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RESEARCH MEMORANDUM

ERROR IN AIRSPEED MEASUREMENT DUE TO STATIC-PRESSURE FIELD AHEAD OF THE WING TIP OF A SWEPT-WING AIRPLANE MODEL AT TRANSONIC SPEEDS

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

March 1, 1951

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SUMMARY

As part of a study of means of airspeed measurement at transonic speeds the use of static orifices located ahead of the wing tip has been investigated for possible application to service or research airspeed installations. The local static pressure and local Mach number have been measured at a distance of 1 tip chord ahead of the wing tip of a model of a swept-wing fighter airplane at true Mach numbers between 0.7 and 1.08 by the NACA wing-flow method. All measurements were made at or near zero lift.

The local Mach number was found to be essentially equal to the true Mach number at true Mach numbers less than about 0.90. The local Mach number was found to be about 0.97 at a true Mach number of 0.95, and to be about 1.04 at a true Mach number of 1.08. The local Mach number provided a reasonably sensitive measure of true Mach number except for a restricted region near a true Mach number of 1.0 where the local Mach number did not change appreciably with true Mach number. The linear theory was found to predict qualitatively the effect of the fuselage on the static pressure ahead of the wing tip but gave a reasonable prediction of the effect of the wing on the static pressure only at Mach numbers below 0.95.

INTRODUCTION

As part of a study of means of airspeed measurement at transonic speeds, the possibility of measuring the static pressure at a point ahead of the wing tip of an airplane has been investigated. For subsonic operation, the static pressure measured at a distance of about 1 tip chord ahead of the wing-tip leading edge has been shown (reference 1) to be subject to only negligible variations from stream static pressure. The continued use of the wing-tip position for static-pressure measurements at transonic speeds may be desirable on airplanes for which the requirements of armament, radar, or propellers prohibit the location of static-pressure orifices ahead of the fuselage nose as discussed in reference 2. Measurements of the static pressure at 1 chord ahead of the wing-tip leading edge of a half model of a swept-wing fighter airplane at or near zero lift have been made by the NACA wing-flow method at Mach numbers between 0.7 and 1.08. The results of this investigation are presented herein to show the magnitude of the static pressure and Mach number error to be expected for this position at transonic speeds, and to interpret where possible the mechanism of the variation of the errors with Mach number in the light of the linearized theory.

APPARATUS

A sketch showing the half model of a swept-wing fighter airplane mounted on the end plate used in the tests is presented as figure 1. A photograph of the model mounted on the wing-flow test panel and alined with the local flow direction is shown as figure 2.

The wing of the model was of aspect ratio 4.5 and taper ratio 0.28 with the quarter-chord line swept back 35°. The airfoil section outboard of the wing-root inlet was an NACA 65-009 in planes normal to the quarter-chord line. The fuselage of the model was of fineness ratio 8.3 with the maximum diameter (excluding cockpit enclosure) located near its midlength position.

A 0.060-inch-diameter static-pressure probe was attached to the wing tip of the model to simulate an airspeed boom, as shown in figures 1 and 2. The orifices in the static-pressure probe were located at 1 tip chord ahead of the wing-tip leading edge and 3.4 fuselage diameters outboard of the fuselage center line directly opposite the position of maximum fuselage diameter. In the detailed sketch of figure 1 the static-pressure probe is seen to be cone-pointed with eight 0.010-inch-diameter orifices located at 1.12 inches (about 19 tube diameters) behind the shoulder of the cone. It is shown in reference 2 that the pressure disturbances arising from the flow about the cone are negligible at a distance of only 5 diameters behind the shoulder of the cone. The static-pressure probe used in the present investigation should be expected, then, to indicate correctly the local static pressure at the position of the orifices. Hence, any difference from free-stream static pressure that is measured can result only from disturbances originating in the flow about the model.

METHOD

Dives were made from high altitude during which the Mach number at the model position varied from about 0.7 to 1.08. Synchronized records were obtained of the static pressure at the orifices in the model airspeed boom and airplane impact and static pressures. Three model configurations were tested: end plate alone, fuselage mounted on the end plate, and the complete model (wing and fuselage) mounted on the end plate.

The true Mach number and true static pressure at the model position are defined as those values measured at the orifices in the model airspeed boom for the case of the end plate alone. For the other two configurations, the true Mach number and the change in static pressure due to the presence of the fuselage or the wing and fuselage were obtained by comparison with the data for the end plate alone at equal values of flight Mach number. Corrections for the effects of airplane lift coefficient on the relation between local Mach number and flight Mach number were not considered necessary, inasmuch as the similarity of the flights resulted in substantially the same airplane lift coefficient at a given flight Mach number.

