To Mr. Charles J. McCarthy, Gen. Mgr., Vought-Sikorsky Airc. Div. United Aircraft Corp.

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WIND-TUNNEL INVESTIGATION OF AN NACA 66,2-216 LOW-DRAG WING WITH SPLIT FLAPS OF VARIOUS SIZES

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SUMMARY

An investigation was conducted in the NACA 19-foot pressure wind tunnel of a rectangular wing having NACA 66,2-216 low-drag airfoil sections and various sizes of simple split flaps. The purpose of the investigation was, primarily, to determine the influence of these flap installations on the aerodynamic characteristics of the wing. Complete lift, drag, and pitching-moment characteristics were determined for a range of test Reynolds numbers from about 2,600,000 to 4,600,000 for each of the installations and for the plain wing.

The results of this investigation indicate that values of maximum lift coefficient similar to those of wings with conventional airfoil sections and split flaps can be expected of wings having the NACA 66,2-216 low-drag sections. The increment of maximum lift due to the split flap was found to be practically independent of the Reynolds number over the range investigated. The optimum split flap on the basis of maximum lift appears to have a chord about 20 percent of the wing chord and a deflection of 60°. The $C_{L_{max}}$ of the wing with the 0.20c partial-span flap deflected 60° is 2.07 at a Reynolds number of 4,600,000 while with the full-span flap it is approximately 2.53; the increment of the maximum lift coefficient due to the flap is approximately proportional to the flap span.

Although the addition of a split flap tends to hasten the stall and to cause it to occur more abruptly, little change in pattern is evidenced by observations of the behavior of wool tufts on the wing.

INTRODUCTION

The present national emergency has, among other things, given impetus to the demands for higher speed aircraft. However, several pressing aerodynamic problems are encountered that have an increasingly adverse effect on performance as the aircraft speed rises. One of these problems is presented by the compressibility burble which has been solved to a great extent for the present needs by the development of the NACA low-drag airfoil sections. These airfoils have, however, proved to be somewhat sensitive to surface irregularities and some doubt exists as to the effectiveness of various high-lift devices used in conjunction with them.

To date very little data are available on the aerodynamic characteristics of low-drag wings with high-lift devices, although some isolated tests for two-dimensional flow have been made. These tests were not extensive and only a few types of flap were tested.

In the NACA 19-foot pressure tunnel, some tests have been made of complete airplane models with wings having NACA low-drag airfoil sections. In these tests a 20-percent-chord split flap and an extensible trailing-edge flap were investigated. Although these tests have been far from conclusive, the results, nevertheless, indicate that values of $C_{L_{max}}$ similar to those of conventional sections with split flaps can be expected from wings having the NACA low-drag sections.

This paper presents the first part of an extensive investigation to determine the effect of various high-lift devices on the aerodynamic characteristics of wings having NACA low-drag airfoil sections. In the present tests the simplest phase of the investigation was carried out. That is, split flaps of various chords and spans were tested on a plain wing of rectangular plan form and the characteristics of the combination determined. The remaining portion of the program will be devoted to the determination of the aerodynamic characteristics of wings of various plan forms using NACA low-drag sections in combination with several different types of high-lift devices.

MODELS

Plain Wing

The plain wing or basic model (fig. 1) was constructed of laminated mahogany, reinforced with steel spars, to the NACA 66,2-216 low-drag airfoil section (fig. 2). The model, rectangular in plan form with elliptical tips, has no dihedral or geometric twist. The span is 15 feet, the aspect ratio 7.0, and the area 32.14 square feet. An "aerodynamically smooth" surface was obtained by spraying the wing with a number of coats of lacquer and then rubbing until smooth with No. 500 water cloth.

Flaps

Simple split flaps of 10, 20, and 30 percent of the wing chord were tested. These flaps were made of 1/16-inch galvanized sheet steel curved to approximate the contour of the flap portion of the wing lower surface. Wooden blocks, cut to the appropriate shape, were attached to the wing lower surface and the flap to obtain each of the desired flap deflections. For the partial-span condition the flaps extended over 53 percent of the wing span. (See fig. 1.) This distance was determined as the distance that exists between the inboard ends of $0.37\frac{b}{2}$ conventional ailerons, should they be used. The full-span arrangement of the flaps extended along 90 percent of the over-all wing span.

TESTS

The tests were conducted in the NACA 19-foot pressure wind tunnel at an absolute pressure of 35 pounds per square inch with the model mounted on the standard wing supports. (See fig. 3.)

Since the plain wing is used as the basis for comparing the merits of the various flap arrangements, a set of complete polar runs was first made for this condition. For these runs the angle of attack was varied from -5° through the stall for dynamic pressures of 13, 20, 40, 70, and 100 pounds per square foot corresponding to test

Reynolds numbers of about 2,100,000; 2,600,000; 3,600,000; 4,600,000; and 5,600,000. Simultaneous measurements of lift and drag were recorded by a six-component electrical-recording balance. In addition to the complete polars, measurements of lift and drag were made through the low-lift range for dynamic pressures of 150 and 175 pounds per square foot.

In order to provide a basis for some comparison of aerodynamic characteristics obtained in these tests with section characteristics obtained in two-dimensional-flow tests, momentum surveys were made in the wing wake at dynamic pressures of 20 and 40 pounds per square foot. These surveys were made with a rake composed of a number of static and total head tubes. Measurements were made at 1-foot intervals along the span except near the wing tips, where intervals of about 2 inches were used. At each of these stations the angle of attack was varied sufficiently to properly bracket the minimum-drag region.

For the partial-span arrangement of the 10-percent-chord flaps, complete polar runs were made for flap deflections of 15°, 30°, 45°, and 60° at dynamic pressures of 20, 40, and 70 pounds per square foot. Complete polar runs were made for the full-span flap arrangement but only at the 60° deflection. Similarly, the wing was tested with 20- and 30-percent chord flaps at the various deflections and dynamic pressures.

In order to study the wing stalling characteristics, wool tufts were fastened with cellulose tape to the wing upper surface at the 20-, 30-, 40-, 50-, 60-, 70-, 80-, and 90-percent-chord points. These tufts were arranged in parallel rows spaced approximately 7 inches apart along the wing span. Slightly closer spacing was used near the tips. Sketches were drawn from visual observations of the behavior of the tufts at various angles of attack through the stall for the plain wing, and for each of the 10-, 20-, and 30-percent-chord flaps deflected 60° in the partial-span arrangement only. The tuft observations were made at a dynamic pressure of 70 pounds per square foot.

RESULTS AND DISCUSSION

Coefficients

The data presented in this report are given in standard nondimensional coefficient form corrected for the effect of model support tare and interference, and for jet-boundary effects.

The coefficients and symbols used herein are defined as follows:

C_L lift coefficient $\frac{L}{qS}$

C_D drag coefficient $\frac{D}{qS}$

$C_{m_{c/4}}$ pitching-moment coefficient about the quarter-chord point of the plain wing $\frac{M}{qSc}$

C_{D_o} wing profile-drag coefficient

c_{d_o} section profile-drag coefficient $\frac{d_o}{qc}$

where q dynamic pressure in the undisturbed air stream $\frac{1}{2}\rho V^2$

S wing area (32.14 sq ft)

c mean wing chord $\frac{S}{b}$ (2.14 ft)

b wing span (15 ft)

ρ mass density of air, slugs per cubic foot

and δ_f flap deflection measured between the lower surface of the wing and the flap

α' geometric angle between the root chord and the horizontal axis of the tunnel

α angle of attack of root chord corrected for jet-boundary interference

R test Reynolds number based on mean wing chord,

$$\frac{\rho V c}{\mu}$$

μ coefficient of viscosity

Precision

The accidental experimental errors as determined from repeat tests are believed to be within the following limits:

α	$\pm\, 0.10^{\circ}$
$C_{L_{max}}$	$\pm\, 0.03$
$C_{m_{c/4}}$	$\pm\, 0.005$
$C_{D(C_L = 0)}$	$\pm\, 0.0003$
$C_{D(C_L = 1)}$	$\pm\, 0.0006$
$C_{D(C_L = 2)}$	$\pm\, 0.002$
$c_{d_o(c_l = 0)}$ wake	$\pm\, 0.0002$
δ_f	$\pm\, 0.5^{\circ}$
Flap position	$\pm\, 0.002c$

The coefficients given are corrected for the effect of support tare and interference as determined for the plain wing. No additional tare tests were made for the flap installations, as the tare increment is believed to be small.

Plain Wing

The aerodynamic characteristics of the basic model as determined in these tests are given in figures 4 through 6 as the zero flap deflection condition. By referring to the lift curves, it can be seen that up to a C_L of about 0.1 the lift curve is straight, but between C_L of 0.1 and 0.5 there is a definite change. Above C_L of approximately 0.5 the lift-curve slope becomes progressively less up to the stall. The slope of this portion of the lift curve increases and the change in slope, as mentioned above, tends to disappear as the Reynolds number increases. Also, with increased Reynolds number the angle of stall is increased.

Because of the variation of the position of the aerodynamic center with C_L, the pitching-moment coefficient was computed about the wing quarter-chord point. Examination of these curves reveals that the pitching-moment coefficient becomes greater positively as the angle of attack is increased and that there is a slight scale effect, the value of the pitching-moment coefficient increasing positively with an increase in Reynolds number.

The section profile-drag coefficients determined by the momentum method are shown in figure 7 for two values of the Reynolds number. From these plots, the wing profile-drag coefficient, C_{D_0}, was determined by integrating the values of $c_{d_0} \times c$ across the span as suggested in reference 1. The minimum wing profile-drag coefficient obtained from these tests at an approximate test Reynolds number of 2,700,000 is 0.0038. The airfoil section profile-drag coefficients shown on the figure are in good agreement with the values obtained from wake measurements of an airfoil with the same low-drag section in the NACA two-dimensional low-turbulence tunnel. It should be pointed out that the turbulence of the 19-foot pressure tunnel is almost as low as that of free air at low test speeds, and increases slightly with increase in tunnel test speed.

Values of minimum profile-drag coefficient of the wing obtained from the force-test measurements were considerably higher than those obtained from the momentum method. The differences are believed to be due to the difficulties involved in accurately measuring the tare forces due to the model supports in the case of the low-drag wing, and to some error in the momentum measurement due to the difficulty of correctly obtaining the tip effects of the wing.

Wing with Flaps

The lift, drag, and pitching-moment characteristics for the wing with the various flap installations are presented in figures 4 to 6, inclusive, where the data are plotted against angle of attack for three values of Reynolds number. The lift curves, in general, are uniform and consistent but there is some variation in the shape at the peak. However, the change in slope that appears to exist at low Reynolds numbers in the lift curves of the plain wing is not evident with flaps deflected. The elimination of this effect may be due to the decrease of a cross flow at the trailing edge over the center portion of the wing when the flaps are deflected. The slope of the lift curve, $\frac{dC_L}{d\alpha}$, appears to decrease with increase in flap deflection, while, on the other hand, for a given deflection, it tends to increase with an increase of Reynolds number.

