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RESEARCH MEMORANDUM

COMPARATIVE DRAG MEASUREMENTS AT TRANSONIC SPEEDS OF AN NACA 65-006 AIRFOIL AND A SYMMETRICAL CIRCULAR-ARC AIRFOIL

By

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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COMPARATIVE DRAG MEASUREMENTS AT TRANSONIC SPEEDS OF AN NACA 65-006 AIRFOIL AND A SYMMETRICAL CIRCULAR-ARC AIRFOIL

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SUMMARY

Measurements have been made at transonic speeds by the freely-falling-body method to compare the drag of a rectangular plan-form airfoil of aspect ratio 7.6 having an NACA 65-006 section with that of an airfoil of identical plan form and maximum thickness having a symmetrical circular-arc section. These measurements, which were made to determine optimum aerodynamic shapes and configurations for use in the transonic- and supersonic-speed ranges, showed the drag for the symmetrical 6-percent-thick circular-arc airfoil to be 16 percent greater at the speed of sound and 11 percent greater at a Mach number of 1.16 than the drag of the NACA 65-006 airfoil. In an effort to simplify the test procedure, both airfoils were mounted on the same body, the circular-arc airfoil to the rear of and at right angles to the NACA 65-006 airfoil. As the effect of this simplification may be assumed to be limited to the interference effect noted in previous tests(in which the measured drag of an airfoil in the front position was slightly greater than the measured drag for an identical airfoil in the rear position), it may be concluded that the actual drag difference is greater than that measured. The most probable value of the drag of the circular-arc airfoil is about 20 to 25 percent greater at a Mach number of 1.0 and 15 to 19 percent greater at a Mach number of 1.16 than the drag of the NACA 65-006 airfoil.

Comparison with the results of previous tests of an NACA 65-009 airfoil showed the drag per unit of frontal area for this airfoil to be about 17 percent greater at supersonic speeds than the drag of the NACA 65-006 airfoil.

INTRODUCTION

One of the many problems encountered in the design of aircraft for the transonic and the supersonic speed ranges is the selection of an airfoil section having low drag in the design high-speed range yet having good low-speed characteristics, especially a high maximum lift so that a safe landing speed may be obtained. Several of the proposed designs for such aircraft have incorporated sharp-nose airfoils of the biconvex or double-wedge type; these airfoils, although having less desirable low-speed characteristics, are assumed to have lower drags at supersonic speeds than conventional rounded-nose airfoils. The assumption that sharp-nose airfoils have lower drag at supersonic speeds is supported by the literature (references 1 to 3) although experimental evidence confirming this assumption is practically nonexistent.

As part of the research program of the National Advisory Committee for Aeronautics to determine airfoil sections, wing plan forms, body shapes, and winged body configurations having a minimum of drag in the transonic and supersonic speed ranges, tests have been made by the Flight Research Division of the Langley Memorial Aeronautical Laboratory to compare the drag of sharp-nose and conventional low-drag airfoil sections at transonic speeds. The results of these tests are presented in the present paper as a comparison of curves showing the measured variations of drag coefficient with Mach number for a rectangular plan-form wing having an NACA 65-006 section and for a wing of identical plan form having a symmetrical circular-arc section of the same maximum thickness. The tests were performed by means of the freely-falling-body method described in references 4 to 6.

APPARATUS AND METHOD

Test body and airfoils.- The general arrangement of the test configuration is shown by the photograph (fig. 1) and the details and dimensions are shown on the line drawing (fig. 2). The two test airfoils had identical rectangular plan forms and frontal areas and differed only in airfoil section; the front airfoil had NACA 65-006 sections and the rear airfoil had symmetrical circular-arc sections with a maximum thickness of 6-percent chord. The test airfoils were constructed of metal, and because of the machining techniques used the leading and trailing edges of the circular-arc airfoil were somewhat flattened. The contour of these edges may be approximated with a maximum discrepancy of 0.001 inch by a radius of 0.005 inch, or about one-fourth of the leading-edge radius of the NACA 65-006 section. The test airfoils (including that part of the airfoil within the body) had an aspect ratio of 7.6 and entered the body through rectangular slots $9\frac{1}{2}$ inches long and 1 inch wide. The body on which the test airfoils were mounted was the same as the body used in the tests of references 5 and 6 with the exception of the tail fairing. The small fairing used previously was replaced for the present test by a cylindrical extension with a flat base so that the pressure acting on such a flat base could be measured.

Measurements.- Measurement of the desired quantities was accomplished as in the previous tests (references 5 and 6) through use of the NACA radio-telemetering system and radar and phototheodolite equipment. The following quantities were recorded at two separate ground stations by the telemetering system:

(1) Force exerted on body by each test airfoil as measured by a spring balance

(2) Total retardation of body and airfoils as measured by a sensitive accelerometer alined with longitudinal axis of body

(3) Pressure acting on flat base of test body as measured by four orifices connected to an aneroid cell.

