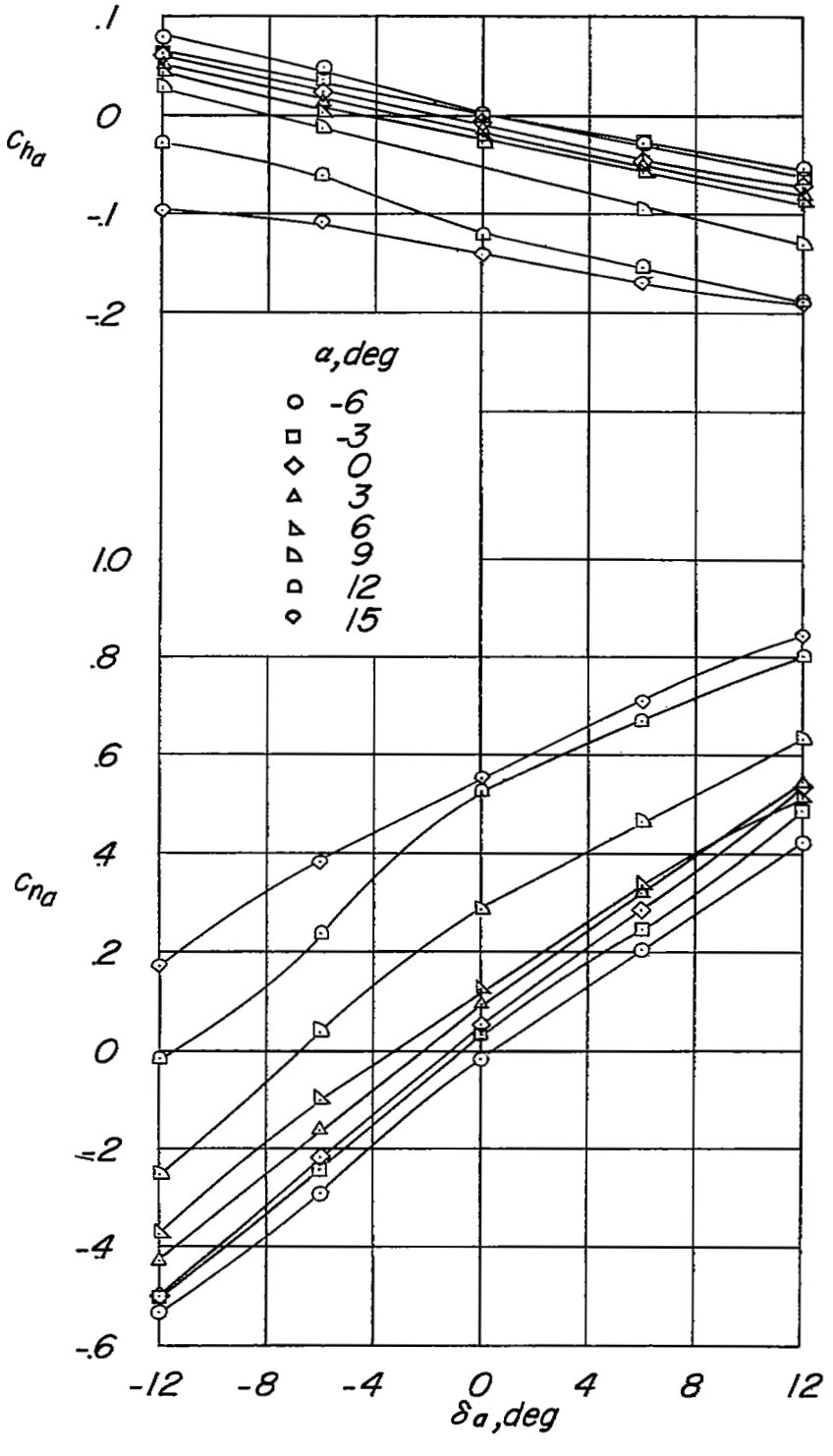


Figure 9.- Concluded.