The static-pressure tube shown to the right of the model in figure 2 was intended to provide a reference Mach number as in reference 3. However, in the subject investigation interference from the model made the indication of the tube unreliable and necessitated the use of the flight Mach number as a reference. This method is considered less accurate than that of reference 3 but should be sufficiently accurate to establish the general magnitude of the errors in Mach number and static pressure caused by the model and the manner in which these errors vary with true Mach number.

It should be kept in mind throughout the discussion that the method of mounting the model with the wing span perpendicular to the airplane-wing surface results in a decrease in stream Mach number at the model of about 0.04 from the root of the wing to the tip. The results of the measurements should, therefore, be expected to differ somewhat from results obtained with a uniform stream, as in flight. The nonuniformity of the stream would be expected to make less abrupt the variation of pressure ahead of the wing with Mach number. In all cases the true Mach numbers quoted are the Mach numbers at the wing-tip position.

RESULTS AND DISCUSSION

The Mach number indicated by the static orifices at 1 chord ahead of the wing-tip leading edge of the fighter model is shown as a function of the true Mach number at that position in figure 3. The error of the indicated Mach number is shown by departure from the line of perfect agreement. It will be noticed that the indicated Mach number is essentially correct at true Mach numbers less than 0.9. For true Mach numbers between 0.9 and 1.0 the indicated Mach number was high by a maximum of 0.015. For true Mach numbers greater than 1.0 the indicated Mach number became steadily less than the true value and reached 1.04 at a true Mach number of 1.08. It should be noted that the indicated Mach number is relatively insensitive to changes in true Mach number near 1.0 as is evidenced by the small slope of the curve in this region.

The data of figure 3 for the complete model have been translated into the variation of pressure coefficient P at the static orifices with true Mach number and are shown in this form by the solid curve in figure 4. The variation of the pressure coefficient (static-pressure error), of course, reflects the variation of the Mach number error; a negative error in static pressure corresponds to a positive error in Mach number.

The pressure coefficient ahead of the wing tip may be thought of as the algebraic sum of the pressure coefficients existing in the flow fields of the wing and fuselage taken separately. The tail surfaces are considered to have a negligible effect on the static pressure ahead of the wing tip. In order to separate the pressure ahead of the wing tip into its major components, the static pressure at this point has been measured with the wing removed to determine directly the pressure coefficient due to the fuselage. The separate pressure coefficient due to the wing was not measured directly but was obtained by subtraction, and thus includes any effect of wing-fuselage interference.

Effect of fuselage.- The geometry of the model configuration was such that the static pressure was measured at 3.4 fuselage diameters outboard of the fuselage center line at a point directly opposite the position of maximum fuselage diameter.

It is to be expected that the fuselage would produce a negative pressure coefficient at this point of its flow field at all subsonic Mach numbers. It is shown in figure 4 that, within the accuracy of measurement, the effect of the fuselage was negligible at Mach numbers below about 0.8. As the Mach number was increased above 0.8, however, the pressure coefficient due to the fuselage became rapidly negative, as the result of the pronounced lateral expansion of the pressure field of the fuselage that is known to take place at high subsonic speeds, and reached a maximum negative value of 0.07 near a Mach number of 1.0.

As the Mach number increased above 1.0, the pressure coefficient became less negative and reached a value of -0.02 at a Mach number of 1.08. This positive increase in pressure coefficient at supersonic speeds may be explained by the simplified description of the flow field provided by the supersonic linearized theory. The pressures at points on the surface of the body are considered to be felt laterally only within the downstream Mach cones from those points on the body. These Mach cones become more and more sweptback as the Mach number is increased. The pressure at the point opposite the maximum thickness position of the fuselage, therefore, becomes influenced more by the positive pressures near the fuselage nose and less by the negative pressures farther to the rear with the result that the pressure coefficient increases positively with Mach number.

The pressure coefficient calculated by the linearized theory (reference 3) at a point in the flow about a sharp-nose body of revolution with parabolic thickness distribution is shown as a function of Mach number in figure 5. The point chosen was in the same relative position with respect to the parabolic body as the point of static-pressure measurement with respect to the model fuselage. The fineness ratio of the parabolic body was equivalent to that of the model fuselage. The type of variation of the pressure coefficient with Mach number is generally similar (fig. 5) for experiment and theory within the range of the test data except, of course, very near a Mach number of 1 where the linear theory predicts an infinite pressure coefficient.

The theoretical results for the simple parabolic body can be used to show qualitatively the variation of the pressure coefficient to be expected beyond the limit of the test data. The computed pressure coefficient of the parabolic body is seen (fig. 5) to increase to a maximum of 0.05 at a Mach number of 1.4 and to decrease smoothly thereafter reaching zero at a Mach number of 1.62 as the Mach line from the body nose passes behind the point at which the pressure was calculated. The smooth decrease of the pressure coefficient to zero would not occur experimentally, inasmuch as a bow shock wave, of which the linearized theory can take no account, will lie ahead of the Mach line from the body nose. The pressure coefficient due to the model fuselage would therefore, in practice, drop abruptly to zero as the bow shock wave passes the point of pressure measurement. The Mach number for the passage of the bow shock wave would be somewhat higher than the value of 1.62 corresponding to the passage of the bow Mach line.

