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Experimental Investigation of the Subsonic High-Altitude Operation of the NASA Lewis 10- by 10-Foot Supersonic Wind Tunnel

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EXPERIMENTAL INVESTIGATION OF THE SUBSONIC HIGH-ALTITUDE OPERATION OF THE NASA LEWIS 10- BY 10-FOOT SUPERSONIC WIND TUNNEL

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SUMMARY

An experimental investigation was conducted in the NASA Lewis 10- by 10-Foot Supersonic Wind Tunnel during subsonic tunnel operation in the aerodynamic cycle to determine the test section flow characteristics near the Advanced Turboprop Project propeller model plane of rotation. The investigation used an eight-probe pitot-static flow survey rake to measure total and static pressures at two locations in the wind tunnel: the test section, where pressures were measured from the centerline of the tunnel to a point 4.5 ft from the centerline; and the bellmouth section, which is upstream of the two-dimensional flexible-wall nozzle, in a line from points 2 ft above the tunnel centerline to 2.5 ft below the centerline. In the test section, pressure measurements were obtained at two azimuthal positions, 180° and 270°, corresponding to locations from the test section centerline to the floor and from the test section centerline to the left sidewall, respectively, as seen from a point upstream of the flow survey rake looking downstream. A cone angularity probe was mounted on the flow survey rake 2 ft from the tunnel centerline to measure the flow angularity in the test section. The evaluation was conducted at tunnel Mach numbers from 0.10 to 0.35 and at three operating altitudes from 2000 to 50 000 ft, which corresponded to tunnel reference total pressures from 1960 to 245 psfa, respectively.

The results of this experimental investigation indicate a total-pressure loss area in the center of the test section and a static-pressure gradient from the test section centerline to the wall. The total- and static-pressure differences in the test section were observed at each of the tunnel operating altitudes investigated, but they diminished at lower tunnel velocities. The total-pressure loss area in the center of the tunnel was present in the bellmouth section as well, which indicated the loss mechanism is not the tunnel nozzle. The flow in the test section is essentially axial since very small flow angles were measured. In addition, the results indicate that because of pressure gradients that exist between the tunnel bellmouth section and test section, a correction to the tunnel total and static pressures must be applied in order to determine accurate free-stream conditions at the test section centerline.

INTRODUCTION

In support of the NASA Lewis Research Center Advanced Turboprop Project propeller model test program, an experimental investigation was conducted in the 10- by 10-Foot Supersonic Wind Tunnel to determine the flow characteristics in the wind tunnel test section during subsonic tunnel operation in the aerodynamic cycle at different pressure altitudes. The supersonic operating characteristics of the wind tunnel have been documented previously (ref. 1). The subsonic tunnel operating conditions are determined from the tunnel total pressure obtained in the wind tunnel bellmouth section, which is upstream of the test section, and from the tunnel static pressure obtained in the test section. The tunnel operating conditions at the test section centerline are assumed to be free-stream conditions.

In propeller testing, the aerodynamic performance of the propeller models is expressed in terms of efficiency. The propeller efficiency is dependent upon several performance parameters, including the propeller power and the free-stream velocity. At constant rotational speed at the propeller takeoff design point, a 1 percent change in free-stream velocity changes the power absorbed by the propeller approximately 1.3 percent and changes the propeller efficiency approximately 0.5 percent. Therefore, an accurate value of velocity at the tunnel centerline must be known to correctly calculate the propeller performance. In addition, if the velocity distribution in the propeller plane is nonuniform, an average value for the velocity must be calculated to obtain the correct propeller performance parameters.

The experiment consisted of identifying any differences between the pressures measured at the test section centerline, where the propeller model operates, and the pressures used to determine the tunnel operating conditions, which are obtained in the bellmouth section and at the test section ceiling. In addition, the pressure and, therefore, velocity distributions were determined in the test section from the centerline to the wall at the location of the propeller model plane. Tunnel Mach numbers ranging from 0.10 to 0.35 and simulated pressure altitudes in the test section from 2000 to 50 000 ft were investigated. Pressures were measured by using a flow survey rake consisting of eight pitot-static probes. The amount of swirl in the flow was also determined by using a cone angularity probe mounted near the middle of the flow survey rake.

