NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

REPORT 1106

THE LANGLEY ANNULAR TRANSONIC TUNNEL

By LOUIS W. HABEL, JAMES H. HENDERSON,
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Langley Field, Va.

National Advisory Committee for Aeronautics

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Conduct, under unified control, for all agencies, of scientific research on the fundamental problems of flight

REPORT 1106

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By Louis W. Habel, James H. Henderson, and Mason F. Miller

SUMMARY

The development of the Langley annular transonic tunnel, a facility in which test Mach numbers from 0.6 to slightly over 1.0 are achieved by rotating the test model in an annular passage between two concentric cylinders, is described.

Data obtained for two-dimensional airfoil models in the Langley annular transonic tunnel at subsonic and sonic speeds are shown to be in reasonable agreement with experimental data from other sources and with theory when comparisons are made for nonlifting conditions or for equal normal-force coefficients rather than for equal angles of attack. The trends of pressure distributions obtained from measurements in the Langley annular transonic tunnel are consistent with distributions calculated for Prandtl-Meyer flow.

INTRODUCTION

The obtaining of experimental aerodynamic information at and very near the speed of sound has involved the use of special techniques in free-fall, rocket, wing-flow, and transonic-wind-tunnel testing. Until the recent development of transonic wind tunnels capable of producing uniform test-section flows continuously from subsonic to low supersonic speeds, the methods for wind-tunnel testing at Mach numbers of and near 1.0 were limited to those utilizing test velocities achieved by induced flow over a bump on the wall of a closed test section or by rotating the test model at high speeds. The latter technique was employed in the Langley annular transonic tunnel, a testing facility designed to provide pressure-distribution data for small two-dimensional models throughout the Mach number range of 0.6 to slightly over 1.0. This tunnel was placed in operation in 1947 and is believed to have yielded the first two-dimensional pressure-distribution data obtained over an airfoil section at a Mach number of 1.0.

The purpose of this report is to describe the development of the Langley annular transonic tunnel and to give an approximate evaluation of the results obtained with this facility by comparison with data from other sources.

SYMBOLS

M	Mach number, V/a
V	test velocity, $\sqrt{V_r^2+V_a^2}$, ft/sec
V_r	velocity of airfoil at center-span section due to rotation, ft/sec
V_a	axial velocity at test section, ft/sec
a	velocity of sound
α	angle of attack of center-span section of airfoil, $\phi-5°$
ϕ	helix angle of the flow, $\tan^{-1}\frac{V_a}{V_r}$, deg
p_0	free-stream static pressure
p	true local absolute static pressure at model airfoil orifice, lb/sq ft
p_m	absolute static pressure indicated by manometer, lb/sq ft
q_0	dynamic pressure, $\frac{1}{2}\rho_0 V^2$
ρ_0	free-stream air density, slugs/cu ft
n	rotation speed of the rotor, rps
g	acceleration due to gravity, ft/sec^2
R	gas constant for air, ft-lb/lb/°F
T	mean temperature of air in rotor tubing, °F abs
r_a	radius to orifice in model airfoil, ft
r_s	radius to orifice in rotating shaft of pressure-transfer device, ft
P	pressure coefficient, $\frac{p-p_0}{q_0}$
P_{cr}	pressure coefficient corresponding to sonic velocity
c_n	section normal-force coefficient
$c_{m_{c/4}}$	section pitching-moment coefficient about airfoil quarter-chord

TUNNEL DEVELOPMENT

GENERAL ARRANGEMENT

The Langley annular transonic tunnel as originally designed is shown schematically in figure 1 and photographically in figure 2. This facility utilizes two concentric circular cylinders arranged with an intervening 3-inch-wide annular passage, which serves as a test section for a two-dimensional model equipped with pressure orifices at its midspan station (see fig. 3). The test model is attached to a rotor, of which the diameter (57 in.) is equal to that of the inner cylinder, and is rotated at velocities up to low supersonic values. An appropriate low-velocity axial flow is induced through the annular passage to control the angle of attack of the model and to prevent the model from operating in its own wake. The model test velocity is equal to the vector sum of the model-rotation and the axial-flow velocities and is continuously variable from intermediate subsonic to low supersonic values. The test Reynolds number for a 4-inch-chord airfoil at a Mach number of 1.0 is of the order of 2.3×10^6.

[1] Supersedes recently declassified NACA RM L8A23, "The Langley Annular Transonic Tunnel and Preliminary Tests of an NACA 66-006 Airfoil" by Louis W. Habel, 1948, and NACA RM L50E18, "Preliminary Investigation of Airfoil Characteristics in the Langley Annular Transonic Tunnel" by Louis W. Habel and James H. Henderson, 1950; also contains pertinent material from NACA RM L9G19, "Analysis of Measured Pressures on Airfoils at Mach Numbers Near 1" by Louis W. Habel and Mason F. Miller, 1949.

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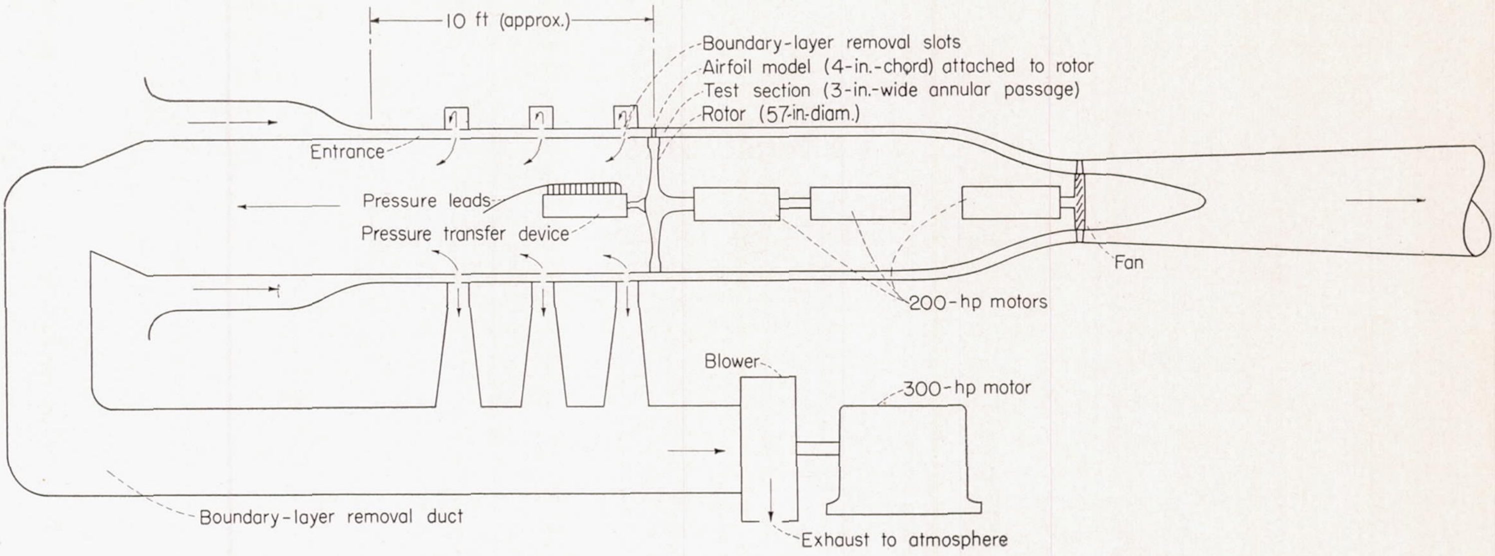


FIGURE 1.—Schematic diagram of the Langley annular transonic tunnel as originally constructed with a long entrance section.