Examination of the pitching-moment curves shows that the pitching-moment coefficient about the quarter-chord point varies with Reynolds number and α but the variation is not consistent. The pitching-moment coefficient does, however, increase negatively as the flap deflection and flap chord are increased. A comparison of the pitching-moment coefficients obtained with a 20-percent-chord split flap on an NACA 23012 airfoil (reference 2), with the results of the present tests, while not strictly comparable, does give values of the same magnitude.

The variation of $C_{L_{max}}$ with Reynolds number is given for the wing with various flaps in figures 8a, 8b, and 8c. A marked scale effect is noticeable both for the plain wing and for the wing with flaps. The curve for the plain wing appears to give an approximately linear variation between Reynolds numbers of 2,000,000 and 6,000,000 with no indication of an immediate leveling off. The curves for the flapped condition appear to deviate somewhat from a linear variation but no consistent change can be determined, so that, in general, there is little scale effect on the increment of $C_{L_{max}}$. The increase of $\Delta C_{L_{max}}$ obtained with the full-span arrangement over that obtained with the partial-span flap is approximately proportional to the increase in flap span.

The variation of $\Delta C_{L_{max}}$ with flap deflection is given in figure 9. At the deflection of about 60° the curves are beginning to level off, indicating that very little gain in lift can be expected beyond this point. A cross plot of these curves (fig. 10) showing the variation of $\Delta C_{L_{max}}$ with flap chord reveals that very little additional lift is obtained by increasing the flap chord beyond 20 percent of the wing chord. From a consideration of these two sets of data, it would seem that a 20-percent-chord split flap deflected about 60° would be about the optimum arrangement from consideration of $C_{L_{max}}$.

Stalling Characteristics

The stall diagrams for the plain wing and for the wing with each of the 10-, 20-, and 30-percent-chord flaps deflected 60° are given in figures 11 to 14. These diagrams show that the stall begins in the rear-center portion of the plain wing, moving forward and outward with increase in angle of attack. The movement appears to be fairly uniform and gradual, indicating desirable stalling characteristics. With the addition of flaps the beginning of the stall is somewhat delayed; once started, however, it develops much more rapidly with complete stall occurring at a lower angle of attack than for the plain wing. The diagrams also indicate that the pattern of the stall is not greatly affected by increases of flap chord. From the visual observations, however, it appeared that the velocity of the inflow near the wing tips was substantially increased as the flap chord was increased. The stall diagrams give the impression that the left side of the wing stalls earlier than the right side, but the difference is small and may be due to a slight asymmetry of the wing rather than to an aerodynamic effect.

CONCLUSIONS

1. The addition of a simple split flap to a rectangular wing with NACA 66,2-216 low-drag airfoil sections gives aerodynamic characteristics that are approximately the same as those obtained with similar flaps on wings having conventional airfoil sections.

2. The most favorable split-flap installation from a standpoint of $C_{L_{max}}$ appears to be one with a chord of about 20 percent of the wing chord and deflected about 60°.

3. The increment of maximum lift due to the split flap was found to be practically independent of the Reynolds number over the range investigated.

4. The $C_{L_{max}}$ of the wing with the 0.20c partial-span flap deflected 60° is 2.07 at a Reynolds number of 4,600,000 and with the full-span flap the $C_{L_{max}}$ is 2.53. The increment of the maximum lift coefficient due to the flap is approximately proportional to the flap span.

5. The addition of the split flap to the rectangular wing, in general, reduced the angle of attack at which the stall occurred but did not appreciably alter the pattern of the stall.

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2. Wenzinger, Carl J., and Harris, Thomas A.: Wind-Tunnel Investigation of an N.A.C.A. 23012 Airfoil with Various Arrangements of Slotted Flaps. Rep. No. 664, NACA, 1939.

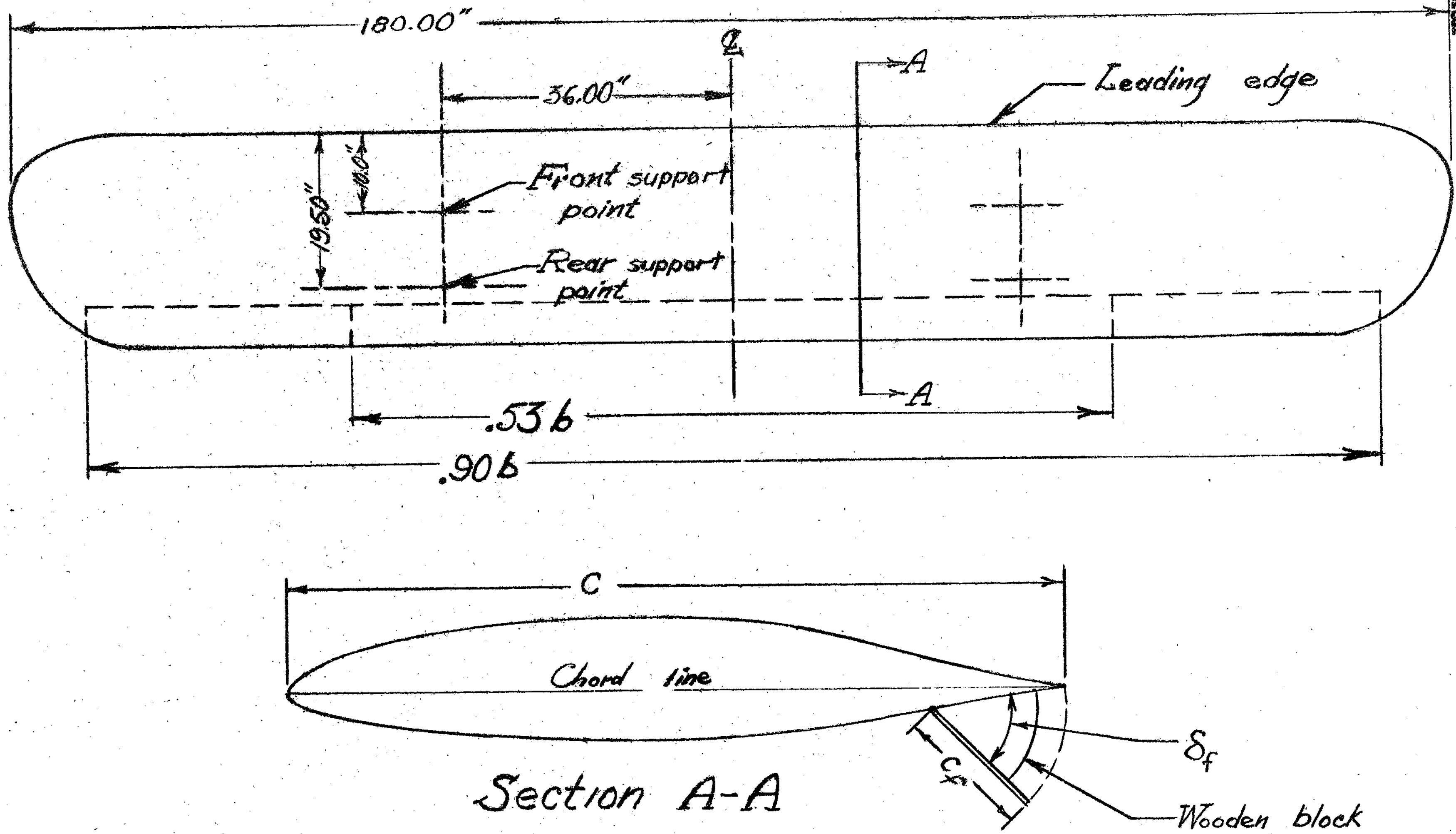
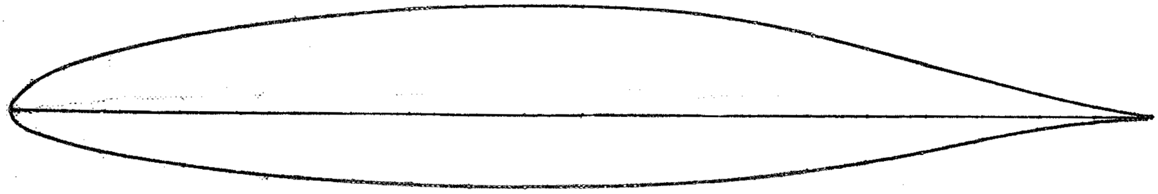


Figure 1.- The rectangular NACA 66,2-216 low-drag wing.



NACA 66,2-216 AIRFOIL SECTION

Ordinates for NACA 66,2-216 airfoil section			
Upper surface		Lower surface	
Station	Ordinate	Station	Ordinate
Percent c	Percent c	Percent c	Percent c
.0	.0	.0	.0
.371	1.242	.629	- 1.112
.607	1.501	.893	- 1.319
1.091	1.886	1.409	- 1.608
2.317	2.615	2.683	- 2.127
4.794	3.701	5.206	- 2.869
7.284	4.563	7.716	- 3.441
9.781	5.308	10.219	- 3.934
14.788	6.500	15.212	- 4.702
19.806	7.428	20.194	- 5.290
24.832	8.155	25.168	- 5.741
29.862	8.708	30.138	- 6.080
34.897	9.098	35.103	- 6.312
39.936	9.356	40.064	- 6.462
44.978	9.471	45.022	- 6.523
50.023	9.431	49.977	- 6.483
55.073	9.224	54.927	- 6.336
60.141	8.800	59.859	- 6.048
65.191	8.084	64.809	- 5.574
70.198	7.068	69.802	- 4.866
75.181	5.889	74.819	- 4.037
80.148	4.585	79.852	- 3.107
85.106	3.265	84.894	- 2.177
90.061	1.937	89.939	- 1.235
95.021	.762	94.979	- .432
100.000	.000	100.000	- .000

Figure 2.- Ordinates for the NACA 66,2-216 low-drag airfoil section.