A time history of the position with respect to ground axes of the body during free fall was recorded by radar and phototheodolite equipment, and a survey of atmospheric conditions applying to the test was obtained from synchronized records of atmospheric pressure, temperature, and geometric altitude taken during the descent of the airplane from which the test body was dropped. The direction and velocity of the horizontal component of the wind in the range of altitude for which data are presented were obtained from radar and phototheodolite records of the path of the ascension of a free balloon.

Reduction of data.- As in the previous tests the velocity with respect to ground axes V_g of the body during free fall was obtained both by differentiation of the flight path determined by radar and phototheodolite equipment and by integration of the vector sums of gravitional acceleration and the directed retardation measured by the longitudinal accelerometer. The true airspeed V was obtained by vectorially adding the velocity V_g and the horizontal wind velocity measured at the appropriate altitude.

The drag D of each airfoil was obtained from the relation

$$D = R + W_T a_e$$

where

R measured reaction between airfoil and body, pounds

W_T weight of airfoil assembly supported on spring balances, pounds

a_e reading of accelerometer (retardation), g

The atmospheric pressure p, the temperature T, and the airfoil frontal area F were combined with simultaneous values of true airspeed and airfoil drag to obtain Mach number M and the ratio D/Fp. The variation of the parameter D/Fp with Mach number affords a simple and convenient means for expressing drag in the transonic-speed range as a function of Mach number, altitude, and size.

Values of conventional drag coefficient based on frontal area C_{D_F} were obtained from the relation

$$C_{D_F} = \frac{D/Fp}{\frac{\gamma}{2} M^2}$$

where the ratio of specific heats γ was taken as 1.4. Drag coefficients based on plan area C_D were obtained by multiplying the values of C_{D_F} by the ratio of frontal area to plan area. Areas used did not include that area enclosed within the body.

RESULTS

A time history of important quantities obtained in the present test is presented as figure 3.

Velocity measurements.- For purposes of comparison the ground speed V_g obtained from each of the two independent methods of measurement is presented in figure 3. The ground-speed data obtained from the accelerometer are shown as a solid line, and the radar and phototheodolite data are represented by test points. From this figure the maximum discrepancy in V_g can be seen to be about 13 miles per hour although the mean discrepancy is considerably smaller. As no systematic discrepancy of the type normally associated with accelerometer errors (a gradual diverging of the curves) is apparent and, as careful study of the records disclosed an intermittent fluctuation of as much as 1 percent in the rate of the clock used to provide a time base for the radar and phototheodolite records, the accelerometer data are considered to be the more reliable. The radar and phototheodolite data presented have been corrected for the average timing error; however, individual points or groups of points may be in error by as much as 1 percent of the velocity because of the intermittent nature of the rate fluctuation The velocity data obtained from the accelerometer, converted to true airspeed V by use of the wind data, are shown in the time history as a dash-line fairing. This velocity was used to compute the Mach number, which is believed to be accurate within ± 0.01. The Mach number corresponding to the ground speed V_g is also shown in figure 3 so that the magnitude of the wind correction may be readily seen.

<u>Base-pressure measurements</u>.- The measurement of base pressure was obtained incidentally to the subject test for use in body-drag research. Although analysis of these data is beyond the scope of the present paper, these measurements are included so that a minimum of delay would be incurred in making the information generally available. The equipment used to measure the base pressure required that this pressure be known at some point during the test. This reference pressure was calculated for a point immediately following the release of the test body by use of the results of reference 7. Results from reference 7, which reports wind-tunnel measurements of the pressure acting on a total-pressure tube at an angle of yaw of 180° at Mach numbers from 0.3 to 0.9, are shown in figure 4 where the ratio of base pressure to atmospheric pressure is plotted against Mach number. The free-fall data, which are also plotted in figure 4 are seen to agree closely with the wind-tunnel data from $M = 0.64$ where the free-fall data were referenced, to the maximum Mach number attained in the tunnel tests $(M = 0.9)$.

<u>Airfoil drag measurements</u>.- The spring balances with which the airfoil drag forces are measured must withstand the high drag forces occurring at supersonic Mach numbers and high pressures (low altitudes) and are therefore necessarily relatively insensitive to the small drag occurring at subsonic Mach numbers and low pressures

(high altitudes). The drag parameters are therefore less accurate at the lowest Mach numbers for which data are presented than in the supersonic range where the drag is high. The values of the ratio D/Fp are believed to be accurate within about ±0.01 at $M = 0.85$, the limit of accuracy decreasing to ±0.004 at $M = 1.16$. Corresponding values of C_D are accurate within ±0.0013 at $M = 0.85$ and within ±0.0006 at $M = 1.16$. These values correspond to about 1 percent of the full-scale balance deflection for values of D/Fp; however, the values of C_D include an additional increment due to the possible uncertainty in Mach number of ±0.01. For these reasons, the range of the balance should be chosen as small as possible so that results of usable accuracy can be obtained near the drag rise. For the tests herein reported, the range was chosen slightly too small with the result that no airfoil drag data were obtained for the last 5 seconds of the drop. (See fig. 3.) As the rate of change of Mach number with time is small near the end of the drop, however, data for only 0.02 of a Mach number were lost. The Reynolds number, based on the airfoil chord, increased from about 0.75×10^6 at release to 5×10^6 at $M = 1.16$.