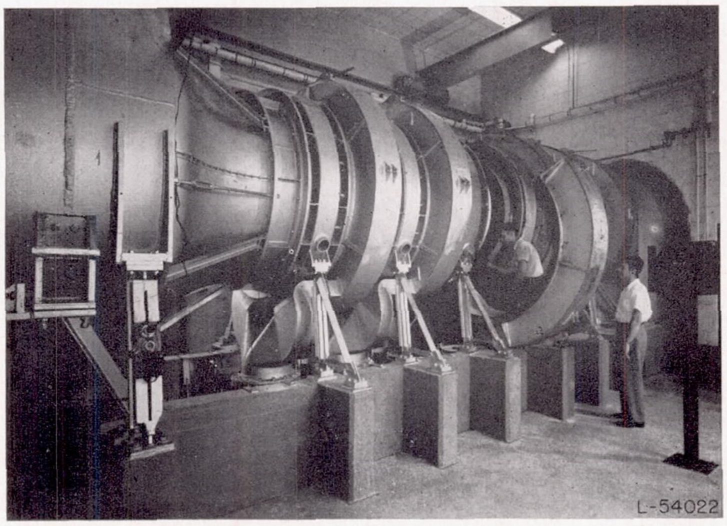


FIGURE 2.—The Langley annular transonic tunnel as originally constructed.

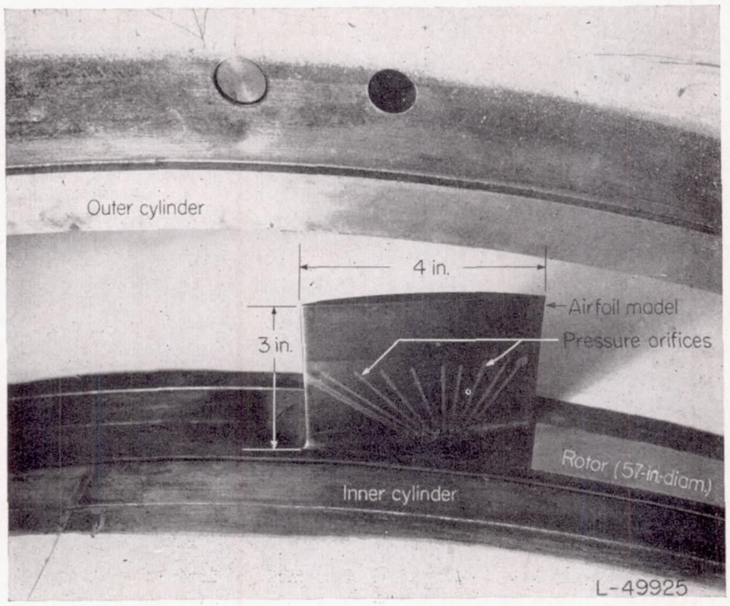


FIGURE 3.—An airfoil mounted in the Langley annular transonic tunnel.

The design of the annular transonic tunnel permits model tests at large ratios of tunnel height to model thickness and is therefore advantageous with respect to the reduction of blockage and choking effects encountered in closed-throat wind tunnels.

ANGLE-OF-ATTACK CONTROL

As was previously mentioned, a continuously variable axial velocity is used to control the angle of attack of the airfoil model and to prevent the model from operating in its own wake. The vector sum of the axial velocity and the model-rotation velocity is equal to the model test velocity. As the model chord line at the midspan station makes an angle of 5° with the plane of rotation of the rotor, the angle between the rotational and test-velocity vectors (the helix angle ϕ) is 5° for an angle of attack of 0°. Helix angles greater than 5° are considered to produce positive angles of attack. Because the maximum axial velocity obtainable through the annular passage at the test section is about 250 feet per second, the maximum angle of attack of model airfoils tested in the Langley annular transonic tunnel is about 13° at a Mach number of 0.7 and about 8° at a Mach number of 1.0.

The airfoil models are twisted so that, when the midspan 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 midspan station is operating at an angle of attack of 5° ($\phi=10°$), the angles of attack of the root and tip sections are within ¼° of the angle of attack at the midspan station if the axial velocity is uniform across the test section.

AXIAL-BOUNDARY-LAYER CONTROL

Removal of some of the boundary layer due to the axial velocity is desirable in order to reduce the spanwise variation in angle of attack. As shown in figure 1, in the original configuration three boundary-layer removal stations upstream of the test section were employed for this purpose. Air entering the slots of the inner cylinder flowed through a duct system to a blower which exhausted to atmosphere. Air which entered the slots in the outer cylinder passed through auxiliary ducts to the main boundary-layer removal duct. During preliminary tests of an NACA 66–006 airfoil in the Langley annular transonic tunnel with the axial-boundary-layer removal system operating at full capacity, it was found that the normal-force curve slopes were lower than would be expected. Axial-velocity surveys across the 3-inch annular passage at the test section indicated that a relatively thick boundary layer existed at the test section even though the boundary-layer removal system was operating at full capacity. In an attempt to reduce the axial-boundary-layer thickness from that indicated by the axial-velocity surveys, the length of the 3-inch annular axial flow path ahead of the rotor was reduced from about 10 feet (fig. 1) to about 2 feet (fig. 4). It was believed that the reduction in spanwise angle of attack of the airfoil associated with a reduction in axial-boundary-layer thickness at the test section would increase the lift-curve slopes measured for airfoil models. Note in figure 4 that the boundary-layer removal slots approximately 12 inches ahead of the test section were retained with the shortened annular entrance length to reduce the boundary-layer thickness at the test section to the minimum value obtainable with this configuration.

Figure 5 illustrates the variation in spanwise angle of attack of a model tested in the Langley annual transonic tunnel for the original long-entrance configuration with and without axial-boundary-layer control and for the short entrance with axial-boundary-layer control. The amount of boundary layer removed during the axial-velocity surveys is the maximum amount which can be removed with the axial-boundary-layer removal system. All three curves shown in figure 5 were computed from results of velocity surveys across the annulus at an axial velocity of approximately 250 feet per second and for an angle of attack of 0° at the center-span station of the airfoil.

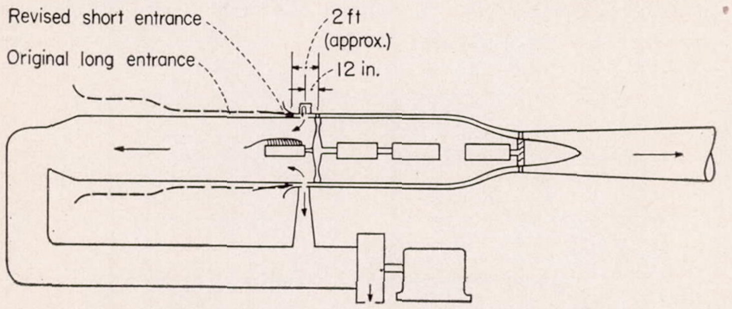


FIGURE 4.—Schematic diagram of the revised short entrance section for the Langley annular transonic tunnel.

