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RESEARCH MEMORANDUM

THE LANGLEY ANNULAR TRANSONIC TUNNEL AND PRELIMINARY TESTS OF AN NACA 66-006 AIRFOIL

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RESEARCH MEMORANDUM

THE LANGLEY ANNULAR TRANSONIC TUNNEL AND PRELIMINARY TESTS OF AN NACA 66-006 AIRFOIL

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SUMMARY

The Langley annular transonic tunnel is essentially an annulus in which an airfoil may be rotated at any Mach number between approximately 0.6 and approximately 1.4. The tunnel was designed to obtain two-dimensional airfoil pressure-distribution data, hence the model very nearly spans the annulus. A mercury-seal pressure-transfer device is employed to transmit the pressures from the rotating model to a stationary manometer.

Additional research is needed in the Langley annular transonic tunnel to determine its limitations. However, preliminary pressure-distribution data obtained for an NACA 66-006 airfoil at a Mach number of 0.75 in the Langley annular transonic tunnel compare well with data obtained in the Langley rectangular high-speed tunnel at a Mach number of 0.75 for a corresponding airfoil section. The local supersonic velocities obtained over the surfaces of the NACA 66-006 airfoil at a test Mach number of unity are shown to be smaller than the local velocities predicted by the Prandtl-Meyer expansion theory. However, the points of maximum velocity as well as the general shapes of the velocity curves are in good agreement for experiment and theory.

The pressure-drag coefficient of the NACA 66-006 airfoil was determined over a range of Mach numbers from approximately 0.65 to 1.00. These data are compared with the total drag coefficient of an NACA 65-006 airfoil which was tested by the freely-falling-body method between Mach numbers of 0.85 and 1.16. At a Mach number of unity, the pressure-drag coefficient of the NACA 66-006 airfoil was found to be approximately 0.03 in the Langley annular transonic tunnel. The freely-falling-body method indicated that at a Mach number of unity the total drag coefficient of an NACA 65-006 airfoil was approximately 0.0315.

INTRODUCTION

Conventional subsonic wind tunnels have been used to produce reliable aerodynamic data up to Mach numbers of approximately 0.96 when special techniques were used and the ratio of tunnel height to model thickness was large (reference 1). Conventional supersonic wind tunnels have been operated at Mach numbers as low as approximately 1.2 (reference 2).

However, in order to obtain reliable aerodynamic data at low supersonic Mach numbers in a conventional supersonic tunnel, not only must the ratio of tunnel height to model thickness be large, but in addition, the model length must be small enough so that shock waves reflected from the tunnel walls will not interfere with the flow over the model. For present-day application, probably the most important Mach number range and certainly the Mach number range about which the least aerodynamic information is available, lies between the upper Mach number limit of the subsonic wind tunnel and the lower Mach number limit of the supersonic wind tunnel. In order to obtain aerodynamic data within this range, numerous methods have been devised, among which are the wing-flow method (reference 3), the free-flight methods (references 4 and 5), and methods whereby a model is rotated at high speeds (reference 6). The purpose of the present paper is to describe the Langley annular transonic tunnel which was designed to obtain two-dimensional airfoil pressure-distribution data at any Mach number from approximately 0.6 to approximately 1.4, to present some preliminary data obtained in the Langley annular transonic tunnel for an NACA 66-006 airfoil, and to discuss briefly some of the limitations of the subject tunnel.

APPARATUS

General arrangement.- A schematic diagram of the Langley annular transonic tunnel is presented in figure 1. Two concentric circular cylinders are arranged with a 3-inch annulus between them. In this annulus a model airfoil can be rotated at any speed from 1000 to 5900 rpm. The center-span station of the airfoil rotates at a radius of 2.5 feet, thus producing velocities up to 1545 feet per second which corresponds to a Mach number of approximately 1.4. The minimum Mach number at which data are obtained has arbitrarily been chosen as approximately 0.6, thus the minimum test rotational speed is approximately 2700 rpm. The flow over the airfoil test section is believed to approach two-dimensional flow as the model very nearly spans the annulus. The choking effects encountered in conventional wind tunnels are believed to be eliminated because the ratio of tunnel height to model thickness is practically infinite. A photograph showing the relative size of the apparatus is presented as figure 2.

Angle-of-attack control.- In order to control the angle of attack of the airfoil and to prevent the model from operating in its own wake, an axial velocity which may be continuously varied from approximately 45 feet per second to approximately 300 feet per second is induced through the annulus. The model test velocity is equal to the vector sum of the rotational velocity and the axial velocity. As the model chord line at the center-span station makes an angle of 5° with the plane of rotation of the rotor, the angle between the rotational- and the test-velocity vectors (the helix angle) is 5° for an angle of attack of 0°. For this condition at a Mach number of unity, the axial velocity is of the order of 100 feet per second.

For tests of symmetrical airfoils, the choice of upper or lower surface is optional. If the choice is made so that helix angles less than 5° are considered to produce positive angles of attack, the approximate angle-of-attack range is from -18° to 1.3° at a Mach number of 0.6. At a Mach number of 1.4 the approximate angle-of-attack range is from -6° to 3.3°. It is believed possible to extend the positive angle-of-attack range somewhat, without changing the negative angle-of-attack range, by reducing the blade angle of the blower used to produce the axial velocity. The airfoil models are twisted so that when the center-span station of the airfoil is operating at an angle of attack of 0°, all other spanwise stations are operating at an angle of attack of 0°. Obviously, the amount of twist can be correct for only one angle of attack. However, when the center-span station is operating at an angle of attack of -5° (or 5° if the choice of upper and lower surfaces of the airfoil is reversed) the angles of attack of the tip and root sections are within $\frac{1}{4}$° of the center-span station angle of attack, providing, of course, that the axial velocity profile is flat at the test section.