The results of the airfoil drag measurements are summarized in figure 5 where curves are presented which show the measured variations of D/Fp, C_{D_F}, and C_D with Mach number for both the NACA 65-006 and the symmetrical circular-arc airfoils. The $\frac{D}{Fp}$- curves of figure 5 show that for the NACA 65-006 airfoil the drag per unit of frontal area rose abruptly from about 0.05 of atmospheric pressure at $M = 0.88$ to 0.36 of atmospheric pressure at $M = 0.98$. The drag per unit of frontal area then increased almost linearly to 0.51 of atmospheric pressure at $M = 1.16$. For the symmetrical circular-arc airfoil, however, the drag per unit of frontal area rose at first less abruptly and then more abruptly than the drag of the NACA 65-006 airfoil, the drag increasing from 0.07 of atmospheric pressure at $M = 0.88$ to 0.42 at $M = 0.98$. The drag of the circular-arc airfoil then increased at about the same rate as the drag of the NACA 65-006 airfoil and reached a value of drag per unit of frontal area of 0.57 of atmospheric pressure at $M = 1.16$. The drag of both airfoils began to rise abruptly at about $M = 0.88$; however, the circular-arc airfoil had a greater drag than the NACA 65-006 airfoil at the lower Mach numbers by an amount approximately equal to the accuracy of the measurement in this region.

DISCUSSION

For purposes of comparison, $\frac{D}{Fp}$-curves for the airfoils, tests of which are presented herein, and for an NACA 16-006 airfoil of aspect ratio 4.9 mounted on a different type of body (reference 4) are shown in figure 6. The $\frac{D}{Fp}$-curves for the NACA 65-006 section and for the NACA 16-006 section agree closely; this result was expected because of the similarity of the profiles but provides confirmation for the measurement. Further confirmation is provided by as yet unpublished results from free-fall tests of NACA 16-006 airfoils mounted in the same manner as the airfoils of reference 4 on a body of considerably higher fineness ratio.

Previous tests of identical rectangular plan-form airfoils tested in both the front and rear positions on the body (references 5 and 6) showed that at supersonic speeds a higher drag was measured for the front airfoil than for the rear airfoil. This difference, which amounted to 0.02 to 0.04 at values of D/Fp of 0.4 to 0.6, was presumably due to the location of the airfoils in different parts of the flow field of the body and/or the effect of the front airfoil on the rear airfoil. Curves of D/Fp from reference 6 are presented in figure 6 to illustrate the magnitude of the resulting interference effect.

Comparison of the $\frac{D}{Fp}$-curves of figure 6 shows that the drag of the airfoil having the symmetrical circular-arc section was greater than the drag of the airfoil having the NACA 65-006 section throughout the tested Mach number range, the measured difference amounting to 0.06 (16 percent) of atmospheric pressure per unit of frontal area at $M = 1$. This difference was constant from $M = 1$ to $M = 1.16$; however, at $M = 1.16$, the difference had decreased to 11 percent. If the interference effect noted in previous tests can be assumed to apply to the present test in which airfoils of different section are mounted on the same body (the NACA 65-006 section in the front position), the actual difference between the drags of the circular-arc and the NACA 65-series sections is somewhat greater than that measured. The most probable value of the drag of the circular-arc section is therefore 20 to 25 percent greater at $M = 1$ and 15 to 19 percent greater at $M = 1.16$ than the value of the drag of the NACA 65-006 section.

Tests of rectangular plan-form airfoils of aspect ratio 2.7 having NACA 65-009 and 9-percent-thick circular-arc sections have been reported in reference 8. These test airfoils were attached to a rocket-propelled body similiar in shape to the body used in the free-fall tests. The NACA 65-009 airfoil was found to have less drag at $M = 1$ than the circular-arc airfoil by about the same percentage shown by the free-fall data; however, at a Mach number of 1.16 the difference had decreased to only 5 percent.

Comparison of the $\frac{D}{F_p}$-curves shown in figure 6 for the NACA 65-006 airfoil and the NACA 65-009 airfoil (reproduced from reference 6) of identical plan form and aspect ratio tested in the same position on the body (thus eliminating the difference in interference effects) shows that the 9-percent-thick 65-series airfoil had about 17 percent more drag per unit of frontal area at supersonic speeds. A similar comparison indicates that the 6-percent-thick circular-arc section had a drag per unit of frontal area slightly greater near $M = 1$ and about equal at $M = 1.16$ to the drag of the NACA 65-009 section previously tested.

The drag results obtained for the symmetrical circular-arc section are not compared with the Ackeret theory as the data do not extend to Mach numbers high enough for the theory to be applicable. According to calculation, an oblique shock wave would not attach to the leading edge of the airfoil (the condition for application of the Ackeret theory) until a Mach number of 1.32 was attained.