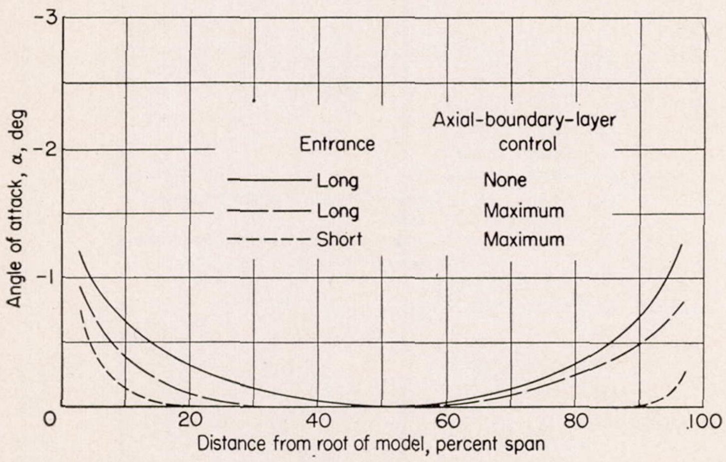


FIGURE 5.—Spanwise angle-of-attack variation of Langley annular-transonic-tunnel models due to boundary layer of the axial flow for various tunnel configurations. V_a=250 fps; α=0°.

The increase in the slope of the normal-force-coefficient curves obtained as a result of decreasing the spanwise angle-of-attack variation by reducing the length of the annular axial-flow entrance section ahead of the rotor is shown in figure 6. The normal-force-coefficient curves are shown for an NACA 66–006 airfoil at a Mach number of 0.625 for the two entrance conditions previously described. Axial-boundary-layer control was employed for both test conditions. For comparison the low-speed section lift curve based on the theoretical lift-curve slope of 2π per radian is shown extrapolated to a Mach number of 0.625 by the Glauert-Prandtl method (ref. 1). The lift-curve slope measured for an NACA 66–006 airfoil in the Langley two-dimensional tunnel at low speed (ref. 2) is within a few percent of 2π per radian. Although shortening the entrance length of the annular transonic tunnel and thereby reducing the axial-boundary-layer thickness caused a marked increase in the normal-force-curve slope of the NACA 66–006 airfoil as measured in the Langley annular transonic tunnel, the measured slope is still lower than the theoretical value.

MODEL INSTALLATION

The models tested in the Langley annular transonic tunnel have approximately 3-inch spans and 4-inch chords

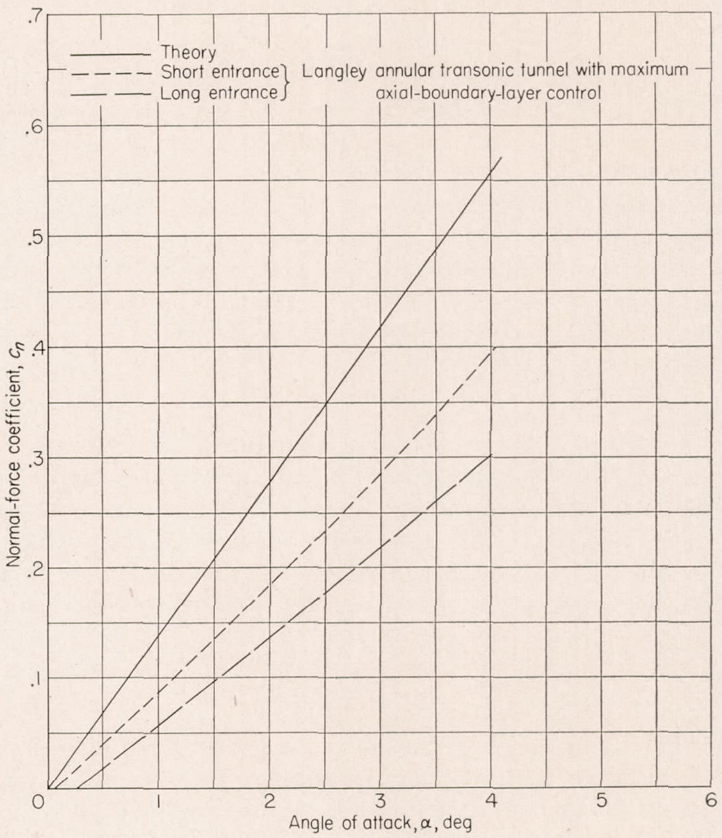


FIGURE 6.—Comparison of normal-force-coefficient curves measured in the Langley annular transonic tunnel with the lift curve based on the theoretical slope of 2π per radian. NACA 66–006 airfoil; Mach number, 0.625

and are equipped with 24 static-pressure orifices at the midspan station. Figure 3 gives a photographic view of an NACA 66–006 airfoil mounted on the tunnel rotor. In this figure the indicated lines on the airfoil surface represent solder-filled slots in which are imbedded stainless-steel capillary tubes leading from the static-pressure orifices at the midspan station. The clearance between the tip of the airfoil and the outer wall is believed to be about 0.010 inch during operation; the large clearance shown photographically in figure 3 is not representative of actual conditions because of the removal of a portion of the outer cylinder.

PRESSURE-TRANSFER DEVICES

Two types of pressure-transfer devices, both of which were developed at the Langley Laboratory of the National Advisory Committee for Aeronautics, have been used for tests in the Langley annular transonic tunnel to transfer the pressures on the rotating model airfoils to a stationary manometer. The first pressure-transfer device employed cells sealed with rotating mercury and is described in reference 3. The second pressure-transfer device, which represented quite an improvement over the first, used synthetic rubber as the sealing medium. A complete description of the latter device is presented in reference 4.

TESTS, PROCEDURES, AND REDUCTION OF DATA

The test velocity and angle of attack for models in the Langley annular transonic tunnel are set simultaneously by bringing both the rotational speed of the model and the axial velocity to predetermined values. The magnitude of the test velocity V is determined from the relation:

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

The rotational velocity of the rotor is determined by comparing the frequency output of a small generator, driven by the rotor shaft, with known frequencies. A pitot-static tube is mounted in the annular passage slightly upstream of the test section to determine the axial velocity.

Because 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 is determined from the relation:

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

Pressure distributions over the airfoil sections are recorded by photographing a multiple-tube manometer which is connected through suitable tubing to the pressure-transfer device. The recorded pressures can be corrected for the effects of centrifugal force on the columns of air in the tubes inside the rotor by the following relation:

$$\frac{p}{p_m}=e^{\frac{2\pi^2n^2}{gRT}(r_a^2-r_s^2)}$$

The temperature of the air in the rotor tubing is assumed to be that indicated by a calibrated temperature gage about 20 inches long installed on the rotor tubing. An electrical signal determined by the temperature at the gage is brought from the rotor through slip rings to a temperature indicator at the control desk.

