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RESEARCH MEMORANDUM

THE EFFECTIVENESS OF A TRAILING-EDGE SPOILER ON A SWEPT-BACK AIRFOIL AT TRANSONIC SPEEDS FROM TESTS BY THE NACA WING-FLOW METHOD

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

THE EFFECTIVENESS OF A TRAILING-EDGE SPOILER ON A SWEPT-BACK AIRFOIL AT TRANSONIC SPEEDS FROM TESTS BY THE NACA WING-FLOW METHOD

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SUMMARY

Tests were made at Mach numbers from 0.61 to 1.05 by the NACA wing-flow method to determine the variation with Mach number of the control effectiveness of a spoiler on the trailing edge of a swept-back airfoil. A trailing-edge spoiler on a swept airfoil is of interest as a possible control device for guided missiles. The airfoil had a sweepback of 45°, an NACA $65_{(112)}$-010, a = 1.0 airfoil section normal to the leading edge, and an aspect ratio of 4.0. The spoiler was normal to the chord of the airfoil and in its neutral position projected 1.75 percent chord above and below the trailing edge. The tests consisted in measuring the floating angle of the model about an axis well forward of the aerodynamic center. Measurements of floating angle were made without a spoiler, with a spoiler along the entire span of the model in a neutral or undisplaced position, with a spoiler along the entire span of the model displaced 0.88 percent chord, and with a spoiler along the middle third of the model span displaced 0.88 percent chord.

The results of the tests indicated that the effectiveness of the trailing-edge spoiler decreased about 10 percent when the Mach number was increased from 0.70 to 0.90. With further increase in Mach number the effectiveness decreased more rapidly and at a Mach number of 1.00 it was about 55 percent of the value obtained at a Mach number of 0.70. The $\frac{1}{3}$ -span spoiler was approximately half as effective as the full-span spoiler.

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INTRODUCTION

In connection with the development of a pilotless aircraft for use as a guided missile, an aerodynamic control device was required for use on a swept-back wing. The principal requirements were simplicity of construction and installation, small hinge moments, and no serious loss in effectiveness at any speed up to and including speeds in the transonic range. A circular arc spoiler on the trailing edge of the wing of the type devised by Wagner in connection with the development of German guided missiles (no published references available) was considered as one possibility because it fulfilled the first two requirements. No information, however, was available on the effectiveness of this control at transonic speeds and tests were therefore undertaken in which the NACA wing-flow method of reference 1 was used. The model tested was a constant-chord airfoil swept back 45° with a 3.5-percent-chord spoiler mounted on the trailing edge. The tests consisted in the measurement of the floating angle of attack of the model about a pivot well forward of the aerodynamic center. The change in floating angle with spoiler deflection was taken to be a measure of the effectiveness of the control. The tests were discontinued when another type of control device was adopted for the guided missile and the scope is therefore limited.

Measurements of the floating angle of the model were made at Mach numbers from 0.61 to 1.05 without a spoiler, with a spoiler along the entire span of the model in a neutral or undisplaced position, with a spoiler along the entire span of the model displaced 0.88 percent chord, and with a spoiler along the middle third of the model span displaced 0.88 percent chord.

SYMBOLS

α_V angle of model with respect to angle of reference vane

α angle of attack of model for zero pitching moment about pivot axis

$\Delta\alpha$ change in α due to deflection of control

b_s span of trailing-edge spoiler

b span of model

h_s displacement of trailing-edge-spoiler center line from model chord

c chord of model; taken parallel to stream direction

M effective Mach number

R Reynolds number based on chord of model

APPARATUS, METHODS, AND TESTS

The tests were made by the NACA wing-flow method of reference 1, in which the model is mounted in the region of high-speed flow over the wing of an airplane. A P-51D airplane was used for the tests.