The lower drag herein reported for the conventional rounded-nose airfoil section at low supersonic speeds and the complete inadaquacy of present theory to predict the characteristics of this type of section even in the higher supersonic-speed range, where reasonably adequate theory is available for sharp-nose sections, shows the necessity for further tests at higher speeds. These tests should determine the extent of the lower drag for the rounded-nose airfoil section into the supersonic-speed range and, at speeds above this range, whether the magnitude of the possible decrease in drag compensates for the less desirable low-speed characteristics of the sharp-nose airfoil sections. The research should be directed toward determining the optimum airfoil for any design condition and therefore should include consideration of control effectiveness and lift characteristics.

CONCLUDING REMARKS

Drag measurements have been made at transonic speeds by the freely-falling-body method for rectangular plan-form airfoils having an aspect ratio of 7.6 and having NACA 65-006 and symmetrical 6-percent-thick circular-arc sections. The results show that the drag per unit of frontal area for the NACA 65-006 airfoil rose abruptly from 0.05 of atmospheric pressure at a Mach number of 0.88 to 0.36 at a Mach number of 0.98 and then increased almost linearly to 0.51 at a Mach number of 1.16. The drag of the airfoil having a symmetrical circular-arc section was greater than the drag of the airfoil having the NACA 65-006 section throughout the tested Mach number range, the measured difference amounting to 0.06 (16 percent) of atmospheric pressure per unit of frontal area at a Mach number of 1. The difference was constant from a Mach number of 1 to a Mach number of 1.16; however, at a Mach number of 1.16 the difference had decreased to 11 percent. If the interference effect noted in previous tests in which identical airfoils were tested in front and rear positions on the body can be assumed to apply to the present test in which different airfoils are mounted in the two positions, the most probable value of the drag of the circular-arc airfoil is about 20 to 25 percent greater at a Mach number of 1 and 15 to 19 percent greater at a Mach number of 1.16 than the drag of the NACA 65-006 airfoil.

Comparison of the NACA 65-006 airfoil with an NACA 65-009 airfoil previously tested in the same position on a similar body (thus eliminating the difference in body-interference effects) showed that the 9-percent-thick airfoil had about 17 percent more drag per unit of frontal area at supersonic speeds.

Further tests at higher speeds should be performed to determine the extent of the lower drag of the rounded-nose airfoil section, herein reported for the transonic and low supersonic-speed ranges, into the higher supersonic-speed range. At speeds at which the rounded-nose section has higher drag these tests should determine whether the magnitude of the possible decrease in drag compensates for the less desirable low-speed characteristics of the sharp-nose sections.

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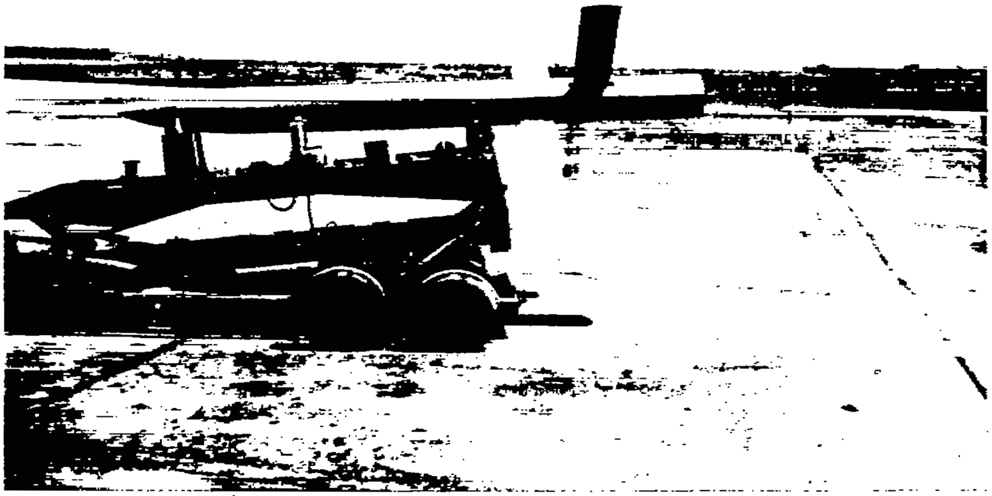

Figure 1.- Side view of the airfoil test body.

