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TECHNICAL NOTE

No. 1152

FLIGHT INVESTIGATION OF THE EFFECT OF A LOCAL CHANGE IN WING CONTOUR ON CHORDWISE PRESSURE DISTRIBUTION AT HIGH SPEEDS

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SUMMARY

Tests were made in high-speed flight with a fighter airplane to determine the effect on the chordwise pressure distribution resulting from a minor modification in the contour of the wing upper surface.

The contour modification consisted in the addition of a faired bulge with a maximum depth of 0.13 percent chord, extending approximately from 35 to 55 percent chord. Chordwise pressure distributions were obtained on the original and modified contours over a range of flight Mach numbers from 0.55 to 0.75.

The tests showed that the change in wing contour had a large effect on the pressure distribution when the critical Mach number was exceeded. The critical Mach number of the modified contour was about 0.01 lower than that of the original contour.

INTRODUCTION

In a number of flight investigations chordwise pressure distributions obtained on wings at high speeds have shown appreciable irregularities although the wing section appeared reasonably fair and gave smooth pressure distributions at low speeds. This irregularity in pressure distribution is believed to result either from slight variations in wing contour or from distortion of the wing contours due to air loads imposed in flight. In order to obtain information on the effect of a change in contour on high-speed pressure distributions, static-pressure measurements were made in flight on a section of the wing of a fighter airplane with the original contour modified to include a faired bulge of known size. The tests were made at Mach numbers from 0.55 to 0.75.

SYMBOLS

p	local static pressure
p_o	free-stream static pressure
q_o	free stream dynamic pressure
P	pressure coefficient $\left(\frac{p - p_o}{q_o}\right)$
M_o	free-stream Mach number
M	local Mach number
C_L	airplane lift coefficient
x	distance along chord from leading edge
y	ordinate of wing contour from chord
Δy	deviation of wing contour from design wing contour
c	wing-section chord

Subscripts:

min	minimum value
cr	corresponding to attainment of local velocity of sound

APPARATUS AND TESTS

The tests were made on the right wing of a fighter airplane at a station 53 percent semispan from the plane of symmetry. (See fig. 1.) At this station the wing section included the ammunition compartment having a door on the upper surface that extended from about 35 to 55 percent chord. Spanwise the door extended from 45 to 65 percent semispan. Tests were made with two doors, one conforming to the original wing contour and the other incorporating a bulge. The doors, constructed of aluminum alloy $\frac{1}{2}$-inch thick, were heavily reinforced at each spanwise end and securely bolted at the front and rear to the wing spars to prevent distortion under heavy air loads in flight. The profiles of a part of the wing section, with each of the doors in place, are shown in figure 2. The deviation of the wing contours from the design wing contour is shown in figure 3. The change in contour was essentially equivalent to the addition of a faired bulge with a maximum depth of 0.13 percent chord, extending from about 35 to 55 percent chord, which eliminated the discontinuity at the leading edge of the original door.

Static-pressure measurements were made on the upper surface of the wing at 53 percent semispan with flush orifices spaced at about $2\frac{1}{2}$-percent-chord intervals from 27 to 50 percent chord. The pressure-distribution measurements were taken in dives from 28,000 to 20,000 feet in which Mach numbers from 0.55 to 0.75 were obtained. A few tests with the revised contour were made in dives from an altitude of 12,000 feet to an altitude of 5,000 feet.

RESULTS AND DISCUSSION

The chordwise pressure distributions for the upper surface of the wing from 27 to 50 percent chord for the original and revised wing contours are compared in figure 4 at the same flight Mach number and airplane lift coefficient. The scale for the corresponding local Mach numbers is shown on the right of each plot. At subcritical Mach numbers minimum pressure for the original contour occurred at the surface discontinuity at 35 percent chord (fig. 3); for the revised contour the position of minimum pressure at 45 percent chord was slightly ahead of the position of the maximum height of the bulge. As the Mach number was increased to the supercritical value of 0.70, minimum pressure moved to 42 percent chord for the original contour and to beyond 50 percent chord for the revised contour. With further increase in supercritical Mach numbers to 0.75 the position of minimum pressure for both contours was beyond 50 percent chord.

