RESEARCH MEMORANDUM

EFFECTS ON CONTROL EFFECTIVENESS OF SYSTEMATICALLY VARYING THE SIZE AND LOCATION OF TRAILING-EDGE FLAPS ON A 45° SWEPTBACK WING AT A MACH NUMBER OF 1.9

By Carl R. Jacobsen

Langley Aeronautical Laboratory
Langley Field, Va.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

December 4, 1951

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

EFFECTS ON CONTROL EFFECTIVENESS OF SYSTEMATICALLY VARYING THE SIZE AND LOCATION OF TRAILING-EDGE FLAPS ON A 45° SWEPTBACK WING AT A MACH NUMBER OF 1.9

By Carl R. Jacobsen

SUMMARY

The effects on control effectiveness of systematically varying the size and location of trailing-edge flaps on a 45° sweptback wing has been investigated at a Mach number of 1.9. The tests were made in the Langley 9- by 12-inch supersonic blowdown tunnel at a Reynolds number of 2.1×10^6. The wing model had an aspect ratio of 2.5, a taper ratio of 0.625, and 6-percent-thick hexagonal airfoil sections.

The most important finding indicated by the experimental results and substantiated by the calculations was the loss in flap effectiveness caused by the effects of the wing tip. The wing tip influenced the loading due to deflecting the flap sufficiently to cause the flap location for maximum rolling-moment effectiveness to move from an outboard to a midsemispan location as the flap chord was increased from 25 to 45 percent of the wing chord. The values of the calculated effectiveness parameters were in qualitative agreement with the experimental values, although the calculated parameters were somewhat higher. The maximum deviation between the experimental and calculated results occurred for those flaps extending inboard to the fuselage.

INTRODUCTION

The characteristics of trailing-edge flaps and spoilers on two related semispan wings in the Langley 9- by 12-inch supersonic blowdown tunnel are being investigated. References 1 and 2 report the results of the initial investigation at a Mach number of 1.9 on an unswept wing. The present paper presents a similar study of trailing-edge flaps on a wing swept back 45°.

[redacted]

The tapered wing had an aspect ratio of 2.5 and 6-percent-thick hexagonal airfoil sections. Flaps having chords and spans which were systematically varied were tested at several spanwise locations.

The investigation was carried out at a Mach number of 1.9 and a Reynolds number of 2.1×10^6. The angle-of-attack range was $\pm 6^\circ$ and the flaps were deflected from 0° to 15°.

Five-component force data are presented and the experimental values of flap lift, rolling-moment, and pitching-moment effectiveness are compared with that predicted by linearized theory.

COEFFICIENTS AND SYMBOLS

All data are presented with respect to the wind axes.

C_L — lift coefficient $\left(\frac{\text{Lift}}{qS}\right)$

C_D — drag coefficient $\left(\frac{\text{Drag}}{qS}\right)$

C_m — pitching-moment coefficient $\left(\frac{\text{Pitching moment about } 0.5\bar{c}}{qS\bar{c}}\right)$

$C_{l_{gross}}$ — gross rolling-moment coefficient $\left(\frac{\text{Rolling moment of the semispan wing}}{2qSb}\right)$

$C_{n_{gross}}$ — gross yawing-moment coefficient $\left(\frac{\text{Yawing moment of the semispan wing}}{2qSb}\right)$

C_l — rolling-moment coefficient due to control-surface deflection $\left(C_{l_{gross}} - C_{l_{gross(\delta=0^\circ)}}\right)$

ΔC_D, ΔC_m, . . . — increment in coefficient due to control-surface deflection

q — free-stream dynamic pressure

S — exposed semispan wing area (10.00 sq in.)

$\bar{c}$ — mean aerodynamic chord of exposed wing area (3.13 in.)

c — local wing chord

c_f — local control-surface chord

b — wing span, twice distance from wing root to wing tip (8.13 in.)

b_f — control-surface span

y_i — spanwise location of inboard end of control surface

y_o — spanwise location of outboard end of control surface

α — angle of attack relative to free-stream direction

δ — control-surface deflection measured in a plane normal to hinge line

R — Reynolds number based on $\bar{c}$

C_{L_α} — rate of change of lift coefficient with angle of attack

C_{L_δ} — rate of change of lift coefficient with control-surface deflection $\left(\frac{\partial C_L}{\partial \delta}\right)$

C_{m_δ} — rate of change of pitching-moment coefficient with control-surface deflection $\left(\frac{\partial C_m}{\partial \delta}\right)$

C_{l_δ} — rate of change of rolling-moment coefficient with control-surface deflection $\left(\frac{\partial C_l}{\partial \delta}\right)$

MODEL

The semispan wing and the half-fuselage mounted in the test section are presented in figure 1. The principle dimensions of the wing and fuselage are presented in figure 2. The wing was swept back 45° at the midchord line. The aspect ratio of the basic wing was 2.5 and the taper ratio was 0.625. The airfoil sections parallel to the airstream were symmetrical 6-percent-chord-thick hexagonal profiles. The profiles were modified slightly by rounding the ridges. Both the leading-edge wedge and the trailing-edge wedge of the section extended 30 percent of the chord. The resultant wedge angle was 11.42°.

The configurations tested included 25-, 35-, and 45-percent-chord plain flaps. The flaps extended from the fuselage intersection at 0.20b/2 to 0.95b/2 and were divided into three 0.25b/2 segments. Flap spans equal to 25, 50, and 75 percent of the wing semispan were achieved by deflecting the segments separately and in combination. The gaps between any segments having the same deflections were sealed and faired. For each flap chord tested, a fine groove along the hinge line was machined on the upper surface of the wing and thereby allowed deflection of the flaps about an axis near the lower surface of the wing. Flap deflections were measured normal to the hinge line.

TESTS

The Langley 9- by 12-inch supersonic blowdown tunnel in which the present tests were made uses the compressed air of the 19-foot pressure tunnel. The air enters at an absolute pressure of about $2\frac{1}{3}$ atmospheres and contains about 0.003 pound of water per pound of air. The free-stream Mach number has been calibrated at 1.90 ± 0.02. This Mach number was used in determining the dynamic pressure. For the tunnel-clear condition, the static pressure in the test section varied about ±1.5 percent. Flow characteristics which might affect the aerodynamic results are discussed in reference 3.

The average dynamic pressure for the tests was 11.0 pounds per square inch. The average Reynolds number was 2.1×10^6. The test Reynolds number decreased about 3.8 percent during the course of each run because of the decreasing pressure of the inlet air.

The investigation was made through an angle-of-attack range from -6° to +6° and through a flap-deflection range of 0° to +15°.

Five-component force measurements were obtained for the wing in the presence of, but not attached to, a half-fuselage. Because of the balance deflections under load, a gap of about 0.015 inch was maintained between the wing and fuselage under a no-load condition (reference 1).

TEST TECHNIQUE

The semispan model used in this investigation was cantilevered from a strain-gage balance which mounts flush with the tunnel wall and rotates with the model through the angle-of-attack range. The half-fuselage is attached to the housing of the balance, and thereby permits the wing to be tested in the presence of, but not attached to, the fuselage.

The development of an acceptable technique for testing semispan wing models in this facility was reported in references 2 and 3. It was found that shimming a half-fuselage away from the tunnel wall minimized wall-boundary-layer effects over the fuselage. However, the gap between the wing and body caused deviations to occur in the wing loading near the wing-fuselage juncture. The deviations were small at angles of attack below 4°. The indications are that, at higher angles of attack, the wing loading would be considerably altered and, therefore, the characteristics of flaps located adjacent to the fuselage could well be in error.

RESULTS

The rolling-moment and yawing-moment coefficients as presented herein apply to a full-span wing with the flap deflected on the left wing panel only. The lift, pitching-moment, and drag data, however, are reduced for flaps deflected on both wing panels. Variations of the aerodynamic coefficients with angle of attack are presented in figures 3 to 7 for the 45-percent-chord flap arrangements. The data for the lift, rolling moment, and pitching moment for the 45-percent-chord flaps are representative of the data for all other control arrangements in that flap effectiveness was almost independent of angle of attack. Consequently, for the 25- and 35-percent-chord flaps only the variations of the aerodynamic coefficients with flap deflection at zero angle of attack have been presented in figures 8 to 12 along with similar data for the 45-percent-chord flaps.

No tare corrections were necessary since the object of the tests was to obtain data on flap effectiveness. The zero shift shown by the data of figure 6 is probably a result of model asymmetry. From a general consideration of balance-calibration accuracy, fluctuations in loads, nonuniformity in the flow, and accuracy in the model setup, it is believed that the data presented are accurate to within about the following limits:

α, degrees	±0.05
δ, degrees	±0.10
C_L	±0.005
C_l	±0.001
C_D	±0.001
C_m	±0.002
C_n	±0.0002

Average experimental values of lift, rolling-moment, and pitching-moment effectiveness parameters, which were obtained arbitrarily between a flap deflection of 0° and 10°, are presented in table I. The data of table I are presented in figures 13 to 15 to illustrate the effects of flap span, flap chord, and flap spanwise location on these effectiveness parameters. Included in both table I and figures 13 to 15 are the calculated effectiveness values of the flaps corrected for wing thickness as obtained by the use of the method of reference 4. Because of the limiting assumptions, the method of reference 4 cannot be used directly for calculating the effectiveness of flaps which extend outboard to 0.95b/2 or which extend inboard to the fuselage. It was possible, however, by certain modifications to the method to obtain approximate effectiveness values for these flaps.

In reference 4, flaps were assumed to be located either at the wing tip or far enough inboard to prevent the outermost Mach cone from the flap from crossing the wing tip. For either location, the innermost Mach line was assumed to lie completely on the adjacent wing panel. In the present investigation, for those flaps located adjacent to the fuselage, the method was modified to consider the fuselage as a reflection plane. For those flaps which extended to 0.95b/2, it was necessary to calculate the effectiveness values of flaps of the same size and geometry for all spanwise locations directly covered by the method of reference 4 (ignoring the effects of the fuselage). From these values a curve was faired through the region not covered by the calculations and approximate values were then obtained at the flap's true location. As an illustration, figure 16 shows how this procedure was used to obtain the approximate values of C_{L_δ} for the 25-percent-semispan flaps which extend to 0.95b/2. The approximate values obtained for the calculated effectiveness parameters in this faired region may differ within approximately the following limits, depending upon the fairing used.

c_f/c	C_{L_δ}	C_{l_δ}	C_{m_δ}
0.25	±0.0001	±0.00002	±0.0001
.35	±.0002	±.00002	±.0001
.45	±.0002	±.00003	±.0002

[redacted]

DISCUSSION

Wing Characteristics

The experimental value of C_{L_α} for the wing with flaps undeflected was 0.041. The center of pressure on the wing at the lower angles of attack was located at 48 percent of the exposed semispan out from the body and at 35 percent of the mean aerodynamic chord of the exposed wing.

Flap Characteristics

The lift, rolling moment, and pitching moment varied almost linearly with flap deflection (figs. 8, 9, and 11) and increased with increasing flap span and with increasing flap chord (within the experimental accuracy). The 50- and 25-percent-semispan flaps of 25-percent chord extending inboard $\left(y_i = 0.20b/2\right)$ to the body had somewhat lower values of lift than would be expected (possibly a result of wing-fuselage gap effects). Except for these configurations, the experimental lift effectiveness for a flap of given span and percent chord decreased slightly as the flap location was moved outboard towards the wing tip (fig. 13). This decrease was in agreement with the calculated effectiveness and was related to the decrease in flap area caused by wing taper and also to wing tip effects in cases for which the outer Mach line from the flap crossed the wing tip. For these cases, the area of the region of carry-over loading progressively decreased as the flap was moved outboard of this location and as its outer Mach cone enclosed more of the wing tip. This Mach cone caused a series of disturbances at its intersection with the wing tip and, consequently, affected the loading over a part of the flap. These effects are possibly caused in part by aeroelastic and viscous effects. Aeroelastic effects would be expected to be small because the wing was solid steel and essentially rigid. The effects of viscosity are believed to be secondary to the tip effects discussed. As the flap chord was increased from 25 to 45 percent, the effectiveness of the flaps extending to 0.95b/2 increased less rapidly with respect to the effectiveness of flaps located further inboard.

The wing tip effects would be expected to change the rolling moment more than the lift because, in addition to the decrease in lift loading, the rolling moment should also be affected by the inboard shift of the center of pressure of the loading relative to the flap. There was some indication from the results that the flap location for maximum effectiveness, particularly for the 25-percent-semispan flaps, moved from the outboard to the midsemispan location as the flap chord was increased from 25 to 45 percent of the wing chord; however, the differences involved were small. The midsemispan flap location for the maximum

[redacted]

rolling effectiveness of the larger chord flaps is in agreement with the findings of other investigations made at transonic and supersonic speeds (references 5 and 6) and at low and high subsonic speeds (reference 7). The data of reference 6 indicate that, when a wing is swept back from 0° to 45°, a control surface of given span should be moved inboard to achieve the maximum effectiveness. The same conclusion is reached when the results reported herein are compared with the results obtained in the parallel investigation of reference 1, since sweep angle is the only variable.

