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FLIGHT CHARACTERISTICS AT TRANSONIC SPEEDS

I - P-80A AIRPLANE INVESTIGATION

By H. H. Brown

Ames Aeronautical Laboratory

Extensive flight tests have been conducted at the Ames Laboratory on the P-80A airplane well up into the transonic range.

Previous to the flight tests, a $\frac{1}{3}$-scale model of the airplane was thoroughly tested in the Ames 16-foot high-speed tunnel up to a Mach number of 0.85. Analyses of these test results indicated that the airplane should possess satisfactory stability and control characteristics up to the maximum Mach number tested. The basis for this opinion is the results shown in the first two figures.

Figure 1 presents the elevator angle for trim in level flight at 20,000 feet as a function of Mach number. The solid curve shows the results based on the wind-tunnel tests. For purposes of comparison the flight test results are shown by the dashed curve. The tucking-under tendency as indicated by the increase in up-elevator angle required above a Mach number of 0.70 was not considered serious since the pilot would presumably have ample warning. The change with Mach number of the tucking under was less severe in flight than indicated from the wind-tunnel tests.

Similarly in figure 2 is shown the variation of required elevator angle with acceleration factor for Mach numbers of 0.80, 0.825, and 0.85. The solid curves are based on the wind-tunnel results and the dashed curves on flight tests. There was nothing shown here which would predict a pitch-up.

In spite of the reassuring nature of the wind-tunnel results and the careful manner in which the flight tests were conducted, a condition was encountered during a dive at a Mach number of about 0.85 which produced a violent and inadvertent stall. This particular feature of this airplane remains one of the major factors which limit operation of the airplane to still higher Mach numbers.

Figure 3 shows a time history of some of the quantities evaluated during this dive. The lift coefficient roughly follows the elevator motion up to about 13.25 seconds. At this point the lift coefficient rapidly began to increase with only a small change in elevator angle and no change in stick force.

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Reexamination of the wind-tunnel data failed to indicate the cause of the pitch-up for two reasons:

(1) The strength of the model limited the C_L to a value of 0.40 at a Mach number of 0.85, which was a lower value than the value encountered during the pitch-up.

(2) Tunnel conditions at the larger lift coefficients and highest Mach numbers were close to choking.

Combining the data obtained during the dive with the wind-tunnel results did afford a partial solution to the problem. The fuselage due to its high critical Mach number was eliminated as a cause of the pitch-up. The wing pressure distributions made during the dive also enabled the pitching-moment coefficient of the wing to be eliminated as a cause. The spanwise loadings derived from the pressure measurements showed only a minor difference compared to the lower Mach number results and therefore downwash changes were ruled out. Lastly, the effects due to the shift in the angle for zero lift of the wing were obtained from the wind-tunnel data.

In figure 4 the total out-of-trim pitching-moment coefficient of the airplane during the pitch-up is shown with C_N as the abscissa. Also shown is the out-of-trim pitching-moment coefficient furnished by the negative shift in the angle for zero lift as the airplane decelerated. The difference between these two curves represents the destabilizing influence that must be attributed to something other than the wing and fuselage.

In order to determine whether the flow conditions at the tail or whether the tail characteristics were responsible, the isolated horizontal tail was tested in the Ames 16-foot high-speed tunnel at the Mach numbers and over the angle of attack and elevator angle range reached during the dive.

The results showed large changes in effectiveness of the tail, especially at the high elevator deflections encountered in the dive. This change in effectiveness, plus the immersion of the tail in the wake at higher lift coefficients, which tends to accentuate its effect, accounts for the unstable action of the airplane in the pull-up.

On the left side of figure 5 the variation of tail pitching moment has been plotted against the Mach number determined from the isolated tail tests. The elevator deflection, which was used in the dive, is 12°, and the tail angles of attack were those which would be encountered by the tail in the process of the pitch-up.

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If it were presumed that the pull-up were made at a constant Mach number of 0.87 and that the tail operated at free-stream Mach number, a crossplot along the vertical line at a Mach number of 0.87 would give the tail contribution to the pitching moment of the airplane. The result is shown by the solid line on the right side of figure 5 with the C_N of the airplane as the abscissa. It can be seen that the tail contribution is stabilizing.

An estimation based on wind-tunnel results of the wake location shown in figure 6 shows, however, that the tail was in all probability passing into the wake as C_N increased.

In the upper part of figure 6, sketch A represents conditions at a low lift coefficient (about 0.10) and the tail is practically out of the wake. Sketch B shows conditions at a lift coefficient of about 0.50, and sketch C shows conditions at a lift coefficient of 0.80 when the tail was well into the wake. The lower part of figure 6 shows the values of q_H/q and decrement in Mach number at the tail corresponding to the upper sketches. Thus as the airplane lift coefficient changed from 0.5 to 0.8 at a constant free-stream Mach number, the tail Mach number decreased by about 0.08.

Figure 5 shows the effect of this Mach number decrease on the tail contribution to the airplane pitching moment.

In this case as the normal-force coefficient C_N at a constant Mach number is increased at a constant Mach number the values shown by the dashed curve are produced taking into account the decrease in Mach number at the tail. This results in a tail contribution to the airplane pitching moment which is neutral or destabilizing above a lift coefficient of 0.7.

Adding this type of tail pitching-moment contribution to the change in trim which occurred due to the decreasing Mach number during the dive means an apparent instability of the entire airplane above a lift coefficient of 0.5.

It is to be emphasized that this instability did not exist at all other elevator deflections. For example, figure 7 shows the results of a similar analysis for an elevator angle of 4.4°, which was required for trim in level flight at a Mach number of 0.87. In this case the influence of the wake is considerably less important and the tail contribution is stabilizing throughout the angle-of-attack range. This accounts for the normal variation of elevator angle against Mach number and elevator angle against acceleration of

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gravity g over the limited range for which it was possible to derive these results from the wind-tunnel tests of the complete model.

Since the airplane is unstable at high Mach numbers, it becomes of interest to determine what rates of elevator response are required of the pilot to forestall such a pitch-up as did occur. By using the conditions at the start of the pitch-up, the change in the normal acceleration factor with time was determined with various rates of elevator motion using a step-by-step solution of the equations of motions. The results are shown in figure 8. With no time delay in the pilot response a rate of elevator motion of 2° per second was sufficient to prevent the acceleration building up to a stall. With a quarter-second time delay, which is about the best response which can be expected from a pilot, a rate of elevator motion of almost 4° per second was needed to prevent a stall. For time delays much longer than a quarter of a second, the required rates became unreasonably large. Actually a pilot responds to a change in acceleration and since the pitch-up motion at first produced only a small acceleration change, the pilot's response time was quite long. As a result, it is improbable that a pilot will be able to prevent such a stall.