This report presents the results of the investigation conducted in the test section and bellmouth section of the 10- by 10-Foot Supersonic Wind Tunnel. The results of the flow survey in the test section and the bellmouth section are presented at each tunnel operating condition investigated. In addition, pressure calibration results that were derived from the test section flow surveys and that are used to determine the operating conditions at the test section centerline are given.

SYMBOLS

M Mach number

P pressure, psf

T temperature, °R

V velocity, ft/s

Subscripts:

cl wind tunnel centerline condition

o free-stream condition

ref wind tunnel reference condition

s static condition

t total condition

APPARATUS AND PROCEDURE

Apparatus

The schematic diagram of the wind tunnel circuit presented in figure 1 shows the general location of the 10- by 10-ft test section and the two-dimensional flexible-wall nozzle used to set Mach number for supersonic tunnel operation.

An enlarged diagram of the test section area is shown in figure 2. In the figure, tunnel station 0 defines the test section floor datum line. The test section begins just in front of the floor datum line. Flow surveys were made in the test section at tunnel station 51 and in the wind tunnel bellmouth section, just upstream of the beginning of the flexible-wall nozzle, at tunnel station -931. The tunnel stations are given in inches from the floor datum line, and the negative sign indicates a location upstream of the datum line. A sketch of the pitot-static rake used for these flow surveys is shown in figure 3. The tunnel station numbers defining the location of the flow survey rake actually define the location of the rake main support tube.

The rake was mounted on the Advanced Turboprop Project propeller test rig (PTR) in the test section at tunnel station 51 as shown in figure 4. This photograph was taken from the test section floor looking up at the flow survey rake located in a horizontal, or 270°, azimuthal position. The left sidewall with the tunnel windows is shown at the left in the photograph. A simple cone-shaped geometry was used as a spinner simulator (figs. 3 and 4) and was attached to the rake at the PTR mounting location to aid in flow transition over the PTR front end. The PTR strut was mounted to the tunnel ceiling-support structure and was located at zero angle-of-attack on the tunnel centerline. The flow survey rake spanned the distance from the tunnel centerline to within 6 in. of the test section wall. In addition, the mounting arrangement allowed the azimuthal position of the flow survey rake to be varied by unlocking the rotating shaft of the PTR.

The first three of the eight probes of the survey rake (fig. 3), those closest to the simulated spinner, were placed 12 in. apart, while the remaining five probes were placed 6 in. apart. In this way, more detail of the flow nearer the test section wall could be obtained. Each probe consisted of four static-pressure taps located 90° apart on the probe housing and a total-pressure tap located at the front and inside of the probe housing. A cone angularity probe (figs. 3 and 4) was attached to the third pitot-static probe away from the center probe on the flow survey rake (fig. 3) to measure flow angles in the test section. The probe consisted of four static-pressure taps located 90° apart on a 10° half-angle cone head with a total-pressure tap located at the cone vertex. Figure 5 gives a closeup view of the cone probe. Two of the static-pressure locations on the probe head can be seen at the top and at the bottom of the probe head. Local flow angles were determined from the probe calibration and the difference in static pressure measured at locations 180° apart on the cone probe.

In the tunnel bellmouth section, the pitot-static flow survey rake was located at tunnel station -931 and was firmly mounted vertically from the tunnel ceiling to the tunnel floor. A photograph of this installation (looking downstream) is shown in figure 6. The tunnel two-dimensional flexible-wall nozzle section begins just behind the pitot-static rake location. The left wall of the tunnel nozzle is seen as the dark region in the photograph. The four tunnel rakes which obtain the tunnel total pressure and total temperature can also be seen in the photograph. One tunnel rake was mounted on each of the tunnel walls. The flow survey rake was mounted so that total-pressure data were obtained up to 2 ft above the tunnel centerline by using four pitot-static probes and down to 2.5 ft below the tunnel centerline by using three probes with the fourth probe on the rake (fig. 3) located on the tunnel centerline.

Instrumentation

The pressure measurements in the wind tunnel are obtained with the Electronically Scanned Pressure (ESP) Measurement System. A single pressure measurement is obtained for each ESP pressure port by using a dedicated pressure transducer. These transducers have a full-scale measuring capability of 30 psi, and they are accurate to within ±0.15 percent of the full scale.