It should be pointed out that the calculated flap characteristics given in reference 1 for outboard flaps are in slight error because the wing tip effects were arbitrarily neglected to hold the computing time within reasonable limits (now circumvented by the use of reference 4). It has been found that, even with wing tip effects taken into consideration, the trends shown in reference 1 would be unaffected. In order to better illustrate this fact and to show the influence of sweepback on tip effects, the following table is presented which lists the calculated values of C_{l_δ} for the 25-percent-semispan flaps having 45-percent chord. On the unswept wing, the wing tip effects slightly decreased the effectiveness of the outboard flap. For the sweptback wing, however, the decrease was so pronounced that the center flap became more effective than the outboard flap. For wings having approximately the same trailing-edge sweep angle, these same effects of the wing tip would be expected to occur at lower Mach numbers for smaller chord flaps.

Flap location, $\frac{y_i}{b/2}$	Calculated C_{l_δ}	
	Wing unswept	Wing swept back 45°
0.20	0.00040	0.00055
.45	.00074	.00061
.70 (considering no tip effects)	.00094	.00073
.70 (considering tip effects)	.00085	.00052

The calculated and experimental pitching-moment effectiveness increased with flap chord, flap span, and with increasing flap spanwise location out from the body (table I(c) and fig. 15). The values of the calculated parameters were, however, somewhat higher than the experimental values (table I(a), (b), and (c), respectively). The maximum deviation between the experimental and calculated results occurred for those flaps extending to the fuselage probably as a combined result of the test technique employed and of wing-body interference.

It should be pointed out that although a center or outboard flap location might be desirable from the standpoint of rolling moment, this location may have disadvantageous yawing-moment and drag characteristics as evidenced by figures 10 and 12.

CONCLUDING REMARKS

An investigation has been made in the Langley 9- by 12-inch supersonic blowdown tunnel at a Mach number of 1.9 to determine the effects of varying the size and location of plain flap-type controls on a low-aspect-ratio sweptback wing.

The most important finding indicated by the experimental results and substantiated by the calculations was the loss in flap effectiveness caused by the effects of the wing tip. The wing tip influenced the loading due to flap deflection sufficiently to cause the flap location for maximum rolling-moment effectiveness to move from an outboard to a midsemispan location as the flap chord was increased from 25 to 45 percent of the wing chord. A comparison of these results and the results of a previous investigation of a related unswept wing shows that sweepback caused a small loss in lift and a related loss in rolling effectiveness, but sweepback caused no change in the general trends of flap lift. Increasing the sweep, however, tended to move the flap location for maximum rolling-moment effectiveness from an outboard to a midsemispan location.

Linearized theory predicted the effects of changing the flap size and location on the characteristic trends of the lift, rolling-moment, and pitching-moment effectiveness parameters. The values of the calculated parameters were, however, somewhat higher than the experimental values. The maximum deviation between the experimental and calculated results occurred for those flaps extending to the fuselage.

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

REFERENCES

1. Mitchell, Meade H., Jr.: Effects of Varying the Size and Location of Trailing-Edge Flap-Type Controls on the Aerodynamic Characteristics of an Unswept Wing at a Mach Number of 1.9. NACA RM L50F08, 1950.

2. Conner, D. William, and Mitchell, Meade H., Jr.: Effects of Spoiler on Airfoil Pressure Distribution and Effects of Size and Location of Spoilers on the Aerodynamic Characteristics of a Tapered Unswept Wing of Aspect Ratio 2.5 at a Mach Number of 1.90. NACA RM L50L20, 1951.

3. Conner, D. William: Aerodynamic Characteristics of Two All-Movable Wings Tested in the Presence of a Fuselage at a Mach Number of 1.9. NACA RM L8H04, 1948.

4. Goin, Kennith L.: Equations and Charts for the Rapid Estimation of Hinge-Moment and Effectiveness Parameters for Trailing-Edge Controls Having Leading and Trailing Edges Swept Ahead of the Mach Lines. NACA TN 2221, 1950. NACA Rep. 1041

5. Strass, H. Kurt, Fields, E. M., and Schult, Eugene D.: Some Effects of Spanwise Aileron Location and Wing Structural Rigidity on the Rolling Effectiveness of 0.3-Chord Flap-Type Ailerons on a Tapered Wing Having 63° Sweepback at the Leading Edge and NACA 64A005 Airfoil Sections. NACA RM L51D18a, 1951.

6. Strass, H. Kurt: The Effect of Spanwise Aileron Location on the Rolling Effectiveness of Wings with 0° and 45° Sweep at Subsonic, Transonic, and Supersonic Speeds. NACA RM L50A27, 1950.

7. Johnson, Harold S.: Wind-Tunnel Investigation at Subsonic and Low Transonic Speeds of the Effects of Aileron Span and Spanwise Location on the Rolling Characteristics of a Test Vehicle with Three Untapered 45° Sweptback Wings. NACA RM L51B16, 1951.

[redacted]

TABLE I

EXPERIMENTAL AND CALCULATED FLAP CHARACTERISTICS OF A

WING MODEL AT A MACH NUMBER OF 1.9

(a) Lift effectiveness parameter, C_{L_δ}

Flap span (percent b/2)	Location of inboard end of flap (percent b/2)		C_{L_δ}					
			Flap chord percent c: 25		35		45	
			Experimental	Calculated	Experimental	Calculated	Experimental	Calculated
75	20		0.0057	0.0057	0.0071	0.0079	0.0095	0.0112
50	45		.0034	.0034	.0042	.0046	.0053	.0064
	20		.0029	.0043	.0046	.0061	.0068	.0087
25	70		.0015	.0015	.0018	.0019	.0022	.0026
	45		.0017	.0019	.0023	.0027	.0032	.0038
	20		.0010	.0023	.0026	.0035	.0035	.0050

NACA

[redacted] SECURITY INFORMATION

TABLE I

EXPERIMENTAL AND CALCULATED FLAP CHARACTERISTICS OF A WING

MODEL AT A MACH NUMBER OF 1.9 - Continued

(b) Rolling effectiveness parameter, C_{l_δ}

Flap span (percent b/2)	Location of inboard end of flap (percent b/2)		C_{l_δ}					
			Flap chord percent c: 25		35		45	
			Experimental	Calculated	Experimental	Calculated	Experimental	Calculated
75	20		0.00077	0.00081	0.00106	0.00114	0.00140	0.00165
50	45		.00060	.00060	.00076	.00080	.00096	.00113
	20		.00044	.00053	.00063	.00078	.00086	.00118
25	70		.00033	.00031	.00037	.00038	.00052	.00052
	45		.00022	.00029	.00039	.00042	.00056	.00061
	20		.00013	.00023	.00026	.00035	.00038	.00055

NACA

[redacted]

TABLE I

EXPERIMENTAL AND CALCULATED FLAP CHARACTERISTICS OF A WING

MODEL AT A MACH NUMBER OF 1.9 - Concluded

(c) Pitching-moment parameter, C_{m_δ}

Flap span (percent b/2)	Location of inboard end of flap (percent b/2)	C_{m_δ}					
	Flap chord percent c	25		35		45	
		Experimental	Calculated	Experimental	Calculated	Experimental	Calculated
75	20 	-0.0020	-0.0021	-0.0022	-0.0026	-0.0022	-0.0029
50	45 	-.0018	-.0018	-.0020	-.0022	-.0023	-.0025
	20 	-.0010	-.0013	-.0009	-.0015	-.0009	-.0019
25	70 	-.0010	-.0010	-.0011	-.0011	-.0012	-.0012
	45 	-.0006	-.0008	-.0009	-.0010	-.0009	-.0013
	20 	-.0003	-.0004	-.0002	-.0005	-.0002	-.0005

NACA

[redacted]

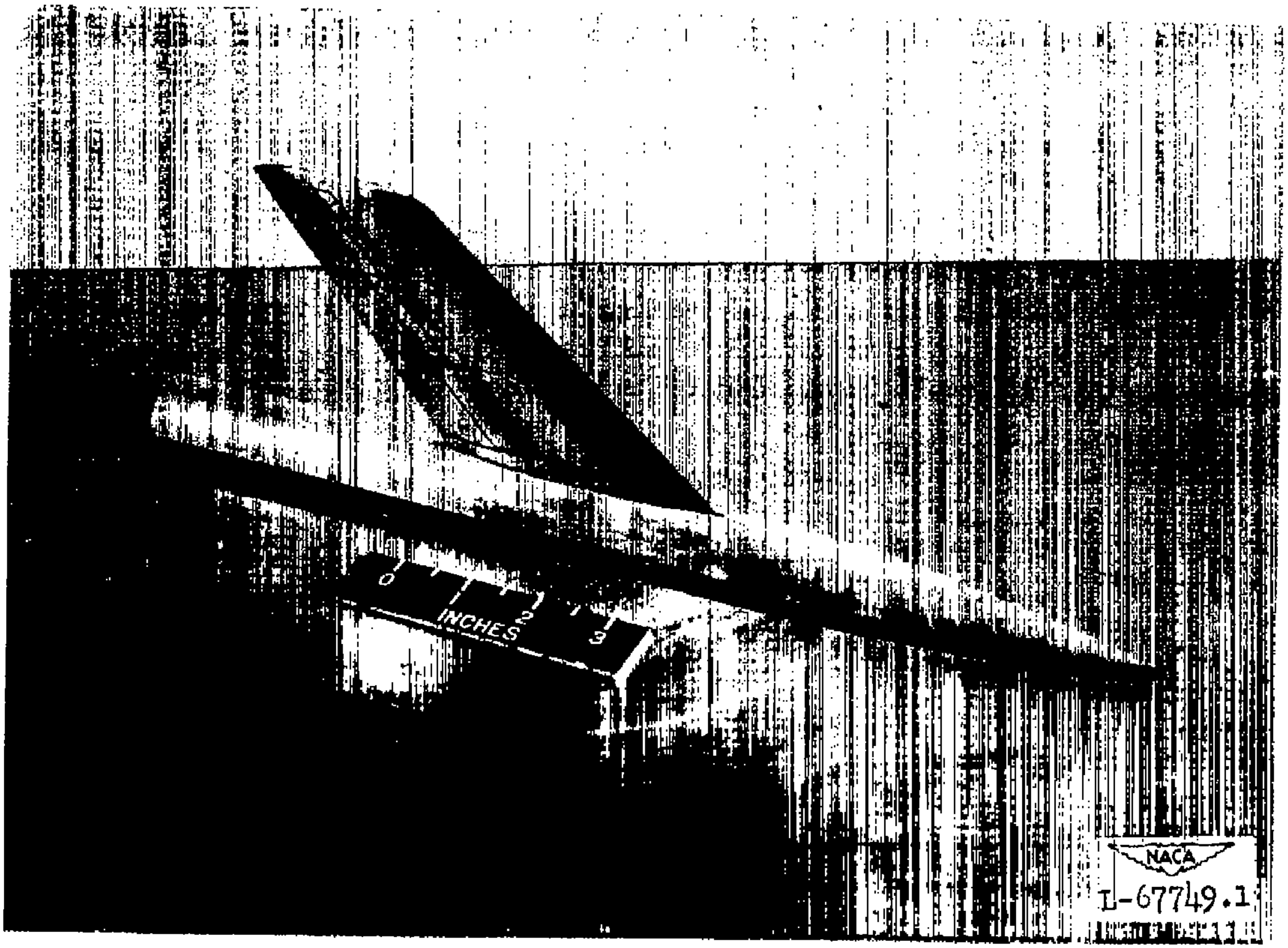
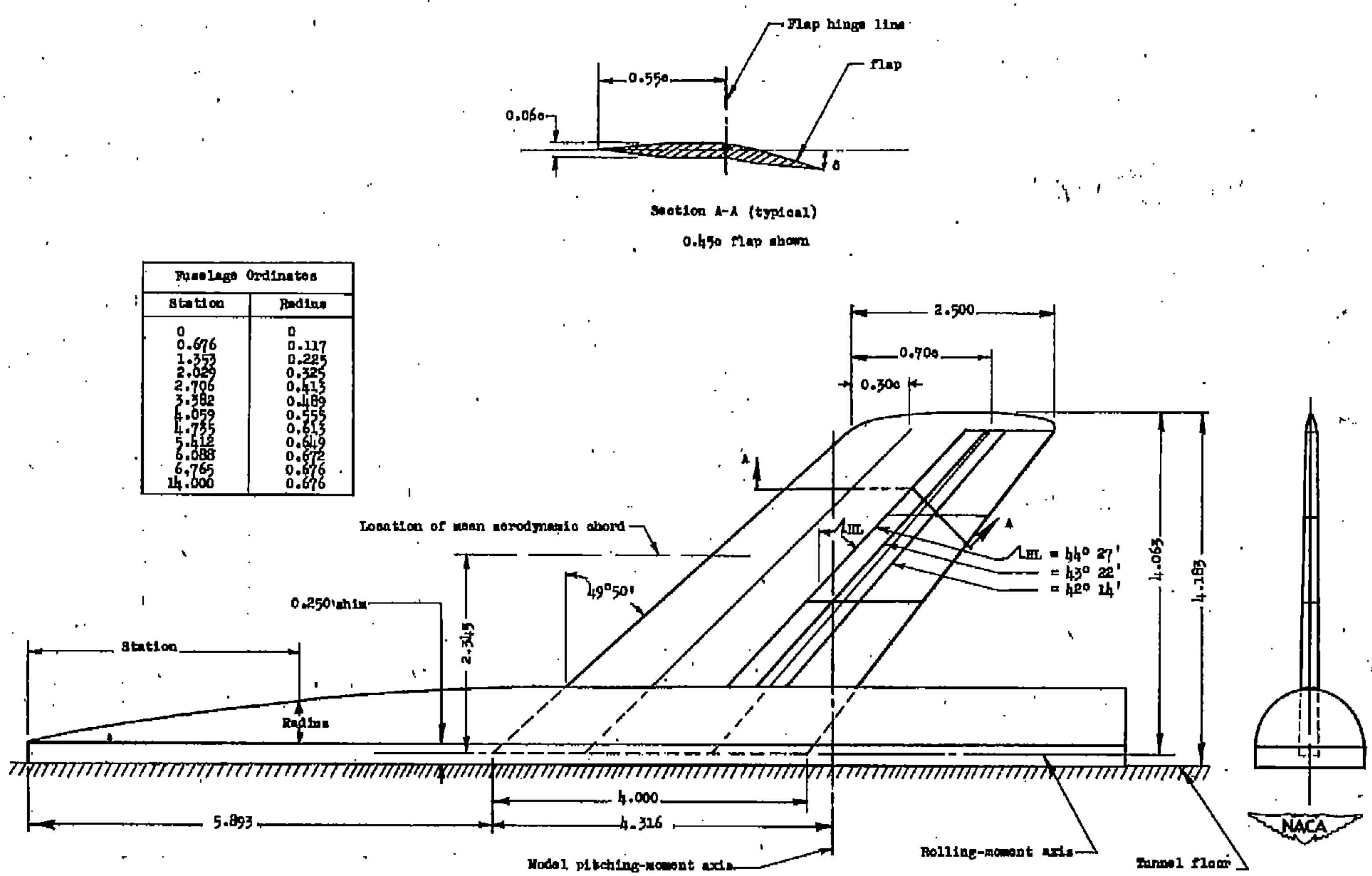