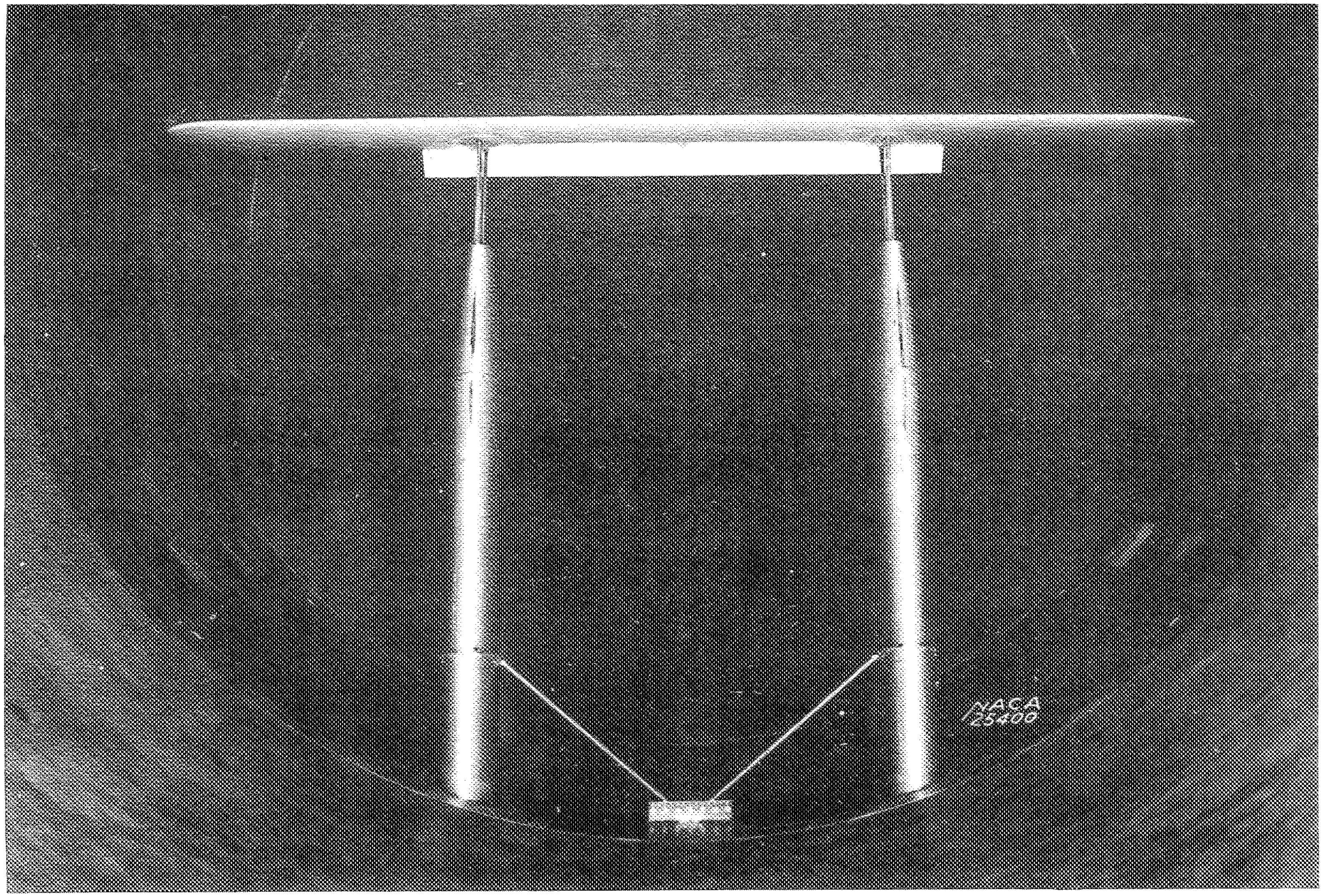
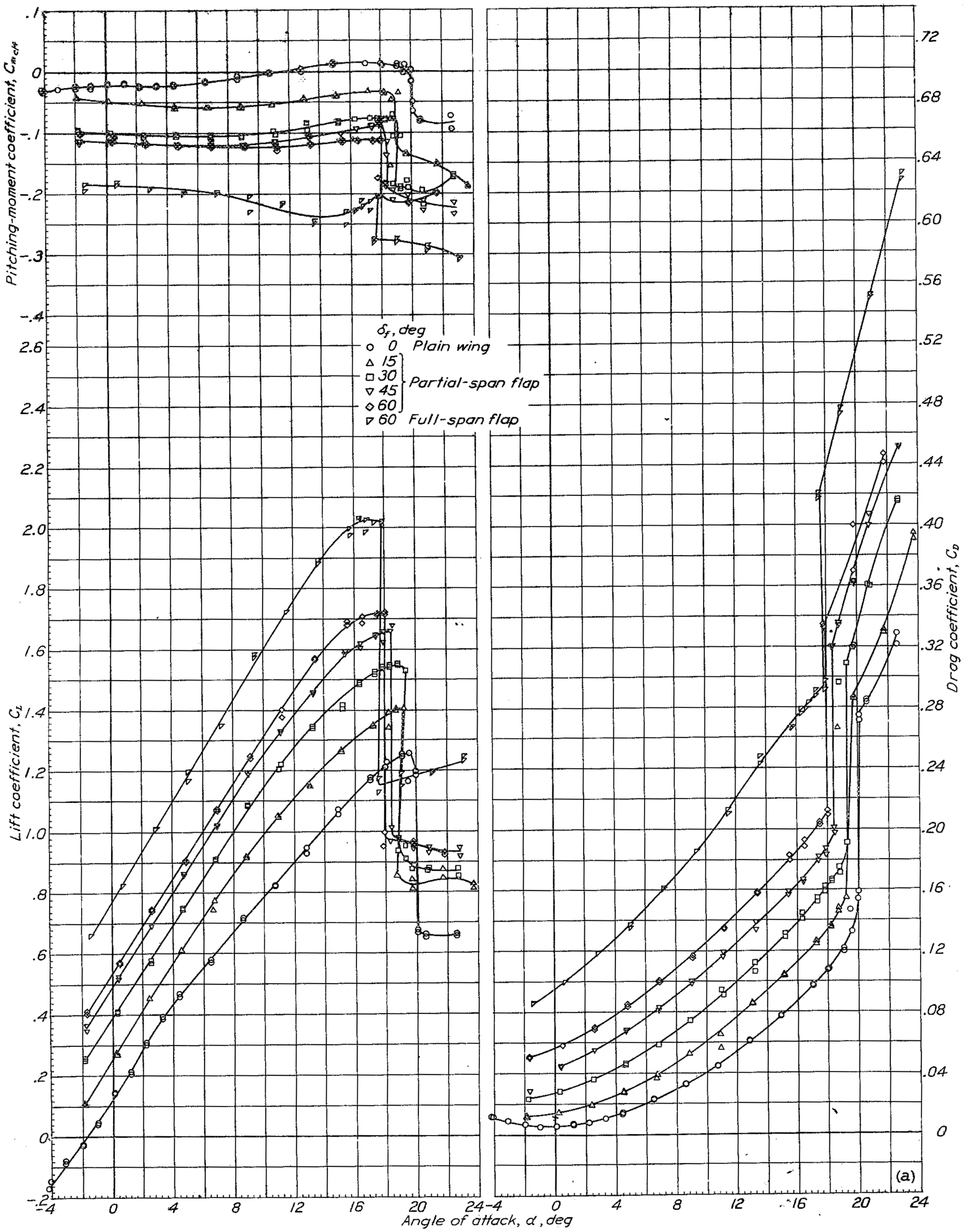
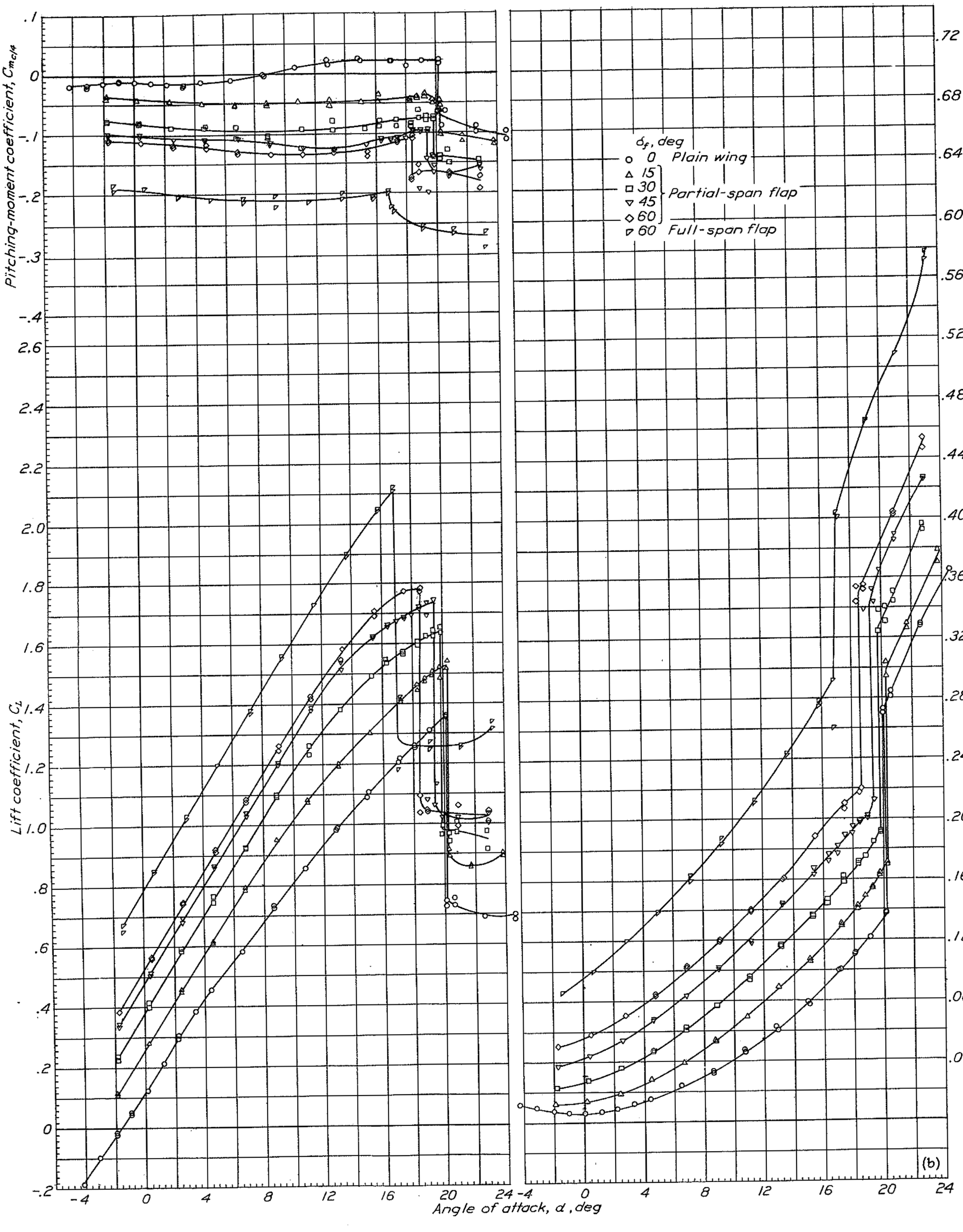


Figure 3.- The rectangular NACA 66,2-216 low-drag wing with split flap mounted on the standard wing support in the 19-foot pressure tunnel.