Model tests for which data are presented in this report include those for the NACA 66–006 and 65–110 airfoils and for 6- and 10-percent-thick symmetrical double-wedge airfoils. Test data for the NACA 66–006 airfoil were used not only for obtaining experimental information concerning the slope of the normal-force-coefficient curve for a thin airfoil in the tunnel (see fig. 6) but also for providing a comparison with available subsonic data from the Langley rectangular high-speed tunnel (ref. 5). Test data for the NACA 65–110 airfoil were obtained to permit comparisons with available flight data at subsonic and sonic speeds. The experimental data for the double-wedge airfoils were obtained to permit comparisons with theory at a Mach number of 1.0 (ref. 6). With the exception of data for the NACA 66–006 airfoil at $\alpha\approx0°$, which were obtained with the long-entrance configuration and no axial-boundary-layer control, all data for the Langley annular transonic tunnel were obtained for a tunnel configuration employing both axial-boundary-layer control and the short entrance section. The annular-transonic-tunnel data were obtained at Mach numbers from about 0.6 to slightly more than 1.0 and at angles of attack from 0° to 4°.

RESULTS AND DISCUSSION

NONLIFTING CONDITIONS

Comparison of pressure distributions with other experimental data.—In figure 7 a comparison is made of pressure distributions measured for the NACA 66–006 airfoil in the Langley annular transonic tunnel at a Mach number of 0.75 and an angle of attack of approximately 0° with pressure distributions measured for this airfoil section in the Langley rectangular high-speed tunnel at the same

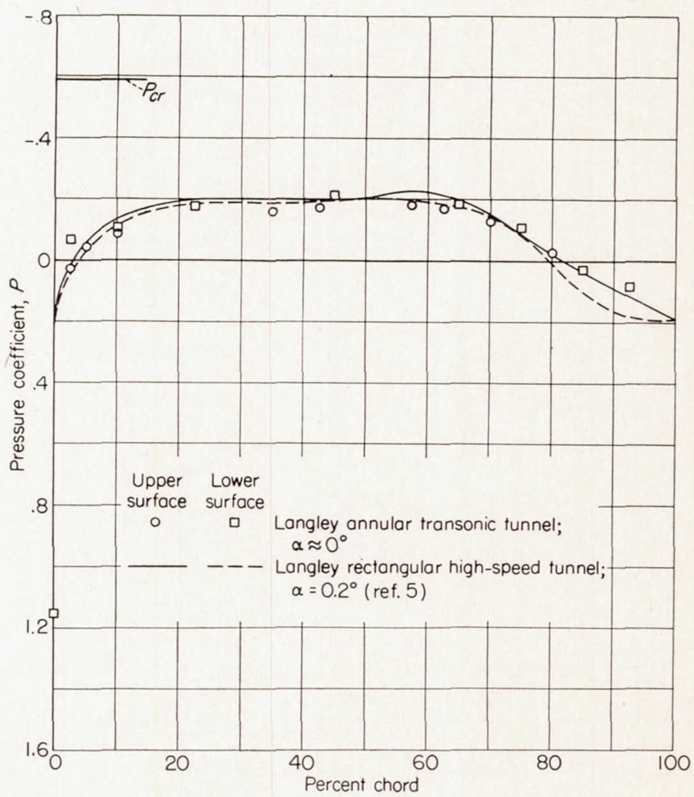
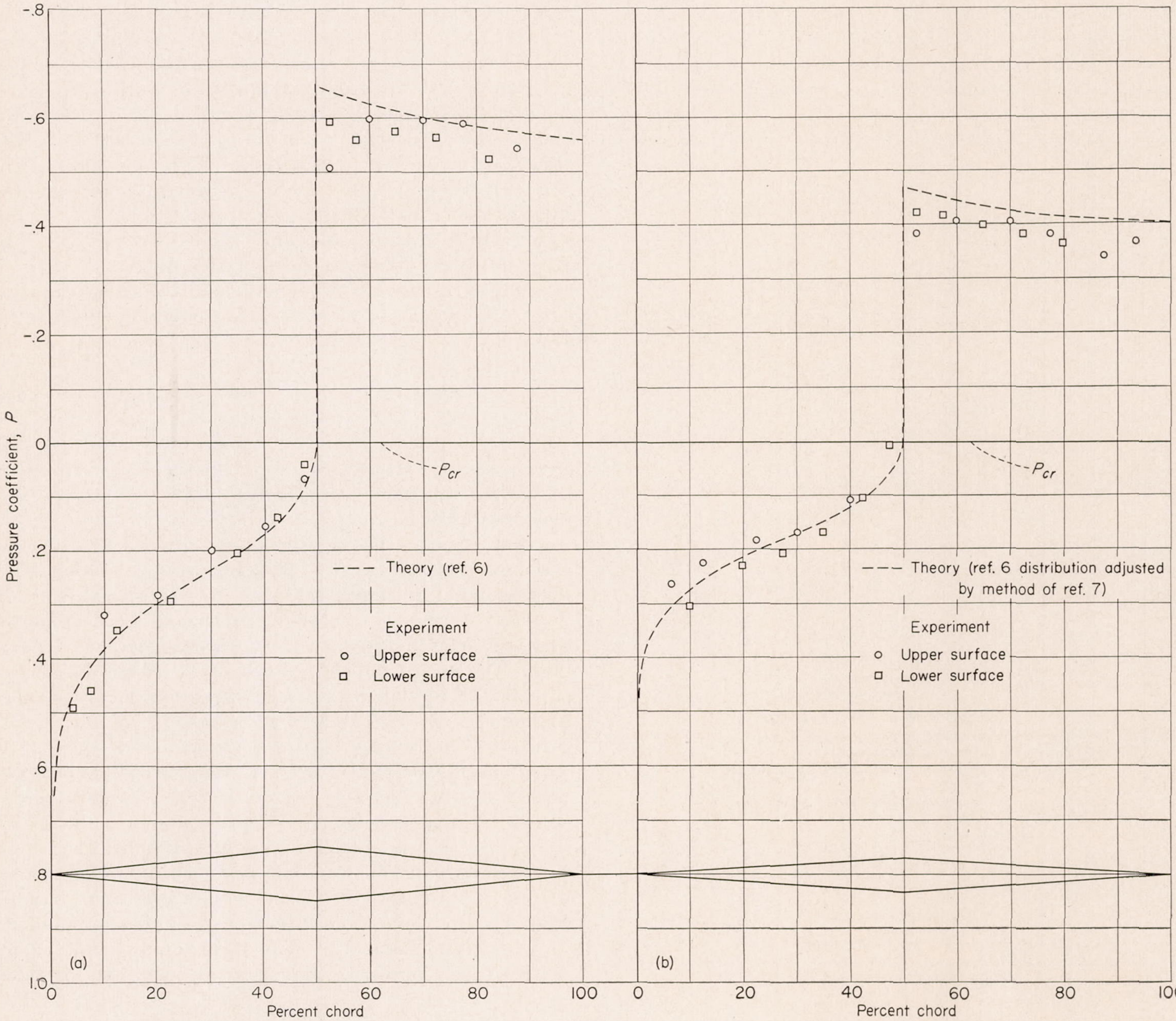


FIGURE 7.—Comparison of pressure distributions measured for an NACA 66-006 airfoil in the Langley annular transonic tunnel (with long entrance section and without boundary-layer control) with those measured in the Langley rectangular high-speed tunnel. $M=0.75$.