Effect of wing.- The pressure coefficient produced by the wing at a point 1 chord ahead of the wing-tip leading edge is shown as a function of true Mach number in figure 4. The pressure coefficient due to the wing was not measured directly but was determined by subtracting the pressure coefficient due to the fuselage from the pressure coefficient for the complete model and, thus, includes any effect of wing-fuselage interference. It is seen in figure 4 that the pressure coefficient due to the wing was negligible at Mach numbers below about 0.95. Between the Mach numbers of 0.95 and 1.0, however, the pressure coefficient increased rapidly to 0.07, and remained constant at this value, within the accuracy of measurement, to a Mach number of 1.08.

The pressure coefficient has been calculated by the linearized non-lifting wing theory (references 4 and 5) at 1 tip chord ahead of the tip of a wing with the plan form and thickness ratio of the model wing but with double-wedge airfoil sections. The theoretical and experimental pressure coefficients are compared as functions of Mach number in figure 6. The theory is seen to predict a very small positive pressure coefficient at low subsonic speeds which increases slightly with Mach number to 0.011 at a Mach number of 1. The measured pressure coefficient is small and of the order of that predicted by theory at Mach numbers below about 0.95. At Mach numbers above 0.95, however, the differences between theory and experiment become marked. Such disagreement between linear theory and experiment is to be expected at transonic Mach numbers since the occurrence of mixed flow invalidates the assumptions of the theory.

The theoretical pressure is highly positive (infinite) at $M = 1.0$ and decreases to high negative values at slightly supersonic speeds as the result of the loss at the point ahead of the wing tip of the effect of the positive pressures near the wing trailing edge. With further increase in Mach number the pressure ahead of the wing increases to positive values as the effect of the negative pressures behind the maximum thickness line is lost. With still further increase in Mach number the positive pressures ahead of maximum thickness progressively lose their effect ahead of the wing tip and the pressure at that point decreases to zero and remains zero for all higher Mach numbers.

The mechanism of the rapid rise in pressure coefficient due to the wing, which was found experimentally to take place subsonically at Mach numbers between 0.95 and 1.00, must be qualitatively similar to the overall change in the theoretical pressure coefficient occurring supersonically between $M = 1.00$ and $M = 1.08$ although the sequence of the changes differs for experiment and theory. In the actual flow, as the critical Mach number of the airfoil sections is exceeded, a region of supersonic flow followed by shock forms near the maximum-thickness position with the result that at points ahead of the wing the effect of the negative pressures behind maximum thickness and the effect of the positive pressures near the trailing edge are lost simultaneously. The loss in the effect of the negative pressures predominates so that the pressure coefficient ahead of the wing increases positively.

According to the simple theory of infinite-span swept wings, the pressure coefficient ahead of the wing depends not upon the stream Mach number but upon the component of stream Mach number normal to the line of sweep. The predicted section critical Mach number of the NACA 65-009 airfoil is shown in reference 6 to be about 0.75 at a lift coefficient of 0.1. The corresponding critical stream Mach number for a 35° yawed infinite-span wing is, then, about 0.92 which is approximately the Mach number at which the pressure coefficient ahead of the wing began to increase (fig. 6). To generalize, it appears reasonable to expect the pressure coefficient due to a wing at a chord ahead of its tip to be small at low Mach numbers and to increase rapidly as the component of Mach number normal to the line of sweep exceeds appreciably the critical Mach number of the airfoil section.

At Mach numbers above the range of the measurements ($M > 1.08$) it is reasonable to expect a variation of pressure coefficient that is qualitatively similar to the theoretical variation; that is, the pressure coefficient should decrease slowly with Mach number. However, the pressure coefficient due to the wing should not be expected to decrease smoothly to zero at a Mach number of 1.25, as in the case of the theory, but to drop abruptly to zero at some Mach number higher than 1.25 as the oblique bow wave from the wing-root leading edge crosses behind the point of pressure measurement. It is important to remember that, after the passage of the wing bow wave, the point ahead of the wing will still lie in the positive pressure field of the fuselage nose and will continue to do so until after the passage of the fuselage bow wave at a Mach number in excess of 1.6.

Comparison with flight-test results.- The variation of indicated Mach number with true Mach number obtained from the wing-flow tests is compared in figure 7 with results obtained in flight tests of a similar airplane (reference 7 and unpublished data). The comparison is not direct. The airplane used for comparison was only generally similar to the wing-flow model insofar as wing geometry and the position of the wing with respect to the fuselage were concerned. The geometry of the fuselages of the model and airplane differed considerably. The main point of interest in figure 7 is that the variations of indicated Mach number with true Mach number obtained in flight and in the wing-flow tests are similar. The flight tests, as in the case of the wing-flow tests, show that the indicated Mach number becomes relatively insensitive to changes in true Mach number for true Mach numbers near 1.0.