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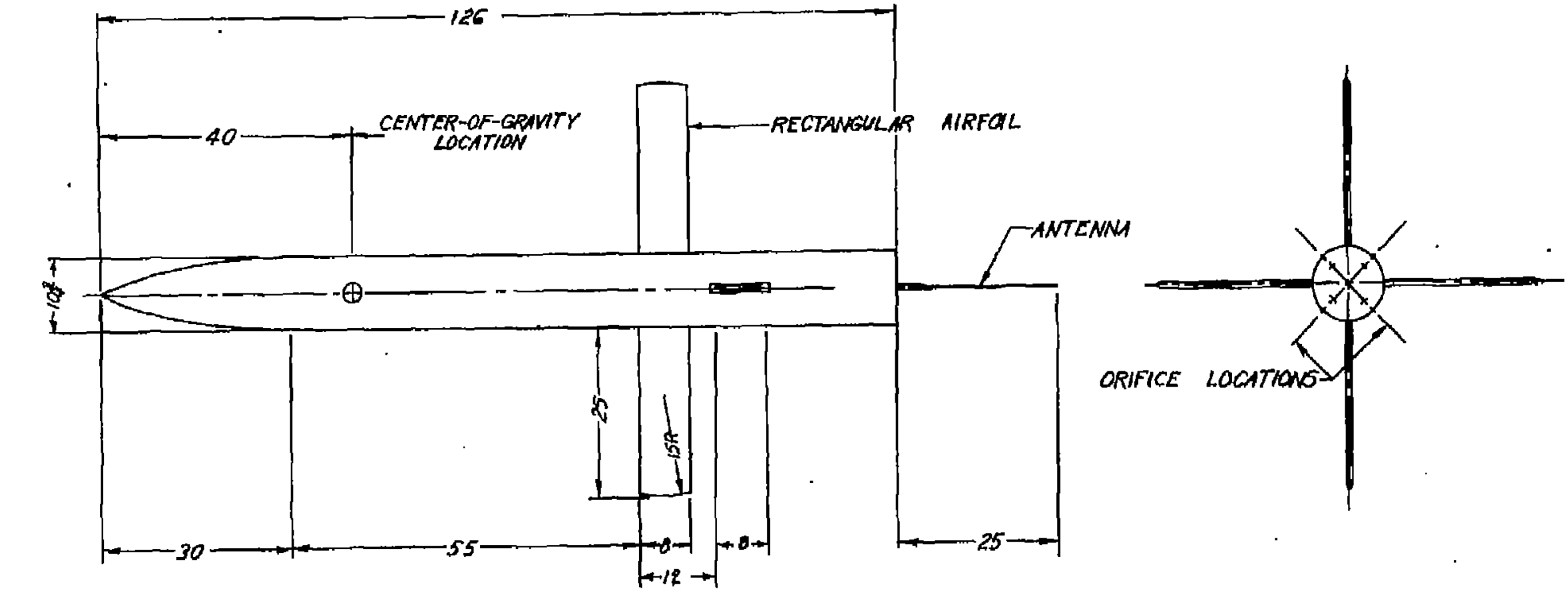

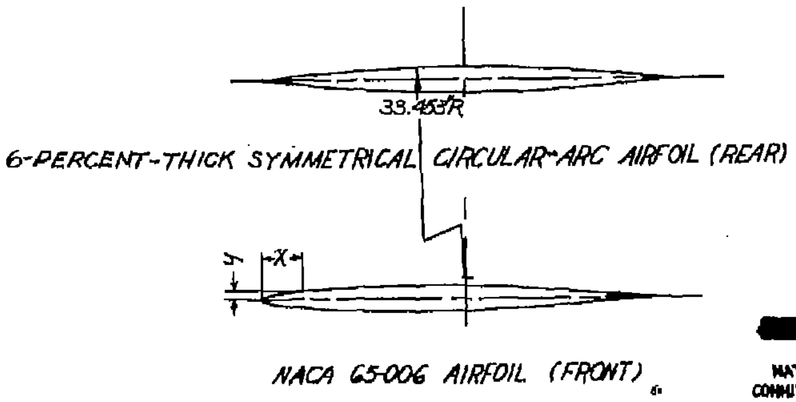

AIRFOIL - SECTION COORDINATES (NACA 65-006 SECTION)					
x	y	x	y	x	y
.000	.000	1.600	.199	5.200	.180
.040	.038	2.000	.216	5.600	.155
.060	.046	2.400	.228	6.000	.128
.100	.057	2.800	.236	6.400	.099
.200	.076	3.200	.240	6.800	.069
.400	.105	3.600	.239	7.200	.041
.600	.127	4.000	.232	7.600	.016
.800	.146	4.400	.219	8.000	.000
1.200	.176	4.800	.202	-----	-----
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Figure 2.- General arrangement and dimensions of the airfoil test body. All dimensions are in inches.

