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RESEARCH MEMORANDUM

LONGITUDINAL CHARACTERISTICS AT MACH NUMBER OF 1.24 OF $\frac{1}{30}$-SCALE SEMISPAN MODEL OF BELL X-5 VARIABLE-SWEEP AIRPLANE WITH WING SWEPT BACK 60° FROM TESTS BY NACA WING-FLOW METHOD

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

October 12, 1950

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LONGITUDINAL CHARACTERISTICS AT MACH NUMBER OF 1.24 OF $\frac{1}{30}$-SCALE SEMISPAN MODEL OF BELL X-5 VARIABLE-SWEEP AIRPLANE WITH WING SWEPT BACK 60° FROM TESTS BY NACA WING-FLOW METHOD

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SUMMARY

Tests were made at low supersonic speeds by the NACA wing-flow method to determine the longitudinal characteristics of a $\frac{1}{30}$-scale semispan model of the Bell X-5 variable-sweep airplane with the wing in the 60° sweptback position and with tail incidences of -2° and -6°. Lift, drag, and pitching-moment results for various angles of attack are presented for a Mach number of 1.24. The Reynolds number of the tests was 1×10^6 ±5 percent based on the mean aerodynamic chord of the wing of the model.

The results indicated that the lift-curve slope was about 0.058 per degree for both tail incidences and that the lift curves remained linear up to an angle of attack of about 9°.

The slope of the pitching-moment curve $dC_m/d\alpha$ varied from about -0.014 at an angle of attack of -4° to about -0.027 at an angle of attack of 8°. These values correspond to a neutral-point variation from 50 to 73 percent mean aerodynamic chord. With a tail incidence of -6°, the model trimmed at an angle of attack of 2.3° which corresponds to a lift coefficient of 0.11. The effectiveness of the tail $\Delta C_m/\Delta i_t$ was about 0.015 per degree.

INTRODUCTION

As part of a program to determine the aerodynamic characteristics of the proposed Bell X-5 airplane incorporating a wing whose angle of sweep can be varied in flight, an investigation is being made at low supersonic speeds by the NACA wing-flow method on a $\frac{1}{30}$-scale semispan model. The results of low-speed wind-tunnel tests to determine the longitudinal stability and control characteristics of a $\frac{1}{4}$-scale full-span model have been reported in reference 1.

This paper presents the results of measurements of normal force, chord force, and pitching moment for tail incidences of -2° and -6° of the semispan model with the wing swept back 60° referred to the 25-percent chord line. The effective Mach number at the wing of the model for the tests was about 1.24.

In the interest of making these data available as soon as possible, they are presented with only a limited analysis.

SYMBOLS

V velocity, feet per second

ρ mass density, slugs per cubic foot

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

S model wing area, semispan (includes area in fuselage between perpendiculars from wing-fuselage intersections to plane of symmetry), square feet

α angle of attack of fuselage, degrees

i_t incidence of horizontal tail (referred to wing chordal plane), degrees

L lift force (resultant force perpendicular to stream velocity), pounds

D drag force (resultant force parallel to stream velocity), pounds

M pitching moment, inch-pounds

C_L lift coefficient (L/qS)

C_D drag coefficient (D/qS)

C_m pitching-moment coefficient referred to $0.26\bar{c}$ $(M/qS\bar{c})$

$\bar{c}$ mean aerodynamic chord of wing; based on the relationship

$$\frac{\int_0^{b/2} c^2 dy}{\int_0^{b/2} c\, dy}$$ where b is wing span and c is chord, inches

y lateral coordinate, inches

$\bar{c}_t$ mean aerodynamic chord of tail, inches

M_L local Mach number at wing surface

M_W effective Mach number at wing of model

M_t effective Mach number at tail of model

R Reynolds number based on mean aerodynamic chord $\bar{c}$

$dC_L/d\alpha$ slope of lift curve, per degree

$dC_m/d\alpha$ slope of pitching-moment curve, per degree

$\Delta C_m/\Delta i_t$ effectiveness of horizontal tail, per degree

APPARATUS AND TESTS

The tests were made by the NACA wing-flow method in which the model is mounted in the region of high-speed flow over the wing of an F-51D airplane.

The contour of the airplane wing in the test region for the present investigation was designed to give a uniform velocity field at Mach numbers near 1.25 rather than through the transonic range.

The configurations tested and reported herein consisted of the semispan model with the wing swept back 60° (referred to the 25-percent chord line) for tail incidences of -2° and -6°.