The fact that airplanes with higher critical Mach numbers have exceeded Mach numbers of 0.85 or 0.86 without similar stability and control troubles may merely indicate postponement of this danger to a higher Mach number. The necessity for testing in the transonic wind tunnels to higher lift coefficients and larger elevator deflections is apparent. If this is not possible because of the limitation of the model or wind tunnel, then recourse can be made to a study based on isolated tail tests and wake profiles similar to that done in this case.

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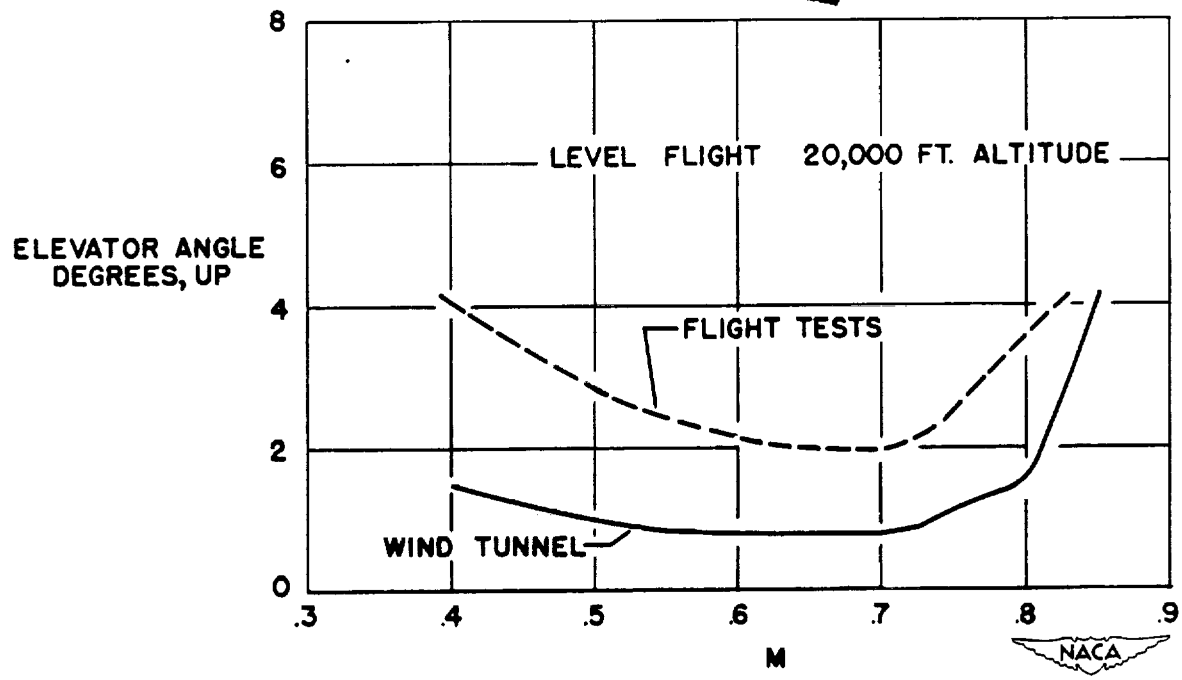

Figure 1.- Variation of elevator angle for trim in level flight at 20,000 feet with Mach number.

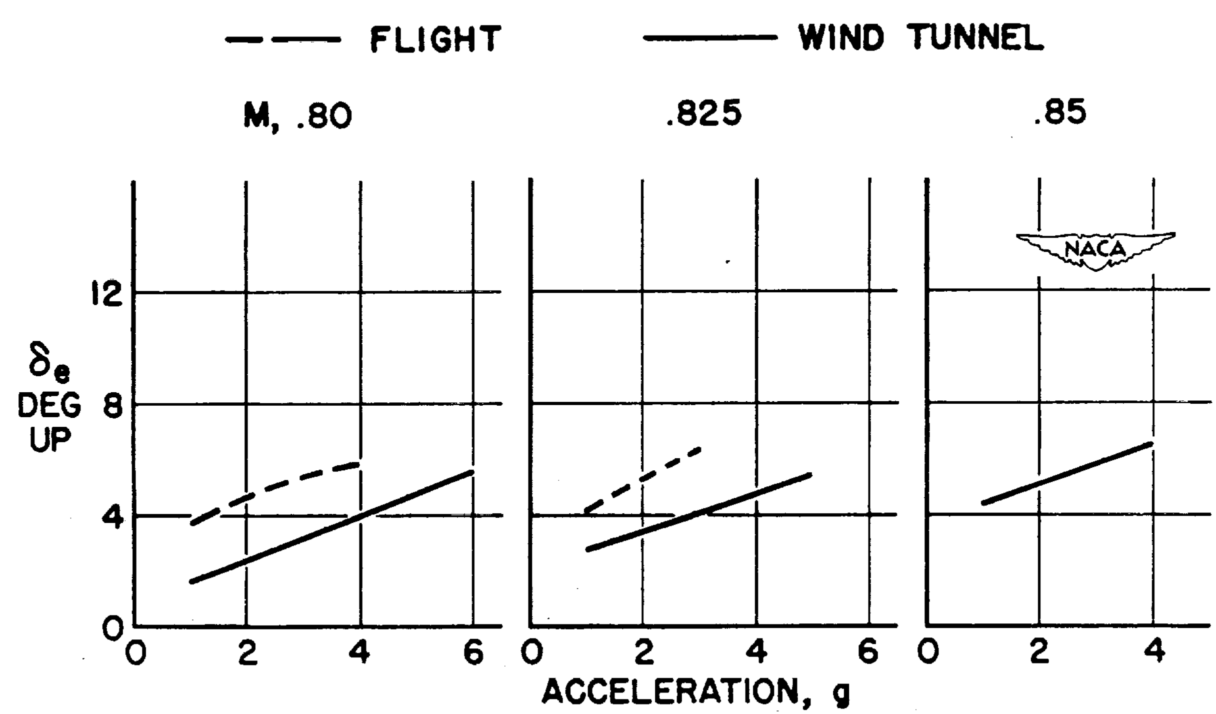

Figure 2.- Variation of required elevator angle with acceleration factor at high Mach numbers.