The semispan model was mounted over the ammunition-compartment door of the airplane, as shown in figures 1 and 2. The model, cut from steel, had a semispan of 5.69 inches, a chord of 2.83 inches, an area of 16.0 square inches, and was swept back 45°. The aspect ratio, if the airplane wing surface is considered as a reflection plane, was 4.0. The airfoil section in the planes normal to the leading edge was NACA $65_{(112)}$-010, a = 1.0. An elliptical end plate having a major axis of 2.83 inches and a minor axis of 2.00 inches was attached to the model at the root. The model was supported on a shank which passed through a slot in the ammunition-compartment door and was mounted on a ball-bearing pivot secured to the interior of the ammunition-compartment door. It was therefore free to float at the angle of attack for zero pitching moment about the pivot axis. The angular position of the model during the tests was determined by recording photographically the deflection of a light beam from a mirror mounted on the shaft of the pivot.

The trailing-edge spoiler was made from a flat plate 0.1 inch wide and 1/32 inch thick and had edges beveled 45°. (See figs. 2 and 3.) The spoiler was soldered to the trailing edge of the airfoil. This spoiler arrangement differed somewhat from the design configuration which required that the front and rear faces of the spoiler have circular arc sections concentric with the hinge axis (located forward in the airfoil) in order to give little or no hinge moment. The design configuration also required a small clearance gap between the trailing edge of the model and the spoiler to allow for its movement. The foregoing differences between the actual model and the design configuration would be expected to have little effect on the effectiveness of the control.

The direction of local air flow was determined by means of a free-floating vane of wedge-shape cross section located 22.5 inches outboard of the model. (See fig. 1.) The method of mounting the vane and of recording its angular position was similar to that used for the model. The direction of the flow at the model location relative to the direction of flow at the vane location was determined in a test of the model without the trailing-edge spoiler. The angle of attack for zero pitching moment about the pivot axis for a given spoiler configuration and Mach number was then taken as the difference in the angle of the model relative to the vane angle for that configuration and for the spoiler-off condition. The spacing of the vane and the model was such that the mutual interference is believed to have been negligible. (See reference 2.)

The method of determining the effective Mach number at the model station from static-pressure measurements at the wing surface was similar to the method described in reference 2. A correction factor of 0.98, based on the result of an incomplete investigation of the variation of static pressure with distance from the wing surface, however, was applied to take account, approximately, of the decrease in velocity with distance from the wing surface.

The tests of each model configuration were made in two high-speed dives. One dive covered an altitude range from 28,000 to 22,000 feet in which Mach numbers at the model from 0.61 to 1.05 and Reynolds numbers from 0.43×10^6 to 0.69×10^6 were obtained. The other dive was from an altitude of 18,000 feet to 12,000 feet, and Mach numbers from 0.61 to 0.91 and Reynolds numbers from 0.60×10^6 to 0.90×10^6 were obtained. The average relation between Reynolds number and Mach number is shown in figure 4. The Reynolds number corresponding to a given Mach number in a given nominal altitude range varied somewhat between different tests, but the variation probably did not exceed 3 or 4 percent. Measurements of the floating angles of the model were made for the following configurations: without a trailing-edge spoiler, with a spoiler along the entire span of the model in a neutral or undisplaced position, with a spoiler along the entire span of the model displaced 0.88 percent chord, and with a spoiler along the middle third of the model span displaced 0.88 percent chord. (See fig. 3.)

RESULTS AND DISCUSSION

Under certain test conditions, the model exhibited a severe high-frequency oscillation for all configurations. These conditions are indicated in figure 5 in which time histories of Mach number M and angle of the model with respect to the reference vane α_V are plotted for all tests. Apparently the oscillations were not due mainly to disturbances created by the spoiler inasmuch as they also appeared in tests with the spoiler off. For the conditions in which the oscillations were of moderate amplitude $(\pm 2.5^{\circ})$ the mean value of the model angle was used in the determination of α.