Mach number and an angle of attack of 0.2° (ref. 5). The data in this figure from the annular transonic tunnel are for the long-entrance configuration with no control of the axial boundary layer; however, since these data are for nonlifting conditions they should be unaffected by the entrance configuration. The data from the rectangular high-speed tunnel are estimated to be essentially free of tunnel-wall effects. The close agreement of the data from the two tunnels (fig. 7) is indicative of the reliability of thin-airfoil data from the Langley annular transonic tunnel for nonlifting conditions at a subsonic Mach number of 0.75.

Comparison with theory.—The quality of pressure-distribution measurements in the Langley annular transonic tunnel for nonlifting conditions at a Mach number of 1.0 is indicated by comparisons with theoretical pressure distributions for 10- and 6-percent-thick double-wedge airfoils. (See fig. 8.) The theoretical pressure distribution shown in figure 8 (a) for the 10-percent-thick symmetrical double-wedge airfoil at an angle of attack of 0° was calculated by Guderley and Yoshihara (ref. 6). The theoretical pressure distribution shown in figure 8 (b) was obtained by adjusting the theoretical pressure distribution for the 10-percent-thick double-wedge airfoil (fig. 8 (a)) to a 6-percent-thick profile by use of the transonic similarity rule (ref. 7). The agreement between the annular-tunnel experimental data for the 10- and 6-percent-thick airfoils with theory is generally very good, especially over the front part of the airfoil. The fact that the experimental pressures are larger than the theoretical pressures over the rear part of the airfoil is believed due to the presence of the boundary layer and to slightly rounded corners of the airfoil at its maximum-thickness station, both of which



(a) Airfoil thickness, 10 percent chord.
(b) Airfoil thickness, 6 percent chord.

FIGURE 8.—Comparisons of pressure distributions measured in the Langley annular transonic tunnel with theoretical pressure distributions for symmetrical double-wedge airfoils. $M=1.0$; $\alpha=0°$.

tend to reduce the extent of the supersonic expansion downstream of the maximum-thickness station. The pressure-drag coefficient corresponding to the experimental pressure distribution for the 10-percent-thick double-wedge section at $M=1.0$, $\alpha=0°$ was found to be 0.081, which is only slightly below the theoretical value of 0.088 obtained by Guderley and Yoshihara (ref. 6). Although the comparisons presented in figure 8 indicated satisfactory agreement between experiment and theory, this evidence is not believed sufficient to warrant a conclusion that data obtained for nonlifting conditions in the Langley annular transonic tunnel near a Mach number of 1.0 are completely reliable.

Comparison with Prandtl-Meyer calculations.—In a further attempt to evaluate the reliability of data from the Langley annular transonic tunnel, Prandtl-Meyer expansions of the supersonic flow over an airfoil were calculated and compared with the experimental data. The calculations based on the methods of reference 8 as applied to the NACA 66–006 airfoil at $\alpha=0°$ and $M=1.0$ are presented in figure 9. The solid line was computed by assuming Prandtl-Meyer flow to begin at the measured sonic-velocity location (approx. 18-percent chord). Although Prandtl-Meyer flow indicates velocities somewhat greater than those measured by experiment, the general shapes of the curves as well as the points of maximum velocity (minimum pressure) are in good agreement. At the sonic-velocity location, Prandtl-Meyer flow indicates an infinite rate of change of velocity (or pressure) with turning angle. The theory of references 9 and 10, however, indicates that the flow through sonic velocity is not subject to the abrupt discontinuity inherent in the Prandtl-Meyer flow. Instead, Mach lines or expansion waves leaving the airfoil surface behind the sonic-velocity location are reflected from the sonic-velocity line as compression waves which, upon reaching the airfoil surface, reduce the local velocity. As a result the flow directly behind the sonic-velocity location on an airfoil is of a complicated nature and the Prandtl-Meyer flow (a purely supersonic concept which neglects the incoming compression waves associated with transonic flow) does not present a true representation of the flow picture.

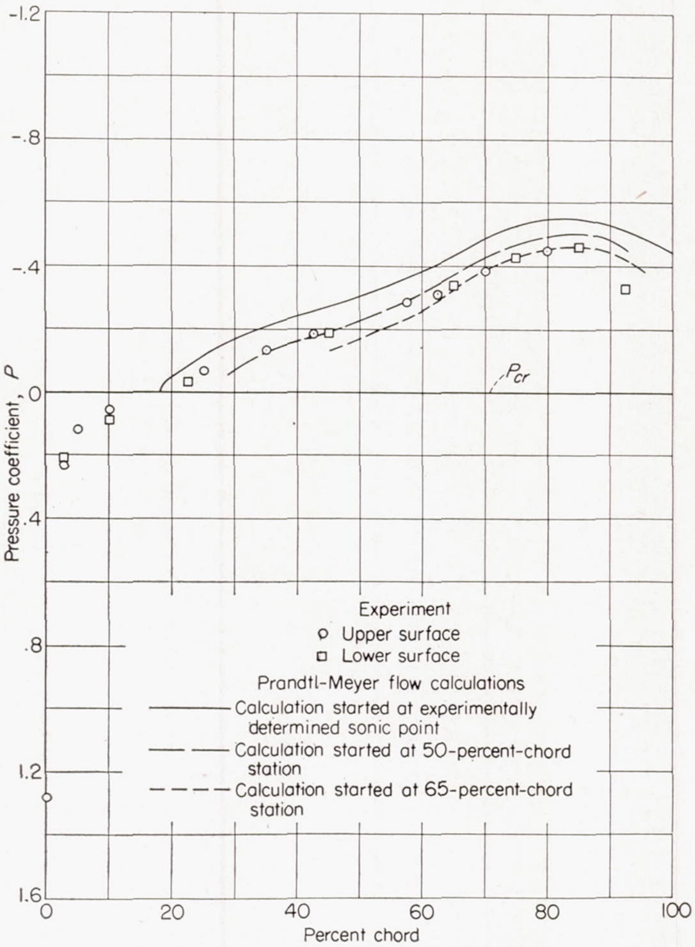


Figure 9.—Comparison of experimental pressure distributions from tests in the Langley annular transonic tunnel with distributions calculated for Prandtl-Meyer expansions in the region of supersonic flow over an NACA 66–006 airfoil. $M=1.0$; $\alpha=0°$.

From the above considerations, it appeared that somewhere behind the sonic-velocity location on the airfoil surface a particular point existed for which the leaving expansion wave would be reflected as a compression wave from the sonic-velocity line and would return to the airfoil surface exactly at the airfoil trailing edge. Then, rearward of the point in the airfoil for which this condition occurs, the Prandtl-Meyer flow should give a good indication of the experimental flow because only the expansion waves or Mach lines need be considered. Accordingly, the experimental pressure at the 50-percent-chord station was used as a starting point to compute the pressures indicated by Prandtl-Meyer flow both forward and rearward of this station (fig. 9).

The agreement of the data from Prandtl-Meyer calculations starting at the 50-percent-chord station with the expermental data is excellent between the 35- and 60-percent-chord stations. Rearward of the 60-percent-chord station the calculated curve indicates lower pressures than those measured.

A third Prandtl-Meyer calculation obtained by using the measured pressure at the 65-percent-chord station as a starting point is shown in figure 9. Agreement between this curve and the measured pressures is good from the 60-percent-chord station to about the 90-percent-chord station. The lack of agreement downstream of this station can probably be attributed to shock waves near the cusp-shaped trailing edge of the NACA 66–006 airfoil.