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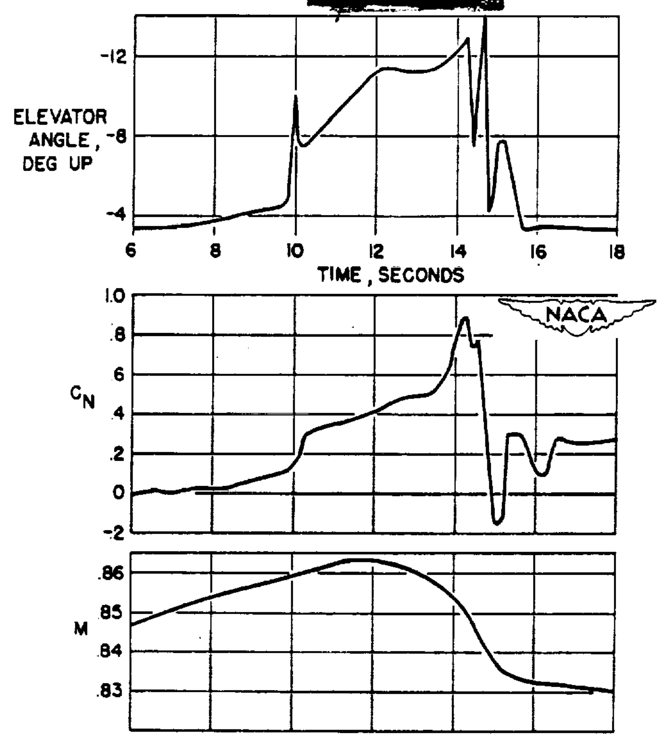

Figure 3.- Time history of dive.

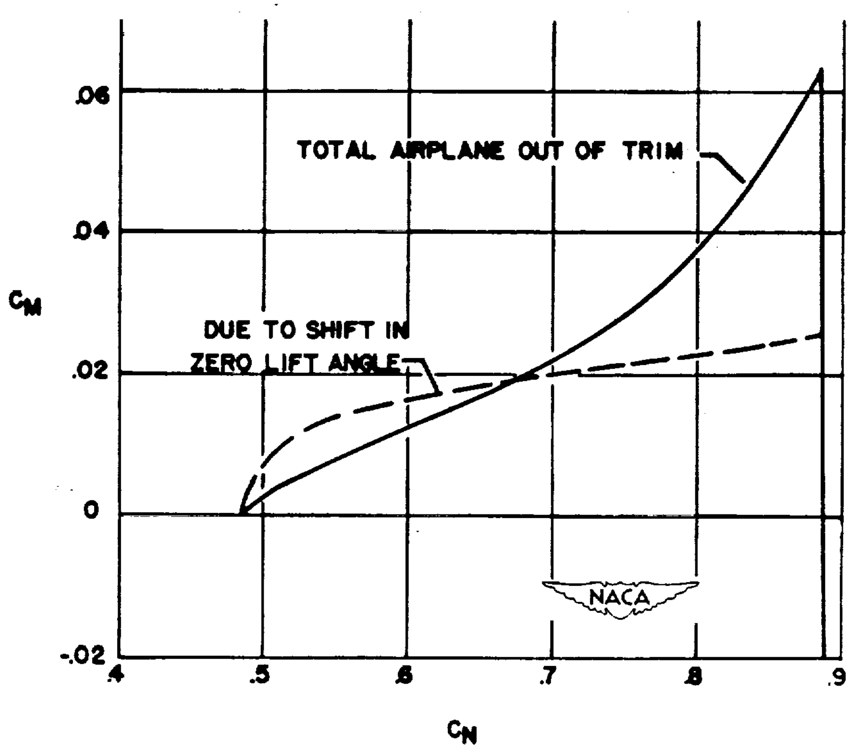

Figure 4.- Out-of-trim pitching-moment coefficients during pitch-up.

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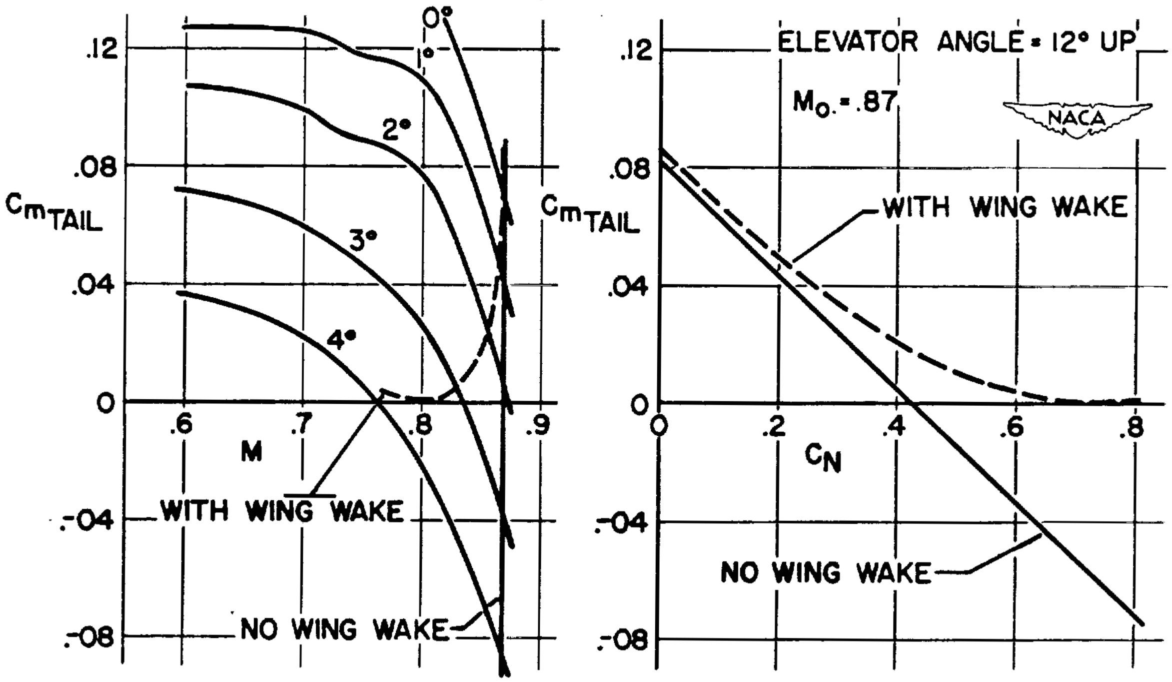

Figure 5.- Tail pitching-moment coefficient with large elevator deflection.

