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RESEARCH MEMORANDUM

LATERAL-CONTROL INVESTIGATION OF FLAP-TYPE CONTROLS ON A WING WITH QUARTER-CHORD LINE SWEPT BACK 45°, ASPECT RATIO 4, TAPER RATIO 0.6, AND NACA 65A006 AIRFOIL SECTION

TRANSONIC-BUMP METHOD

By

Raymond D. Vogler

Langley Aeronautical Laboratory
Langley Air Force Base, Va.

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SUMMARY

As part of an NACA research program, an investigation by the transonic-bump method through a Mach range of 0.6 to 1.2 has been made in the Langley high-speed 7- by 10-foot tunnel to determine the lateral control characteristics of 30-percent-chord flap-type controls of various spans. The wing of the semispan fuselage-wing combination had 45° of sweepback of the quarter-chord line, a taper ratio of 0.6, an aspect ratio of 4.0, and an NACA 65A006 airfoil section parallel to the free air stream.

Rolling and pitching moments and lift were obtained through a small range of control deflections. The majority of the data are presented as control-effectiveness parameters to show their variation with Mach number. In the Mach number region of 0.85 to 1.0, the results showed a decided decrease in the lift- and rolling-effectiveness parameters and a relatively smaller decrease in the negative values of the pitching-effectiveness parameters.

INTRODUCTION

The urgent need for aerodynamic data in the transonic speed range and the paucity thereof have led to the establishment of an integrated program for transonic research. As part of the NACA transonic research program, a series of wing-fuselage configurations having wing plan form as the chief variable are being investigated in the Langley high-speed 7- by 10-foot tunnel by using the transonic-bump test method.

This paper presents the results of a lateral-control investigation of a semispan wing-fuselage model employing a wing with the quarter-chord line swept back 45°, an aspect ratio of 4, a taper ratio of 0.6, and an NACA 65A006 airfoil section. The purpose of this investigation was to obtain lateral-control data with flap-type controls of 30-percent chord and various spans. The results of a previous investigation of the same wing fuselage without controls, giving additional aerodynamic data, may be found in reference 1.

MODEL AND APPARATUS

The semispan wing had 45° of sweepback at the quarter-chord line, a taper ratio of 0.6, an aspect ratio of 4.0, and an NACA 65A006 airfoil section (reference 2) parallel to the free air stream (fig. 1). The wing was made of beryllium copper and the fuselage was made of brass with all surfaces polished. The wing was mounted in the center of the fuselage vertically and had no dihedral or incidence. The fuselage, which was semicircular in cross section and in conformity with the ordinates of figure 2, was bent to the contour of the bump. For the purpose of determining the effect of fuselage shape on control characteristics, a few tests were made with the wing mounted on a cylindrical body with an ogive nose in anticipation of such a fuselage being used in free-flight tests of the wing. The drawing and the ordinates of the cylindrical body and the location of the quarter chord of the mean aerodynamic chord are also given in figure 2.

The controls (aileron or flap) were made integral with the wing by cutting grooves 0.03-inch wide along the 70-percent-chord line on the upper and lower surfaces of the wing (fig. 3). After setting the control at the desired deflection by bending the metal along the grooves, the grooves were filled with wax, thus giving a close approach to a 30-percent-chord sealed plain flap-type control surface. The entire control from fuselage surface to wing tip was divided into four equal spanwise segments. (See fig. 3.)

The model was mounted on an electrical strain-gage balance wired to calibrated galvanometers in order to measure the aerodynamic forces and moments. The balance was mounted in a chamber within the bump, and the chamber was sealed except for a small rectangular hole through which an extension of the wing passed. This hole was covered by the fuselage end plate which was approximately 0.03 inch above the bump surface.

COEFFICIENTS AND SYMBOLS

C_L lift coefficient $\left(\frac{\text{Twice lift of semispan model}}{qS}\right)$

C_l rolling-moment coefficient at plane of symmetry $\left(\frac{\text{Rolling moment of semispan model}}{qSb}\right)$

C_{l_a} rolling-moment coefficient produced by the control (rolling-moment coefficient with control deflected minus rolling-moment coefficient without deflection)

C_m pitching-moment coefficient referred to $0.25\bar{c}$ $\left(\frac{\text{Twice pitching moment of semispan model}}{qS\bar{c}}\right)$.

q effective dynamic pressure over span of model, pounds per square foot $\left(\frac{1}{2}\rho V^2\right)$

S twice wing area of semispan model, 0.125 square foot

b twice span of semispan model, 0.707 foot

$\bar{c}$ mean aerodynamic chord of wing, 0.180 foot $\left(\frac{2}{S}\int_0^{b/2} c^2 dy\right)$

c local wing chord, feet

y spanwise distance from plane of symmetry

y_i spanwise distance from plane of symmetry to inboard end of control

ρ	mass density of air, slugs per cubic foot
V	free-stream air velocity, feet per second
M	effective Mach number over span of model
M_a	average chordwise local Mach number
M_l	local Mach number
R	Reynolds number of wing based on $\bar{c}$
α	angle of attack, degrees
δ	control deflection relative to wing-chord plane, measured perpendicular to control hinge axis (positive when trailing edge is down), degrees
b_a	control span measured perpendicular to plane of symmetry

$$\left.\begin{aligned} C_{L_\delta} &= \left(\frac{\partial C_L}{\partial \delta}\right)_\alpha \\ C_{l_\delta} &= \left(\frac{\partial C_l}{\partial \delta}\right)_\alpha \\ C_{m_\delta} &= \left(\frac{\partial C_m}{\partial \delta}\right)_\alpha \end{aligned}\right\} \text{The subscript } \alpha \text{ indicates the factor held constant.}$$

CORRECTIONS

The rolling-effectiveness parameters presented herein represent the aerodynamic effects on a complete wing produced by the deflection of the control on only one semispan of the complete wing. Reflection-plane corrections have been applied to the data throughout the Mach range tested. The correction factors which were applied to the parameters are given in figure 4. The values of the correction factors given in figure 4 were obtained from unpublished experimental low-speed data and theoretical considerations. Although the corrections were based on low-speed considerations and are valid for the low Mach numbers only, it was believed that the results obtained by applying the corrections would give a better representation of true conditions than uncorrected data.

The lift- and pitching-effectiveness parameters represent the aerodynamic effects of deflection in the same direction of the controls on both semispans of the complete wing, and hence no reflection-plane corrections are necessary for the lift and pitching-moment data.

No corrections were applied for any twisting or deflection of the wing caused by the air load. These effects were believed to be small, however.

TEST TECHNIQUE

The tests were made in the Langley high-speed 7- by 10-foot tunnel using an adaptation of the NACA wing-flow technique for obtaining transonic speeds. The technique used involves placing the model in the high-velocity flow field generated over the curved surface of a bump on the tunnel floor (reference 3). Typical contours of local Mach number in the vicinity of the model location on the bump with model removed are shown in figure 5. The contours indicate that there is a Mach number variation of about 0.04 over the wing semispan at low Mach numbers and about 0.07 at high Mach numbers. The chordwise Mach number variation is generally less than 0.01. The effective Mach number over the wing semispan is estimated to be 0.02 higher than the effective Mach number where 50-percent-span outboard ailerons normally would be located. No attempt has been made to evaluate the effects of this chordwise and spanwise Mach number variation. The long-dash lines near the root of the wing in figure 5 indicate a local Mach number 5 percent below the maximum value and represent the estimated extent of the bump boundary layer. The effective test Mach number was obtained from contour charts similar to those presented in figure 5 by using the relationship

$$M = \frac{2}{S}\int_0^{b/2} cM_a \, dy$$

The variation of the mean test Reynolds number with Mach number is shown in figure 6. The boundaries on the figure are an indication of the probable range in Reynolds number caused by variations in test conditions during the course of the investigation.

