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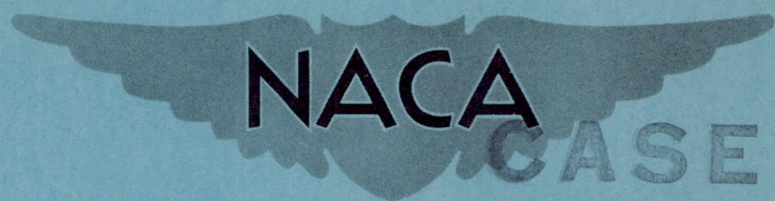
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RESEARCH MEMORANDUM

FLIGHT INVESTIGATION OF THE EFFECT OF BOUNDARY-
LAYER SUCTION ON PROFILE-DRAG COEFFICIENT
AT SUPERCRITICAL MACH NUMBERS

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

WASHINGTON
September 20, 1949

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RESEARCH MEMORANDUM

FLIGHT INVESTIGATION OF THE EFFECT OF BOUNDARY—

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SUMMARY

Flight tests were conducted with a fighter airplane to study the effect of boundary—layer suction aft of the shock wave on airfoil drag at supercritical Mach numbers and high Reynolds numbers. A suction slot was placed at about 70—percent chord, approximately 7—percent chord aft of the shock location at the highest test Mach number. Airfoil chord force was determined from pressure—distribution measurements obtained at Mach numbers of 0.70 to 0.83 in steady dives. Wake survey measurements were also made but over the lesser Mach number range from 0.70 to 0.78. The approximate Mach number for drag divergence was 0.73.

Results of the tests showed no measurable effect of suction for the suction coefficient available. Even under conditions where flow separation was present the drag increase with Mach number was due primarily to pressure changes associated with supersonic flow on upper and lower surfaces which resulted in increased pressure drag.

INTRODUCTION

As has been known for some time, an abrupt rise in the drag coefficient of an airfoil occurs at high subsonic Mach numbers due to flow changes associated with the occurrence of local regions of supersonic flow near the airfoil. One of these flow changes is the boundary—layer growth or separation accompanying the shock formation which terminates a supersonic region. This boundary—layer behavior develops because of the steep adverse pressure gradient at the base of the shock. In view of this action of the boundary layer, there has been renewed interest in the possibilities of boundary—layer control

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as a means of reducing airplane drag at high Mach numbers, especially those Mach numbers at which separation occurs behind the shock. As far as is known, research to date on boundary-layer control at high Mach numbers has been confined to tests on very small models. References 1 and 2 present the results of tests where suction control was applied behind the shock on 2-inch-chord airfoils yielding a drag reduction of the order of 50 percent.

The purpose of the present investigation was to study at large scale the effect of boundary-layer suction on airfoil drag at super-critical Mach numbers and to determine whether the drag increase due to separation could be reduced. Accordingly, an airplane was fitted with a suction slot on the upper surface of the left wing at about 70-percent chord. Measurements were made of profile drag (by the wake survey method) and of pressure distribution (to evaluate chordwise force) at Mach numbers beyond that of drag divergence to determine the effect at these speeds of boundary-layer removal on drag.

TEST EQUIPMENT

The tests of boundary-layer control reported herein were carried out on a portion of the left wing of a jet-propelled fighter airplane. A picture of the airplane as instrumented for the tests is shown in figure 1.

The slot configuration used in the tests is shown in figures 2 and 3. As can be seen from the figures, the slot was located at about 70-percent chord. The slot was 10.3 percent (24 in.) of the wing semi-span and 2.52 percent (2 in.) of the local wing chord and was located at about 42 percent (8 ft, 1-1/4 in.) of the wing semispan from the fuselage center line. Air flow was induced through the slot by the low pressure existing at the duct exit. (See figs. 2 and 4.) A wing root bump was installed to increase the pressure difference between the slot entrance and the exit. For a limited series of tests the slot length was reduced by one-half in an effort to increase the flow coefficient obtainable.

Standard NACA recording instruments synchronized by a standard NACA timer were used to record the following variables: indicated airspeed, pressure altitude, normal acceleration, wake-survey total-head decrement and static pressures, boundary-layer total head and static pressures, and the amount of suction air flow. All recording instruments were installed in the nose compartment except the accelerometer which was installed in the pilot's cockpit. The air temperature

used in the Reynolds number calculations was obtained from radiosonde
data. In addition, a 16-millimeter gunsight aiming-point camera was
installed in the canopy to photograph tuft action on the test panel.

For the recording airspeed system a freely swiveling airspeed
head was mounted on the end of an airspeed boom attached to the left
wing tip and extending two chord lengths ahead of the wing leading
edge, as shown in figures 1 and 2. The airspeed calibration error for
the installation was almost negligible, the maximum correction to the
measured Mach number throughout the test range being 0.01.

The profile-drag rake (shown in fig. 4) was mounted in line with
the center line of the test section on the end of a cantilevered strut
extending outward from the fuselage as shown in that figure. The rake
contained 54 total head tubes and 6 static tubes with the tube openings
located 14.2-percent chord (11-1/4 in.) aft of the wing trailing edge.

TESTS AND RESULTS

The pressure-distribution and wake-survey measurements were
obtained in steady dives of substantially linear flight path. The
wake surveys were conducted at a pressure altitude of approximately
15,000 feet over a Mach number range from 0.70 to 0.78 (airplane lift
coefficient varied from 0.12 to 0.08). (Maximum speed was limited by
rake vibration and expansion of the wake, which exceeded the limited
extent of the rake at Mach numbers above 0.78.) The pressure-distribution
tests were conducted at an approximate pressure altitude of 30,000 feet
over a Mach number range from 0.70 to 0.83 (airplane lift coefficient
varied from 0.21 to 0.12). The Reynolds number – Mach number relation
for both sets of tests is shown in figure 5. As can be seen from the
figure, the Reynolds number range for the higher altitude was from
13×10^6 to 16×10^6 and for the lower altitude was from 20×10^6 to
23×10^6 based on the test section mean chord.

A study of the flow conditions on the test panel upper surface
was made with the following results:

1. From the boundary-layer surveys, transition was found to occur
at about 20-percent chord due to surface waviness.

2. Inspection of the pressure distributions indicated that the
shock location varied from about 52- to 63-percent chord. Hence the
shock was always forward of the slot by at least 7-percent chord.

3. The tuft study indicated that a pronounced cross flow over the

test panel developed progressively above a Mach number of 0.76.

4. The measurements gave no indication of boundary-layer separation until about 0.79 Mach number, although the Mach number for drag divergence is about 0.73.

The suction coefficient[1] attained with the full-length slot was constant over the Mach number range and equal to about 0.00175. This value of suction coefficient corresponds to removal of about one-quarter of the air in the boundary layer at the slot location at the lower Mach numbers and is about one-third of the suction coefficient it had been hoped to attain at the beginning of the tests. More suction could not be attained due to the limited pressure drop available and the high duct losses present. In an effort to increase the percentage of boundary-layer air removed, the slot length was reduced from each end by 25 percent (reducing the total length to 1 ft). This change resulted, approximately, in a doubling of the suction coefficient due to the 50-percent reduction in affected area, since the volume rate of flow remained about the same. No further attempt to increase the flow rate by additional mechanical means was made in view of the conclusions drawn from a study of the pressure-distribution results presented hereinafter.

Although the wake-survey measurements were made to 0.78 Mach number, the results were questionable above a Mach number of 0.76 due to the probable invalidating influence of the boundary-layer cross flow on the boundary layer and wake measurements above that Mach number. Accordingly, since the wake measurements were considered unreliable and since the frictional drag becomes an increasingly smaller percentage of the total drag at supercritical Mach numbers, it was decided to study the effect of suction on the pressure drag as determined from the pressure-distribution measurements. It was realized that a rigorous study of pressure drag would involve the measurement of angle of attack α and the determination of the drag coefficient C_d by the equation

$$C_d = C_c \cos \alpha + C_n \sin \alpha$$

where C_c and C_n are the chordwise- and normal-force coefficients. In view of the difficulties of measuring angle of attack in flight, however, the chordwise-force coefficient was used to study drag changes due to suction, since at the low angles of attack which existed for these tests the magnitude of the $C_n \sin \alpha$ term was small. This

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[1] Suction coefficient is defined as Q/VA where Q is the volume of air removed under the test conditions of temperature and pressure, V is the true airspeed of the airplane, and A is the total wing area ahead and behind the slot.

term was estimated to be roughly 6 percent of the $C_c \cos \alpha$ term at a Mach number of 0.78 for the conditions of these tests. Actually, even if the absolute magnitude of C_d were not closely represented by C_c for these tests, the change in C_d with Mach number would still be given very closely by changes in C_c, since the tests were conducted at an essentially constant angle of attack near zero α. (The angle-of-attack deviation from the mean value over the test range was only $\pm 0.6^\circ$ as estimated from the measured values of lift coefficient.) The estimated change in $C_n \sin \alpha$ due to Mach number (applying Glauert factor to average test C_n) over the test range is of insignificant magnitude compared to the change in $C_c \cos \alpha$.

Curves of chordwise force coefficient and profile drag plotted against Mach number are presented in figure 6 for the original slot to show the suction effect. In addition, data for the reduced length slot is shown in figure 6(a). The chordwise force coefficients were obtained from integration of thickness-wise pressure distributions. The profile drag coefficients were obtained from the wake-survey data using the method of reference 3. The momentum loss of the removed boundary-layer air (determined from a boundary-layer survey at the slot entrance) was added to that measured by the survey rake[2] in order to obtain the total section profile drag[3]. As can be seen from the figure, any reduction in drag or chordwise force due to boundary-layer removal is within experimental accuracy and would appear to be negligible.

DISCUSSION

A qualitative idea of the causes for the drag rise (and the failure of the suction to modify this rise) can be obtained from examination of figures 7 and 8 which present representative chordwise and thickness-wise pressure distributions for selected test Mach numbers throughout the test range with and without suction. The chordwise distributions show that the critical pressure coefficient (for local Mach number equal 1.0) was exceeded on the upper surface over the entire range of these tests. The lower-surface pressures, however, did not exceed the critical until a Mach number of about 0.76 was attained. The shock apparently first formed on the upper surface near the 50-percent-chord station,

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[2] Such an evaluation of the drag with suction applied neglects the drag equivalent of the suction power (assumes 100-percent duct efficiency).

[3] Comparison should be made only to a Mach number of 0.76 because of boundary-layer cross flow. Points affected by cross flow are indicated by the use of flags in figure 6(b).

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which is 10 percent behind the point of maximum thickness. The upper-surface shock then moved back with increase in Mach number reaching about 59-percent chord at 0.76 Mach number and about 63-percent chord at 0.83 Mach number. The lower-surface shock apparently forms near the maximum thickness point initially and then moves back much more abruptly than the upper-surface shock, arriving at about 71-percent chord at 0.83 Mach number. As can be seen from the figure, the wing-surface pressure rise became more abrupt as the shock moved rearward. Examination of the trailing-edge pressures for the various distributions presented shows separation to have begun between a Mach number of 0.78 and a Mach number of 0.80 and to be present at higher Mach numbers, separation being indicated by the degree of trailing-edge pressure recovery. The meaning of these changes in terms of changes in airfoil pressure drag can be seen by referring to the thickness-wise pressure distributions also presented in the figures.