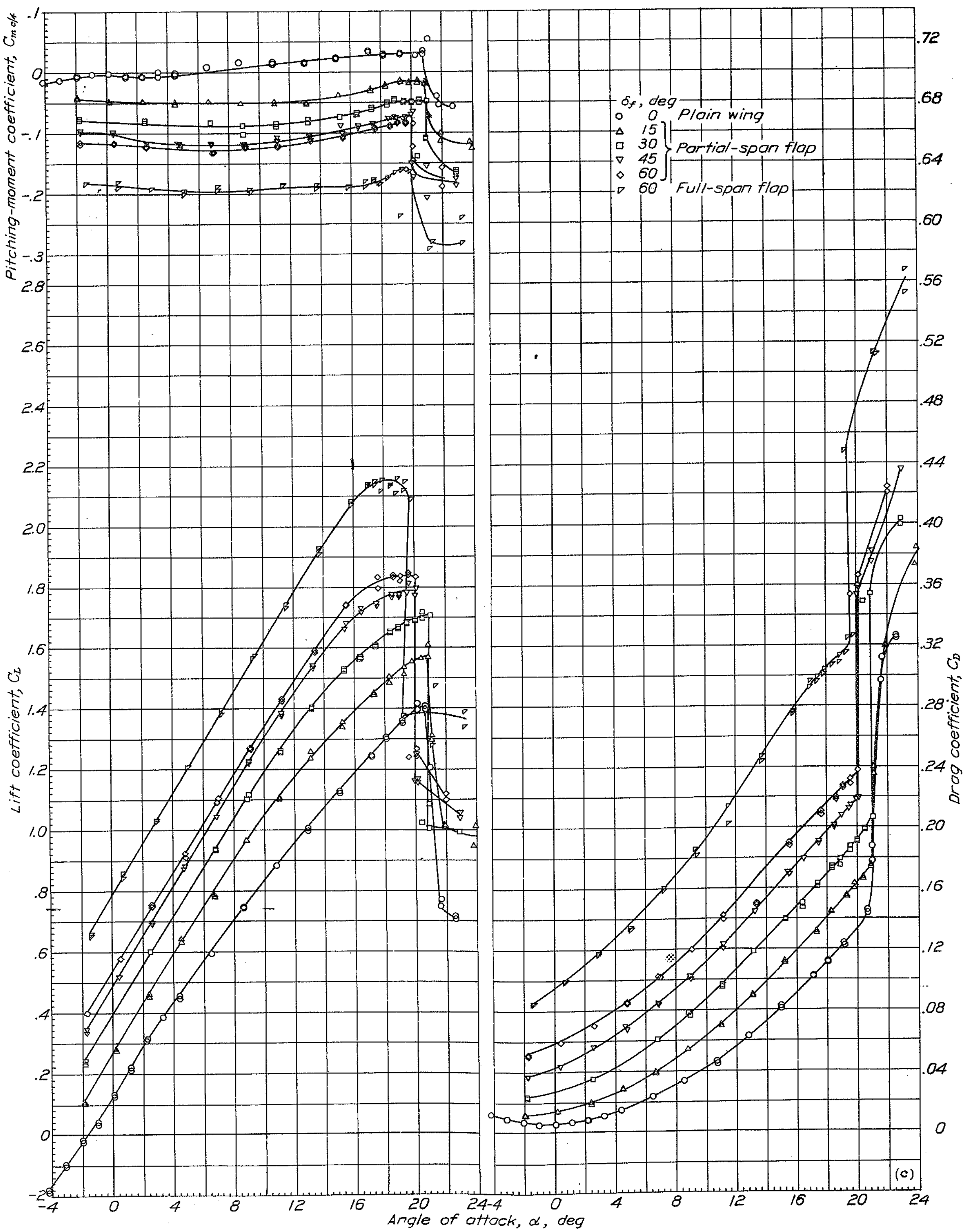


(a) R = 2,600,000.

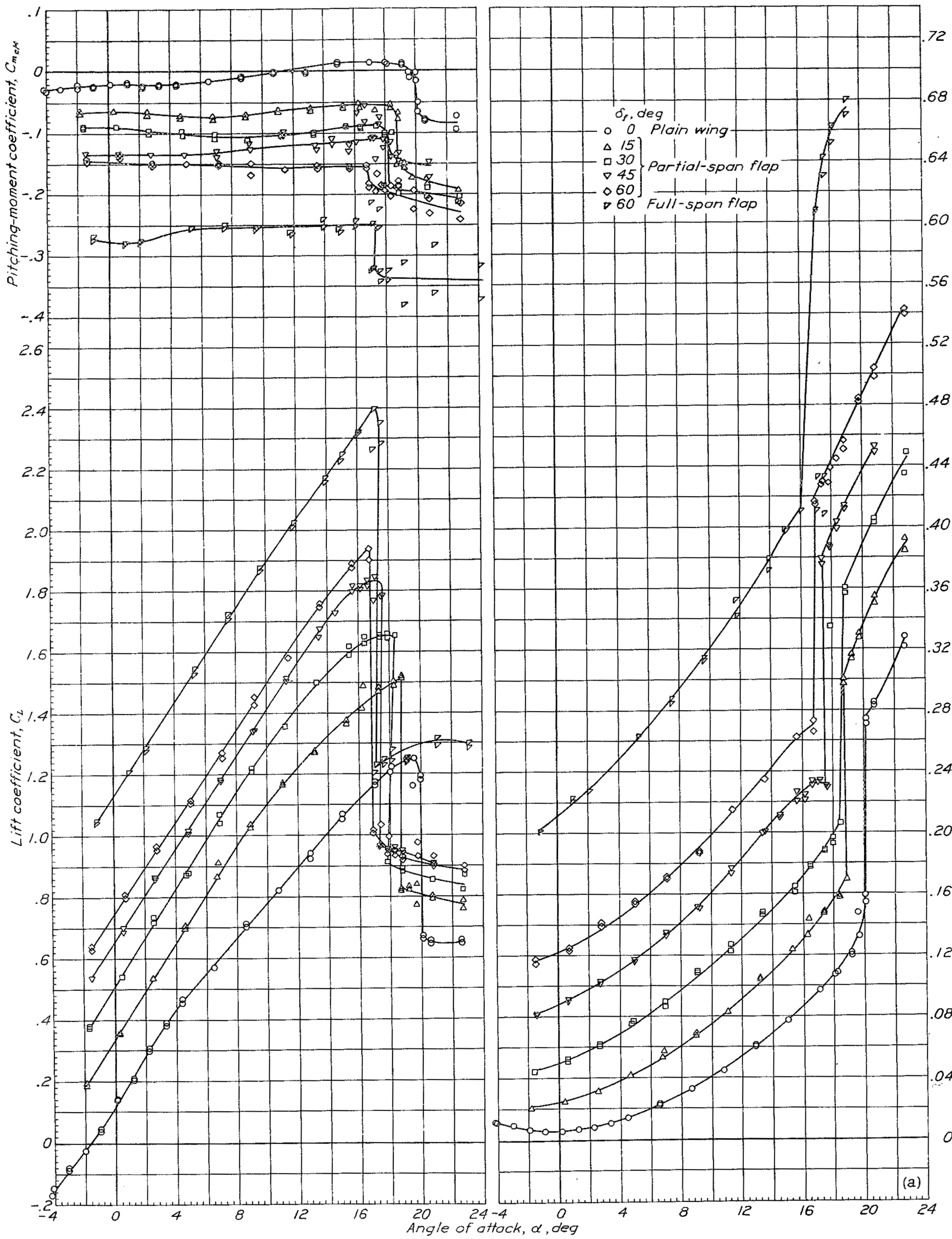
Figure 4a to c.- Aerodynamic characteristics of a rectangular NACA 66,2-.216 low-drag wing with 0.10c split flap.



(b) R = 3,600,000.
Figure 4.- Continued.

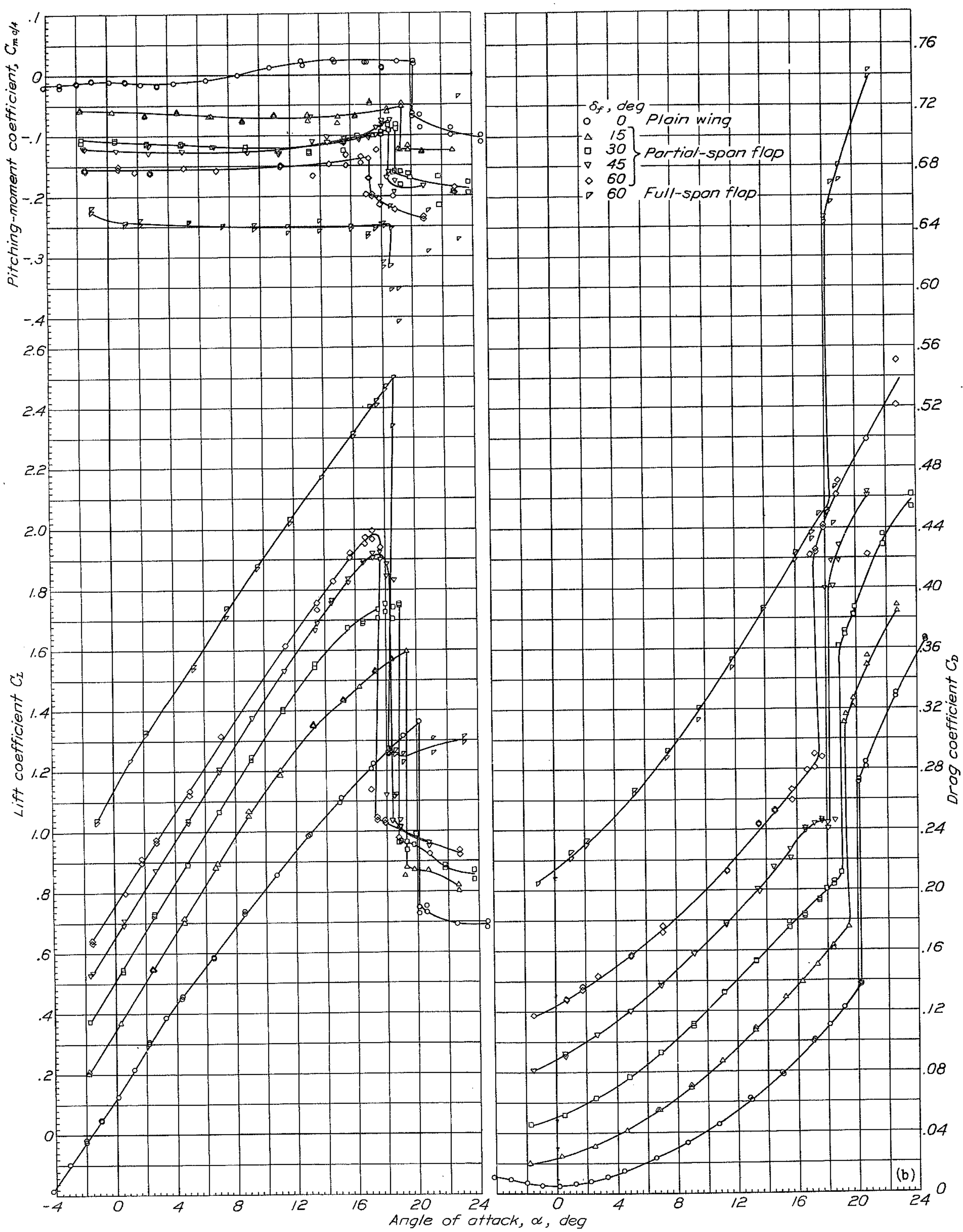


(c) R = 4,600,000.
Figure 4.- Concluded.