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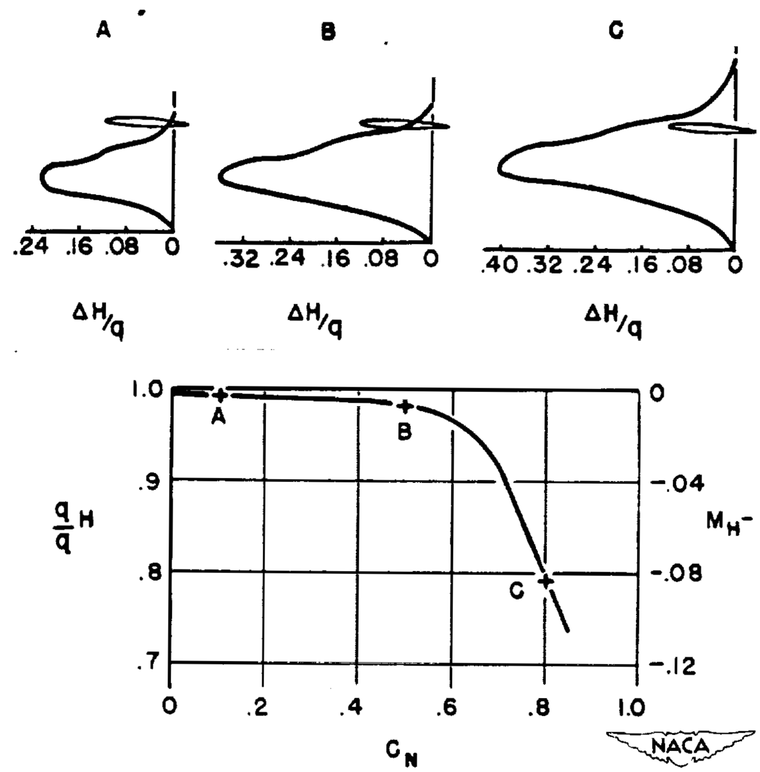

Figure 6.- Estimated wake at the tail and the effect on local Mach number.

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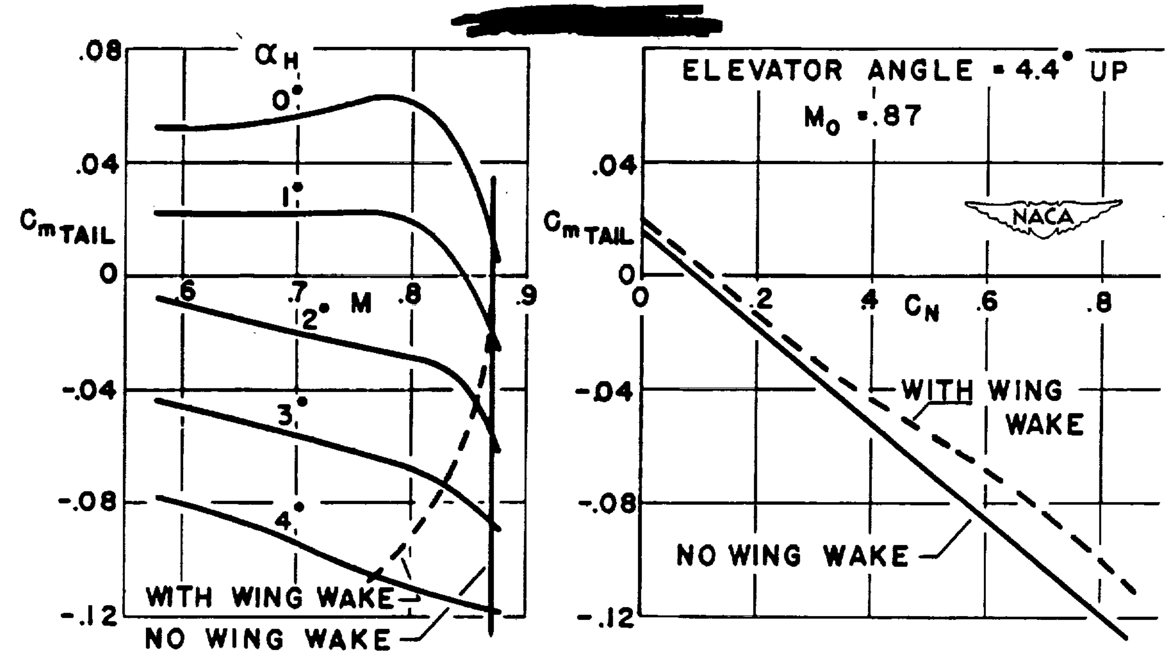

Figure 7.- Tail pitching-moment coefficient with a moderate elevator angle.

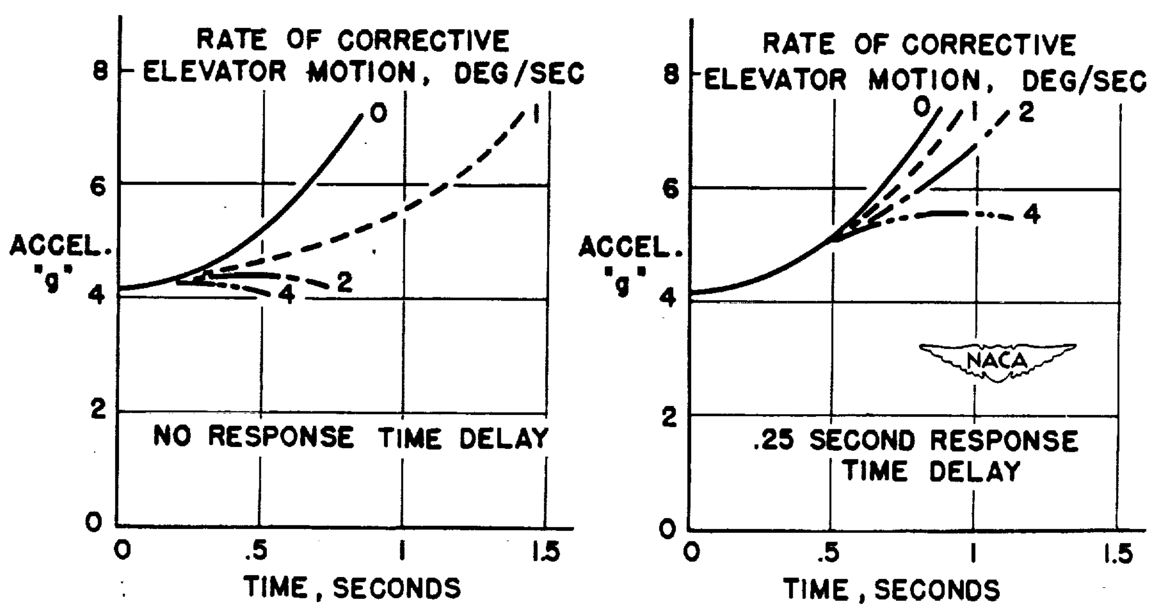

Figure 8.- Effect of rate of response and time delay in preventing a stall.