Figure 1.- Photograph of semispan wing model.

[redacted]



Fuselage Ordinates	
Station	Radius
0	0
0.676	0.117
1.353	0.225
2.029	0.325
2.706	0.413
3.382	0.489
4.059	0.555
4.735	0.613
5.412	0.649
6.088	0.672
6.765	0.676
14.000	0.676

Figure 2.- Details of semispan wing model. All dimensions are in inches.

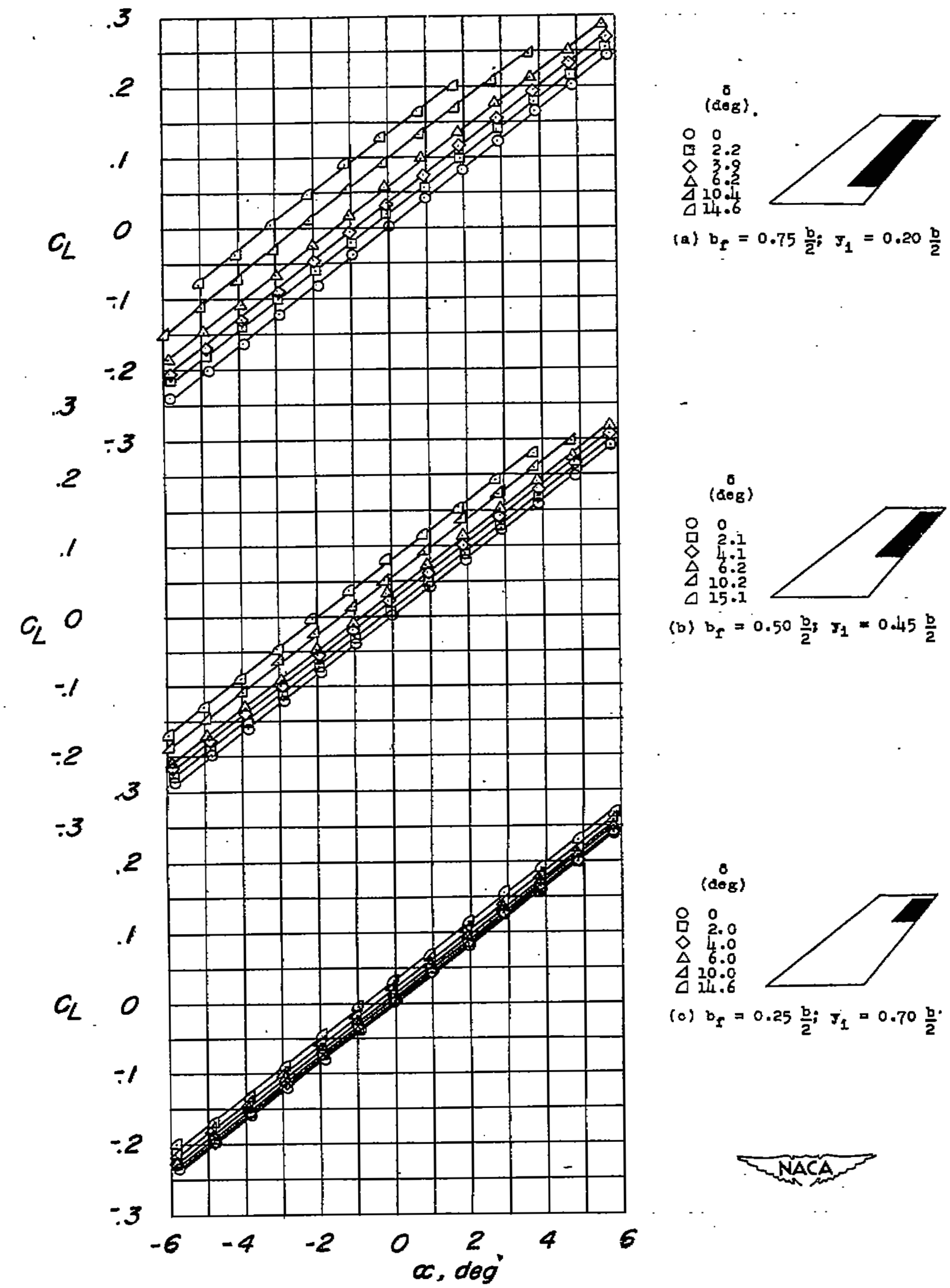


Figure 3.- Lift characteristics of a semispan wing with 45-percent-chord flaps. $R = 2.10 \times 10^6$; $M = 1.9$.

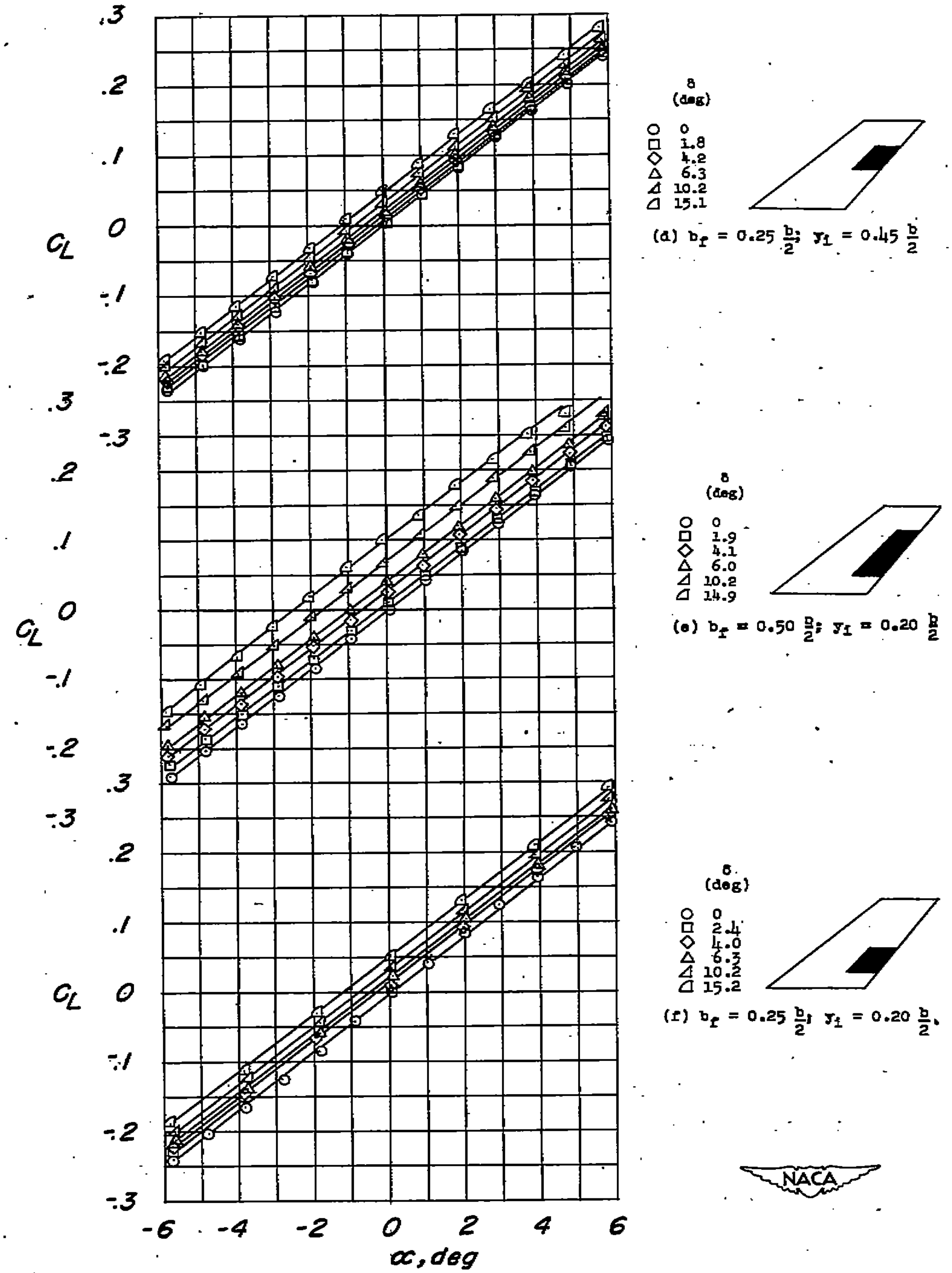


Figure 3.- Concluded.

[redacted]

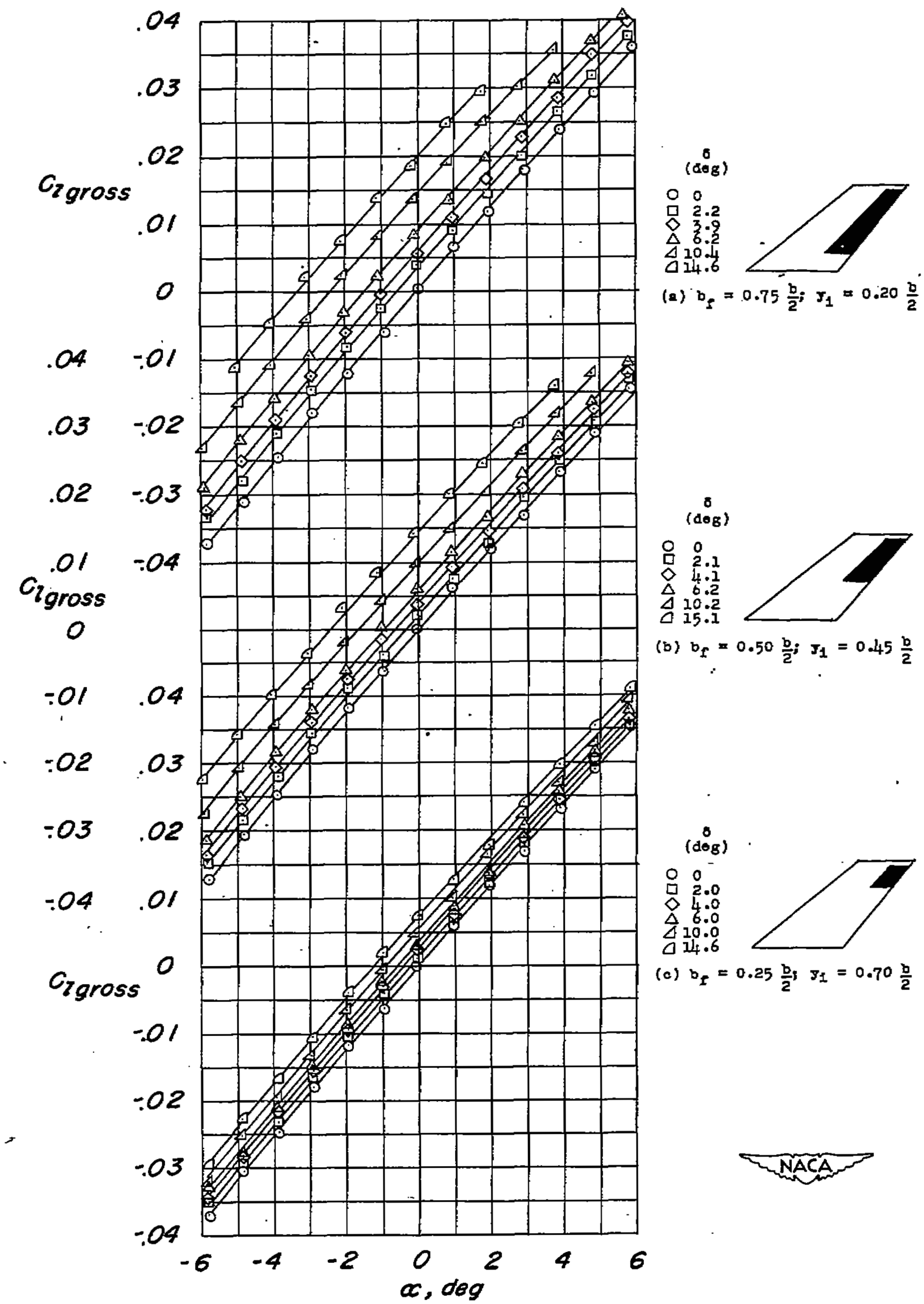


Figure 4.- Rolling-moment characteristics of a semispan wing with 45-percent-chord flaps. $R = 2.10 \times 10^6$; $M = 1.9$.

