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RESEARCH MEMORANDUM

A WIND-TUNNEL INVESTIGATION OF THE LOW-AMPLITUDE
DAMPING IN YAW AND DIRECTIONAL STABILITY
OF A FUSELAGE-TAIL CONFIGURATION
AT MACH NUMBERS UP TO 1.10

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NATIONAL ADVISORY COMMITTEE
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A WIND-TUNNEL INVESTIGATION OF THE LOW-AMPLITUDE DAMPING IN YAW AND DIRECTIONAL STABILITY OF A FUSELAGE-TAIL CONFIGURATION AT MACH NUMBERS UP TO 1.10

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SUMMARY

A low-amplitude free-oscillation investigation was made at zero angle of attack of the damping in yaw and directional stability of a fuselage-tail configuration through a range of reduced frequency from about 0.03 to 0.11 and Reynolds numbers from about 4×10^6 to 22×10^6 based on body length at Mach numbers up to 0.94. Limited additional tests were made at Mach numbers up to 1.10.

The results indicate that the damping in yaw decreased appreciably at Mach numbers above 0.8 and became a minimum near Mach numbers of 1.0. At the higher test Mach numbers, the damping increased to a value at least as great as that at subsonic speeds. The directional-stability parameter decreased rapidly at high subsonic speeds.

INTRODUCTION

Certain high-speed aircraft configurations have experienced a lack of damping of low-amplitude oscillations in yaw at high subsonic Mach numbers. (See ref. 1, for example.) This undamped lateral oscillation is difficult to control and reduces the efficiency of the aircraft as a gun platform.

The purpose of the present investigation is to determine the effect of Mach number and Reynolds number on the oscillatory damping in yaw and directional stability of an airplane model through a range of reduced frequency at zero angle of attack. The configuration chosen for the investigation was a fuselage-tail model of a research airplane which is known to be subject to snaking oscillations at transonic speeds.

SYMBOLS

The data are referred to the stability axes, the origin of which is taken as the longitudinal location of the quarter-chord point of the mean aerodynamic chord of the normal wing position of the model tested. The symbols are defined as follows:

a value of damping, ft-lb/radian/sec

b assumed wing span, 2 ft

C mechanical spring constant, ft-lb/radian

$$C_n = \frac{\text{Yawing moment}}{qSb}$$

$$C_{n_r} = \frac{\partial C_n}{\partial\left(\frac{rb}{2V}\right)}$$

$$C_{n_{\dot{r}}} = \frac{\partial C_n}{\partial\left(\frac{\dot{r}b^2}{4V^2}\right)}$$

$$C_{n_\beta} = \frac{\partial C_n}{\partial\beta}$$

$$C_{n_{\dot{\beta}}} = \frac{\partial C_n}{\partial\left(\frac{\dot{\beta}b}{2V}\right)}$$

I_Z yawing moment of inertia of model, ft-lb-sec^2

k reduced-frequency parameter, $\omega b/2V$

M Mach number

P_∞ tunnel stagnation pressure, in. Hg

q dynamic pressure, $\frac{1}{2}\rho V^2$, lb/sq ft

R	Reynolds number based on body length
$r,\dot{\psi}$	yawing velocity, $d\psi/dt$, radians/sec
$\dot{r}$	yawing acceleration, $d^2\psi/dt^2$, radians/sec^2
S	assumed wing area, 0.663 sq ft
$T_{1/2}$	time to damp to one-half amplitude, sec
t	time, sec
V	free-stream velocity, ft/sec
β	angle of sideslip, radians

$\dot{\beta} = d\beta/dt$, radians/sec

ω	circular frequency, radians/sec
ρ	mass density of air or Freon-12, lb-sec^2/ft^4
ψ	angle of yaw (equal to $-\beta$ for wind tunnel), radians

Subscripts:

f	tare value due to mechanical friction
ω	value measured during oscillatory motion
∞	undisturbed free-stream conditions

APPARATUS AND MODEL

The tests were made in the Langley low-turbulence pressure tunnel. The tunnel can accommodate tests in air at pressures of 1 to 10 atmospheres at Mach numbers up to 0.4 and in Freon-12 at pressures of 1/10 to 1 atmosphere at equivalent air Mach numbers from 0.4 to 0.94. Slots in the ceiling and floor of the test section can be opened to permit tests up to a Mach number of approximately 1.1 in Freon. Freon as a test medium gives the added advantage of permitting lower oscillating frequencies for a given reduced frequency.