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FLIGHT CHARACTERISTICS AT TRANSONIC SPEEDS

II - RESEARCH AIRPLANES

By Walter C. Williams

Langley Memorial Aeronautical Laboratory

INTRODUCTION

The Air Forces, the Navy, and the NACA have been engaged in a cooperative program for the development and procurement of a series of research airplanes which would have potential characteristics necessary for level flight in the transonic- and supersonic-speed zones. This program was undertaken in anticipation of the increased importance of flight research in the transonic-speed range where the aerodynamic characteristics of airplanes were known to show large and sudden changes. The range of airplane configurations flying or under construction include straight-wing types with conventional airfoil sections, straight-wing types with a supersonic airfoil section, sweptback-wing types, and tailless sweptback-wing types. The manufacturers involved are Bell Aircraft Corp., Douglas Aircraft Co., and Northrup Aircraft Corp. Two types of these airplanes are flying: the Douglas D-558 Phase I airplane procured by the Navy; and the Bell XS-1 procured by the Air Forces. The airplanes represent the first phase of the program and are being used to explore the limits to which an airplane of relatively conventional design can be flown.

The Navy procured from the Douglas Aircraft Co. the D-558 Phase I airplane. This airplane has a 10-percent-thick straight wing with an aspect ratio of 4 and an 8-percent-thick horizontal tail. The power plant is a TG-180 turbojet engine. The Douglas Aircraft Co. recently turned one of these airplanes over to the NACA at Muroc, Calif. The installation of NACA recording instrumentation has been completed and flight tests of this airplane are expected to begin this week. This airplane will be used for the measurement of stability and control characteristics and over-all aerodynamic loads by use of strain gages throughout the allowable speed range of the airplane. It is expected that a second D-558 Phase I airplane will be delivered to the NACA within the next several weeks and this airplane will be used for detailed measurements of the pressure distribution on the wing and on the horizontal tail.

The Bell XS-1 airplane was procured by the Air Forces. This airplane has a straight wing with an aspect ratio of 6 and is powered by an RM-1 liquid oxygen-alcohol rocket engine. Two of these airplanes have been completed. One airplane has a 10-percent-thick wing

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and an 8-percent-thick horizontal tail; whereas, the other airplane has an 8-percent-thick wing and a 6-percent-thick tail.

The acceptance tests on the XS-1 airplane conducted by the Bell Aircraft Corp. have been completed. During these tests NACA instruments were installed to measure stability and control characteristics and aerodynamic loads up to a Mach number of 0.8 which was the contractural limit of the tests. These tests showed that the airplane had good handling qualities with no unusual characteristics.

Upon completion of the acceptance tests, one XS-1 airplane (with a 10-percent-thick wing and an 8-percent-thick horizontal tail) was assigned to the NACA for a systematic step-by-step investigation of flight to exploit the full capabilities of the type in the transonic-speed range. The same instrumentation used in the acceptance tests will be used in the early phase of these tests. Later tests will include detail pressure-distribution measurements. These tests are just getting under way having been delayed by mechanical difficulties. The other XS-1 airplane (with an 8-percent-thick wing and a 6-percent-thick horizontal tail) was taken over by the Flight Test Division at Wright Field for use in an accelerated transonic-flight program. These tests would differ from NACA tests in that no detailed investigations would be made, and as large an increase in Mach number as compatible with safety would be made in each flight. If necessary, flight would be made at extreme altitudes (50,000 to 60,000 ft). This is a cooperative program between the Wright Field Flight Test Division and the NACA. NACA instrumentation is used on all flights, data reduction and analysis are performed by NACA personnel, and the flying is done by a Wright Field Flight Test Division pilot. The instrument installation, however, is not as comprehensive as in the NACA XS-1. Telemetering and recording instruments are used to measure airspeed altitude, elevator, right aileron and stabilizer position, normal, transverse, and longitudinal acceleration, shear and bending moment on the right horizontal tail, and bending moment on the right wing. These tests have been in progress several months and the data presented herein are results obtained in the accelerated transonic program up to a Mach number of 0.92.

TESTS, RESULTS, AND DISCUSSION

A preliminary airspeed calibration was made during the acceptance tests. These results showed the static-pressure error to be of

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the order of 1/2 percent up to a Mach number of 0.8. As the flights of the airplane progressed to higher speeds, calibration of the static-pressure error of the airspeed head was made. The calibration was made using radar to obtain true altitude. The results of the calibration up to a Mach number of 0.92 are given in figure 1 where the error is expressed as the ratio of error in Mach number to corrected Mach number and is plotted as a function of corrected Mach number. Figure 1 shows that below a Mach number of 0.83 the airspeed head is indicating static pressure lower than true static pressure; whereas above a Mach number of 0.83, an increasing error in static pressure above true static is indicated. It is believed that this variation in static-pressure error above a Mach number M of 0.8 is caused by the formation of a shock on the airspeed head itself and the shock is moving back on the head towards the static holes. No correction was applied to the total-head measurement since the total-head measurement is not expected to be affected by shock-wave formation until a free-stream Mach number of at least unity is reached. The airspeed head used in this case is a Kollsman Type D-1 high-speed head mounted on a boom 1-chord length ahead of the left wing tip.