Force and moment data were obtained with controls of various spans through a Mach number range of 0.60 to 1.20, an angle-of-attack range of -6° to 6°, and a control-deflection range of 0° to 10°, plus same data on the 43-percent-span outboard control up to a deflection of 30°. Some additional tests were made with the cylindrical body in place of the transonic-research fuselage.

RESULTS AND DISCUSSION

The data presented were obtained using the wing-fuselage combination except in figure 15 where the cylindrical body is used. In figures 7, 8, and 9 are curves of lift, rolling-, and pitching-moment coefficients plotted against control deflection up to 30° for the outboard 43-percent-span control at a wing angle of attack of 2°. In all other configurations the maximum control deflection was 10°. Inasmuch as the wing was symmetrical, data obtained at negative angles of attack and positive control deflections were considered, with appropriate regard to signs, to be equivalent to data that would be obtained at positive angles of attack and negative control deflections and were plotted as such. The curves of figures 7 to 9 are typical of the curves of each of the other control configurations tested.

Control-effectiveness parameters.- The control-effectiveness parameters presented in figures 10 to 12 were obtained from figures 7 to 9 and similar plots of the test data for the various control configurations. The control effectiveness for all configurations had a linear variation with control deflection for the deflection range of ±10°, and it was within this range that the slopes were measured.

A marked decrease in rolling and lift effectiveness occurs between Mach numbers of 0.85 and 1.0, and a relatively smaller decrease in the negative values of the pitching-effectiveness parameter occurs in the same Mach number region (figs. 10 to 12).

The effectiveness of controls of various spans starting at the tip (fig. 13) indicates that the outboard 21-percent-span control gives very low rolling effectiveness. Although there are considerable differences in rolling effectiveness for a given span control with increasing Mach number, the general shape of the curves remains the same. This would indicate that the relative effectiveness of a partial-span control to a full-span control is little affected by Mach number. On the other hand, the pitching-effectiveness data (fig. 13) indicate greater relative loss in effectiveness at supersonic Mach numbers for controls near the wing tip than for controls near the root.

A comparison of the values of C_{l_δ} obtained at low Mach numbers in this investigation with those estimated by the method of reference 4 shows fair agreement (fig. 14).

Effect of fuselage shape.– A comparison of the rolling-moment coefficients resulting from a 5° deflection of both a 43-percent-span and 86-percent-span outboard control on the wing with the regular transonic-research fuselage and with the cylindrical body of figure 2 shows little effect of fuselage shape. (See fig. 15.) A decrease in effectiveness of the 86-percent-span control can be noted for angles of attack from -4° to -6° when the cylindrical body is used, otherwise the differences are small and within the experimental accuracy of the tests.

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

REFERENCES

1. Weil, Joseph, and Goodson, Kenneth W.: Aerodynamic Characteristics of a Wing with Quarter-Chord Line Swept Back 45°, Aspect Ratio 4, Taper Ratio 0.6, and NACA 65A006 Airfoil Section. Transonic-Bump Method. NACA RM L9A21, 1949.

2. Loftin, Laurence K., Jr.: Theoretical and Experimental Data for a Number of NACA 6A-Series Airfoil Sections. NACA TN 1368, 1947.

3. Schneiter, Leslie E., and Ziff, Howard L.: Preliminary Investigation of Spoiler Lateral Control on a 42° Sweptback Wing at Transonic Speeds. NACA RM L7F19, 1947.

4. Lowry, John G., and Schneiter, Leslie E.: Estimation of Effectiveness of Flap-Type Controls on Sweptback Wings. NACA TN 1674, 1948.

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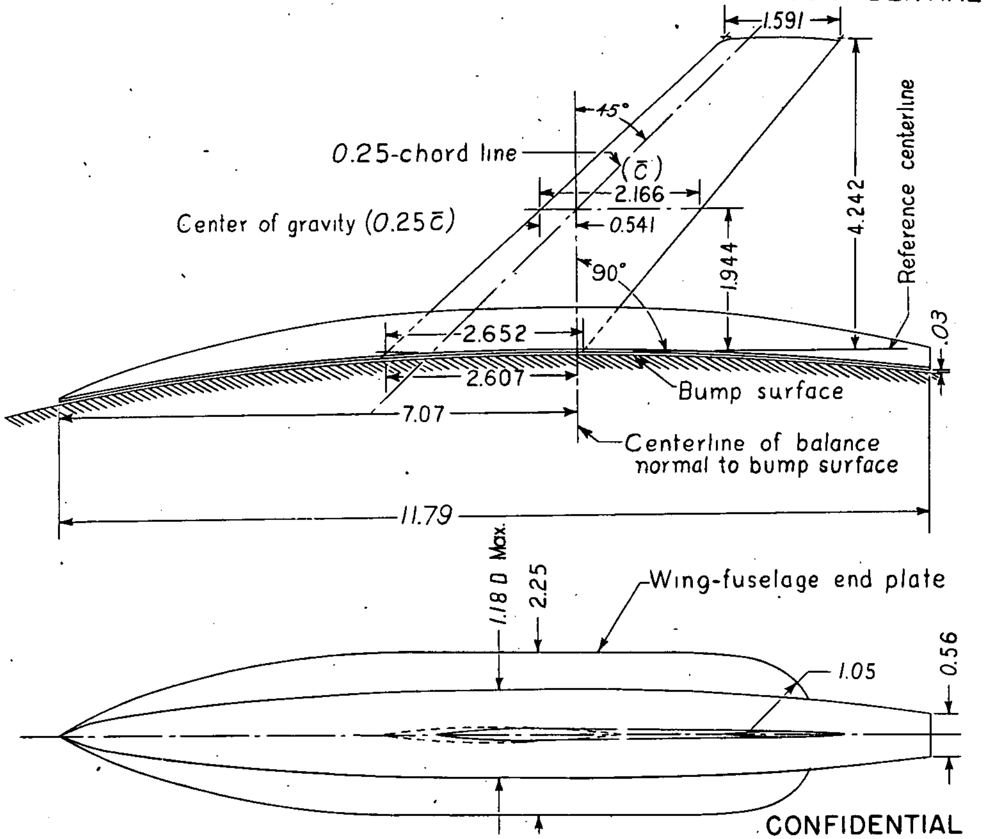

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TABULATED DATA

Wing	
Twice semispan area	0.125 sq ft
Aspect ratio	4.0
Taper ratio	0.60
Mean aerodynamic chord	0.180 ft
Incidence	0°
Dihedral	0°
Airfoil section parallel to free airstream	NACA 65A006

0 1 2
Scale, inches

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Figure 1.– General arrangement of model with 45° sweptback wing, aspect ratio 4, taper ratio 0.6, and NACA 65A006 airfoil.