Photographs of the semispan model equipped with an end plate at the fuselage center line are given in figures 1 and 2. The geometric characteristics of the model wing and horizontal tail surfaces are given in table I; other details of the model are shown in figure 3. The fuselage of the model was constructed of mahogany, whereas duralumin was used for the wing and tail surfaces. A duct was included in the model fuselage to simulate the air intake and flow through the jet engine of the full-scale airplane. The airfoil section perpendicular to the 38-percent chord line (wing pivot point of the full-scale airplane) was an NACA $65_{(10)}$A011 at the root (through pivot point) and tapered to NACA $65_{(08)}$A008.6 at the tip. The horizontal tail had an NACA 65A006 airfoil section parallel to the free stream. The $\frac{1}{4}$-chord line of the tail was swept back 45°. The aspect ratio of the wing, considering the F-51D airplane wing surface as a reflection plane, was 2.18. The semispan model, curved to conform to the curvature of the wing in the test region, was mounted close to the airplane wing surface and was connected to a balance enclosed within the wing. Because the model and balance were arranged to oscillate as a unit, the balance measured the forces both normal and parallel to the fuselage reference line of the model at all angles of attack. For each test, continuous measurements were made of angle of attack, normal force, chord force, and pitching moment as the model was oscillated through an angle-of-attack range of -4° to 16° with each of the two tail angles -2° and -6°. A free-floating vane was used to determine the direction of flow at the model as described in reference 2.

A typical chordwise velocity gradient in the test region on the airplane wing as determined from static pressure measurements at the wing surface with the model removed is indicated in figure 4. From static pressure measurements made with a static-pressure tube located at various distances above the surface of the test section, the vertical Mach number gradient was found to be 0.009 per inch up to a distance of 6 inches above the surface. The effective dynamic pressure at the model wing q, the effective Mach number at the model wing M_w, and the effective Mach number at the model tail M_t were determined from an integration of the velocity distribution over the area covered by the wing and tail of the model. For the chordwise velocity distribution shown in figure 4, M_w and M_t were, respectively, 1.24 and 1.23. A compression shock passes over the model location at an effective Mach number somewhat lower than 1.22; an upper limit of 1.26 is determined by the maximum airplane Mach number at which the F-51D airplane may be safely operated. A more complete discussion of the method of determining the Mach number and dynamic pressure at the model can be found in reference 2.

The tests were made in high-speed dives of the F-51D airplane. The Reynolds number was about $1 \times 10^6 \pm 5$ percent based on the mean aerodynamic chord of the wing of the model.

RESULTS AND DISCUSSION

The aerodynamic characteristics are presented in figure 5 for a tail incidence of -2° and in figure 6 for a tail incidence of -6°. Data points are shown for both increasing and decreasing angles as the model was oscillated through the range of angles of attack from -4° to 16°. Pitching-moment data were obtained only from -4° to 6° for the -2° tail and only from 0° to 10° for the -6° tail because of limitations in the capacity of the pitching-moment element of the balance. A comparison of the results for the two tail incidences is presented as figure 7, and the following discussion is based on this figure.

The lift curves for both tail incidences remain linear up to about 9° angle of attack or a lift coefficient of about 0.5. The angle for zero lift of the model is about -0.2° with the -2° tail and 0.4° with the -6° tail setting. The lift-curve slopes $dC_L/d\alpha$ for both tail incidences are the same, about 0.058 per degree. A low-speed value of 0.045 for the lift-curve slope was obtained in wind-tunnel tests of a $\frac{1}{4}$-scale model of the same airplane with similar configuration (reference 1).

The value of the slope of the pitching-moment curve $dC_m/d\alpha$ varies from about -0.014 to about -0.027 over a range of angle of attack from -4° to $+8^{\circ}$. These values are independent of tail incidence within the accuracy of determination and correspond to a neutral point variation from about 50 to 73 percent $\bar{c}$. With a tail incidence of -6°, the model trimmed at an angle of attack of 2.3° which corresponds to a lift coefficient of 0.11. For the -2° tail the angle of attack and lift coefficient for trim are, respectively, -1.1° and -0.06°. The effectiveness of the tail $\Delta C_m/\Delta i_t$ is about 0.015 per degree.

The absolute values of drag are not reliable because they include the drag of the end plate and are subject to an unknown effect of the semispan configuration on the drag of the fuselage; both of these factors tend to substantially affect the measured drag. Duct losses would also be expected to affect the drag measurements somewhat. The results are presented because the drag variation with lift coefficient is believed, on the basis of other investigations, to be essentially unaffected by these factors and are therefore considered to be of interest.

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REFERENCES

1. Kemp, William B. Jr., Becht, Robert E., and Few, Albert G. Jr.: Stability and Control Characteristics at Low Speed of a $\frac{1}{4}$-Scale Bell X-5 Airplane Model. Longitudinal Stability and Control. NACA RM L9K08, 1950.

2. Johnson, Harold I.: Measurements of Aerodynamic Characteristics of a 35° Sweptback NACA 65-009 Airfoil Model with $\frac{1}{4}$-Chord Plain Flap by the NACA Wing-Flow Method. NACA RM L7F13, 1947.