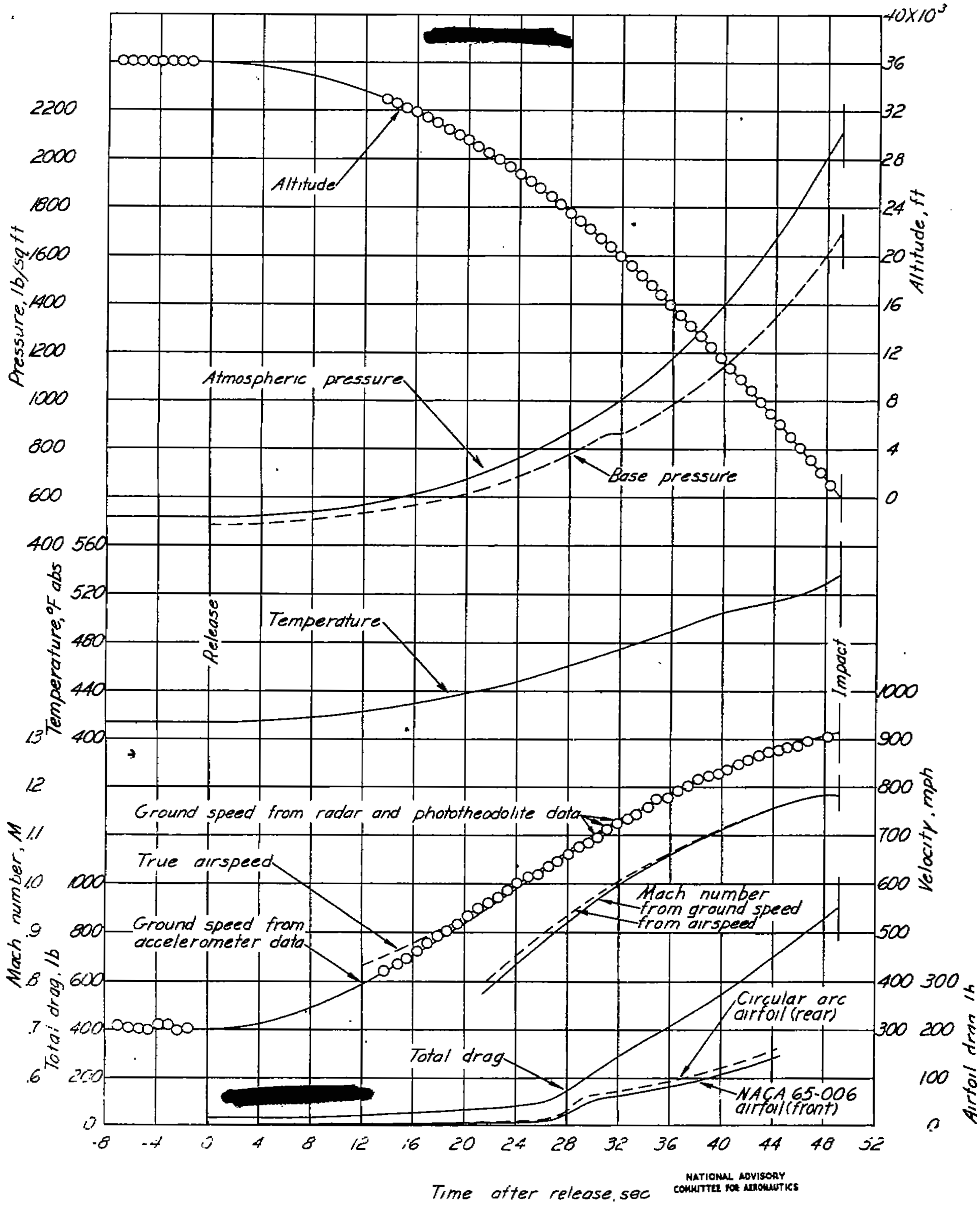

Figure 3.- Time history of important quantities obtained during the free fall of test body.