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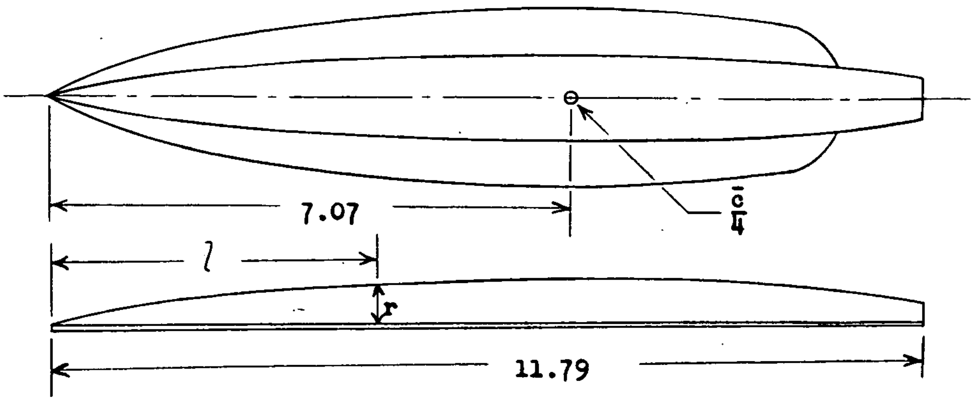
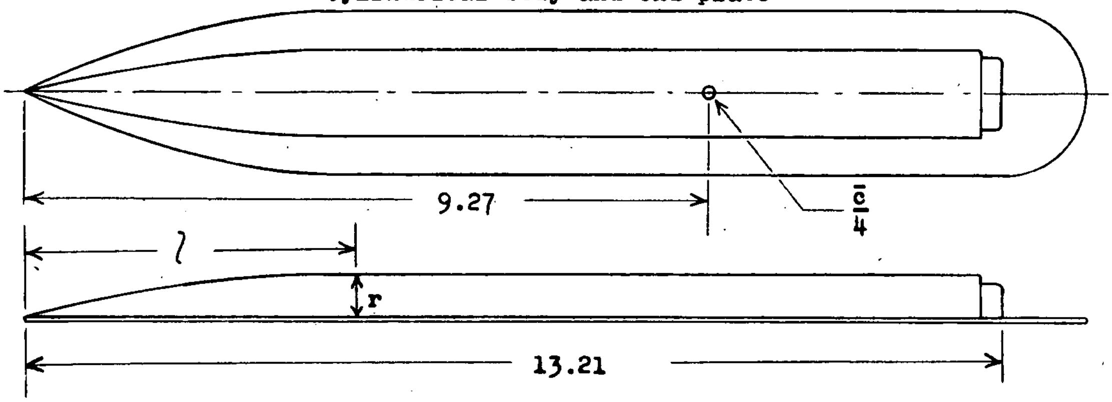
Transonic-research fuselage and end plate



Cylindrical body and end plate



Transonic-research fuselage

Ordinates, inches			
l	r	l	r
0	0	5.657	.575
.071	.033	6.364	.586
.106	.042	7.071	.589
.177	.060	7.778	.584
.354	.102	8.485	.569
.707	.170	9.192	.543
1.061	.228	9.899	.504
1.414	.279	10.606	.442
2.121	.367	11.314	.357
2.828	.437	11.792	.283
3.535	.490		
4.243	.529		
4.950	.556		

Cylindrical body

Ordinates, inches	
l	r
0	0
.59	.144
1.18	.271
1.77	.372
2.36	.462
2.95	.533
3.54	.575
4.13	.589
7.00	.589
10.50	.589
12.92	.589
12.92	.495
13.21	.495

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Figure 2.– Drawings and ordinates of the fuselage and cylindrical body.

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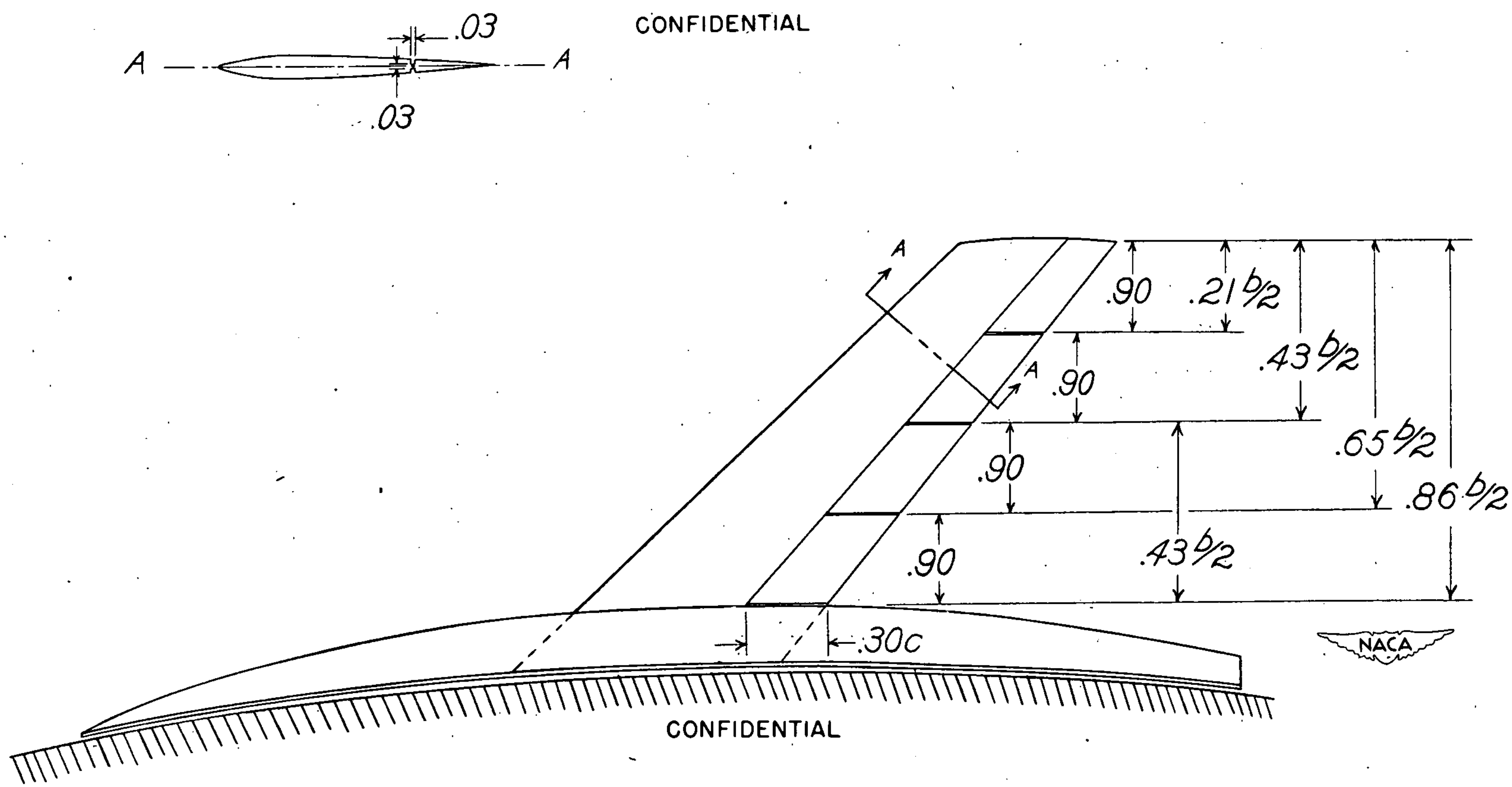

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Figure 3.- Details of controls tested.