CONCLUSIONS

Studies made by the NACA wing-flow method at Mach numbers between 0.7 and 1.08 of the measurement of airspeed by the use of static-pressure orifices located 1 tip chord ahead of the wing tip of a model of a swept-wing fighter airplane near zero lift have indicated that:

1. The local Mach number at the position ahead of the wing was essentially equal to the true stream Mach number for true Mach numbers less than about 0.90, became a maximum of 0.015 high at a true Mach number of 0.95, and became less than true Mach number with further increase in speed, reaching 1.04 at a true Mach number of 1.08.

2. The local Mach number ahead of the wing provided a sufficiently sensitive measure of the true Mach number except for a restricted region near a true Mach number of 1.0 where the local Mach number did not change with true Mach number.

3. The part of the pressure coefficient due to the fuselage was negligible at true Mach numbers less than about 0.8, reached a value of -0.07 near a Mach number of 1.0, and increased positively thereafter becoming -0.02 at a Mach number of 1.08.

4. The effect of the wing on the pressure ahead of the wing tip was negligible at Mach numbers below about 0.95. Between the Mach numbers of 0.95 and 1.0 the pressure coefficient due to the wing increased abruptly to 0.07 and remained at this value to a Mach number of 1.08, the limit of the tests.

5. The effect of the fuselage on the static pressure ahead of the wing tip was predicted qualitatively by the linear theory. However, the linear theory gave a reasonable prediction for the effect of the wing on the static pressure ahead of the wing tip only at Mach numbers below 0.95.

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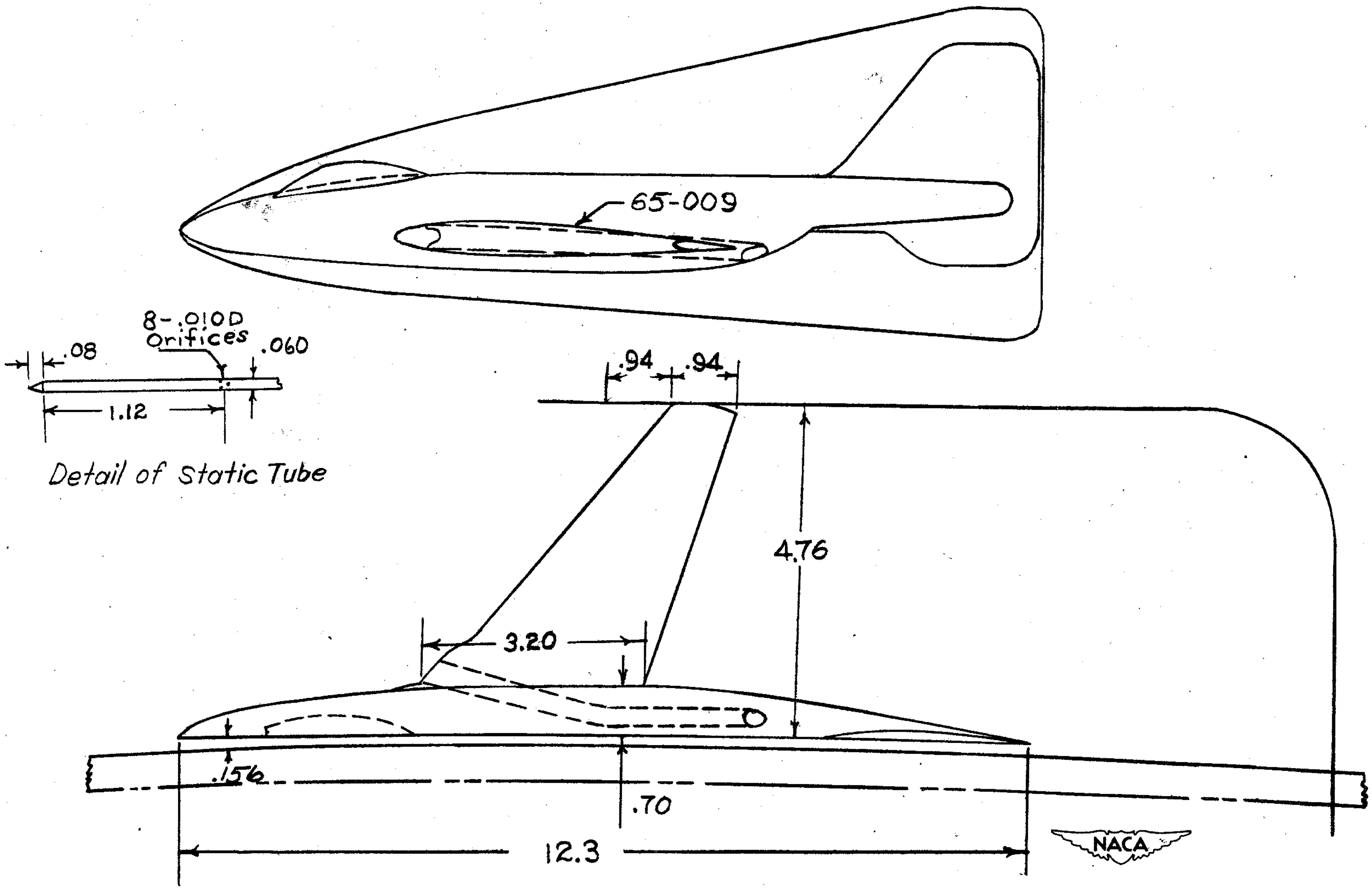