The variation with Mach number of the angle of attack for zero pitching moment α for the spoiler arrangements tested is shown in figure 6. The data for the spoiler-neutral condition provide an indication of the probable accuracy with which the spoiler could be set. The change in α resulting from displacements of the spoiler is influenced by drag moment about the pivot and by change in pitching moment about the aerodynamic center of the model. The predominating effect, however, is that due to the change in angle of attack of zero lift. Variation in α can therefore be considered as indicative of variations in effectiveness of the spoiler in changing the angle of zero lift. The effectiveness of the spoiler appeared to be consistently greater at high altitude than at low altitude, probably because of the difference in Reynolds number which was greater at the low altitude. The trends, however, are generally similar over the range of Mach number for which results were obtained at each altitude.

With a full-span spoiler displaced 0.88 percent chord the effectiveness of the spoiler for the high-altitude test decreased approximately 10 percent as the Mach number increased from 0.70 to 0.90. With further increase in Mach number the effectiveness decreased more rapidly, and at a Mach number of 1.00 it was approximately 55 percent of the value obtained at a Mach number of 0.70. The effectiveness of the $\frac{1}{3}$-span spoiler displaced 0.88 percent chord appeared to be approximately half that of the full-span spoiler and showed a similar variation with Mach number.

A comparison is shown in figure 7 of the variation with Mach number of the control effectiveness of the trailing-edge spoiler on a swept-back airfoil and of three other control arrangements tested by the NACA wing-flow method. (See references 1 and 3.) The results are presented as plots of $\Delta\alpha/\Delta\alpha_{M=0.7}$ against Mach number, where $\Delta\alpha_{M=0.7}$ is the value of $\Delta\alpha$ at a Mach number of 0.7. The comparison indicates that large differences may be expected in the effect of compressibility on the effectiveness of different control configurations at transonic speeds.

CONCLUDING REMARKS

The results of the tests of a trailing-edge spoiler on a swept-back airfoil indicated that the control effectiveness decreased about 10 percent when the Mach number was increased from 0.70 to 0.90. With further increase in Mach number the effectiveness decreased more rapidly and at a Mach number of 1.00 it was about 55 percent of the value obtained at a Mach number of 0.70. The $\frac{2}{3}$-span spoiler was approximately half as effective as the full-span spoiler.

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2. Zalovcik, John A., and Adams, Richard E.: Preliminary Tests at Transonic Speeds of a Model of a Constant-Chord Wing with a Sweepback of 45° and an NACA 65(112)-210, a = 1.0 Airfoil Section. NACA ACR No. L5J16a, 1945.

3. Daum, Fred L., and Sawyer, Richard H.: Tests at Transonic Speeds of the Effectiveness of a Swept-Back Trailing-Edge Flap on an Airfoil Having Parallel Flat Surfaces, Extreme Sweepback, and Low Aspect Ratio. NACA CB No. L5H01, 1945.

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Figure 1.- Model mounted on wing of P-51D airplane. Spoiler along entire span; spoiler deflected 0.88 percent chord; reference vane in background.