Axial boundary-layer control.- Removal of some of the axial boundary layer is desirable so that the spanwise variation in angle of attack of the model is small. As shown in figure 1, three boundary-layer removal stations upstream of the test section are employed for this purpose. Air entering the slots of the inner cylinder flows through a duct system to a blower which exhausts to atmosphere. Air entering the slots in the outer cylinder flows through auxiliary ducts to the main boundary-layer removal duct. With this arrangement, at an axial velocity of approximately 250 feet per second, it is possible to remove up to 42 percent of the axial boundary layer that is present at the test section when boundary-layer control is not employed. In figure 3, the effect of axial boundary-layer control on spanwise angle-of-attack variation is shown. The curve presented for maximum axial boundary-layer control was determined from a velocity survey made across the 3-inch annulus at an axial velocity of approximately 250 feet per second. For an angle of attack of 0° at an airfoil center-span station at a Mach number of approximately 1.4 the axial velocity would be only 140 feet per second. Hence, for small angles of attack at the airfoil center-span station, the spanwise variation in angle of attack may be slightly less than indicated in figure 3.

Model airfoils.- The model for which data are presented in this paper is an NACA 66-006 airfoil with a span of approximately 3 inches and a chord of 4 inches. The model is equipped with 24 pressure orifices. A photograph showing an NACA 66-006 airfoil mounted on the Langley annular transonic tunnel rotor is presented as figure 4. The chord line at the center-span station of the airfoil in the photograph makes an angle of 15° with the plane of rotation of the rotor, while as previously stated, the corresponding angle of the model for which data are presented in this paper is 5°. The photograph gives a distorted view of the arrangement because the camera could not be placed nearer the inner cylinder. The annulus, however, as previously pointed out, is 3 inches wide. The clearance between the tip of the airfoil and the outer wall is believed to be

approximately 0.010 inch during operation. The pressure orifices may be seen in the model at the center-span station. The diagonal light lines shown in the model are solder which was used to make a smooth surface over the stainless-steel tubes in the airfoil.

Pressure-transfer device.- A photograph of the interior of a pressure-transfer device similar to the one connected to the hub of the Langley annular transonic-tunnel rotor is presented as figure 5. Small-diameter stainless-steel tubes, which are connected to the tubes leading from the model orifices, were installed in axial slots cut in the shaft of the transfer device. Each tube in the shaft is open to an individual space between the rotor disks shown in figure 5. The projections on the upper half of the stator fit between the disks and match similar projections on the lower half of the stator. Mercury in the device rotates with the disks, thus forming mercury-sealed cells between the disks. The pressures are transmitted from the rotating shaft through the cells to holes drilled in the projections on the stator. The pressures are led from these holes to a multiple-tube manometer for visual observation and photographic recording. A more complete description of the pressure-transfer device is presented in reference 7.

TEST PROCEDURE AND REDUCTION OF DATA

The testing procedure consisted of setting the rotational speed of the rotor at predetermined values and then adjusting the axial velocity to a value determined from the rotational speed of the rotor and the desired angle of attack of the model airfoil. Pressure distributions over the airfoil surfaces were recorded by photographing a manometer which was connected to the pressure-transfer device. The recorded pressures can be corrected for the effect of centrifugal force by the following relation:

$$\frac{p}{p_m} = e^{\frac{2\pi^2 n^2}{gRT}\left(r_o^2 - r_s^2\right)} \tag{1}$$

where

p true local absolute static pressure at model airfoil orifice, pounds per square foot

p_m absolute static pressure indicated by manometer, pounds per square foot

e base of empirical system of logarithms

n rotational speed of the rotor, revolutions per second

g acceleration of gravity, feet per second per second

R gas constant for air, foot pounds per pound per °F

T mean temperature of air in rotor tubing, °F absolute

r_o radius to orifice in model airfoil, feet

r_s radius to orifice in rotating shaft of pressure-transfer device, feet

In order to obtain the temperature of the air in the rotor tubing, a calibrated temperature gage was installed on the tubes inside the rotor. However, the temperature gage failed before the data presented in this paper were obtained. The corrections for the effect of centrifugal force on the air in the rotor tubing were therefore determined for these data by computing the stagnation pressure for each velocity at which data were obtained, and then assuming that the pressure orifice installed in the leading edge of the airfoil should record stagnation pressure when the airfoil was at an angle of attack of 0°. The exponential term in equation (1) was then equal to $\frac{p_h}{(p_h)_m}$ where p_h is the computed stagnation pressure and $(p_h)_m$ is the absolute pressure indicated by the manometer tube reading for the orifice installed in the airfoil leading edge. Equation (1) may then be rewritten

$$\frac{p}{p_m} = \frac{p_h}{(p_h)_m} = e^{\frac{2\pi^2 n^2}{gRT}\left(r_o^2 - r_s^2\right)} \tag{2}$$

Recently a new calibrated temperature gage was installed on the rotor tubing. The new gage was designed to obtain the average temperature along the tubes and is approximately 20 inches long. It is believed that the reasons for the previous gage failure have been eliminated, hence for future tests it should be possible to compare the two previously described methods of correcting the pressures for the effect of centrifugal force.

The magnitude of the test velocity V was determined from the following relation:

$$V = \sqrt{V_r^2 + V_a^2}$$

where

V_r velocity of airfoil at center-span section due to rotation, feet per second

V_a axial velocity at the test section, feet per second

The rotational velocity of the rotor is determined by comparing the frequency output of a small generator driven by the rotor shaft with a

known frequency. Standard Lissajous patterns are obtained on the screen of a cathode-ray oscilloscope. A pitot-static tube is mounted in the annulus slightly upstream of the test section to determine the axial velocity.

As the airfoil chord line at the center-span station makes an angle of 5° with the plane of rotation of the rotor, the airfoil angle of attack α was determined from the relation

$$\alpha = 5^{\circ} - \phi$$

where

ϕ helix angle of the flow $\left(\phi = \tan^{-1} \frac{V_a}{V_r}\right)$

Limitations of the equipment.- There are several possible sources of error which might influence the data obtained during tests of airfoils in the Langley annular transonic tunnel. Although these effects are believed to influence the results only slightly, it is necessary that they be investigated. Among the investigations believed necessary are the following:

1. A study of the effect of centrifugal force on the airfoil boundary layer and on the disturbed flow which occurs behind a normal shock on the airfoil.