TABLE I

GEOMETRIC CHARACTERISTICS OF $\frac{1}{30}$ - SCALE SEMISPAN MODEL OF BELL X-5 VARIABLE-SWEEP AIRPLANE WITH WING IN 60° SWEEP POSITION

Wing dimension:	
Section (root)	NACA $65_{(10)}$A011
Section (tip)	NACA $65_{(08)}$A008.6
Semispan, in.	3.88
Mean aerodynamic chord, in.	3.64
Chord at tip, in.	1.84
Chord at plane of symmetry, in.	4.10
Area (semispan), sq in.	13.79
Aspect ratio	2.18
Sweepback (0.25 chord line), deg	60
Dihedral (chordal plane), deg	0
Incidence (chordal plane), deg	0
Horizontal tail:	
Section	NACA 65A006
Semispan, in.	1.91
Mean aerodynamic chord, in.	1.43
Chord at tip, in.	0.72
Chord at plane of symmetry, in.	1.95
Area (semispan), sq in.	2.55
Aspect ratio	2.86
Length ($0.26\bar{c}$ to $0.25\bar{c}_t$), in.	6.83
Height (above wing chord), in.	0.56

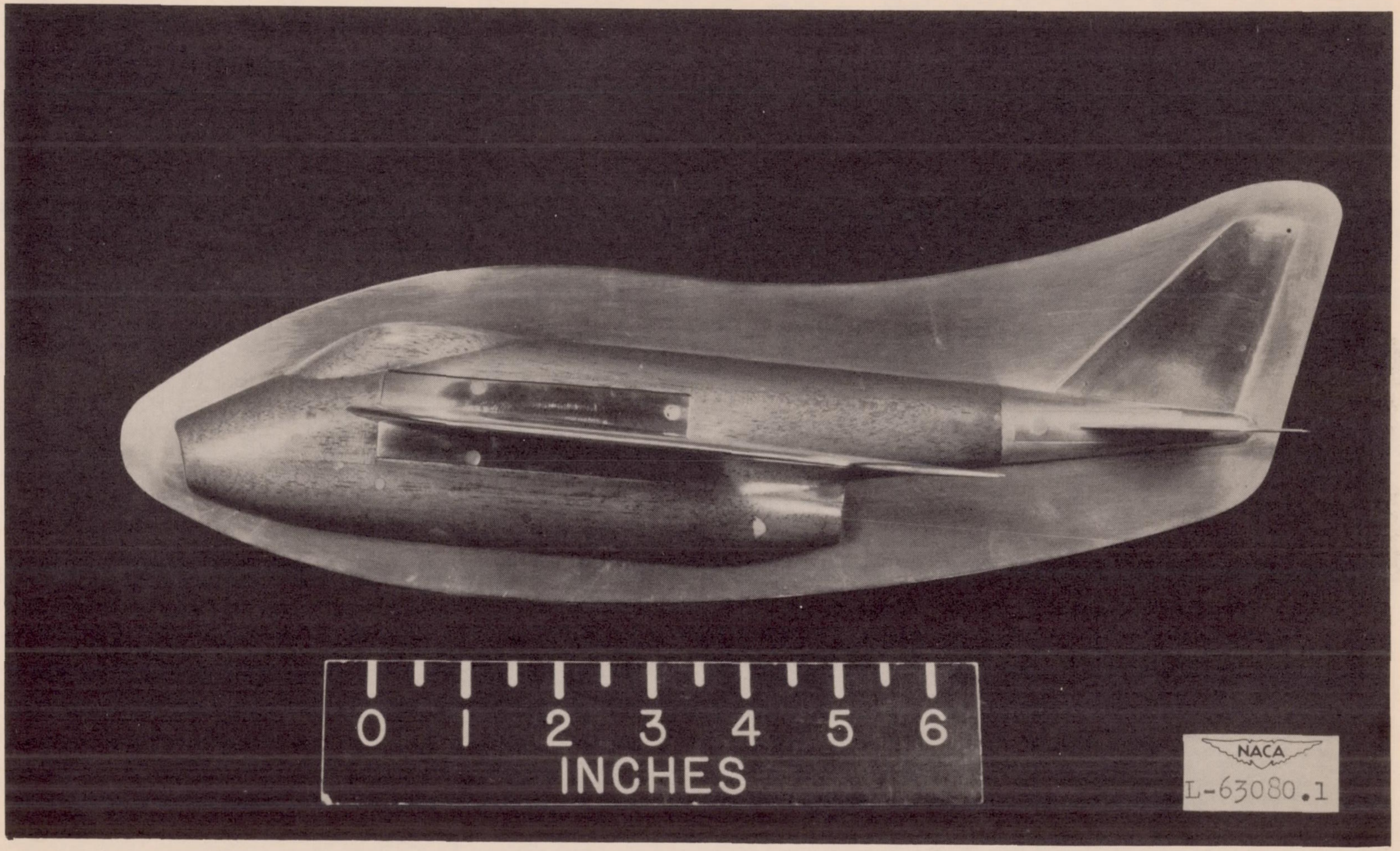

Figure 1.- Side view of semispan wing-flow model of the Bell X-5 variable-sweep airplane.

