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RESEARCH MEMORANDUM

PRESSURE-DISTRIBUTION MEASUREMENTS OVER AN EXTENSIBLE
LEADING-EDGE FLAP ON TWO WINGS HAVING LEADING-
EDGE SWEEP OF 42° AND 52°

By

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

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RESEARCH MEMORANDUM

PRESSURE-DISTRIBUTION MEASUREMENTS OVER AN EXTENSIBLE
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EDGE SWEEP OF 42° AND 52°

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SUMMARY

An investigation of the pressure distribution over a leading-edge flap was conducted in the Langley 19-foot pressure tunnel. The tests were made on 42° and 52° sweptback wings of NACA 64₁-112 sections, the 42° wing being used in conjunction with a circular cross-section fuselage in a high-wing combination. The pressure-distribution data for the 52° sweptback wing were obtained at various angles of attack and angles of yaw for a Reynolds number of 4.4×10^6 and a Mach number of 0.08, for both split flaps deflected and neutral configurations with upper-surface fences installed. The 42° sweptback wing was tested at zero yaw for various angles of attack at a Reynolds number of 5.12×10^6 and a Mach number of 0.11 with split flaps deflected.

The pressure-distribution measurements over the leading-edge flap indicated that the rate of increase of the flap normal-force coefficient C_{N_f} with lift coefficient was nearly constant for the conditions tested, but the hinge-moment coefficient C_{h_f} increased with lift coefficient at an increasing rate. The maximum values of C_{N_f} and C_{h_f} obtained at zero yaw with the split flaps deflected were 3.24 and 1.62, respectively, for the 42° sweptback wing, and 3.12 and 1.68, respectively, for the 52° sweptback wing. The maximum values of C_{N_f} and C_{h_f} were lower when the split flaps were neutral. Yawing the 52° sweptback wing increased the maximum values of C_{N_f} and C_{h_f} on the leading wing panel and caused a decrease for the trailing wing panel.

INTRODUCTION

Experimental investigations such as references 1 and 2 have shown that extensible leading-edge flaps can increase the maximum lift and improve the stability characteristics of sweptback wings. Practical application of such flaps on any aircraft would require some knowledge

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of the magnitudes of the aerodynamic forces on the flaps. Pressure-distribution measurements were therefore made over a leading-edge flap on two wings of 42° and 52° sweepback. The 42° sweptback wing had NACA 641-112 airfoil sections normal to the 0.273 chord line and the 52° sweptback wing had the same sections normal to the 0.282 chord line. (The 0.273 and 0.282 chord lines correspond to the 0.25 chord line of a similar wing panel which has zero sweep at the 0.25 chord line.) Both wings had taper ratios of 0.625 but differed in aspect ratio, the 42° wing having an aspect ratio of 4.01 and the 52° wing an aspect ratio of 2.88. In addition to showing the effect of yaw on the leading-edge flap loads, the results at various yaw angles might be used in the prediction of leading-edge flap loads on wings of different sweepback. The original data obtained on the 42° swept wing, previously published in reference 3, have also been included in this report for purposes of comparison.

COEFFICIENTS AND SYMBOLS

α	angle of attack of wing chord line measured in a plane parallel to the plane of symmetry
Λ	sweep angle of wing leading edge
ψ	angle of yaw, positive when right wing is back
C_L	lift coefficient (Lift/ qS)
C_D	drag coefficient (Drag/ qS)
C_m	pitching-moment coefficient referred to quarter-chord point of mean aerodynamic chord (Moment/ $qS\bar{c}$)
c_{n_f}	section normal-force coefficient of leading-edge flap $\left(\int P d\left(\frac{x}{c_f}\right) \right)$
C_{N_f}	normal-force coefficient of complete leading-edge flap $\left(\int c_{n_f} d\left(\frac{y_f}{b_f}\right) \right)$
c_{h_f}	section hinge-moment coefficient of leading-edge flap about trailing edge of flap $\left(\int P \frac{x}{c_f} d\left(\frac{x}{c_f}\right) \right)$
C_{h_f}	hinge-moment coefficient of complete leading-edge flap about trailing edge of flap $\left(\int c_{h_f} d\left(\frac{y_f}{b_f}\right) \right)$

c.p.	center of pressure of leading-edge flap in percent of flap chord measured from leading edge of flap $\left(100\left(1 - \frac{c_{h_f}}{c_{n_f}}\right)\right)$
R	Reynolds number (based on wing mean aerodynamic chord)
S	wing area
q	free-stream dynamic pressure
\bar{c}	wing mean aerodynamic chord
c'	wing chord normal to 0.273 chord line of 42° wing and 0.282 chord line of 52° wing
p	local static pressure
P	pressure coefficient $\left(\frac{p - p_0}{q}\right)$
p ₀	free-stream static pressure
y _f	distance measured along span of leading-edge flap from inboard end
b _f	span of leading-edge flap
x	distance measured along leading-edge-flap chord from trailing edge of flap (perpendicular to leading edge of flap)
c _f	leading-edge-flap chord measured perpendicular to flap leading edge.

MODELS

The models used in the present tests had been previously used in the investigations reported in references 1 and 2, in which they are described in detail. Figure 1 shows the location of the leading-edge flaps on each of the models and also the geometric parameters of the models. The 42° sweptback wing had NACA 64₁-112 airfoil sections normal to the 0.273 chord line, and the 52° wing had the same sections normal to the 0.282 chord line. (The 0.273 and 0.282 chord lines correspond to the 0.25 chord line of a similar wing panel which has zero sweep at

the 0.25 chord line.) The 42° wing had an aspect ratio of 4.01 and the 52° wing, 2.88. Both wings had taper ratios of 0.625. The 42° wing was tested with half-span split flaps and in combination with a circular cross-section fuselage of fineness ratio 10 to 1. The 52° sweptback wing was also tested with half-span split flaps and an upper-surface fence, which made the wing longitudinally stable. The fence was located 45 percent of the semispan from the plane of symmetry, had a constant height of 6 percent of the local airfoil chord, and extended over the rear 95 percent of the chord. The same leading-edge flaps were used on both models with equal deflection angles of 50° from the wing chord plane being maintained. The geometry of the flaps and the location of the orifices, which were on the right-hand flap only, are given in figure 2. Figure 3 shows photographs of the 42° and 52° sweptback wings in the Langley 19-foot pressure tunnel.

TESTS

The tests were made in the Langley 19-foot pressure tunnel with the air compressed to approximately 33 pounds per square inch absolute. The flap pressures were recorded photographically from a multiple-tube manometer. The 52° sweptback wing was mounted on the single-support system, and the data were obtained at yaw angles of 10° , 0° , -10° , and -20° at various angles of attack for the split flaps both deflected and neutral. The 42° sweptback wing was mounted on a two-support system and was tested in conjunction with a fuselage, forming a high-wing combination. The flap pressure data were obtained at zero yaw for various angles of attack for the split-flaps-deflected configuration only.

The pressure data for the 52° wing were obtained at a Reynolds number of 4.4×10^6 with a corresponding Mach number of 0.08, and the tests on the 42° wing were made at a Reynolds number of 5.12×10^6 with a corresponding Mach number of 0.14.

The force data presented were obtained with all connector tubing removed. The usual wind-tunnel corrections (the same as in references 1 and 2) were applied.

RESULTS AND DISCUSSION

The lift, drag, and pitching-moment characteristics of both the 42° and 52° sweptback wings are presented in figure 4. The chordwise pressure distributions for each of the five spanwise stations along the leading-edge flap are given in figure 5 for the 42° sweptback wing and in figures 6 and 7 for the 52° sweptback wing. The dotted portions of some of the curves of figure 5 are interpolations based on the existing data, as no data were obtained for the lower surface of the flap at those

sections. Pressure coefficients obtained along the lower surface were faired point to point and no attempt to reach stagnation pressure was made since the effect on the force coefficient would be negligible.

Figures 5 to 7 show that at the lowest angles of attack the flap loads for both the 42° and 52° sweptback wings were small and were concentrated near the trailing edge of the flap. As the angle of attack was increased, large negative pressure peaks developed at the leading edge of the flaps, accompanied by a forward movement in the center of pressure. A maximum value of negative pressure coefficient measured was -10.75 for the 52° sweptback wing at the 72-percent flap span station (measured outboard from the inboard end of the flap) at an angle of attack of 25.2° and zero yaw with the split flaps deflected. It is believed that the fence would not appreciably affect the pressures on the leading-edge flap. It is also believed that the pressure distribution over the fuselage on the 42° sweptback wing has a negligible effect on the pressures over the leading-edge flap since the model was tested at zero yaw.

The spanwise variations of the flap normal-force and hinge-moment coefficients for the 42° and 52° sweptback wings are presented in figures 8 to 10. The supporting members of a flap are usually designed for the maximum load obtainable on the flap, and, therefore, the span-load distribution in the higher angle-of-attack range will be of most interest. Figures 8 to 10 indicate that the loading was, in general, well distributed over the flap and that the maximum loading was near the center of the flap, being shifted slightly toward the inboard end by negative yaw (which tends to decrease the sweep angle) and toward the outboard side for positive yaw. The spanwise center of loading was determined for the configurations at zero yaw and was found to vary from about 45 percent of the flap span at low angles of attack to about 50 percent in the high angle-of-attack range.

The normal-force and hinge-moment coefficients of the leading-edge flap are presented as functions of the lift coefficient in figure 11. The lift coefficients for the right wing panel of the 52° sweptback wing were estimated for various angles of yaw from the following empirical relationship:

$$C_{L(\text{yawed panel})} = C_{L(\psi=0)} \left[\frac{\cos(\Lambda + \psi)}{\cos \Lambda} \right]^{0.92}$$

This relationship gave satisfactory agreement when used to check values of lift coefficient obtained from pressure-distribution measurements of a 45° sweptback wing at various angles of yaw (reference 4) and was considered sufficiently accurate (see fig. 12) for presenting the flap load data.

Figure 11 shows that in general the rate of increase of the normal-force coefficient with lift coefficient was nearly constant for the conditions tested.

The figure also shows that the hinge-moment coefficient increased with lift coefficient but at an increasing rate. The hinge-moment curves were nearly parallel, however, for identical trailing-edge-flap configurations. In general, deflecting the split flaps resulted in a marked decrease in the leading-edge flap normal-force and hinge-moment coefficients at constant lift coefficient. Since the split flaps were of partial-span, the lift at the inboard end of the wing would be increased, whereas the outboard half would be carrying a smaller load, thereby reducing the leading-edge flap loads. The inboard end of the leading-edge flap may also be affected by the change in chordwise pressure distribution, due to the split flap.

For any given lift coefficient, the values of the normal-force and hinge-moment coefficients varied considerably with sweepback and yaw. Figure 11 shows that an increase of about 0.30 and 0.32 occurred for C_{N_F} and C_{h_F} , respectively, for an increase in sweepback from 42° to 52° . The coefficients C_{N_F} and C_{h_F} also increased with angle of yaw, for constant lift coefficient, in the range of yaw angles from $\psi = -20^\circ$ to $\psi = 10^\circ$, so that the incremental increase for each 10° of yaw was greater.

The effects due to yaw or sweepback can be explained quite readily when it is realized that the forces on the flap and the lift of the wing are proportional to the dynamic pressure normal to the leading edge. The rate of increase of the flap normal force with lift coefficient would, therefore, be constant, regardless of the sweep angle if all secondary effects, such as cross flow over the wing, are neglected. However, the curves showing the variation of the flap normal force with lift would be displaced by an amount depending on the sweep angle. The flap was deflected down 50° , and a negative force proportional to the dynamic pressure normal to the leading edge acted upon it when the wing was at zero lift. Since the angle of attack for zero lift is unaffected by sweep, the initial force at zero lift would be changed by varying the sweep angle.

The maximum values of the normal-force and hinge-moment coefficients obtained on the 52° wing varied with angle of yaw and trailing-edge-flap deflection. Figure 13 shows that the maximum values of C_{N_F} and C_{h_F} at zero yaw were about 3.12 and 1.68, respectively, for the split-flaps-deflected configuration and that a decrease of about 0.65 in C_{N_F} and 0.30 in C_{h_F} occurred for 10° of yaw. For the negative yaw angles, an increase of approximately 0.34 in C_{N_F} and 0.05 in C_{h_F} occurred for -20° of yaw. At all angles of yaw, the maximum values of C_{N_F} and C_{h_F} were lower when the split flaps were neutral.

When the 52° sweptback wing was yawed -10° , the angle of sweepback of the right wing panel was equal to the sweepback of the 42° sweptback wing at zero yaw. From figure 11 it can be seen that the curves of C_{N_F} and C_{h_F} against C_L for the two conditions are in fairly good agreement

when the differences in the two model configurations are considered. If the data for the 42° wing had been obtained without a fuselage, it is expected that the agreement would be better, since the lift coefficients would be increased. The maximum values of C_{N_f} and C_{h_f} for the 42° wing were 3.24 and 1.62, which were about equal to the values for the 52° wing at -10° yaw.

The maximum load on a similar flap on any sweptback wing could probably be estimated from the data contained herein. However, particular attention should be given to any devices, such as trailing-edge flaps, which affect the spanwise loading of the wing and also to the original deflection angle of the flap (with respect to the wing chord line), the effect of which was not isolated in this report.

CONCLUSIONS

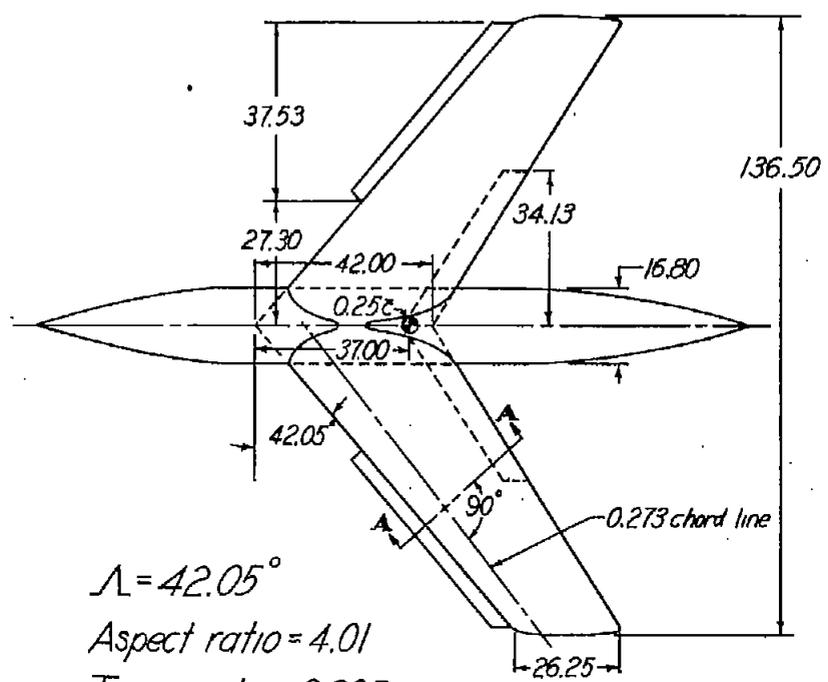
An investigation of the pressure distribution over an extended leading-edge flap on wings of 42° and 52° sweepback indicated that:

1. The rate of increase of the normal-force coefficient C_{N_f} with lift coefficient was nearly constant for the conditions tested, but the hinge-moment coefficient C_{h_f} increased with lift coefficient at an increasing rate.
2. The maximum values of C_{N_f} and C_{h_f} obtained at zero yaw with the split flaps deflected were 3.24 and 1.62, respectively, for the 42° sweptback wing, and 3.12 and 1.68, respectively, for the 52° sweptback wing.
3. The maximum values of the normal-force and hinge-moment coefficients for the leading-edge flap were lower when the split flaps were neutral.
4. Yawing the 52° sweptback wing increased the maximum values of C_{N_f} and C_{h_f} on the leading wing panel and caused a decrease for the trailing wing panel.