The model, which is shown in figure 1, was sting mounted in the tunnel. It is a 1/14-scale model of the Bell X-1 airplane. Inasmuch as wings have little effect on damping in yaw or directional stability at zero angle of attack, the model was tested without wings. Horizontal and vertical tails of the model had NACA 65-008 airfoil sections parallel to free stream.

The model was supported at its center of gravity by means of flexure plates which permitted freedom in yaw with restraint in pitch and roll. Electrical strain gages were attached to these flexure plates to record on pen-type recorders the time history of the yaw deflections as the model oscillated freely. An initial displacement of 1° in yaw was produced by an eccentric cam and a direct-current motor located in the model nose and attached to the sting. A photograph of the model with nosepiece off is shown in figure 2. At a given Mach number and Reynolds number, the frequency of oscillation was varied by use of three interchangeable pairs of flexure plates identified herein as flexure plates 1, 2, and 3, which had mechanical spring constants C of 217, 100, and 26 foot-pounds per radian, respectively.

TESTS

The tests consisted of deflecting the model 1° in yaw and then releasing it. The resulting free oscillation was recorded against time. These tests were made through a range of reduced frequency $(\omega b/2V)$ by variation of the free-stream velocity, stagnation pressure, and spring constant of the yaw flexure plates. (See fig. 3(a).) The range of test Reynolds number and Mach number converted to equivalent air Mach number by the method of reference 2 is shown in figure 3(b). A slight variation in stagnation pressure from test to test caused small variations in Reynolds number at each nominal pressure. The model angle of attack was 0° for the entire test.

EFFECTS OF TECHNIQUE

The test technique used in this investigation was selected at a time when more positive methods of measuring aerodynamic damping were unavailable. The technique is severely limited in accuracy in those test ranges where the model motion is influenced by such factors as: (1) turbulence in the free stream, (2) resonant conditions in the free stream caused by presence of the tunnel walls (ref. 3), (3) tunnel-wall reflected shock waves, and (4) sting flexibility.

Unpublished tests made with a hot-wire anemometer in the Langley low-turbulence pressure tunnel show that the level of airstream turbulence in this tunnel is low throughout the Mach number range with the test section slots either open or closed. On the basis of these results, it is believed that the low level of airstream turbulence in the low-turbulence pressure tunnel did not appreciably affect the damping trends determined.

The conditions for resonance of the airstream between tunnel walls have been predicted theoretically in reference 3. For the present tunnel test conditions, the critical condition of resonance occurs near a Mach number of unity. The theory was developed, however, for the two-dimensional case. Inasmuch as the chord length of the lifting surface of the vertical tail was small in comparison with the tunnel width (1:10) and the aspect ratio of the lifting surface was only 2, it appears probable that the effects of resonance of the airstream on model response would be minor for the present investigation.

Calculations on the position of the bow-shock reflections from the tunnel walls (ref. 4) indicate that reflections strike the nose of the model forward of the maximum body diameter only at Mach numbers of 1.03 to 1.05. Since the model was located in the center of the tunnel and was performing oscillations of small amplitude, the reflected waves would be expected to strike the model on opposite sides at nearly the same longitudinal location and with about the same strength. The effect of shock reflections on model response is believed negligible, therefore, even in the range of Mach number from 1.03 to 1.05.

In order to obtain adequate amplitude of oscillations while maintaining scale-model afterbody shape, sting diameter at the base of the model was quite small and resulted in a relatively flexible sting (fig. 1). Initial displacement of the model by the eccentric cam prestressed the sting so that a sting oscillation resulted when the cam was released for each test. Although the model was mechanically coupled to the sting through the relatively weak flexure plates, oscillation of the model on the sting caused very little sting excitation because the mass center of gravity of the model was maintained on the oscillation axis and because of the divergence between the resonant frequencies of the model and of the sting support. Oscillograph records verified the small response of the sting when the model was oscillated. A dynamic analysis made of the first-order effects of the above factors on the measured damping indicated that sting response due to initial model displacement and to model-sting coupling could affect the measured values of damping from approximately 3 to 20 percent. This possible error in measurement would not, of course, affect the trends of measured damping as presented in this paper.