The thickness-wise pressure distributions show the contribution of the various regions along the airfoil contour to the chordwise pressure force.[4] The area between the forebody and afterbody pressure distributions has been crosshatched to show whether the resultant differential force at any given vertical ordinate is a thrust or a drag force. Integration of the areas so enclosed yields the contribution to chordwise force coefficient. The large drag area roughly centered about the line of zero thickness and with its centroid to the left of the line of zero pressure coefficient is due largely to the basic pressure distribution of the airfoil (unmodified by any boundary-layer or separation effects). It should be noted that most of this area would exist at low Mach numbers as well as the Mach numbers of these tests. The primary effect of flow separation at the Mach number of 0.83 is to increase the area above the low-speed size by making the trailing-edge pressures on the upper surface more negative. The drag areas near the points of maximum thickness and with centroids to the right of the line of critical pressure coefficient are due to the occurrence of supersonic flow and attendant rearward shock movement. These drag areas are due to the combined effect of pressure decrease ahead of the shock associated with the change from subsonic to supersonic flow and the decreasing vertical ordinates back of the point of maximum thickness.

An over-all inspection of the curves just discussed shows that the drag rise to 0.76 Mach number can be traced to the rearward movement of shock on the upper surface with no increase due to separation since separation has not yet formed. The additional drag rise to 0.83 Mach

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[4] Chordwise pressure force is used for the reasons given in the text of the section on Tests and Results.

number can be traced in large part to the formation of shock and sub-
sequent rearward shock movement on the lower surface, although some
of the increase (roughly 30 percent) is due to the upper-surface sepa-
ration then present. It would appear from these results that the major
portion of the drag rise on the test airfoil comes from the formation
of supersonic regions on the upper and lower surfaces (as has also been
noted for other airfoils in references 4 and 5). Since the beneficial
effect of suction is anticipated to derive solely from the elimination
of separation, these results indicate the limited possibilities of
boundary-layer suction on the test airfoil even if fully effective in
causing reattachment of the separated wake.

CONCLUDING REMARKS

The results of a limited flight investigation to study the effect
of boundary-layer suction behind the shock wave on airfoil drag at
supercritical Mach numbers showed no measurable effect on the test air-
foil for the suction coefficient attainable. An inspection of the pres-
sure distributions revealed that the major portion of the drag increase
with Mach number was due to pressure changes directly associated with
supersonic flow on upper and lower surfaces, and a minor portion (never
exceeding 30 percent) was attributable to upper-surface separation
behind the shock wave. Thus, the possible drag reduction due to elimi-
nation of flow separation behind the upper-surface shock wave by means
of suction was limited. The extent to which this condition applies to
other airfoils or other angles of attack cannot be inferred from the
results of the subject tests.

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      National Advisory Committee for Aeronautics,
            Moffett Field, Calif.

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1. Regenscheit, B.: Drag Reduction by Suction of the Boundary Layer
       Separated Behind Shock Wave Formation at High Mach Numbers. NACA
       TM 1168, 1947.

2. Fage, A., and Sargent, R. F.: Effect on Aerofoil Drag of Boundary-
       Layer Suction Behind a Shock Wave. R. & M. No. 1913, Oct. 26,
       1943.

3. Baals, Donald D., and Mourhess, Mary J.: Numerical Evaluation of the Wake-Survey Equations for Subsonic Flow Including the Effect of Energy Addition. NACA ARR, Nov. 1945.

4. Oswatitsch, K.: The Drag Increase at High Subsonic Speeds. R.A.E. TN No. Aero 1919, Oct. 1947.

5. Davies, H.: Flight Research at High Subsonic Speeds. Jour. of the Royal Aero. Society, Aug. 1948, vol. 52, no. 452, pp. 483–512.
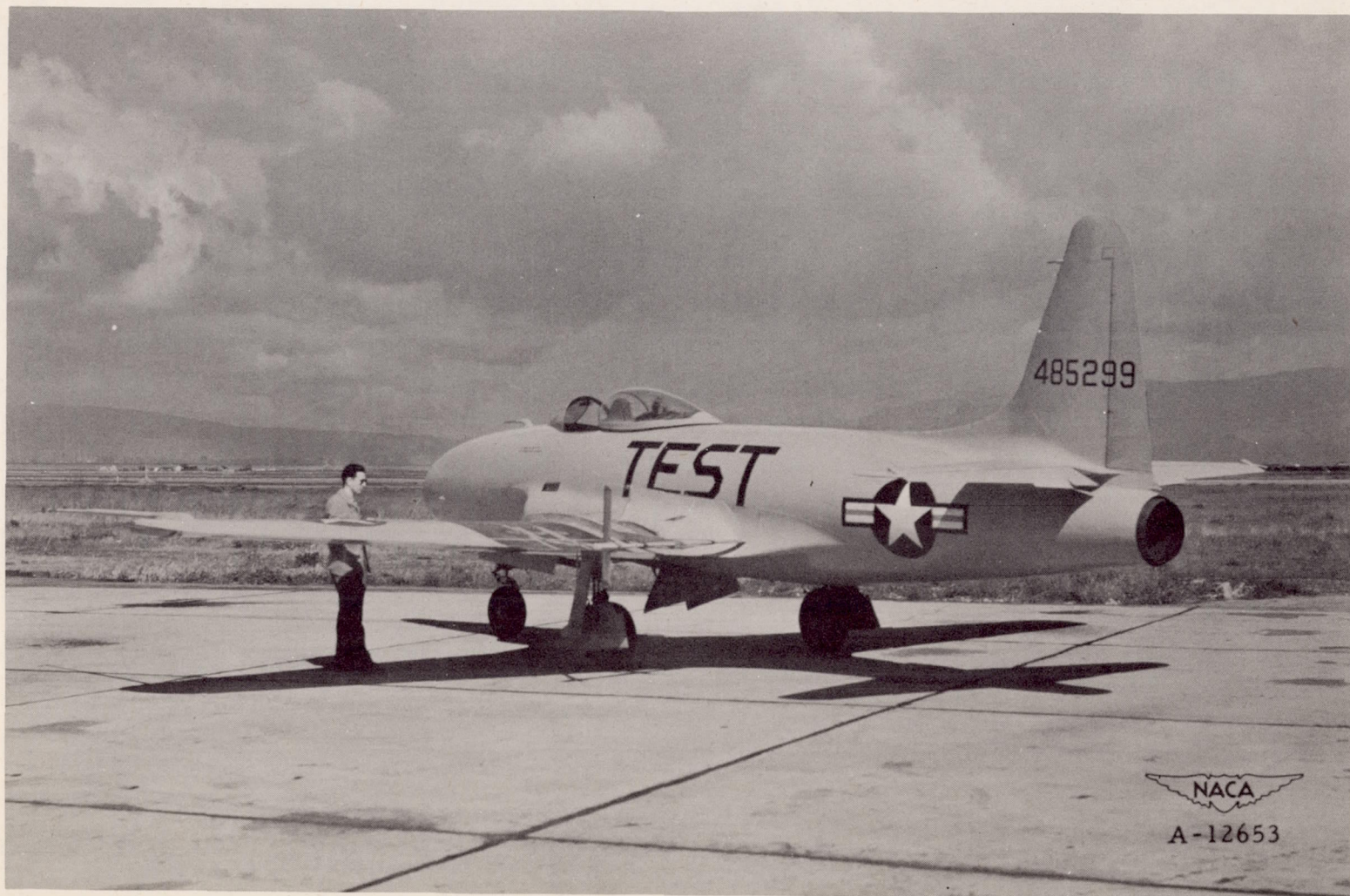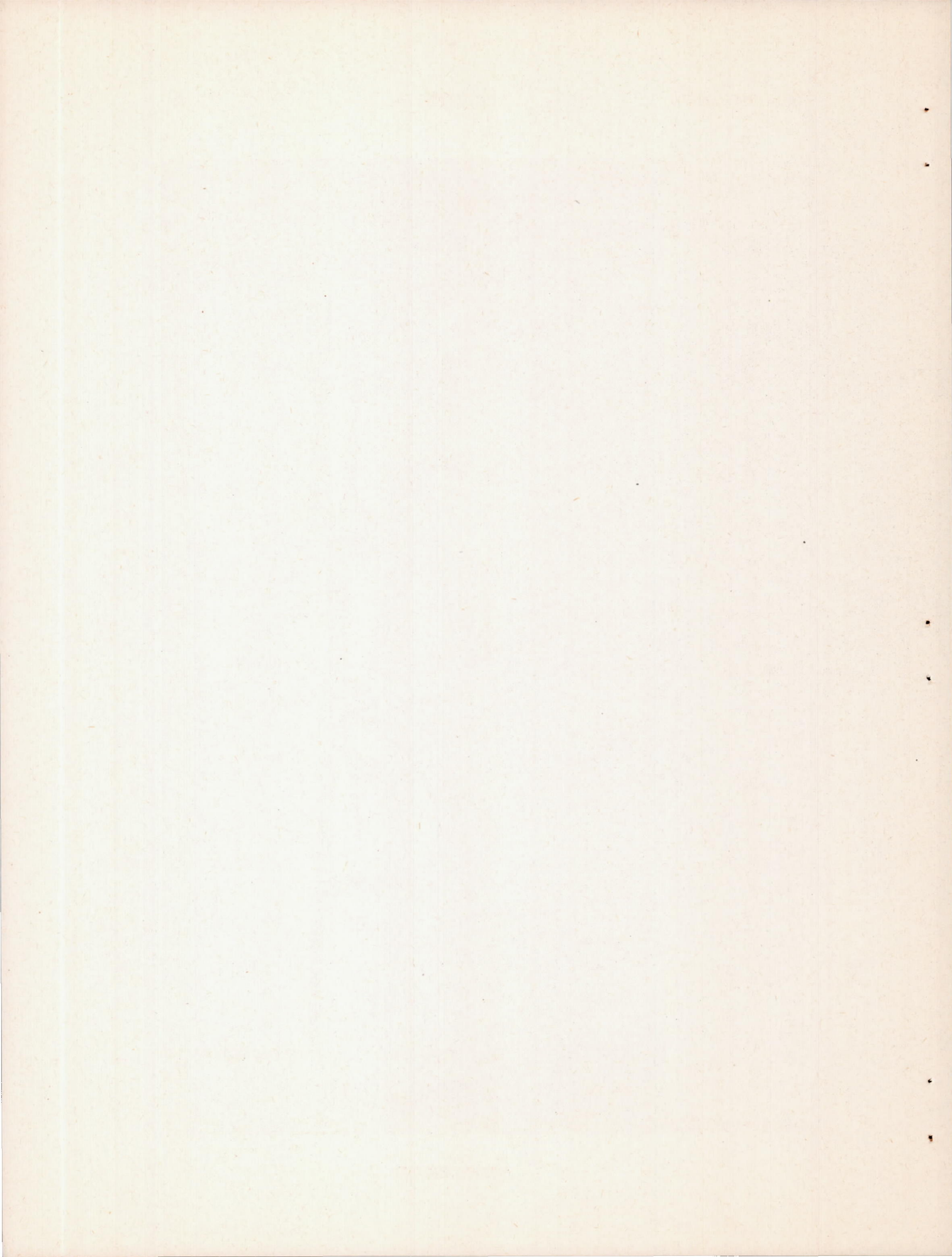
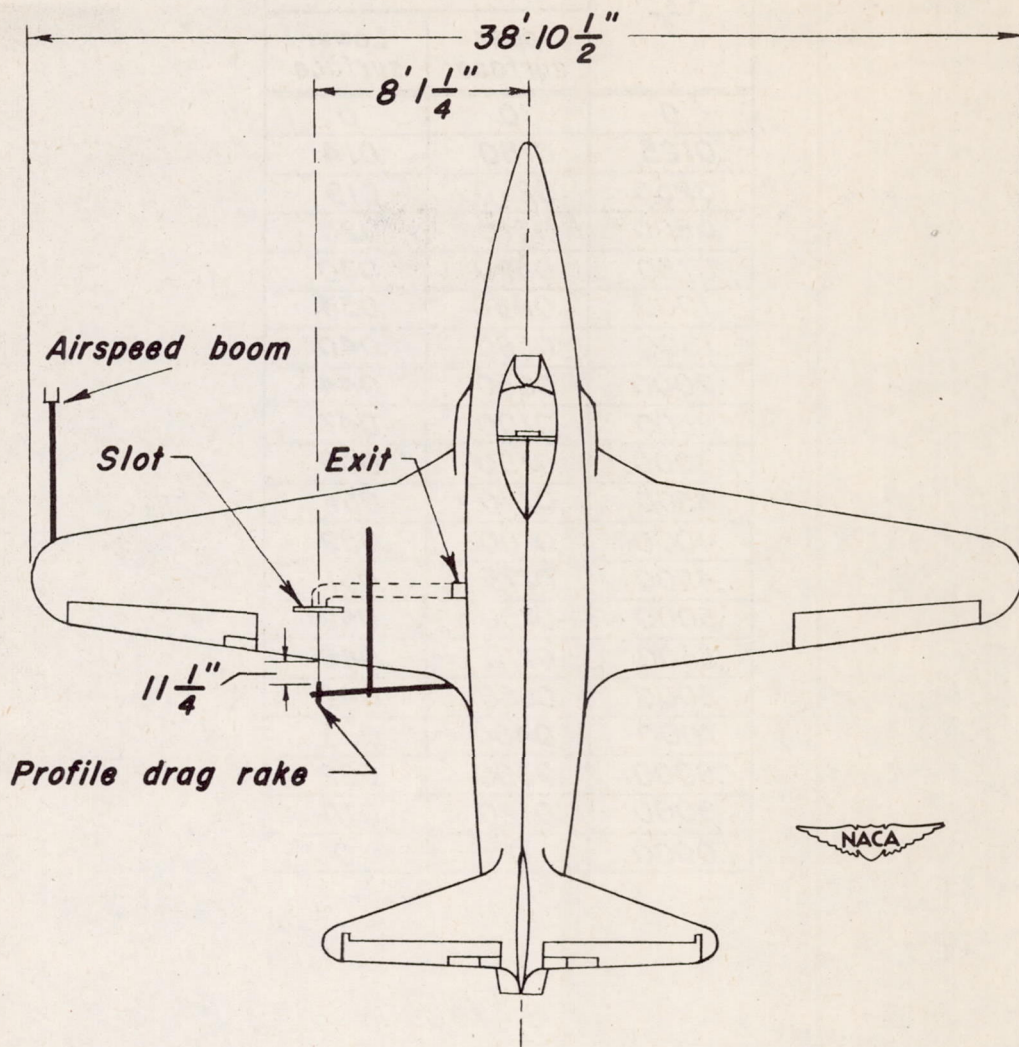Figure 1.— Test airplane as instrumented for the flight tests.