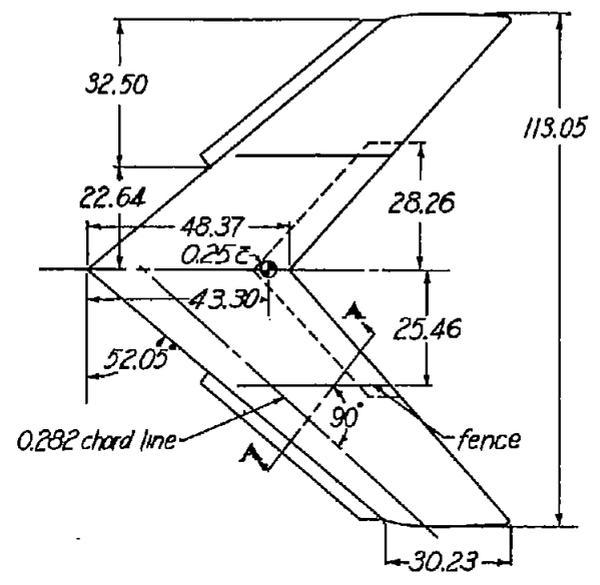
Langley Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Air Force Base, Va.

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2. Graham, Robert R., and Conner, D. William: Investigation of High-Lift and Stall-Control Devices on an NACA 64-Series 42° Sweptback Wing with and without Fuselage. NACA RM No. L7G09, 1947.
3. Conner, D. William, and Foster, Gerald V.: Investigation of Pressure Distribution over an Extended Leading-Edge Flap on a 42° Sweptback Wing. NACA RM No. L7J03, 1947.
4. Jacobs, W.: Druckverteilungsmessungen an zwei Pfeilflügeln konstanter Tiefe ($\varphi = 30^\circ, 45^\circ$) bei unsymmetrischer Anströmung. UM Nr. 2110, Deutsche Luftfahrtforschung (Braunschweig), 1944.



$\Lambda = 42.05^\circ$
 Aspect ratio = 4.01
 Taper ratio = 0.625
 M.A.C. = 34.7 in.



$\Lambda = 52.05^\circ$
 Aspect ratio = 2.88
 Taper ratio = 0.625
 M.A.C. = 39.97 in.

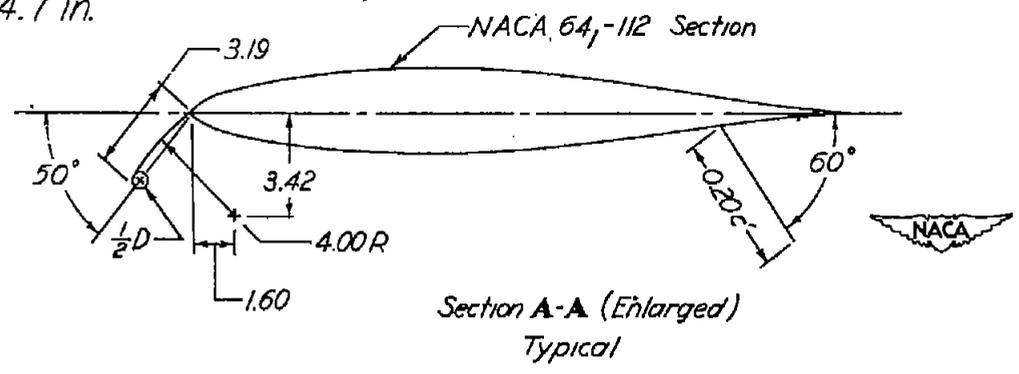


Figure 1.- Geometry of 42° and 52° sweptback wing models. All dimensions in inches.

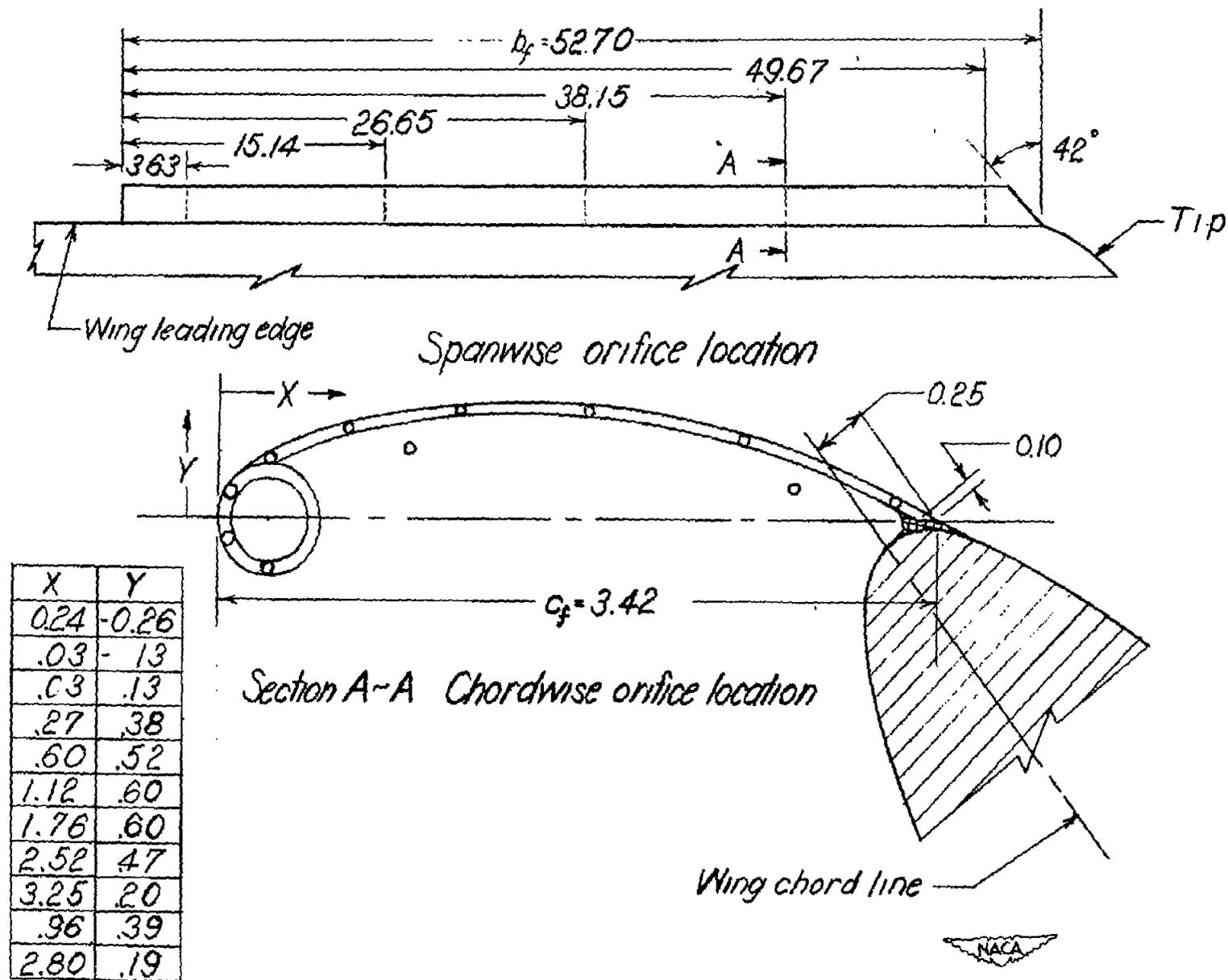
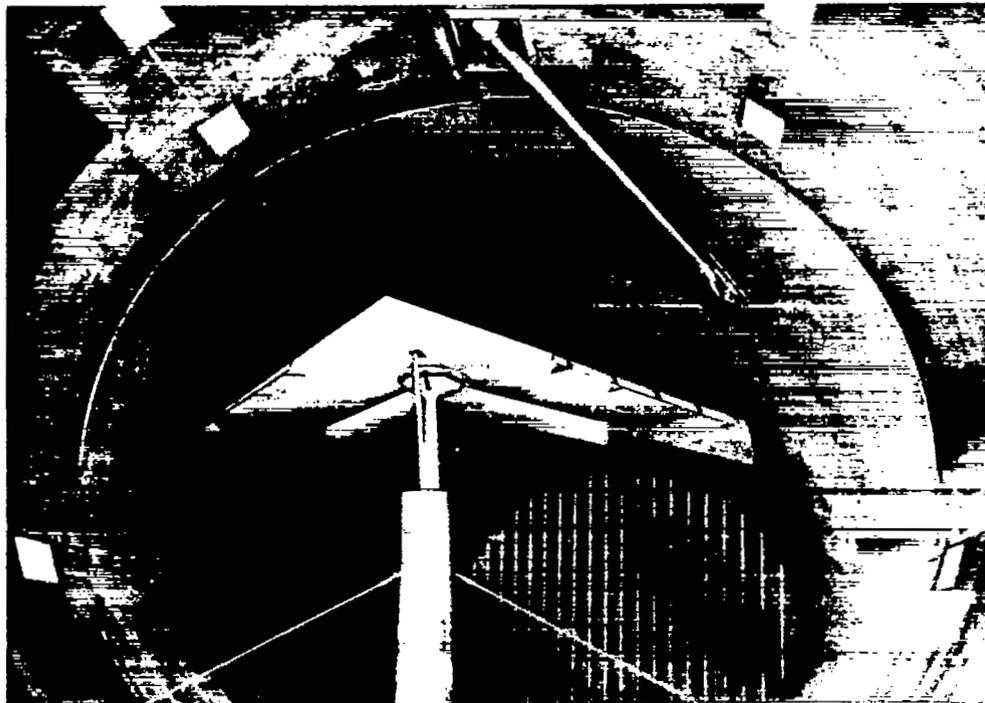


Figure 2.— Location of orifices on leading-edge flap. (All dimensions in inches.)



(a) 42° sweptback wing.



(b) 52° sweptback wing.

Figure 3.— Photographs showing the 42° and 52° sweptback wings.

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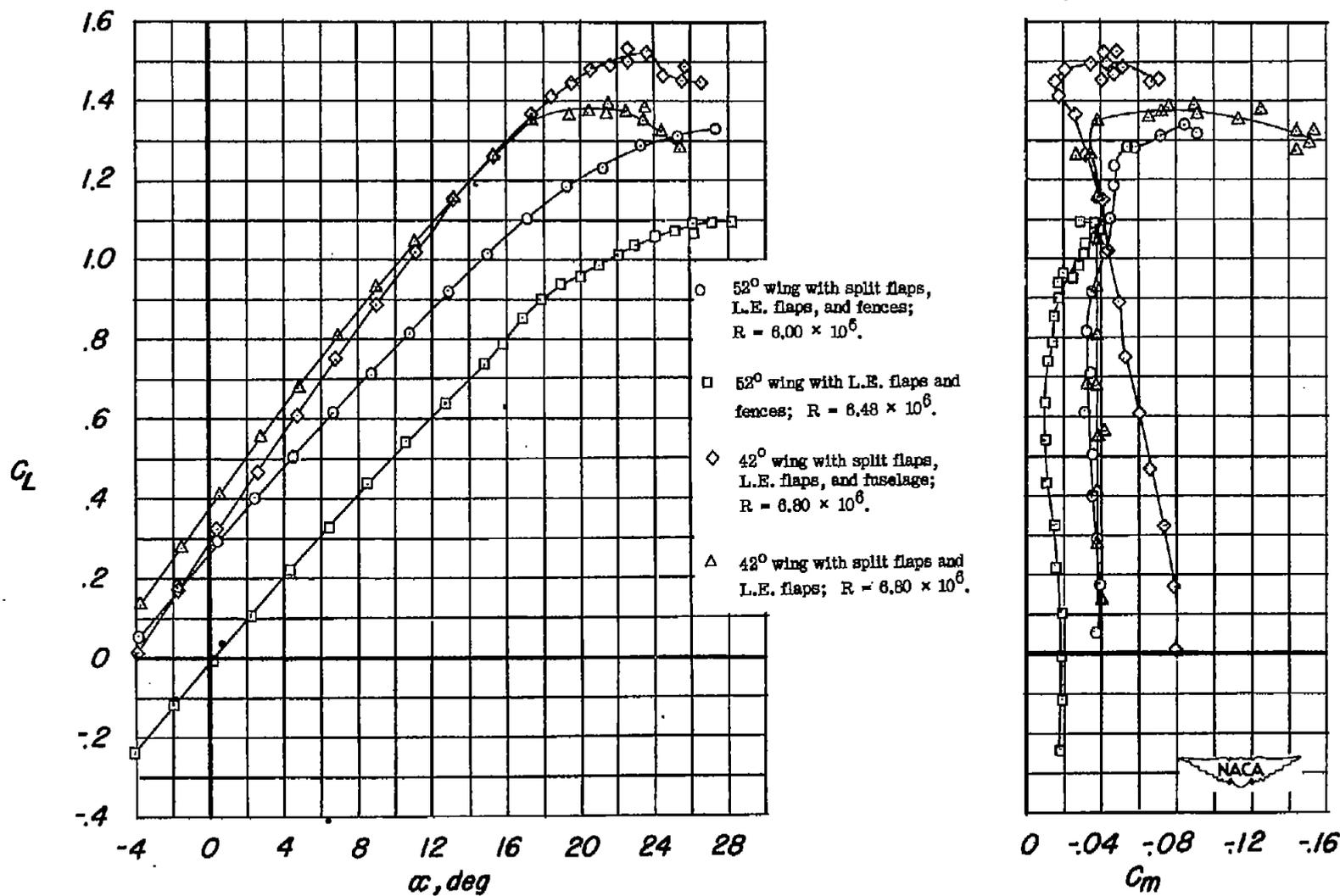
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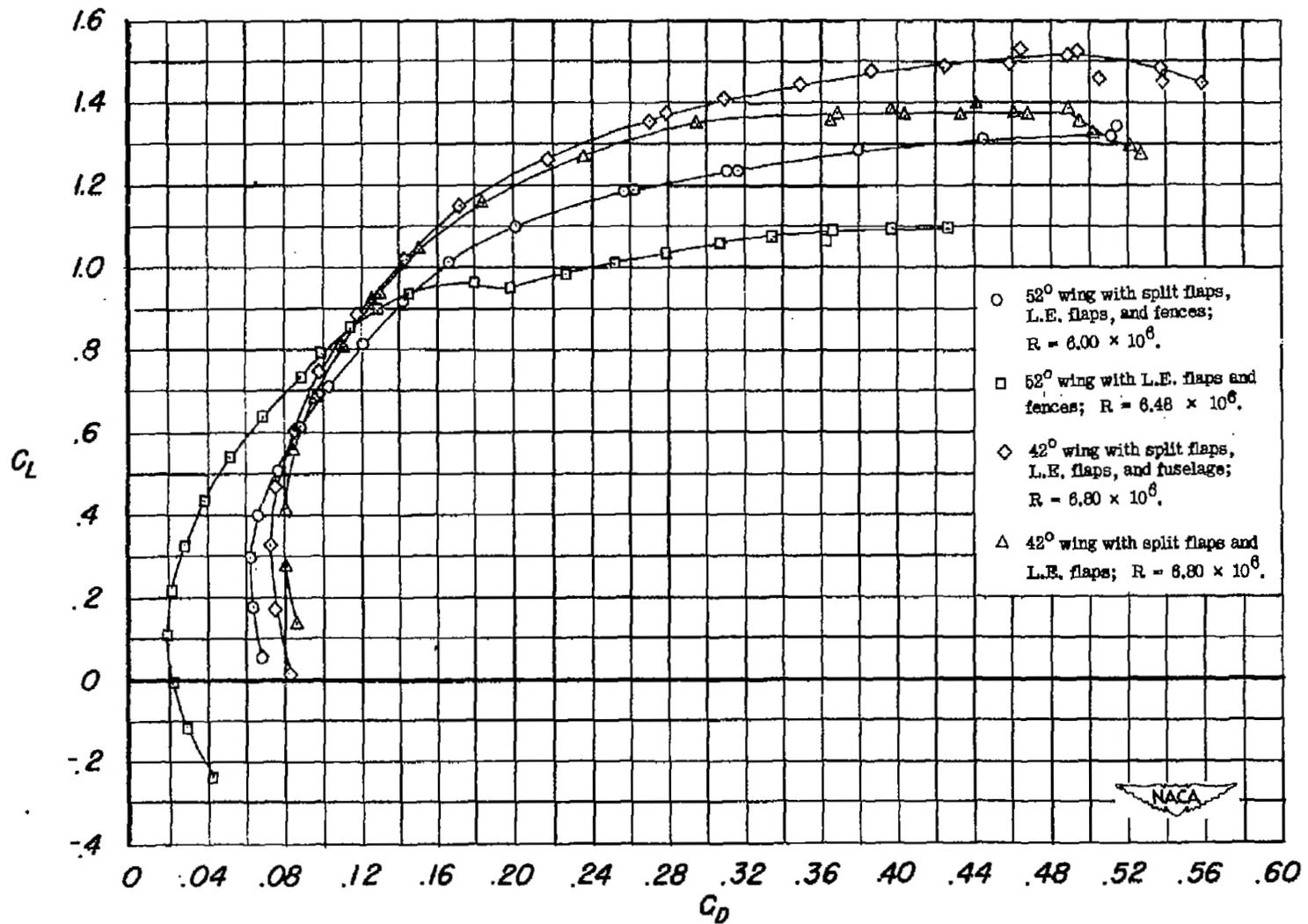
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(a) C_L against α and C_m .