It is recognized that the preceding effects on aerodynamic damping measured with the free-decay oscillation technique will become more pronounced as the actual model damping decreases.

REDUCTION OF DATA

Determination of the damping-in-yaw and directional-stability derivatives by use of the free-oscillation technique is discussed in reference 5. In nondimensional form, they may be expressed, respectively, as

$$\left(C_{n_r}\right)_\omega - \left(C_{n_{\dot{\beta}}}\right)_\omega = \frac{-4I_ZV\left(a - a_f\right)}{qSb^2}$$

and

$$\left(C_{n_\beta}\right)_\omega + \left(k^2C_{n_{\dot{r}}}\right)_\omega = \frac{I_Z}{qSb}\left(\omega^2 - \omega_f^2\right)$$

where the subscript f denotes wind-off test values. The values of a and a_f are determined from the relation

$$a = \frac{\log_e 2}{T_{1/2}}$$

Wind-off test runs were made with various parts of the model removed in order to determine the variation of a_f with frequency for each flexure plate. Values of a_f for use in the damping-in-yaw equation were then interpolated at the wind-on frequency.

The values of spring constant C for the three flexure plates were determined from static calibration. The value of I_Z for the model was then determined from the wind-off tests by use of the expression

$$I_Z = \frac{C}{\omega_f^2}$$

RESULTS AND DISCUSSION

Damping in Yaw

Figure 4 shows the variation of the damping-in-yaw parameter $\left(C_{n_r}\right)_\omega - \left(C_{n_{\dot{\beta}}}\right)_\omega$ with Mach number for the various flexure plates and stagnation pressures. The scatter of test points is an indication of the accuracy of the data based on repeatability and appears to be generally within ±0.05. Within this accuracy, there appears to be no effect of Reynolds number (stagnation pressure). Increasing the reduced frequency (from flexure plate 3 to 1) generally reduced the damping in yaw. It should be noted, however, that this apparent trend is contrary to theory (ref. 6) and is almost within the estimated accuracy of the data. The most important trend to be noted is the change in damping with Mach number as the Mach number approaches unity. The damping begins decreasing at a Mach number of 0.8 and decreases progressively to near the closed-tunnel choke Mach number of about 0.94. For the one test at a Mach number of 1.0 at a stagnation pressure of 19 inches mercury (tunnel slots open), the aerodynamic damping in yaw was apparently sufficiently negative to offset the tare damping of the system so that the model experienced a sustained oscillation of about ±1°. A negative value of aerodynamic damping probably results from the turbulent energy in the airstream which, although very low, can influence the model motion when the model aerodynamic damping approaches zero. At Mach numbers of 1.05 and 1.10, the damping increased to a value equal to or greater than that measured at subsonic speeds.

Directional Stability

Variation of the directional-stability parameter with Mach number is shown in figure 5 for the various flexure plates and stagnation pressures. There was an increase in the stability parameter $\left(C_{n_\beta}\right)_\omega + \left(k^2 C_{n_{\dot{r}}}\right)_\omega$ with a decrease in reduced frequency. The indicated increase in stability at Mach numbers up to 0.85 includes the effect of decreasing reduced frequency (fig. 3(a)). The actual increase with Mach number is therefore less than that shown in figure 5 although some increase would be expected from the Mach number effect on lift-curve slope of the vertical tail. The decrease in stability at Mach numbers from 0.85 to 0.95 is due to the force break on the 8-percent-thick vertical tail in this Mach number range.

Within an estimated accuracy based on repeatability of ±0.03, there is no effect of change in Reynolds number (stagnation pressure) on the directional-stability parameter for the range of test conditions.