Most transonic flight tests have been limited by the changes in longitudinal stability and trim, and these data have been of primary concern in the XS-1 tests. The results obtained are presented in figure 2 where elevator position and force are plotted as functions of Mach number for two stabilizer positions. The elevator positions shown here are measured relative to the stabilizer, and the stabilizer positions are relative to the fuselage reference line. With the stabilizer set at an angle of incidence i_t of 1.0°, the pilot stopped at a Mach number of about 0.88 because of the large trim forces required and the forward position of the stick. A nose-down trim change is indicated at the highest Mach number. With the stabilizer set at an angle of incidence of 2.2° the pilot continued flight up to a Mach number of 0.92. In going to this speed, three trim changes were encountered. The first, which began at a Mach number of about 0.8 was in the nose-down direction which the pilot corrected with up elevator. Above a Mach number of about 0.87, the nose-down trim condition is alleviated and the airplane tends to pitch upward, and then, the pilot corrected with down elevator; at the highest Mach number the airplane is again showing a tendency toward nose-down trim position. Most tests have terminated somewhere in the region of the first trim change because with conventional fighters the control forces involved are large. In the present case the range of forces in the trim changes is of the order of 10 pounds. The changes in elevator angle for trim were also not large (of the order of 4°). Because of the small control forces and motions, the

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pilot did not object to the unusual trim changes. The forces are low in this case because the tests were run at a moderately high altitude (about 30,000 ft) and because the elevators are very small. With a larger airplane or at a lower altitude these control characteristics would probably be objectionable. Data from the Langley 8-foot high-speed-tunnel tests and from wing-flow tests of XS-1 models are in general qualitative agreement with flight data.

From the elevator positions required for trim with two stabilizer positions, a measure of the relative effectiveness of the elevator was obtained. These data are shown in figure 3 where the ratio of the change in stabilizer incidence $\Delta\alpha_t/\Delta\delta_e$ to change in elevator position is plotted as a function of Mach number M. Between a Mach number of 0.72 and 0.87 the relative elevator effectiveness is reduced by more than 50 percent.

This reduction in elevator effectiveness in the speed range tested affects the magnitude of the trim changes as noted by the pilot, but in figure 4, where the pitching-moment coefficient of the wing-fuselage combination is plotted as a function of Mach number, it can be seen that the trim changes are being caused by changes on the wing. These data were obtained by using measured values of horizontal tail loads. Very little data have been obtained to show the longitudinal stability in accelerated flight but it is indicated that the stability as evidenced by the pilot, that is the elevator motion required to produce a given acceleration, is greatly increased above a Mach number of 0.85. Some of this increase in the elevator angle per lift coefficient C_L is caused by the decrease in elevator effectiveness, but the data though meager, indicate that the airplane is becoming more stable. These characteristics will be investigated in detail during the NACA tests of the XS-1.

Difficulties have been experienced in recent tests at transonic speeds with one-dimensional flutter or buzz. There has been no evidence of buzz in the data of the XS-1 tests. One probable contributing factor to the absence of this oscillation in addition to the thin wing section is the large amount of friction in the aileron control system. The friction in the ailerons is of the order of 20 foot-pounds. The aerodynamic hinge-moment coefficient for the dynamic pressure q corresponding to a Mach number of 0.85 at 30,000 feet and neglecting the effects of Mach number on the hinge-moment coefficient is of the order of 6.9 foot-pounds per degree. Hydraulic dampers are installed but have not been used. There has been no evidence of abrupt changes in the floating tendencies of the ailerons. The pilot did report a right wing heaviness which he noticed at a Mach number of about 0.88 and which continued up to a

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Mach number of 0.92. Figure 5 shows that the right aileron angle for trim increased in the downward direction with increasing Mach number.

An unusual unclamped lateral oscillation has occurred in some flights. Because of the usual stability boundaries it would be expected that the airplane would be stable because the directional stability is very high and the lateral stability is moderate. The oscillations have occurred in steady gliding flights and in turns from a Mach number of 0.7 to a Mach number of 0.85. It was thought that these oscillations were possibly caused by fuel sloshing. A series of tests was made therefore with varying amounts of fuel on board. These tests showed that the fuel had little effect on the damping of the short period oscillation.

Another difficulty which has limited the Mach number at which airplanes are flown has been buffeting. The buffet boundary and the limit lift for the XS-1 are shown in figure 6 as a function of Mach number. These data were obtained in level flight or in gradual turns with the stabilizer set at an incidence angle of 2.2°. Limit lift has been determined from measurements where lift ceased to increase although increasing up elevator was being applied. Although buffeting has been experienced in level flight, it has not been disconcerting to the pilot because the buffeting is not severe. The maximum buffeting tail loads were obtained at limit lift from a Mach number of 0.76 to a Mach number of 0.80 and were of the order of ±400 pounds. At Mach numbers above 0.80 the buffeting tail loads decreased, and up to a Mach number of 0.92 the buffeting tail loads were less than ±250 pounds.

CONCLUSIONS

The data obtained for the XS-1 airplane show that most of the difficulties expected in the transonic range have been experienced, and although conditions are not normal, the airplane can be flown satisfactorily at least to a Mach number of 0.92. The following results have been noted in detail:

1. The airplane has experienced longitudinal trim changes in the speed range from a Mach number of 0.8 to a Mach number of 0.92, but the control forces associated with these trim changes have been small. The pilot has been able, therefore, to control the airplane.

2. The elevator effectiveness has decreased by more than 50 percent in going from a Mach number of 0.7 to a Mach number of 0.87.

This loss in elevator effectiveness has affected the magnitude of the trim changes, but the actual trim changes have been caused by changes in the wing-fuselage moment.

3. No aileron buzz or associated phenomena has been experienced up to a Mach number of 0.92. The airplane becomes right wing heavy but can be trimmed with aileron.

4. Buffeting has been experienced in level flight but has been very mild up to a Mach number of 0.92. The tail loads associated with the buffeting have been small.

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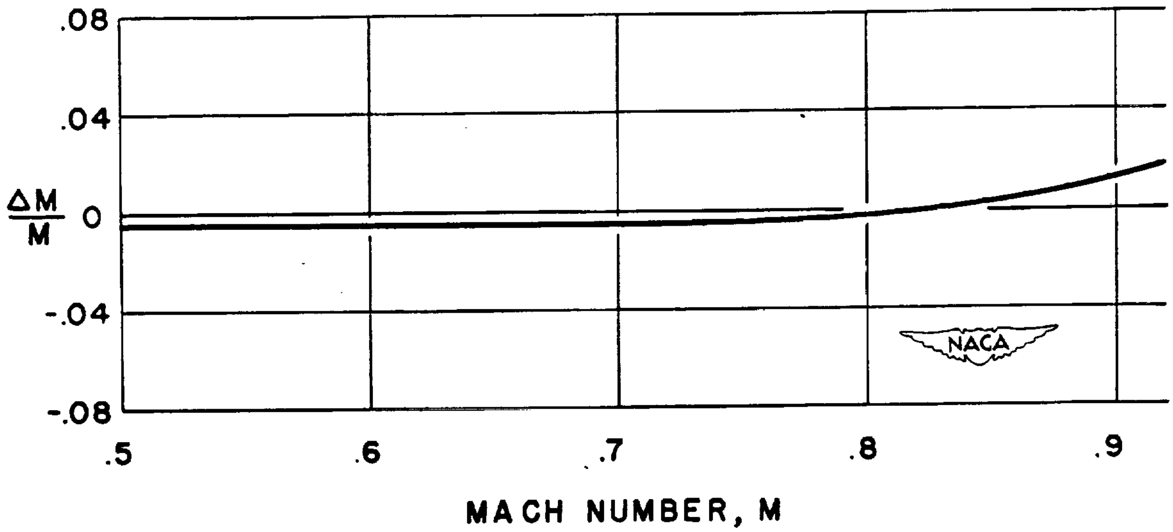