Figure 4.- Aerodynamic characteristics of the 42° and 52° sweptback wing models.



(b) C_L against C_D .

Figure 4.— Concluded.

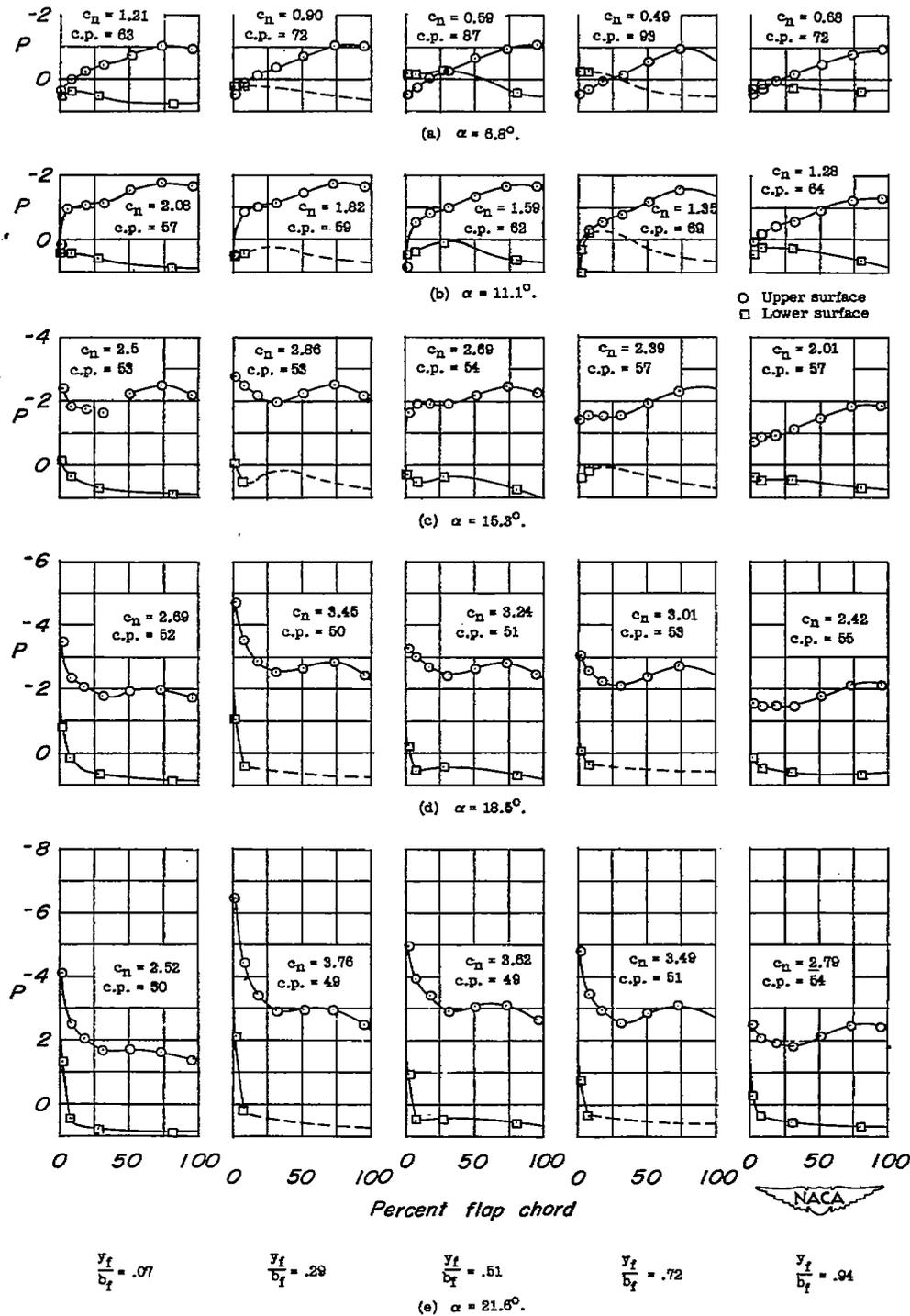


Figure 5.— Chordwise pressure distribution over the leading-edge flap on the 42° sweptback wing at five spanwise stations. Split flaps deflected; $R = 5.12 \times 10^6$.

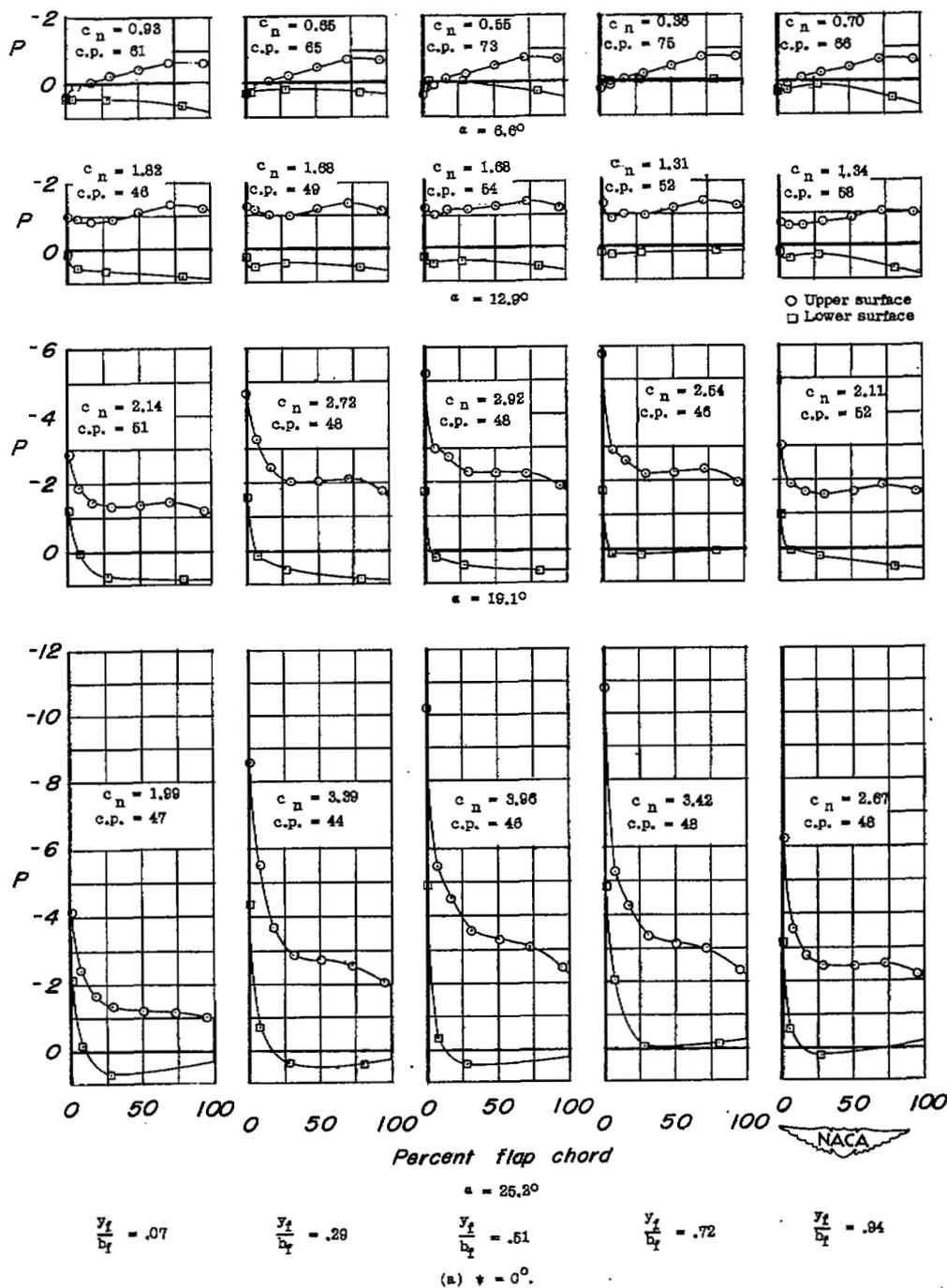


Figure 6.— Chordwise pressure distribution over the leading-edge flap on the 52° sweptback wing at five spanwise stations. Split flaps deflected and fences installed; $R = 4.4 \times 10^6$.

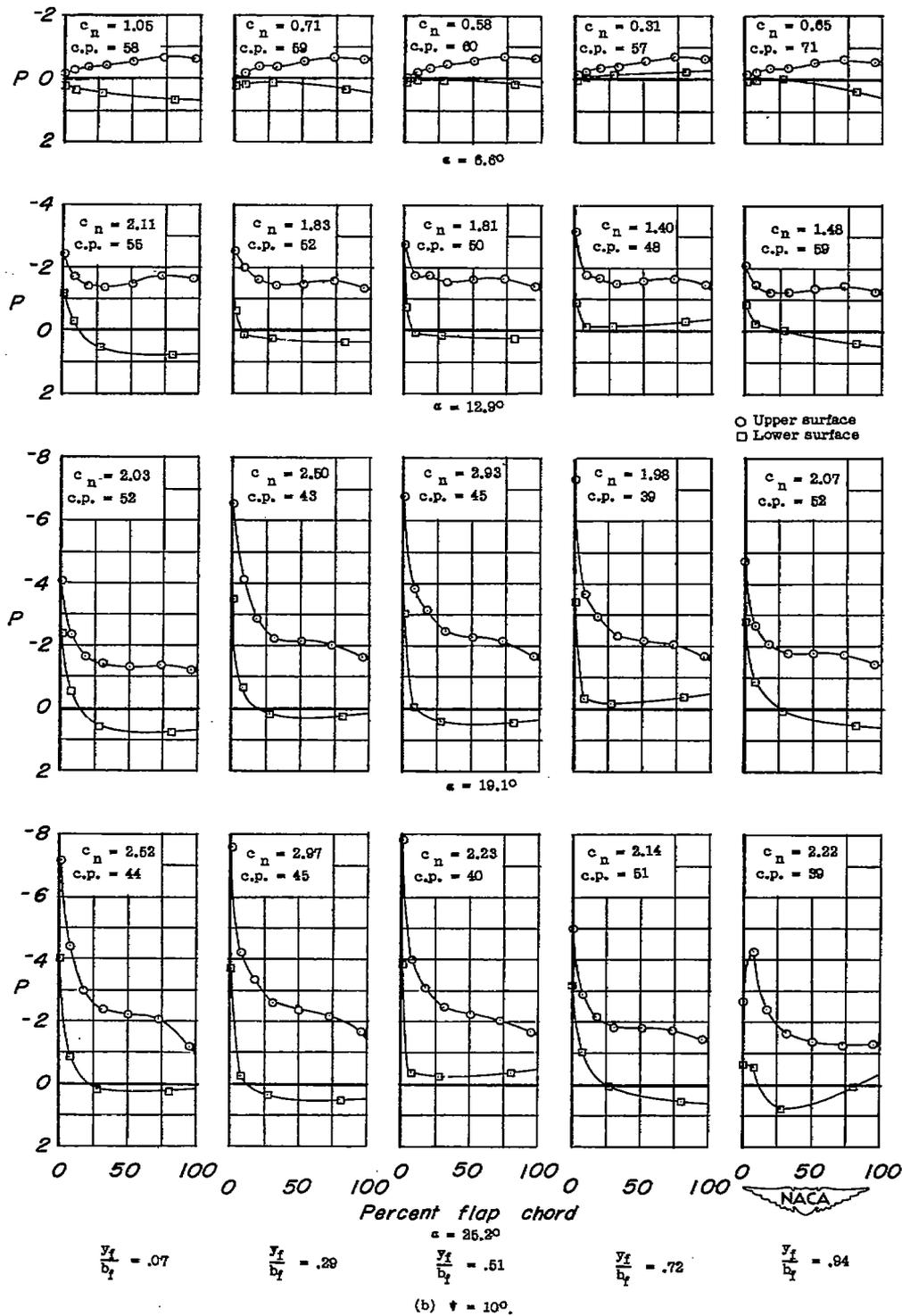


Figure 6.- Continued.