CONCLUSIONS

A low-amplitude free-oscillation investigation was made at zero angle of attack of the damping in yaw and directional stability of a fuselage-tail configuration through a range of reduced frequency from about 0.03 to 0.11 and Reynolds number from about 4×10^6 to 22×10^6 at Mach numbers up to 0.94. Limited additional tests were made at Mach numbers up to 1.10. The results indicate the following:

1. The damping in yaw decreased appreciably at Mach numbers above 0.8 and became a minimum near Mach numbers of 1.0. Further increase in Mach number to 1.10 increased the damping to a value at least as great as that measured at subsonic speeds.

2. The directional-stability parameter decreased rapidly at Mach numbers from 0.85 to 0.95.

3. An increase in reduced frequency caused a decrease in damping in yaw and a decrease in directional stability for the test frequency range.

4. Within the accuracy of the data, the damping in yaw and directional stability were not affected by change in Reynolds number.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., March 5, 1957.

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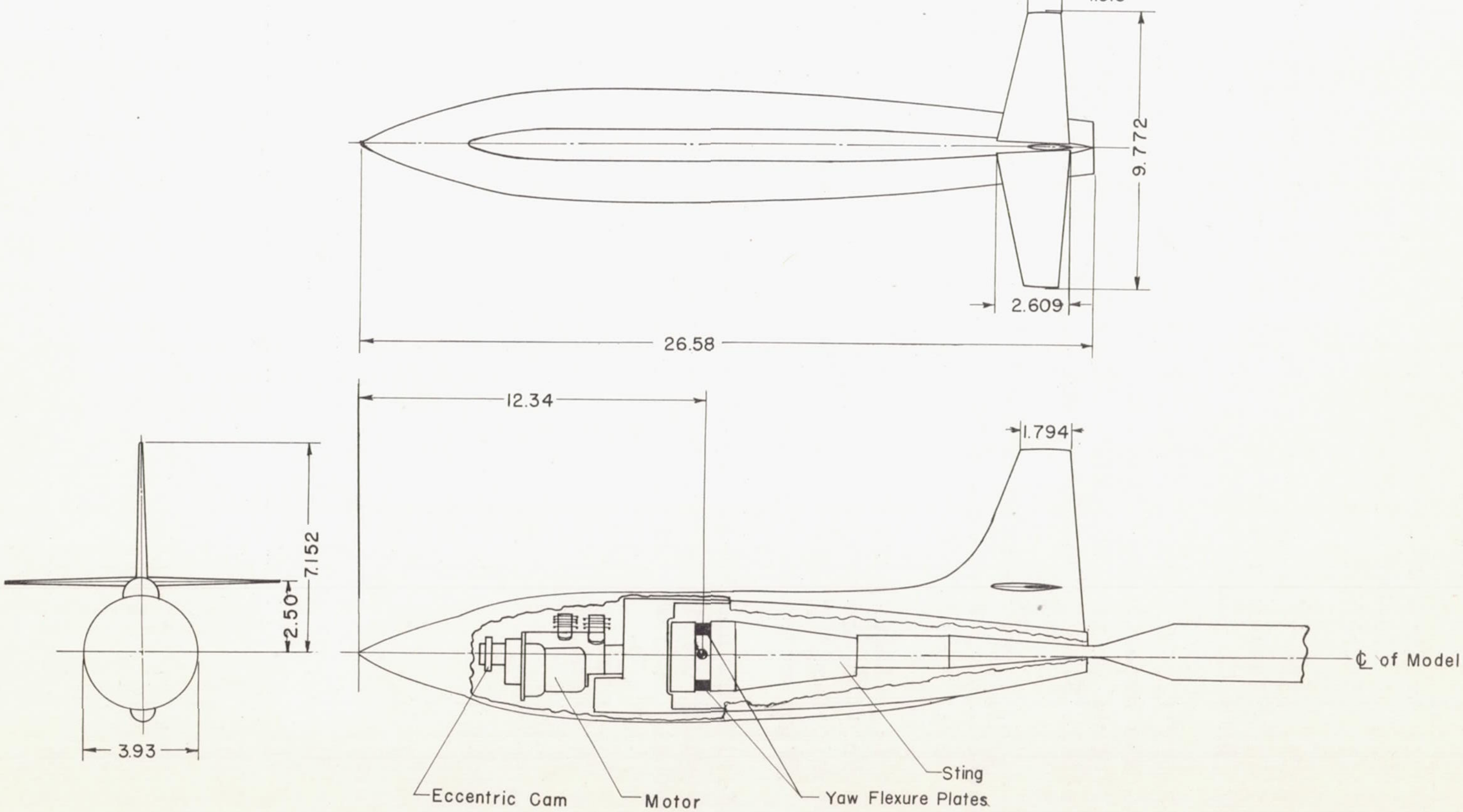


Figure 1.- Sketch of model. (All dimensions in inches.)