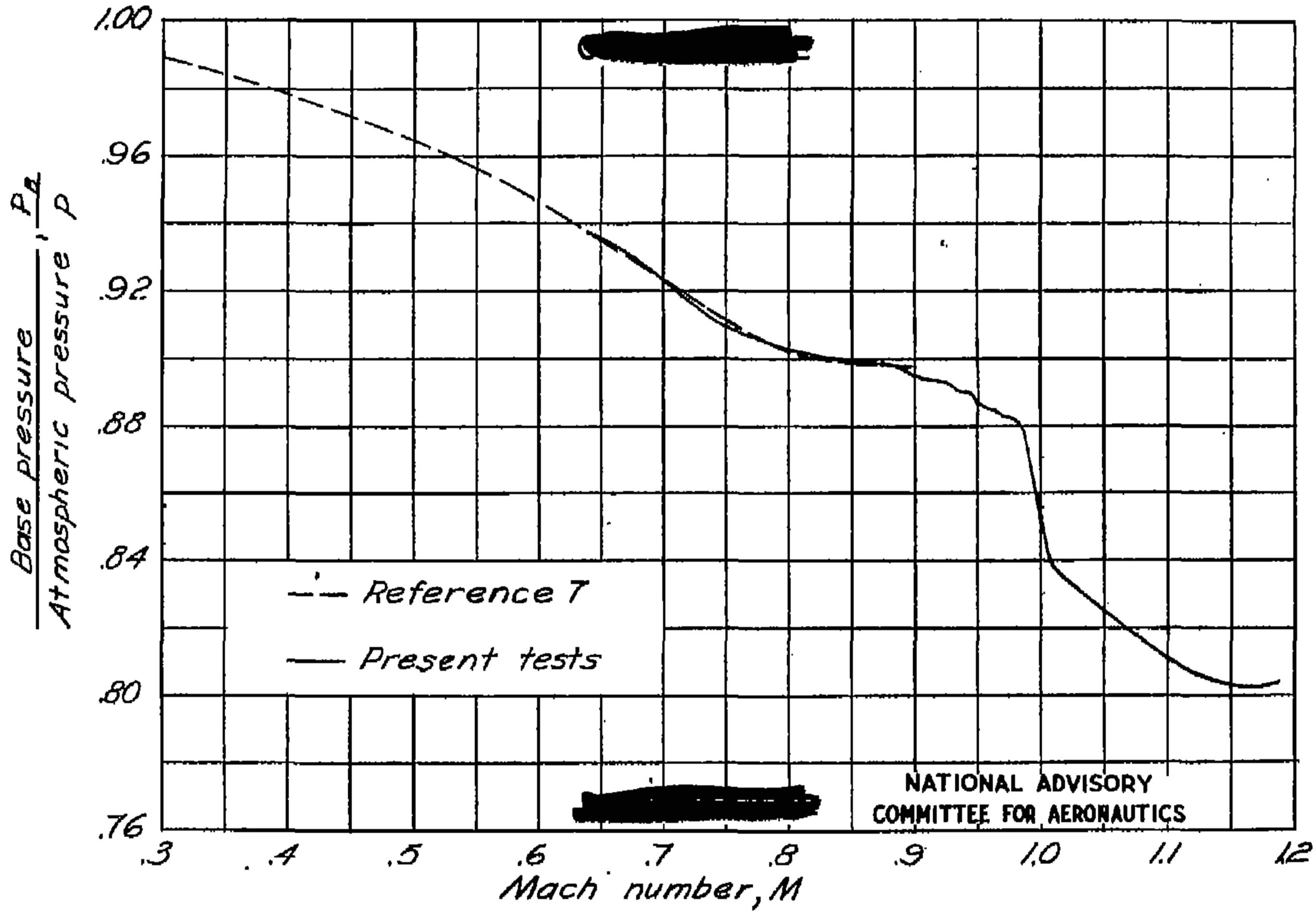

Figure 4.- The variation with Mach number of the ratio of the pressure measured on the flat base of the airfoil test body to atmospheric pressure. Data from reference 7 is shown for comparison.