The permanent wind tunnel instrumentation, which obtains the tunnel operating conditions, consists of the four rakes in the wind tunnel bellmouth section at tunnel station -964 (fig. 6) and a static-pressure tap located in the ceiling of the test section at tunnel station 32. Each of the four tunnel bellmouth rakes measures total pressure and total temperature. The total temperature measurements in the wind tunnel are obtained from thermocouples mounted in the bellmouth rakes. In addition, two differential pressure transducers measure the difference between the tunnel bellmouth rake total pressure at tunnel station -964 and the test section static pressure at tunnel station 32. Each differential pressure transducer has a different full-scale range capability (2.22 psid for one and 0.5 psid for the other) with a full-scale accuracy of ±0.15 percent for both.

The ESP measurement system also obtained total- and static-pressure data from each pitot-static probe and total-pressure data from the cone angularity probe. A more accurate means of obtaining pressure difference data was provided by installing differential pressure transducers between several locations: (1) the flow survey rake total pressure and static pressure at the tunnel centerline probe location by using 2.5 psid ±0.15 percent and 0.5 psid ±0.15 percent differential pressure transducers; (2) the tunnel bellmouth total pressure and the total pressure at the tunnel centerline probe by using a 0.2 psid ±0.15 percent differential pressure transducer; (3) the test section static pressure and the static pressure at the tunnel centerline probe by using a

2.0 psid ±0.15 percent differential pressure transducer; and (4) the cone angularity probe static-pressure differences by using two 2.0 psid ±0.15 percent differential pressure transducers.

Procedure

The investigation consisted of an evaluation of the flow characteristics in both the wind tunnel test section and the bellmouth section. The test section evaluation was conducted at tunnel station 51 which closely corresponded to the location of the Advanced Turboprop Project high-speed propeller model plane of rotation at tunnel station 46. The pitot-static flow survey rake obtained total- and static-pressure data at two azimuthal locations. These locations were 180° (from the tunnel centerline to the tunnel floor) and 270° (from the tunnel centerline to the left tunnel sidewall) as seen from a position upstream and looking downstream (fig. 4). The pressure surveys were taken across a range of tunnel Mach numbers from 0.10 to 0.35 and a range of tunnel total pressures from 1960 to 245 psfa. In addition, the flow angularity in the test section was measured during the test section investigation approximately 2 ft from the tunnel centerline by using the cone angularity probe mounted to the flow survey rake (fig. 3). Table I summarizes the test conditions at which data were taken in the test section for each flow survey rake azimuthal position.

The pressure survey data obtained during the evaluation define the flow conditions in the test section. During wind tunnel operation, a single value of tunnel total pressure and tunnel total temperature is determined for each tunnel operating condition by arithmetically averaging the four measurements of total pressure and total temperature obtained from the tunnel bellmouth rakes. The tunnel static pressure is calculated from the tunnel total pressure and a total pressure minus static pressure measurement obtained from the differential pressure transducers. Incompressibly, this pressure difference is equal to the tunnel dynamic pressure and is referred to as the "tunnel Q". The smaller range differential pressure transducer measured the tunnel Q at lower tunnel flow velocities and at higher tunnel operating altitudes. The tunnel total and static pressures, the tunnel Q, and the tunnel total temperature (all of which define the tunnel operating conditions) are referred to as the tunnel reference conditions, or just the reference conditions. From the reference conditions, a reference flow velocity in the test section is calculated. The tunnel centerline total- and static-pressure data measured in the test section with the centerline probe from the pitot-static flow survey rake are then compared with the reference pressures. Based on this comparison, appropriate corrections to account for any pressure differences between the tunnel bellmouth section and the tunnel test section are then applied to the reference pressures to obtain the actual operating conditions present at the test section centerline. These centerline operating conditions are the free-stream conditions.

In addition, based on the pitot-static survey rake pressure and velocity distribution results obtained, corrections to allow for any transverse pressure gradients in the test section can be applied to the free-stream conditions to obtain the local flow conditions at any location in the flow survey plane. In this way, the total- and static-pressure gradients present in the tunnel which affect the flow characteristics in the test section and, therefore, the calculation of the free-stream conditions, are taken into account.