Figure 2.- Plan form of test airplane showing location of suction slot and duct exit.

$\dfrac{x}{c}$	$\dfrac{y}{c}$	
	Upper surface	Lower surface
0	0	0
.0125	.0150	.014
.0250	.0220	.019
.0500	.0310	.025
.0750	.0390	.030
.1000	.0460	.034
.1500	.0560	.040
.2000	.0650	.044
.2500	.0700	.047
.3000	.0750	.050
.3500	.0770	.051
.4000	.0780	.052
.4500	.0775	.051
.5000	.0750	.049
.5500	.0710	.046
.6000	.0660	.042
.7000	.0480	.033
.8000	.0300	.022
.9000	.0120	.010
1.0000	0	0

$c = 79.35$ in.

Suction slot
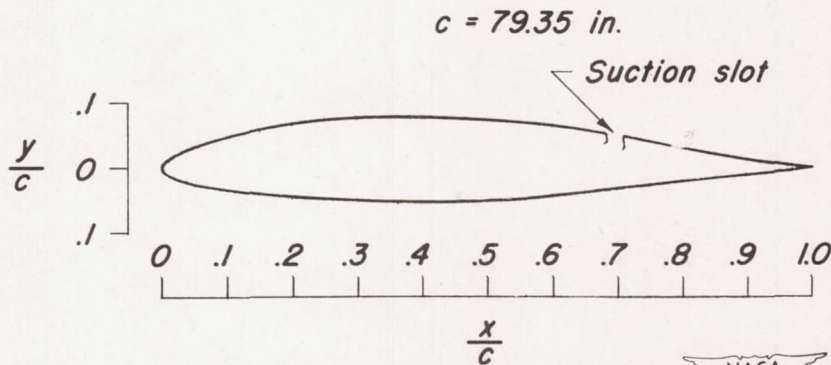
Figure 3.– Wing cross section along test panel center line showing slot location. NACA 65$_{(112)}$– 213, $a = 0.5$.