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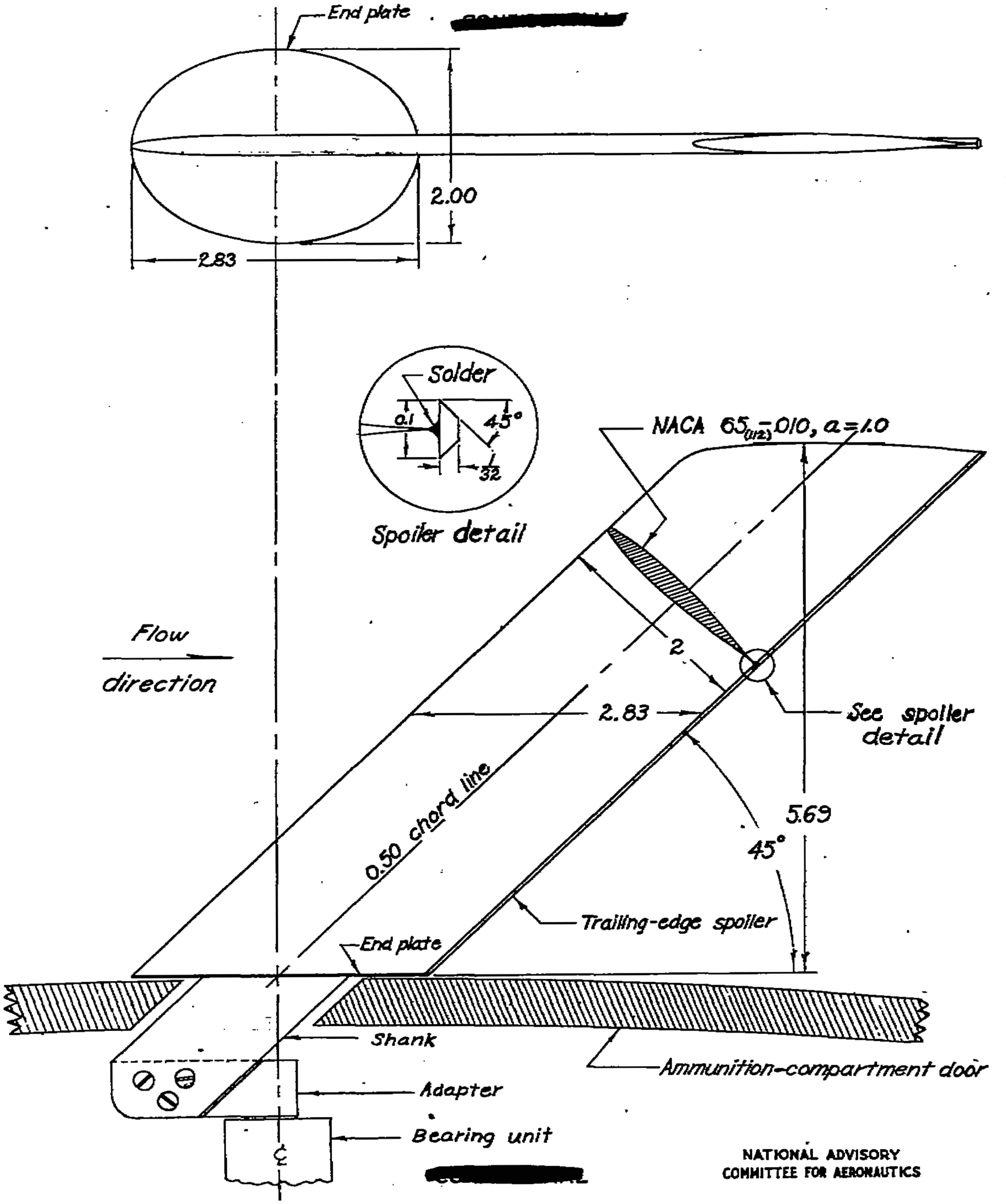

Figure 2.- Details of model and trailing-edge spoiler.
(All dimensions are in inches.)