The tunnel bellmouth section evaluation using the pitot-static flow survey rake mounted at tunnel station -931 included an evaluation of only the total-pressure characteristics of the tunnel at 1960-psfa reference total pressure and at reference Mach numbers of 0.15, 0.25, and 0.35. The tunnel conditions at which the bellmouth section data were obtained are also shown in table I.

RESULTS AND DISCUSSION

Test Section Pressure Distribution Results

The test section evaluation was conducted at altitudes from 2000 to 50 000 ft, corresponding to reference total pressures from 1960 to 245 psfa, for reference Mach numbers from 0.10 to 0.35, and at two azimuthal positions, 180° and 270°. The results of the evaluation using the pitot-static flow survey rake in the test section at 2000-ft altitude, or 1960-psfa reference total pressure, are shown in figures 7 to 9.

Figure 7 shows the distribution of total pressure present in the test section from the tunnel centerline to the wall at reference Mach numbers from 0.10 to 0.35 at 1960-psfa reference total pressure. As indicated in the figure, a variation in total pressure is present in the test section with the total-pressure ratio results indicating a total-pressure loss area exists at the center of the test section. The total pressure increases outward from this low to a peak, and then it decreases toward the tunnel wall or ceiling. This total-pressure variation is present at both the 180° and the 270° azimuthal locations, and is probably due, in part, to the losses associated with the wind tunnel primary compressor and cooler and the large butterfly valve which are all located upstream of the test section. These are identified as Compressor 1, Cooler 2, and the Butterfly Valve in figure 1. At the 180° location, which is from the test section centerline to the floor, the highest total pressure is near a point 3.5 ft from the test section centerline. Beyond this point, the total pressure starts to drop again indicating the edge of the tunnel wall boundary layer. However, at the 270° location, or from the test section centerline to the sidewall, the highest total pressure is near a point 3 ft from the centerline. The discrepancy in the location of highest total pressure is probably due to the larger boundary layer present along the tunnel walls at both the 90° and 270° azimuthal locations; the larger boundary layer is most likely associated with the tunnel two-dimensional flexible-wall nozzle.

Also evident in figure 7 is the trend toward more uniform flow in the test section, meaning a smaller total-pressure loss, at lower reference Mach numbers. However, even at the lowest reference Mach number of 0.10, a small total-pressure loss area is still present. The largest loss in total pressure occurs at a reference Mach number of 0.35 and is on the order of 0.37 percent from the center of the test section to the location of peak total pressure, based on the flow survey results at the 180° survey position. The 270° survey position results indicate a total-pressure loss on the order of 0.30 percent at the center of the test section. Table II summarizes the magnitude of the total-pressure loss near the center of the test section for each reference Mach number investigated at both azimuthal flow survey rake positions.

Figure 8 shows the distribution of static pressure present in the test section from the tunnel centerline to the wall at reference Mach numbers from

0.10 to 0.35 at 1960-psfa reference total pressure. The static-pressure ratio results indicate a static-pressure gradient from the center of the test section to the walls. The data show the static pressure continually falls from the test section centerline to the walls, and the gradient is somewhat larger at 270° than at 180° at Mach number 0.35. Also indicated is the smaller static-pressure gradient at lower reference Mach numbers. The largest static-pressure difference again occurs at a reference Mach number of 0.35, is approximately a decrease of 0.36 percent in static pressure outward from the test section centerline to the sidewall, and is on the order of a 0.27 percent decrease in static pressure outward from the test section centerline to the floor. A summary of the static-pressure differences at each flow survey location for each reference Mach number is also presented in table II.

The distribution of velocity in the test section from the tunnel centerline to the tunnel walls is shown in figure 9 at reference Mach numbers from 0.10 to 0.35 at 1960-psfa reference total pressure at both azimuthal positions. The velocity-ratio results indicate a large variation in velocity across the test section; these results are similar to the total-pressure distribution results. As shown in the figure, the velocity drops off in the test section wall boundary layer, which begins approximately 3.5 ft from the center of the test section at 180° and 3 ft from the center of the test section at 270°. The velocity increases on the order of 3.0 percent from the test section centerline to the edge of the boundary layer at the 270° location at a Mach number of 0.35 and increases approximately 3.4 percent at the 180° position. The velocity profile in the test section does not vary appreciably as the reference Mach number changes. This is consistent with the total- and static-pressure results presented earlier in this section, since the velocity is a function of both the total and the static pressure. A summary of the velocity change at each flow survey location for each reference Mach number investigated is included in table II.