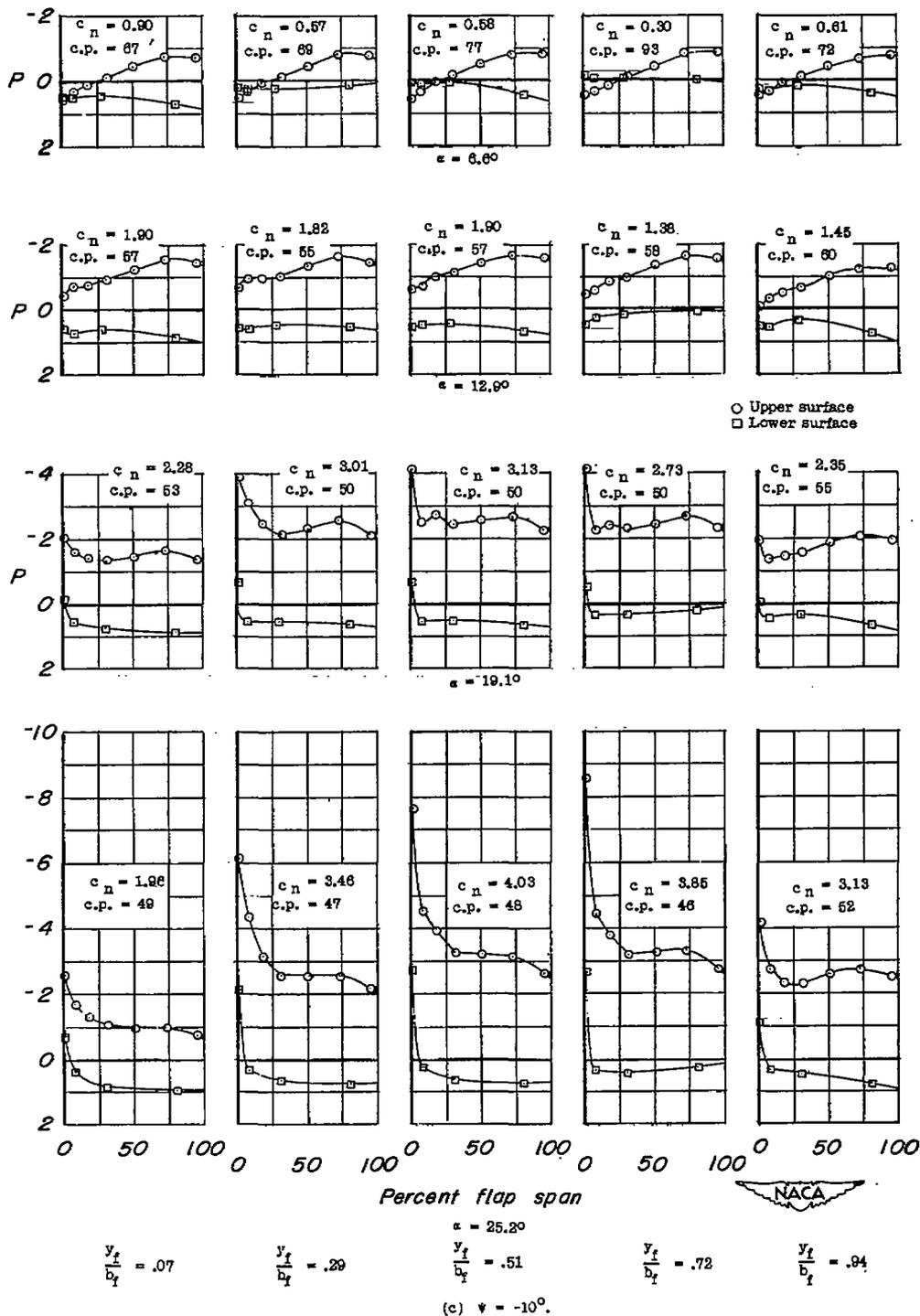


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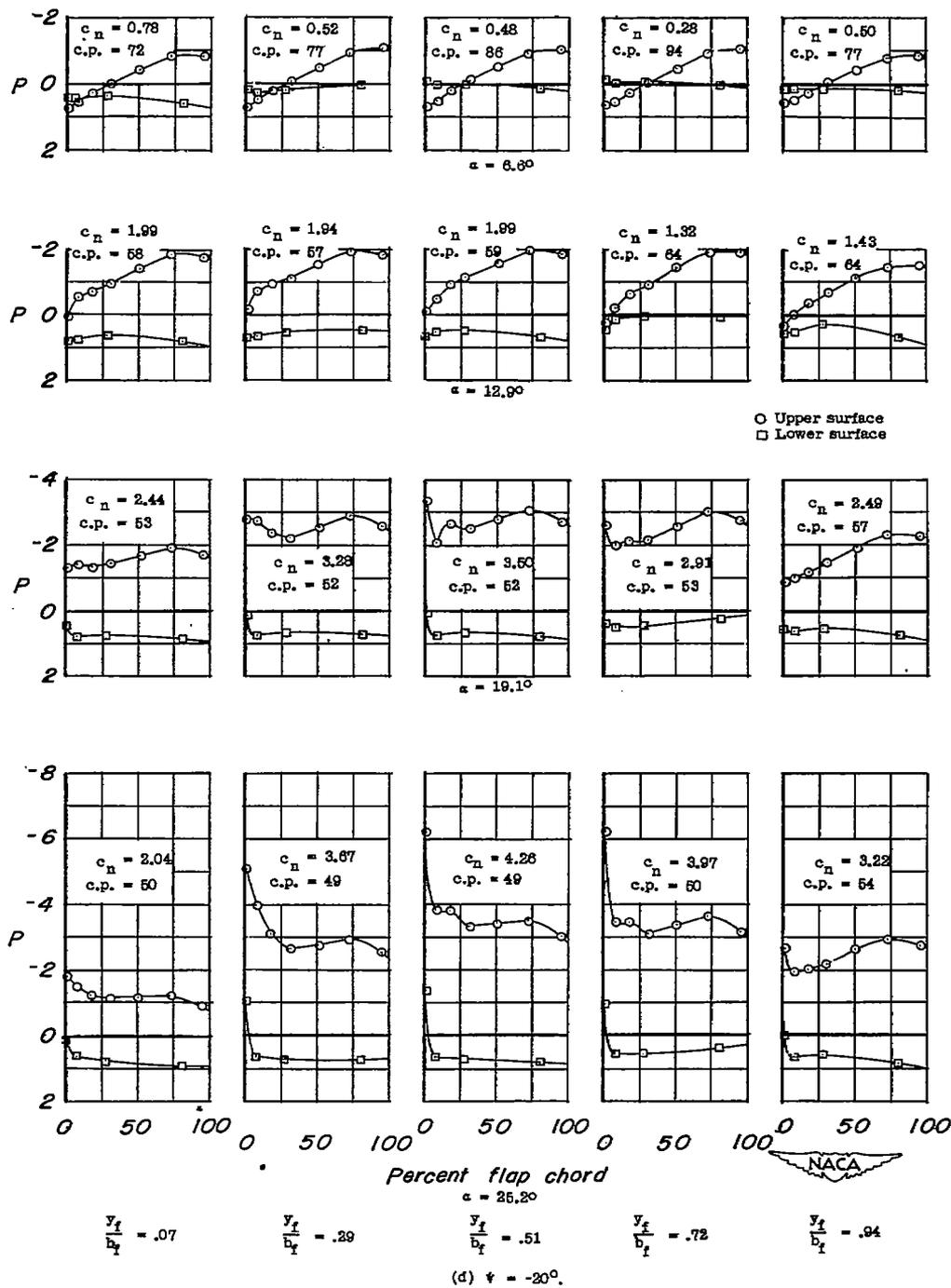


Figure 6.— Concluded.

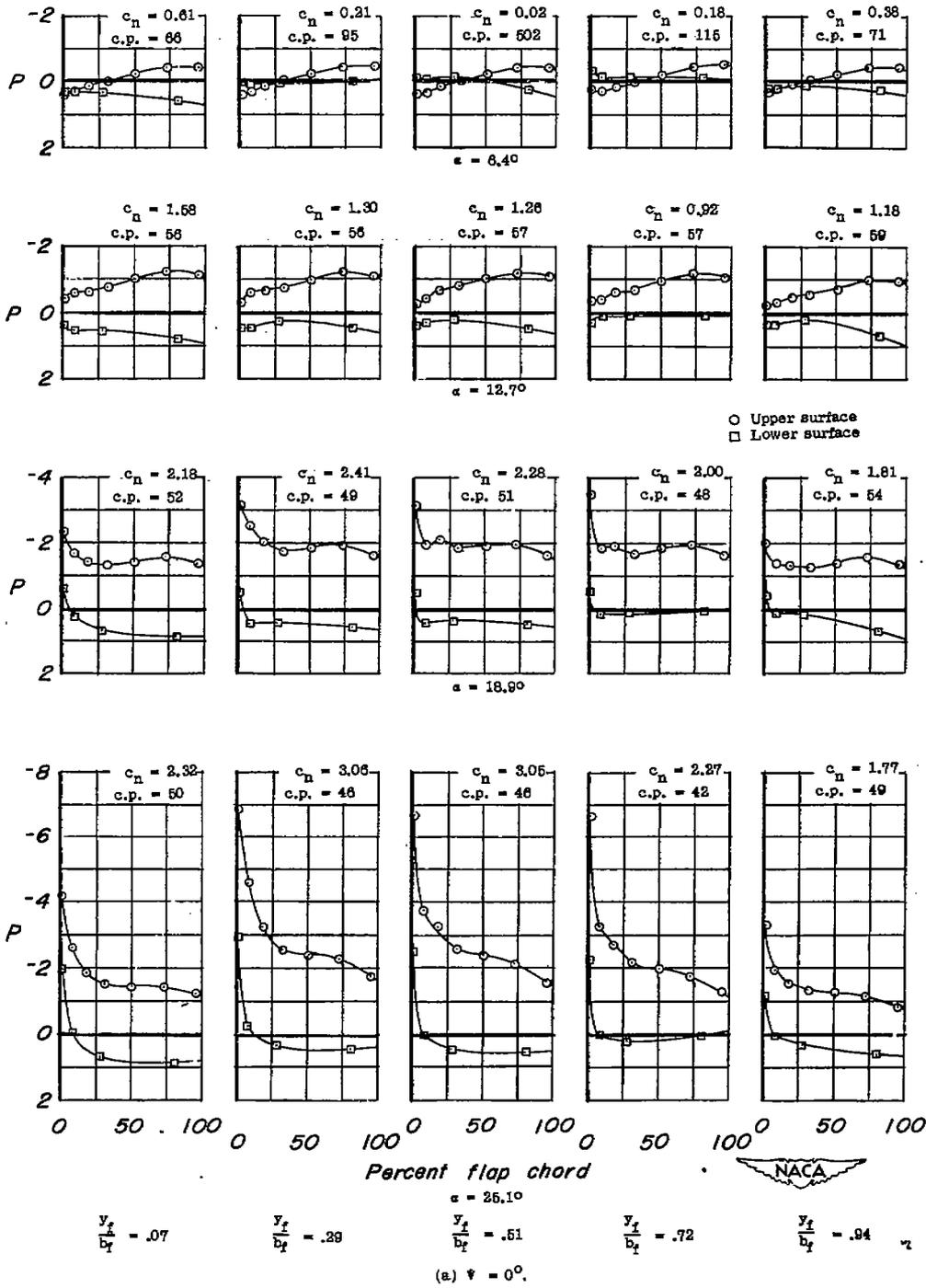


Figure 7.— Chordwise pressure distribution over the leading-edge flap on the 52° sweptback wing at five spanwise stations. Split flaps neutral, fences installed; $R = 4.4 \times 10^6$.