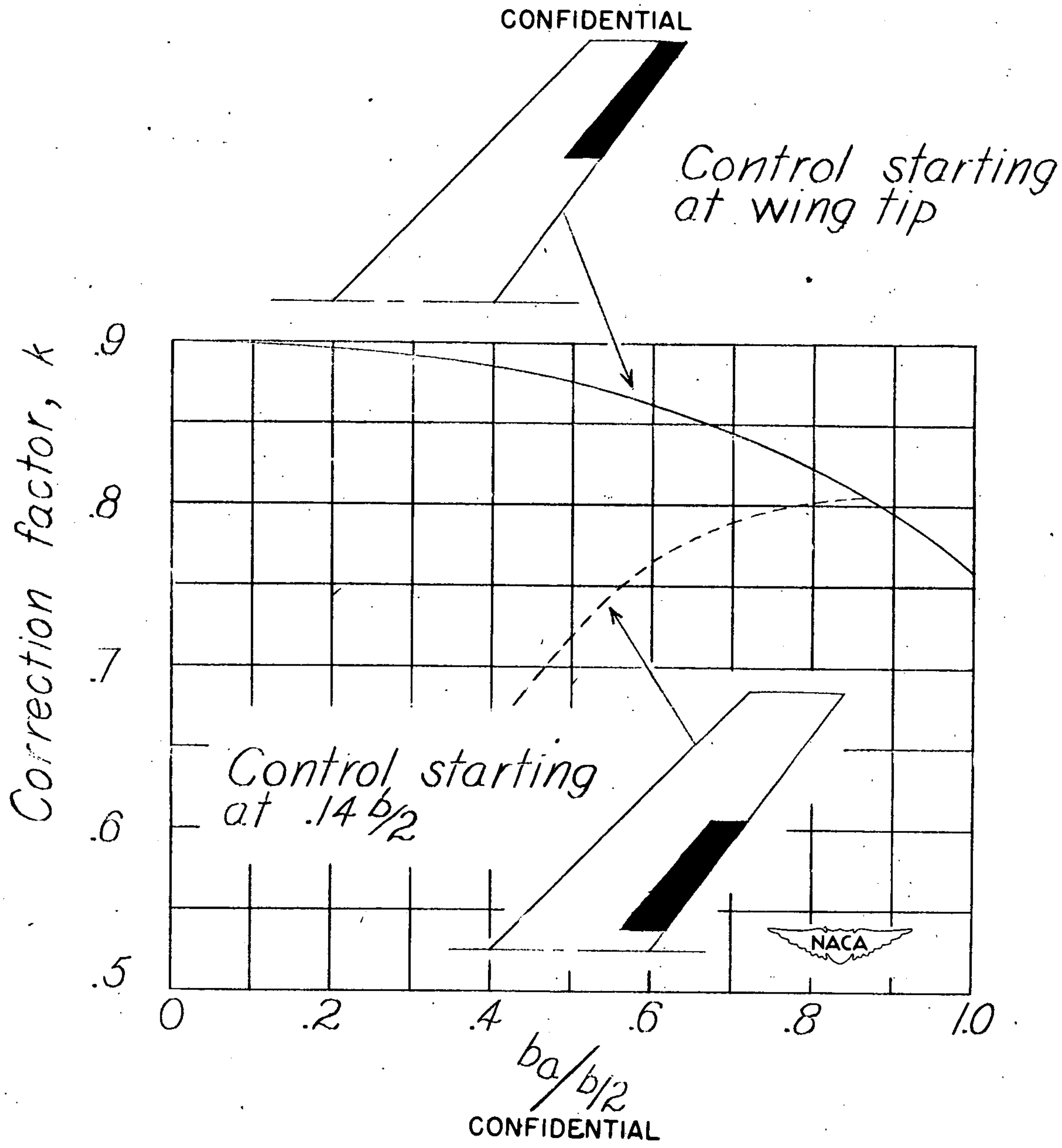

Figure 4.- Reflection-plane correction factors for inboard and outboard controls of various spans for a wing of 45° of sweepback, aspect ratio 4, and taper ratio of 0.6.

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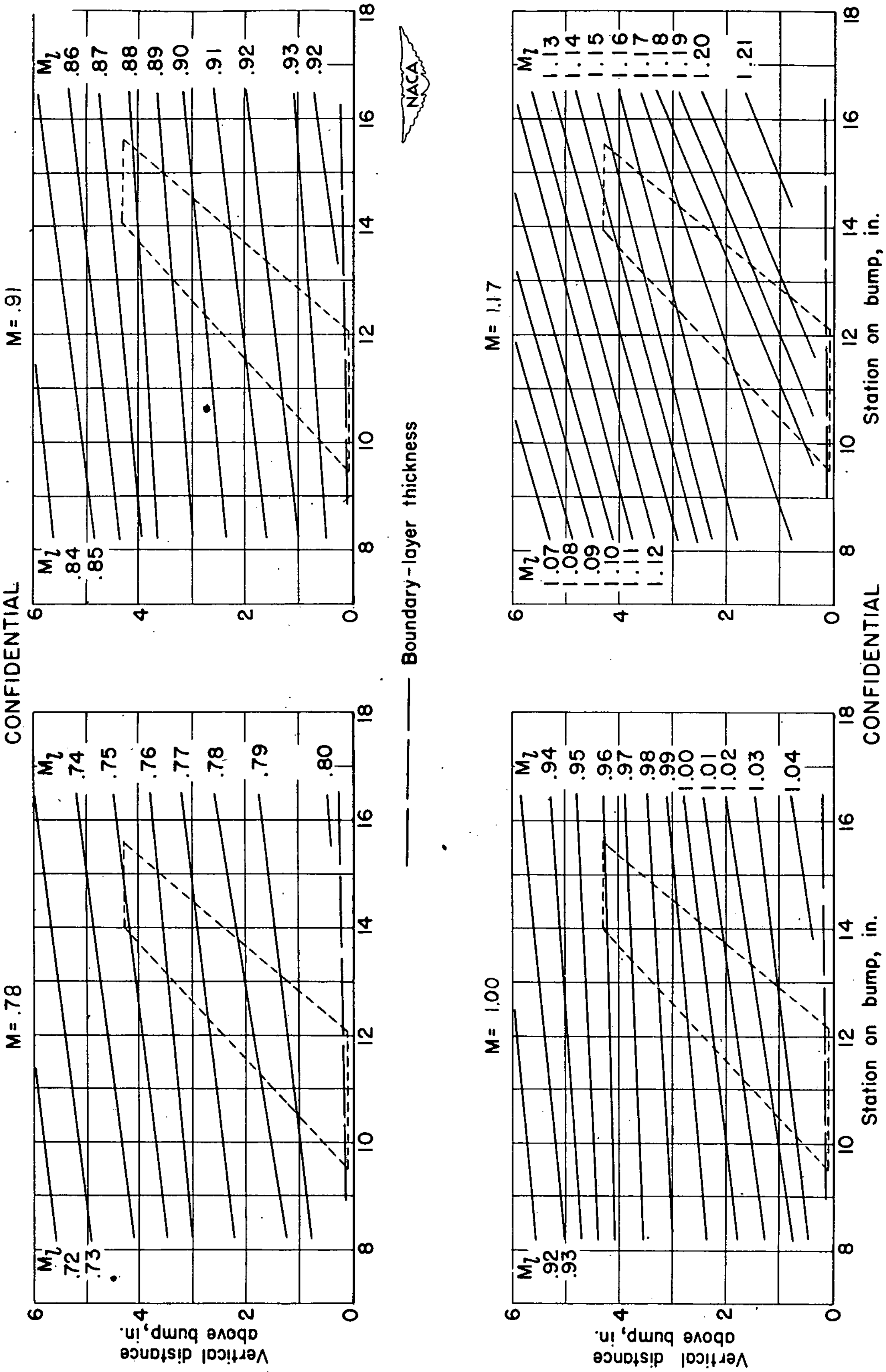

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Figure 5.- Typical Mach number contours over transonic bump in region of model location.