Figures 10 to 12 show the results of the evaluation using a pitot-static flow survey rake in the test section at 35 000-ft altitude, corresponding to 500-psfa reference total pressure, at reference Mach numbers from 0.10 to 0.35, and at both azimuthal flow survey rake positions. The evaluation procedure was the same as the procedure followed for the investigation at the 1960-psfa reference total-pressure condition. However, the results obtained at this reference total pressure exhibited increased scatter relative to the results at 1960 psfa. This is due, in part, to the data acquisition system; at the lower reference total pressure, the accuracy of the data acquisition system has a larger effect on the data. Therefore, only general trends in the results are considered at the 500-psfa reference total pressure.

The distribution of total pressure in the test section at 500-psfa reference total pressure is shown in figure 10. Again, the total-pressure ratio results at reference Mach numbers from 0.10 to 0.35 and both flow survey rake azimuthal positions are presented. As seen in the figure, the same trend in the distribution of total pressure exhibited at 1960-psfa reference total pressure is present at 500 psfa. The largest total pressure was measured near a point 2.5 ft from the center of the test section for both the 180° and 270° flow survey rake positions. The relative total-pressure loss at the center of the test section is close to the level measured at the 1960-psfa reference total-pressure condition. The trend of a smaller total-pressure loss at the lower reference Mach numbers is apparent in the figure. The thickness of the boundary layer in the test section can also be determined from the location of the peak total pressure. The boundary layer thickness is larger near the test section walls than the floor, which is consistent with the 1960-psfa reference total pressure results. In addition, the boundary layer is thicker at the 500-psfa reference total pressure condition than at 1960 psfa.

Figure 11 shows the distribution of static pressure in the test section at 500-psfa reference total pressure for Mach numbers of 0.10 to 0.35 and both azimuthal positions. The existence of a static-pressure gradient from the center of the test section to the walls at both azimuthal positions can be seen in the results. Generally, the static-pressure gradient measured is comparable to the static-pressure gradient seen at the 1960-psfa reference total-pressure condition. The figure also shows that the static-pressure gradient at the 270° location is larger than at the 180° location at Mach number 0.35. Once again, as the Mach number decreases, the static-pressure gradient at both locations decreases until, at Mach numbers below 0.15, the variation in static pressure becomes almost negligible. These results are similar to the static-pressure results obtained at 1960-psfa reference total-pressure condition.

Velocity distributions in the test section at 500-psfa reference total pressure are shown in figure 12 for the Mach numbers 0.10 to 0.35 at both azimuthal positions. The velocity distributions from the center of the test section outward follow the shape of the velocity distributions at 1960 psfa. As at 1960 psfa, the velocity distributions also follow the trend of the total-pressure distributions. The edge of the boundary layer, near a point 3 ft from the test section centerline at 180° and 2.5 ft from the centerline at 270° at Mach number 0.35, is apparent in the figure.

Additional total-pressure, static-pressure, and velocity distribution data were obtained in the test section at 50 000-ft altitude, or 245-psfa reference total pressure. However, a large amount of scatter was introduced into the test section calibration results; the data acquisition system used could not measure the low pressures at this pressure altitude with the accuracy needed to calculate the small differences from two nearly equal pressures. Therefore, pressure and velocity distribution results are not presented for the 245-psfa reference total-pressure condition.

Results from the data obtained by using the cone angularity probe mounted on the flow survey rake indicated that the flow is essentially axial at a point 2 ft from the center of the test section. The results indicated the maximum flow angles measured less than 1°. Since the flow angles measured were so small, the results are not shown here, and essentially axial flow is assumed to exist in the test section at all values of reference Mach numbers and reference total-pressure conditions.