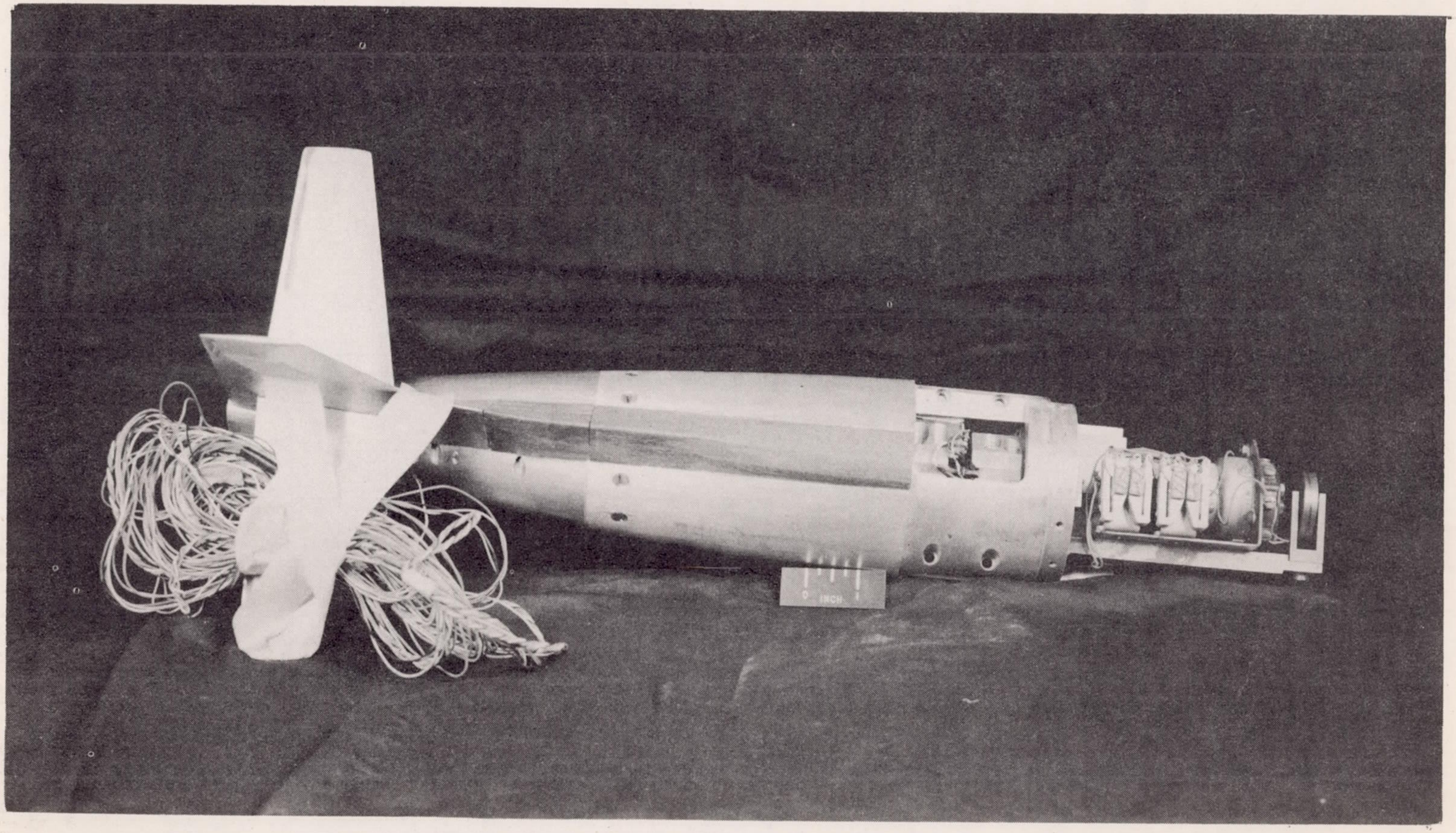