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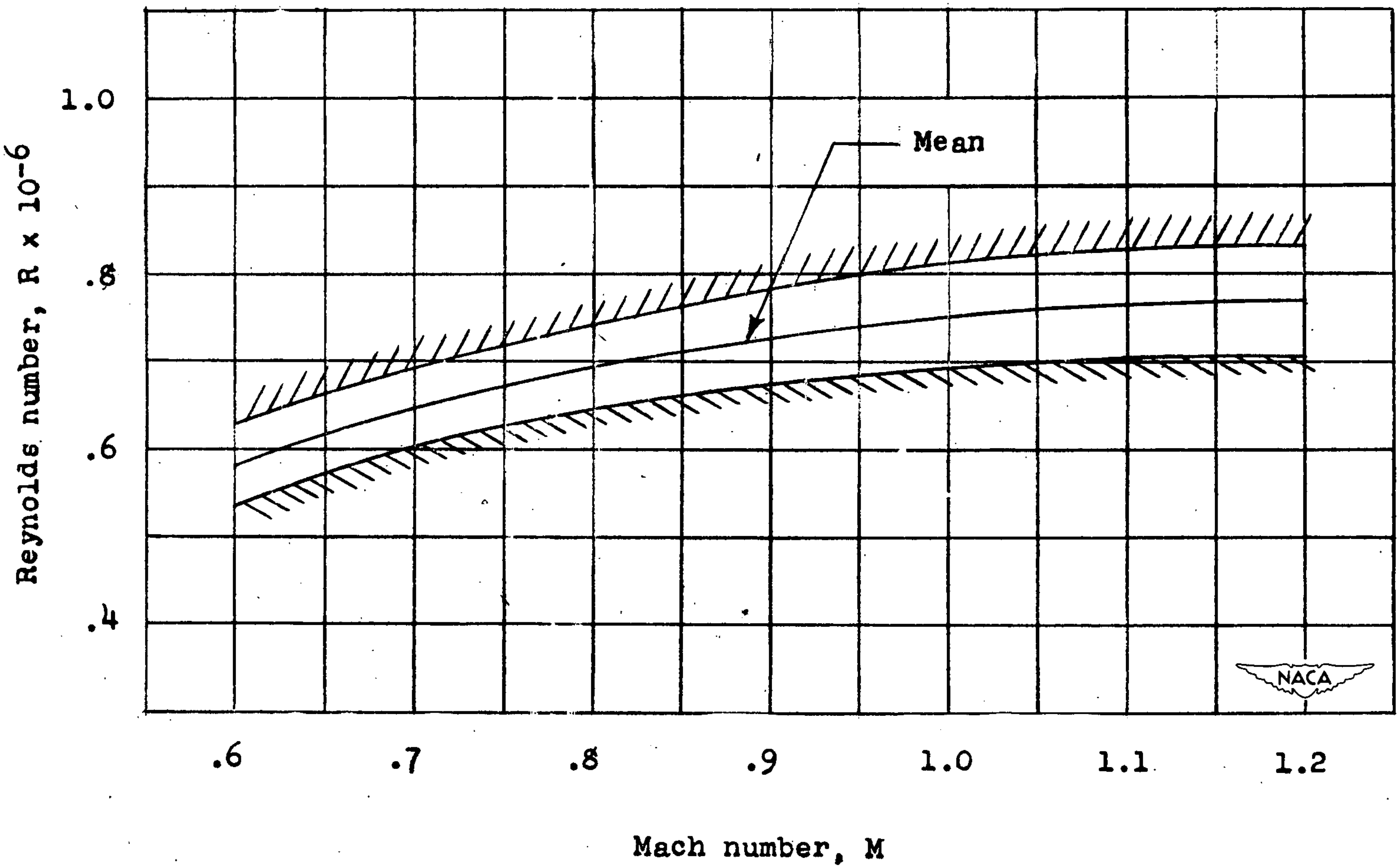

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Figure 6.– Variation of test Reynolds number with Mach number for model with 45° sweptback wing, aspect ratio 4, taper ratio 0.6, and NACA 65A006 airfoil.

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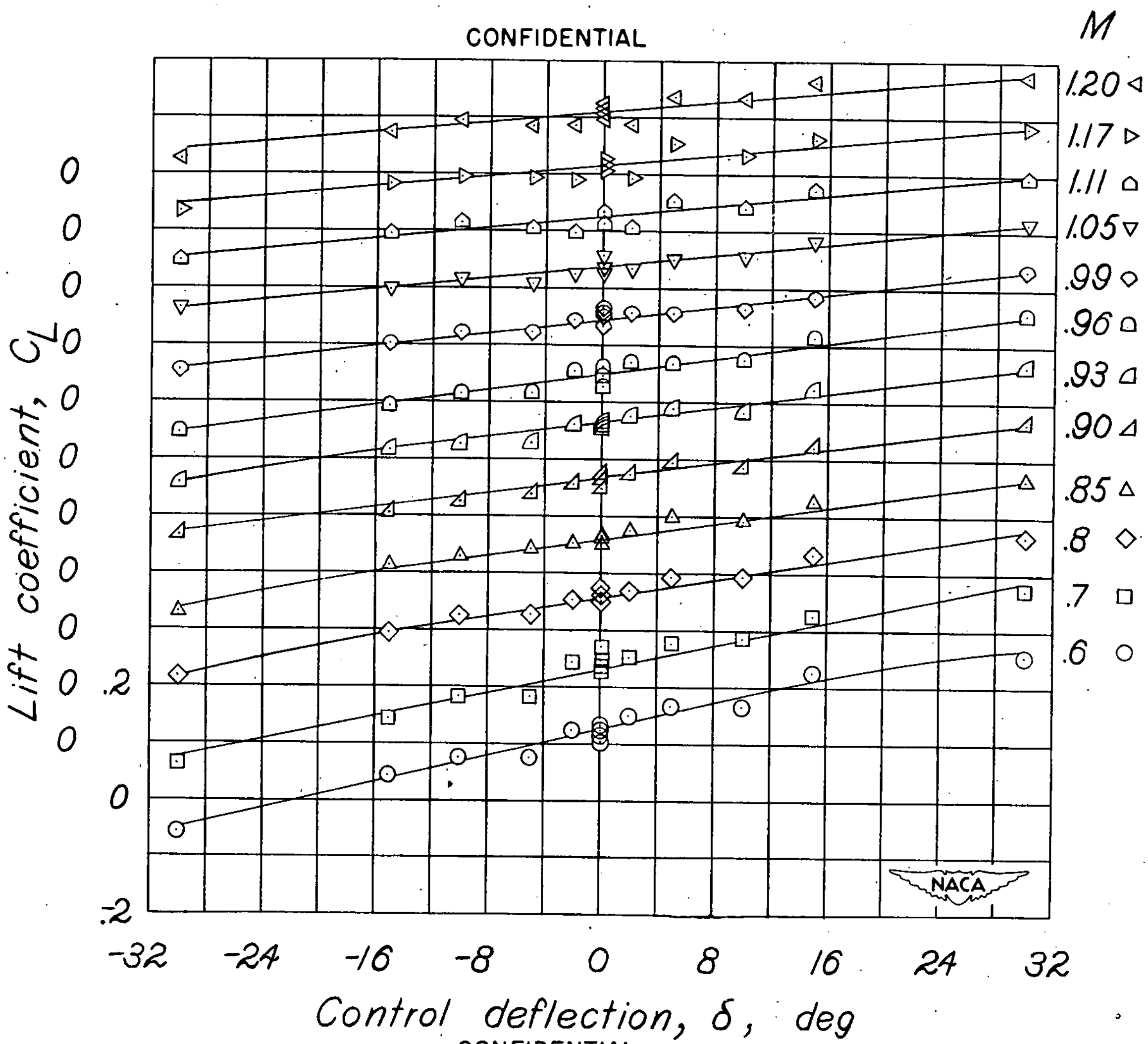

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Figure 7.– Variation of lift coefficient with control deflection for various Mach numbers. $b_a = 0.43\frac{b}{2}$, outboard; $\alpha = 2^{\circ}$.