2. A determination of the interference on the model caused by disturbances set up during the preceding revolution of the model. As the circumference of the path on which the model rotates is approximately 15.7 feet and the model chord line at the airfoil center-span station makes an angle of attack of 5° with the plane of rotation of the rotor, the system can be thought of as a cascade with a gap of approximately 47 chords and a stagger angle of 85° (for an airfoil angle of attack of 0°). It is believed that information concerning these interference effects can be obtained by testing the airfoil with an additional model mounted on the rotor 180° from the original model in a space provided for balance weights, thus cutting the cascade gap in half. The interference effects of the skin-friction wake are believed negligible because at an airfoil angle of attack of 0° the wake is moved approximately 4.1 chords downstream during one revolution of the rotor. At supersonic velocities, however, shock waves from the airfoil surfaces theoretically extend to infinity and the air at the test section may be whirled by the rotating shock waves. A survey made without a model in the tunnel indicated that the direction of the velocity at the test section made an angle of 90° to the plane of rotation of the rotor. A similar survey made while the model is rotating should indicate if the velocity remains axial.

3. A study of the effects of spanwise Mach number variation. When the model is rotated, the tip section of the model operates at a velocity

approximately 5 percent greater than the center-span station velocity. The root section operates at a velocity approximately 5 percent smaller than the center-span station velocity. During tests, this spanwise Mach number gradient produces higher positive and negative pressures over the outboard sections of the airfoil than those obtained over sections farther inboard. In the positive pressure region the centrifugal forces tend to appose the pressure gradient, hence, little or no spanwise flow would be expected. In the negative pressure region, however, the centrifugal forces and the pressure gradient may combine to produce an outward spanwise flow.

RESULTS AND DISCUSSION

Pressure-distribution data obtained for an NACA 66-006 airfoil at a Mach number of 0.75 in the Langley annular transonic tunnel are presented in figure 6. The ordinate p/p_h is the local static pressure divided by the stagnation pressure and is plotted against stations along the airfoil chord. The data obtained in the Langley annular transonic tunnel are shown to agree well with data presented for an NACA 66-006 airfoil tested in the Langley rectangular high-speed tunnel (reference 8).

The airfoil angle of attack α at which the rectangular high-speed tunnel data were obtained was 0.2°. The angle of attack of the airfoil center-span station for the Langley annular transonic-tunnel tests was 0° but the axial boundary-layer control system was not in use for these preliminary tests. The lack of axial boundary-layer control resulted in a larger spanwise variation of angle of attack than would be obtained had the axial boundary-layer control system been used. However, as may be seen in figure 3, the center 50 percent of the airfoil span was operating at an angle of attack of 0.2° or less.

The pressure distribution over the NACA 66-006 airfoil at a Mach number of 1.00 as obtained in the Langley annular transonic tunnel with the center-span station of the airfoil operating at an angle of attack of 0° is presented in figure 7. The flow over the airfoil reattained sonic velocity at the 18-percent chord station and continued to increase in velocity to approximately the 85-percent chord station. From the 85-percent chord station to the trailing edge the flow decreased in velocity because the NACA 66-006 airfoil is cusped. Shock probably occurred off the airfoil trailing edge. To make an approximate comparison between experiment and theory, the Prandtl-Meyer expansion (described in reference 9) was computed from the 18-percent chord station. It may be seen that the general shapes of the curves as well as the locations of the points of maximum velocity are in good agreement. The reasons for the difference in magnitude of velocities have not been definitely determined. It is interesting to note, however, that the difference is in the same

direction as would be obtained if expansion waves originating at the airfoil surfaces were reflected from a sonic boundary line as compression waves which upon returning to the airfoil surfaces would reduce the local velocities to values lower than those predicted by the Prandtl-Meyer expansion theory in which an infinite supersonic flow field is assumed.

Figure 8 presents the pressure-drag coefficient of the NACA 66-006 airfoil as determined from tests in the Langley annular transonic tunnel at Mach numbers from approximately 0.65 to 1.00. At a Mach number of unity a pressure-drag coefficient of 0.03 was found. Also presented in figure 8 is the total drag coefficient of an NACA 65-006 airfoil which was tested by the freely-falling-body method over a Mach number range from approximately 0.85 to 1.16 (reference 4). At a Mach number of unity the total drag coefficient of the NACA 65-006 airfoil was found to be 0.0315. Ordinarily, one might expect the total drag coefficient of the NACA 65-006 airfoil to be appreciably higher than the pressure-drag coefficient of the NACA 66-006 airfoil which does not include the skin-friction drag coefficient. However, the NACA 65-006 airfoil tested by the freely-falling-body method had a finite aspect ratio of 7.6 which would cause the total drag coefficient to be lower than would be obtained if the airfoil was tested with the infinite aspect ratio approximated in the Langley annular transonic-tunnel tests.

CONCLUDING REMARKS

The Langley annular transonic tunnel was designed to obtain two-dimensional airfoil pressure-distribution data at any Mach number between approximately 0.6 and approximately 1.4. Although the data obtained in the Langley annular transonic tunnel for an NACA 66-006 airfoil at a Mach number of 0.75 agree well with data obtained in a conventional subsonic wind tunnel for a corresponding airfoil section at a Mach number of 0.75, additional research is needed in the Langley annular transonic tunnel to determine its limitations.

It is shown that at a Mach number of unity the Prandtl-Meyer expansion theory predicts higher local supersonic velocities over the NACA 66-006 airfoil than were experimentally realized. However, the experimental point of maximum velocity as well as the general shape of the experimentally determined supersonic portion of the velocity curve was accurately predicted by the Prandtl-Meyer relation.

At a Mach number of unity, reasonable agreement was obtained between the pressure-drag coefficient of the NACA 66-006 airfoil as determined by Langley annular transonic-tunnel tests and the total drag coefficient of an NACA 66-006 airfoil as found by the freely-falling-body method.