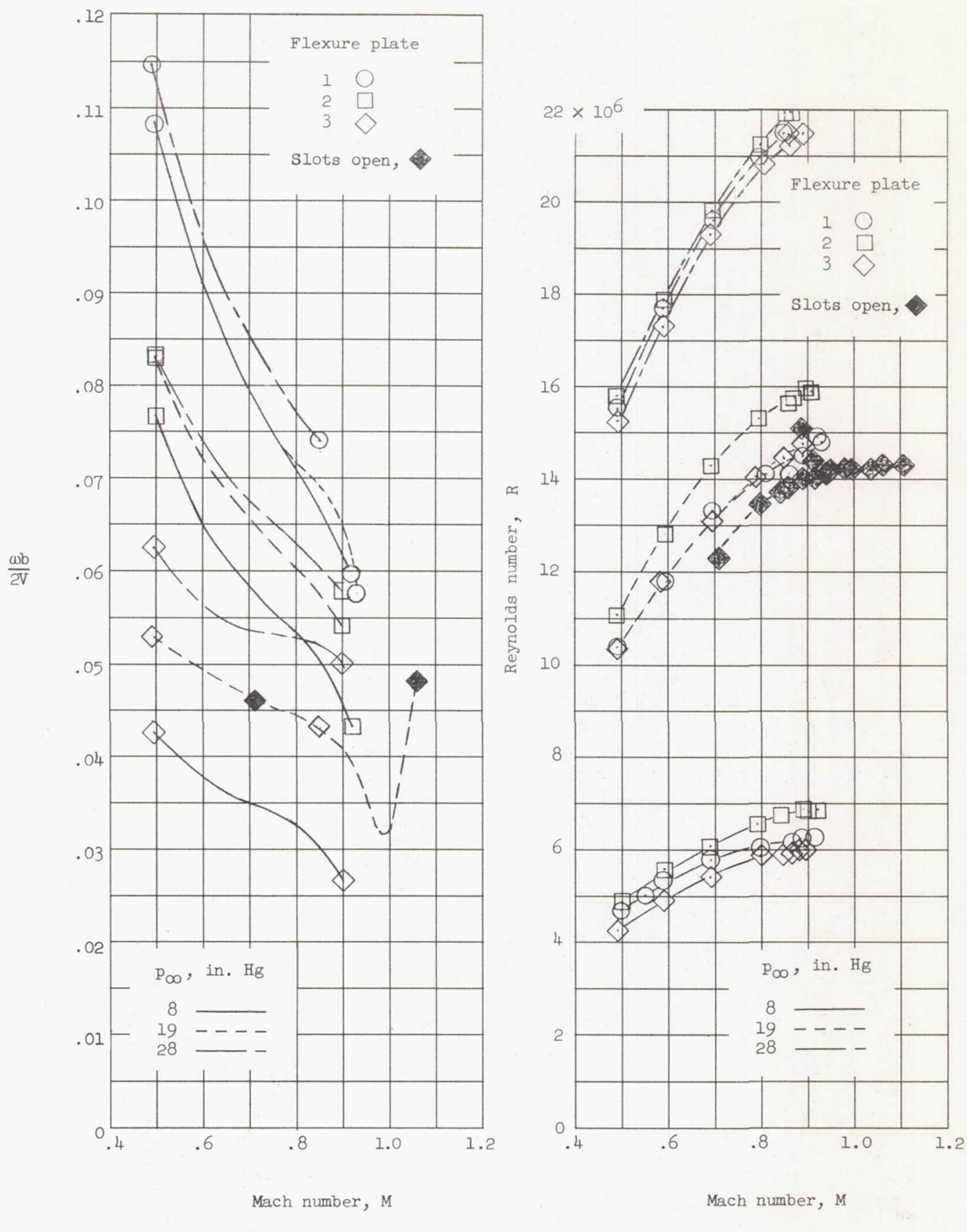