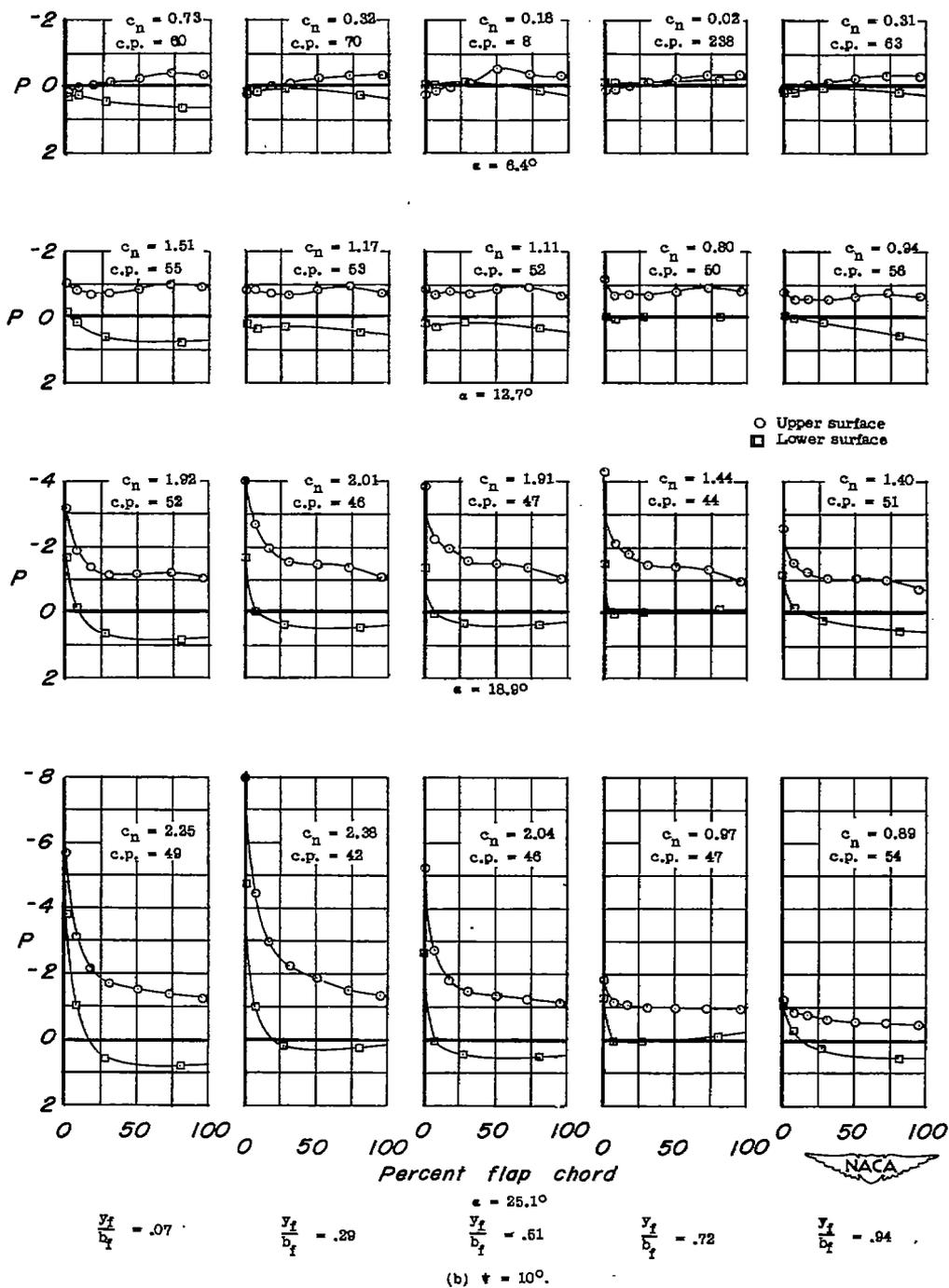
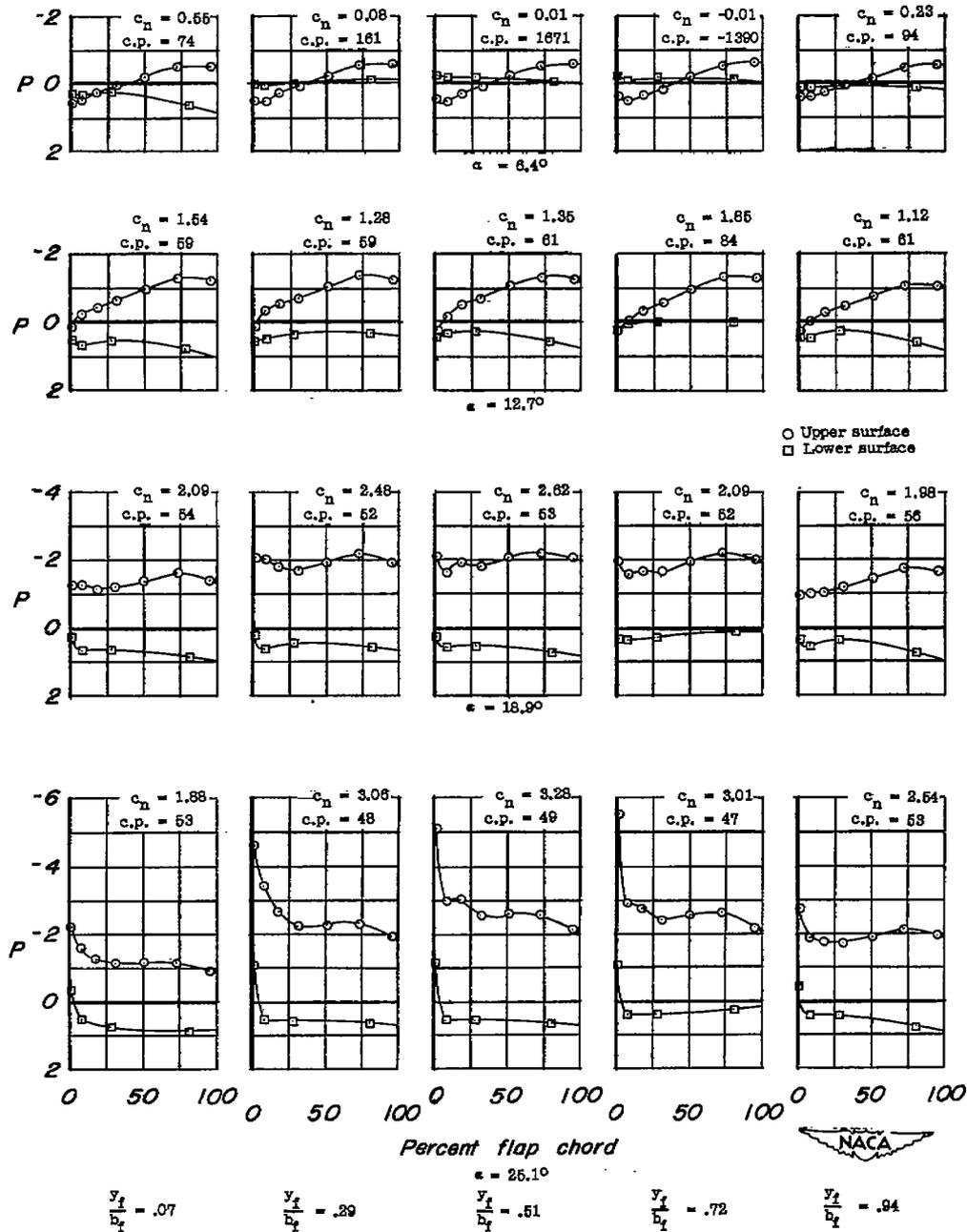


Figure 7.-- Continued.



(c) $\alpha = -10^\circ$.

Figure 7.- Continued.

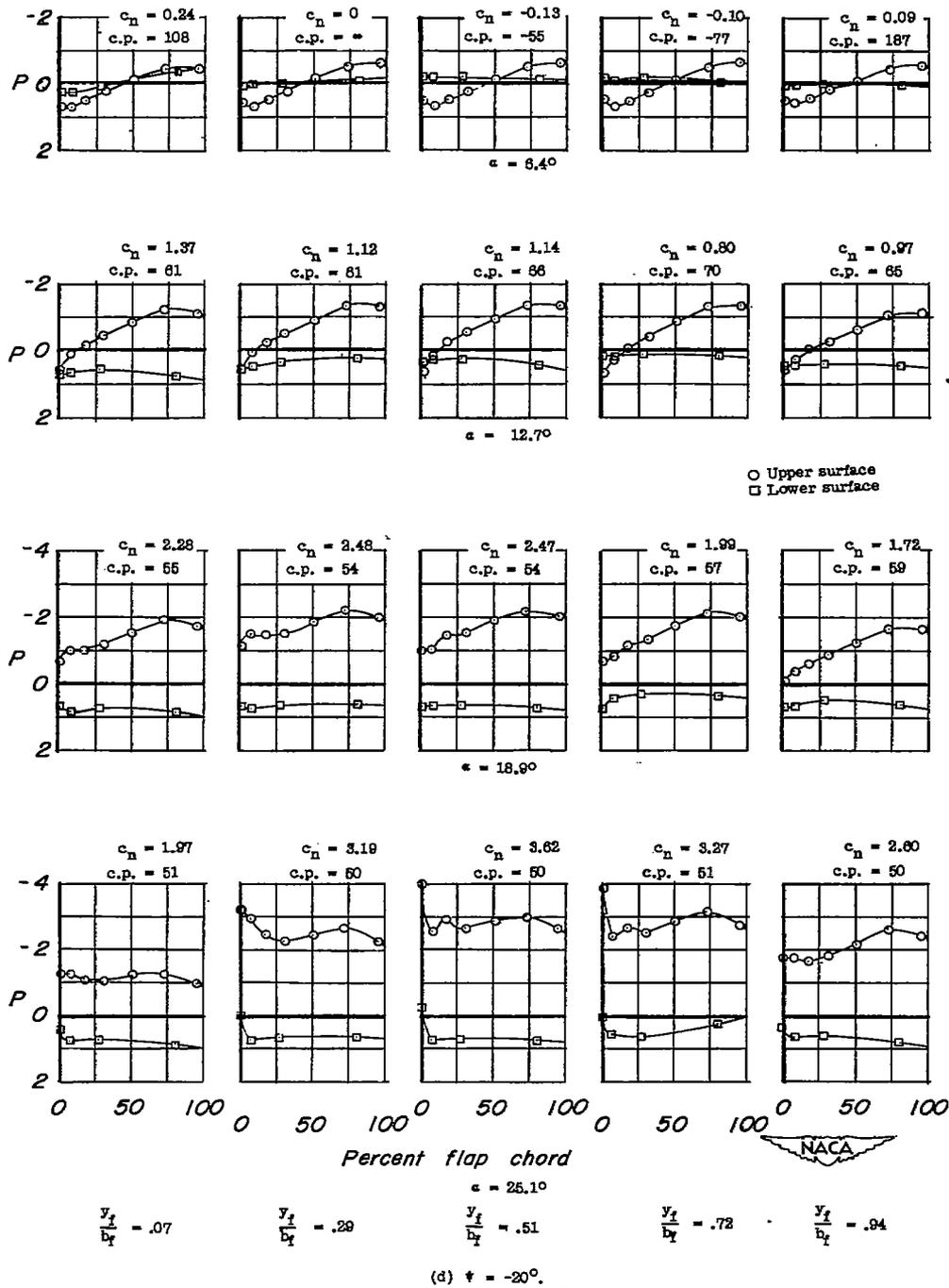


Figure 7.-- Concluded.

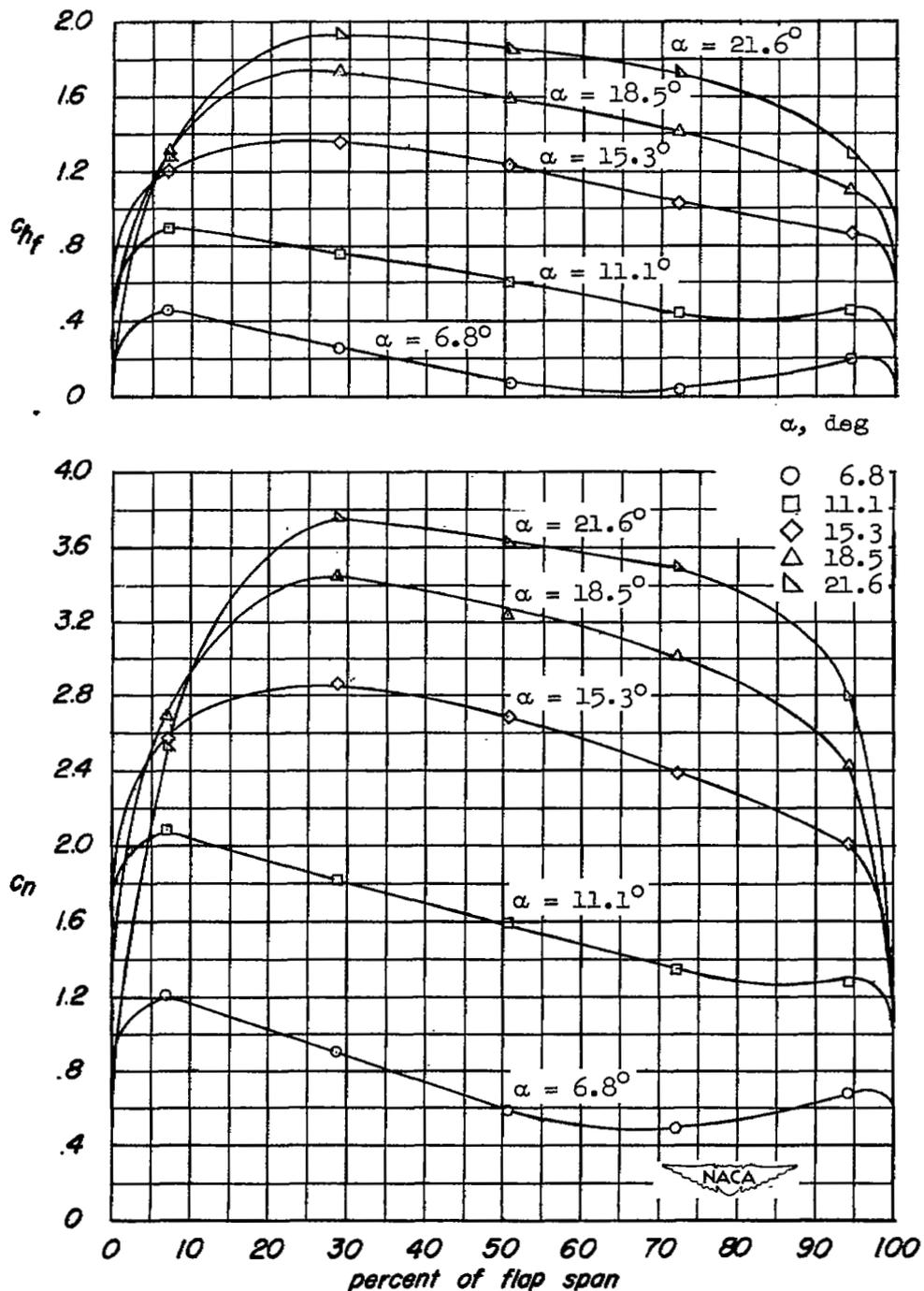


Figure 8.— Variation of the normal-force and hinge-moment coefficients across the leading-edge flap span for the 42° sweptback wing. Split flaps deflected; $R = 5.12 \times 10^6$.

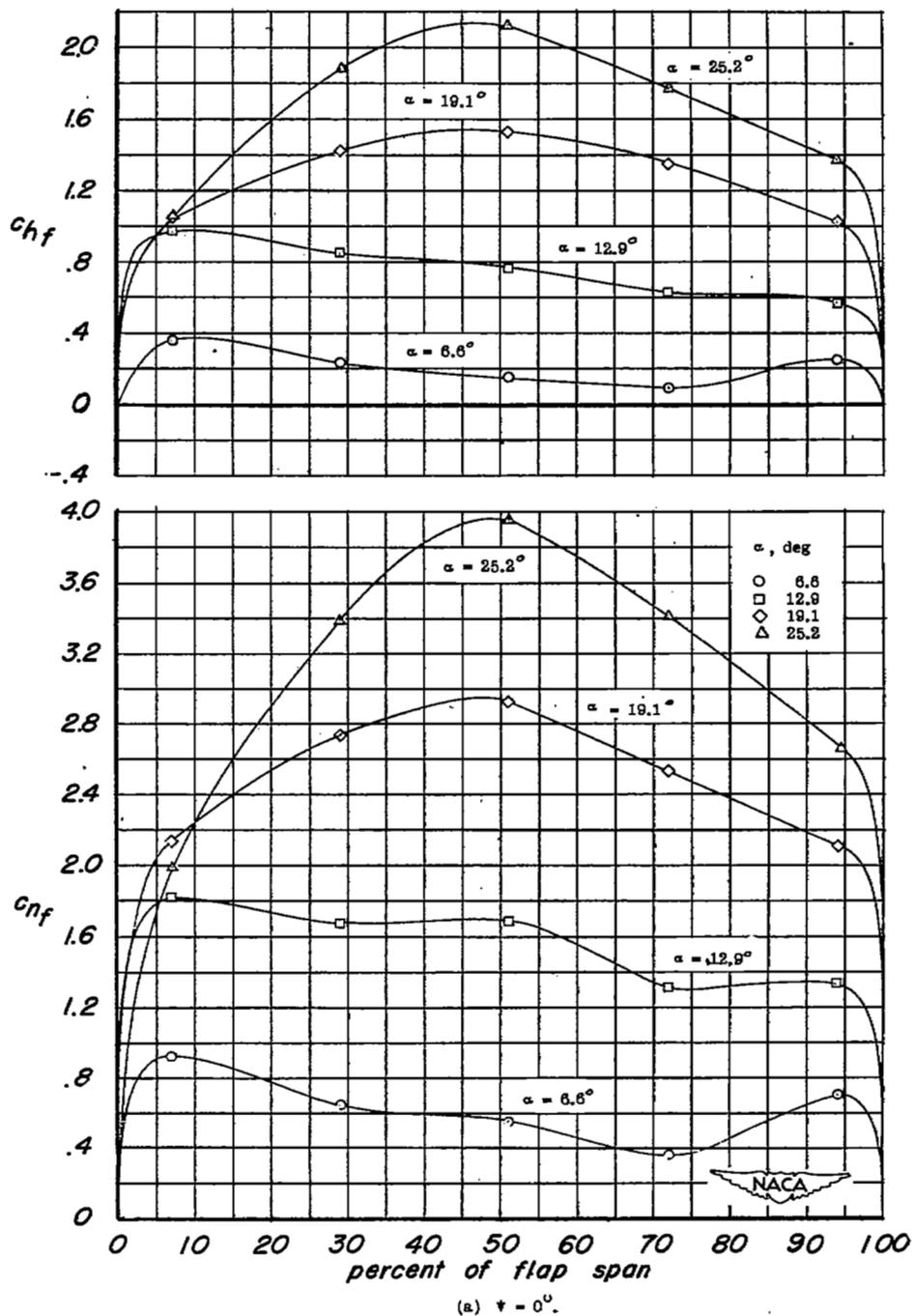


Figure 9.— Variation of the normal-force and hinge-moment coefficients across the leading-edge flap span for the 52° sweptback wing. Split flaps deflected and fences installed; $R = 4.4 \times 10^6$.