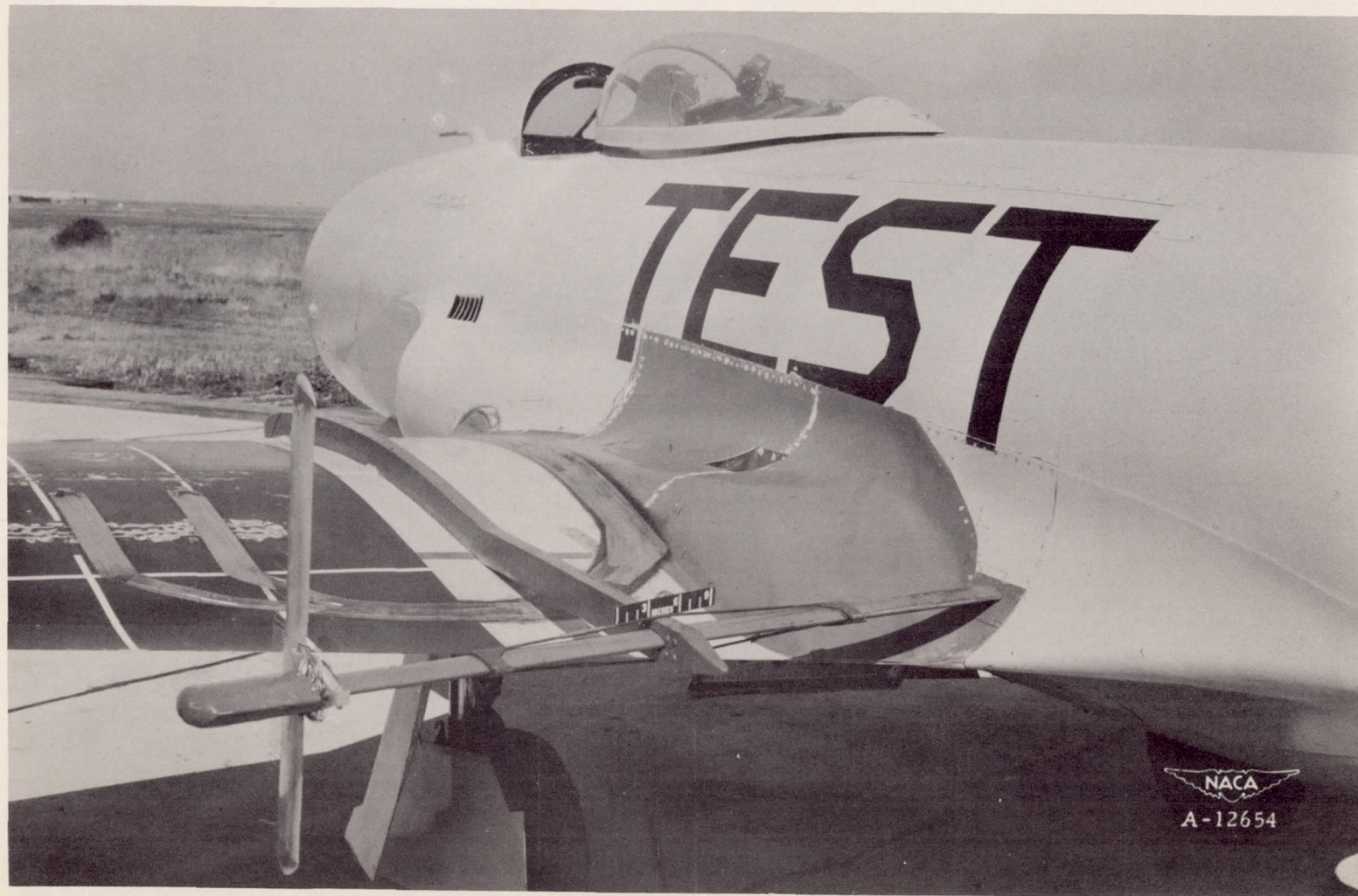
Figure 4.— Test equipment showing wing root bump installed for the tests.
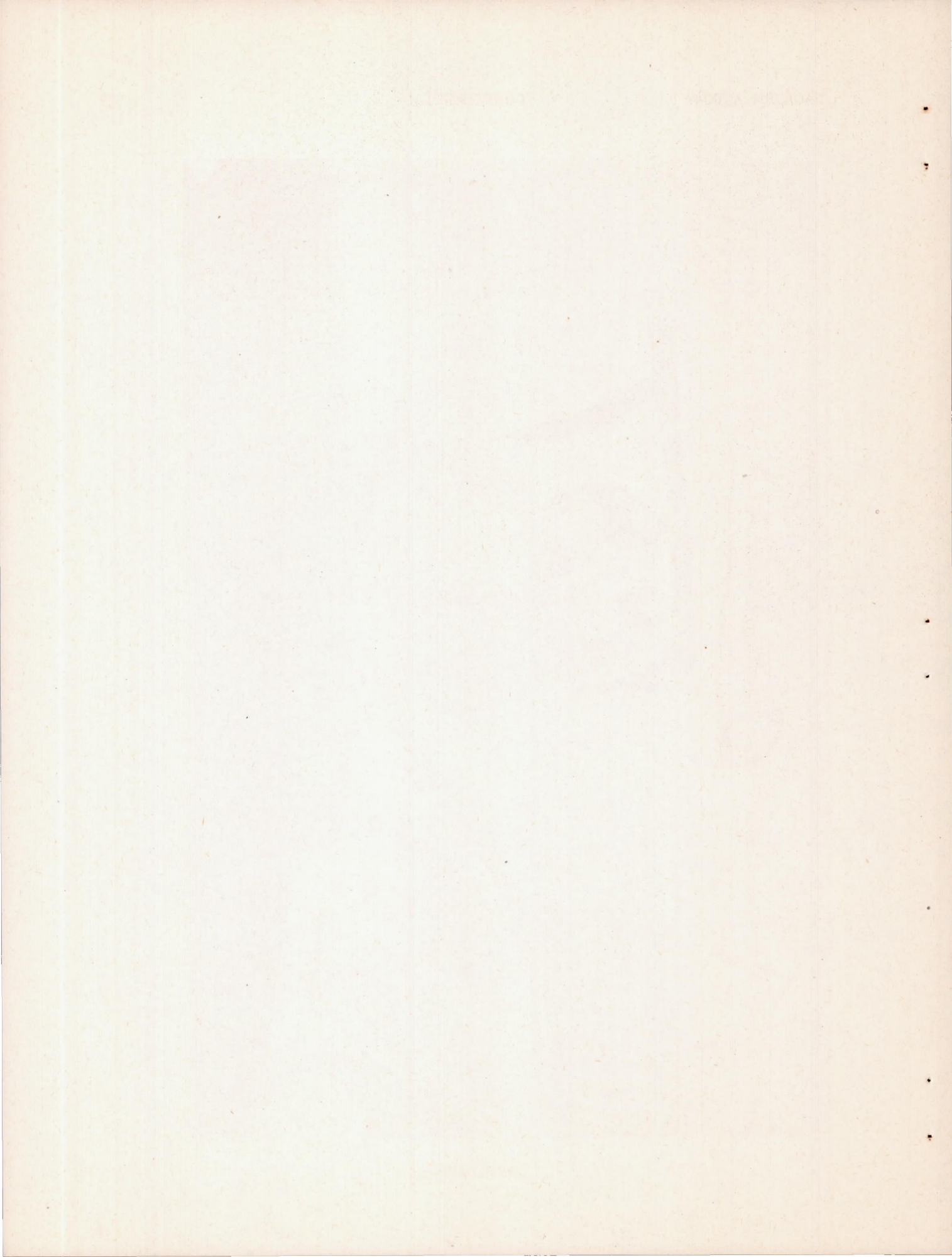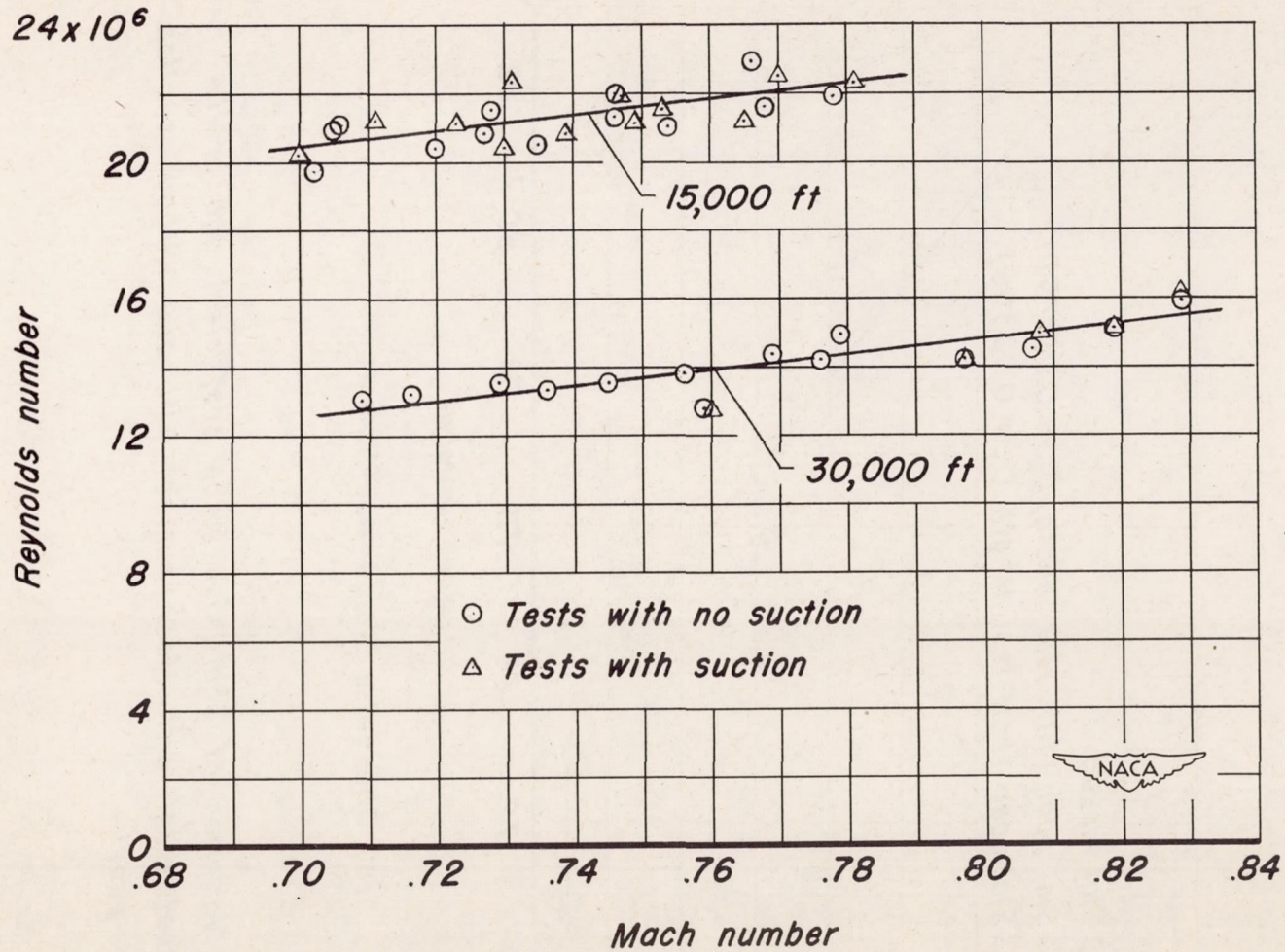
Figure 5.- Relation between Reynolds number and Mach number for the two test altitudes.
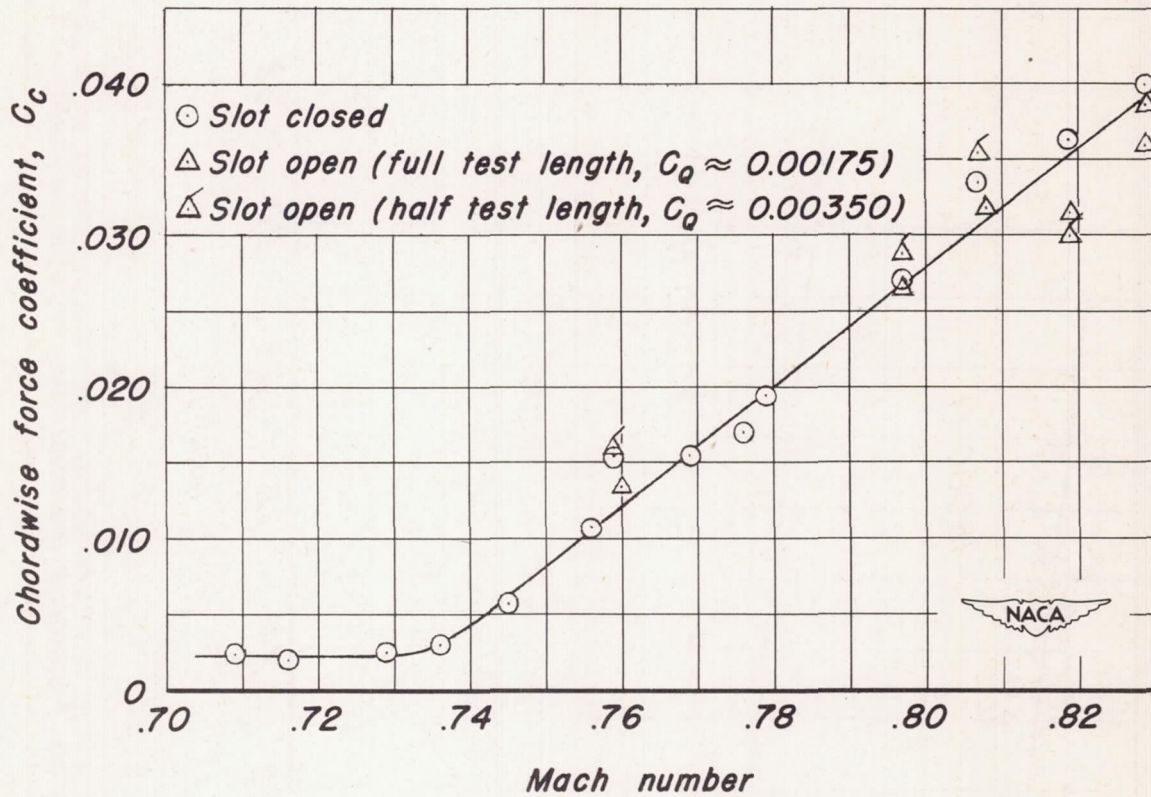
(a) Chordwise force coefficient.

Figure 6.- Variation of chordwise force coefficient and profile
    drag coefficient with Mach number with and without suction
    on test panel.

Note: Cross flow present on all flagged
points.
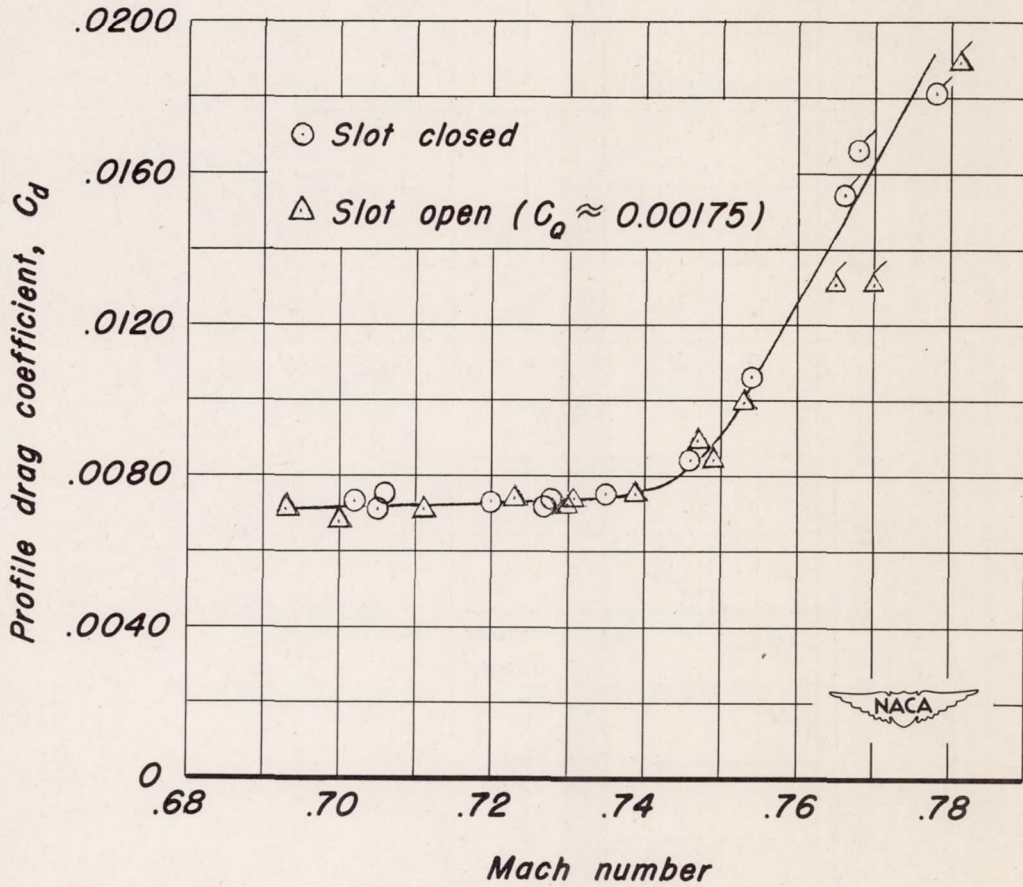
(b) Profile drag coefficient.

Figure 6.— Concluded.