Figure 2.- Photograph of the model with nose piece off. L-94895



(a) Reduced frequency. (b) Reynolds number.

Figure 3.- Variation with Mach number of the reduced frequency and Reynolds number for various flexure plates and stagnation pressures. (Symbols are for identification and not actual data points.)

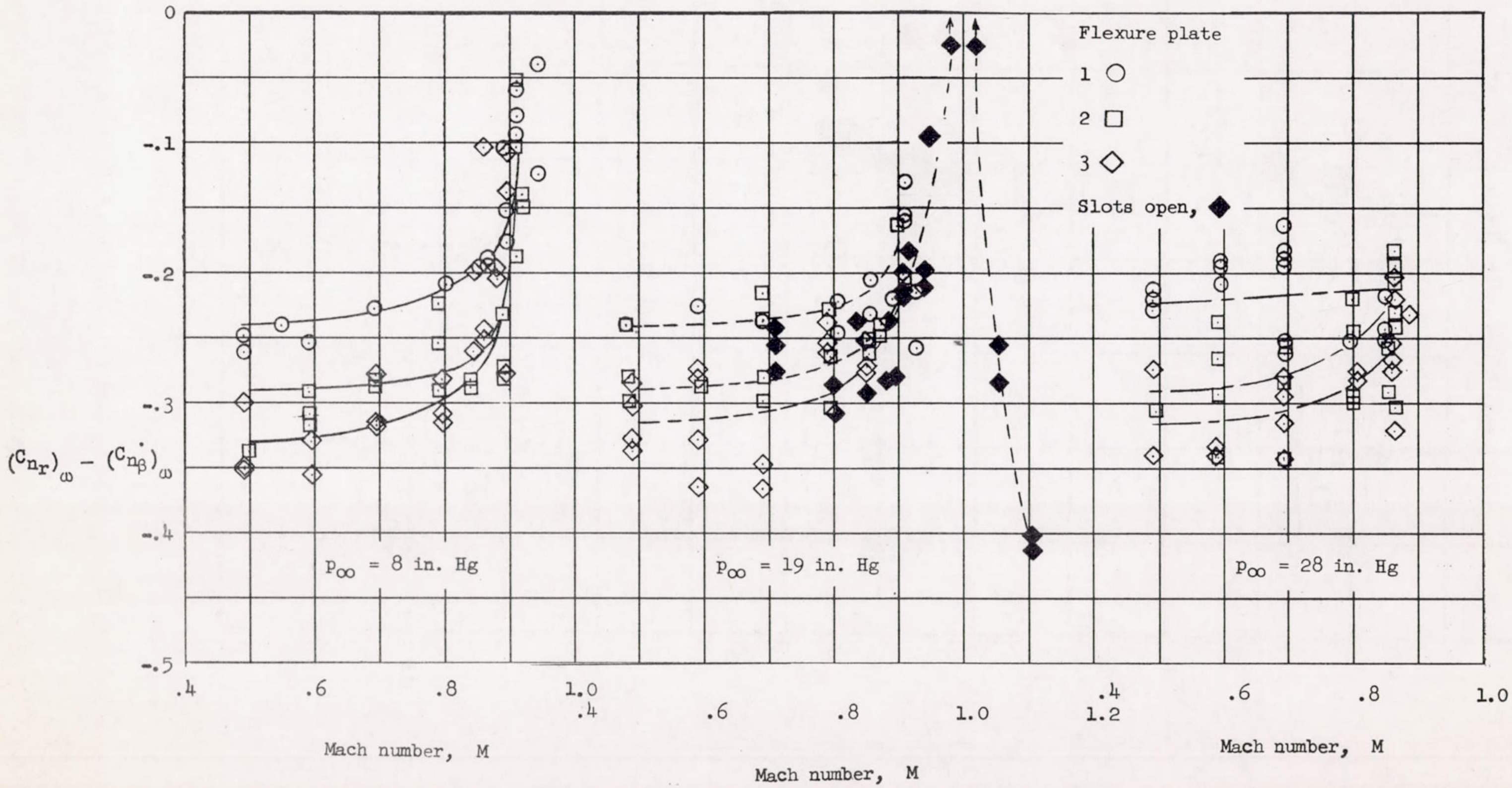


Figure 4.- Variation with Mach number of the damping-in-yaw parameter for various stagnation pressures and flexure plates.