In order to indicate that the pressures corresponding to the Prandtl-Meyer flow are representative of those experienced in the supersonic region of the flow over an airfoil section, comparisons similar to those presented in figure 9 for data from the Langley annular transonic tunnel are shown in figure 10 for transonic propeller data. These data are for an NACA 16–307 airfoil section and were obtained from measurements at the 0.8 radius of a rotating propeller tested in the Langley 16-foot high-speed tunnel. Only upper-surface pressures are presented and the indicated angle of attack has been corrected for induced flow through the propeller. The agreement of these propeller data with Prandtl-Meyer flow (fig. 10) is similar to that obtained in the Langley annular transonic tunnel, and thus the applicability of the Prandtl-Meyer calculation is substantiated.

LIFTING CONDITIONS

Comparisons of pressure distributions measured in the Langley annular transonic tunnel with other experimental and theoretical data.—In figure 11, the pressure distributions measured for the NACA 66–006 airfoil in the Langley annular transonic tunnel at a Mach number of 0.694 and an angle of attack of 2.2° are compared with theoretical pressure distributions. The solid lines represent the theoretical pressure distributions based on an angle of attack of 2.2° and were obtained by calculating the low-speed pressure distrubution by use of the methods described in reference 2 and adjusting the low-speed distributions to a Mach number of 0.694 by the use of reference 11. The low-speed lift coefficient for which the theoretical pressure distribution was computed was determined by using the theoretical value of the lift-curve slope (2π per radian). Although the shape of the theoretical curve is in agreement with that obtained by experiment, the theoretical pressure distribution calculated for an angle of attack of 2.2° represents considerably more lift than is indicated by the experimental pressure distribution. The dashed line in figure 11 represents a "theoretical" pressure distribution determined by using, in the method of reference 2, a reduced lift coefficient such that, after the compressibility corrections were applied, the lift coefficient would be the same as the experimental. The shapes of the theoretical and experimental pressure distributions are shown to be in good agreement although the absolute values of the measured pressures are slightly greater than those indicated by the theoretical curve.

In figure 12 the pressure distributions measured in the Langley annular transonic tunnel for an NACA 65–110 airfoil section are compared with the pressure distributions measured at the same normal-force coefficients in flight for a similar airfoil at Mach numbers of 0.79 (fig. 12 (a)) and 1.00 (fig. 12 (b)). The flight airfoil section was an NACA 65–110 section modified to remove the trailing-edge cusp. The flight measurements were made near the midspan

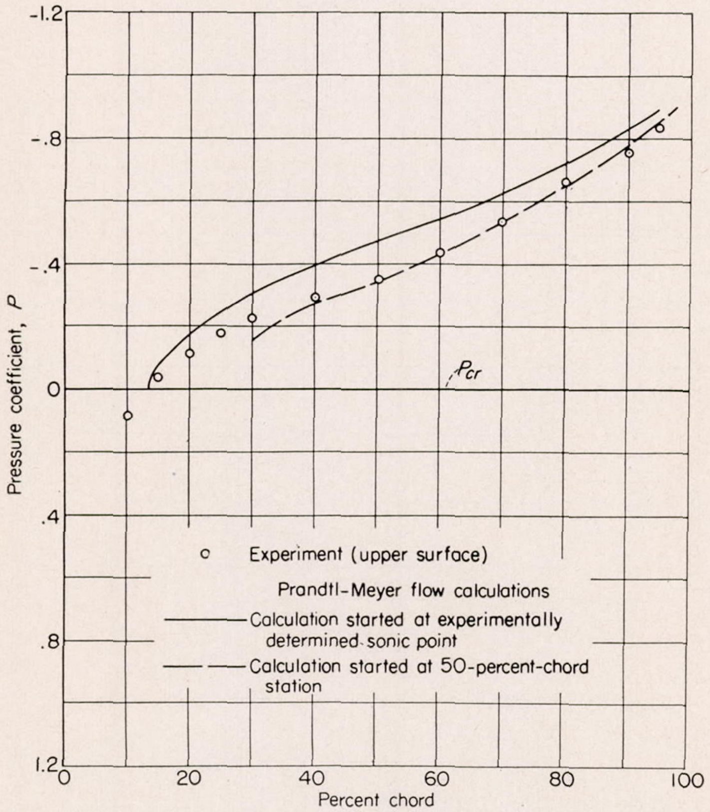


FIGURE 10.—Comparison of experimental pressure distributions from tests of a rotating propeller blade in the Langley 16-foot high-speed tunnel with distributions calculated for Prandtl-Meyer flow. NACA 16–307 airfoil section; M=1.0; α=0.35°.