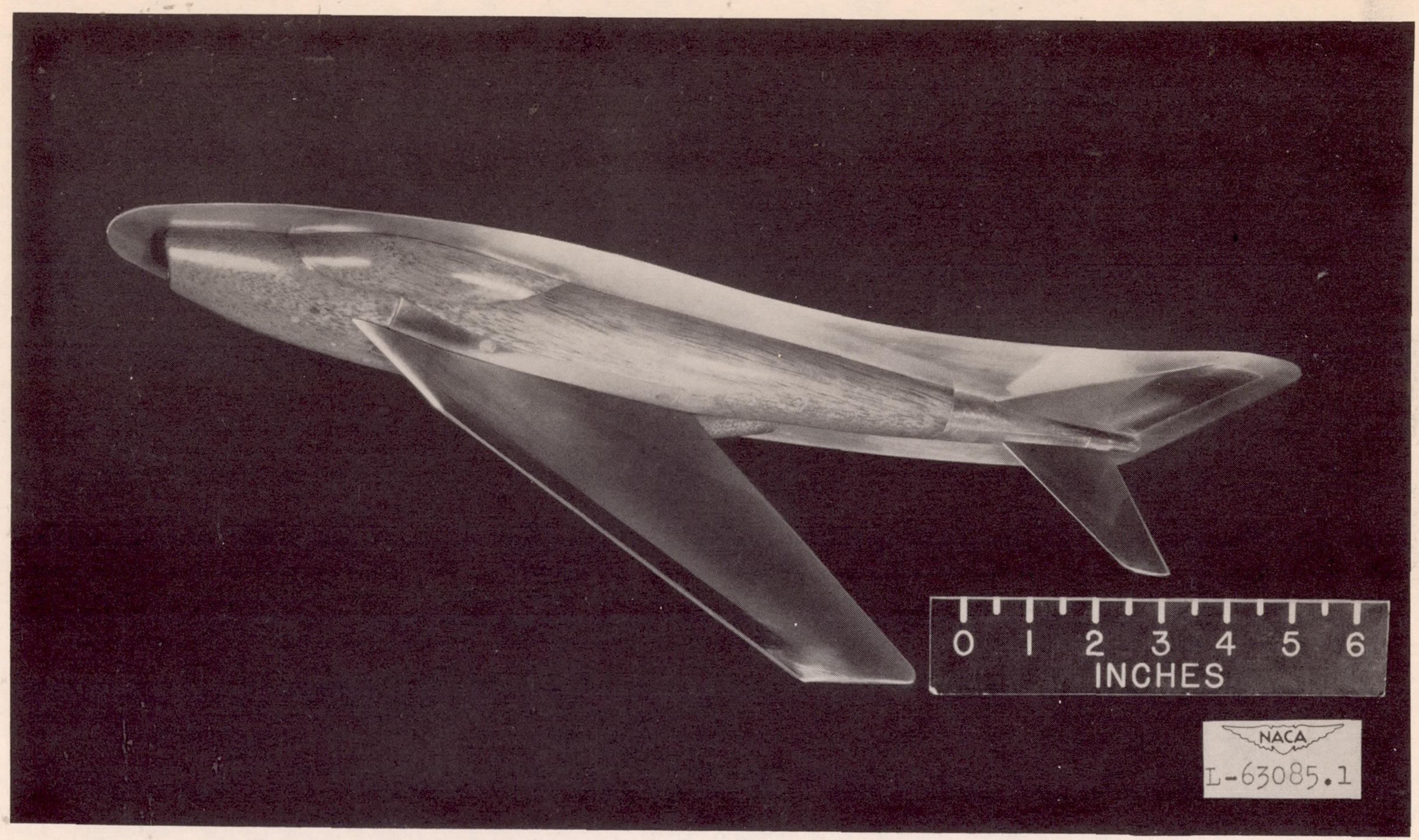

Figure 2.- Semispan wing-flow model of the Bell X-5 variable-sweep airplane with wing in 60° position.