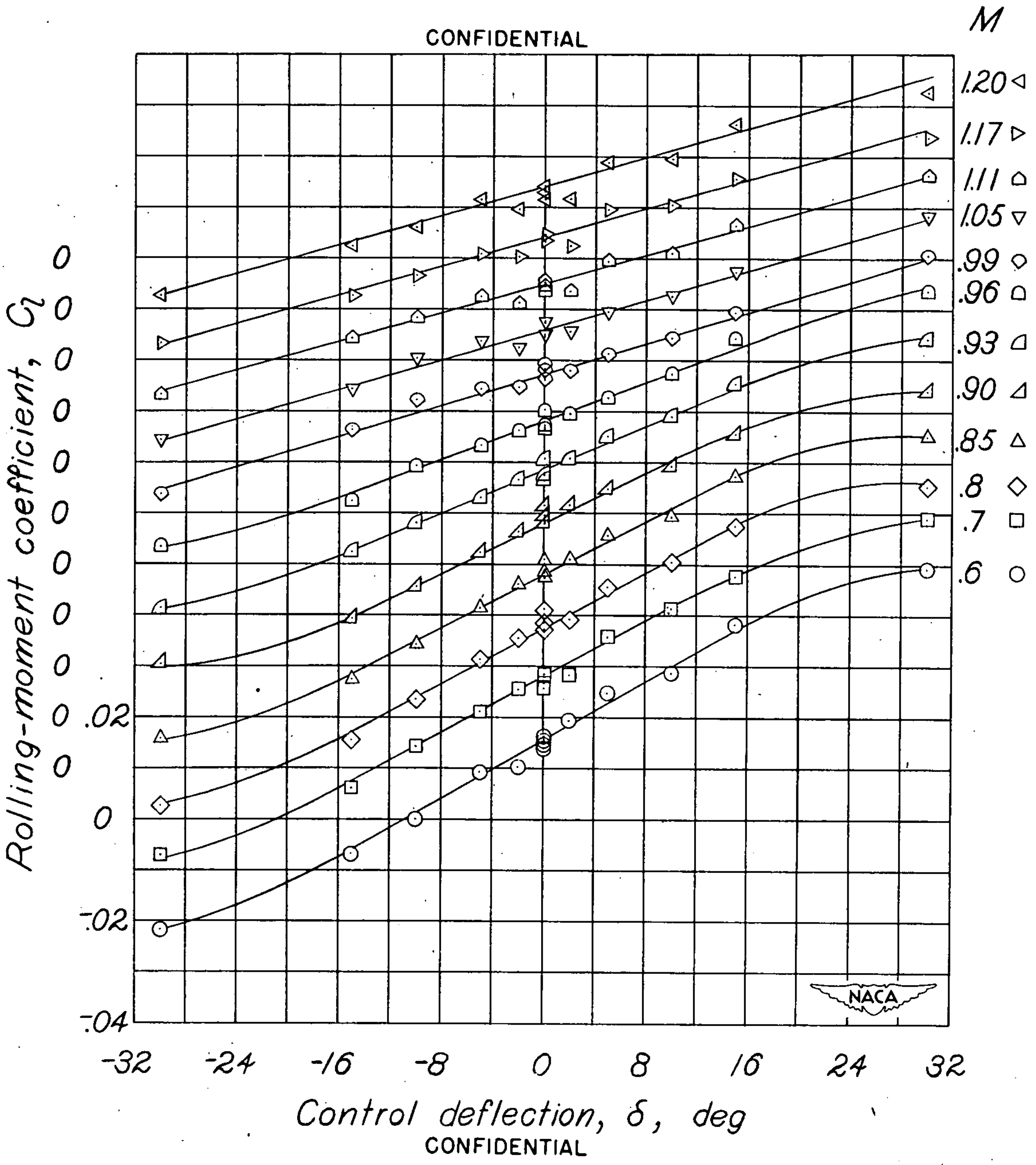

Figure 8.– Variation of rolling-moment coefficient with control deflection for various Mach numbers. $b_a = 0.43\frac{b}{2}$, outboard; $\alpha = 2^{\circ}$.

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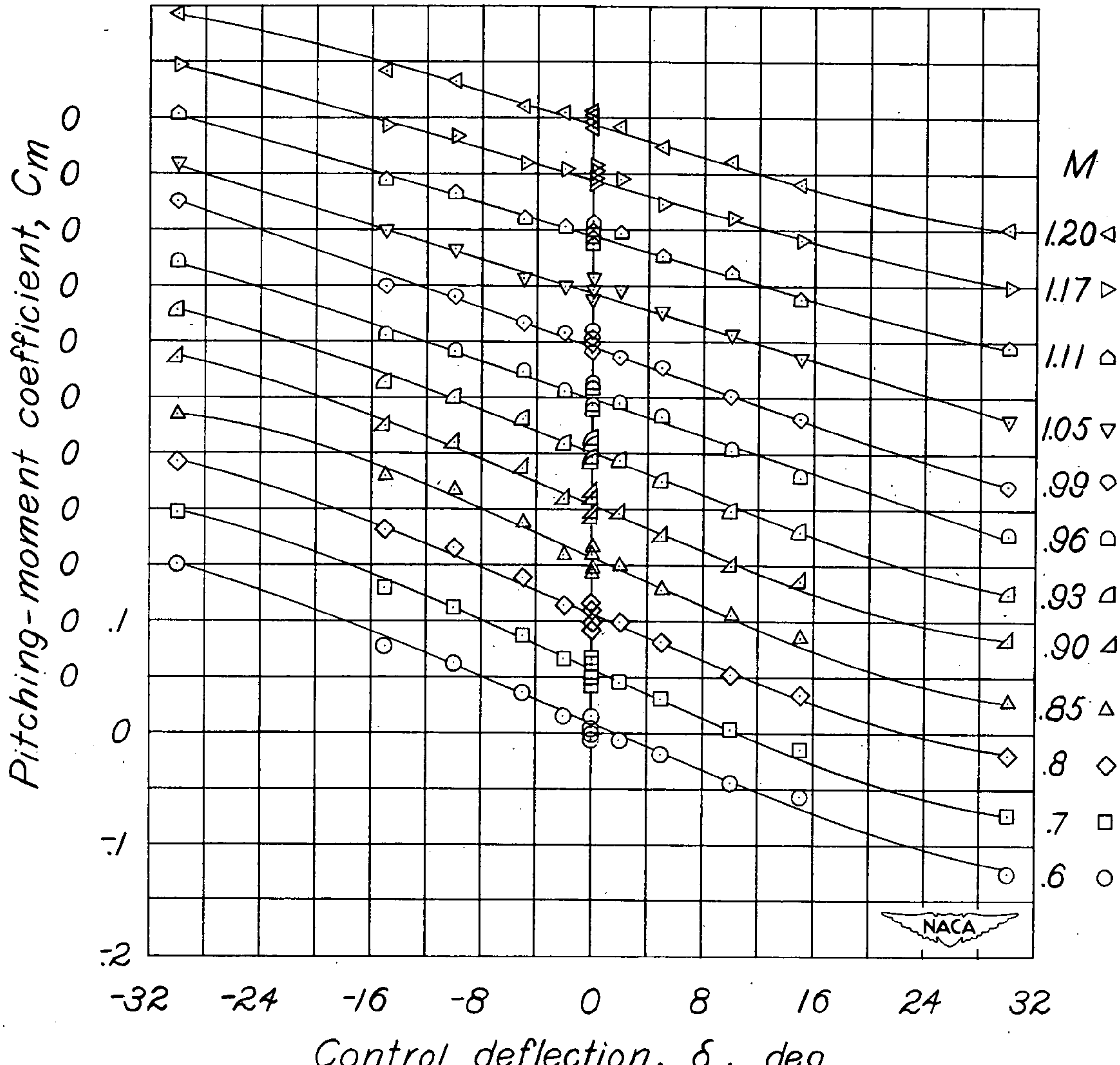

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Figure 9.- Variation of pitching-moment coefficient with control deflection for various Mach numbers. $b_a = 0.43\frac{b}{2}$, outboard; $\alpha = 2^{\circ}$.