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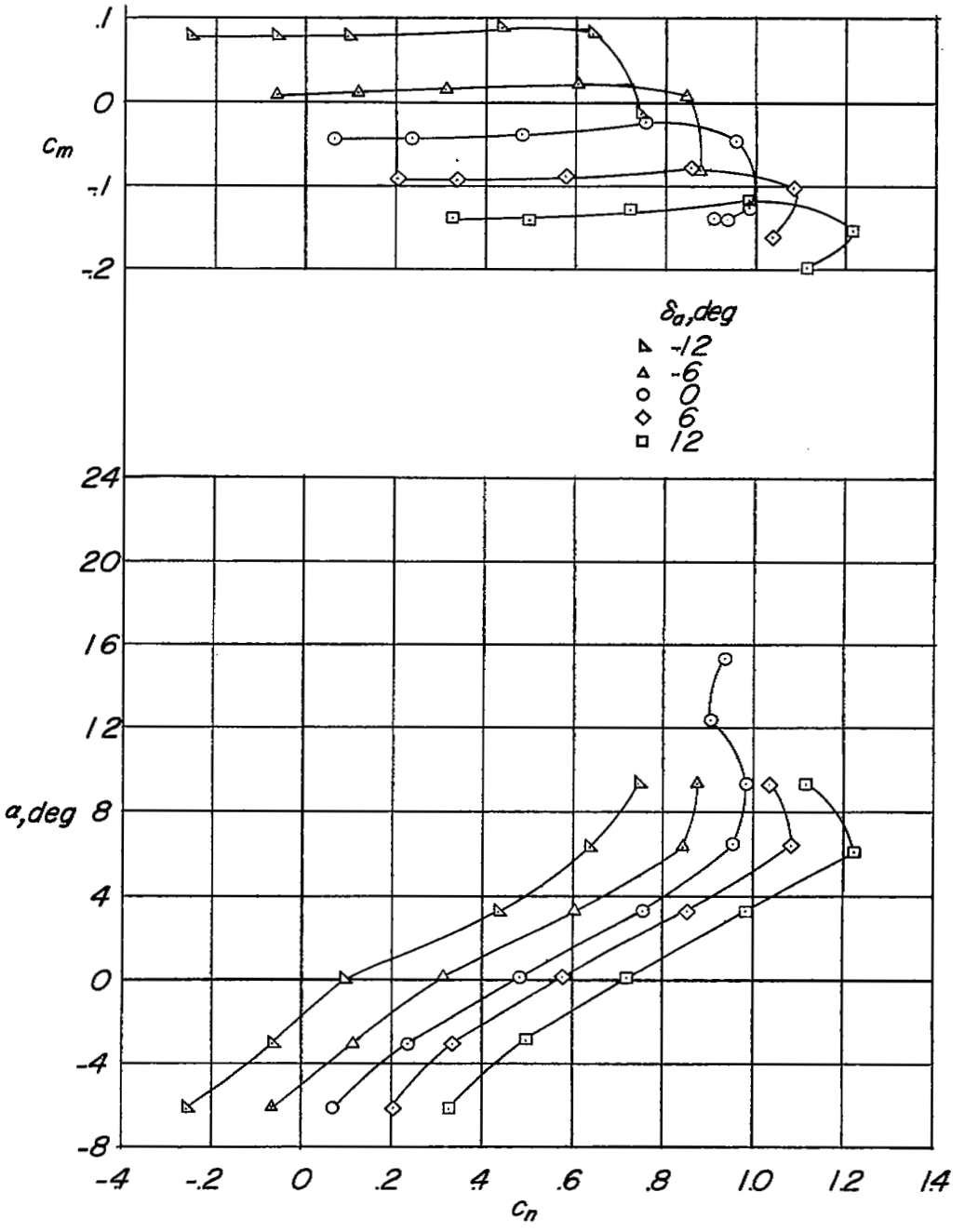

Figure 10.- Effect of aileron deflection on wing section normal-force and section pitching-moment coefficients and flap section normal-force and flap section hinge-moment coefficients as determined from pressure-coefficient data on a 4-percent-thick, aspect-ratio-4 wing of a 1/4-scale model of the Bell X-1 airplane. Station 0.805b/2; $\delta_f = 35^\circ$.