(a) M = 0.70   (C_L = 0.21).
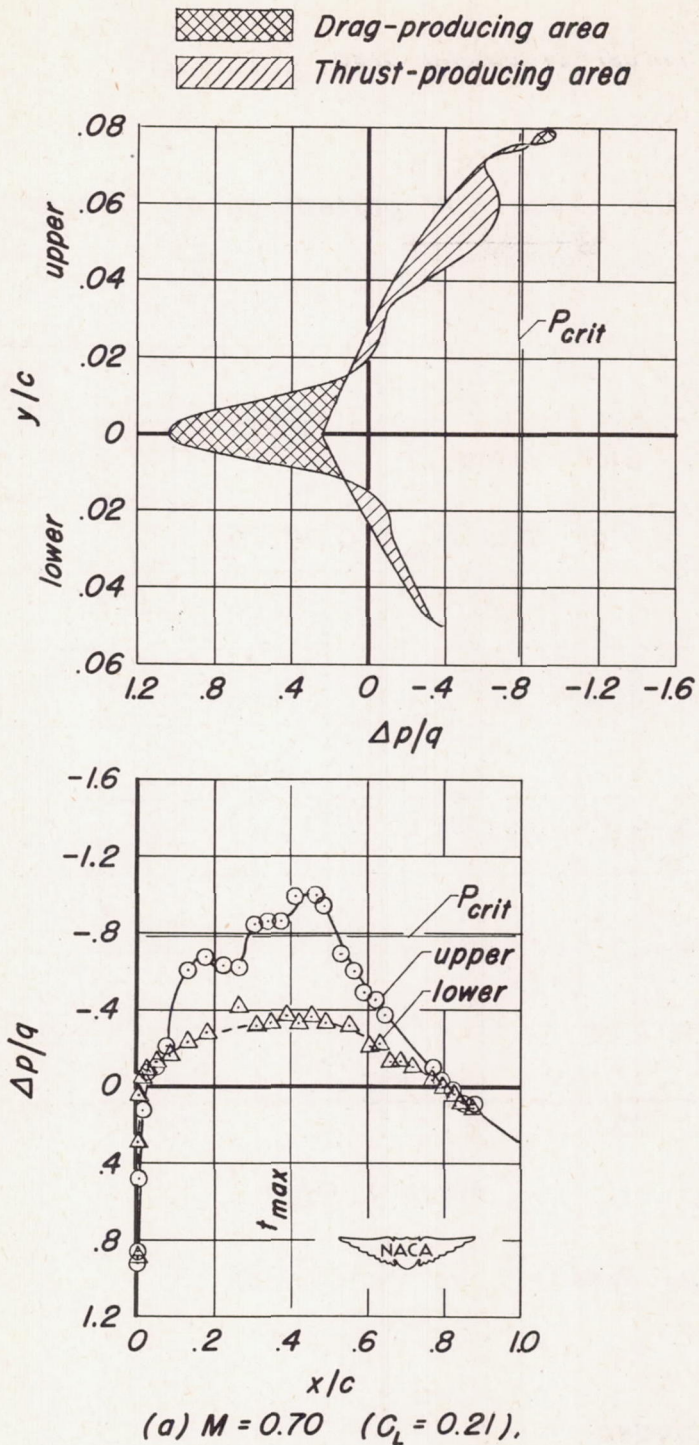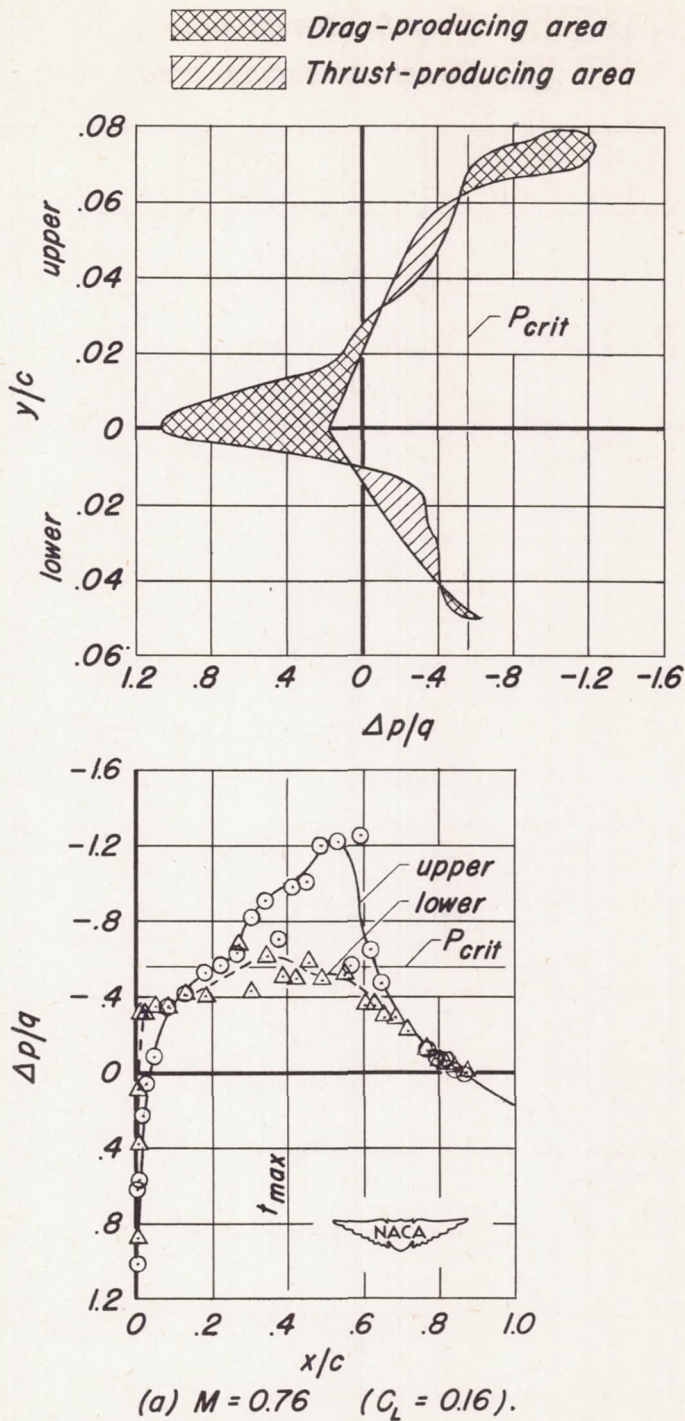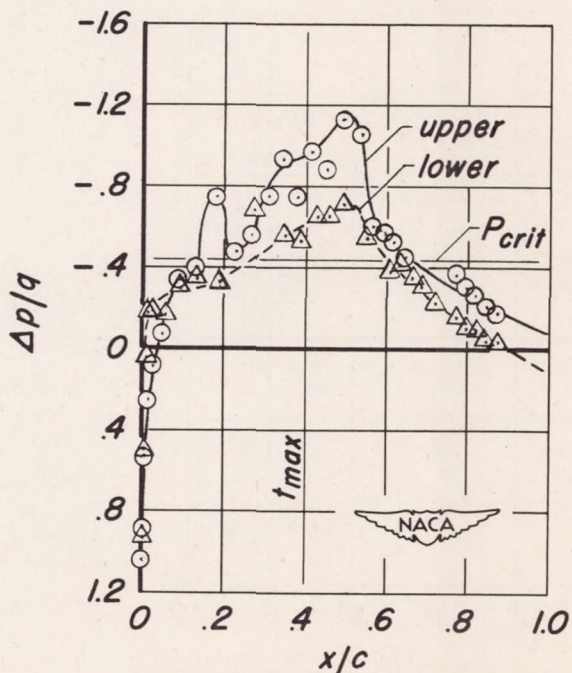
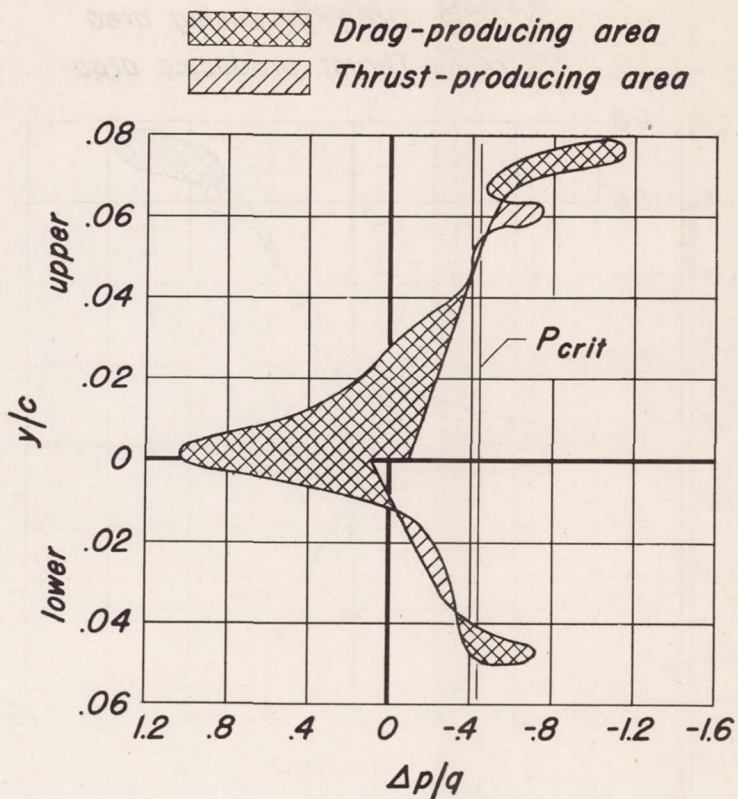Figure 7.- Curves of measured chordwise and thickness-wise pressure distributions over the test panel at selected Mach numbers in the test range. No suction.

Drag-producing area

Thrust-producing area

P_{crit}

$\Delta p/q$

upper

lower

P_{crit}

t_{max}

NACA

$\Delta p/q$

x/c

(b) $M = 0.73$ ($C_L = 0.20$).

Figure 7.– Continued.