Figure 1.- Variation of Mach number error $\Delta M/M$ with corrected Mach number. XS-1 airplane.

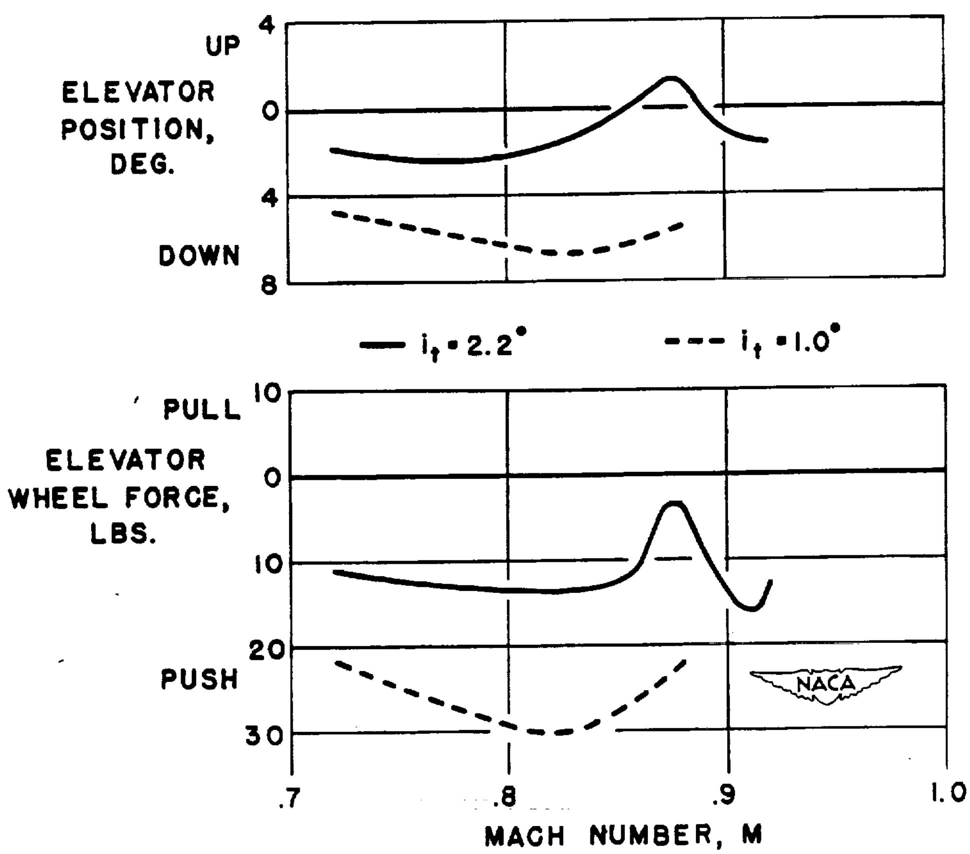

Figure 2.- Variation of elevator position and force with Mach number. XS-1 airplane.

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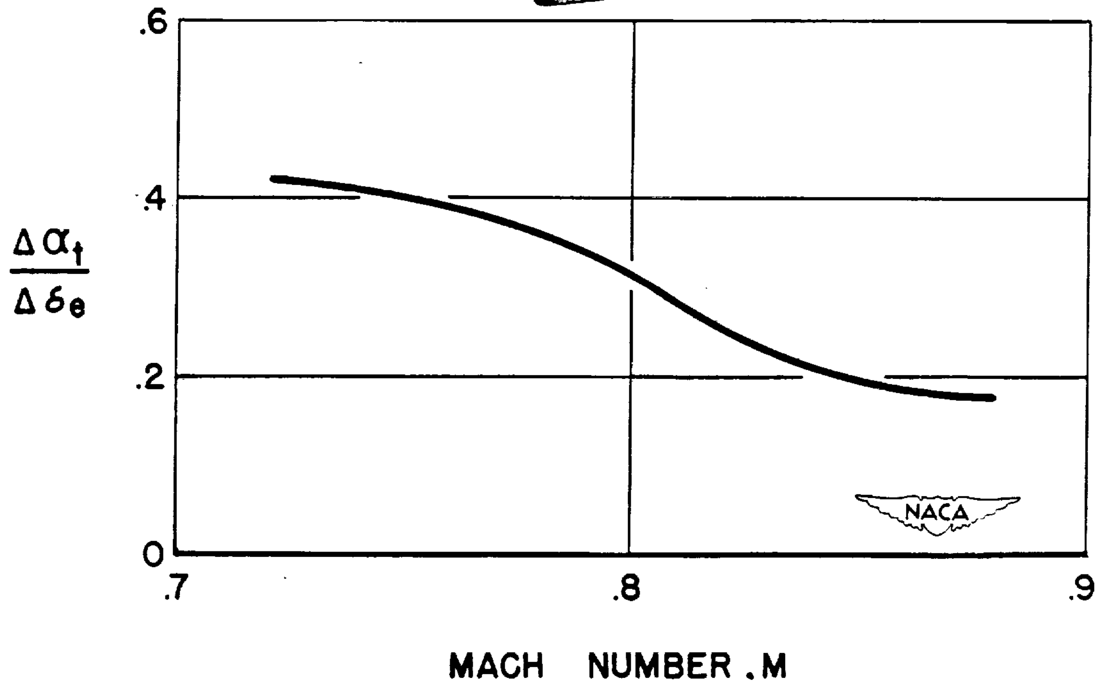

Figure 3.- Variation of elevator effectiveness factor $\Delta\alpha_t/\Delta\delta_e$ with Mach number.