Figure 1.- Half model of swept-wing fighter airplane showing location of static orifices on wing-tip boom. All dimensions are in inches.

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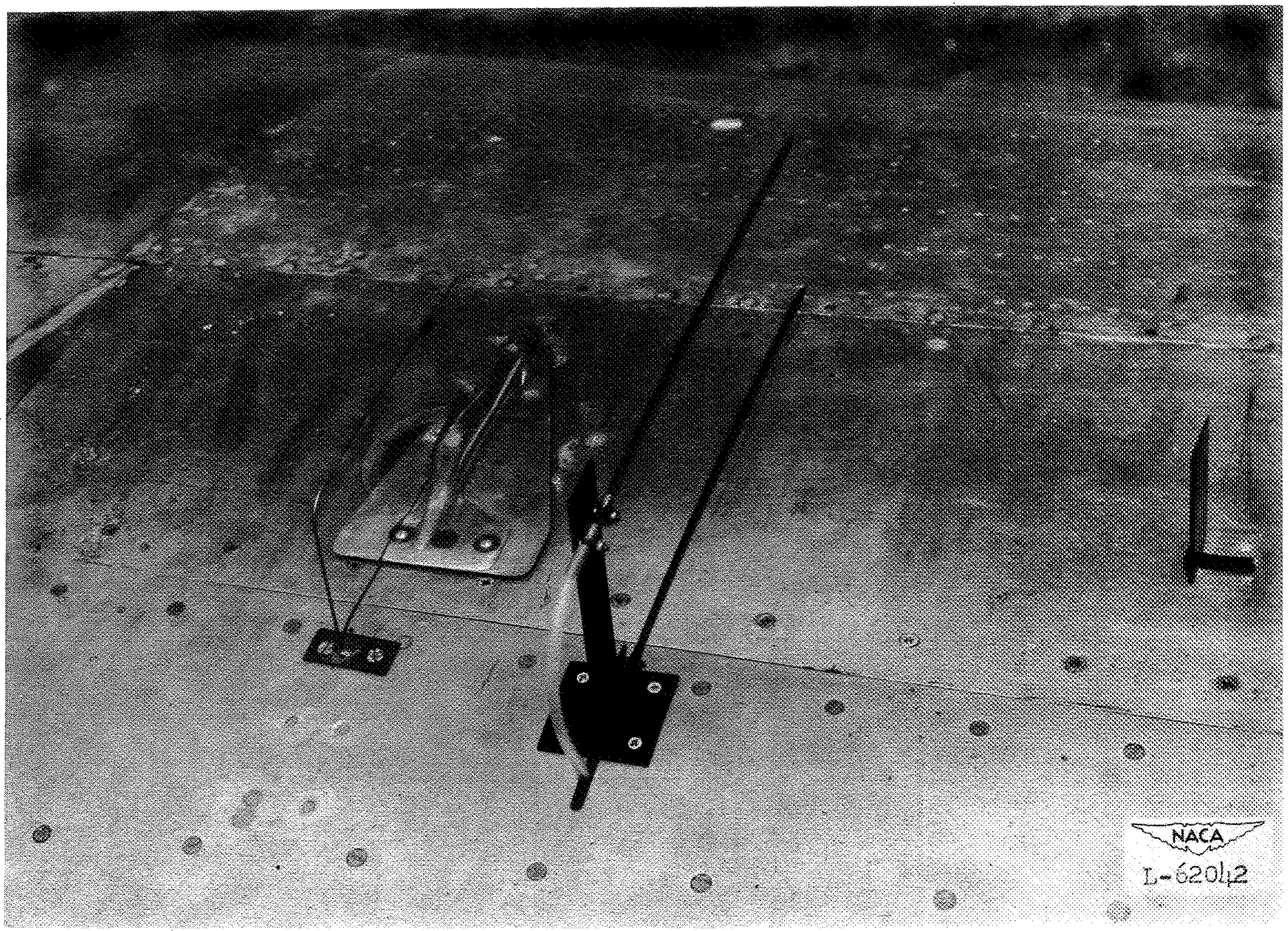

Figure 2.- Half model of swept-wing fighter airplane mounted on test panel.