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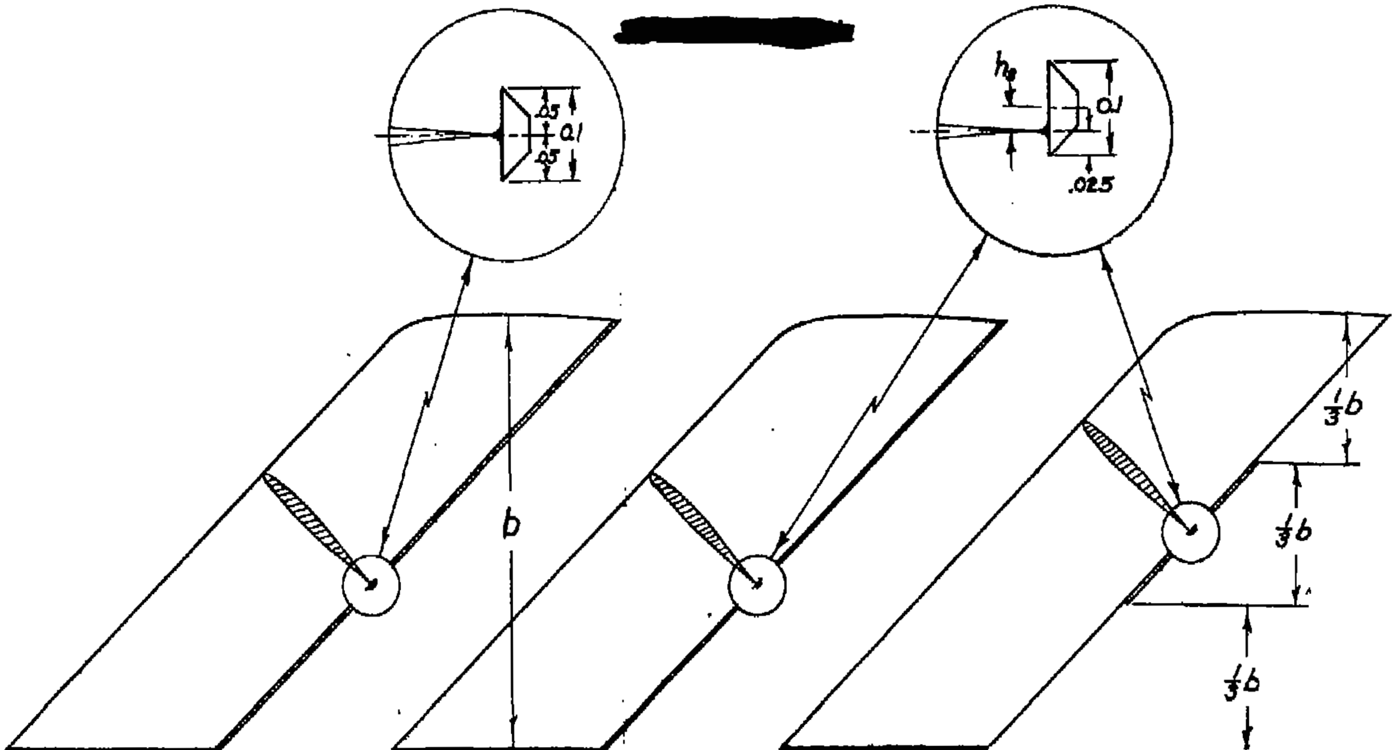

(a) $\frac{b_s}{b} = 1$; $\frac{h_s}{c} = 0$. (b) $\frac{b_s}{b} = 1$; $\frac{h_s}{c} = 0.0088$. (c) $\frac{b_s}{b} = \frac{1}{3}$; $\frac{h_s}{c} = 0.0088$.

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Figure 3.- Spoiler configurations tested.