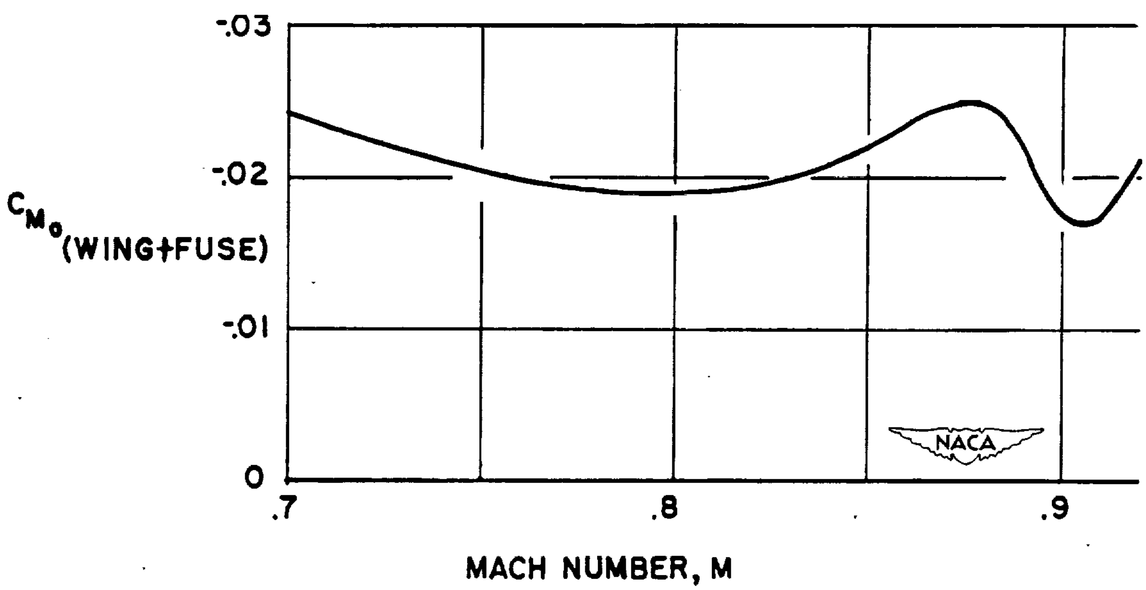

Figure 4.- Variation of wing-fuselage pitching-moment coefficient with Mach number. XS-1 airplane.

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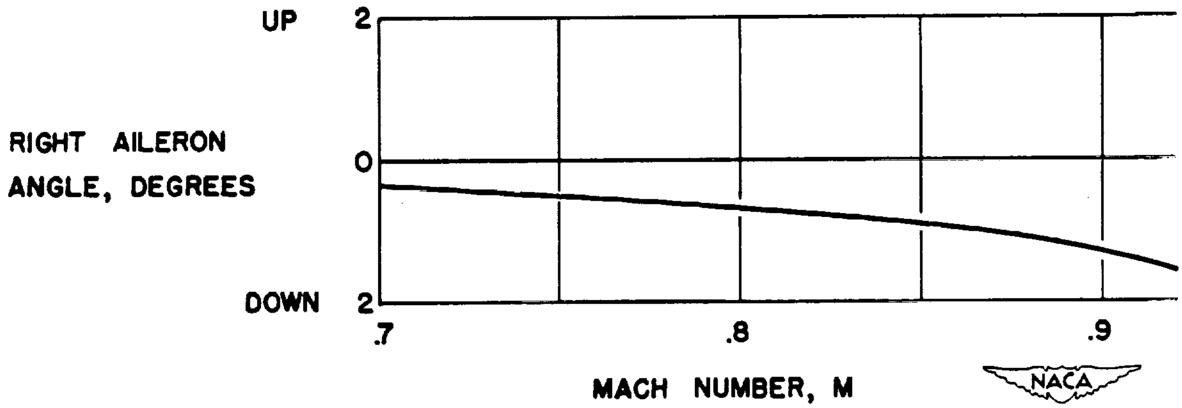

Figure 5.- Variation of right aileron angle with Mach number. XS-1 airplane.

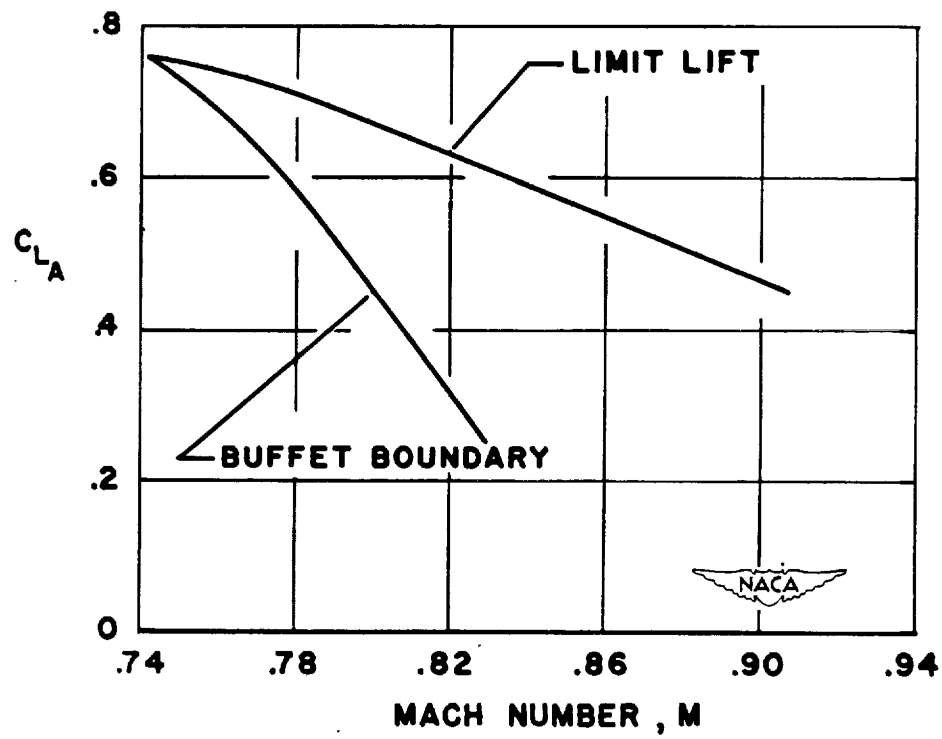

Figure 6.- Buffet boundary and limit lift. XS-1 airplane.

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