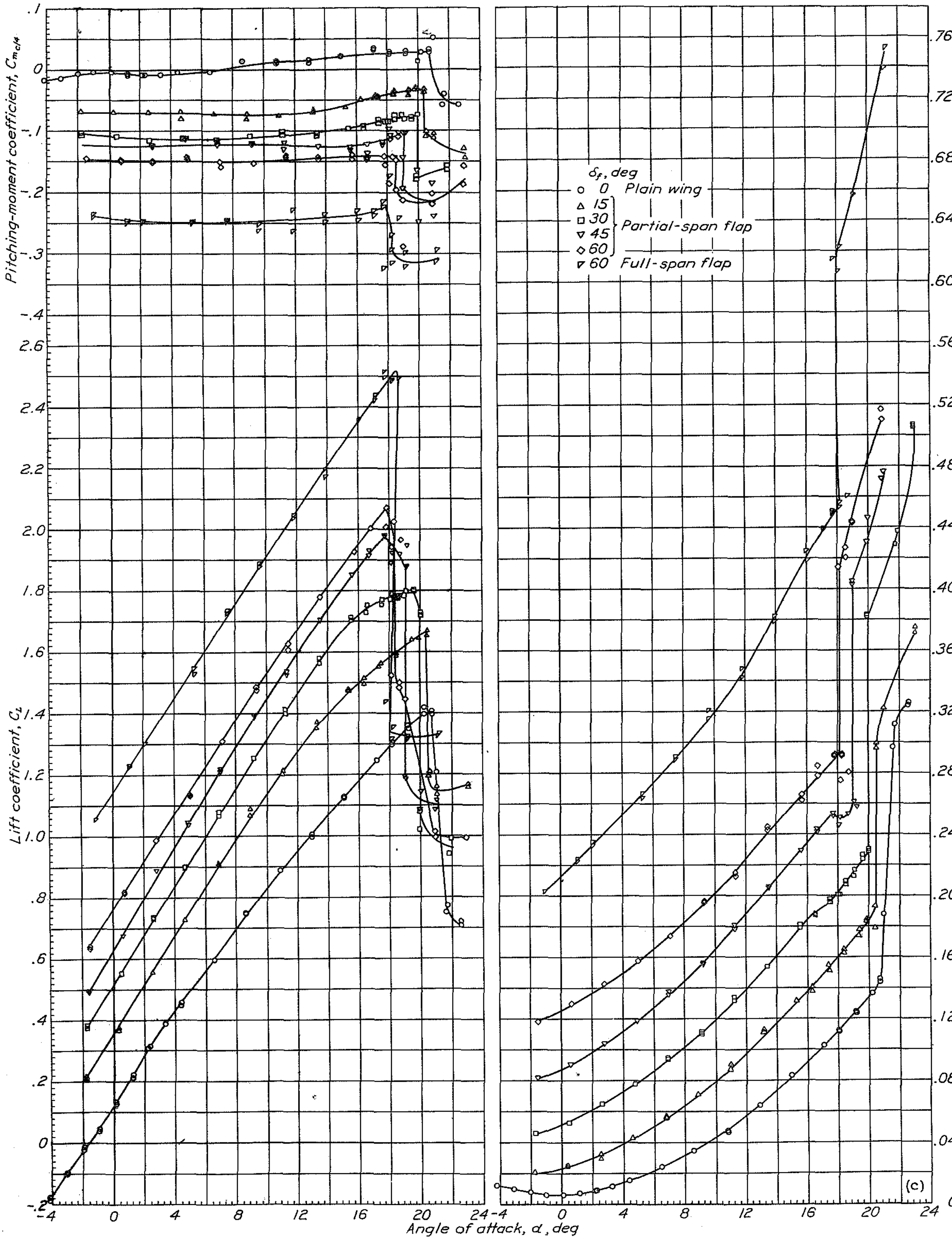


(a) R = 2,600,000.

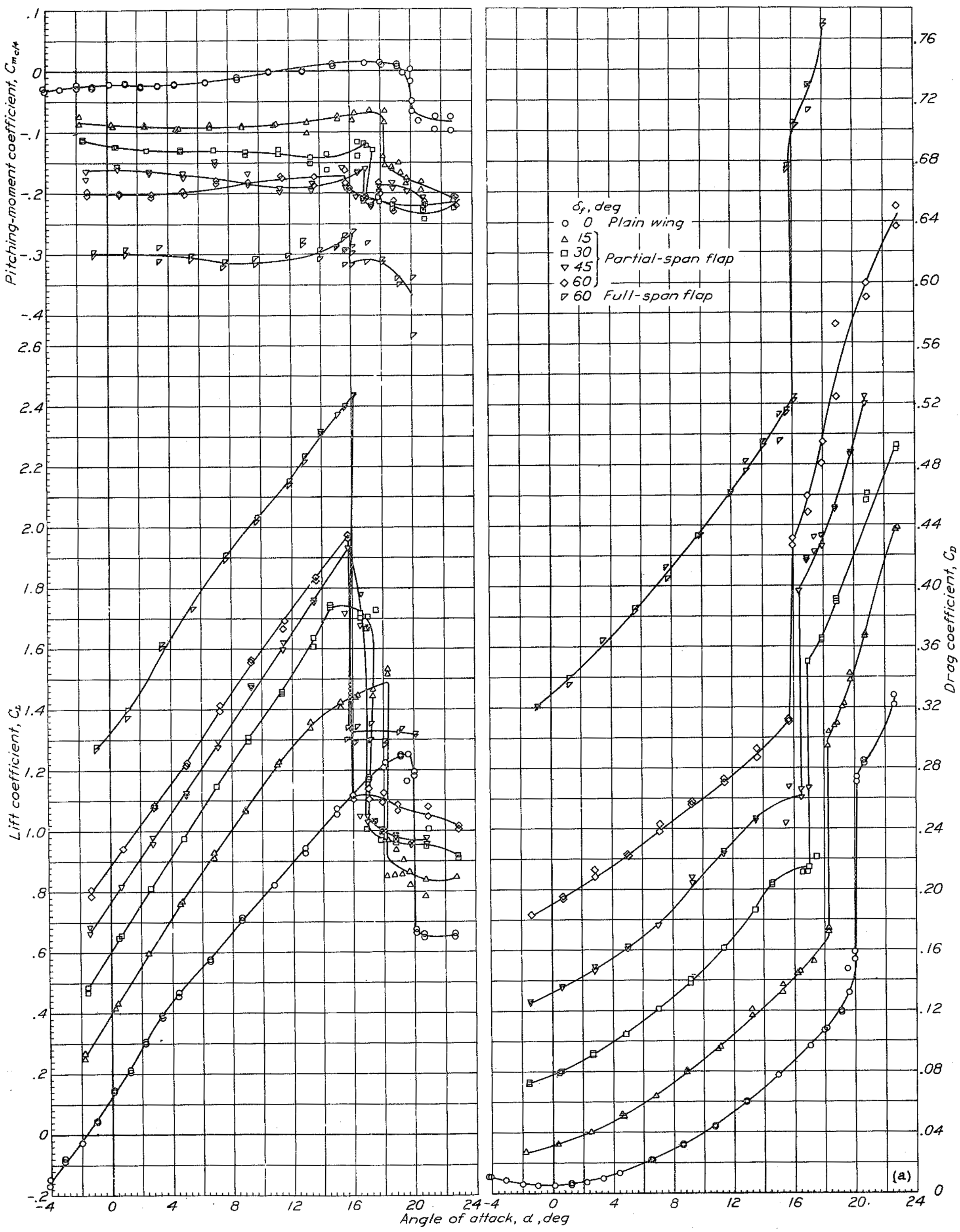
Figure 5a to c.- Aerodynamic characteristics of a rectangular NACA 66,2- 216 low-drag wing with 0.20c split flap.



(b) R = 3,600,000.
Figure 5.- Continued.

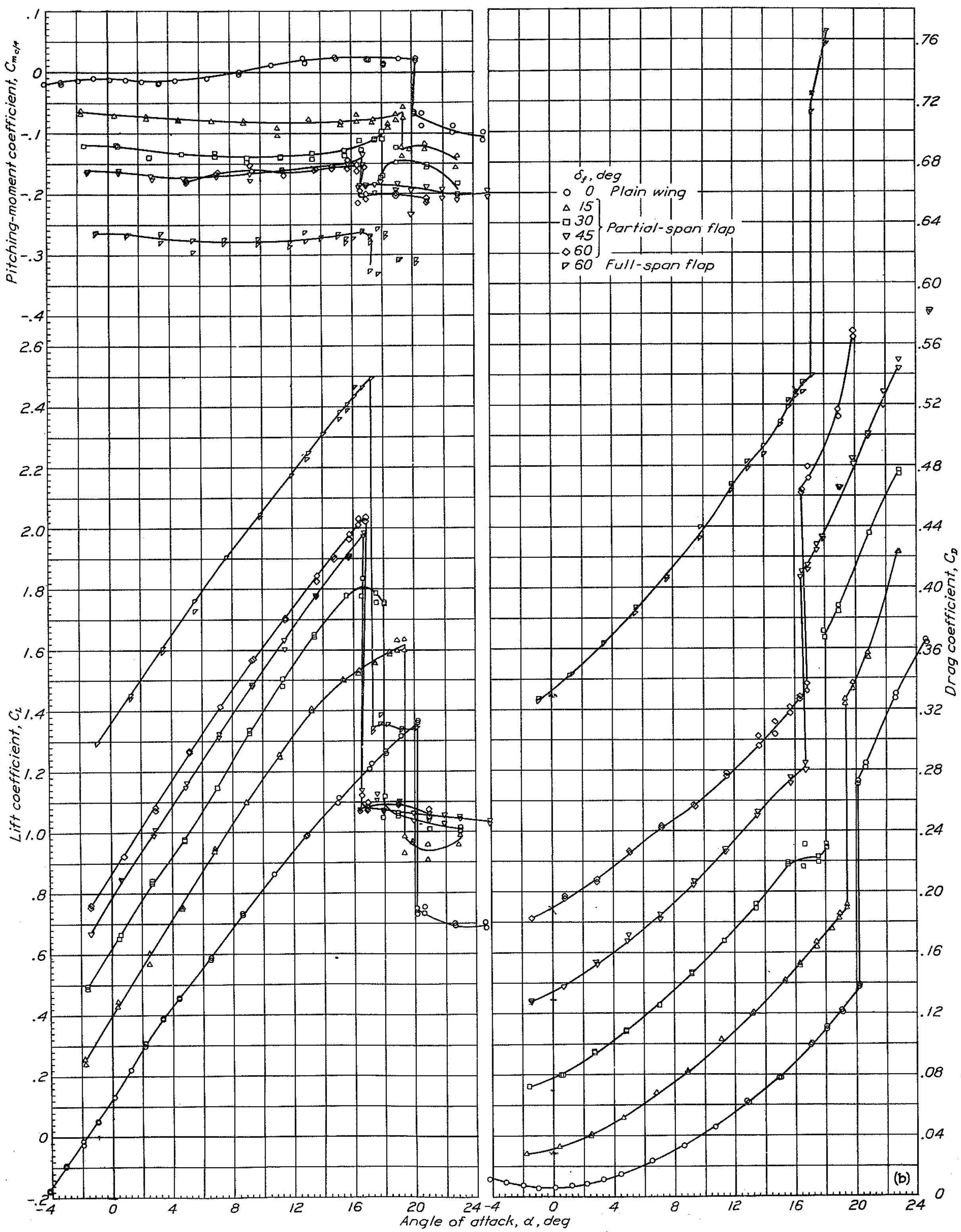


(c) R = 4,600,000.
Figure 5.- Concluded.