[redacted]

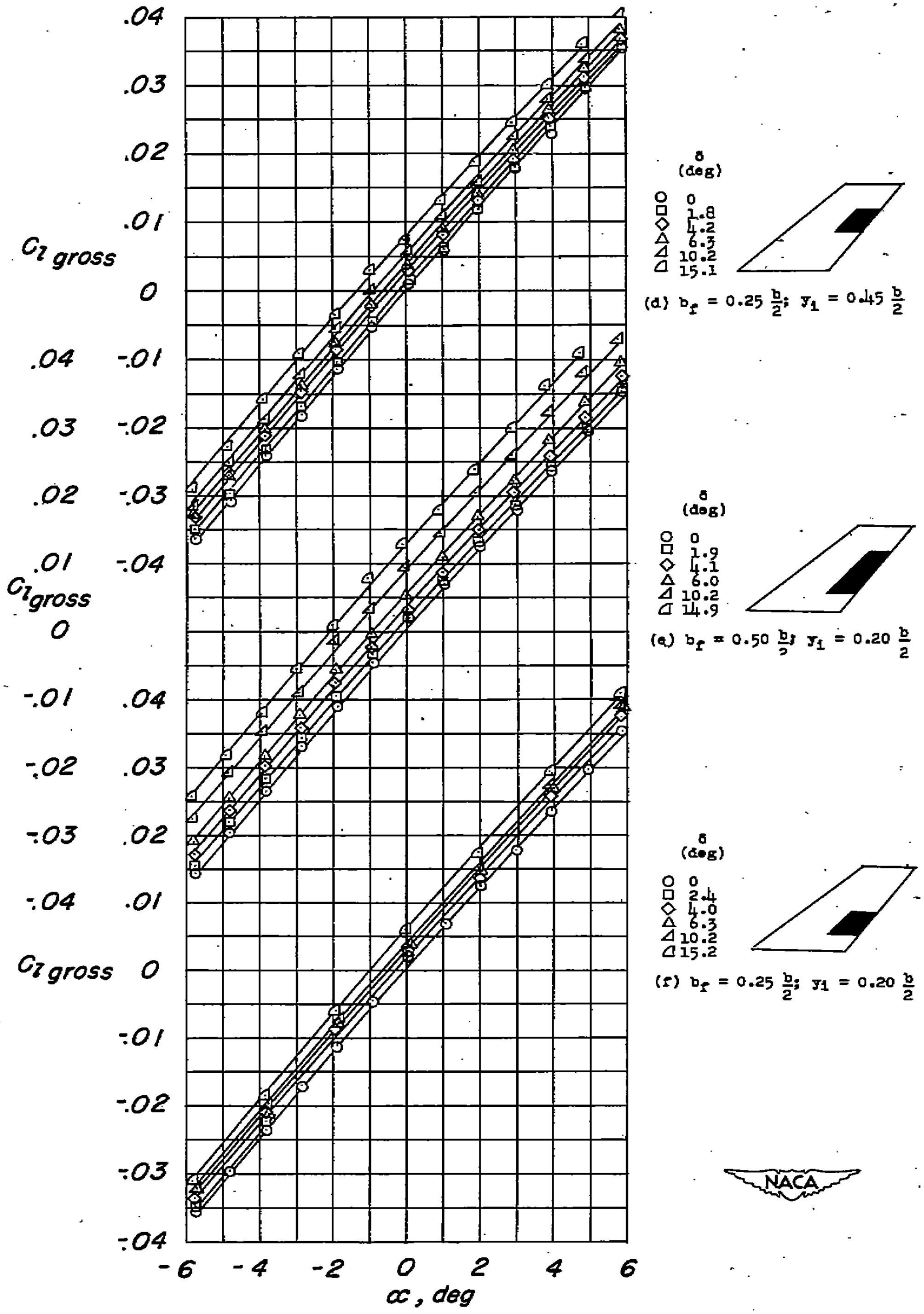


Figure 4.- Concluded.

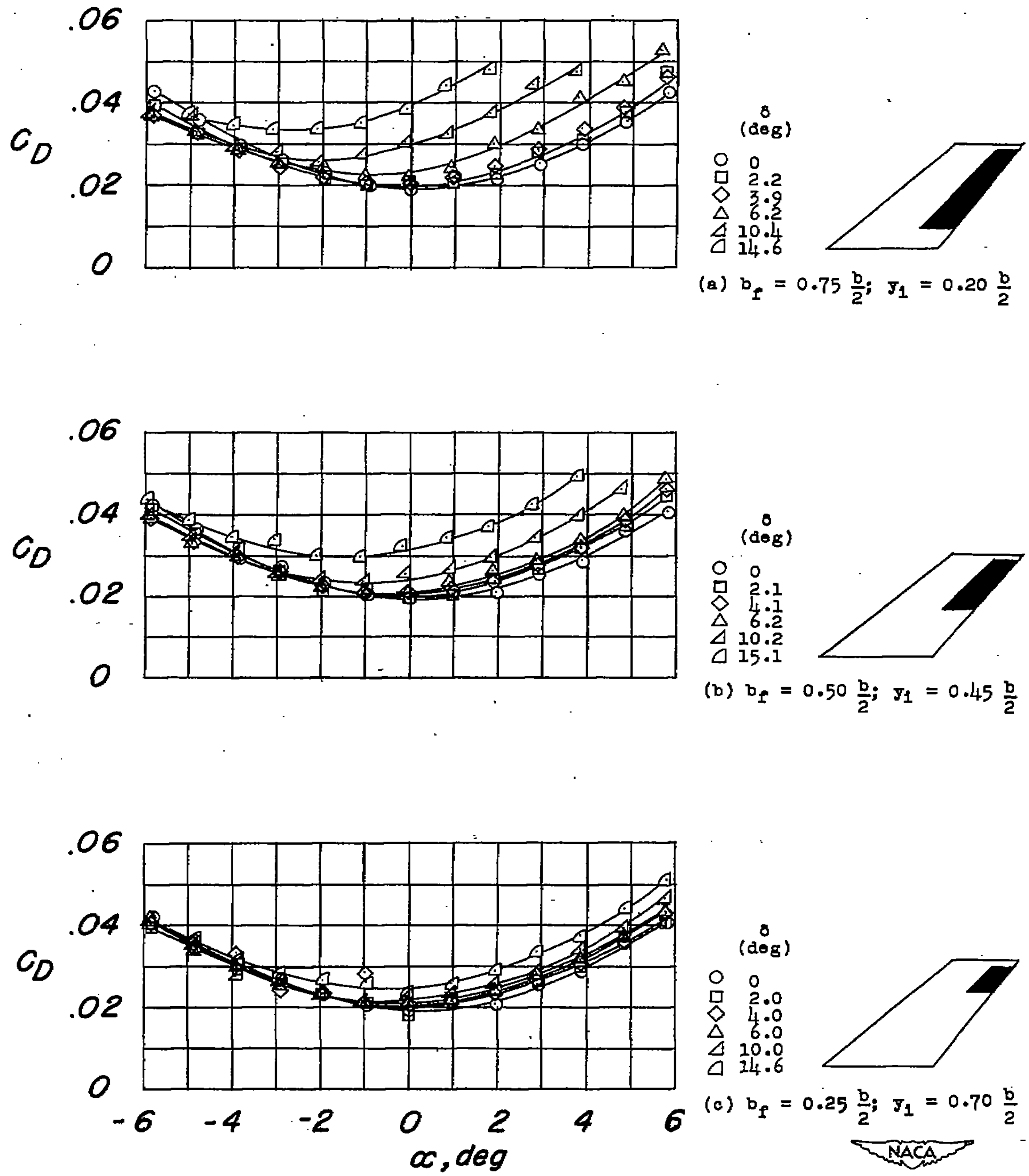


Figure 5.- Drag characteristics of a semispan wing with 45-percent-chord flaps. $R = 2.10 \times 10^6$; $M = 1.9$.

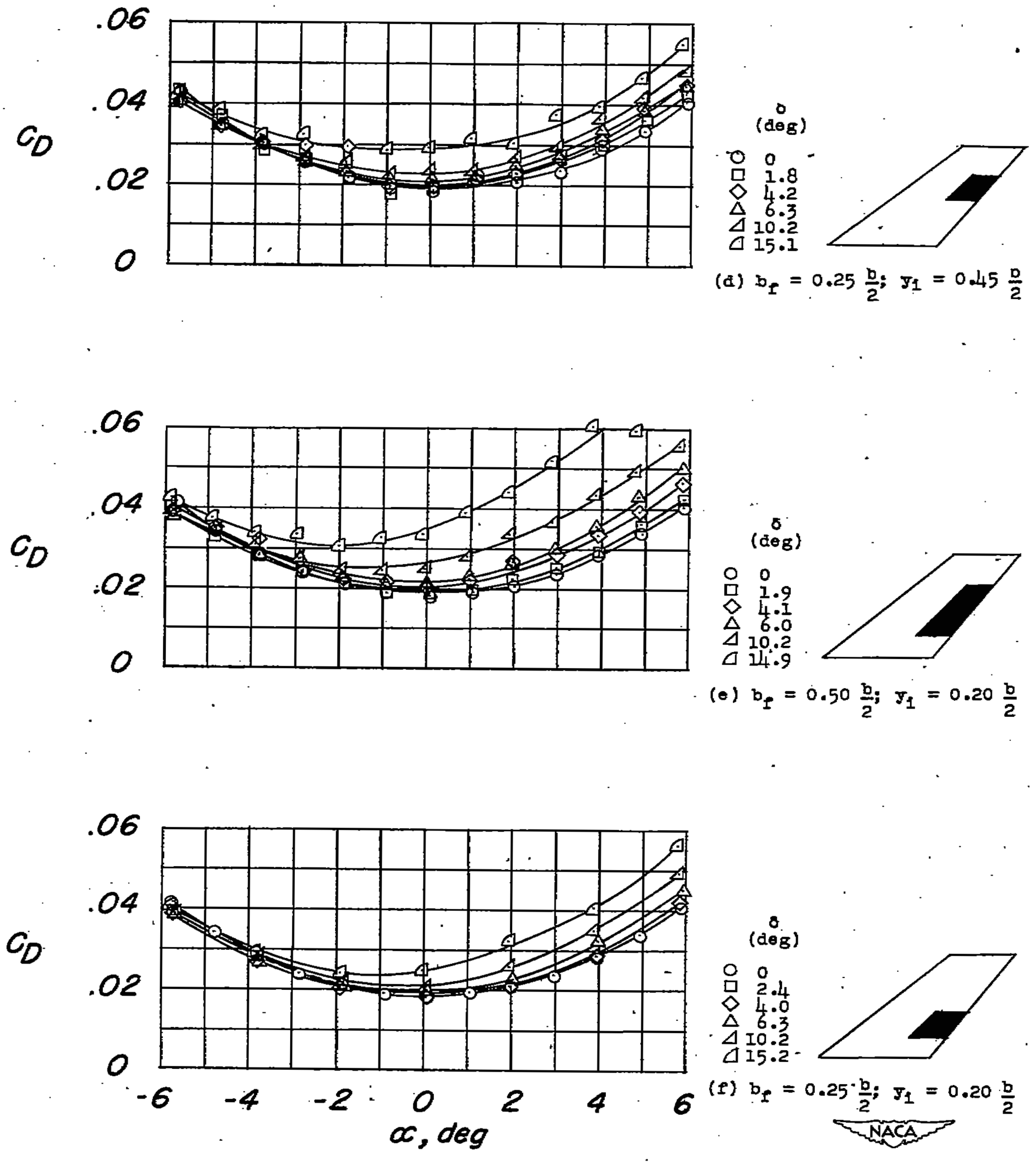


Figure 5.- Concluded.