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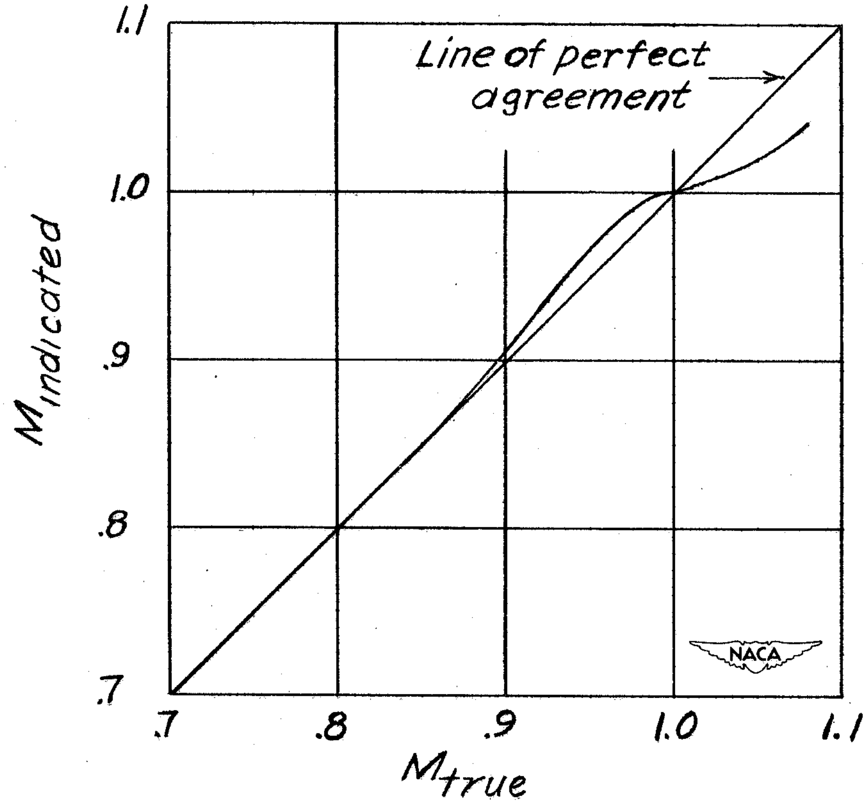

Figure 3.- Indicated Mach number at wing-tip boom on model fighter airplane.

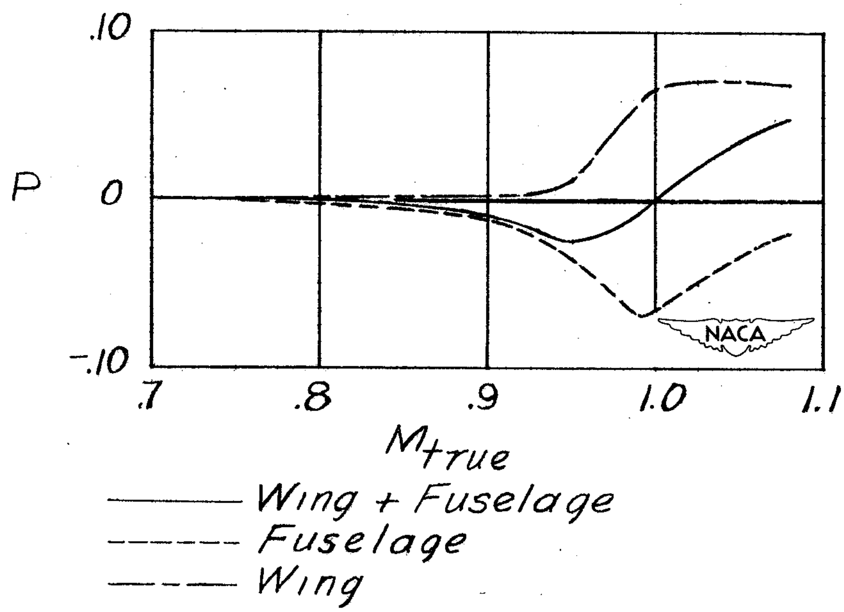

Figure 4.- Static-pressure coefficient at wing-tip boom for wing plus fuselage, fuselage alone, and wing alone.