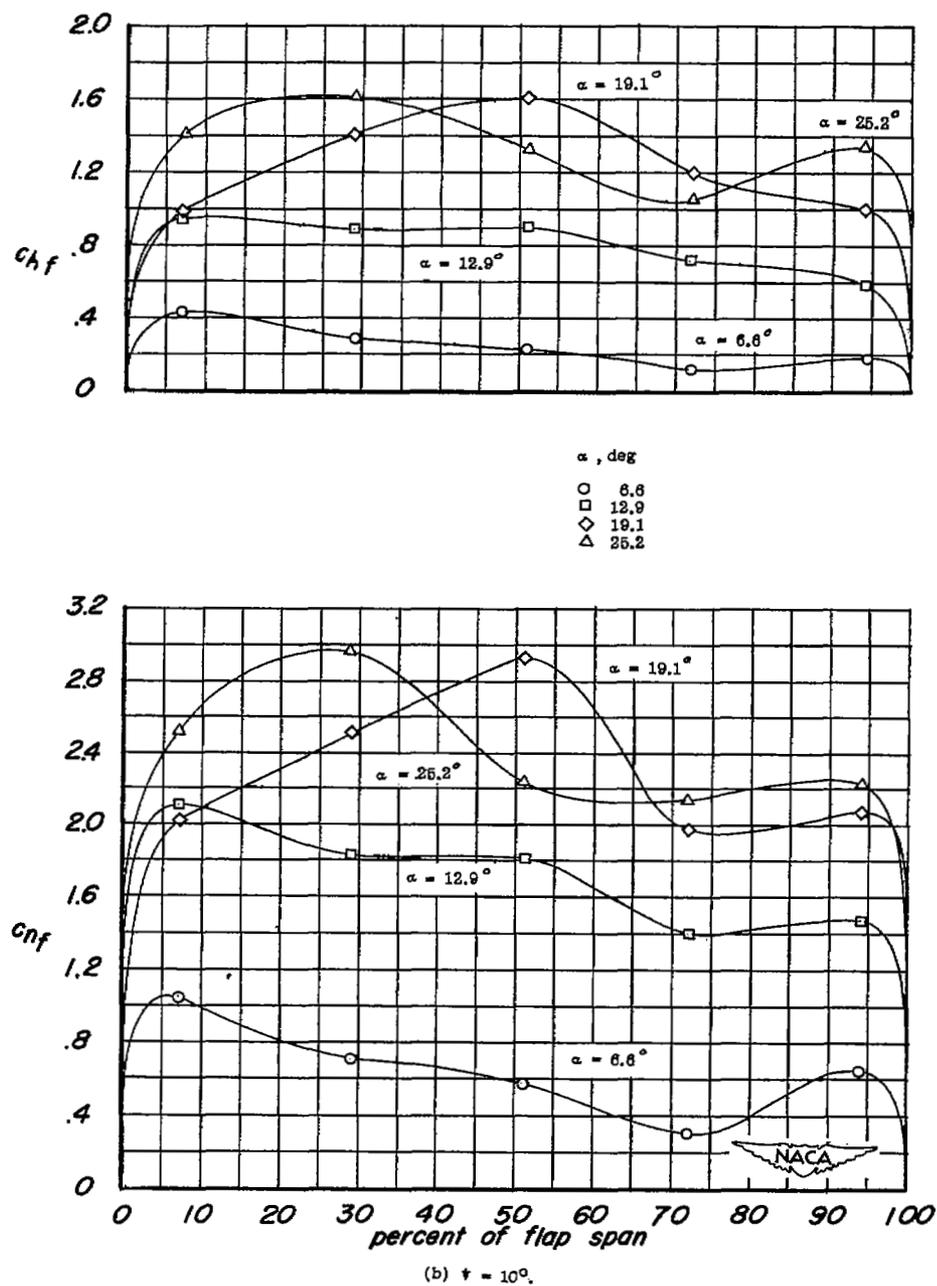


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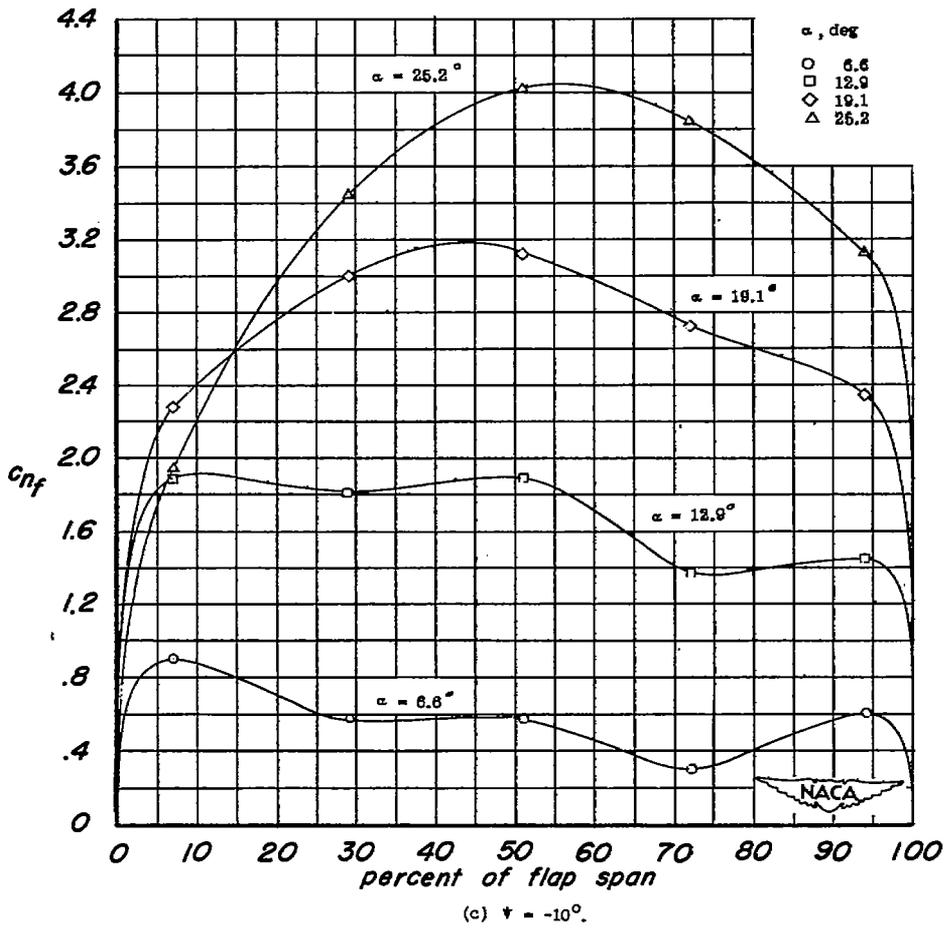
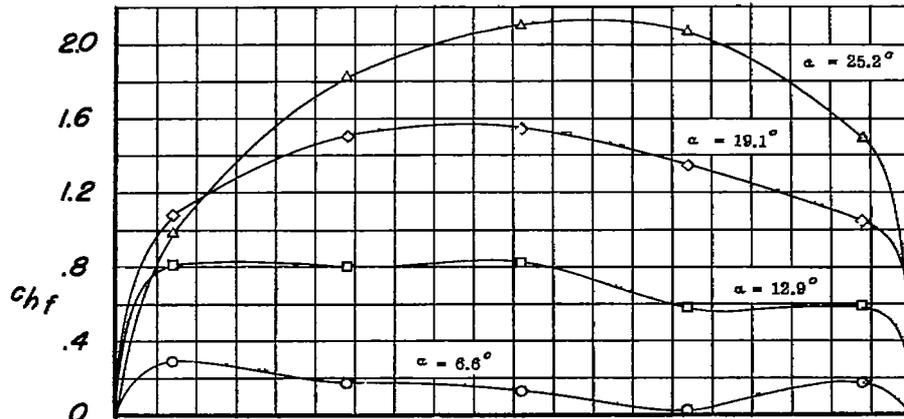


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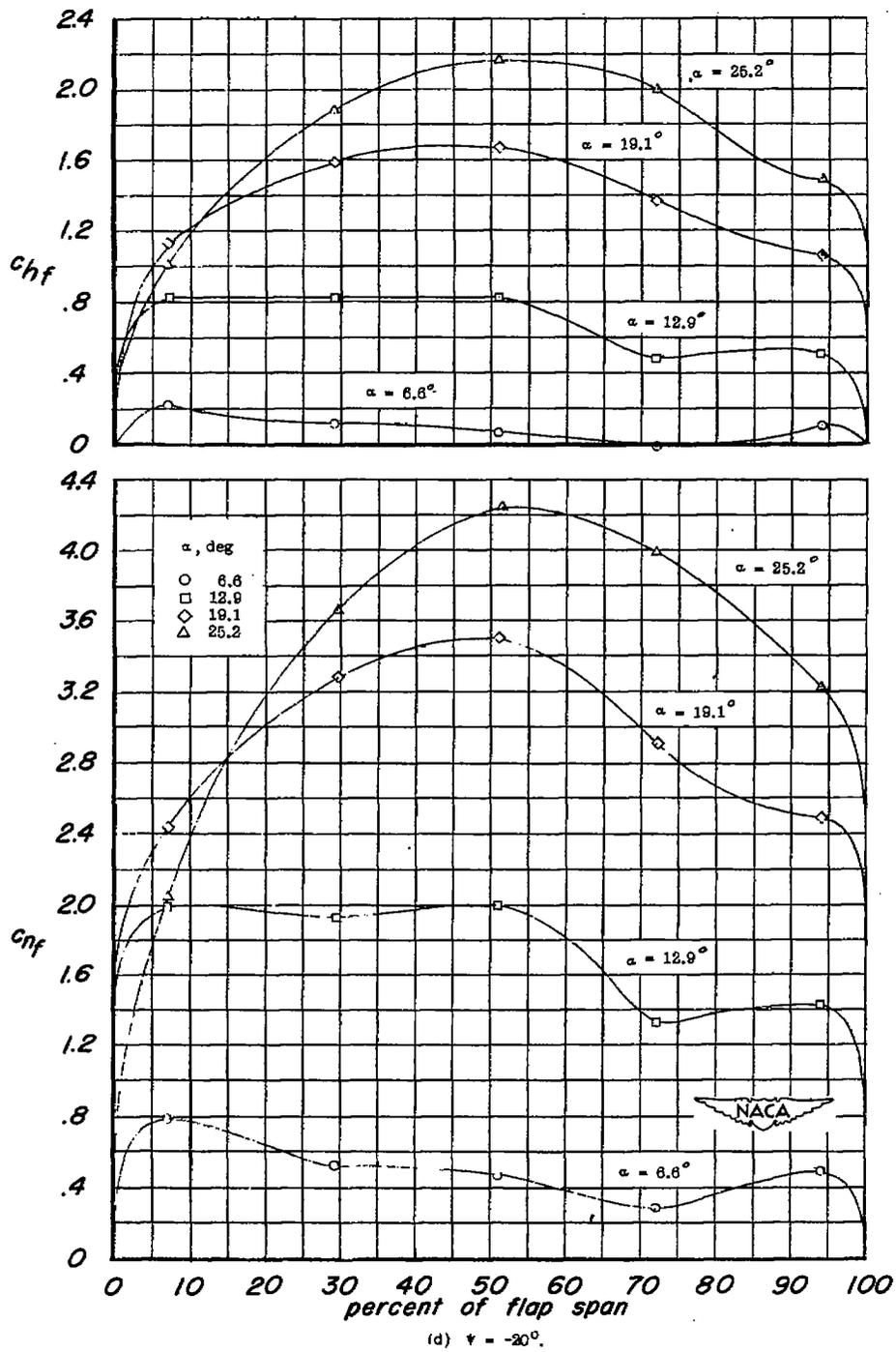