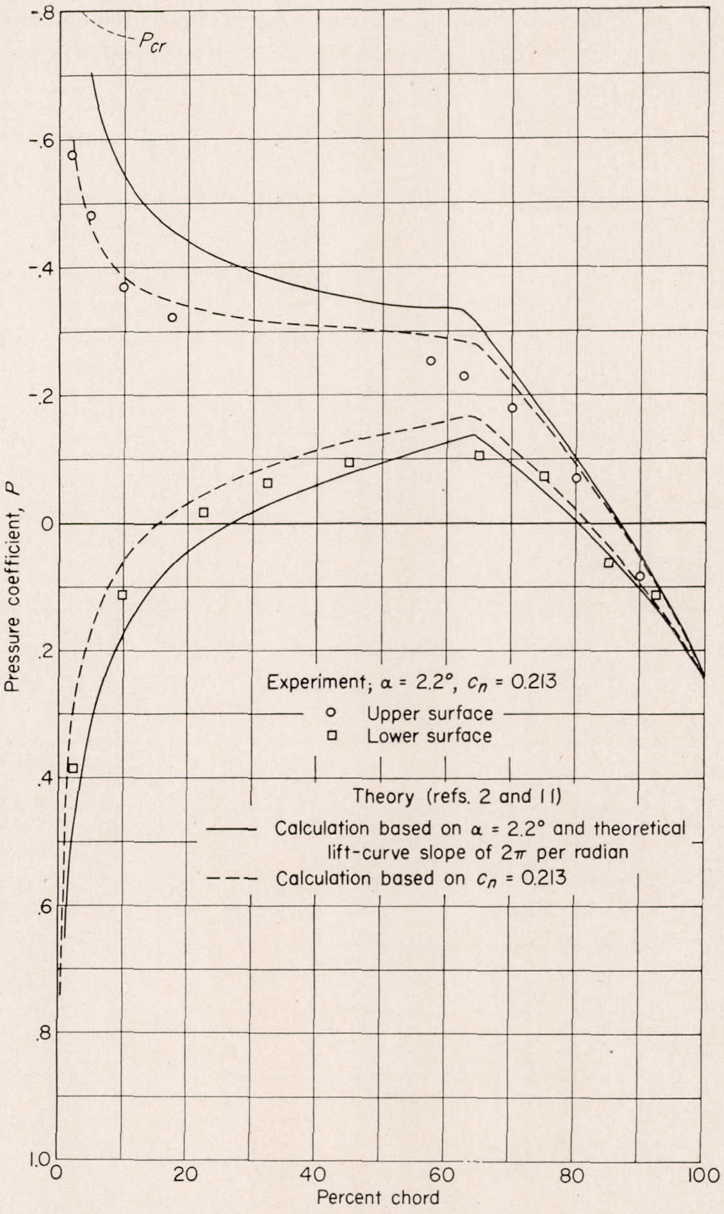


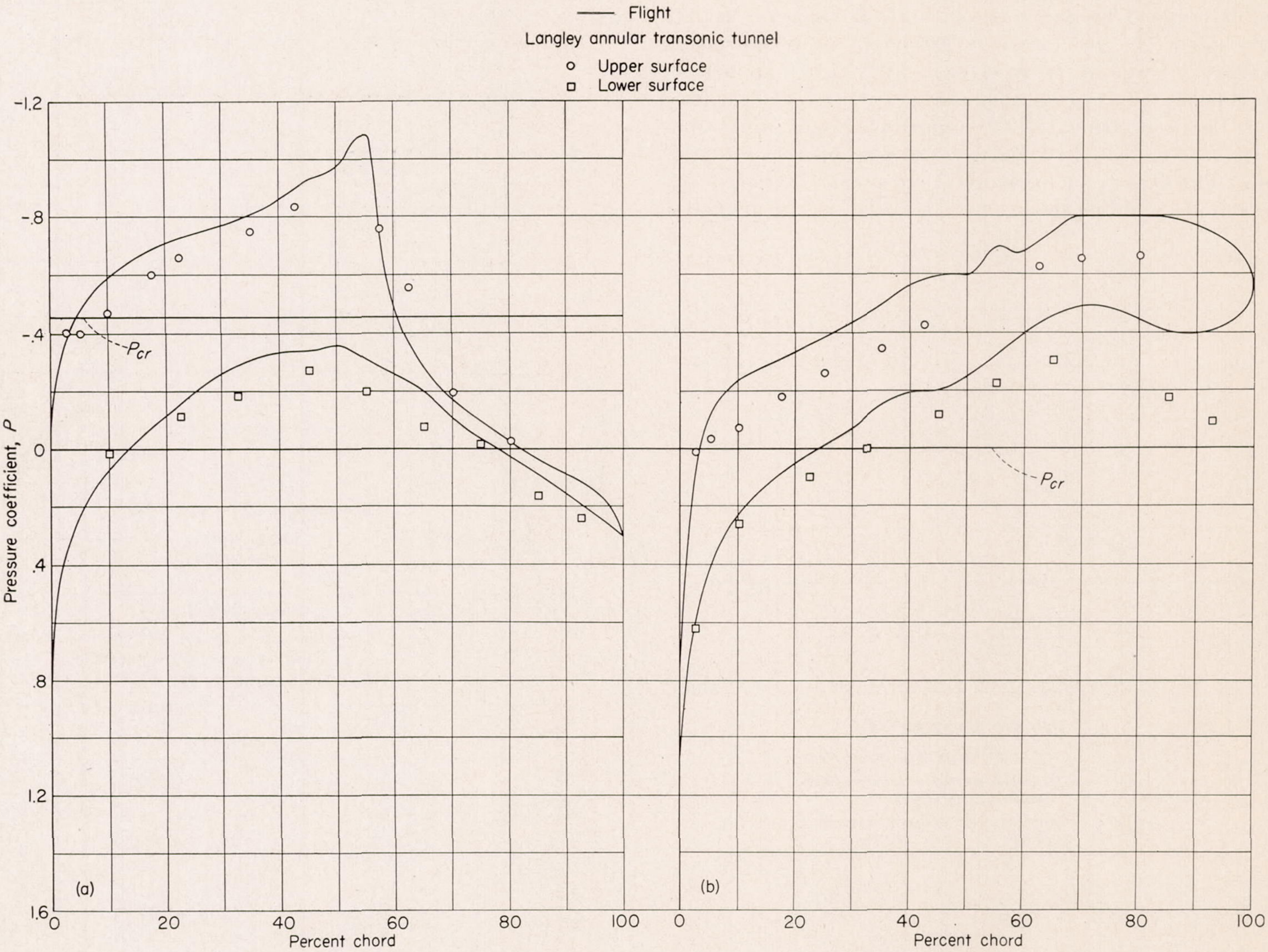
FIGURE 11.—Comparison of pressure distributions measured in the Langley annular transonic tunnel with theoretical pressure distributons for an NACA 66–006 airfoil. M=0.694.

station of a wing where fuselage and tip effects would be expected to be at a minimum. The comparisons of figure 12 indicate relatively good agreement between the shapes of the pressure-distribution curves at both subsonic and sonic speeds, although larger absolute pressures were indicated for the distributions from the Langley annular transonic tunnel. At sonic speed, the pressure coefficients over the rearward part of the airfoil would have been expected to be in better agreement for the two tests if the airfoil section used in the flight tests had been cusped near the trailing edge.

Typical normal-force and pitching-moment data.—As an indication of the type of data obtainable in the Langley annular transonic tunnel with the short entrance, the normal-force and pitching-moment coefficients measured for the NACA 66–006 airfoil are presented as a function of Mach number for various values of angle of attack in figure 13. The normal-force data (fig. 13 (a)) obtained for the lifting conditions are characteristic of curves of normal-force coefficient plotted against Mach number in that, as the Mach number is increased, the normal-force coefficient increases to a peak, then decreases rapidly, and levels off near a Mach number of 1.0. The angles of attack indicated in the figure are believed to be in error, and the curves shown are believed to be representative of those which would be expected for angles of attack slightly lower than the indicated values.

In figure 13 (b) the pitching-moment coefficients about the quarter-chord position are presented for the NACA 66–006 airfoil as a function of Mach number for several angles of attack. The moment coefficients generally remain near zero but diverge to negative values as the angle of attack and the Mach number are increased. The angles of attack indicated in figure 13 (b) are, as in figure 13 (a), believed to be in error.

Angle-of-attack error.—The data presented in the preceding figures indicate that data from the Langley annular transonic tunnel are in relatively good agreement with data from other sources and with theory when comparisons are made for nonlifting conditions or for equal normal-force coefficients rather than for equal angles of attack. It is