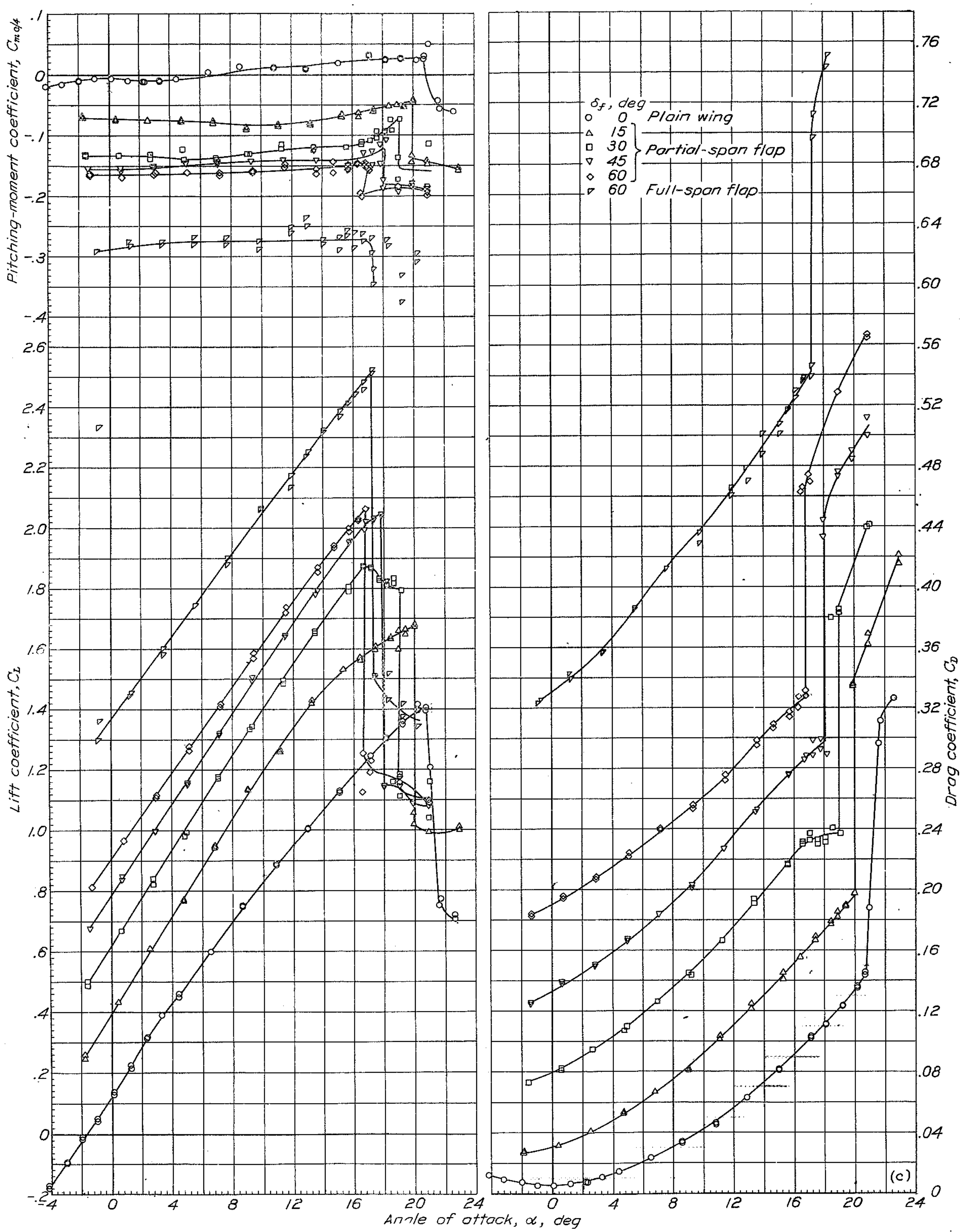


(a) R = 2,600,000.

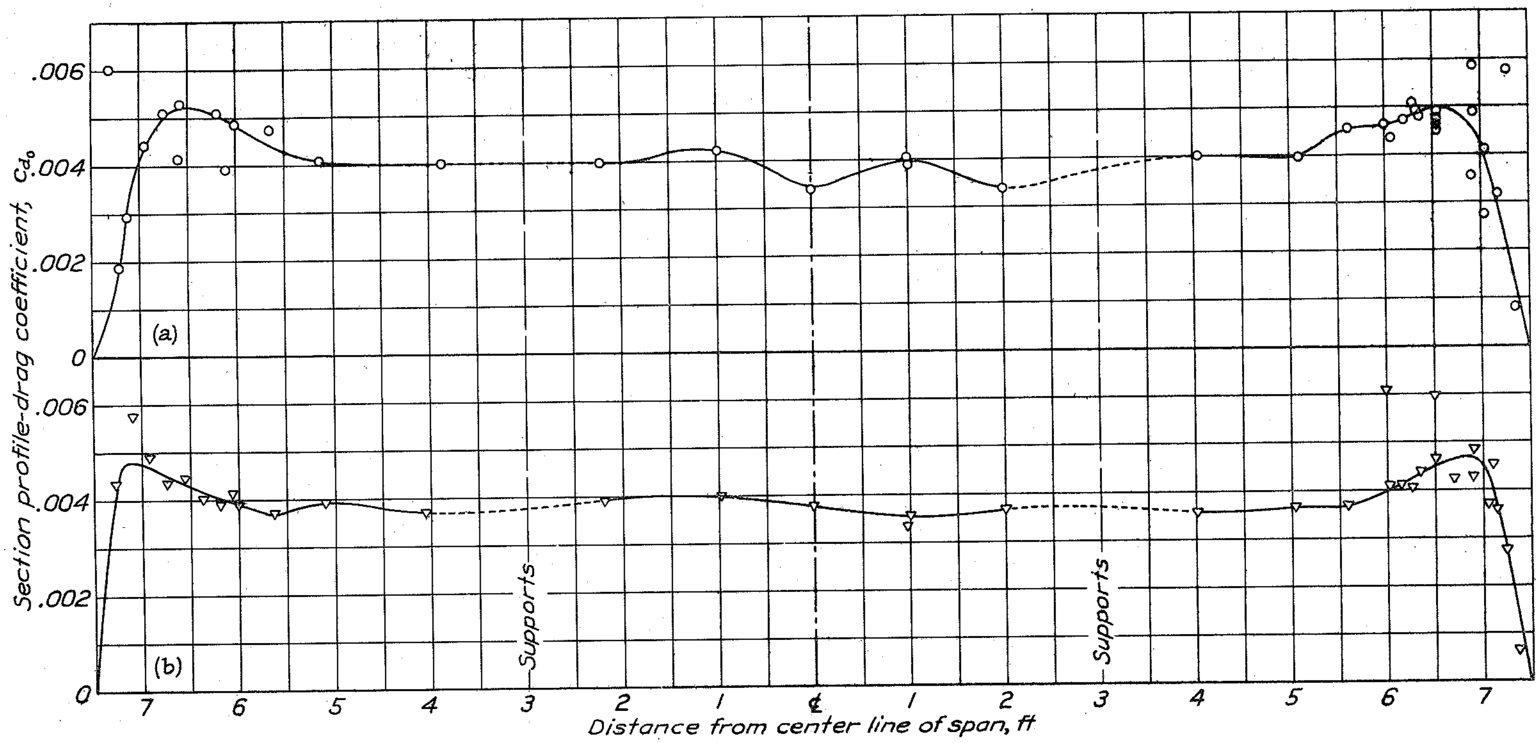
Figure 6a to c.- Aerodynamic characteristics of a rectangular NACA 66,2-.216 low-drag wing with 0.30c split flap.



(b) R = 3,600,000.
Figure 6.- Continued.



(c) R = 4,600,000.
Figure 6.- Concluded.



(a) R = 2,600,000. (b) R = 3,600,000.

Figure 7.- The variation of section profile-drag coefficient along the span of a rectangular NACA 66,2-.216 low-drag wing.

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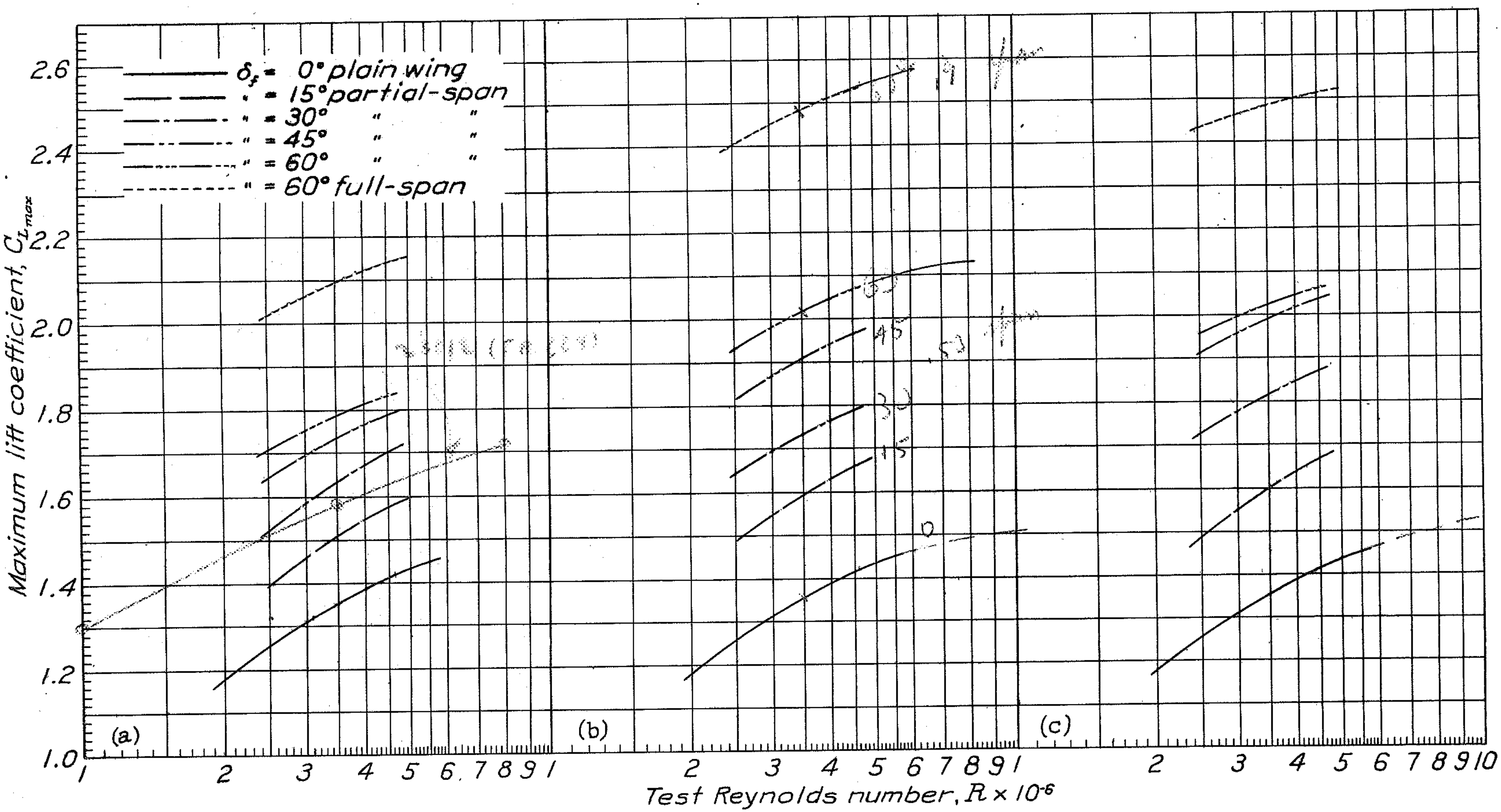

(a) With 0.10c split flap.