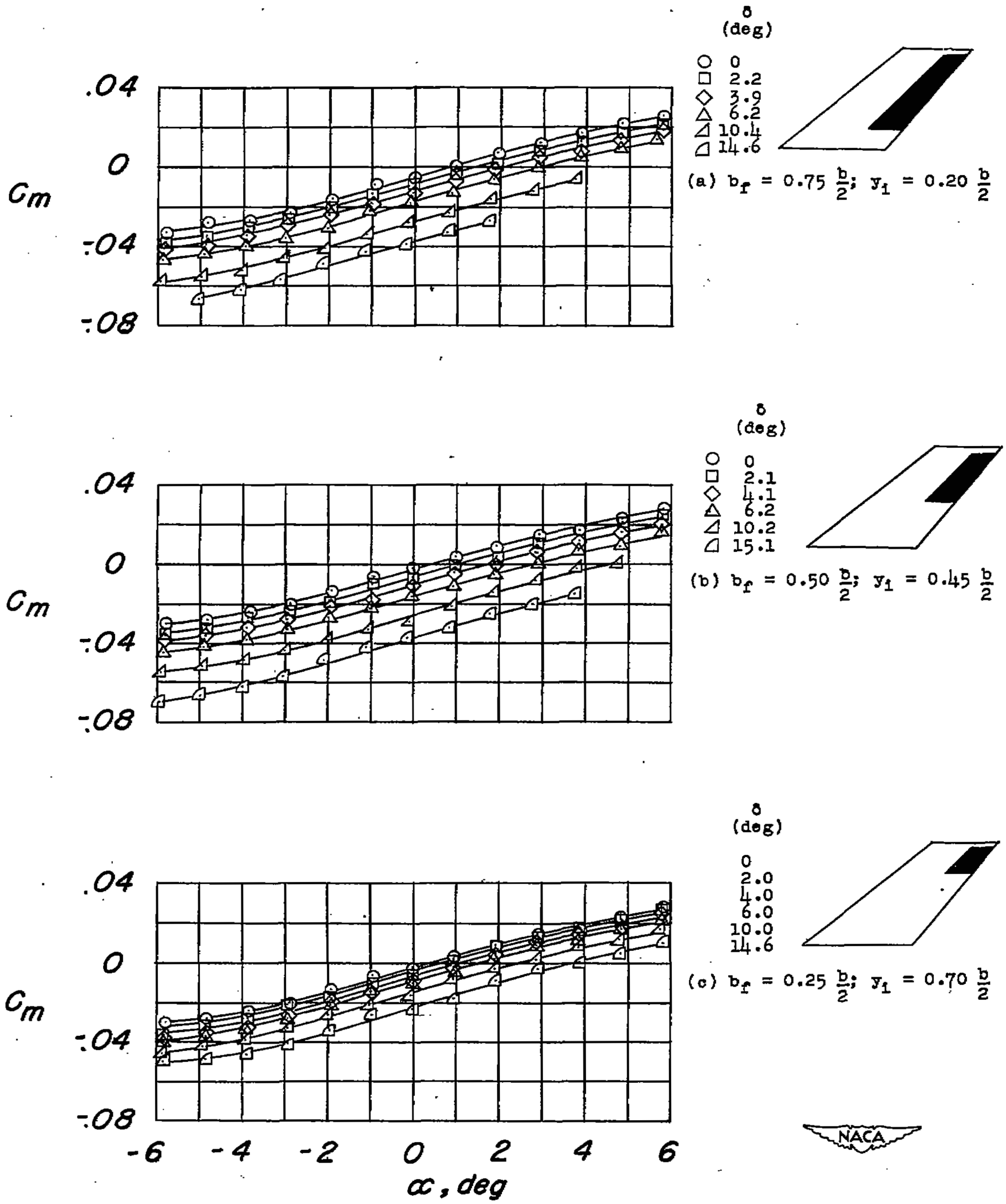


Figure 6.- Pitching-moment characteristics of a semispan wing with 45-percent-chord flaps. $R = 2.10 \times 10^6$; $M = 1.9$.

[redacted]

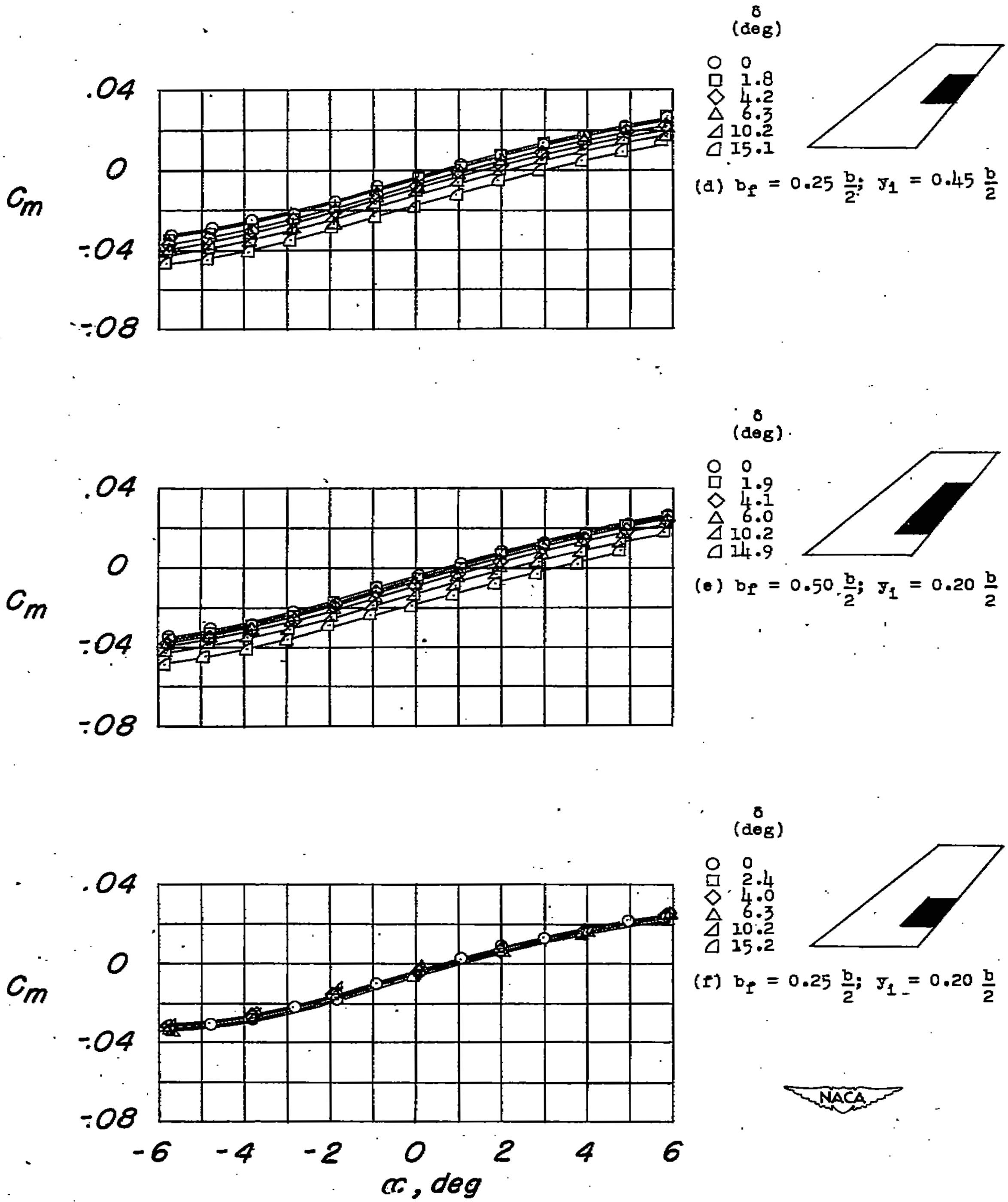


Figure 6.- Concluded.

[redacted]

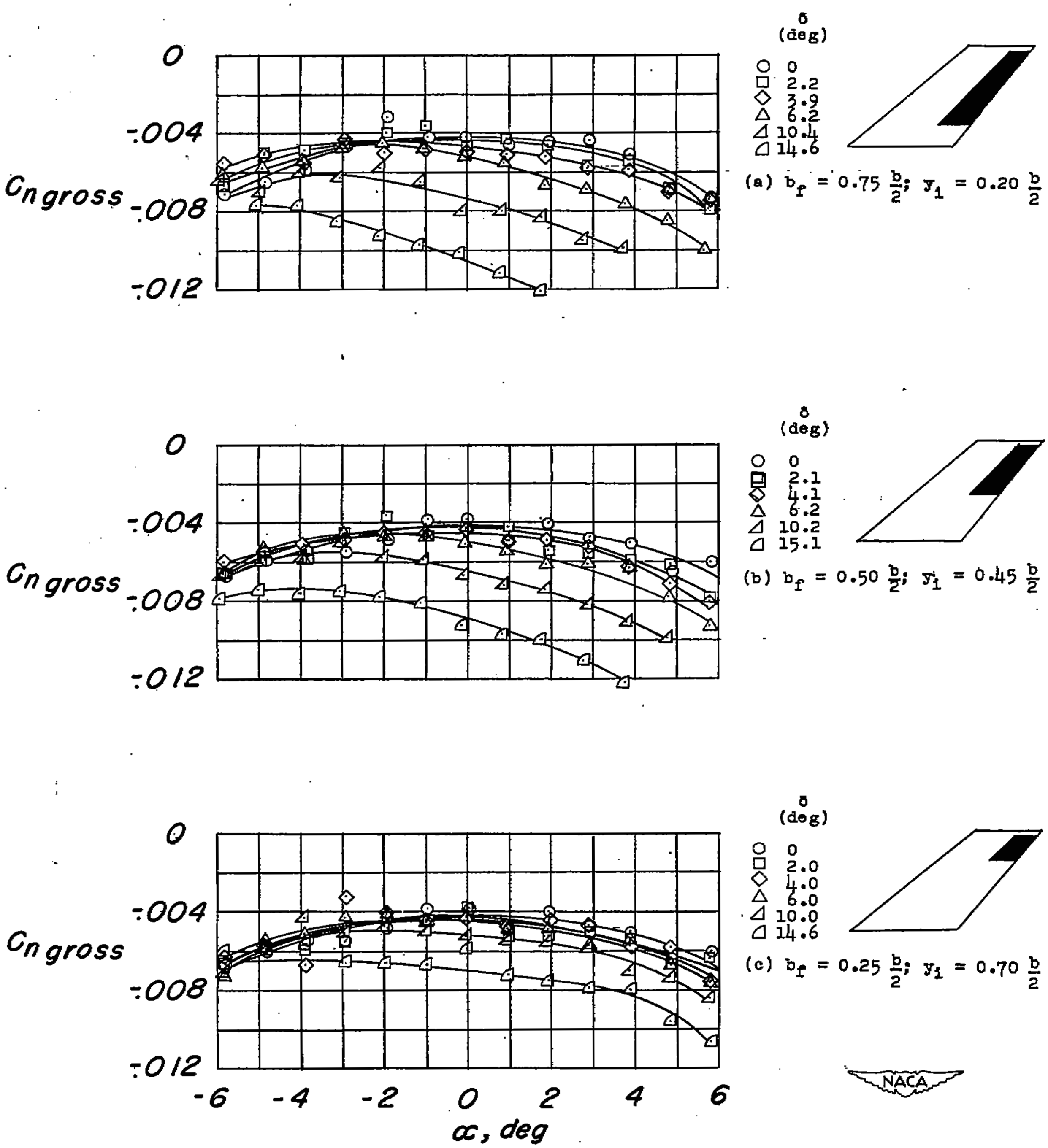


Figure 7.- Yawing-moment characteristics of a semispan wing with 45-percent-chord flaps. $R = 2.10 \times 10^6$; $M = 1.9$.