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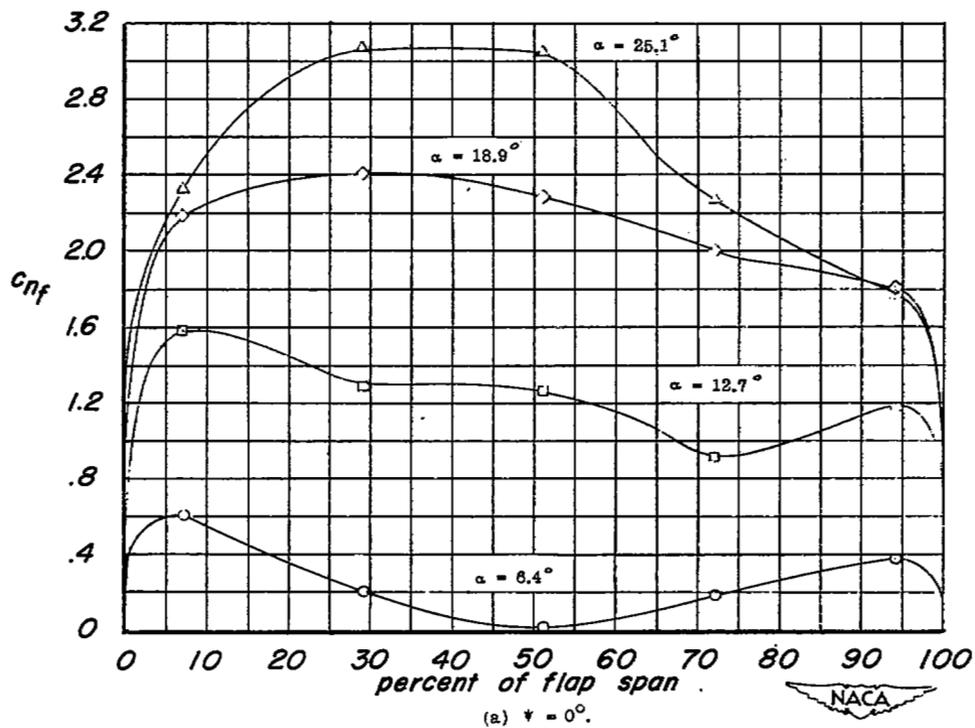
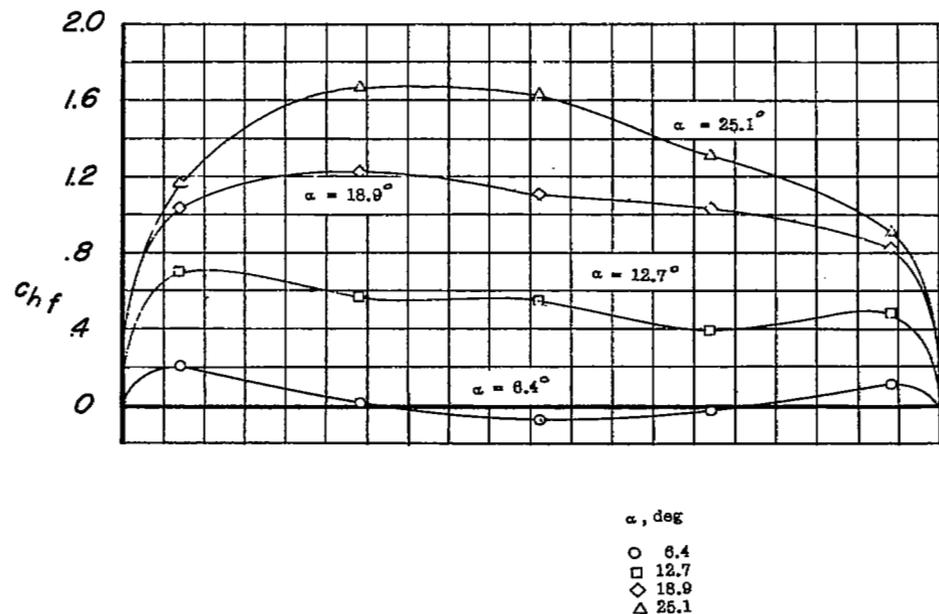


Figure 10.— Variation of the normal-force and hinge-moment coefficients across the leading-edge flap span for the 52° sweptback wing. Split flaps neutral, fences installed; $R = 4.4 \times 10^6$.

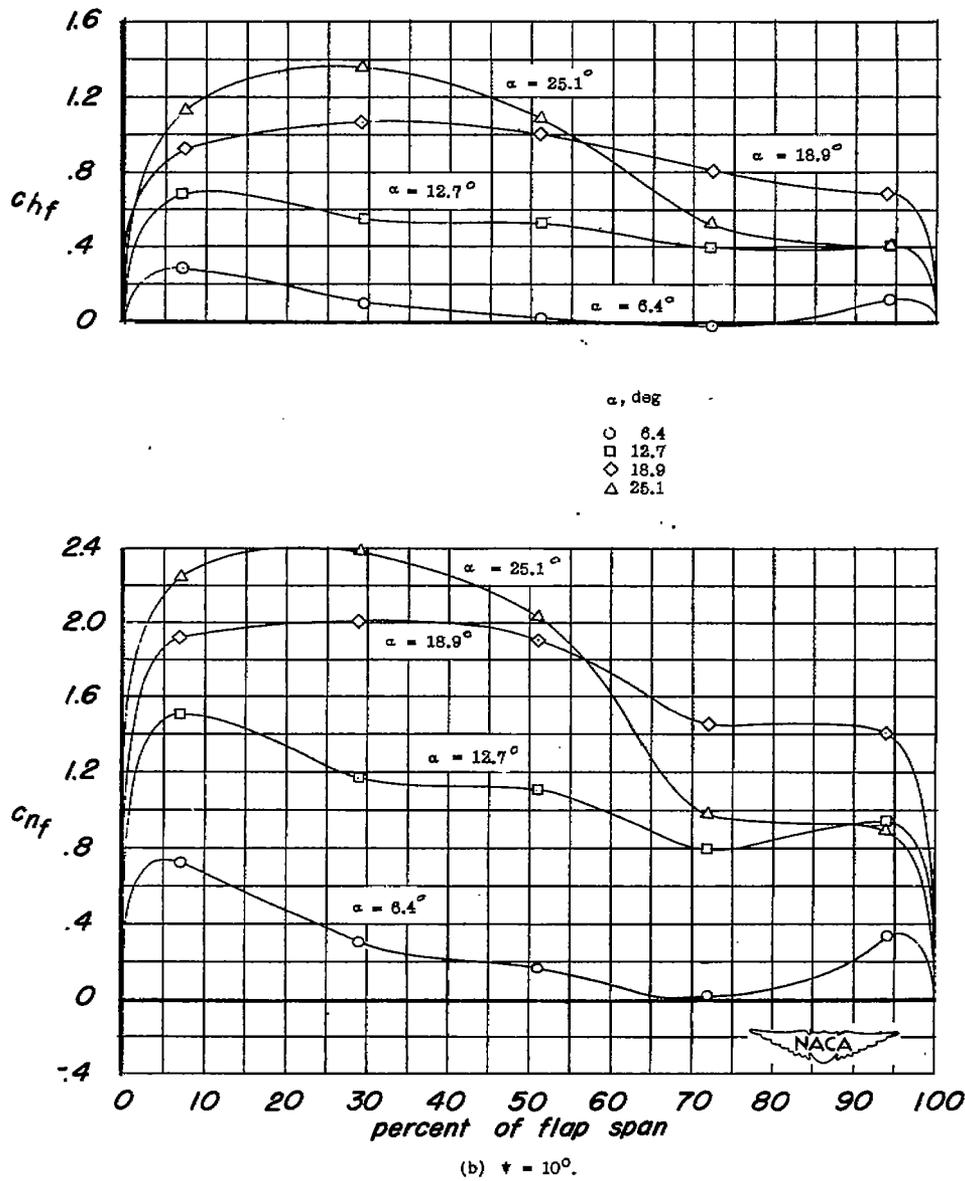


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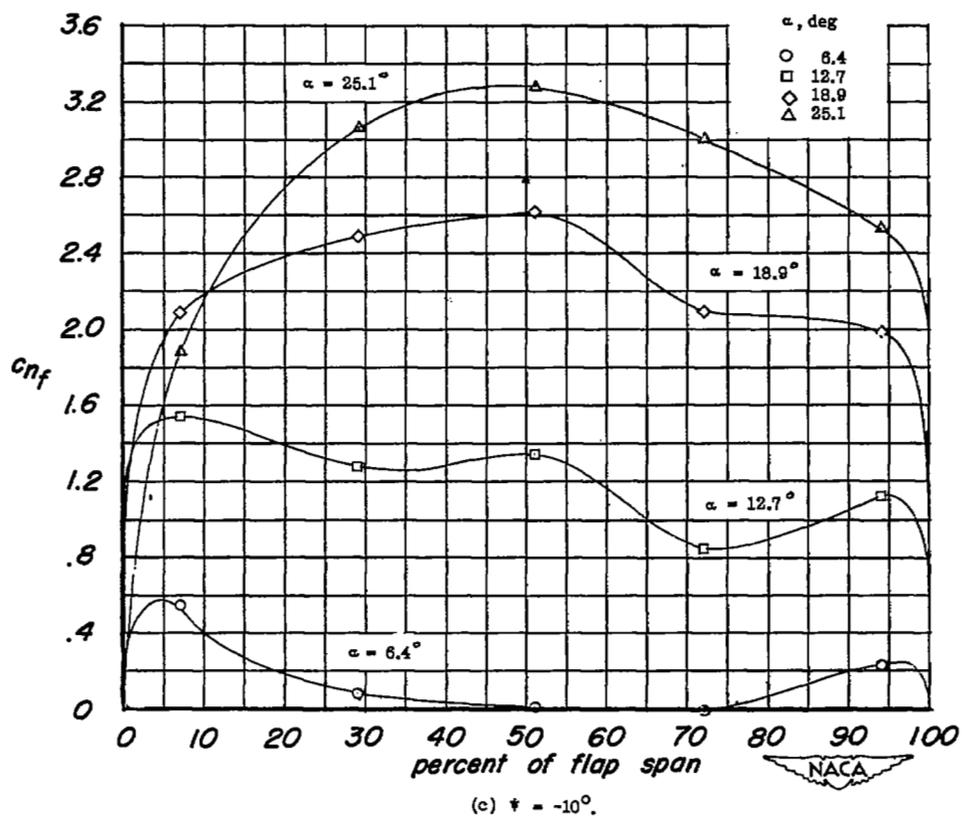
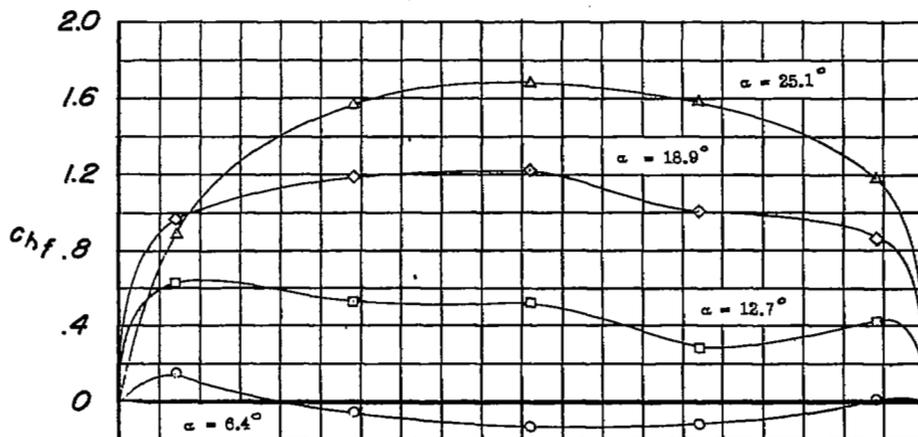


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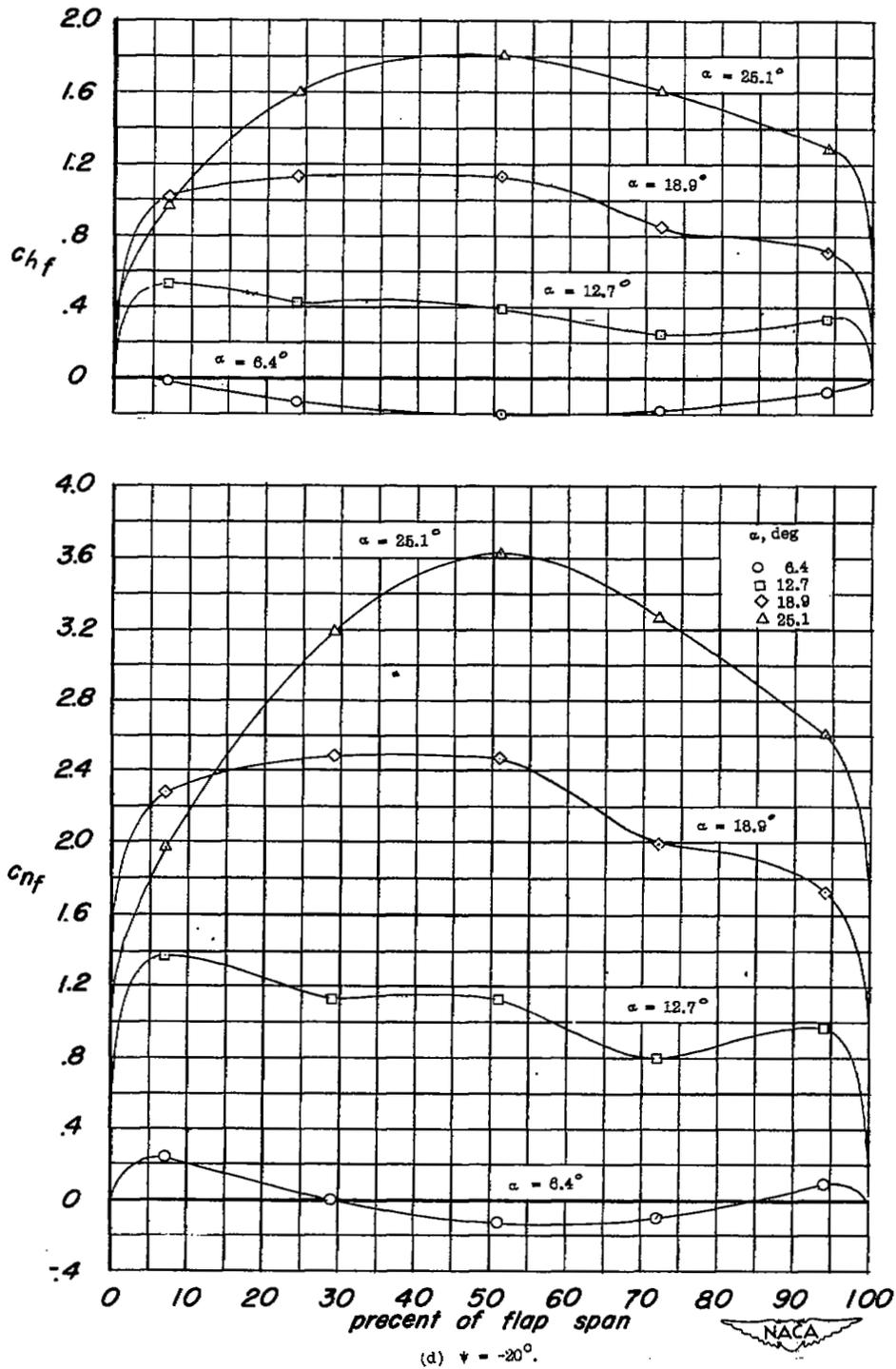