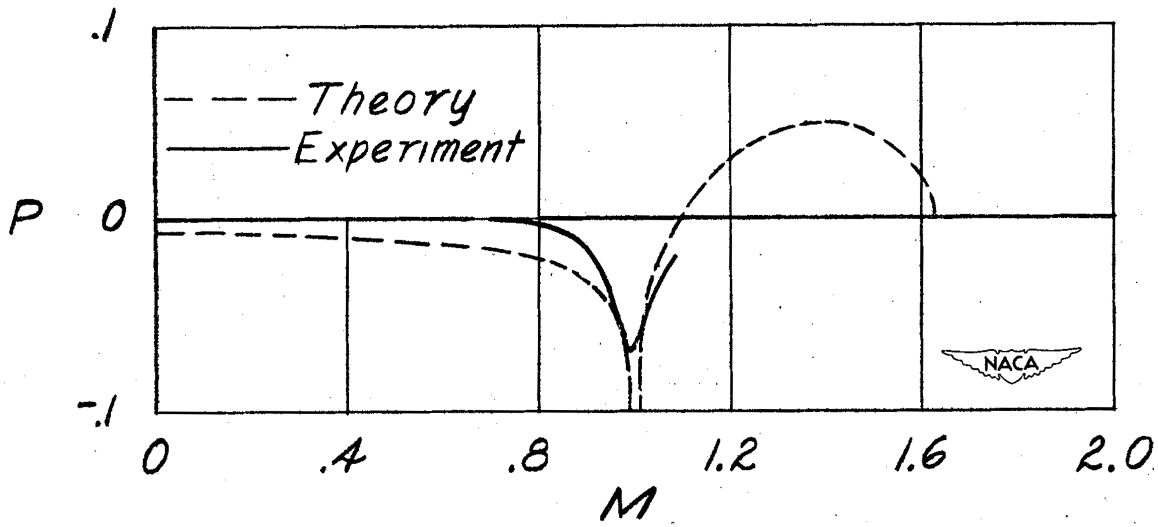

Figure 5.- Comparison of experimentally obtained variation with Mach number of the pressure coefficient due to the fuselage with results obtained by the linearized theory.