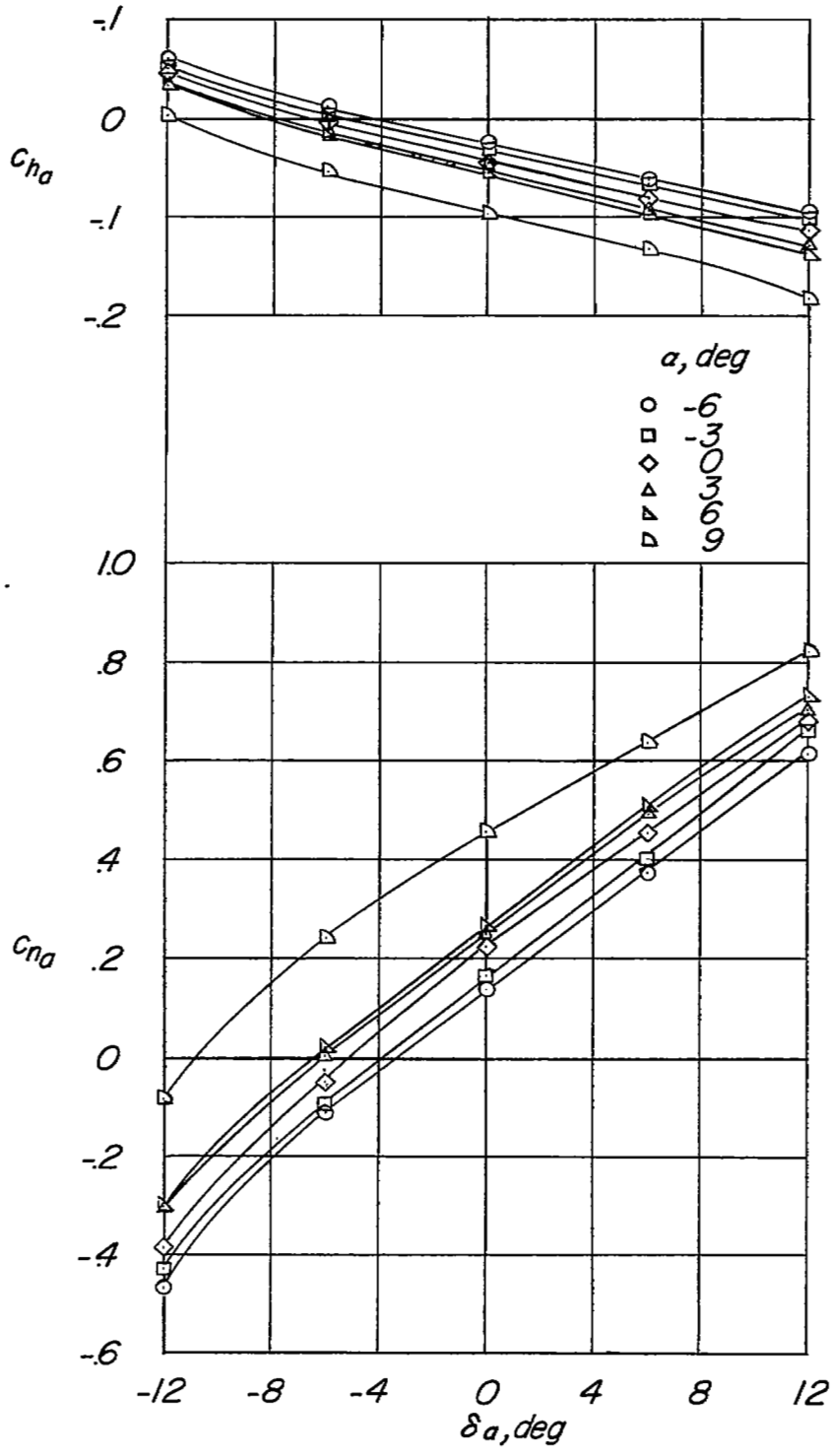

Figure 10.- Concluded.