Drag-producing area

Thrust-producing area
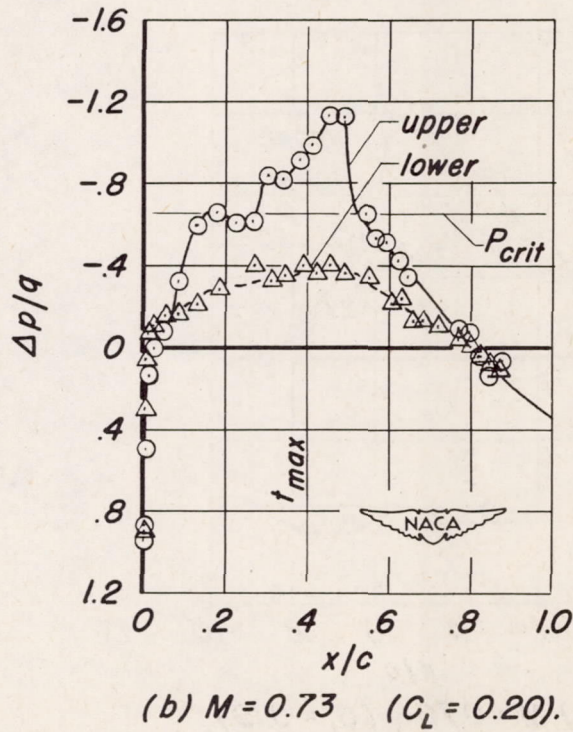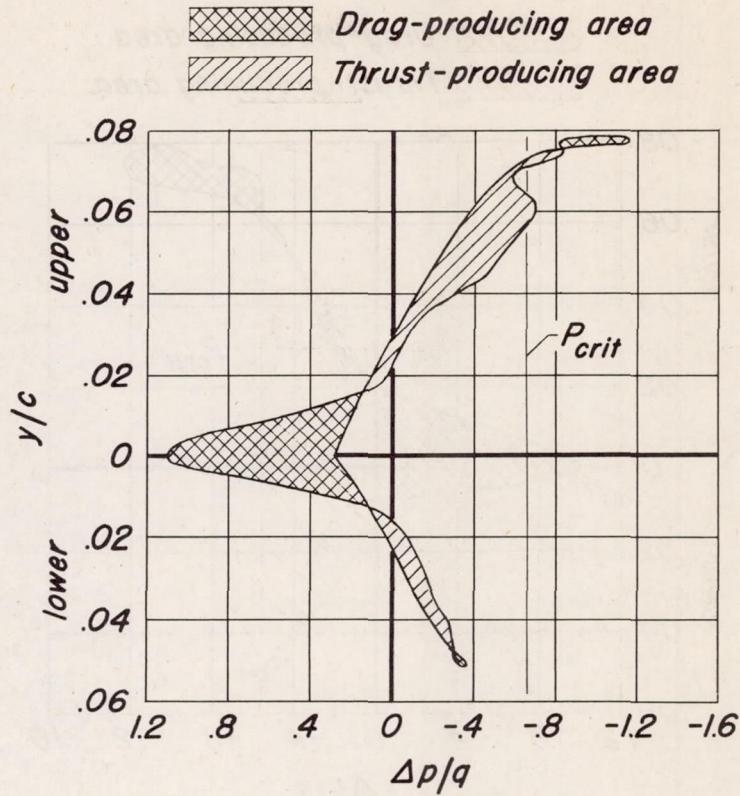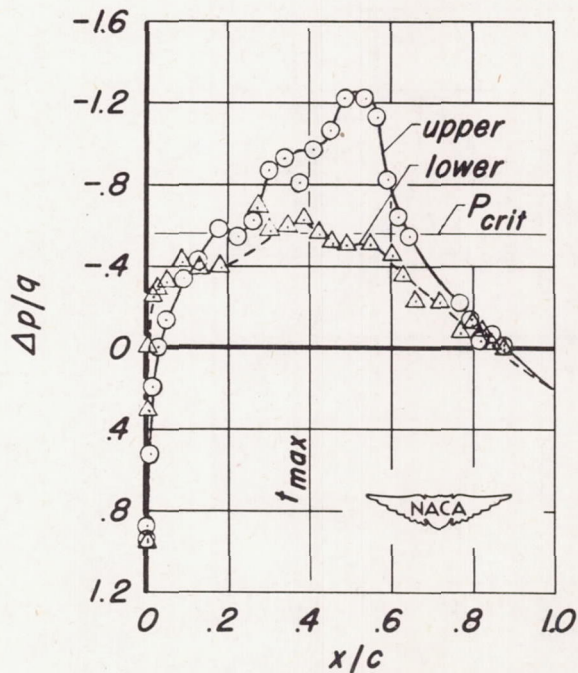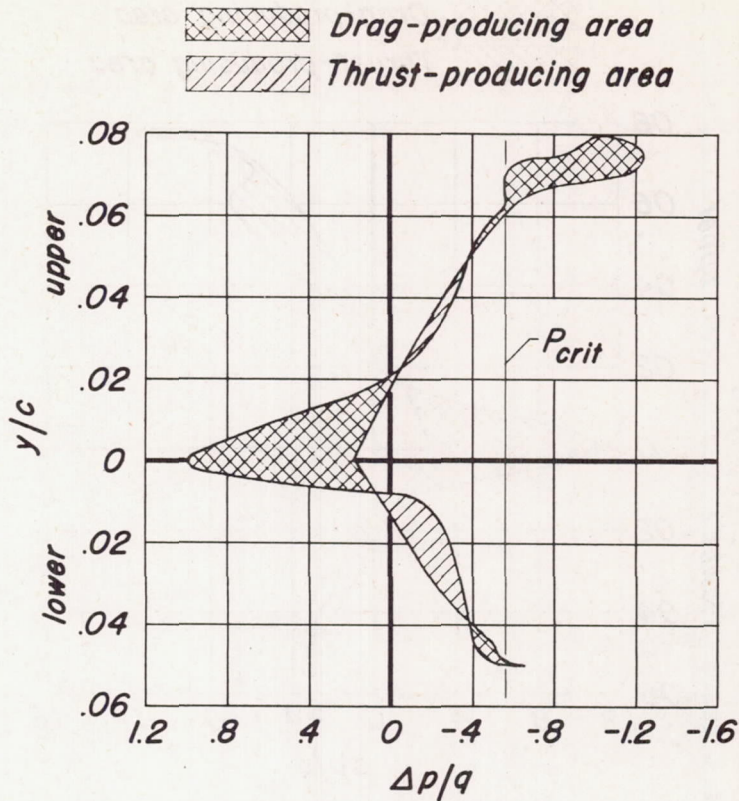
(c) M = 0.76   (C_L = 0.19).

Figure 7.-Continued.
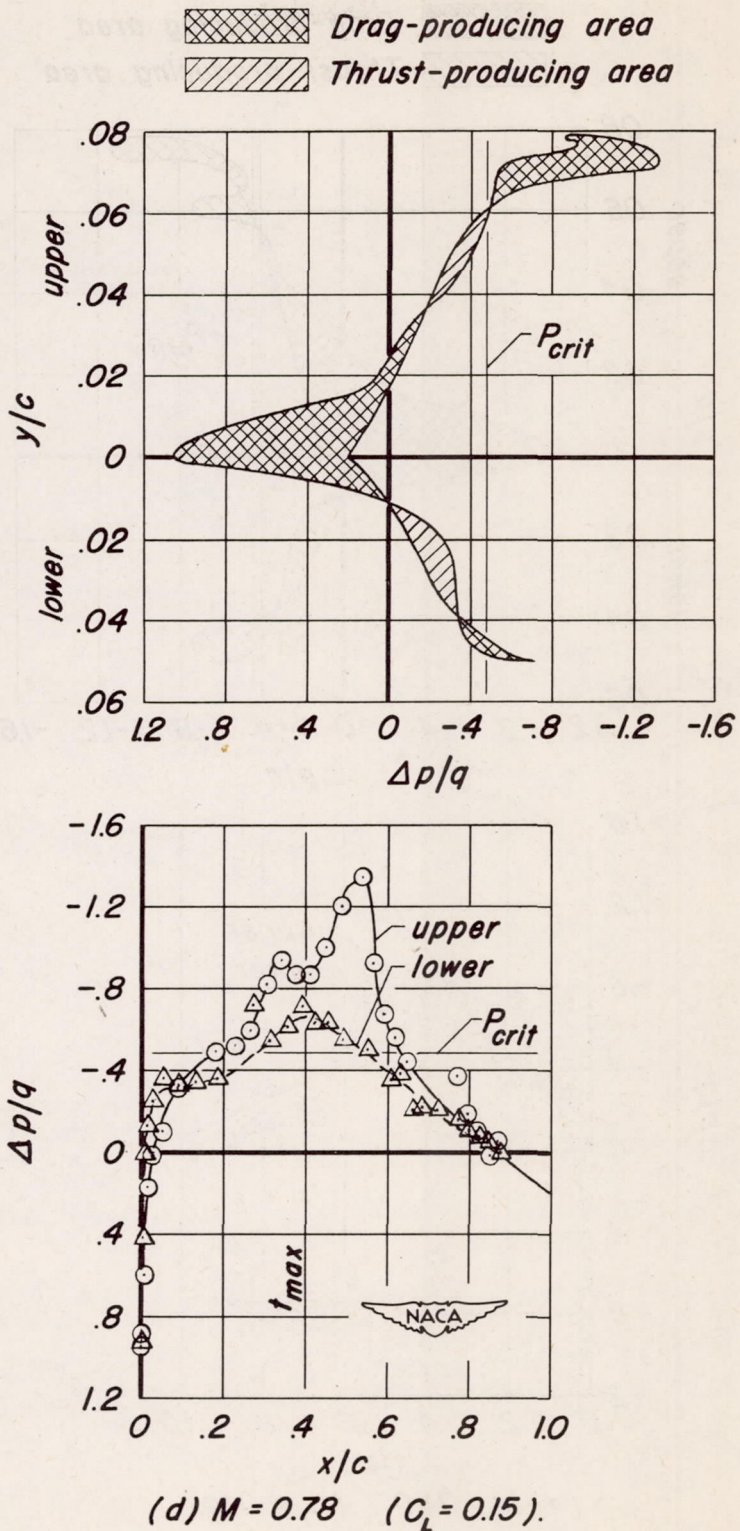
(d) M = 0.78   (C_L = 0.15).

Figure 7.- Continued.

(e) M = 0.80   (C_L = 0.14).

Figure 7.- Continued.

(f) M = 0.81   (C_L = 0.17).

Figure 7.- Continued.

Drag-producing area
Thrust-producing area

(g) M = 0.82   (C_L = 0.21).

Figure 7.- Continued.