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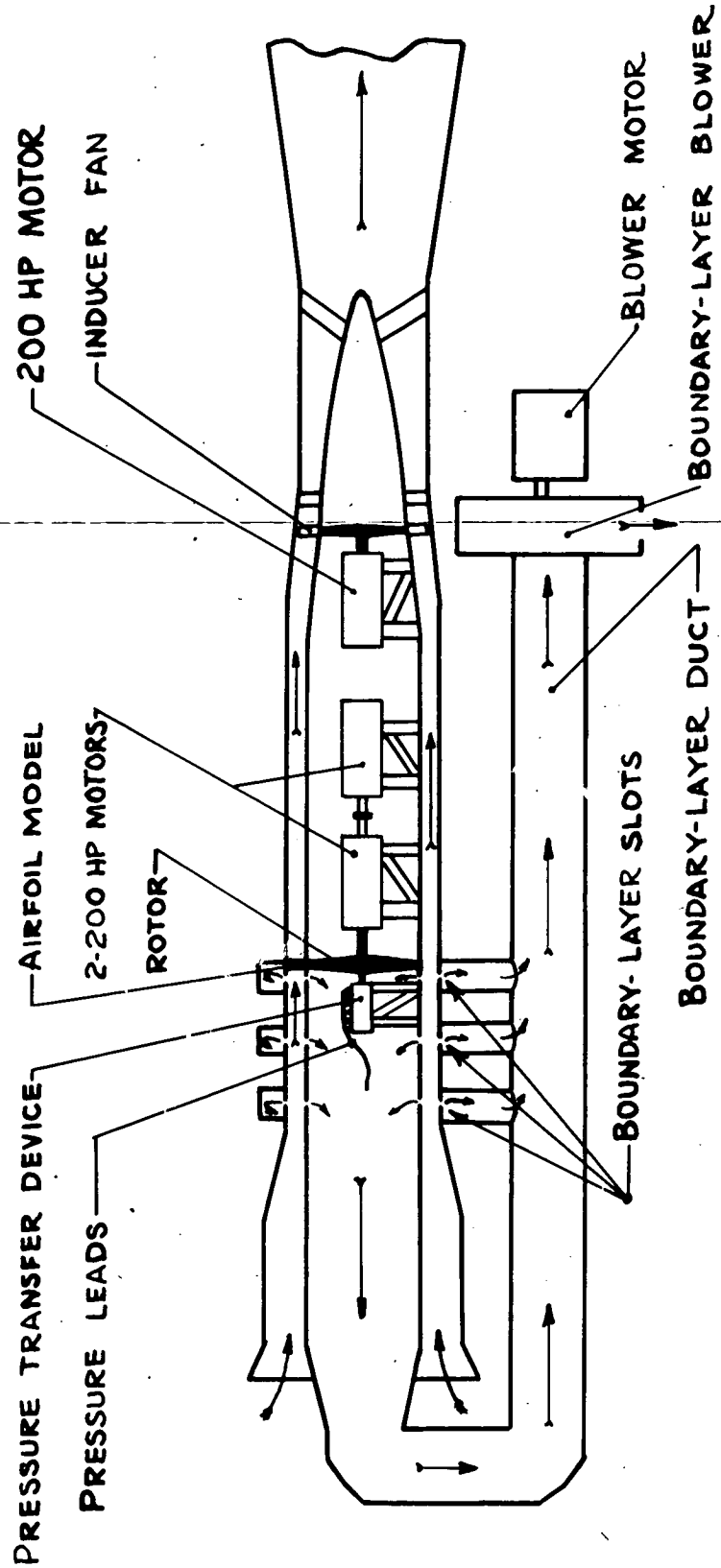

FIGURE 1.- SCHEMATIC DIAGRAM OF LANGLEY ANNULAR TRANSONIC TUNNEL.

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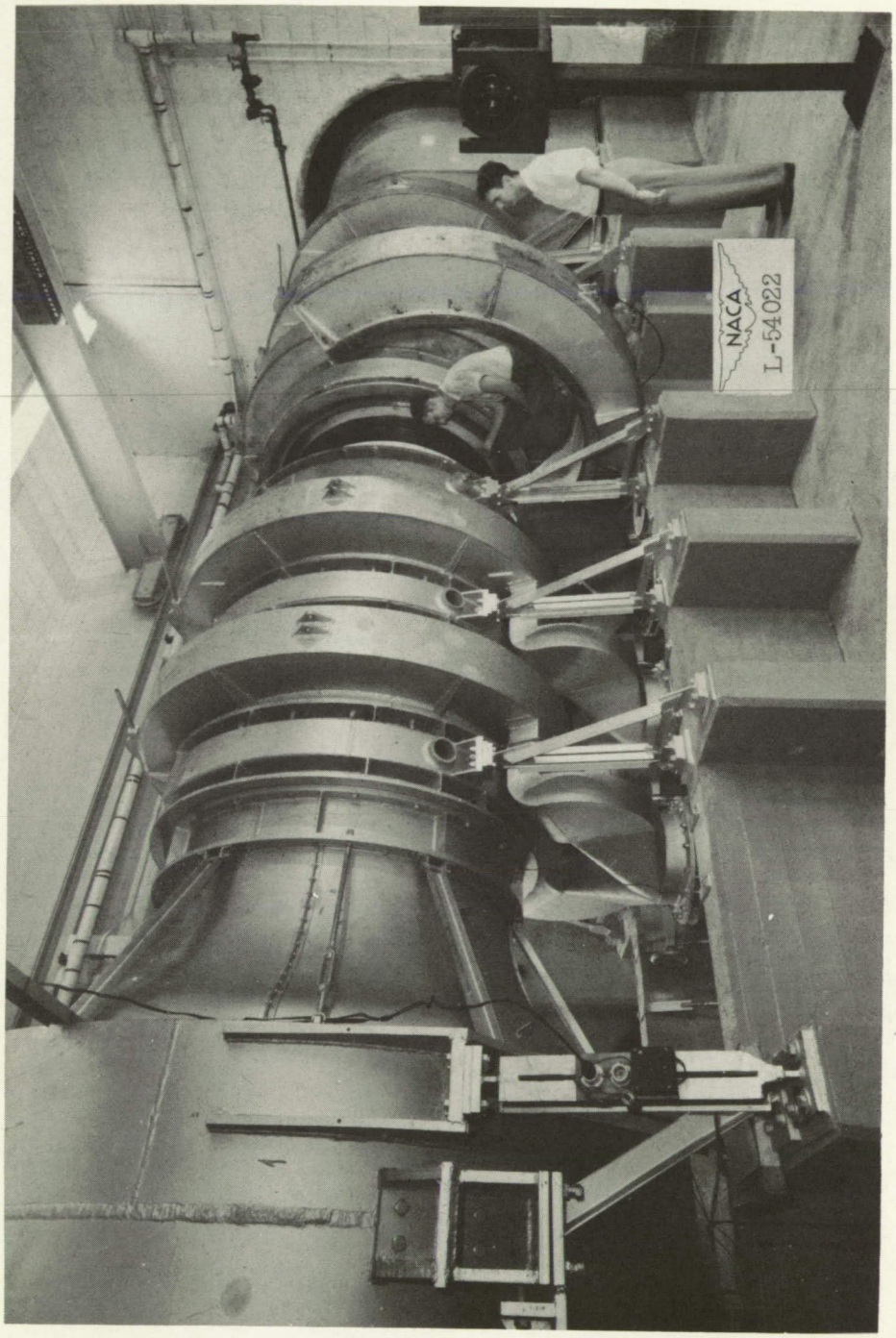

Figure 2.- Photograph of the Langley annular transonic tunnel.