The difference in the pressure distribution of the two contours was particularly marked as the critical Mach number (local Mach number of 1.00) was exceeded. At a Mach number of 0.70 and a lift coefficient of 0.11 the pressure for the original contour was lower by about 26 percent free-stream dynamic pressure at 40 percent chord and higher by about 56 percent free-stream dynamic pressure at 50 percent chord than the pressures at the same positions for the revised contour. (See fig. 4(c).)

The change in critical Mach number due to the change in contour is illustrated in figure 5 where the minimum pressure coefficients for the two contours and the critical pressure coefficient are plotted against flight Mach number. The corresponding airplane lift coefficients are plotted above the pressure curves. The critical Mach number for the revised contour may be seen to be about 0.01 lower than the critical Mach number for the original contour.

The fact that the air loads imposed in flight had practically no effects on the contour of the doors, and hence on the results shown in figures 4 and 5, is indicated in figure 6 where pressure distributions over the revised wing contour are presented for the same Mach number and lift coefficient but for free-stream dynamic pressures differing by a factor of about 2. The results are presented for subcritical and supercritical conditions and in both cases the pressure distributions at the two values of dynamic pressure agree, in general, within the experimental accuracy. The dynamic pressures for each set of flight conditions in figure 4 were of the same order of magnitude for both contours and therefore the change in pressure distribution due to modification of the wing contour was not affected by air loads.

The comparisons shown in figures 4 to 6 are believed to be almost unaffected by differences in aileron position. Inasmuch as the orifices were inboard of the aileron, the effects of aileron deflection would be secondary. Furthermore, changes in yaw and in lateral position of the airplane center of gravity were too small to require any appreciable differences in the mean aileron position during the dives. Other tests have shown that the variation of aileron deflection in the flight conditions covered in the present investigation was very small (less than ±1°). An indication that the effect of the aileron on the results of figures 4 to 6 was negligible is obtained from figure 4(c) where, for the same flight conditions, the pressures measured in two separate flights for the revised contour were in agreement within the experimental error.

The measured change in pressure distribution due to the change in contour is compared for a condition near the critical Mach number ($M = 0.66$, $C_L = 0.16$) with the change in pressure distribution computed by the method of reference 1 in conjunction with the Kármán-Tsien relation (reference 2). (See fig. 7.) The calculated pressure distribution was determined by adding the increments of induced velocity due to the contour change, computed by the method of reference 1, to the measured velocity distribution for the original contour reduced from a Mach number of 0.66 to 0 by the Kármán-Tsien relation. The resulting pressure distribution was then converted back to a Mach number of 0.66. The comparison indicates that near the critical Mach number the measured change in pressure distribution due to the change in contour was considerably greater than that computed. As a result the critical Mach number would be overestimated when using the computed distribution.

CONCLUDING REMARKS

Flight tests made on a wing section of a fighter airplane indicated that the addition of a faired bulge with a maximum depth of 0.13 percent chord and extending from about 35 to 55 percent chord caused a large change in pressure distribution when the critical Mach number was exceeded. The pressures at a given chordwise position and for the same flight condition differed by as much as 56 percent of the free-stream dynamic pressure. The critical Mach number of the wing section was decreased by 0.01.

Comparison near the critical Mach number of the measured and calculated change in pressure distribution due to the change in contour indicated that the measured change was considerably greater than that computed.

Langley Memorial Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Field, Va., August 14, 1946

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2. Von Kármán, Th.: Compressibility Effects in Aerodynamics. Jour. Aero. Sci., vol. 8, no. 9, July 1941, pp. 337-356.