(b) With 0.20c split flap.

(c) With 0.30c split flap.

Figure 8.- Variation of maximum lift coefficient with Reynolds number of a rectangular NACA 66, 2-216 low-drag wing.

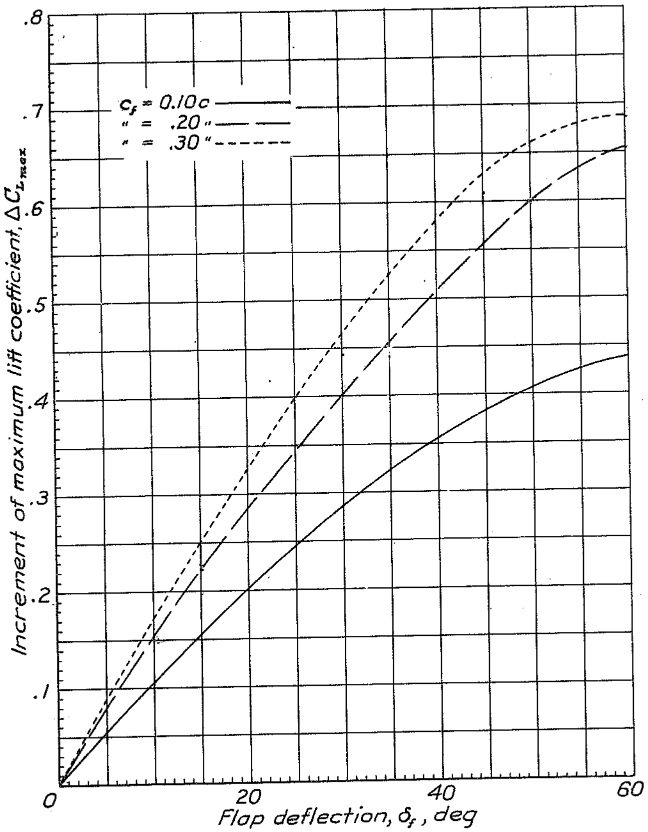

Figure 9.- Effect of flap deflection on increment of maximum lift coefficient for various sizes of partial-span split flap on a rectangular NACA 66,2- 216 low-drag wing.

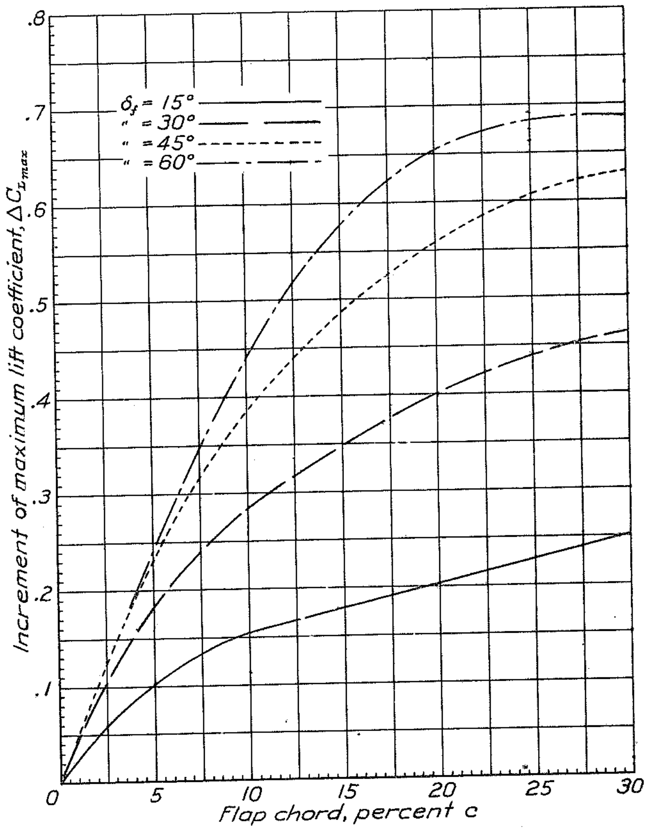

Figure 10.- Effect of flap chord on increment of maximum lift coefficient for various deflections of a partial-span split flap on a rectangular NACA 66,2-216 low-drag wing.

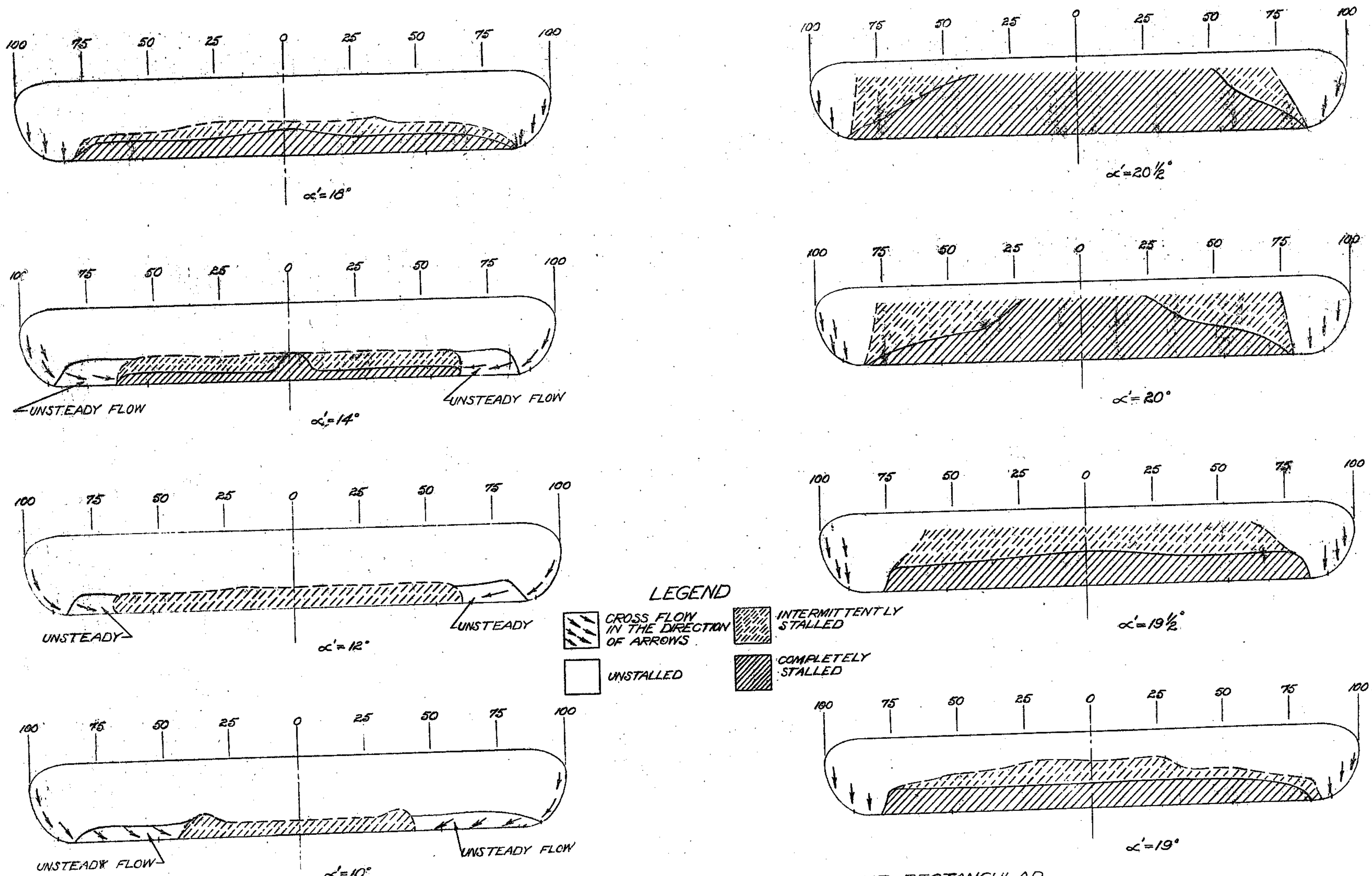

FIGURE 11.- STALL DIAGRAMS OF THE RECTANGULAR LOW-DRAG WING. R = 4,600,000. NACA 66,2-216 AIRFOIL SECTIONS.