Test Section Pressure Calibration Results

As previously shown, variations in total and static pressure are present in the test section from the centerline outward to the tunnel walls. The reference static pressure is obtained at the test section ceiling at tunnel station 32. Total-pressure variations also are present from the test section centerline to the tunnel bellmouth section, where the reference total pressure is obtained at tunnel station -964. Thus, to calculate accurate free-stream conditions at each tunnel operating condition during subsonic testing, the reference pressures must be corrected for the differences between the test section centerline pressures and the reference pressures that were observed during the investigation. Figures 13 and 14 present results representing the pressure corrections necessary to obtain accurate free-stream total and static pressures at the test section centerline. The pressure ratio results at the 1960-psfa reference total-pressure condition are curvefit, and the curves produced are known as the pressure calibration curves. Figure 13 shows the total-pressure calibration results at the 1960-psfa reference total-pressure condition. The total-pressure correction is presented as a function of the normalized reference Q. As indicated in the figure, the total-pressure loss, expressed as the test section centerline to reference total-pressure ratio, increases with an increase in the reference Q. The total-pressure calibration results are presented in the appendix in equation form, and they are valid for all reference total-pressure conditions during subsonic tunnel operation.

The static-pressure calibration results are shown in figure 14 at the 1960-psfa reference total-pressure condition. The static-pressure correction is presented as a function of the normalized reference Q. The results have been corrected for the upstream variations in static pressure brought about by an increase in test section blockage caused by the PTR. This was done by modeling the forward section geometry of the PTR (which includes the spinner, nacelle, and windscreen) and the wind tunnel walls and by analyzing the configuration with an axisymmetric potential flow program. The local blockage correction to the centerline static pressure was approximately a 0.094 percent decrease in the centerline static pressure at Mach number 0.35, but it decreased at lower Mach numbers. The static-pressure correction results, expressed as a test section centerline to reference static-pressure ratio, indicate an increase in the centerline static pressure with an increase in the reference Q at the 1960-psfa reference total-pressure condition. The static-pressure calibration results are presented in the appendix in equation form and are valid for all reference total-pressure conditions during subsonic tunnel operation.

Bellmouth Section Results

Figure 15 presents total-pressure distribution results in the tunnel bellmouth section for reference Mach numbers of 0.35, 0.25, and 0.15 at the 1960-psfa reference total pressure. The results indicate a loss in total pressure exists in the center of the tunnel which is similar to the total-pressure loss observed in the center of the test section at each of the reference Mach numbers investigated. The tunnel-centerline-to-reference-total-pressure ratio results in the bellmouth section are shown in figure 16. The results are given as a function of the normalized reference Q to allow comparison with the total-pressure calibration results obtained in the test section. The loss in total pressure at the tunnel centerline (fig. 16) is slightly larger than the loss observed in the test section by about 0.05 percent at a tunnel Mach number of 0.35. The bellmouth section results, therefore, verify that the total-pressure loss observed in the test section is not, in fact, an effect brought about by the tunnel two-dimensional flexible-wall nozzle but rather an effect brought about by another loss mechanism located upstream of the bellmouth section.

CONCLUDING REMARKS

The results obtained in this investigation show the existence of both total-pressure losses and static-pressure gradients in the test section of the NASA Lewis 10- by 10-Foot Supersonic Wind Tunnel during subsonic tunnel operation in the aerodynamic cycle. These pressure differences exist at every tunnel pressure altitude, or reference total-pressure condition, but diminish at lower tunnel velocities. Additional results from the investigation conducted in the tunnel bellmouth section indicate that the area of total-pressure loss at the center of the tunnel is present in the bellmouth section as well as in the test section. The total-pressure loss mechanism, therefore, is located upstream of the tunnel two-dimensional flexible-wall nozzle and is not a result of nozzle flow. The loss mechanism may be associated with the large butterfly bypass valve, the primary compressor, the coolers, or all three located in the tunnel circuit.

In addition, the results indicate that a pressure difference exists between the total pressure measured at the test section centerline and the reference total pressures measured in the tunnel bellmouth section, and also between the centerline static pressure measured and the reference static pressure obtained at the test section ceiling. Once again, these pressure differences diminish at lower tunnel velocities. Therefore, with the existence of these differences between the reference pressures and the test section centerline pressures, the reference pressures must be corrected by using calibration equations (appendix) in order to calculate accurate free-stream conditions at the test section centerline.