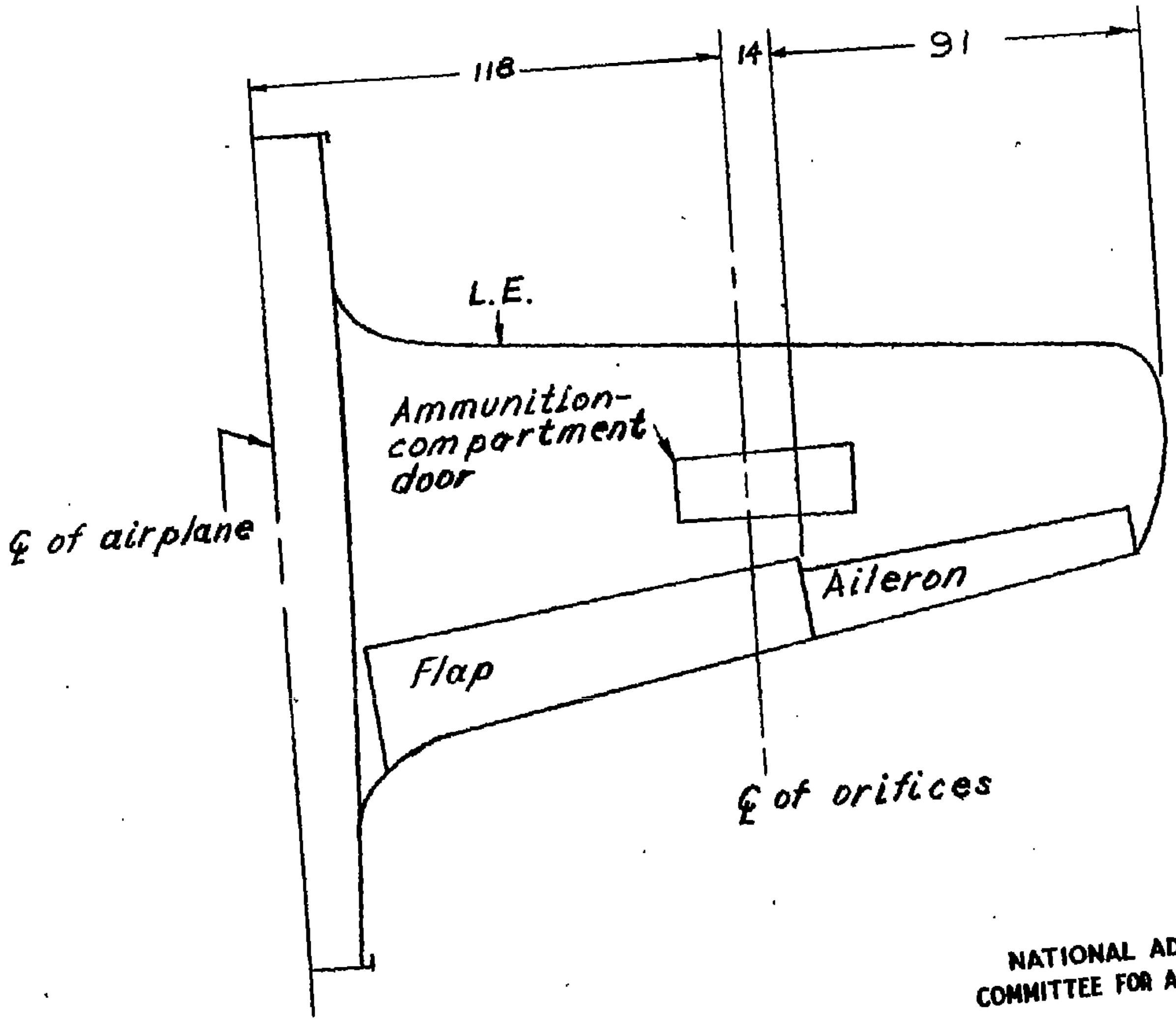

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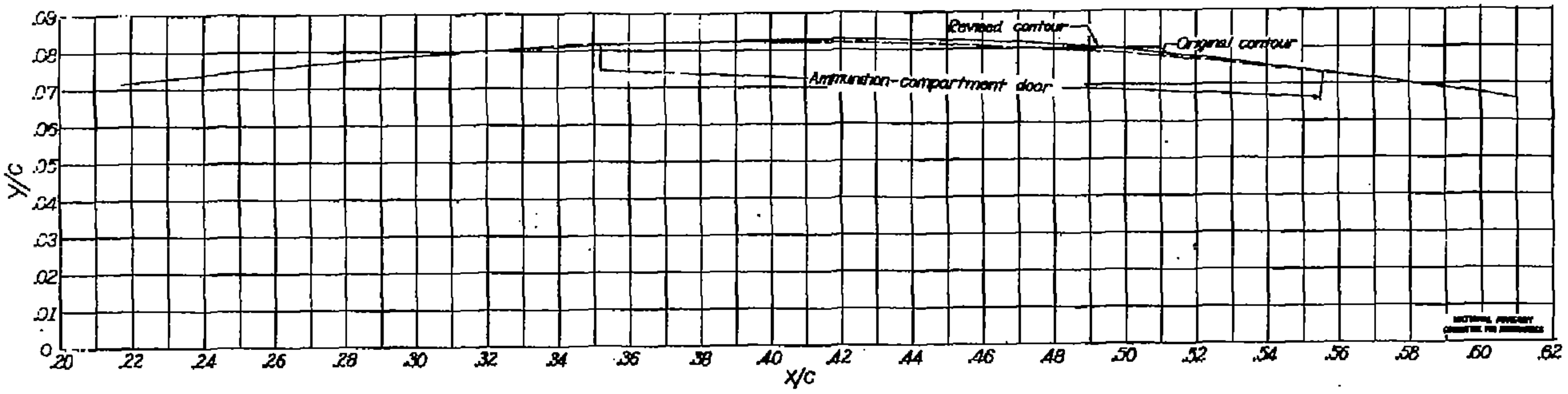
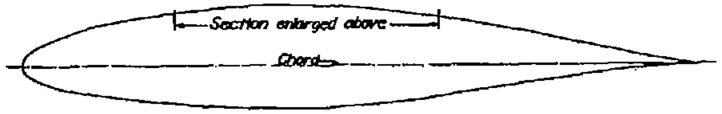
Figure 1.- Plan form of wing showing location of pressure orifices and ammunition compartment. (All dimensions are in inches.)



(a) Original and revised wing contours.



(b) Wing section at test station, 118 inches from plane of symmetry (53 percent semispan).

Figure 2.— Comparison of original and revised wing contours.