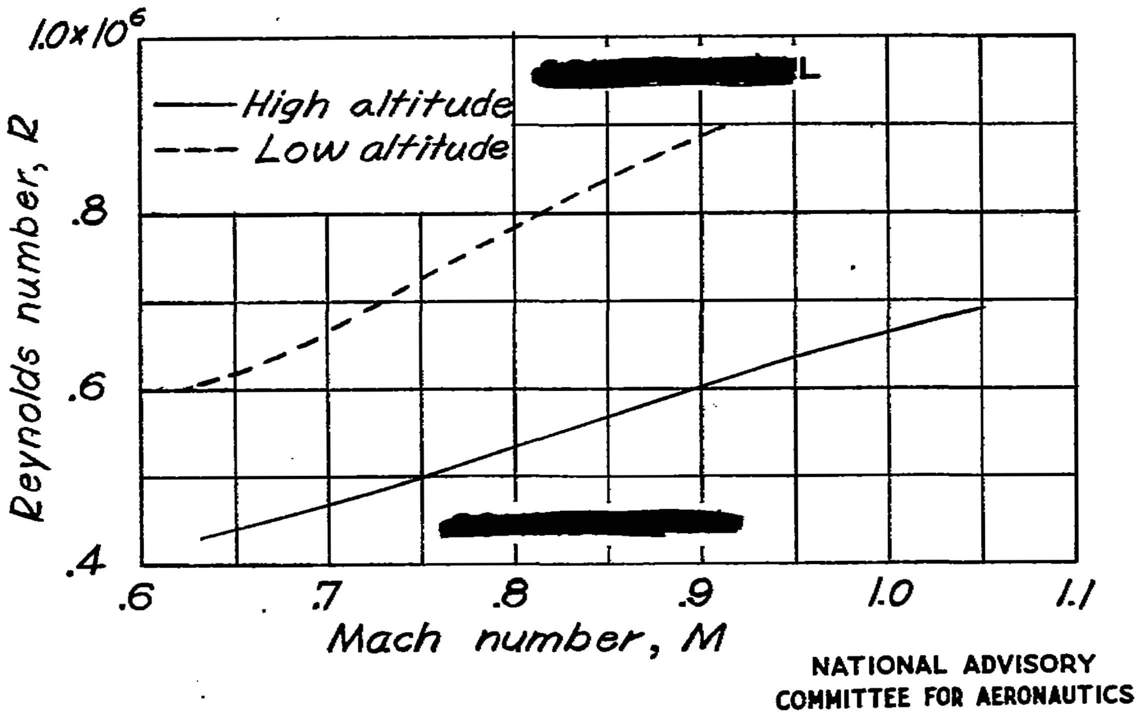

Figure 4.- Average relation between Reynolds number and Mach number during tests.