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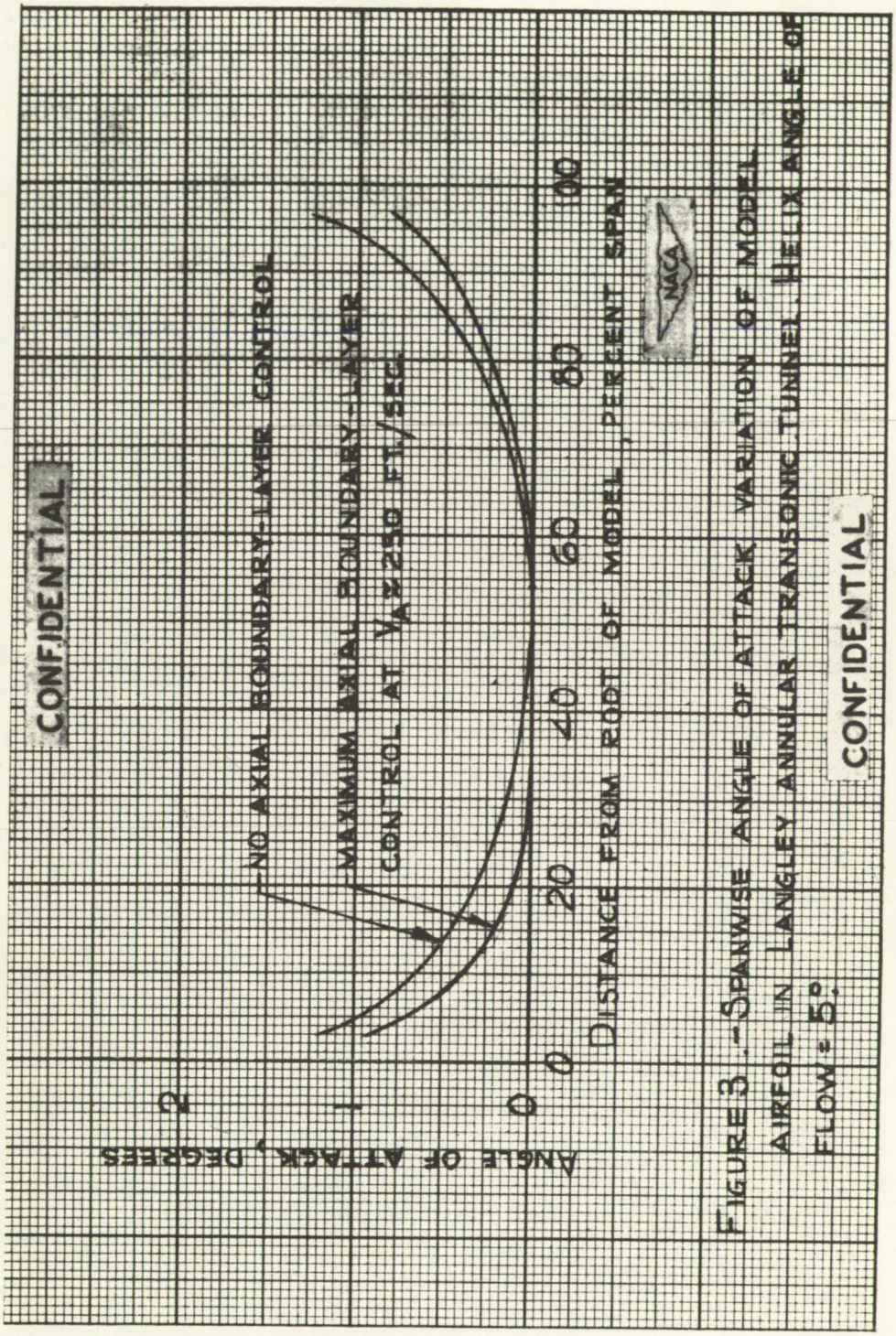

FIGURE 3.- SPANWISE ANGLE OF ATTACK VARIATION OF MODEL AIRFOIL IN LANGLEY ANNULAR TRANSONIC TUNNEL. HELIX ANGLE OF FLOW = 5°.