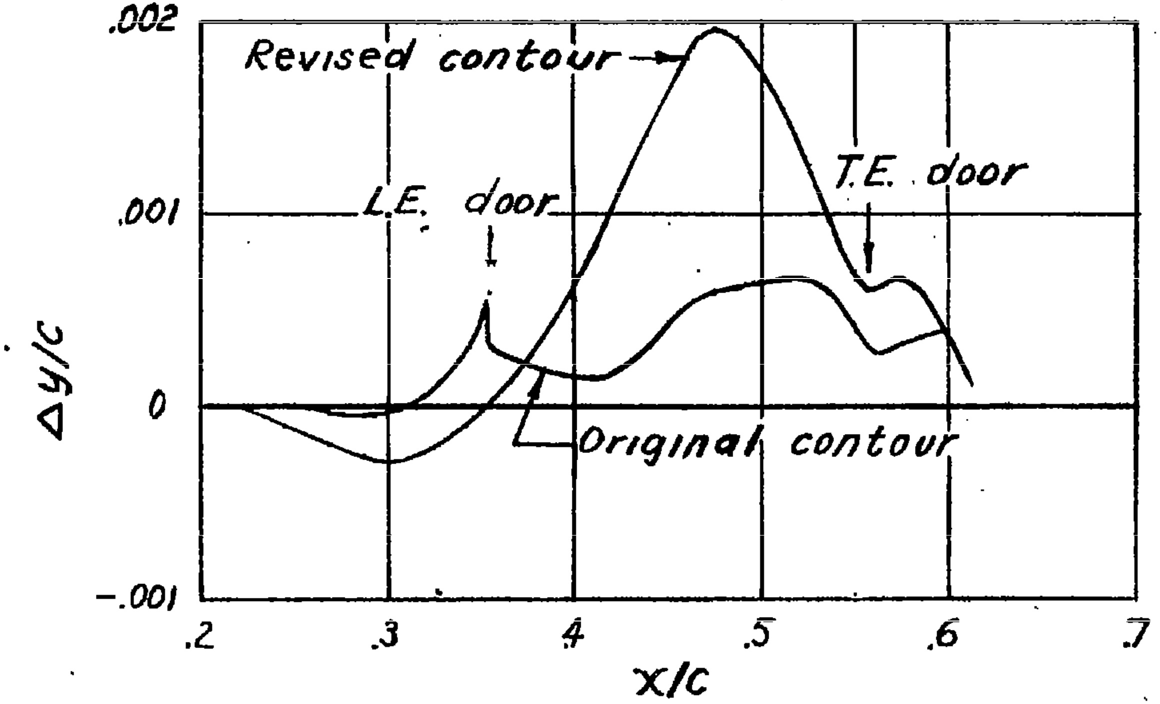


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