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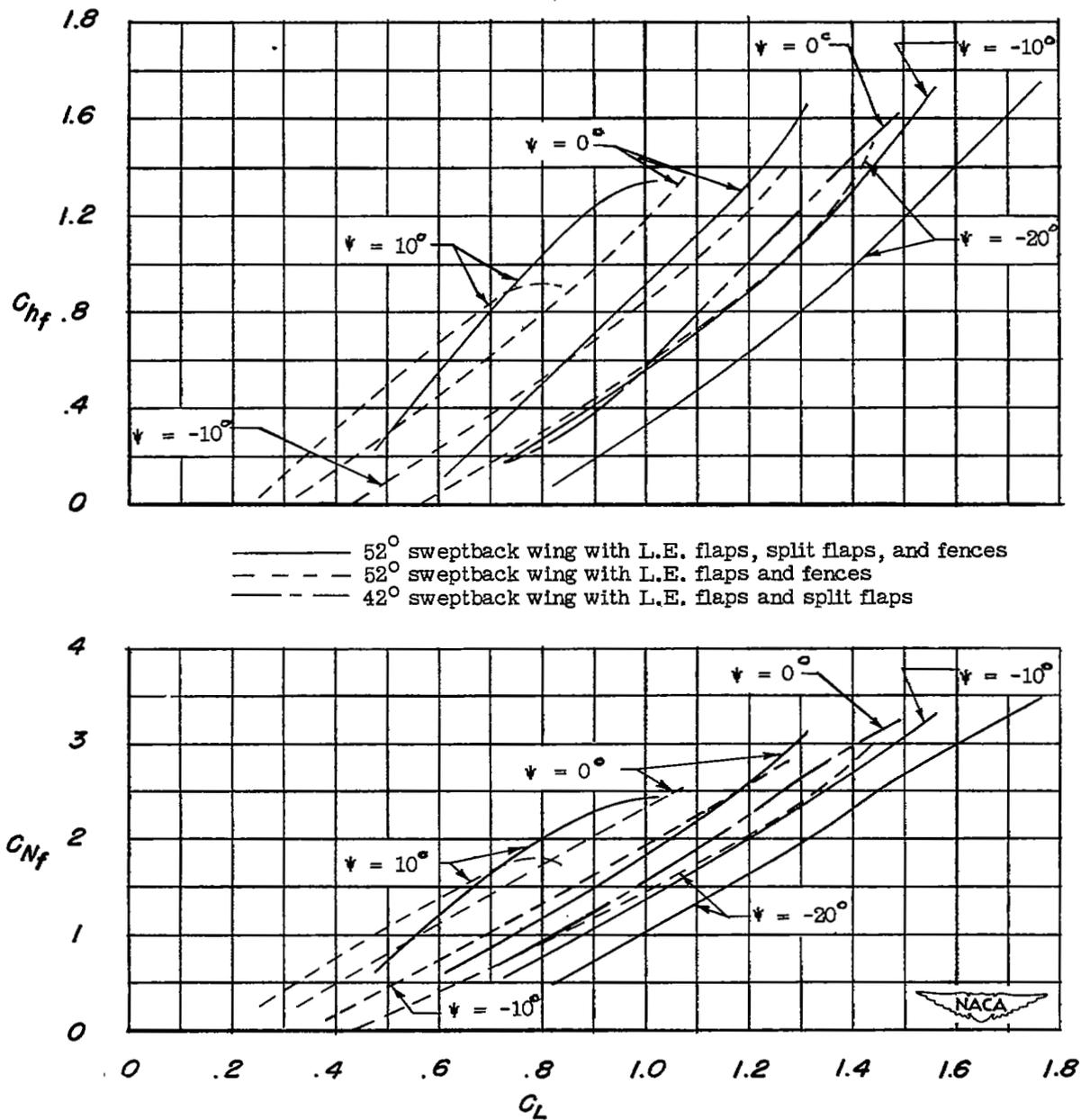


Figure 11.— Variation of C_{N_f} and C_{h_f} with C_L for an extended leading-edge flap on wings of 42° and 52° sweepback for various angles of yaw with and without split flaps deflected.

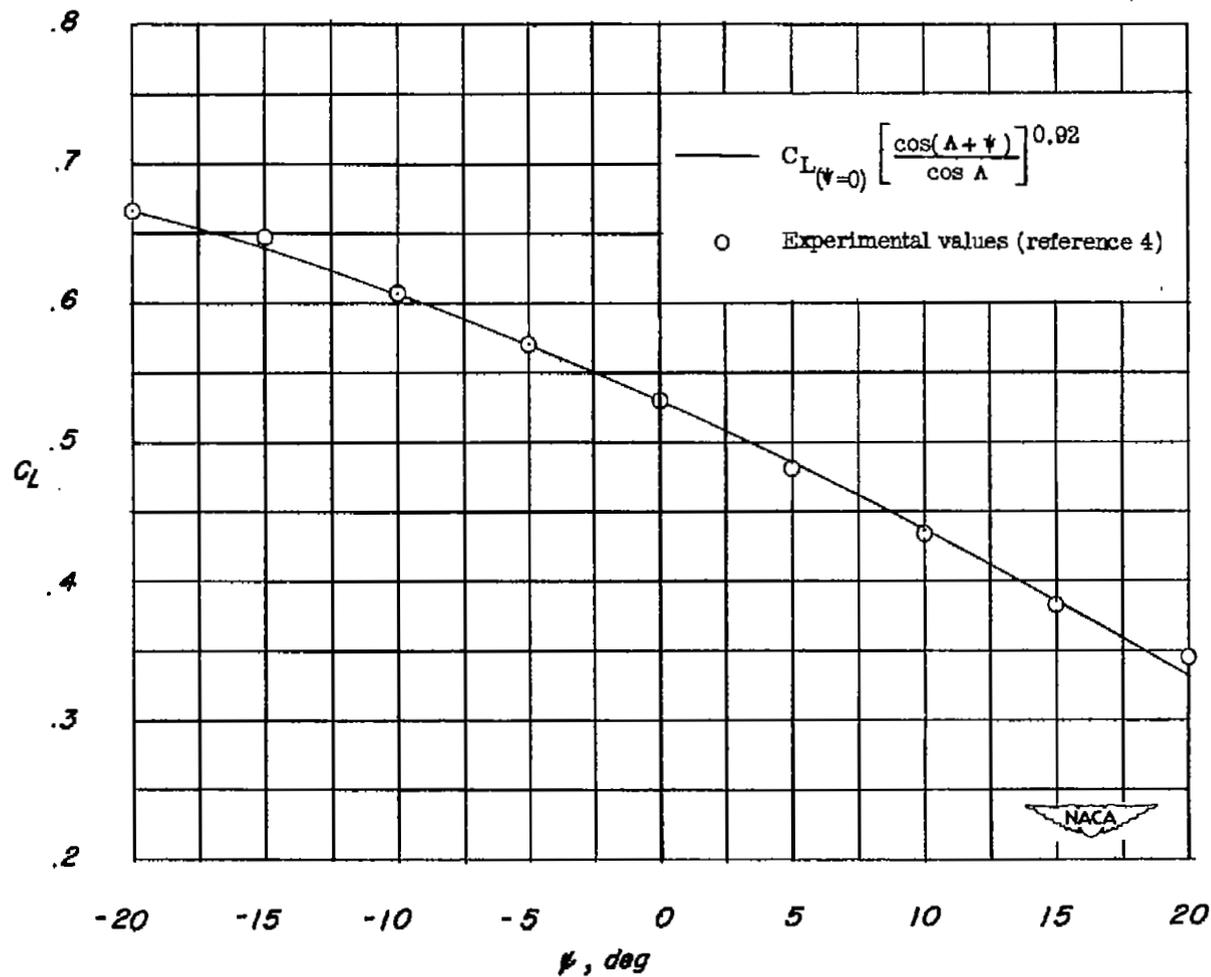
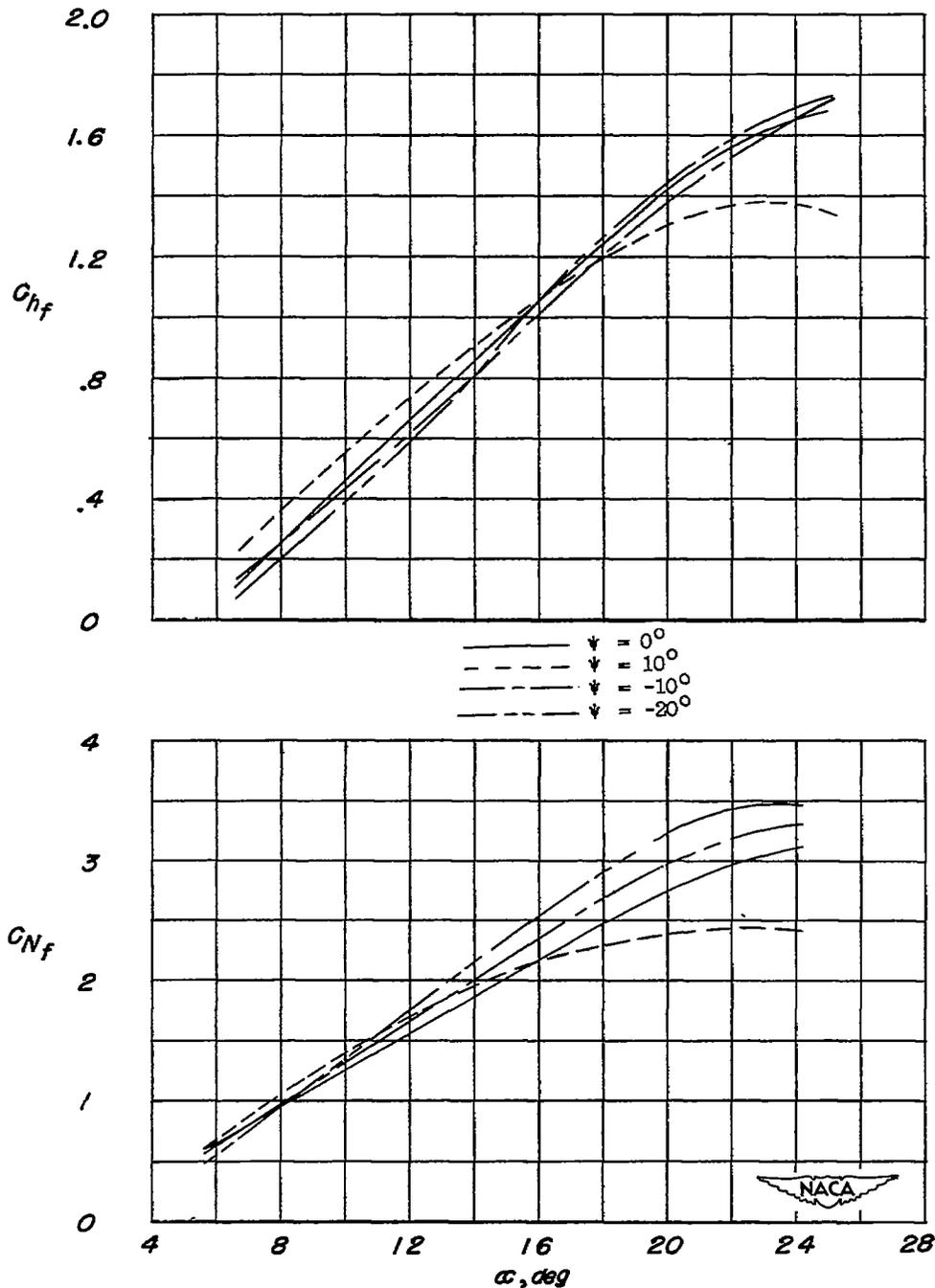
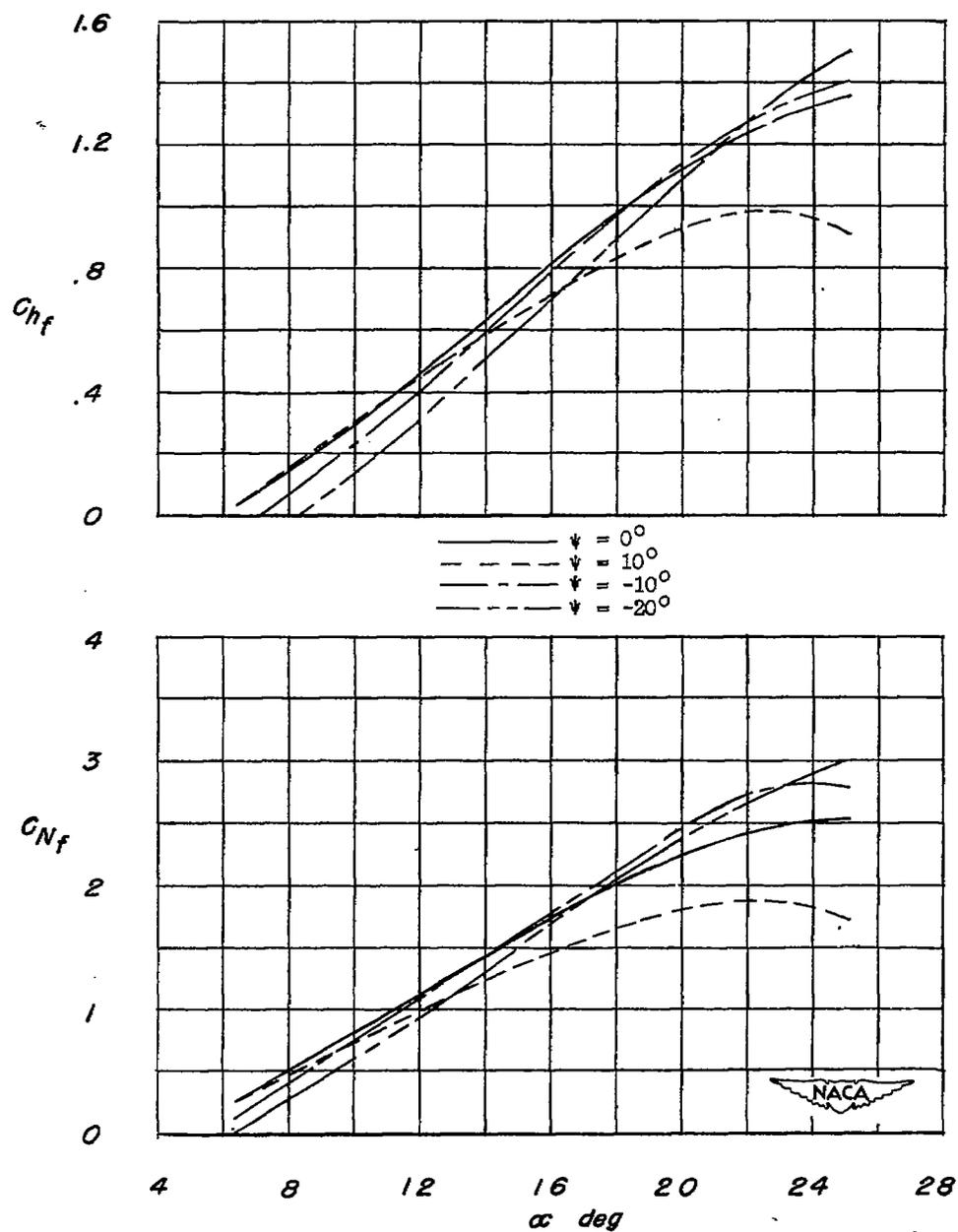


Figure 12.— Comparison of calculated values and experimental values based on pressure distributions for the variation of lift coefficient with angle of yaw for one wing panel of a 45° sweptback wing. (Reference right wing panel.)



(a) Split flaps deflected.

Figure 13.— Variation of the total flap normal-force and hinge-moment coefficients with angle of attack for the 52° sweptback wing and fences installed; $R = 4.4 \times 10^6$.



(b) Split flaps neutral.

Figure 13.- Concluded.

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