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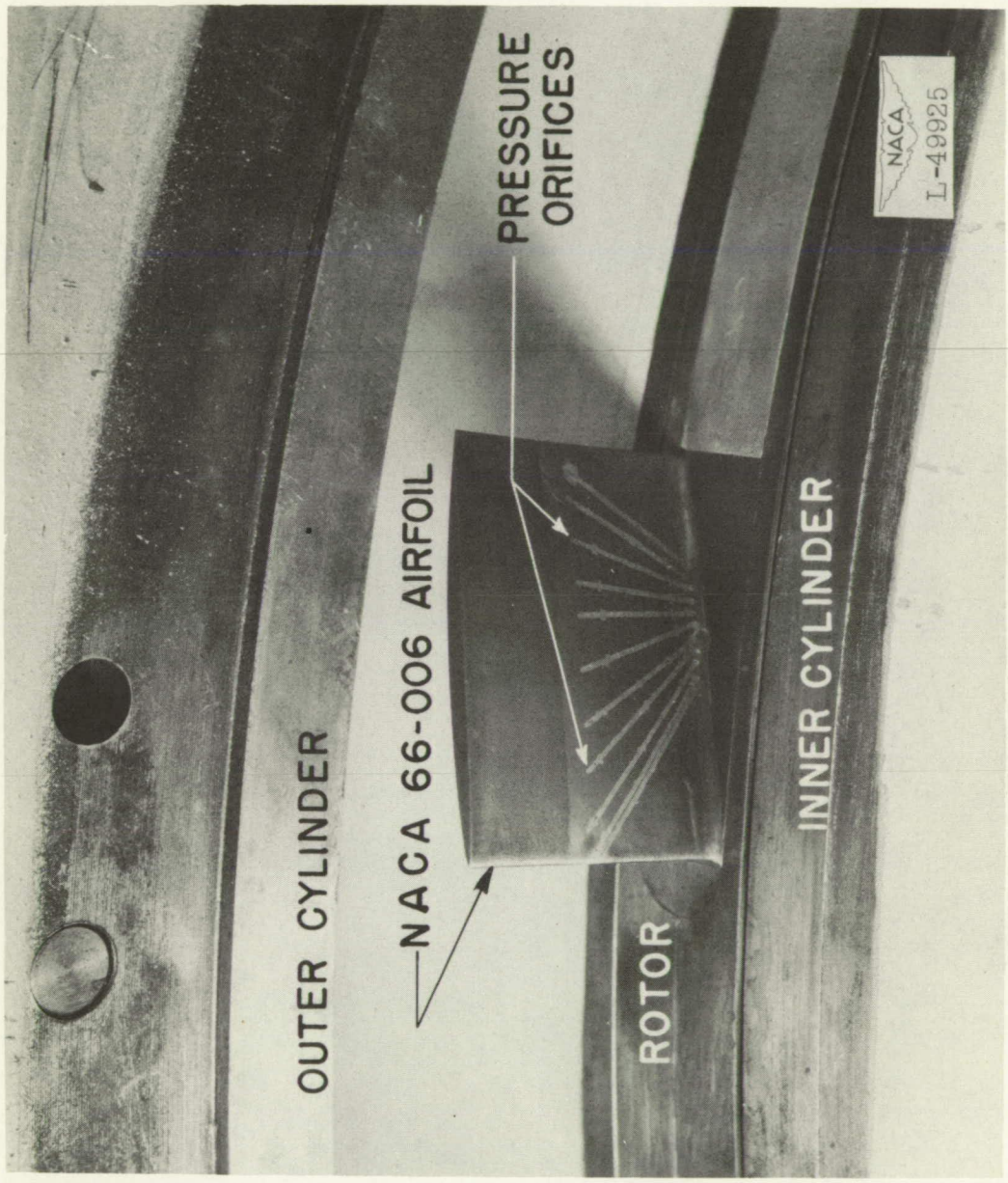

Figure 4.- An NACA 66-006 airfoil mounted in the Langley annular transonic tunnel.

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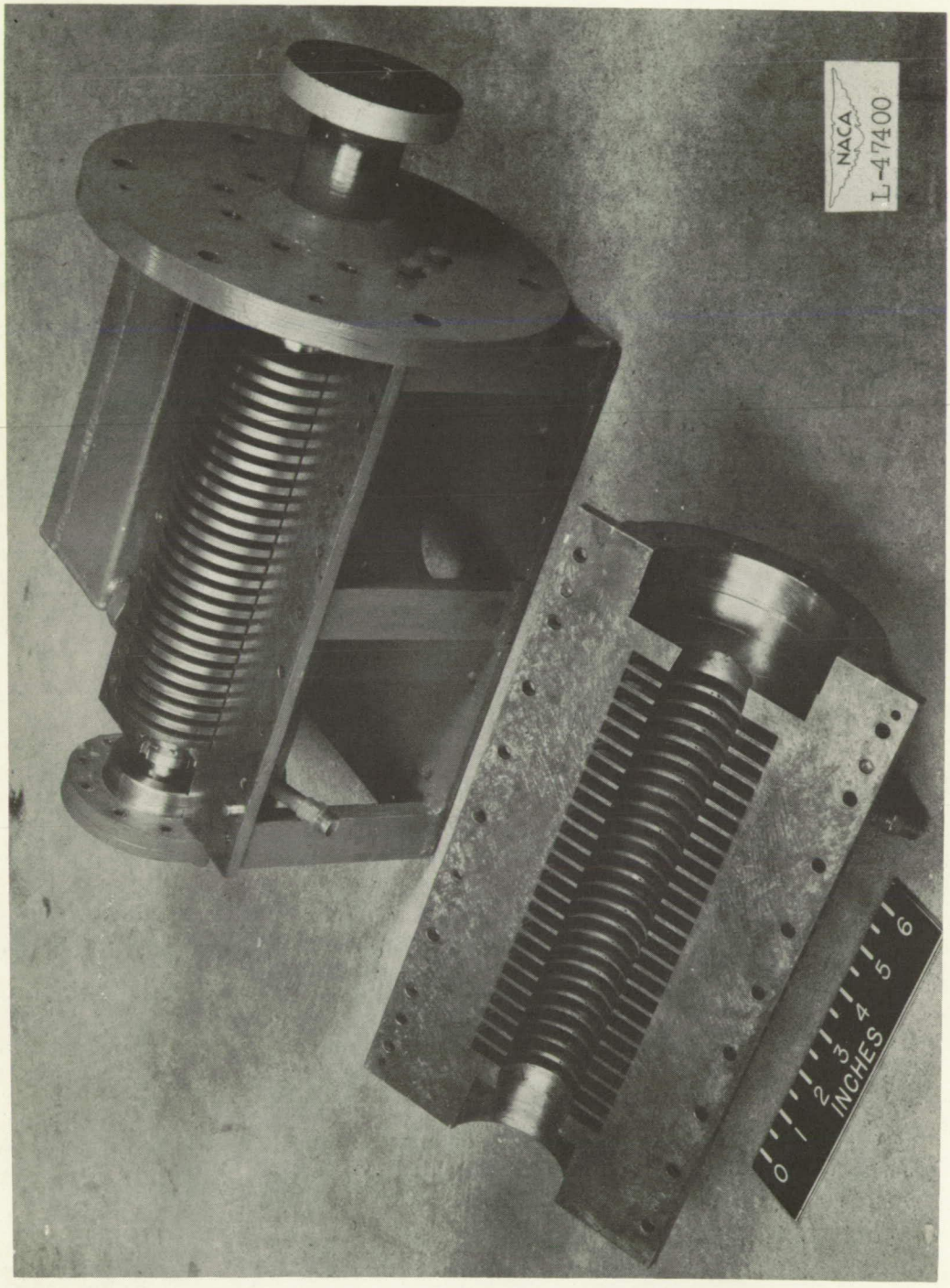

Figure 5.- Internal view of a mercury-seal pressure-transfer device.