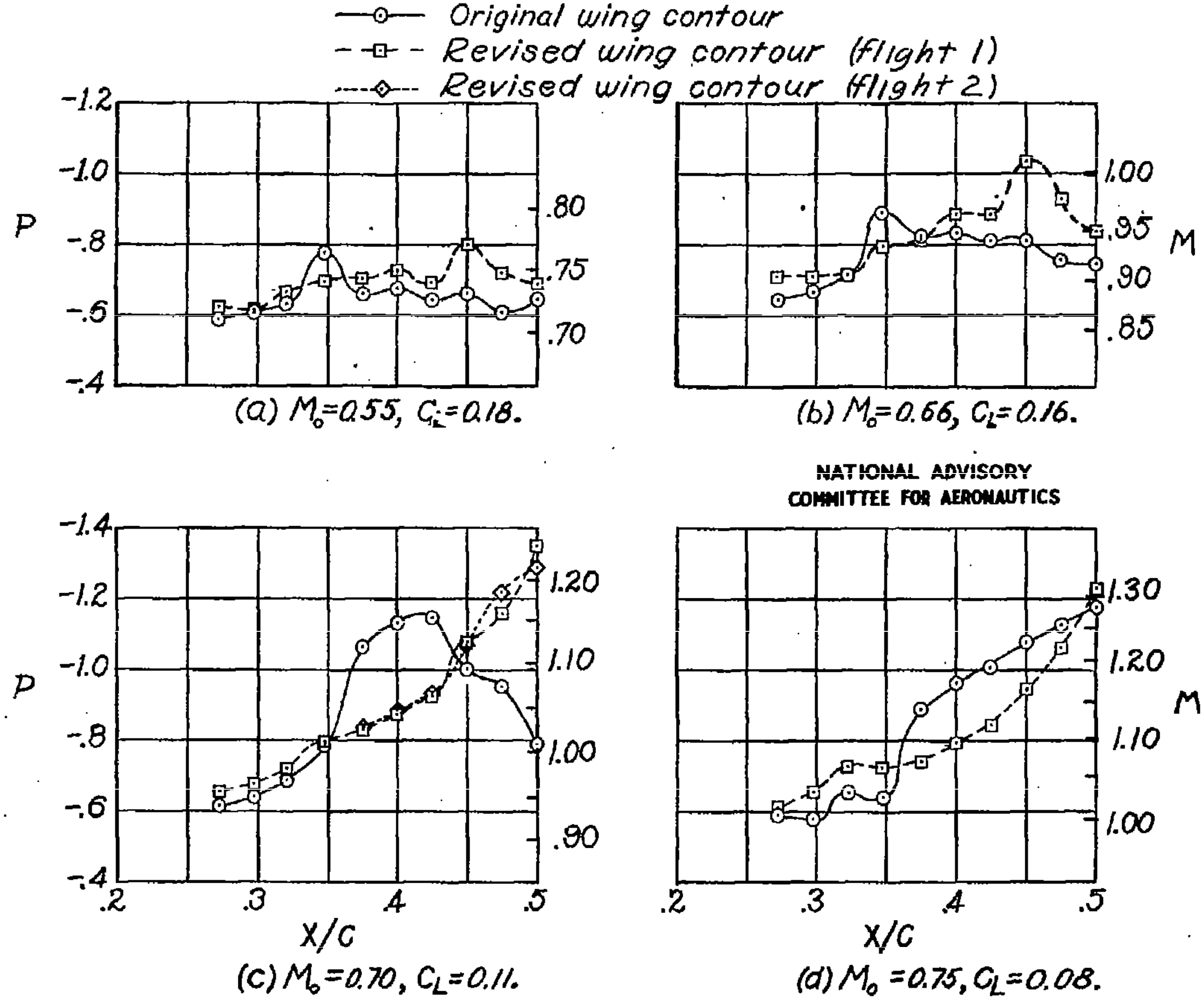
Figure 3.- Deviation of original and revised wing contours from design wing contour.