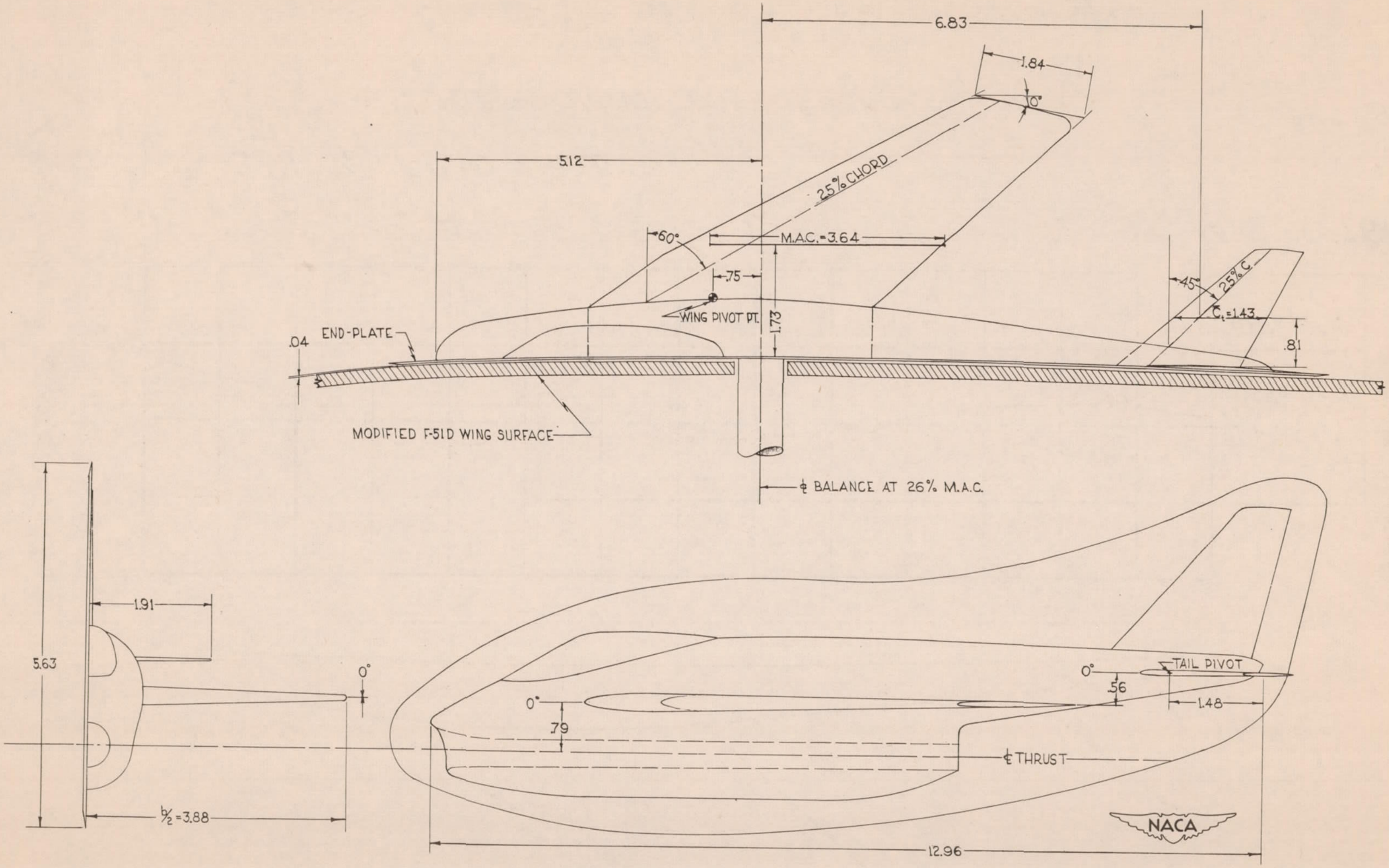

Figure 3.- Details of the semispan model of the Bell X-5 variable-sweep airplane with wing in 60° sweep position. (All dimensions are in inches.)