Drag-producing area
Thrust-producing area

(h) M = 0.83   (C_L = 0.16).
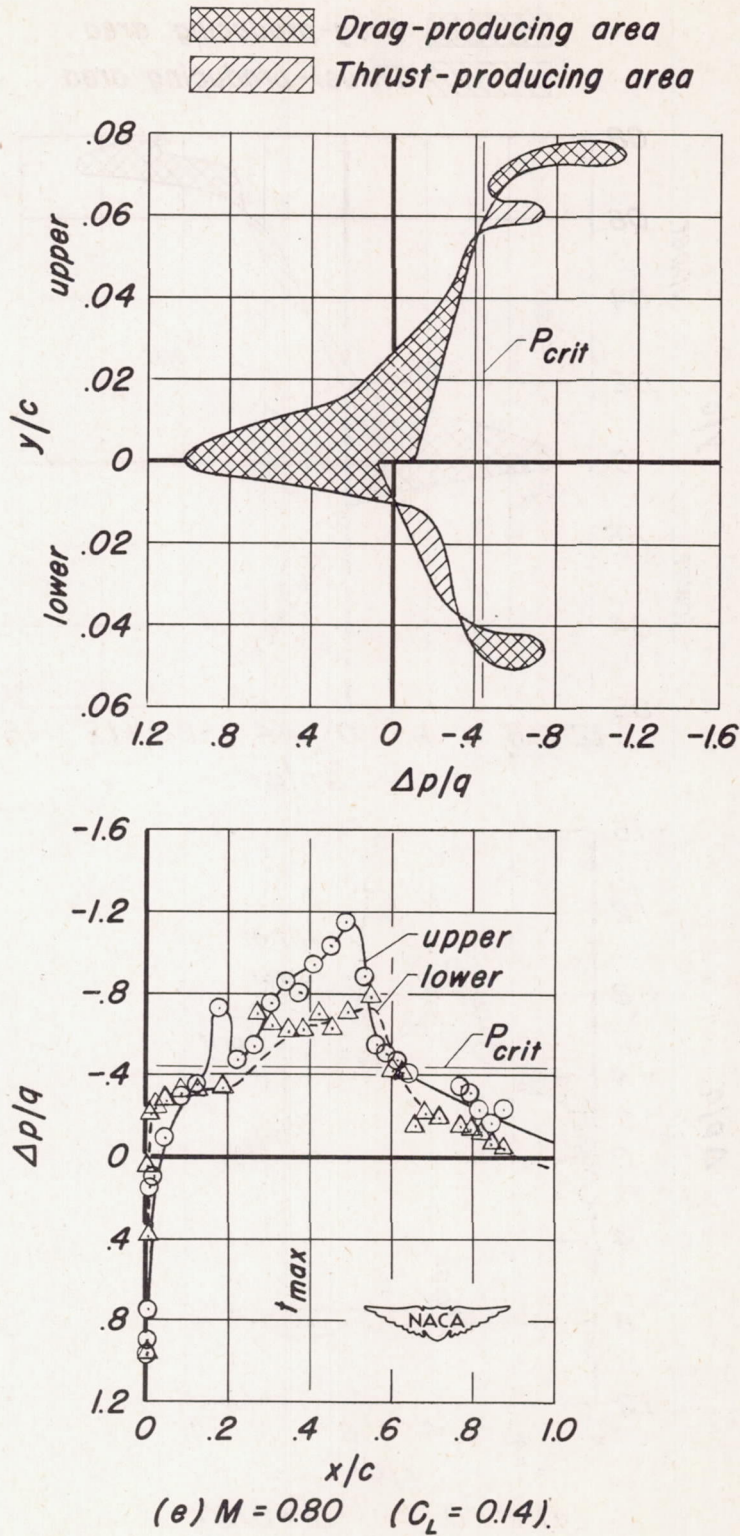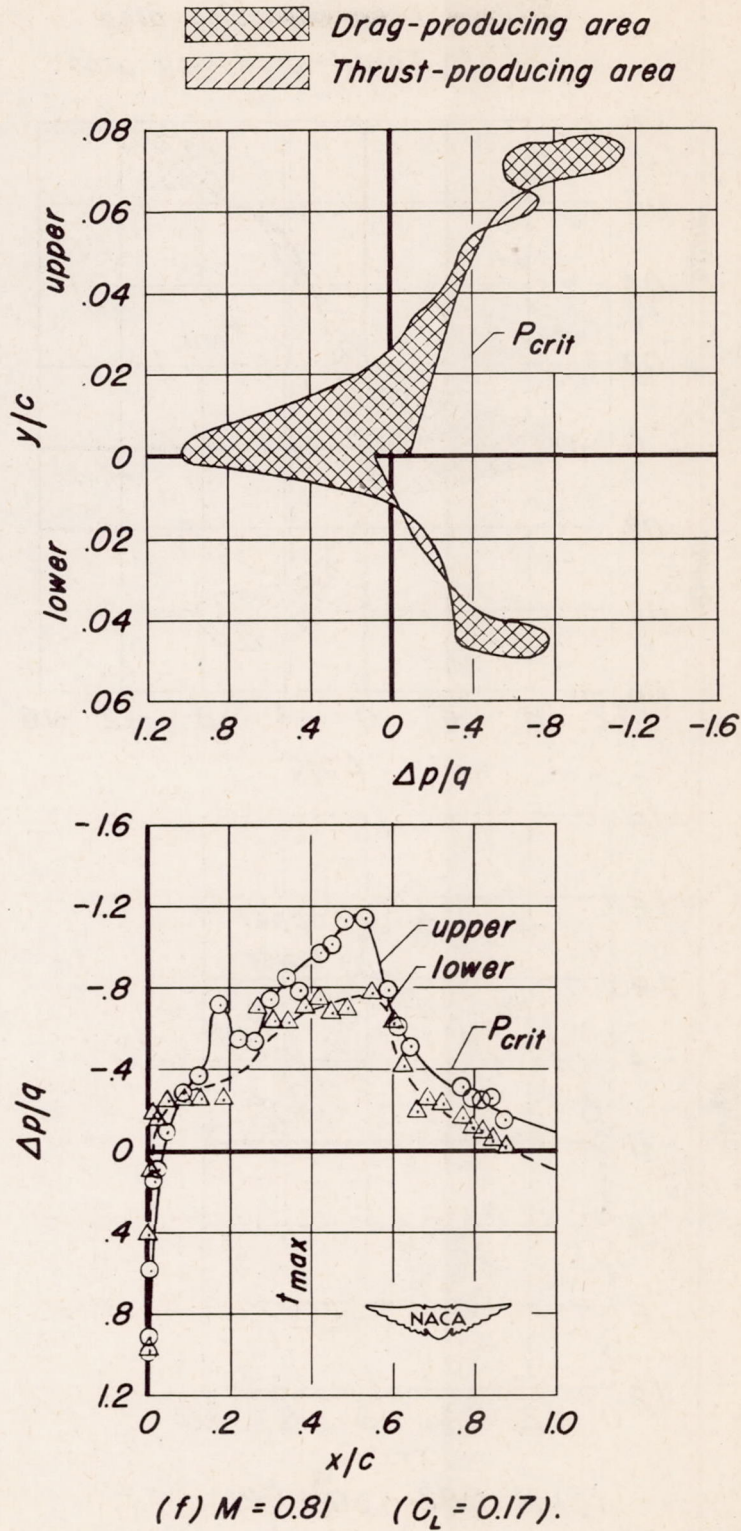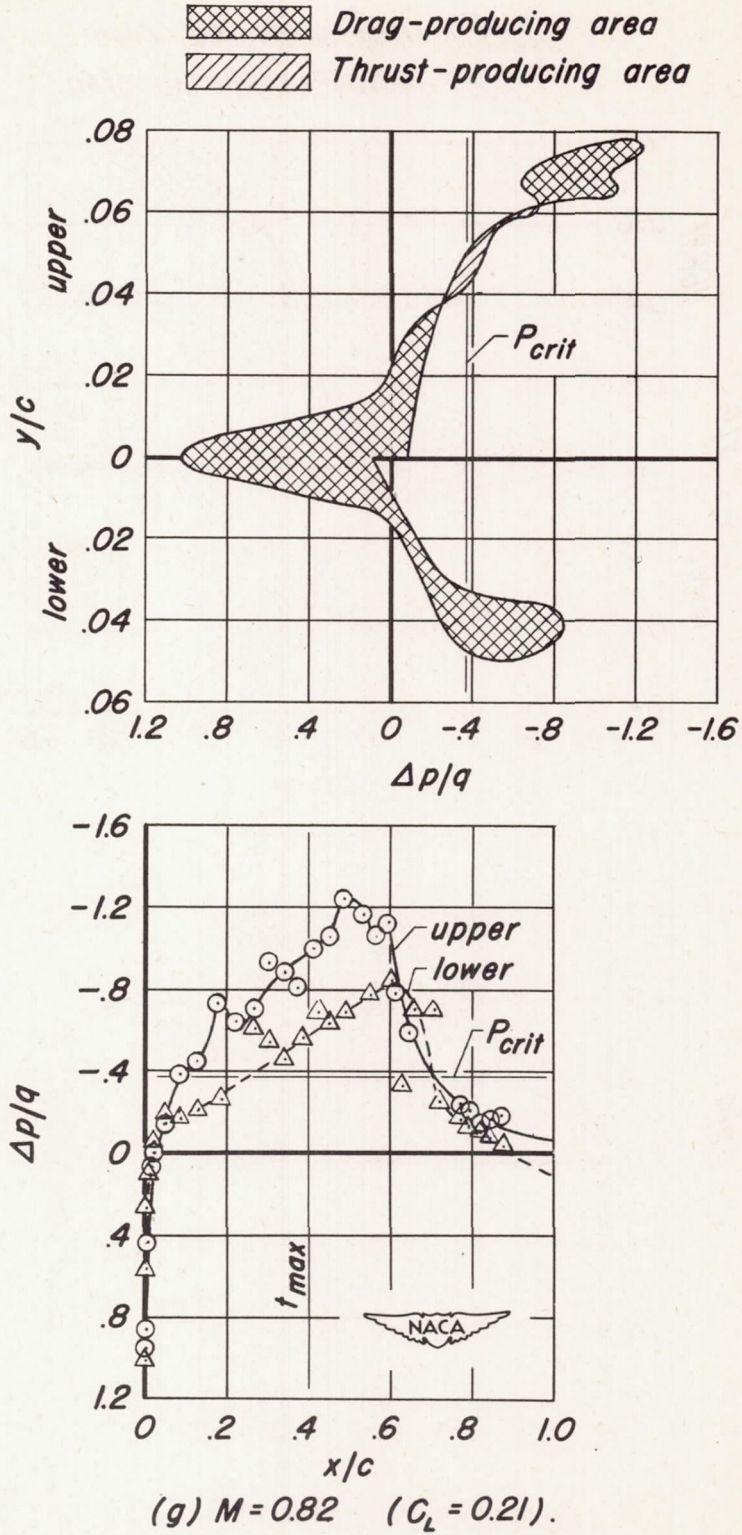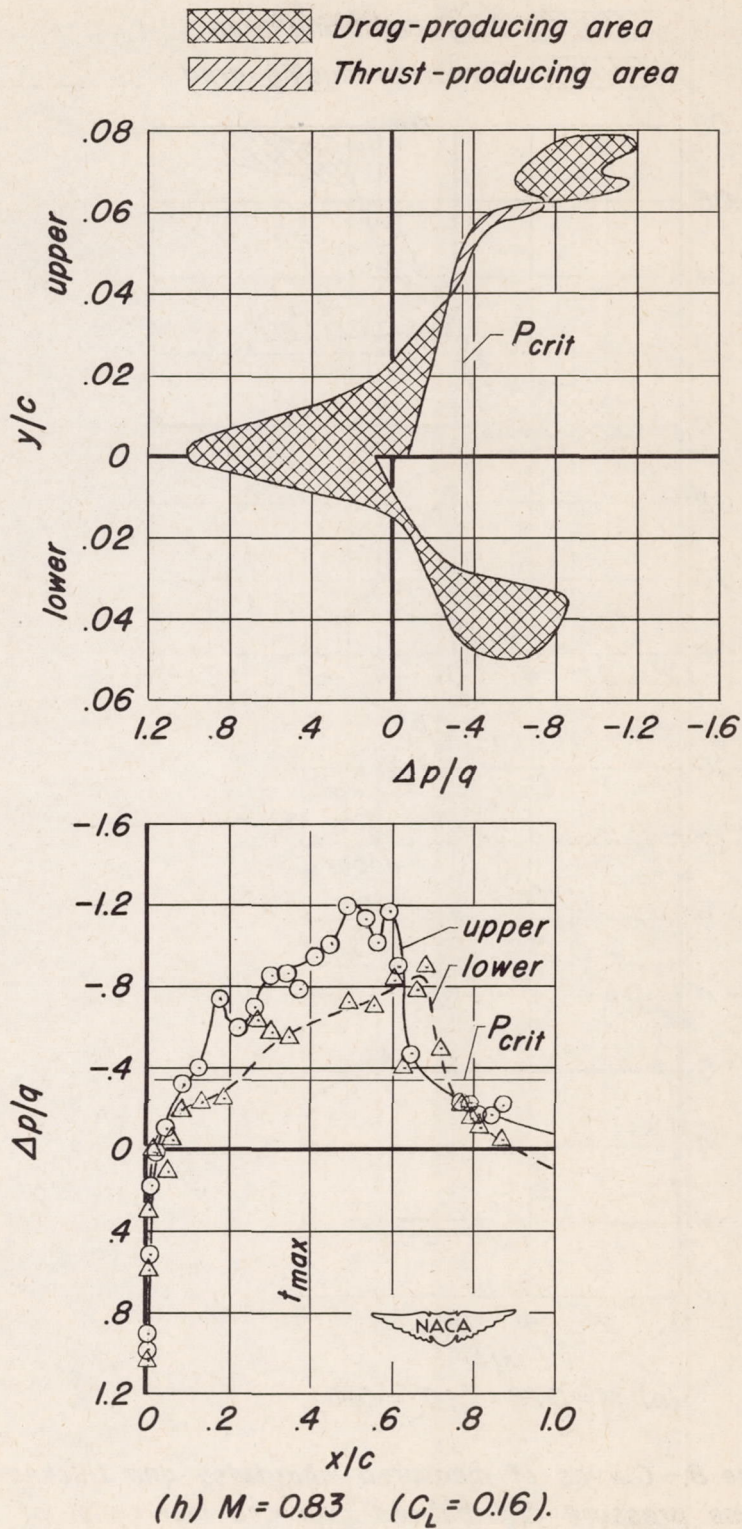
Figure 7.- Concluded.