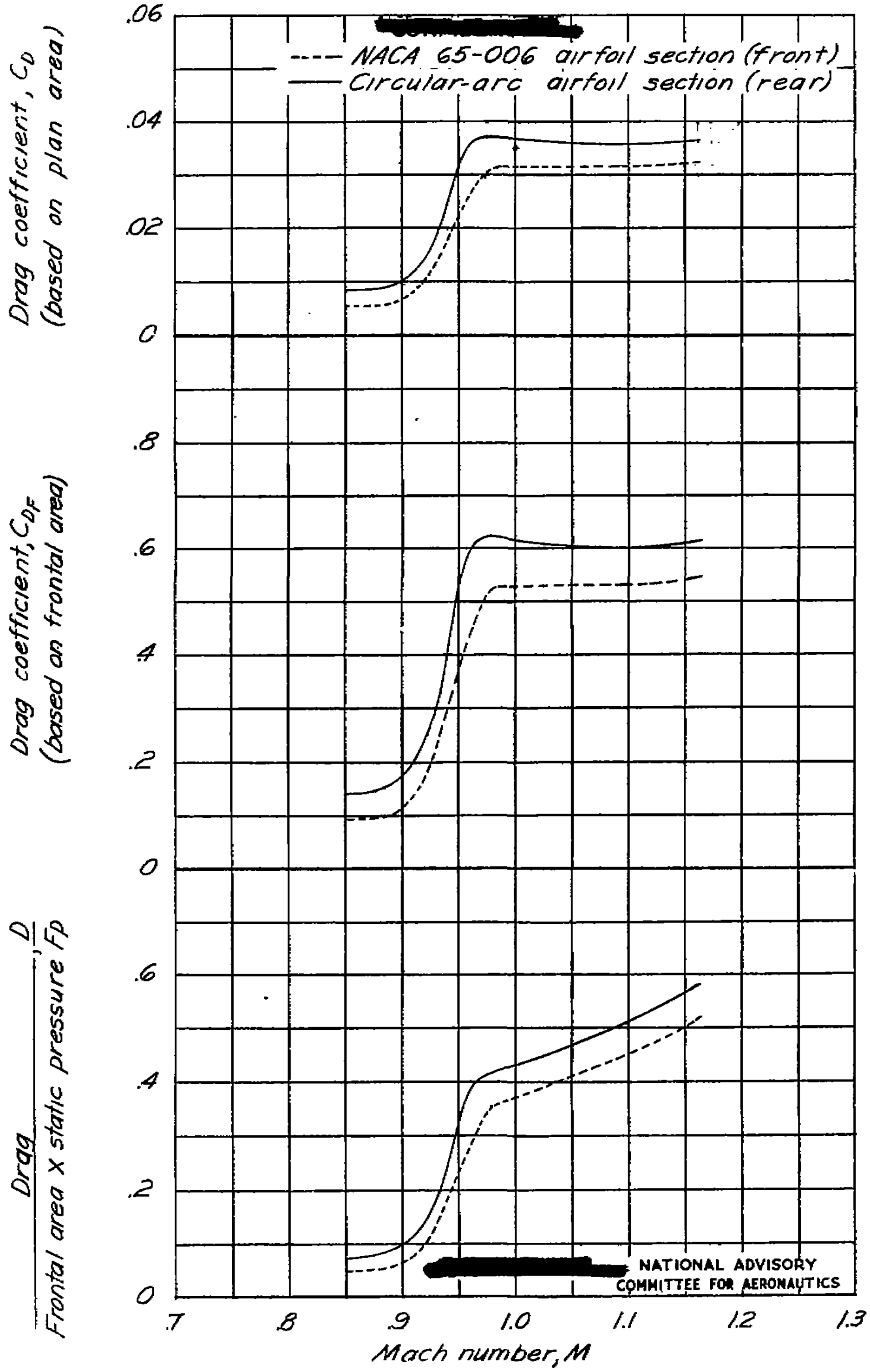

Figure 5.- The measured variation with Mach number of drag coefficients and D/Fp for NACA 65-006 and symmetrical circular-arc airfoils of aspect ratio 7.6.

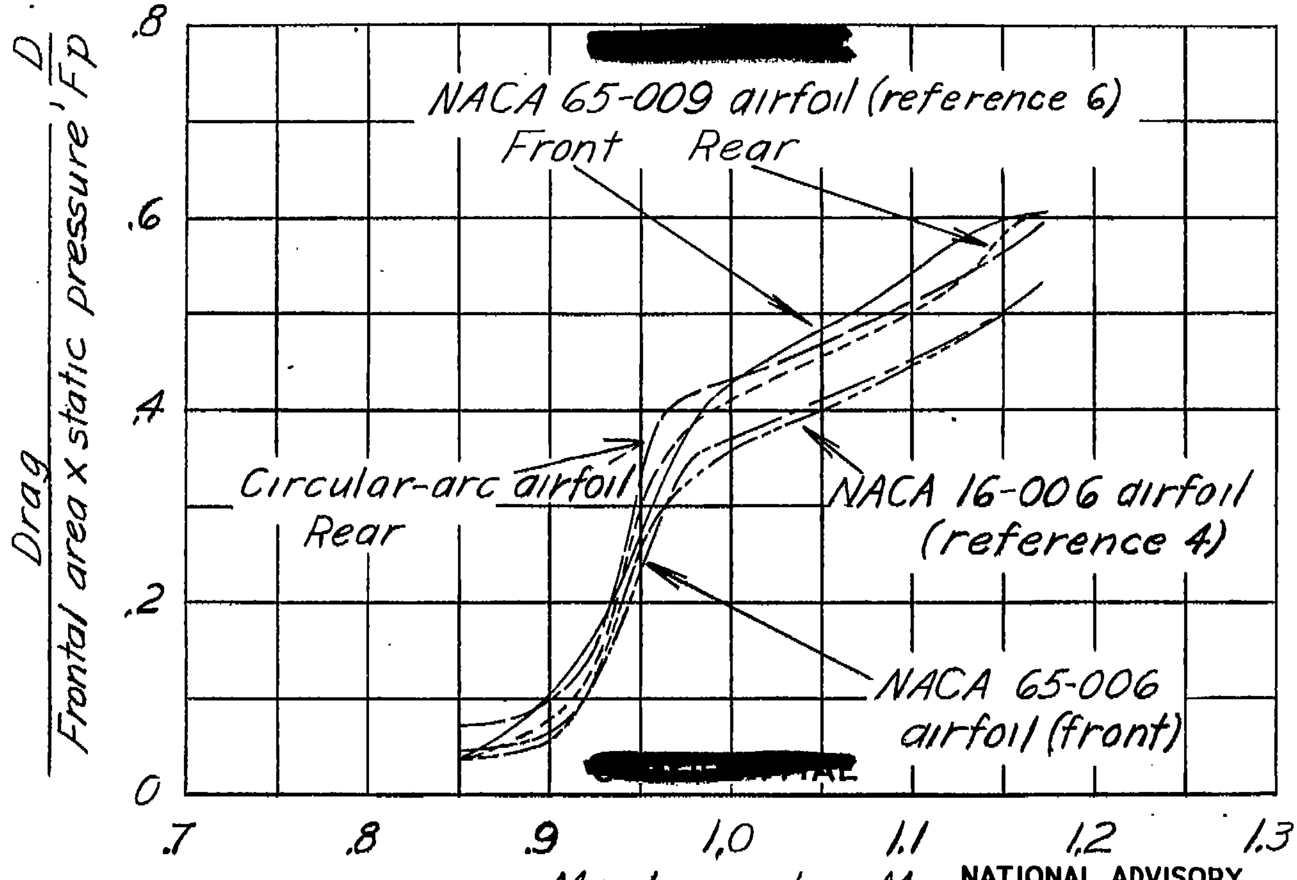

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Figure 6.- Comparison of results with those of previous tests.