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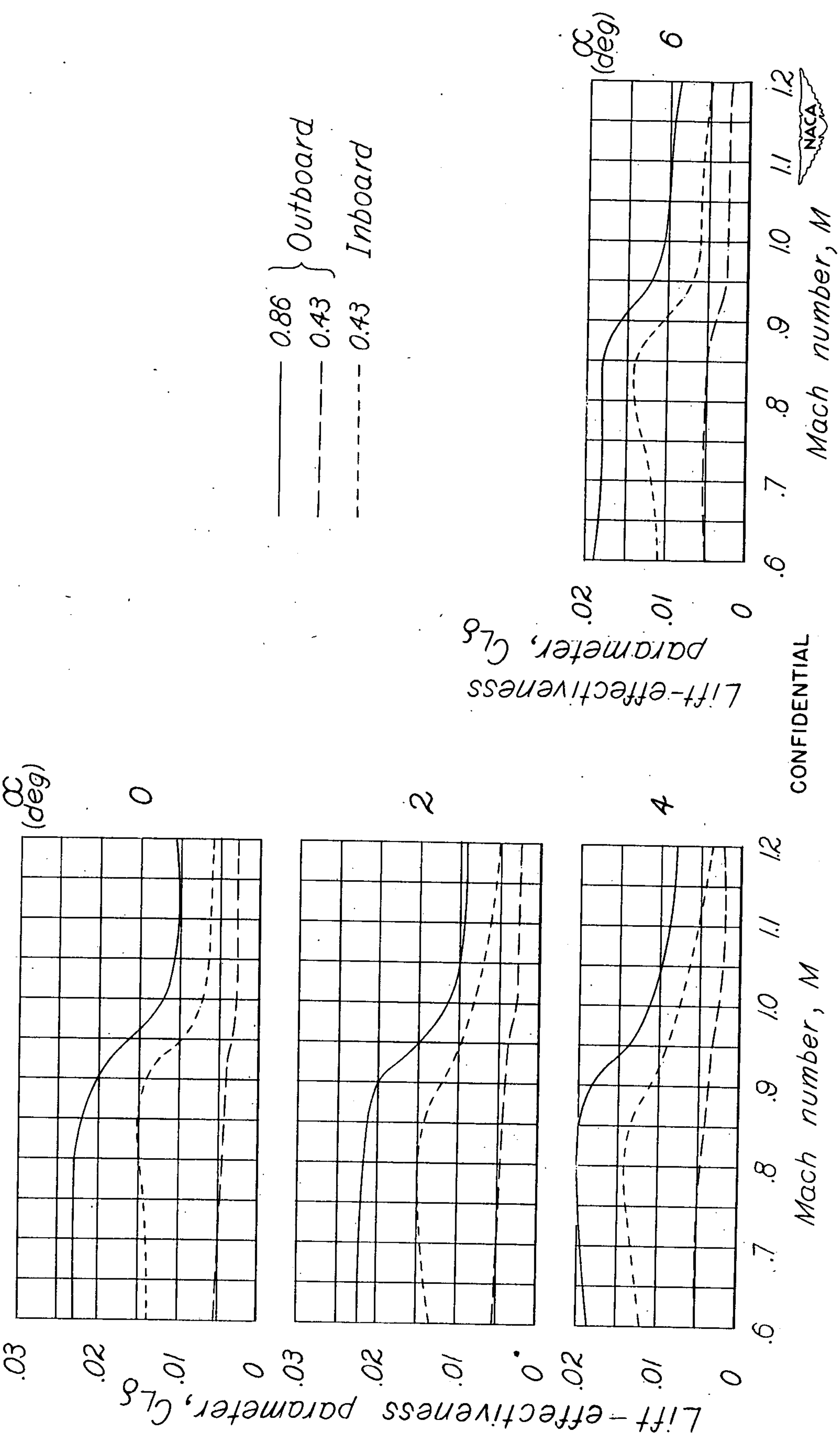

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Figure 10.- Variation of lift-effectiveness parameter with Mach number.

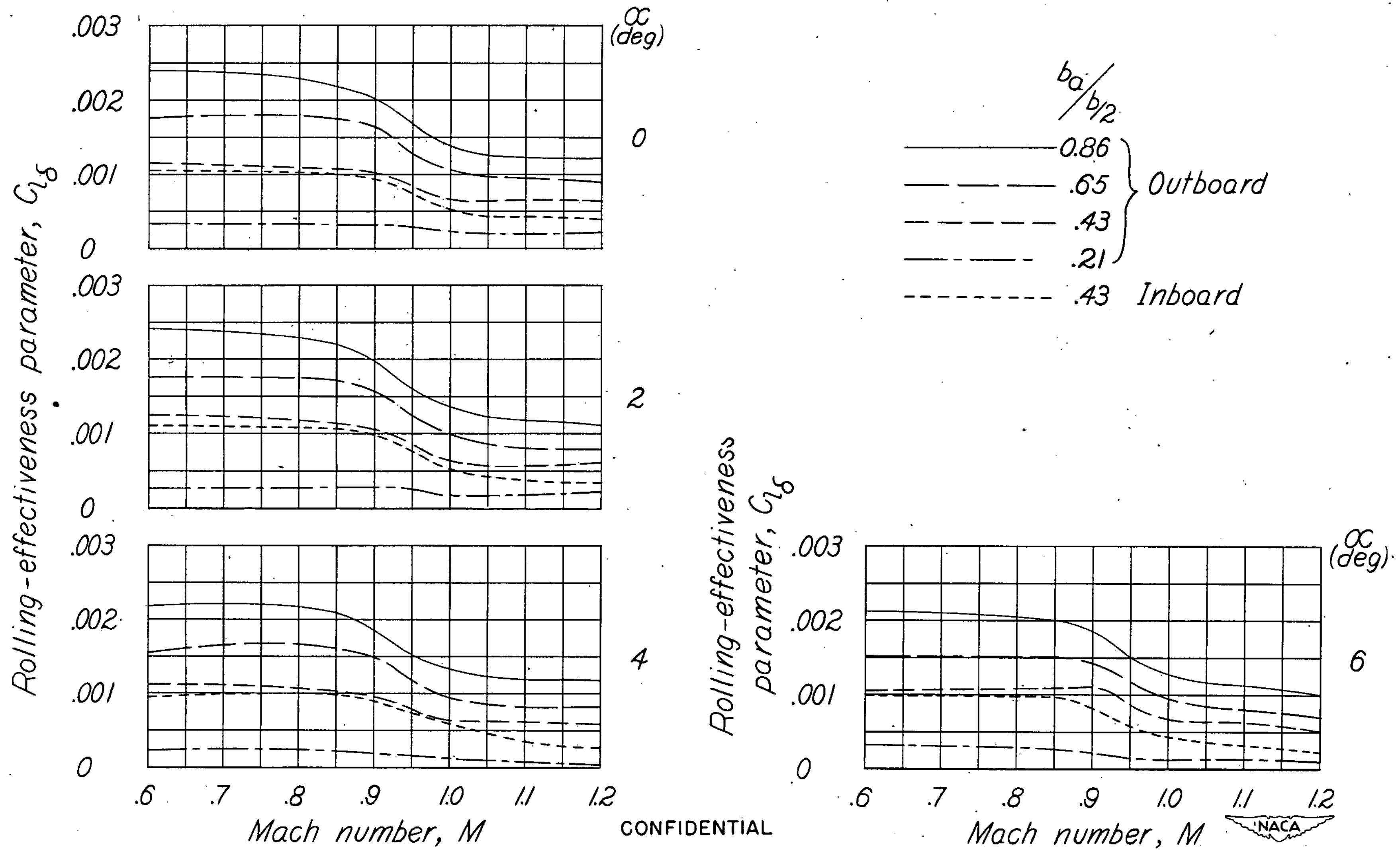

Figure 11.- Variation of rolling-effectiveness parameter with Mach number.