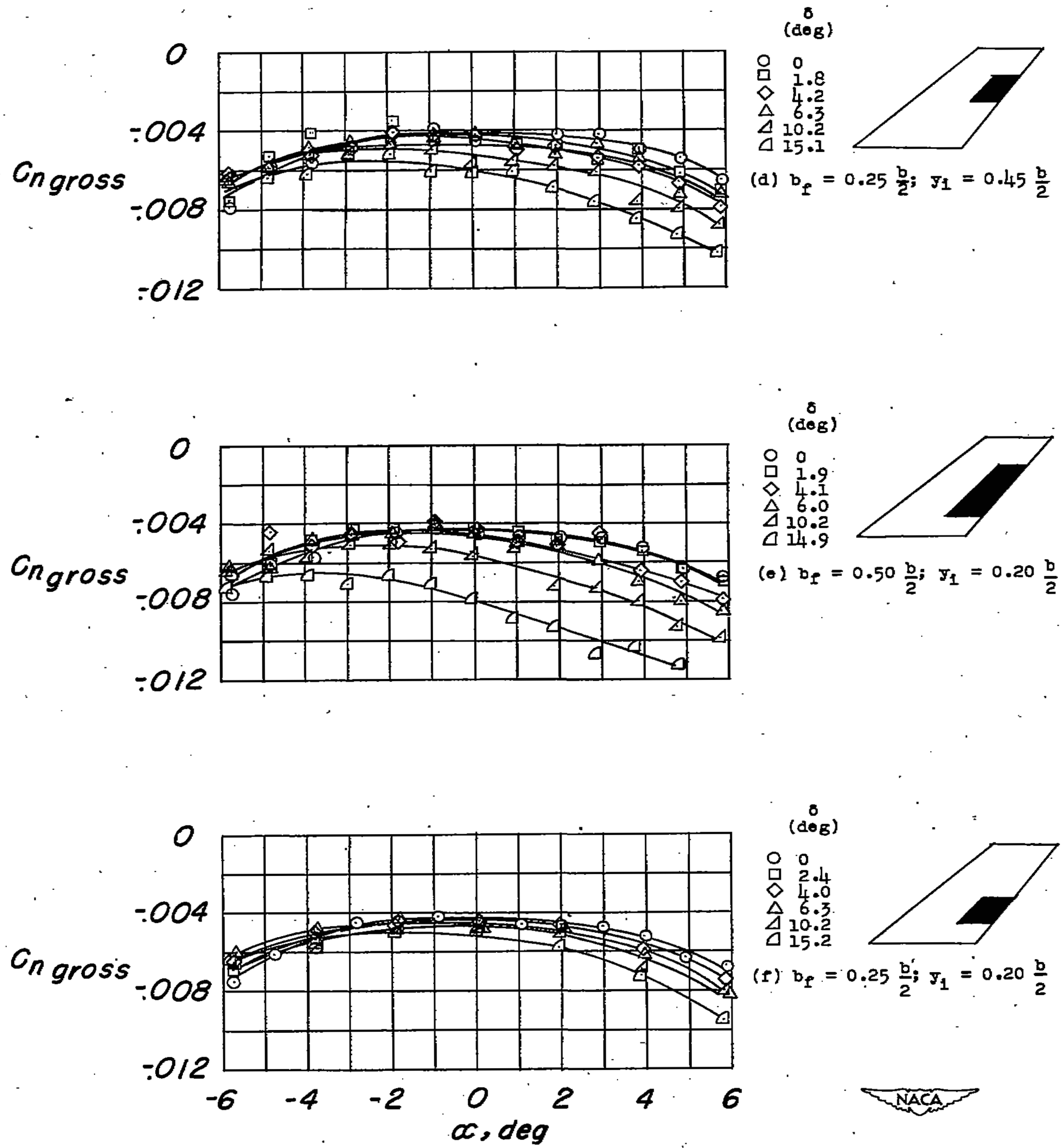


Figure 7.- Concluded.

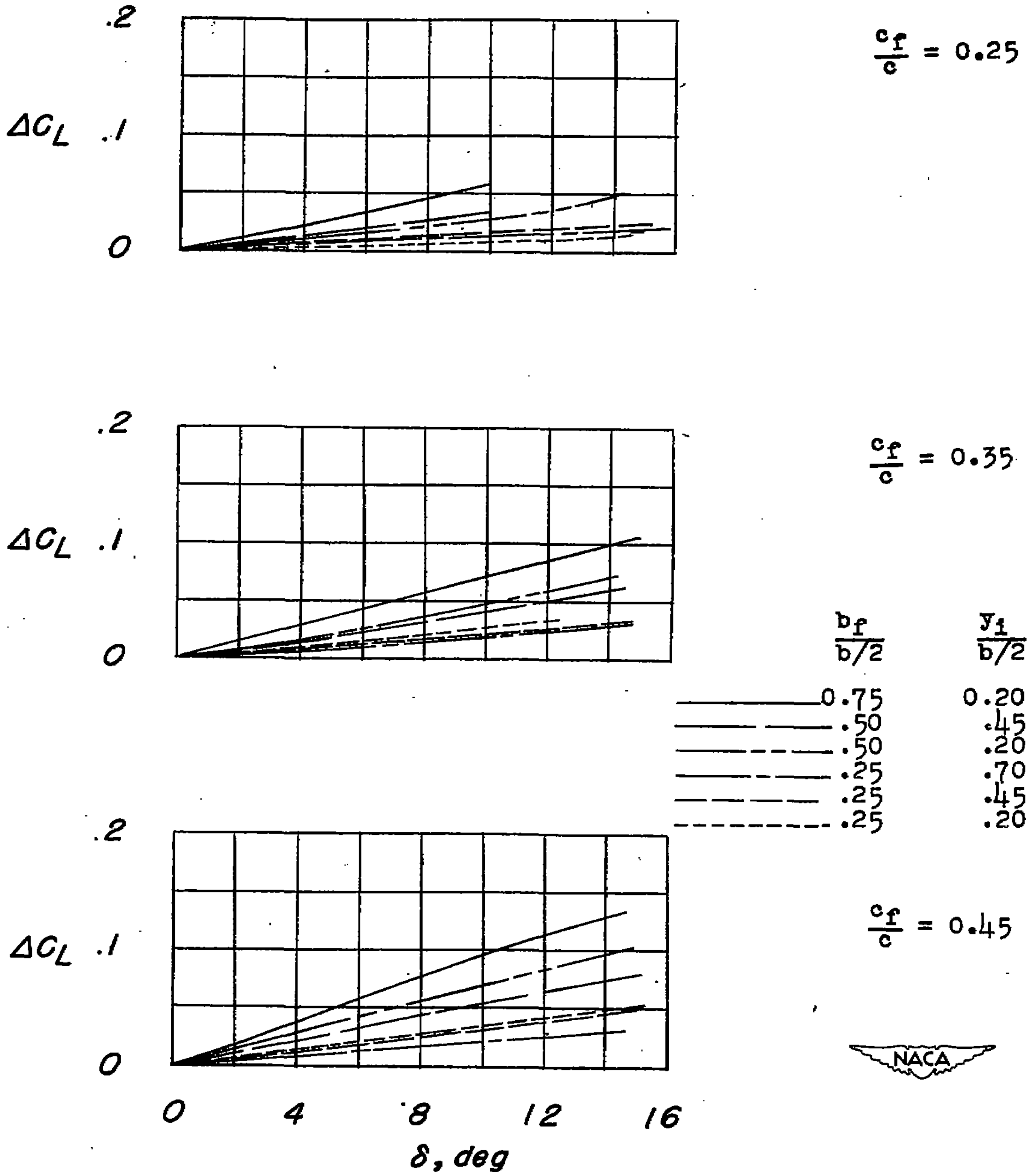


Figure 8.- Lift characteristics due to flap deflection of a semispan wing with 25-, 35-, and 45-percent-chord flaps. $\alpha = 0^{\circ}$; $R = 2.1 \times 10^{6}$; $M = 1.9$.

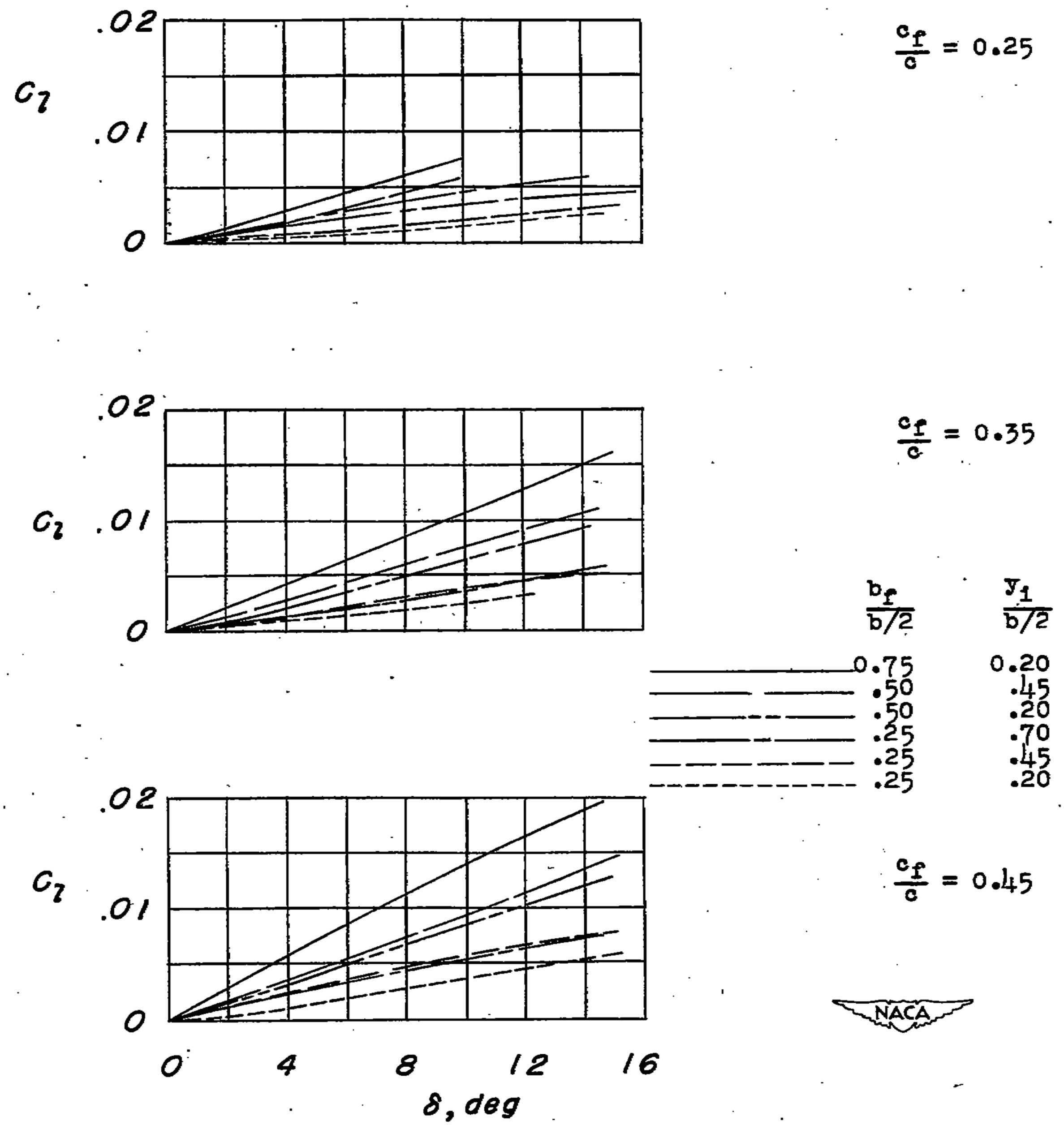


Figure 9.- Rolling-moment characteristics due to flap deflection of a semispan wing with 25-, 35-, and 45-percent-chord flaps. $\alpha = 0^{\circ}$; $R = 2.1 \times 10^{6}$; $M = 1.9$.

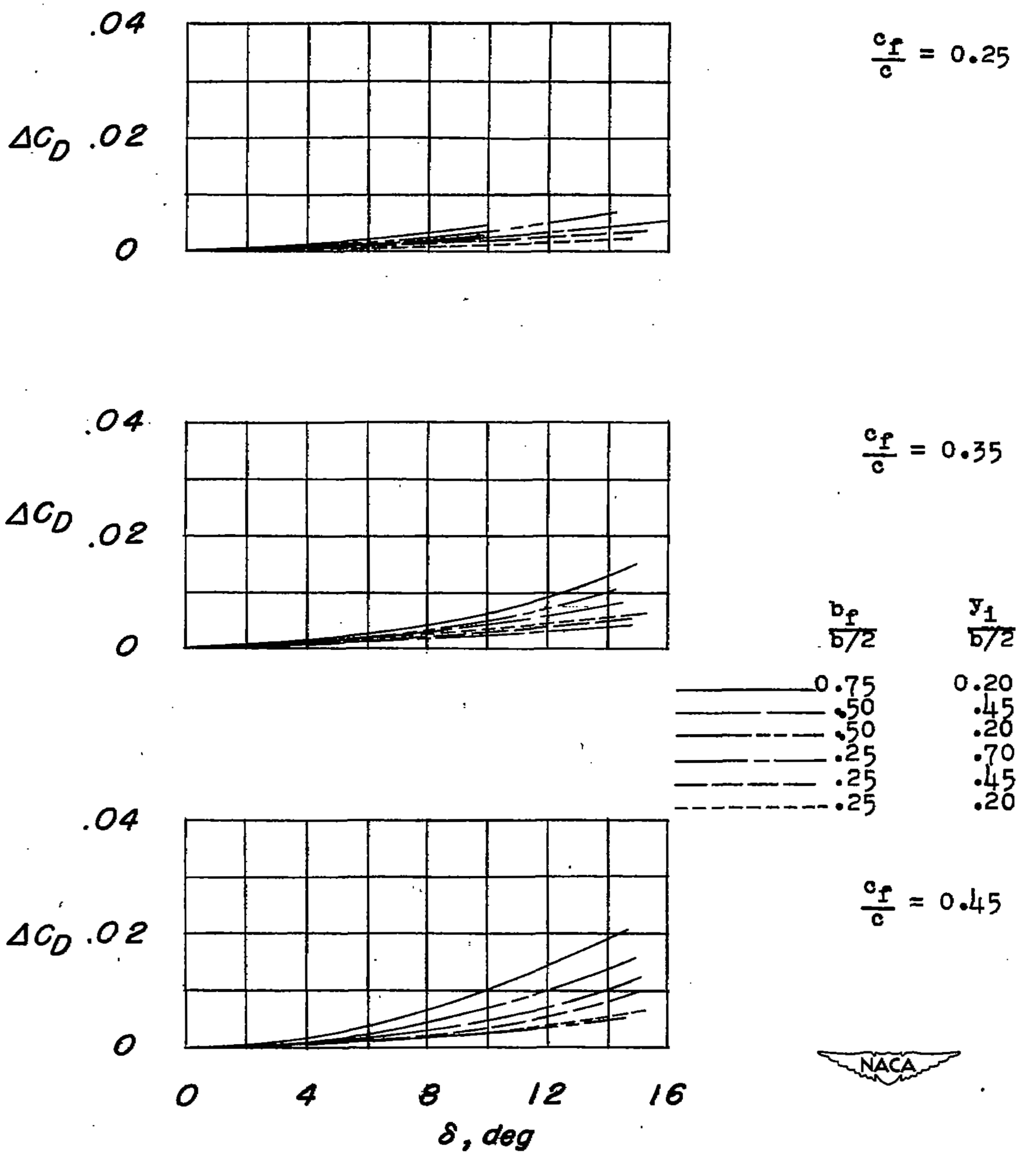


Figure 10.- Drag characteristics due to flap deflection of a semispan wing with 25-, 35-, and 45-percent-chord flaps. $\alpha = 0^{\circ}$; $R = 2.1 \times 10^{6}$; M = 1.9.