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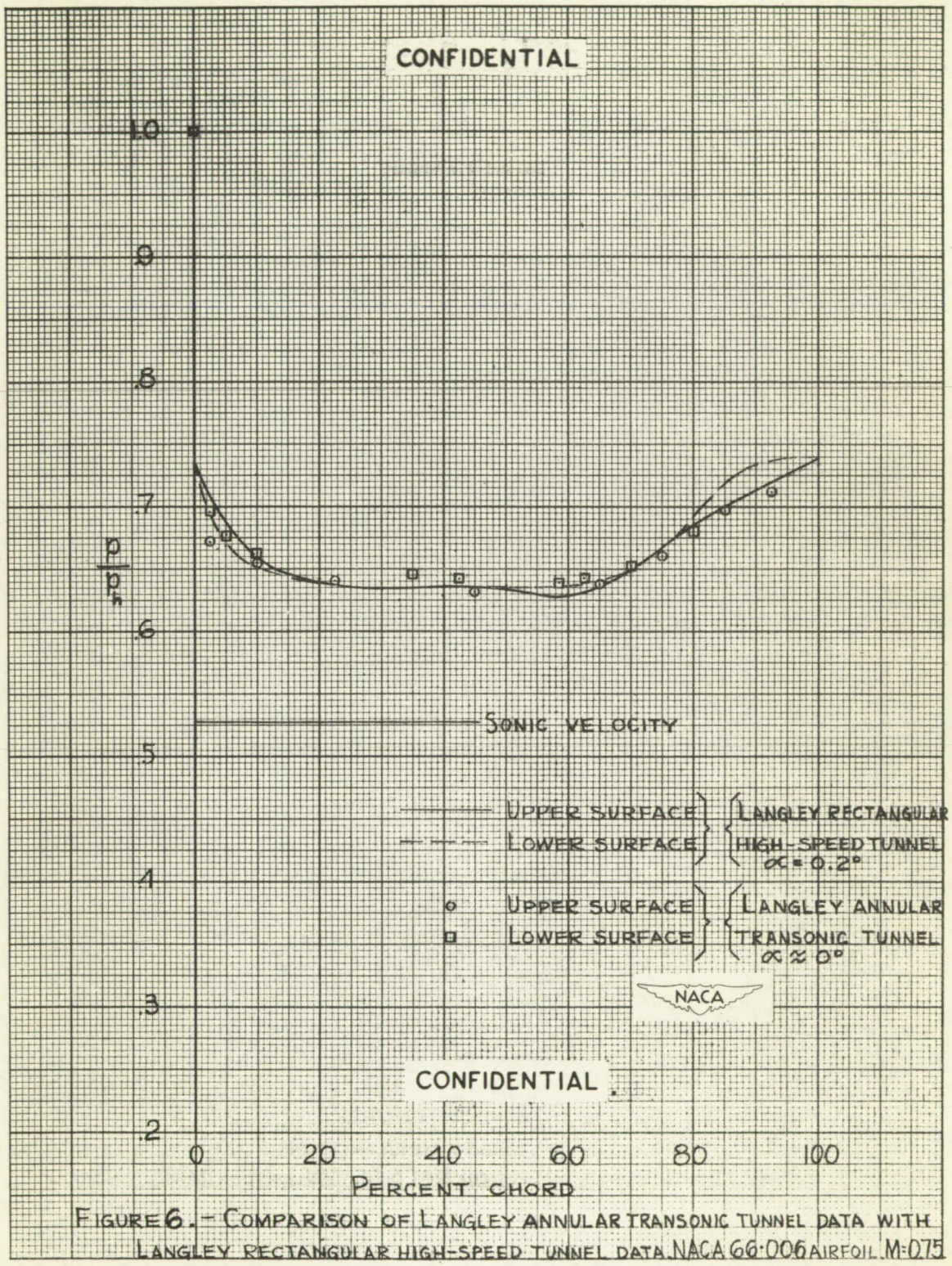

FIGURE 6.- COMPARISON OF LANGLEY ANNULAR TRANSONIC TUNNEL DATA WITH LANGLEY RECTANGULAR HIGH-SPEED TUNNEL DATA. NACA 66-006 AIRFOIL. M=0.75