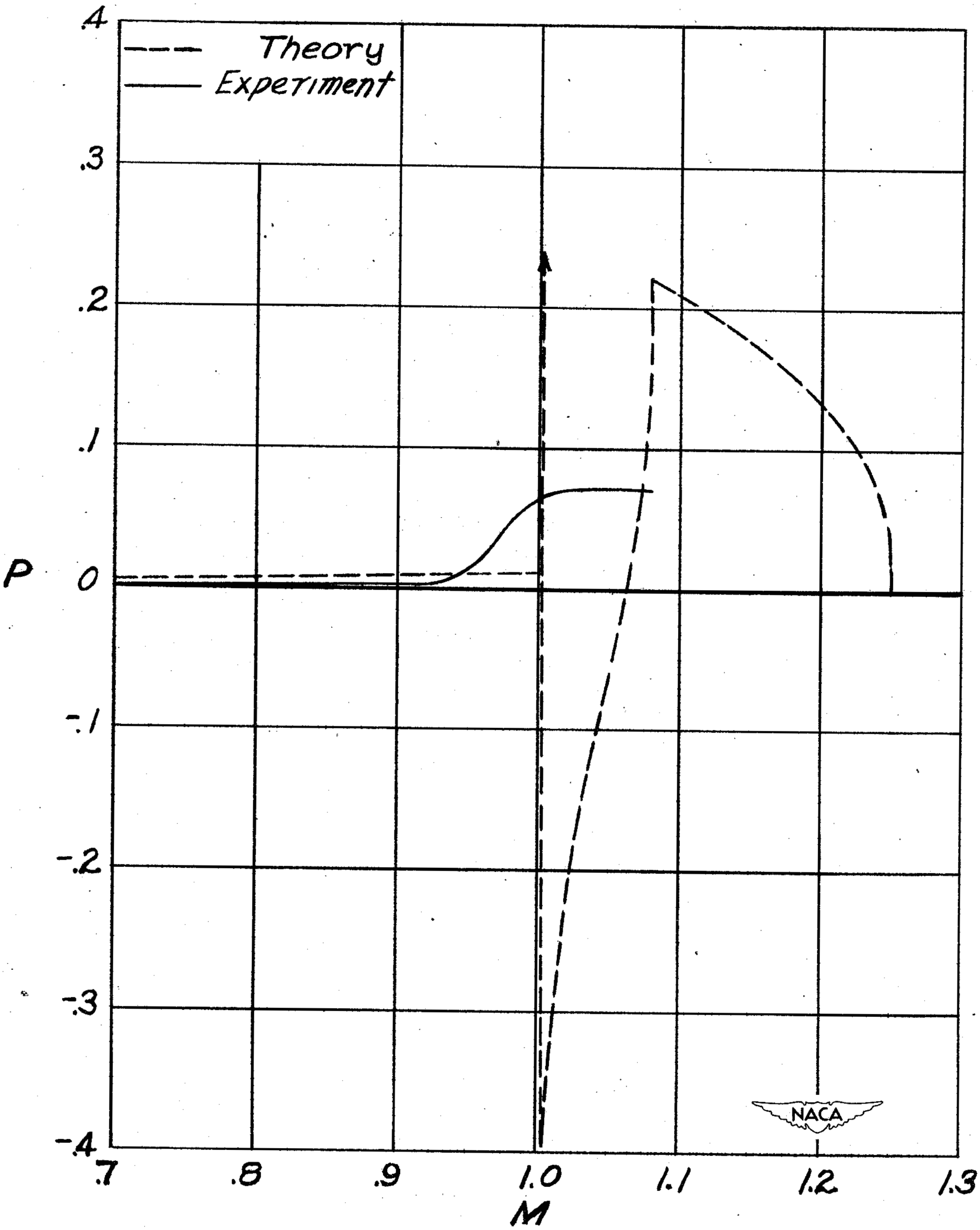

Figure 6.- Comparison of experimentally obtained variation with Mach number of the pressure coefficient due to the wing with results obtained by the linearized theory.

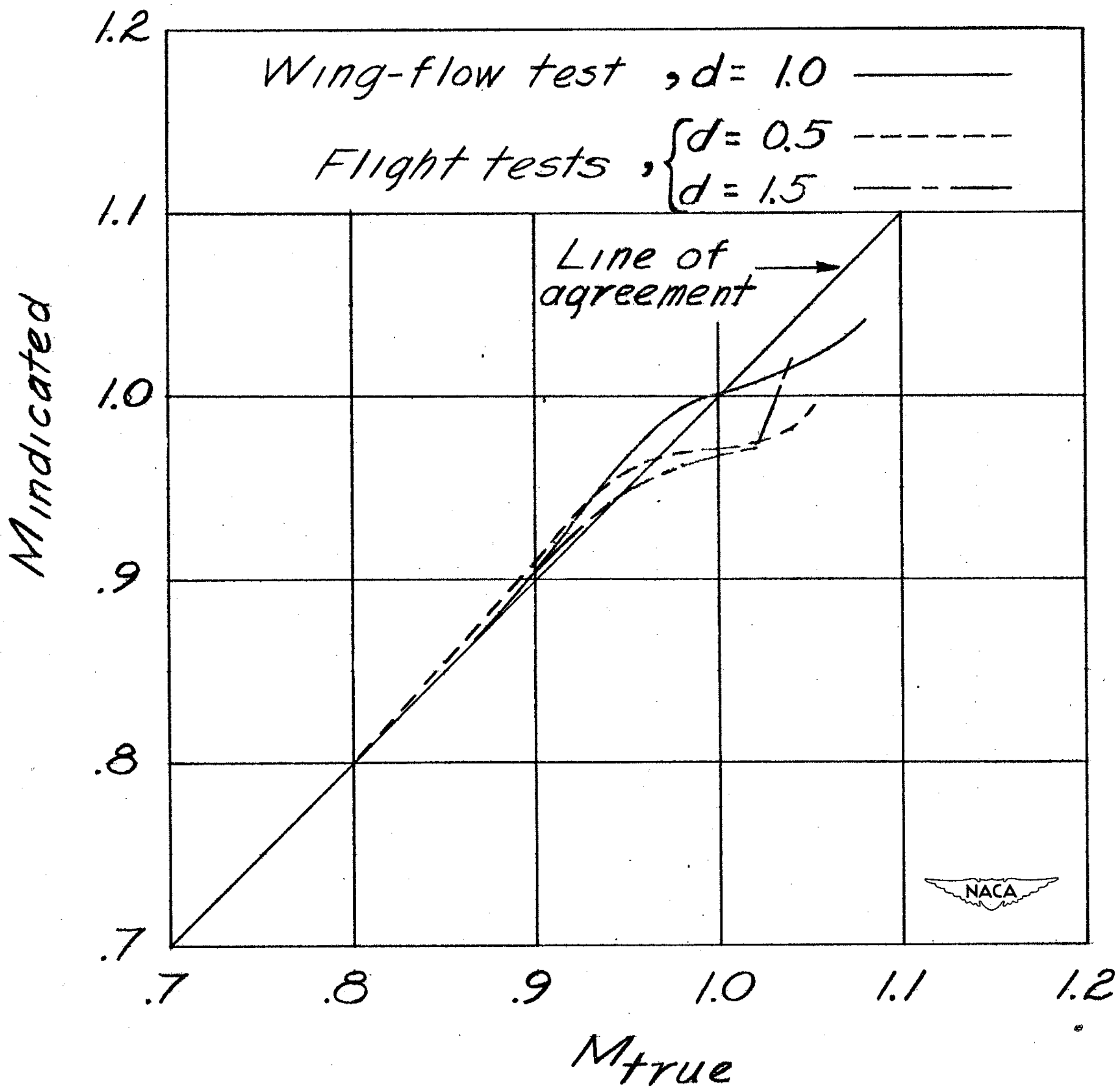

Figure 7.- Variation of indicated Mach number with true Mach number from tests of wing-flow model and flight tests of similar airplane (d = Distance of orifices ahead of wing tip expressed in wing-tip chords).

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