(a) M_o=0.55, C_L=0.18.

(b) M_o=0.66, C_L=0.16.

(c) M_o=0.70, C_L=0.11.

(d) M_o=0.75, C_L=0.08.

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Figure 4.- Comparison of chordwise pressure distributions for original and revised wing contours at several Mach numbers.

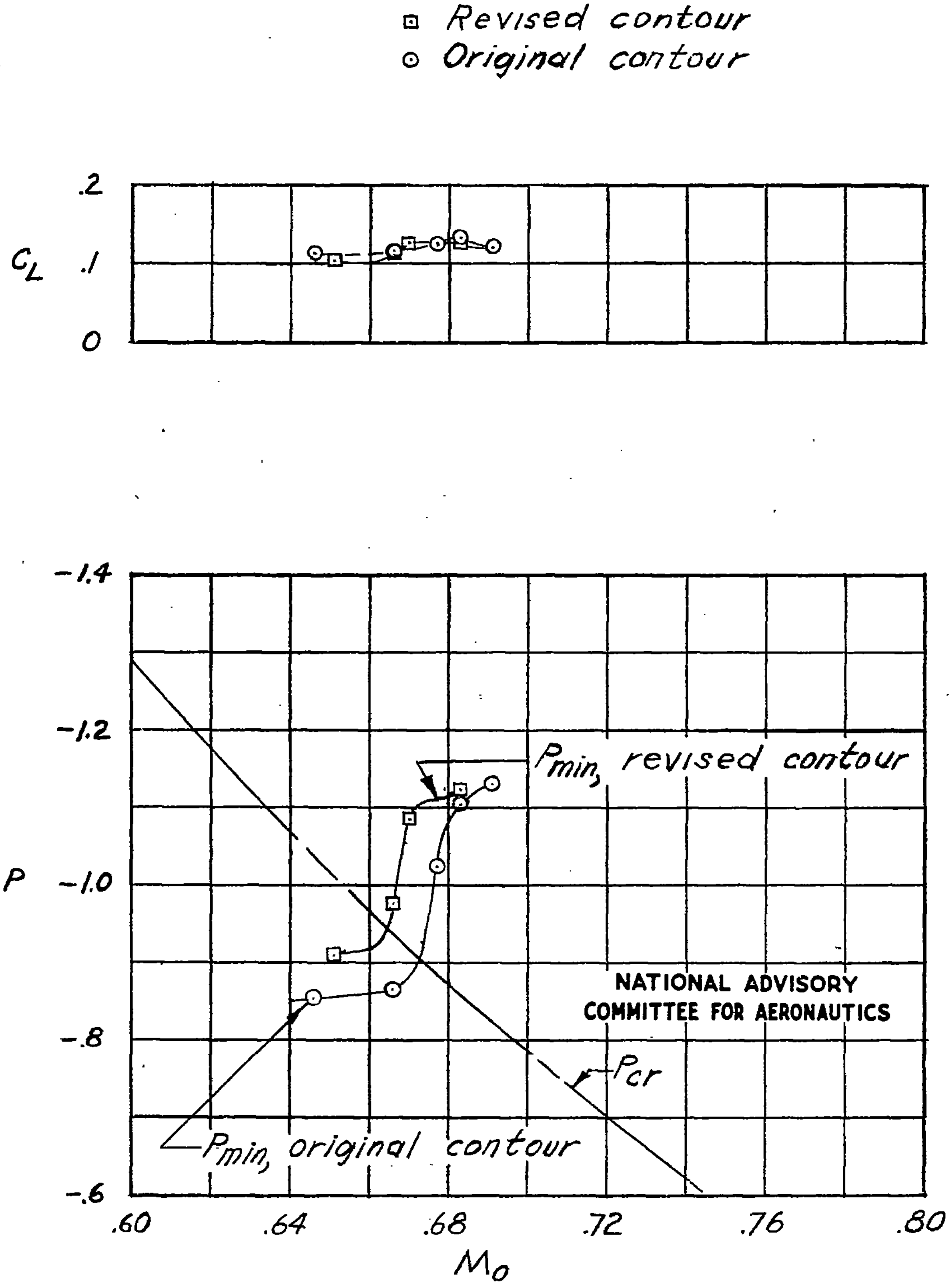
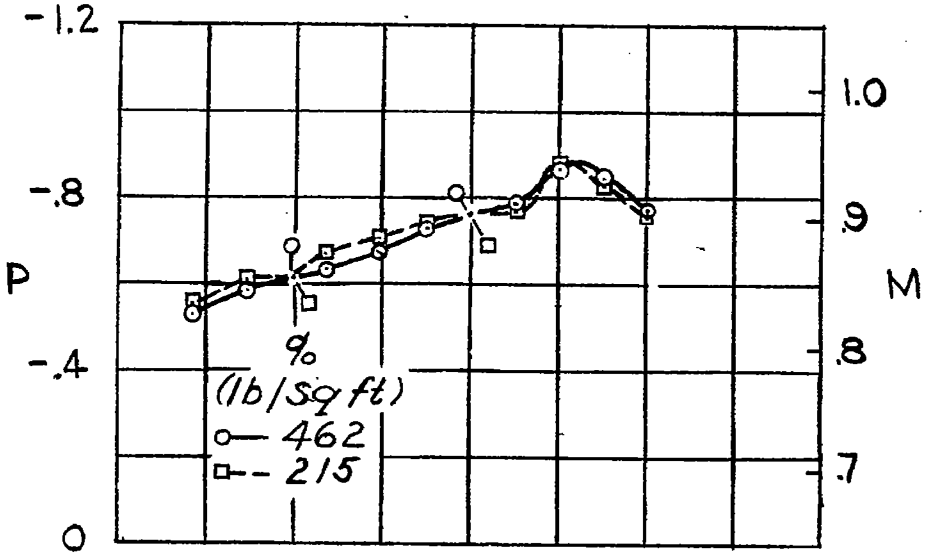