(a) M = 0.76   (C_L = 0.16).

Figure 8.– Curves of measured chordwise and thickness-
    wise pressure distributions over the test panel at
    selected Mach numbers in the test range. With suction,
    full span slot.

(b) $M = 0.80$ ($C_L = 0.18$).
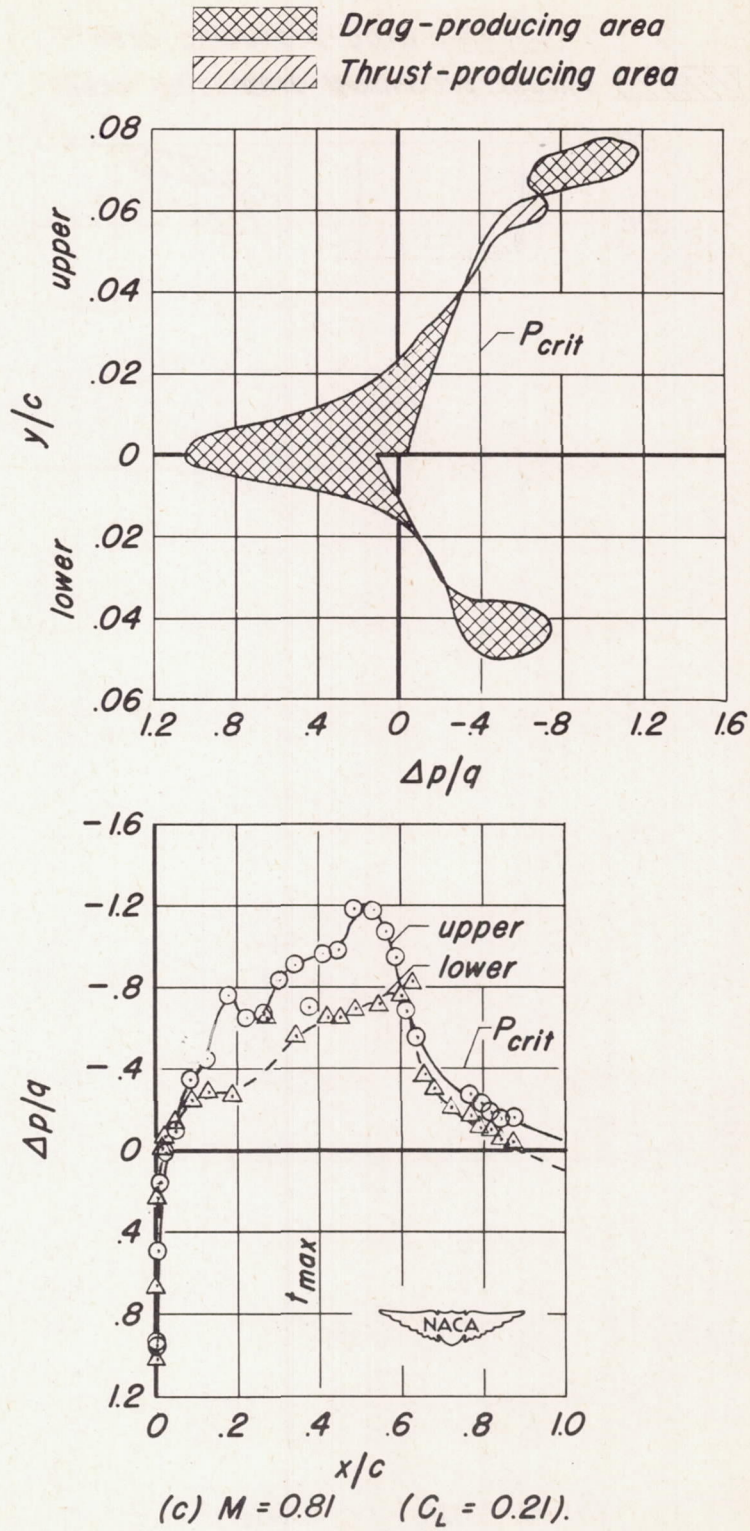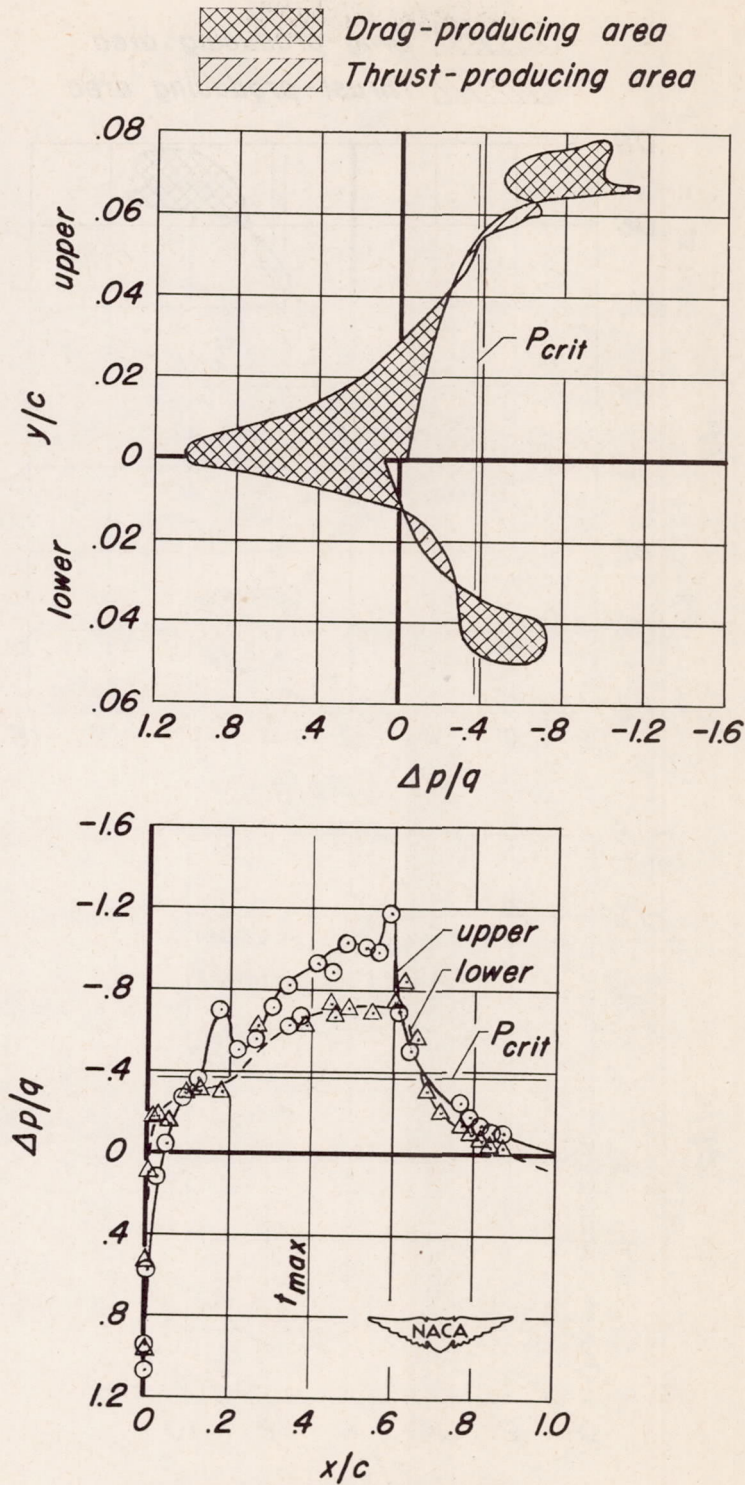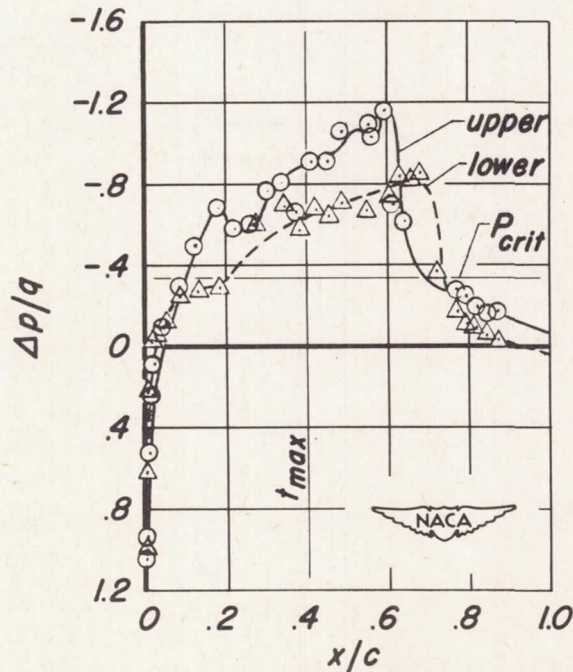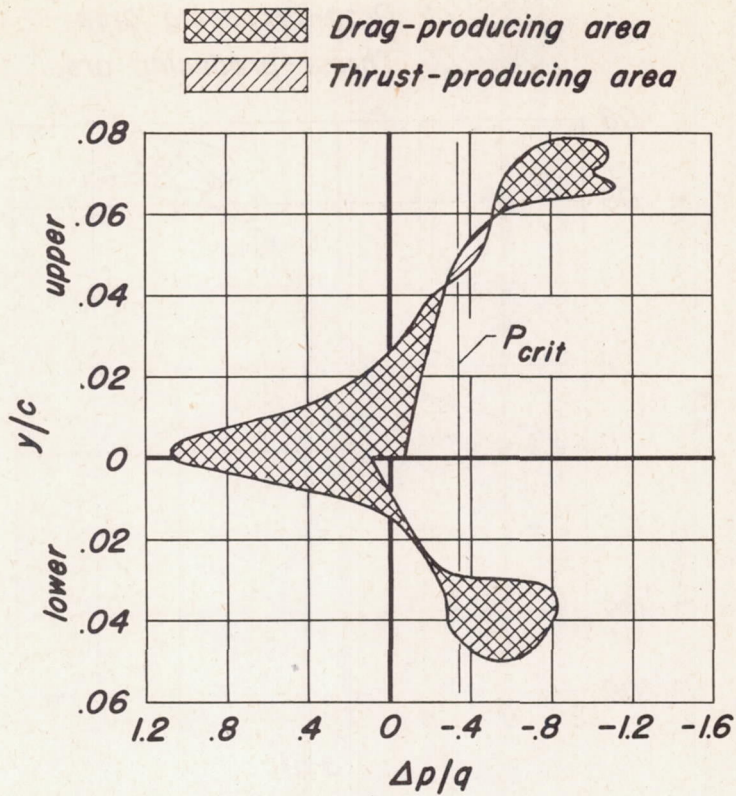
Figure 8.— Continued.

(c) M = 0.81   (C_L = 0.21).

Figure 8.– Continued.

Drag-producing area
Thrust-producing area

(d) M = 0.82 (C_L = 0.12).

Figure 8.- Continued.

(e) M = 0.83   (C_L = 0.13).

Figure 8.- Concluded.

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