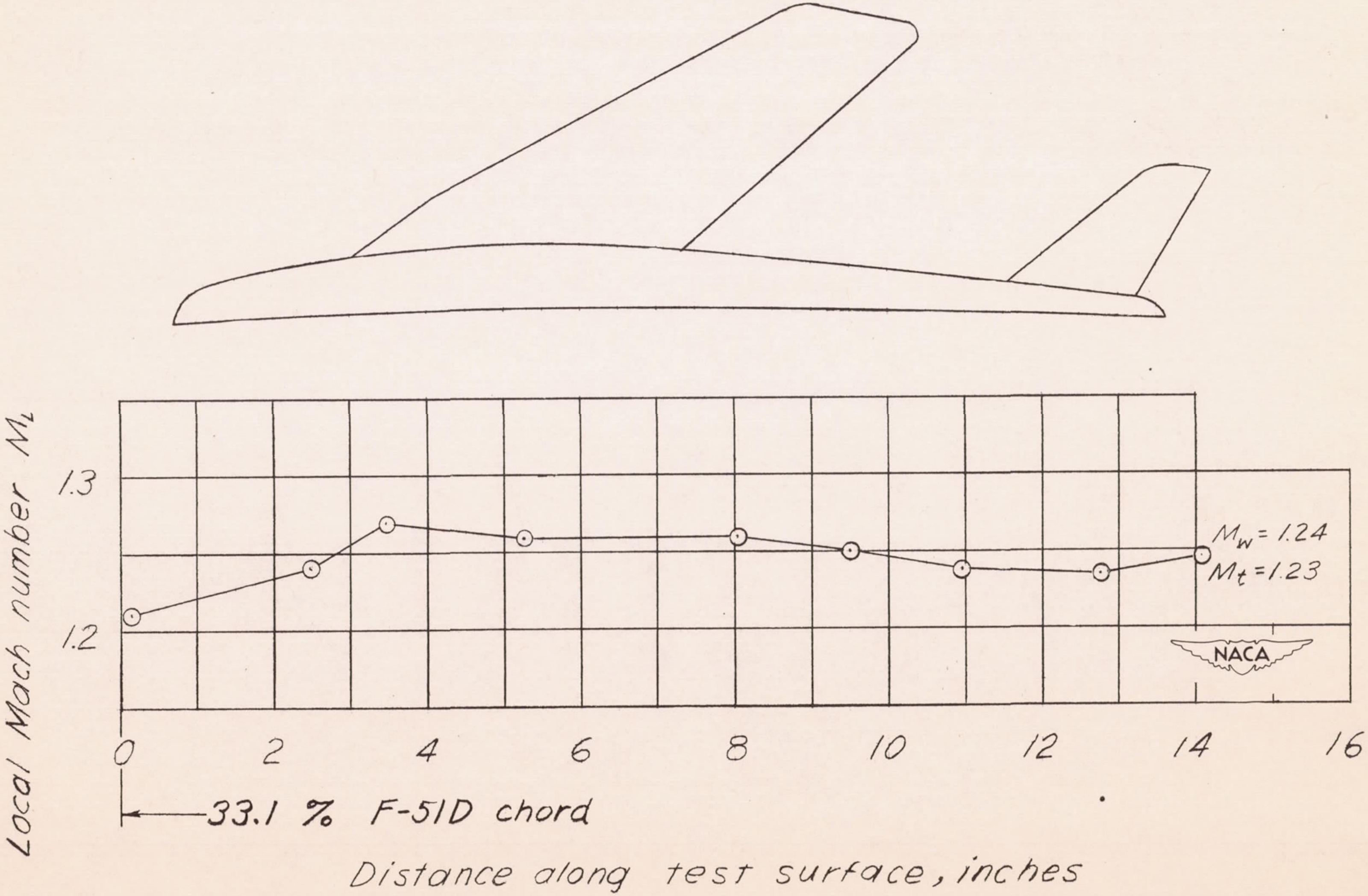

Figure 4.- Typical chordwise local Mach number variation measured at surface of test section. Chordwise location of model also shown.