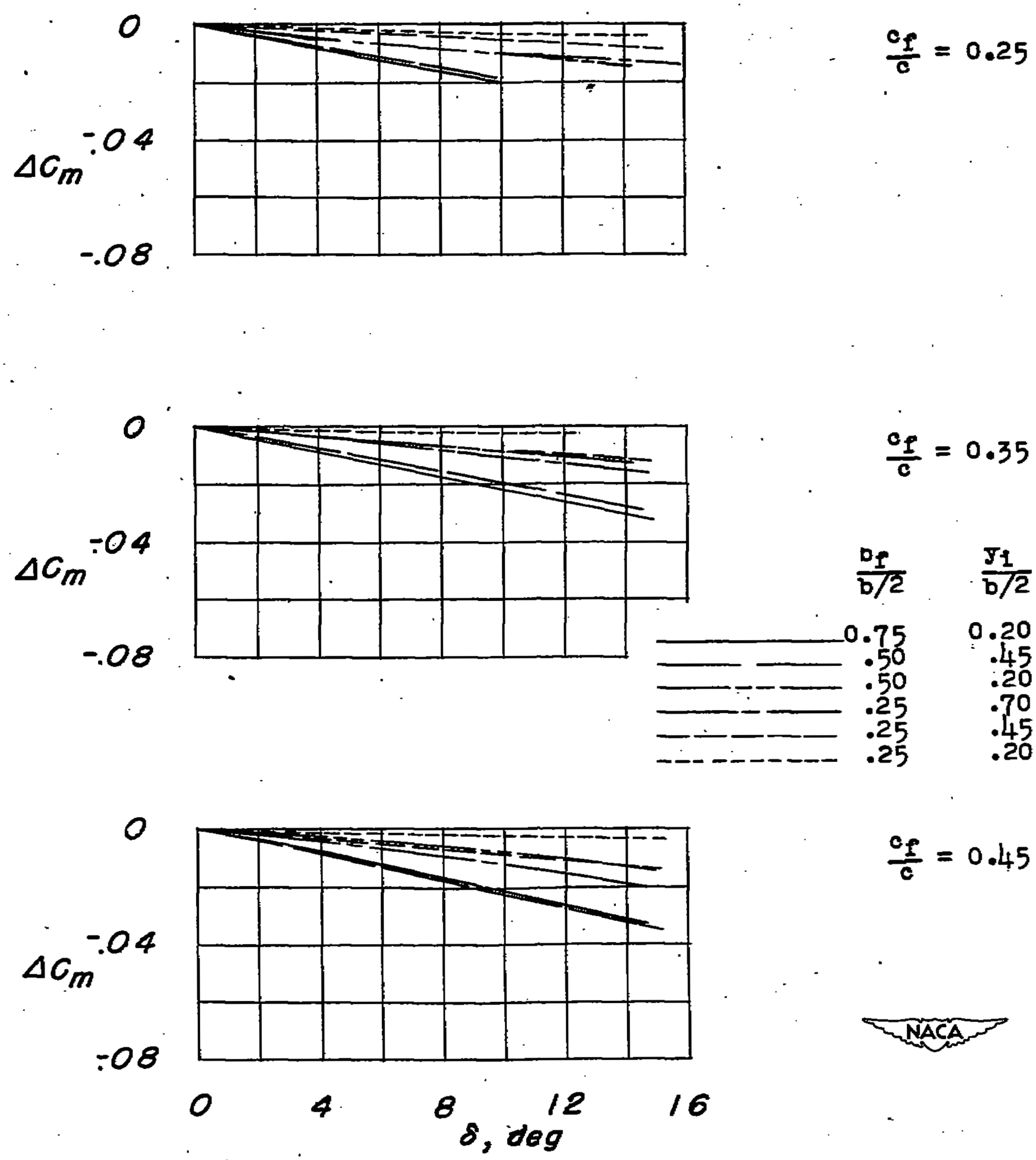


Figure 11.- Pitching-moment characteristics due to flap deflection of a semispan wing with 25-, 35-, and 45-percent-chord flaps. $\alpha = 0^{\circ}$; $R = 2.1 \times 10^6$; $M = 1.9$.

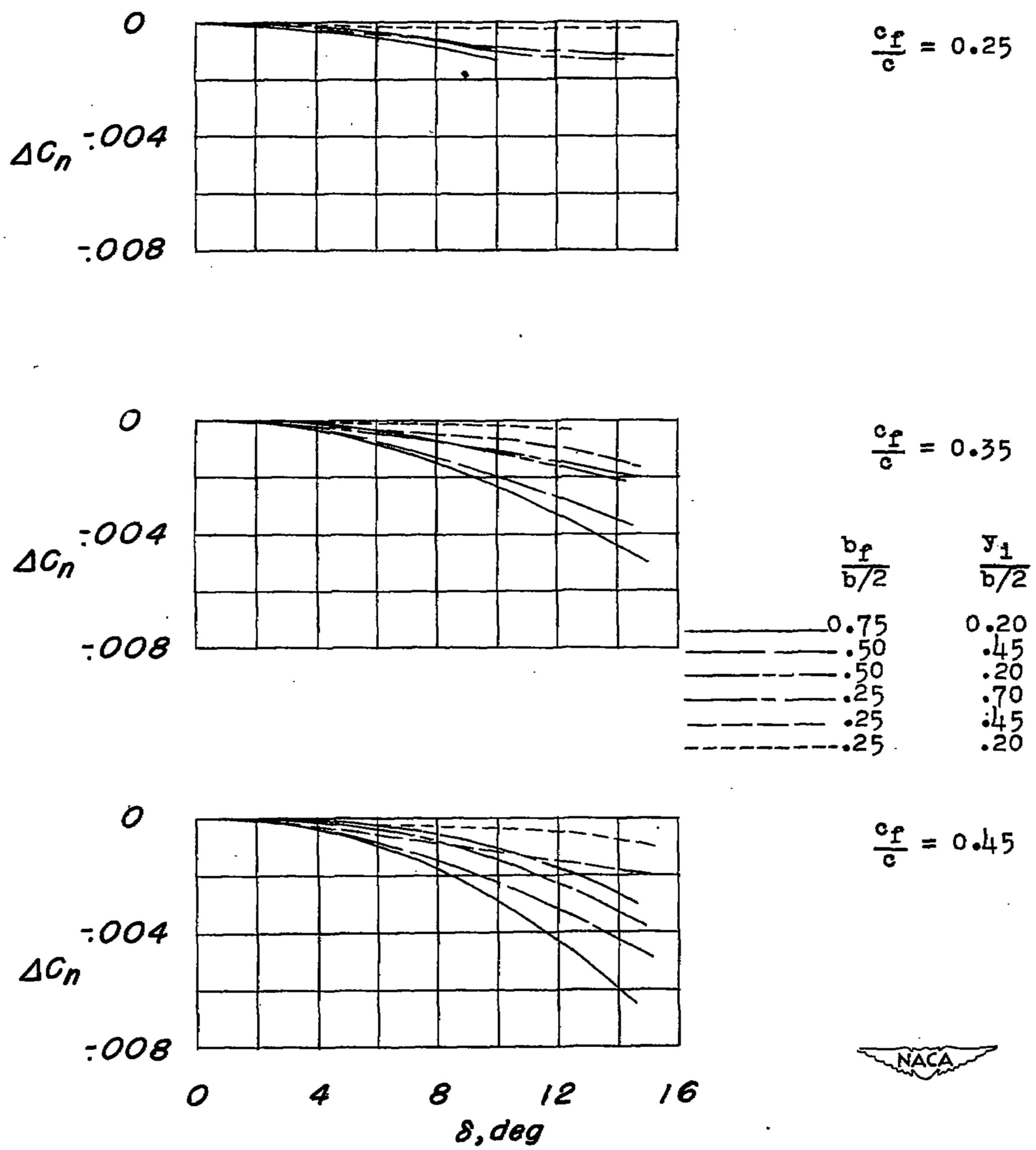


Figure 12.- Yawing-moment characteristics due to flap deflection of a semispan wing with 25-, 35-, and 45-percent-chord flaps. $\alpha = 0^{\circ}$; $R = 2.1 \times 10^{6}$; $M = 1.9$.

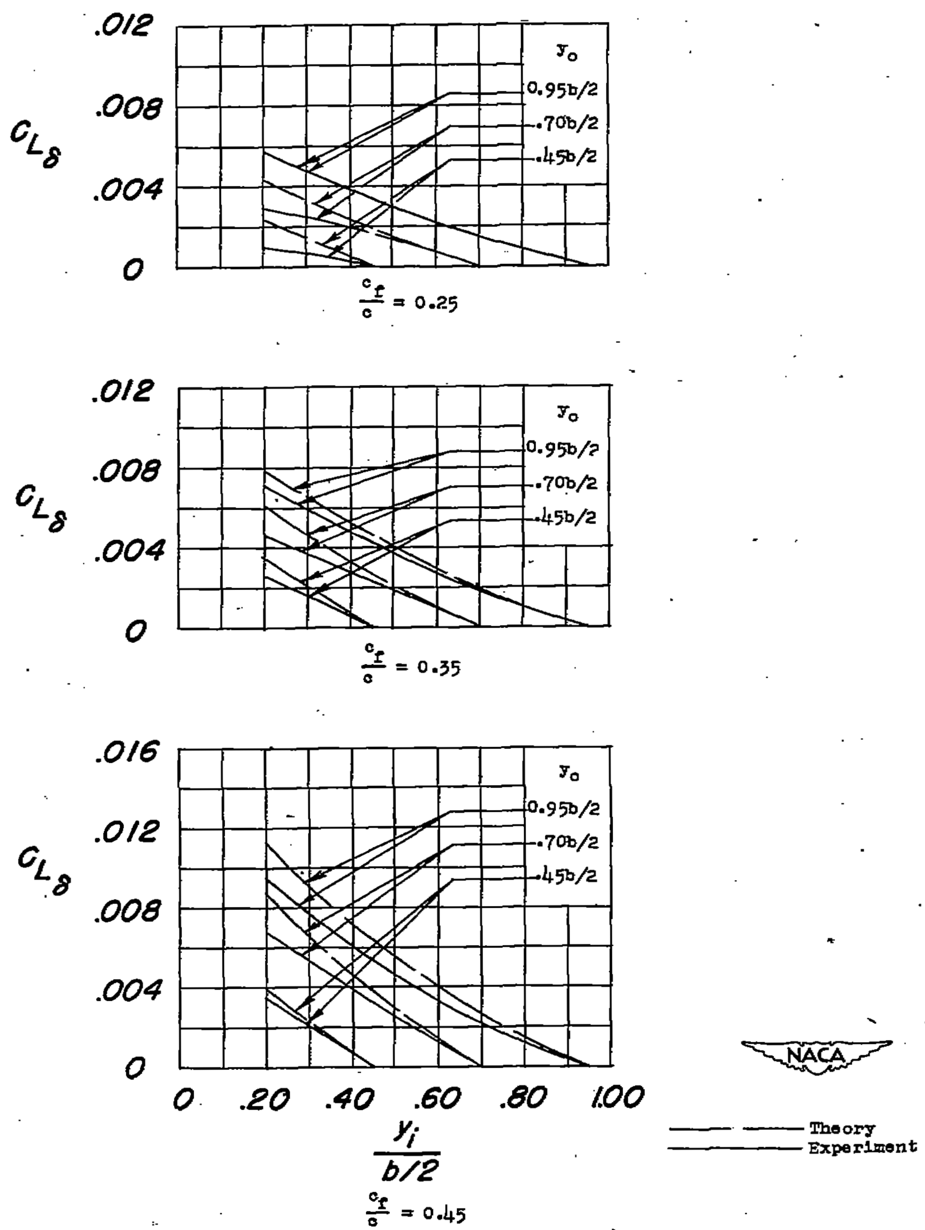


Figure 13.- Effects of varying flap span and flap spanwise location on lift effectiveness parameter C_{L_δ}. $\alpha = 0^\circ$; $R = 2.1 \times 10^6$; $M = 1.9$.