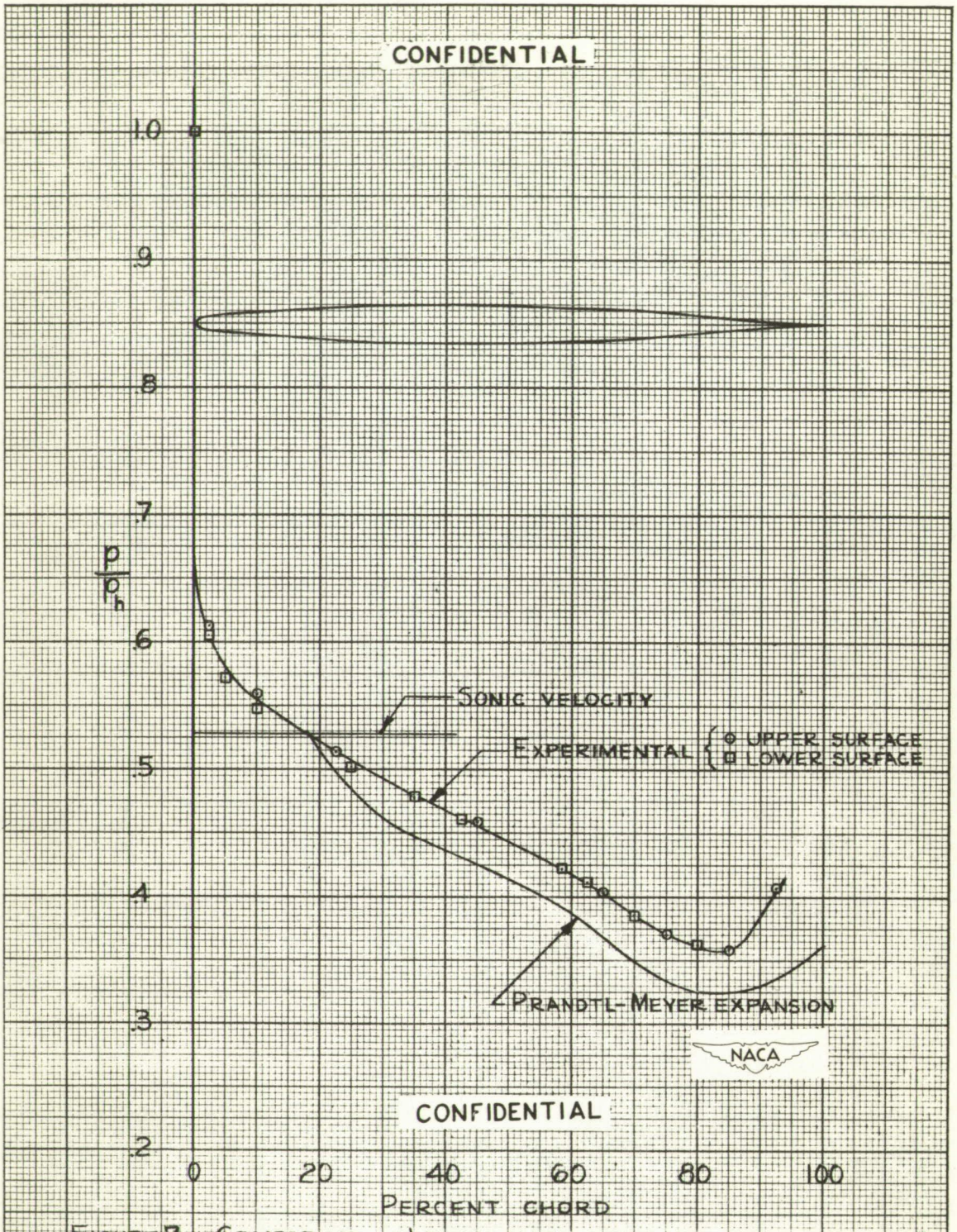

FIGURE 7.—COMPARISON OF LANGLEY ANNULAR TRANSONIC TUNNEL DATA WITH THE PRANDTL-MEYER EXPANSION NACA 66-006 AIRFOIL. $M = 1.00$, $\alpha \approx 0°$

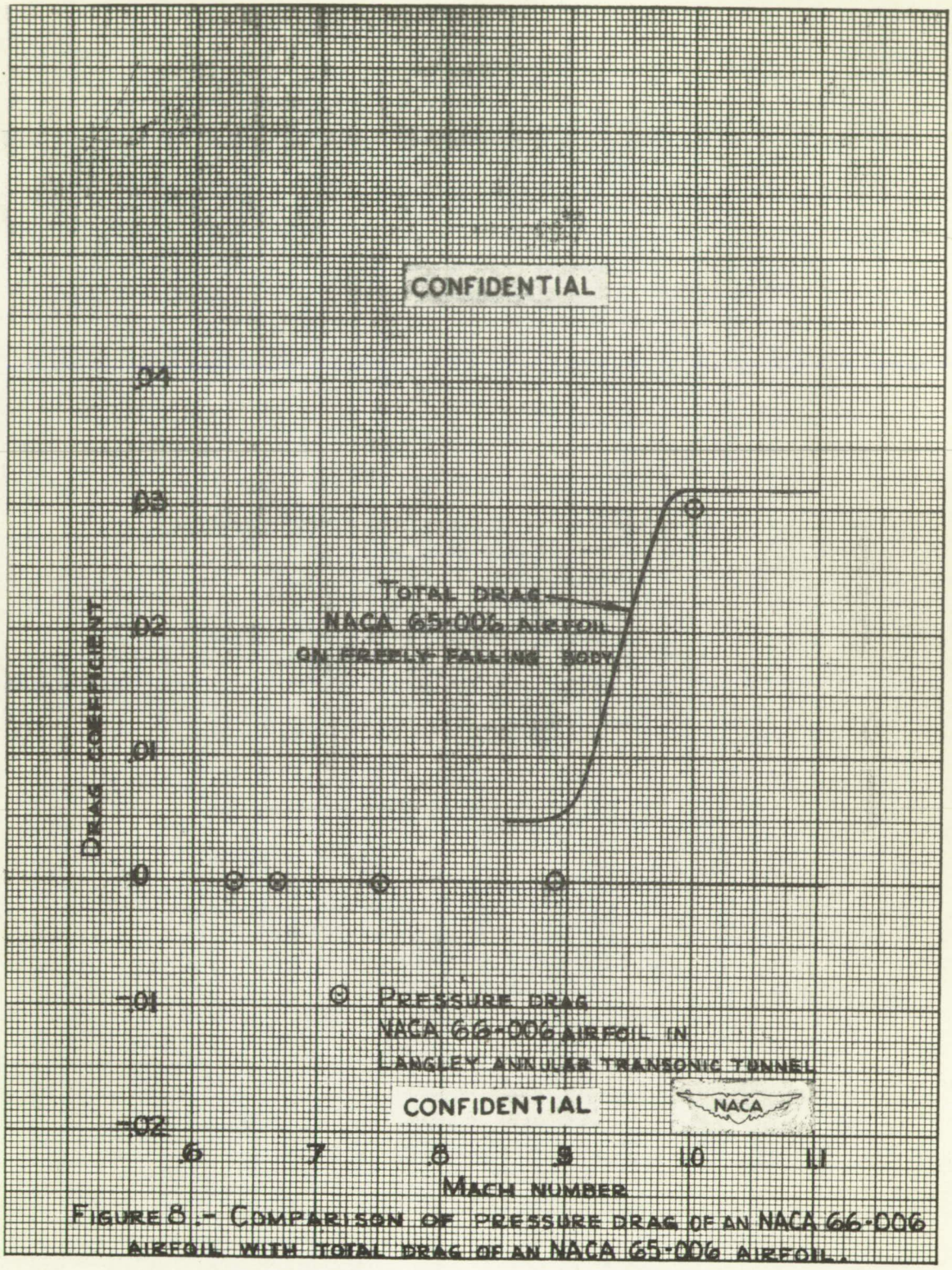

FIGURE 8.- COMPARISON OF PRESSURE DRAG OF AN NACA 66-006 AIRFOIL WITH TOTAL DRAG OF AN NACA 65-006 AIRFOIL.

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