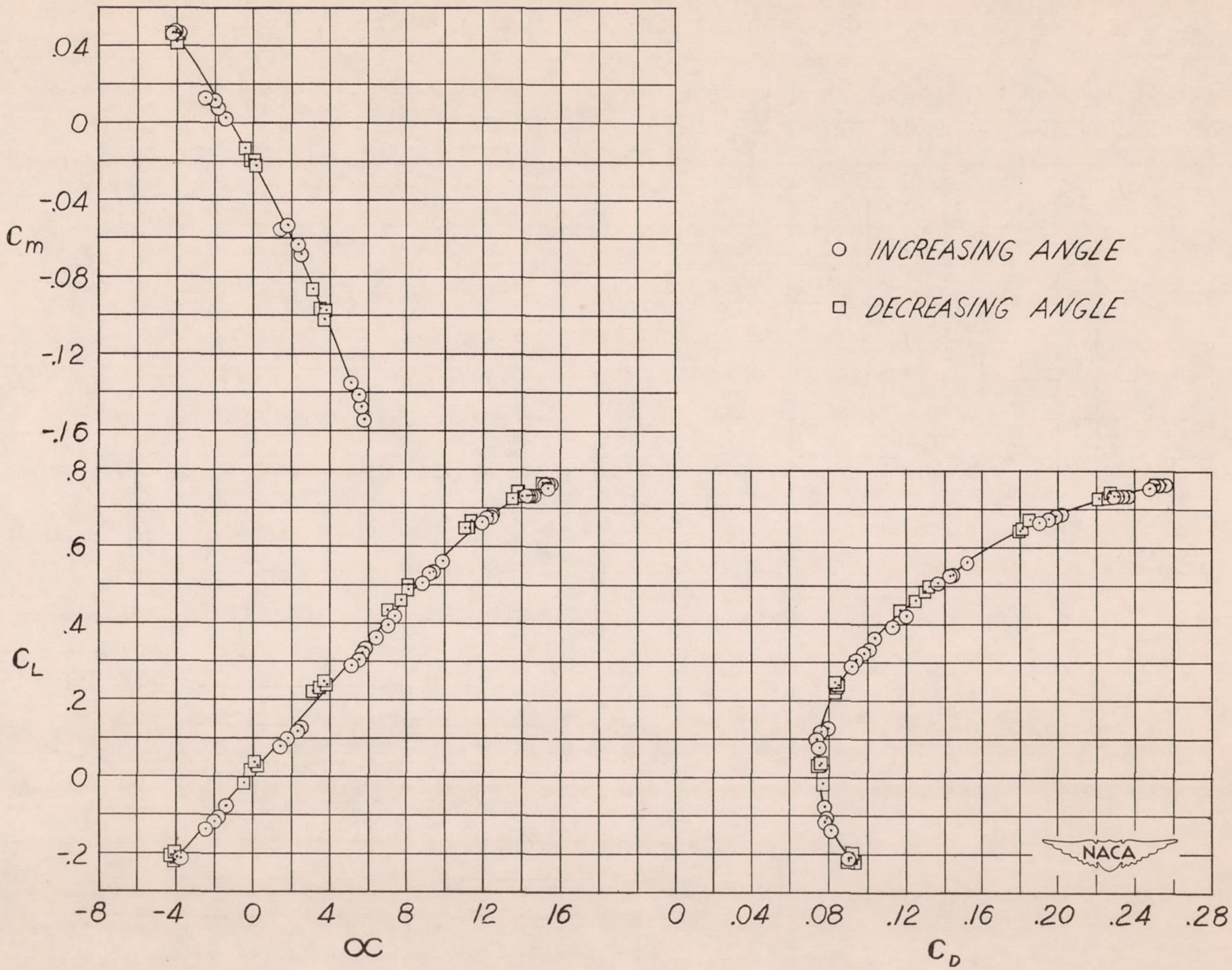

Figure 5.- Aerodynamic characteristics of semispan model of Bell X-5 variable-sweep airplane. Sweepback angle 60°; $i_t = -2^{\circ}$; $M_W = 1.24$.