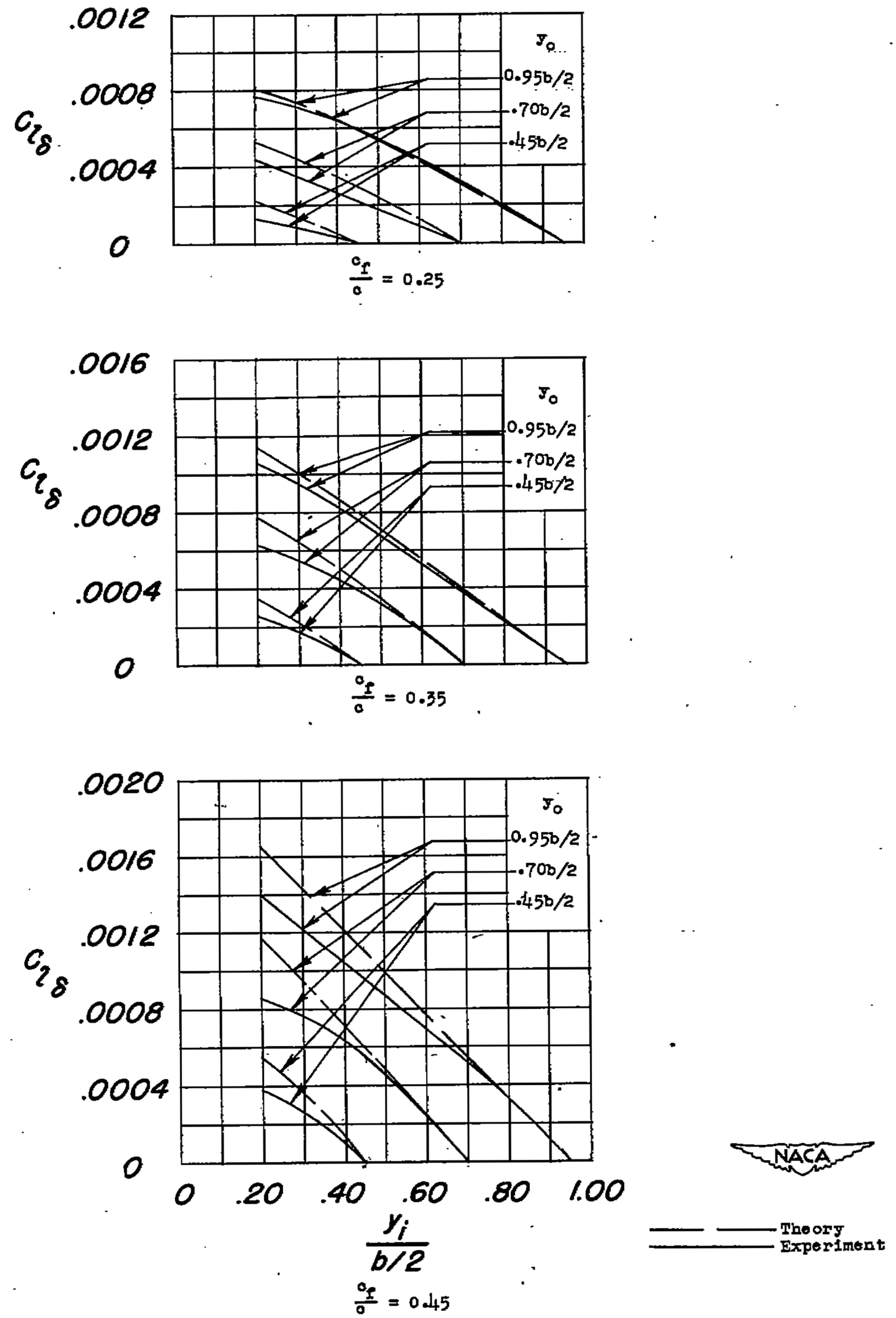


Figure 14.- Effects of varying flap span and flap spanwise location on rolling-moment effectiveness parameter C_{l_δ}. $\alpha = 0^\circ$; $R = 2.1 \times 10^6$; $M = 1.9$.

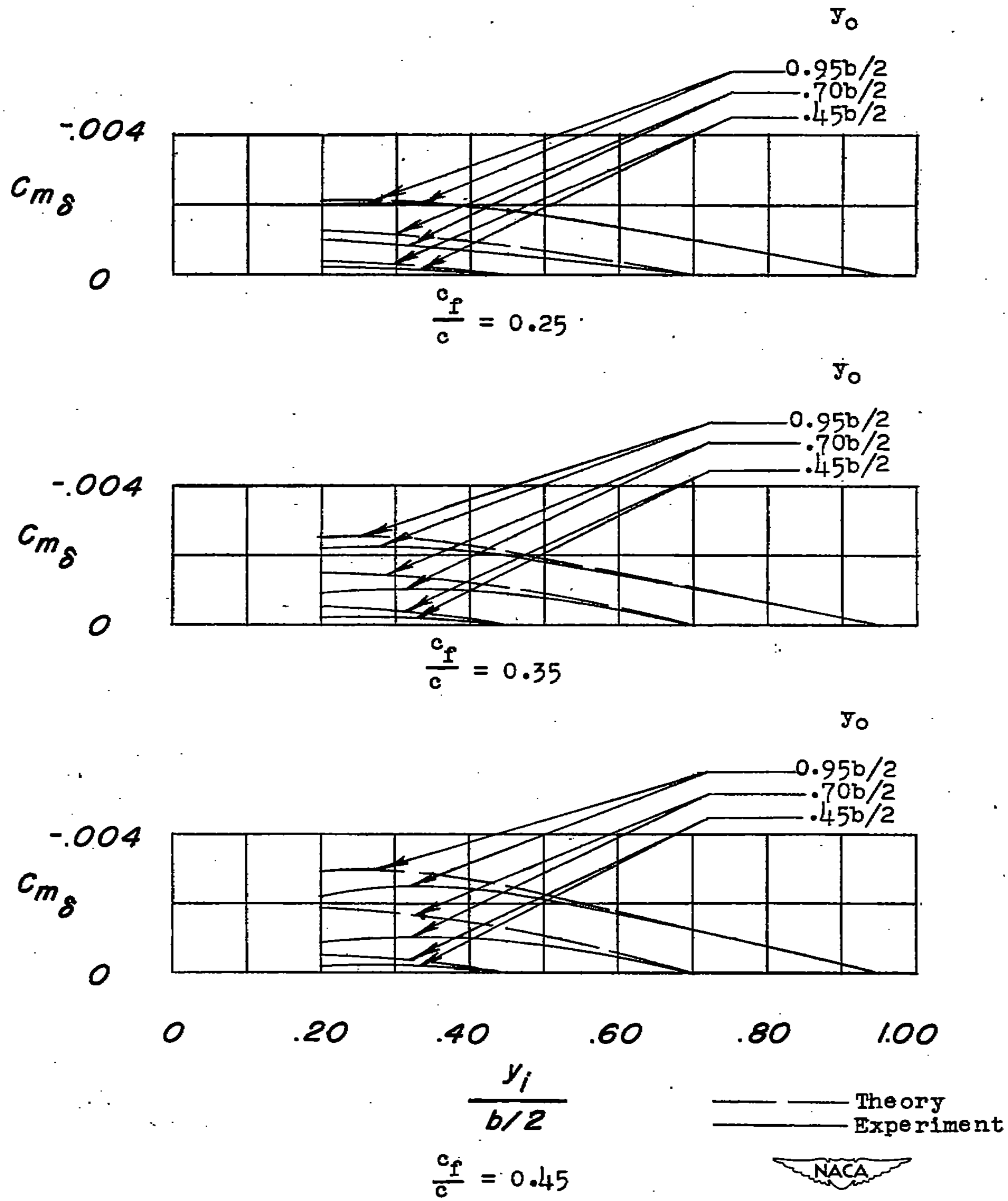


Figure 15.- Effects of varying flap span and flap spanwise location on pitching-moment effectiveness parameter C_{m_δ}. $\alpha = 0^{\circ}$; $R = 2.1 \times 10^{6}$; $M = 1.9$.

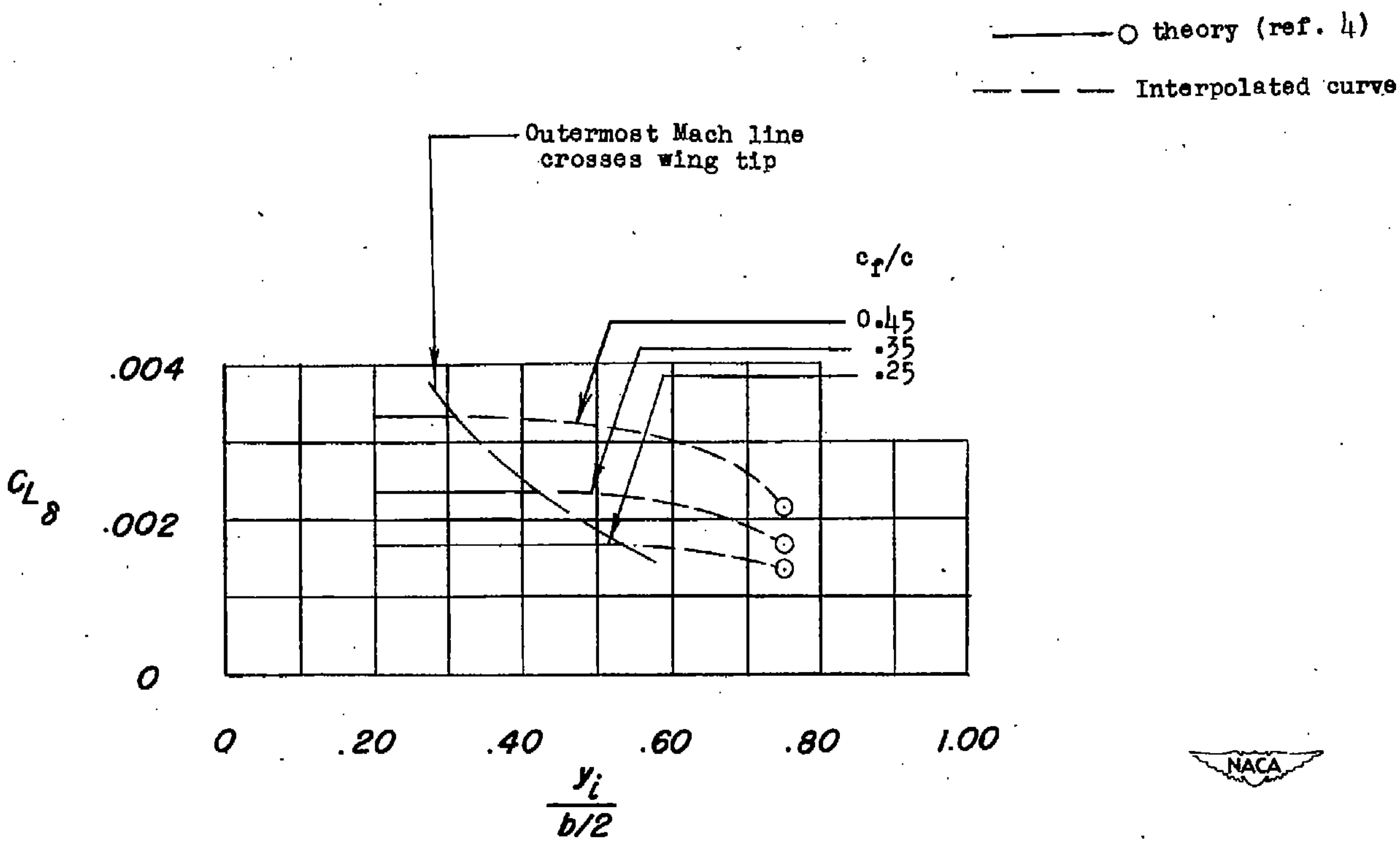


Figure 16.- Calculated effects of varying the location of a constant-percent-chord 25-percent-semispan flap on the lift effectiveness parameter C_{L_δ}. $\alpha = 0^\circ$; M = 1.9.

NACA-Langley - 12-4-51 - 325

SECURITY INFORMATION

[redacted]DENTIAL