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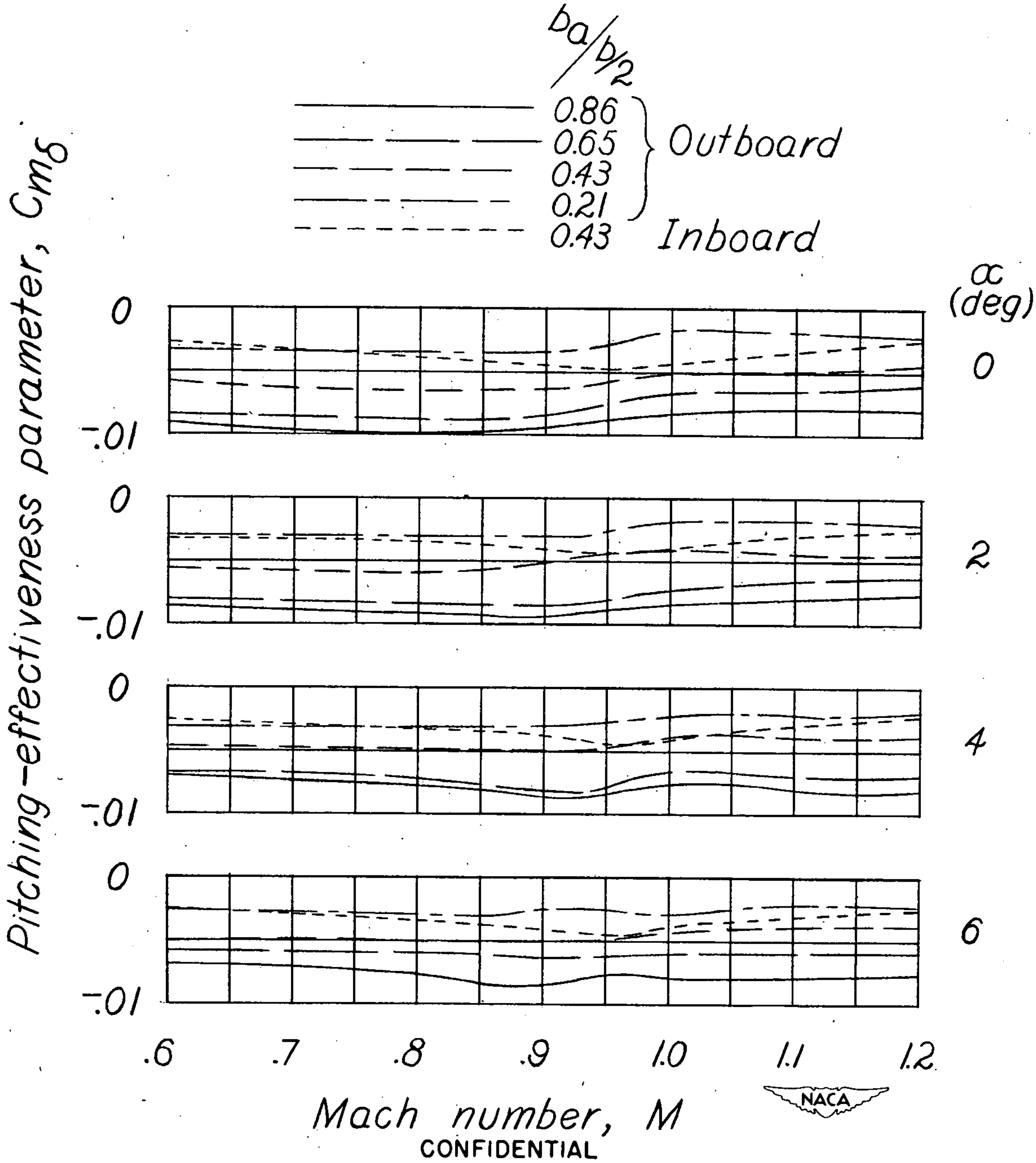

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Figure 12.– Variation of pitching-effectiveness parameter with Mach number.

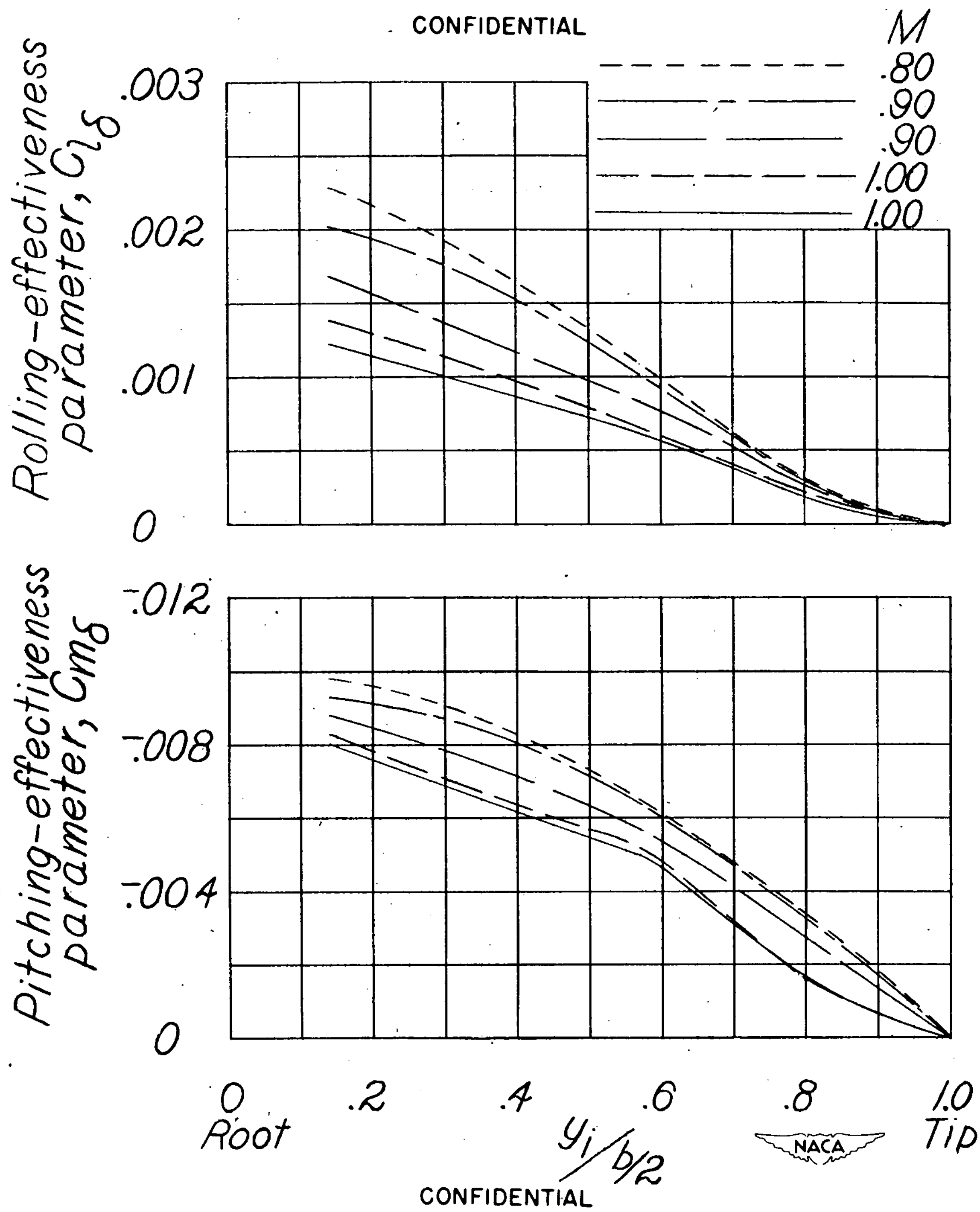

Figure 13.- Variation of control-effectiveness parameters with control span for various Mach numbers. $\alpha = 0^{\circ}$.