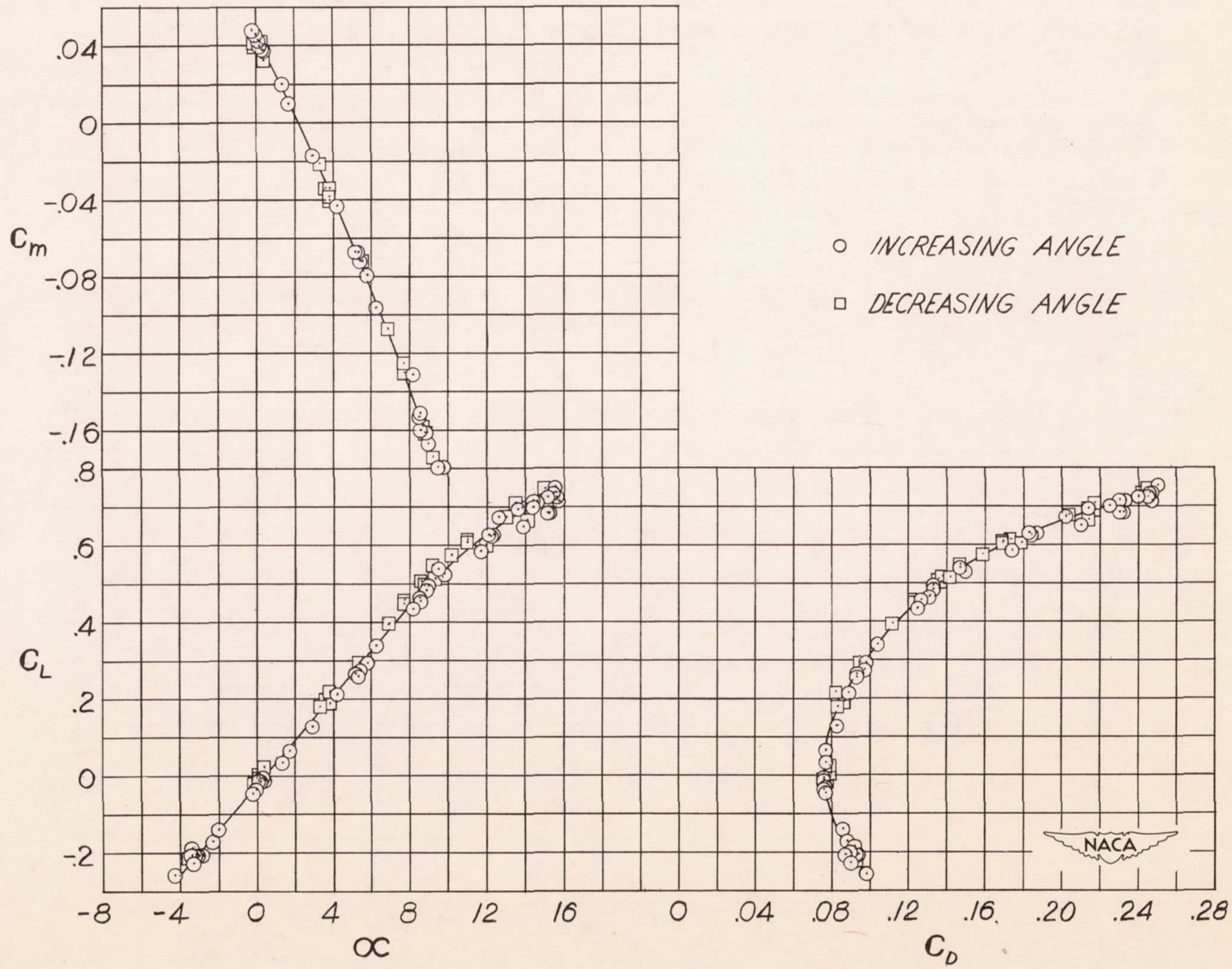

Figure 6.- Aerodynamic characteristics of semispan model of Bell X-5 variable-sweep airplane. Sweepback angle 60°; $i_t = -6°$; $M_w = 1.24$.

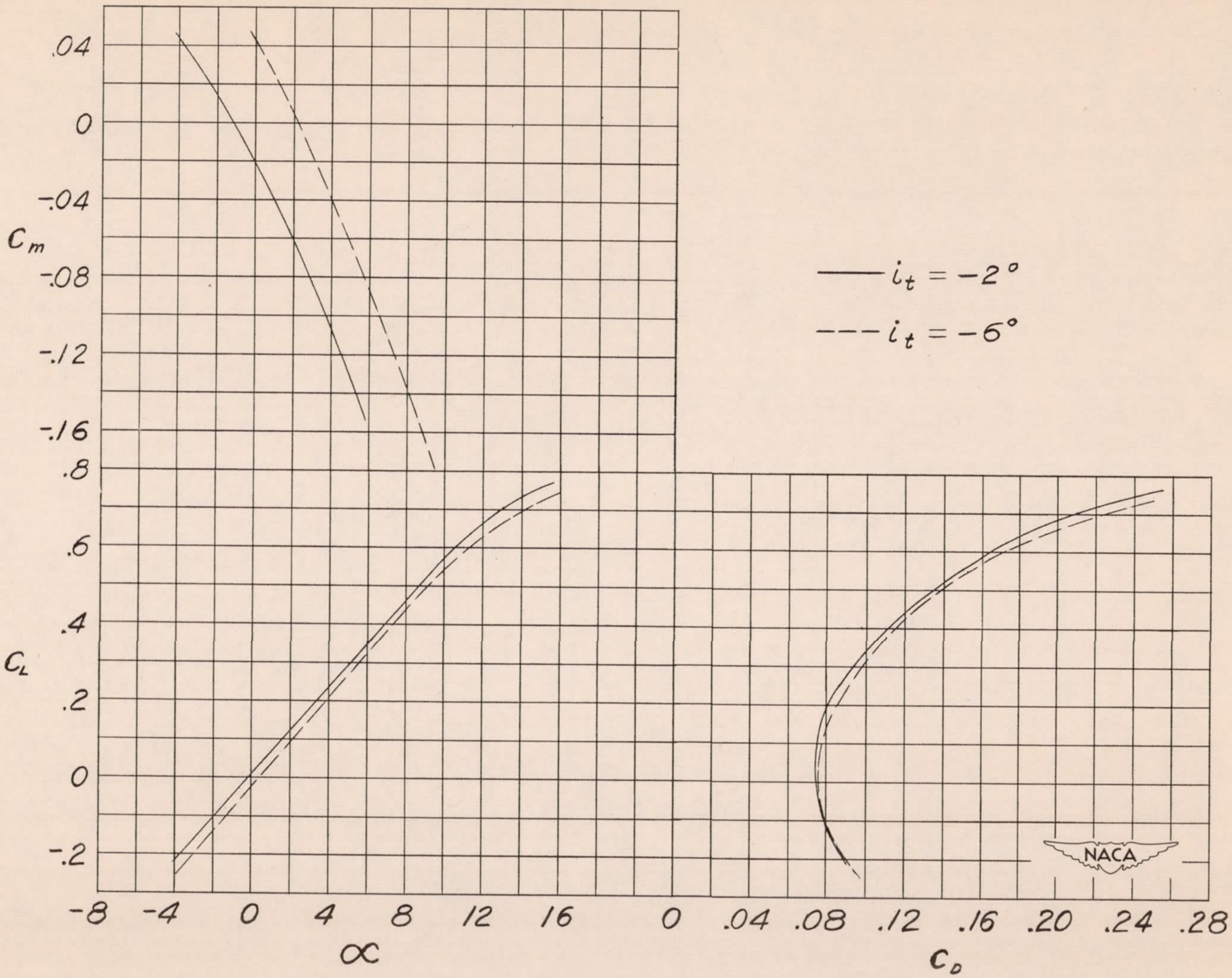

Figure 7.- Comparison of aerodynamic characteristics of the semispan model with each tail setting. $M_W = 1.24$.