Figure 5.- Variation of minimum pressure for original and revised wing contours with Mach number. Airplane lift coefficients are plotted above minimum-pressure curves.



(a) $M_0 = 0.64$, $C_L = 0.07$.

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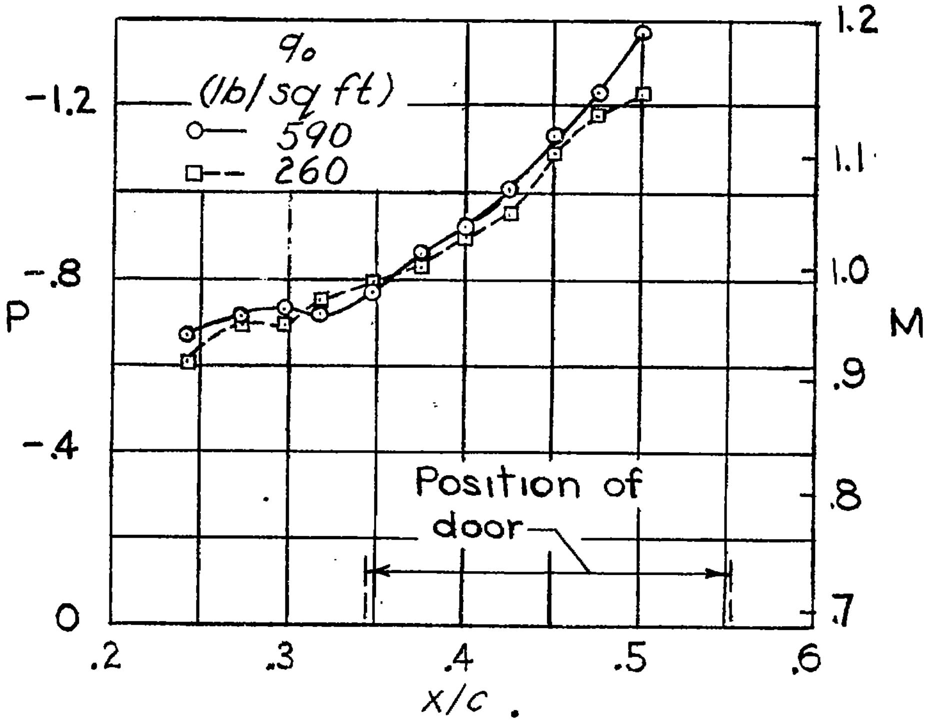

(b) $M_0 = 0.69$, $C_L = 0.14$.