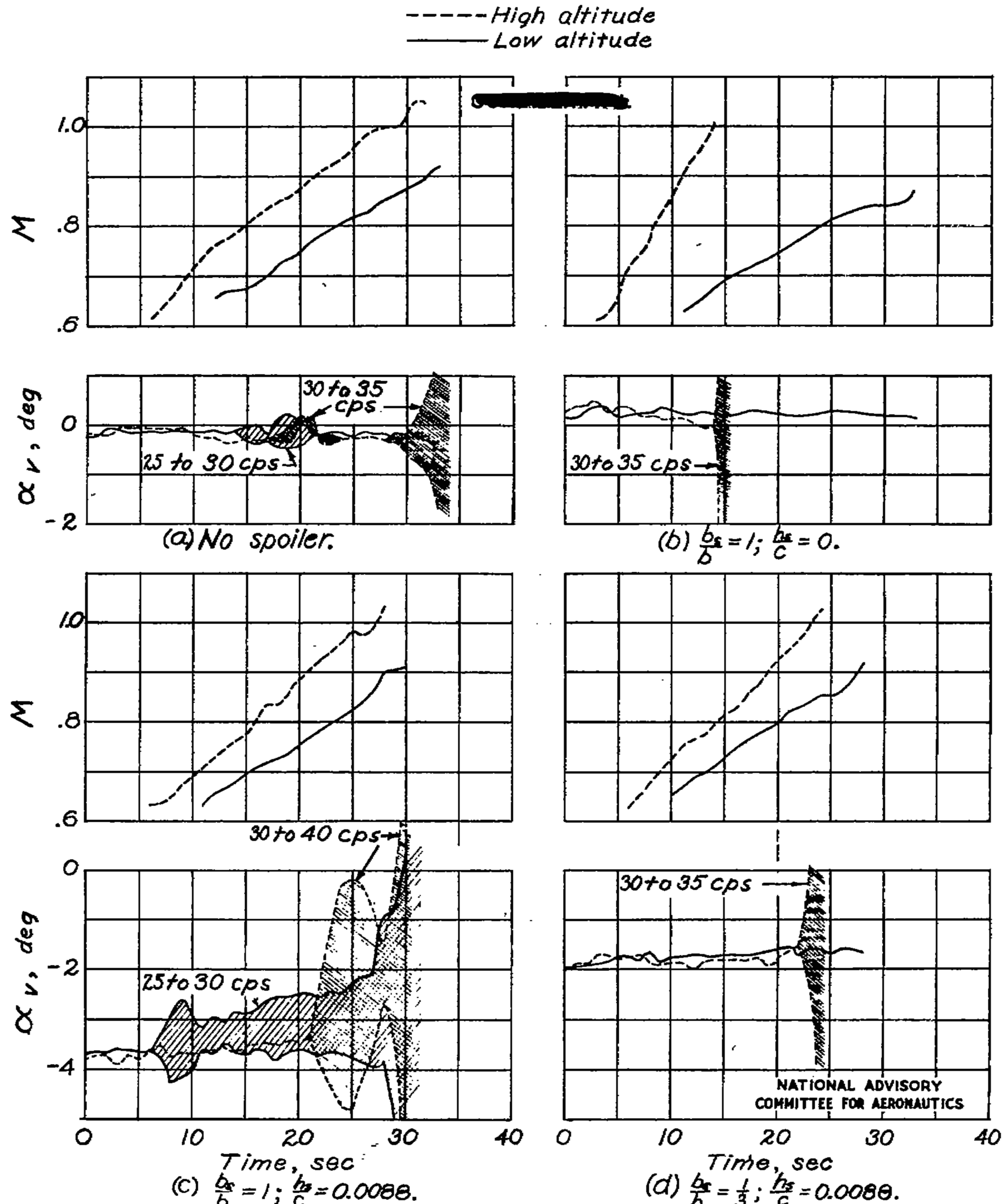

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Figure 5.- Time histories of Mach number and of angle of model relative to angle of reference vane. Shaded areas indicate amplitude of high-frequency oscillation of model.

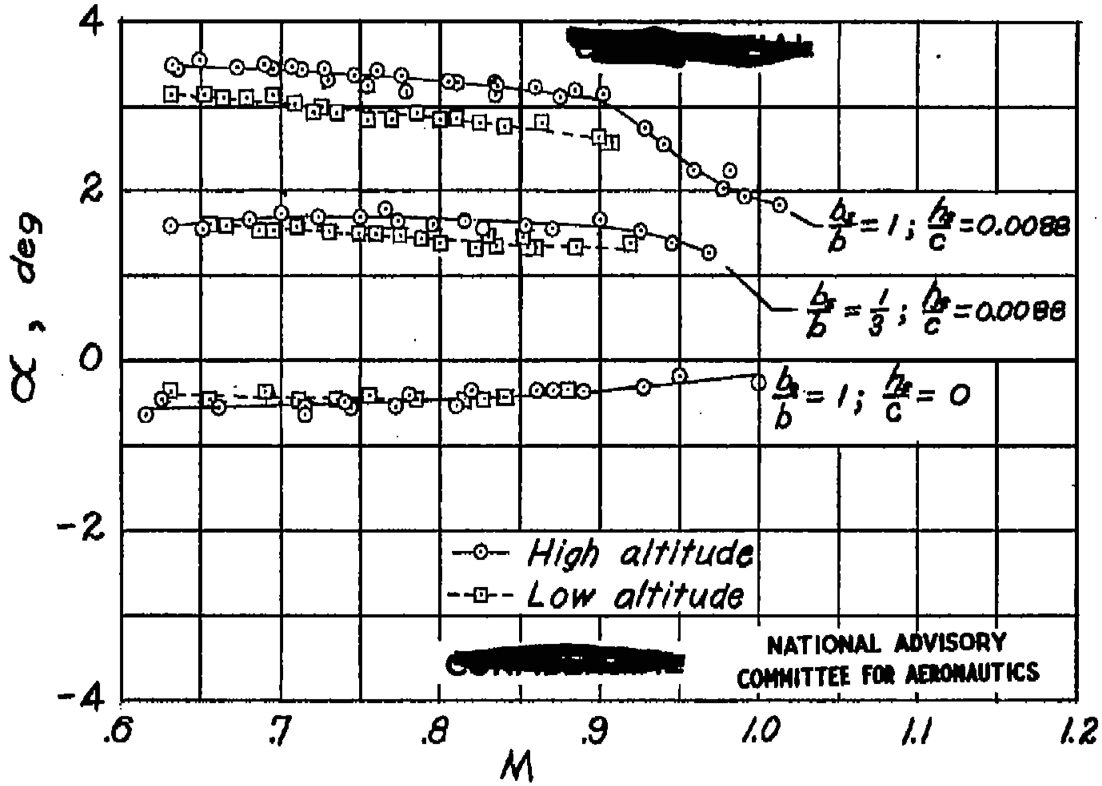

Figure 6.- Variation with Mach number of angle of attack of model for zero pitching moment about the pivot axis.