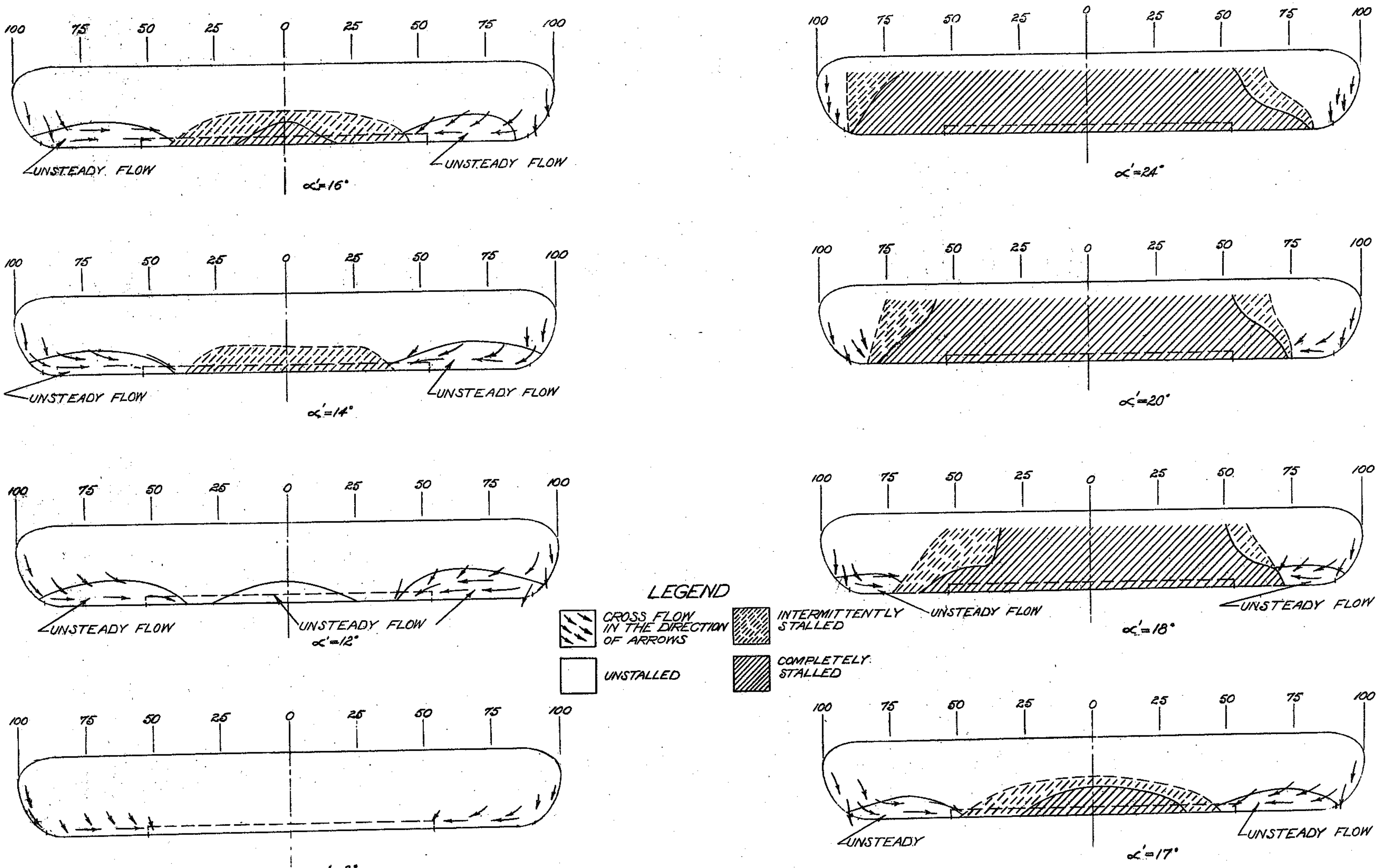

FIGURE 12.-STALL DIAGRAMS OF THE RECTANGULAR LOW-DRAG WING WITH 0.10c PARTIAL-SPAN SPLIT FLAPS DEFLECTED 60°. R = 4,600,000. NACA 66,2-216 AIRFOIL SECTION.

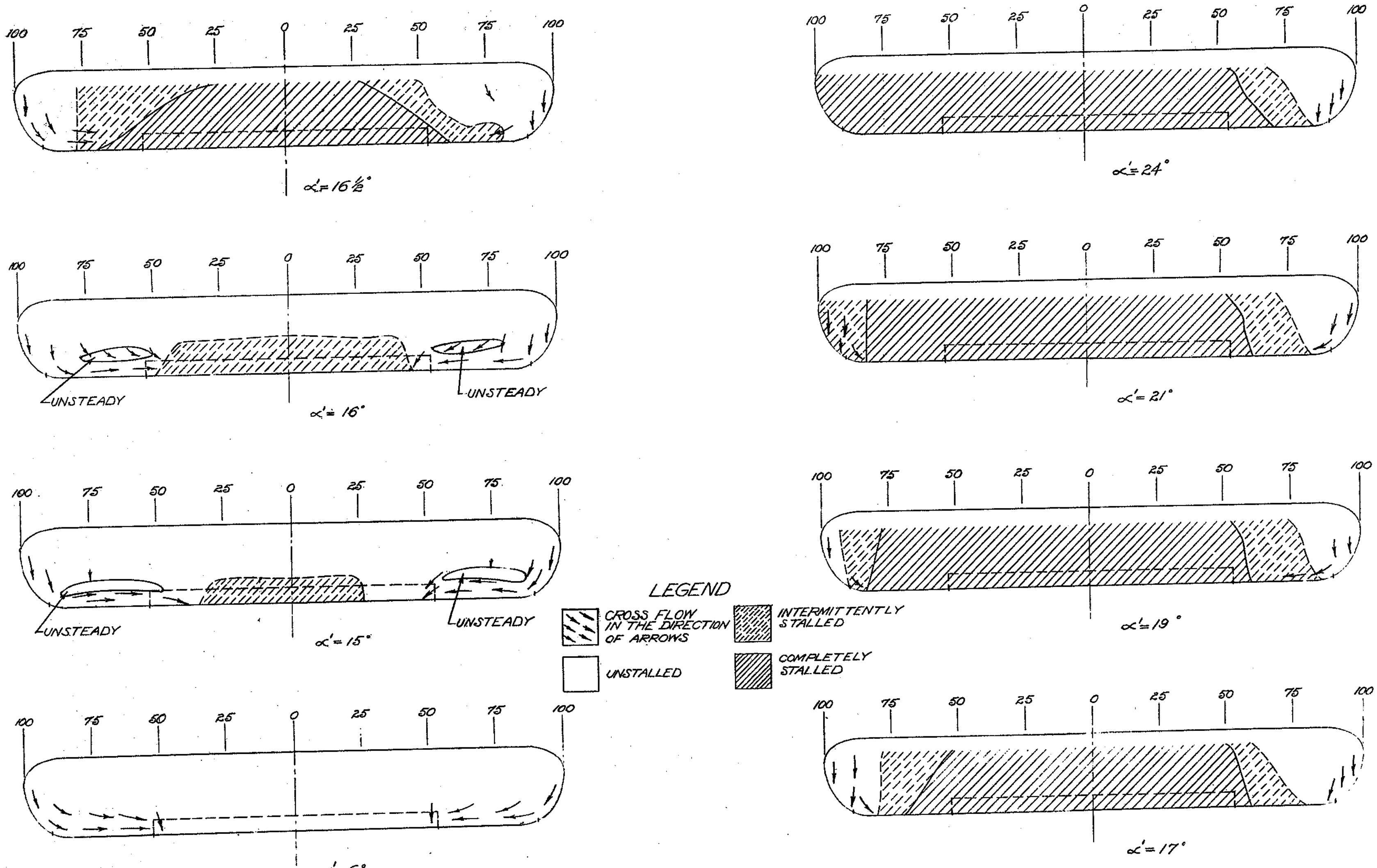

FIGURE 13.-STALL DIAGRAMS OF THE RECTANGULAR LOW-DRAG WING WITH 0.20c PARTIAL-SPAN SPLIT FLAPS DEFLECTED 60°. R = 4,600,000. NACA 66,2-216 AIRFOIL SECTIONS.

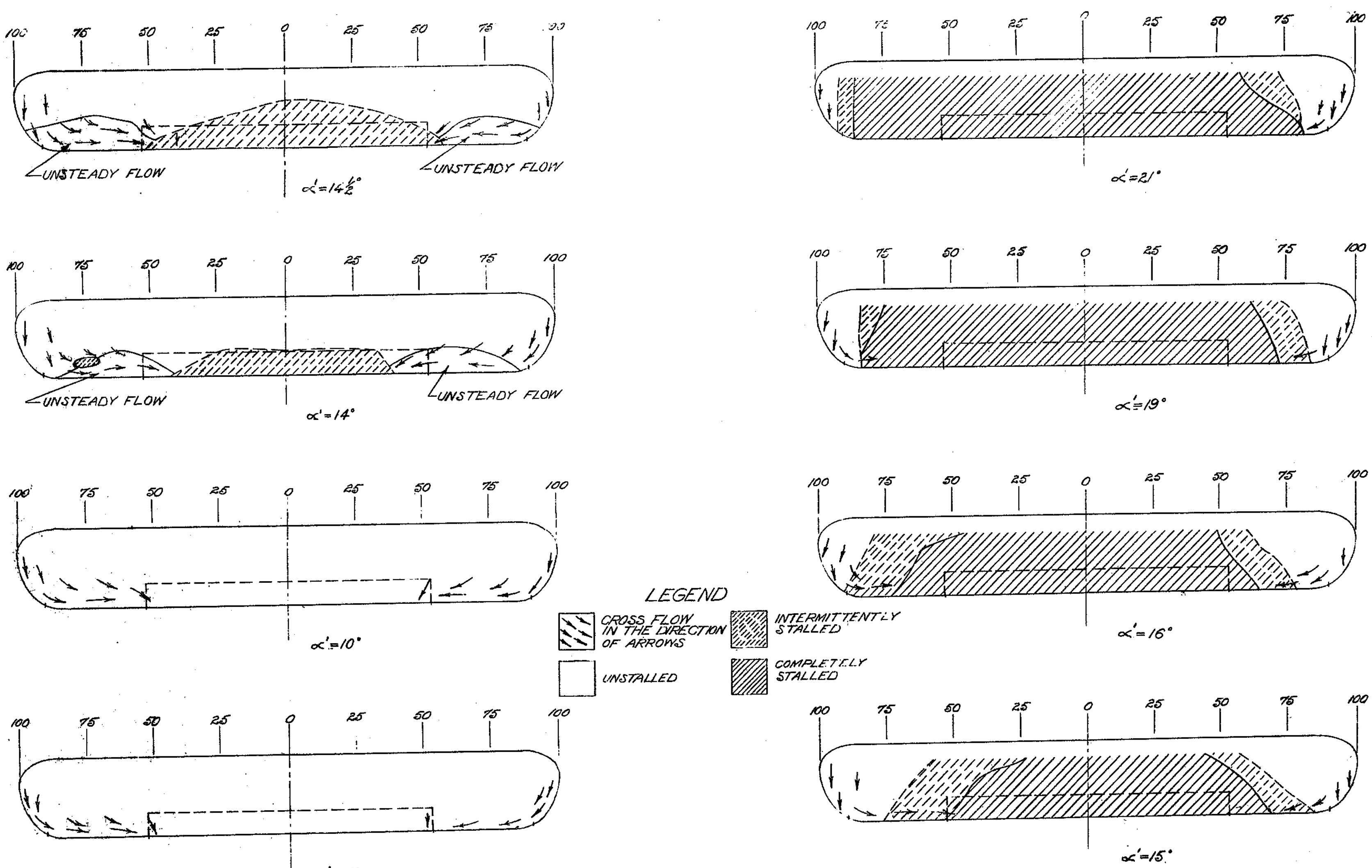

FIGURE 14.-STALL DIAGRAMS OF THE RECTANGULAR LOW-DRAG WING WITH 0.30c PARTIAL-SPAN SPLIT FLAPS DEFLECTED 60° R = 4,600,000. NACA 66,2-216 AIRFOIL SECTIONS