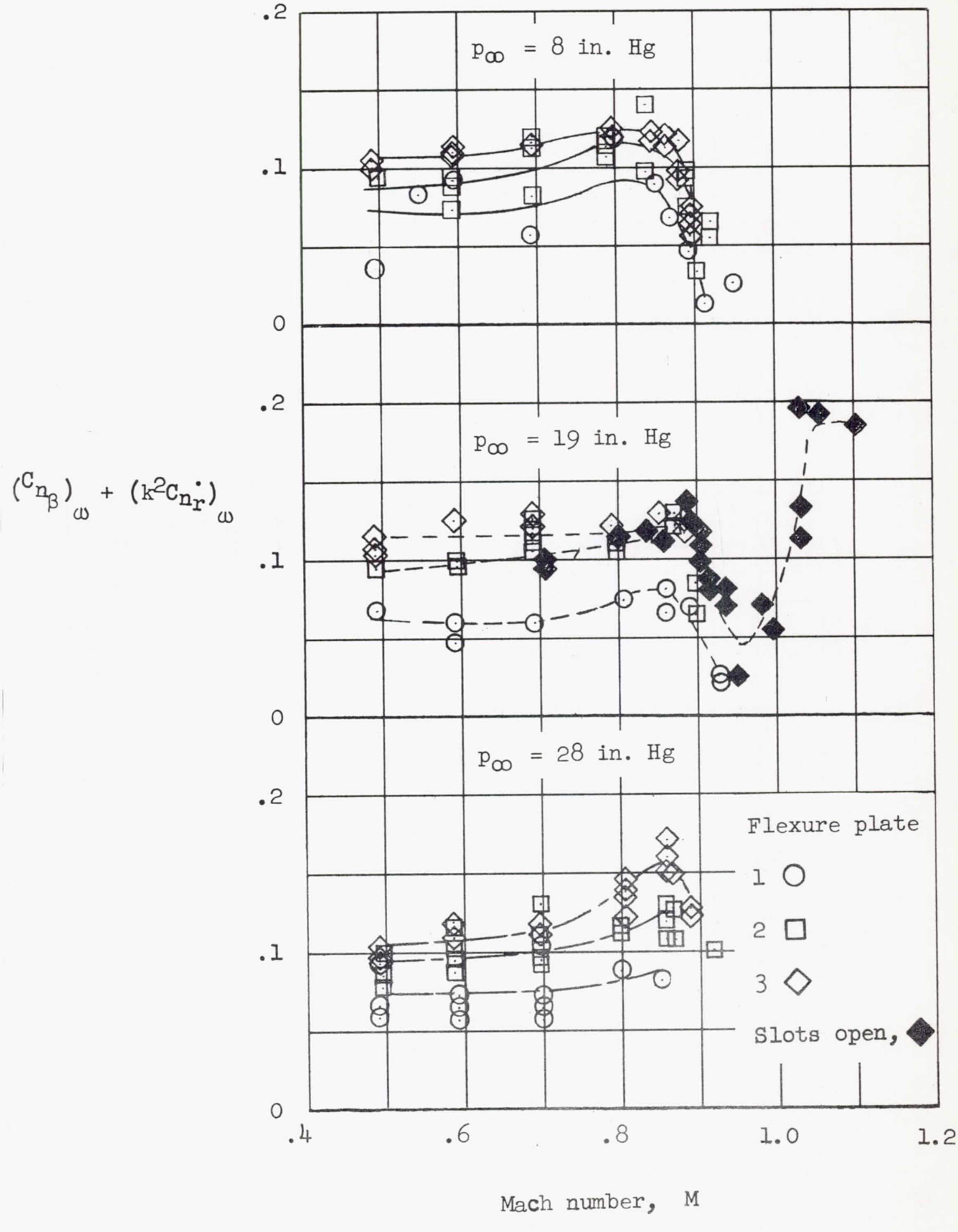


Figure 5.- Variation with Mach number of the directional-stability parameter for various flexure plates and stagnation pressures.