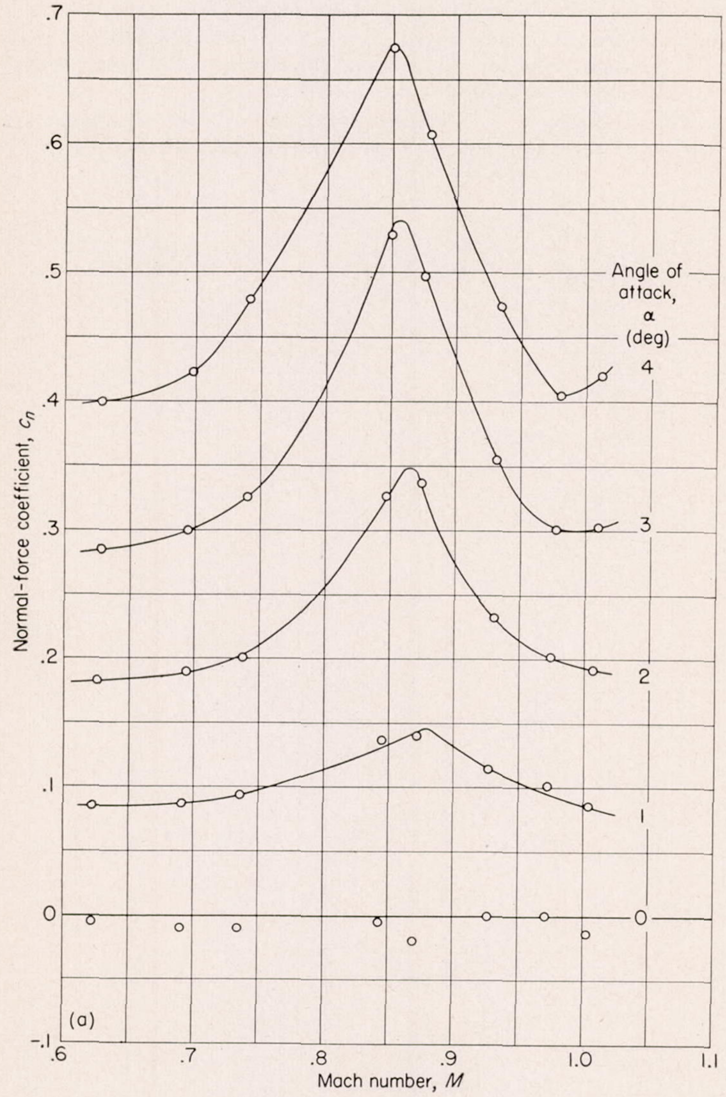
(a) M=0.79; c_n=0.386.

(b) M=1.00; c_n=0.379.

FIGURE 12.—Comparisons of pressure distributions measured in the Langley annular transonic tunnel and in flight for the NACA 65–110 airfoil.

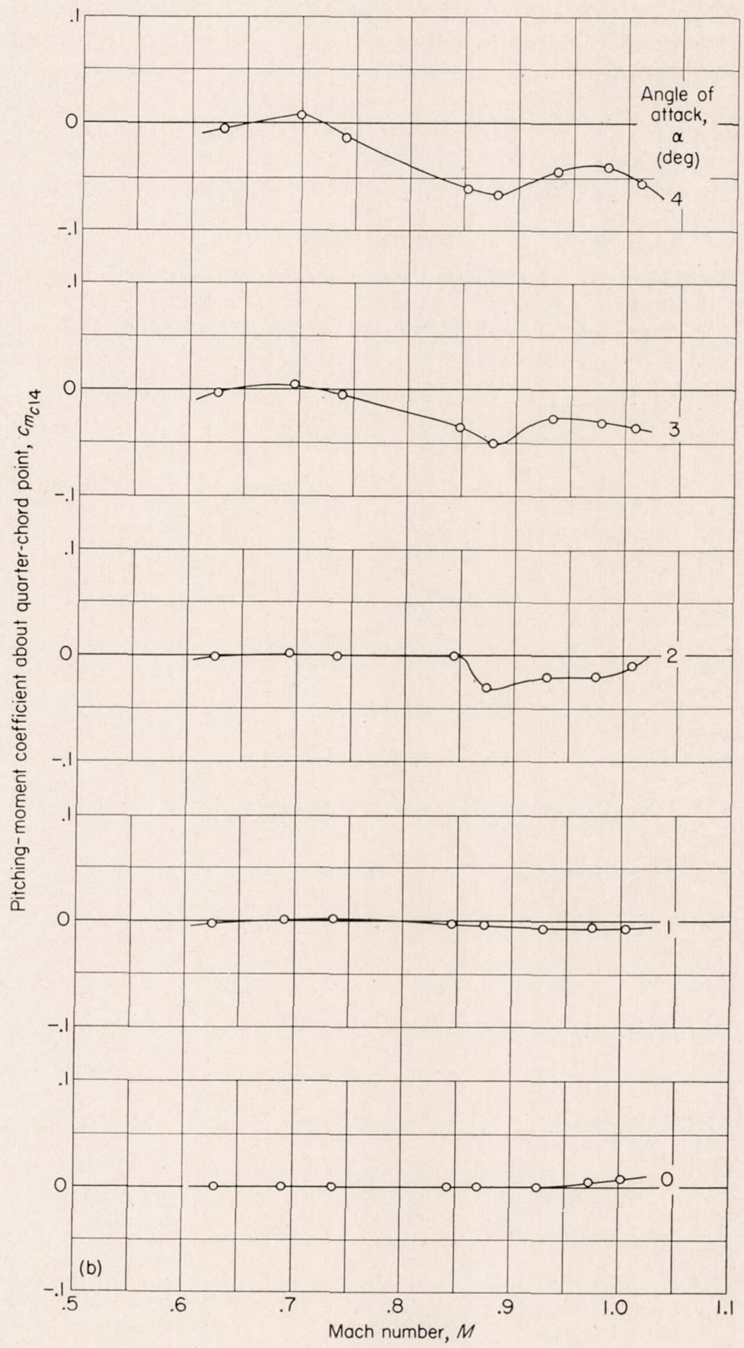
thus believed that the largest source of error in the data from the Langley annular transonic tunnel is an error in the angle of attack. In view of the large increase in the slope of the normal-force-coefficient curve gained by reducing the axial-boundary-layer thickness, and thus the spanwise angle-of-attack variation, further reduction of the axial-boundary-layer thickness might be expected to result in further increases of the slope of the normal-force-coefficient curve. Experience with airfoils in other test facilities has indicated, however, that a finite boundary-layer effect may still remain, unless continuous suction can be applied to the immediate area about which the airfoil is attached. Practical considerations did not permit this type of boundary-layer removal in the annular transonic tunnel.

Inasmuch as the angle of attack is determined directly from the helix angle which, in turn, is a function of the measured rotational and axial velocities, the indicated test velocities may be in error. The rotational velocity is measured by comparing the frequency output of a small alternator driven by the rotor shaft with known frequencies and is believed to be in error by considerably less than 1 percent. The axial velocity is measured with a conventional pitot-static tube located in the annular passage slightly upstream of the test section. Although the pitot-static tube indicates the average axial velocity in the annular passage, the average axial velocity may not be the axial velocity from which the helix angle should be computed. The possibility exists that, owing to the nonuniform spanwise lift distribution caused by the spanwise Mach number gradient, the finite tip clearance, and other effects which are unknown, an induced velocity may be present which is not indicated by the pitot-static tube.



(a) Variation of normal-force coefficient with Mach number.

FIGURE 13.—Typical normal-force and pitching-moment data obtained from pressure-distribution measurements for the NACA 66-006 airfoil at several angles of attack in the Langley annular transonic tunnel.



(b) Variation of pitching-moment coefficient with Mach number.

FIGURE 13.—Concluded.

CONCLUDING REMARKS

The development of the Langley annular transonic tunnel, a wind-tunnel facility in which Mach numbers from 0.6 to slightly over 1.0 were achieved by rotation of a two-dimensional test model in an annular passage between two concentric cylinders, has been described, and comparisons have been presented of data obtained from this facility with data from other sources.

Data obtained for several two-dimensional airfoil models in the Langley annular transonic tunnel at subsonic and sonic speeds were found to be in reasonable agreement with experimental data from other sources and with theory for nonlifting conditions and for lifting conditions when comparisons are made for equal normal-force coefficients; however, angle-of-attack measurements appeared to be subject to errors of uncertain origin.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., *January 19, 1953.*

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