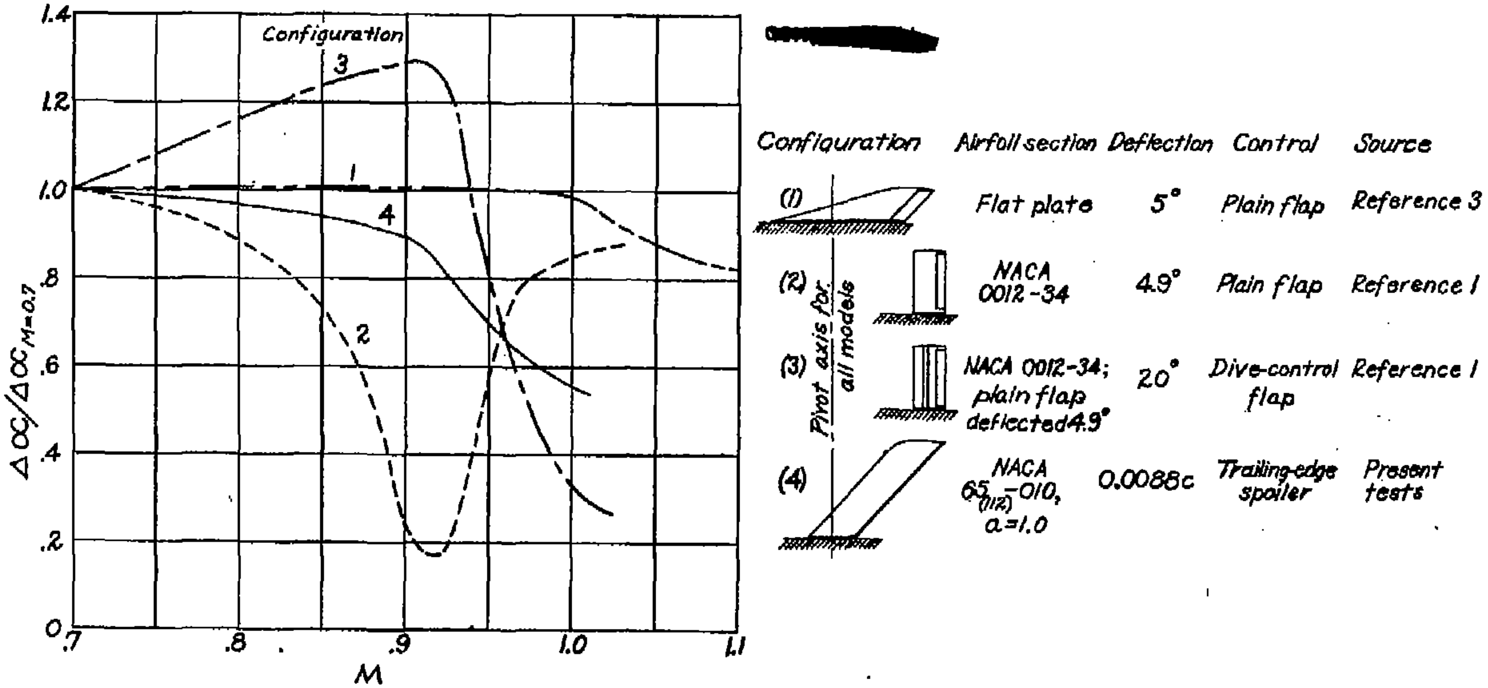

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Figure 7.- Variation with Mach number of control effectiveness for various arrangements of airfoils and control surfaces as determined from floating angle tests.