Figure 6.- Pressure distribution over revised wing contour for same Mach number and lift coefficient but for different values of free-stream dynamic pressures.

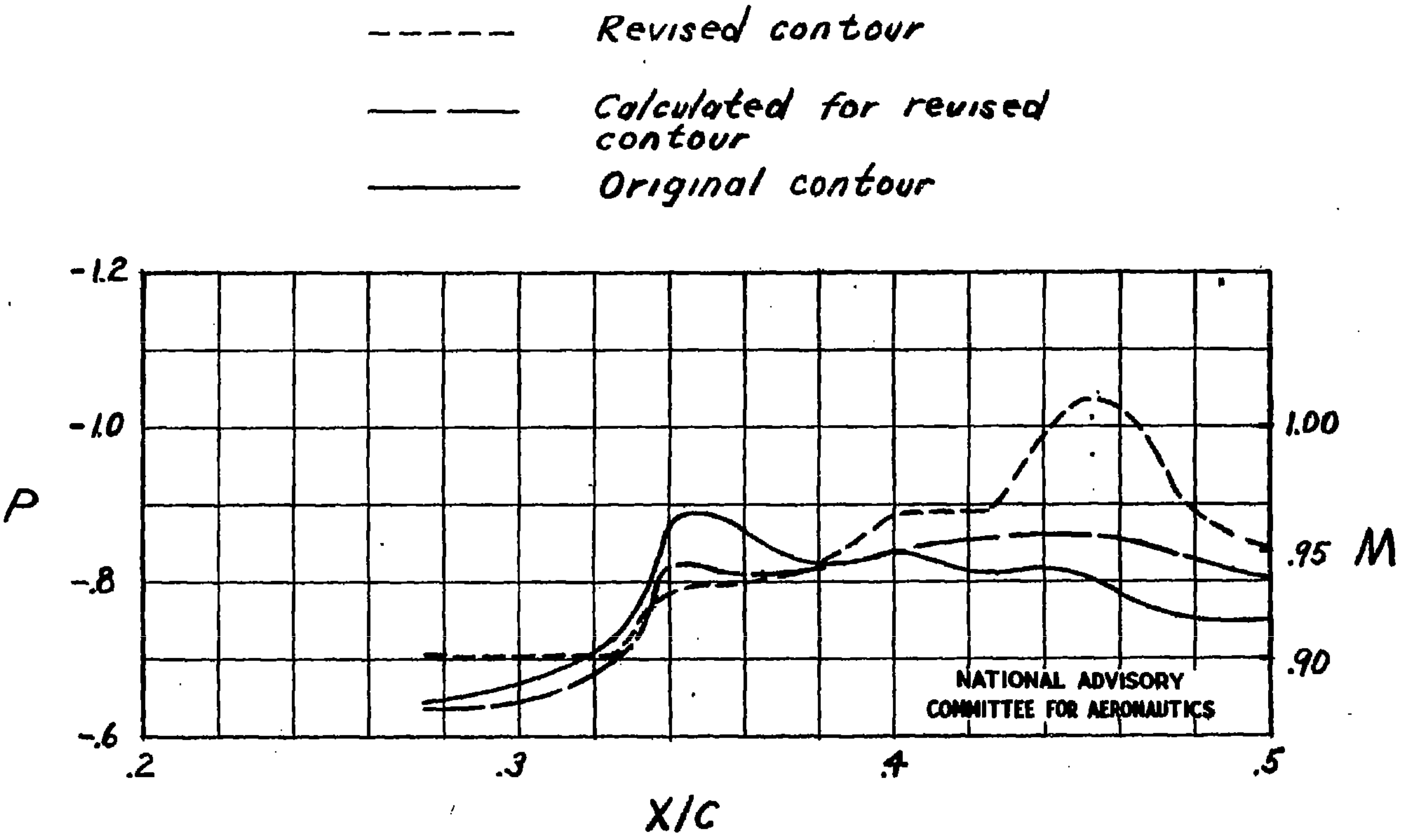

Figure 7.- Comparison at a Mach number of 0.66 of the measured pressure distribution over the original and revised contours with the calculated distribution over the revised contour.