The total- and static-pressure variations present in the test section from the centerline to the tunnel walls, as well as from the test section centerline to the tunnel bellmouth section, are sufficient to produce large variations in the velocity distributions in the test section. These velocity variations contribute to errors that may be significant, depending on the size of the propeller model being tested, in terms of the amount of error introduced into the calculation of the propeller model efficiency. At the center of the test section, the pressure gradients are small enough to allow testing of the Advanced Turboprop Project high-speed propeller models at zero angle-of-attack with only very small errors introduced into the propeller efficiency calculations. At angle-of-attack, or during testing of the low-speed propeller models, an averaged local velocity must be calculated.

SUMMARY OF RESULTS

An experimental investigation was conducted in the NASA Lewis 10- by 10-Foot Supersonic Wind Tunnel during subsonic tunnel operation to determine the test section flow characteristics near the Advanced Turboprop Project propeller model plane of rotation. The investigation involved using an eight-probe pitot-static flow survey rake to measure total and static pressures at two locations in the wind tunnel: the test section, where pressures were measured from the tunnel centerline to within 0.5 ft of the tunnel wall; and the bellmouth section upstream of the tunnel two-dimensional flexible-wall nozzle, where pressures were measured in a line from 2 ft above the tunnel centerline to 2.5 ft below the centerline. In the test section, pressure measurements were obtained at two azimuthal positions, 180° and 270°; these corresponded to surveys from the test section centerline to the floor and from the test section centerline to the left sidewall, respectively (as seen from a point upstream of the flow survey rake looking downstream). A cone angularity probe was mounted on the flow survey rake 2 ft from the tunnel centerline to measure the flow angularity in the test section. The evaluation was conducted at tunnel Mach numbers from 0.10 to 0.35 and at three pressure altitudes from 2000 to 50 000 ft, corresponding to tunnel total pressures from 1960 to 245 psfa, respectively. The results of the investigation are summarized as follows:

1. An area of total-pressure loss exists in the center of the test section and near the tunnel walls at all tunnel pressure altitudes. The near-wall loss is associated with the wall boundary layer which varies in size because of flow characteristics associated with the tunnel two-dimensional flexible-wall nozzle. The maximum total-pressure loss at the center occurs at a Mach number of 0.35 and is approximately 0.37 percent of the test section centerline total pressure at an azimuthal location of 180° and 0.30 percent at the 270° azimuthal location. The total-pressure loss diminishes and the wall boundary layer thickness decreases at lower tunnel velocities.

2. A static-pressure gradient exists from the center of the test section outward to the tunnel walls at all tunnel pressure altitudes. The static pressure decreases outward from the test section centerline. The maximum static-pressure difference occurs at Mach number 0.35 and is approximately 0.27 percent of the test section centerline static pressure at the 180° location and 0.36 percent at the 270° location. The pressure gradient diminishes at lower tunnel velocities.

3. A difference in pressure exists between the test section centerline total and static pressures and the tunnel reference total and static pressures. Between the tunnel bellmouth section and the test section centerline, the total pressure decreases approximately 0.46 percent at a tunnel Mach number of 0.35. At the same Mach number, the static pressure decreases approximately 0.44 percent between the test section centerline and the tunnel reference on the test section ceiling. Both pressure differences decrease at lower tunnel velocities.

4. A total-pressure loss in the center of the tunnel is also present in the tunnel bellmouth section, which is upstream of the tunnel two-dimensional flexible-wall nozzle. At tunnel Mach number 0.35, the total-pressure loss is approximately 0.51 percent of the reference total pressure, and it also diminishes at lower tunnel velocities.

5. The test section boundary layer is fairly large, and its thickness varies from the tunnel floor to the sidewalls. At tunnel Mach number 0.35, the boundary layer is approximately 1.5 ft thick on the tunnel floor and approximately 2 ft thick on the tunnel sidewall, but the boundary layer thickness at both locations decreases at lower tunnel Mach numbers. The thicker sidewall boundary layer is due to the tunnel two-dimensional flexible-wall nozzle that is upstream of the test section.

6. The test section flow is essentially axial since flow angles of less than 1° were measured.

REFERENCES

1. Aiello, R.A.: NASA Lewis 10- by 10-Foot Supersonic Wind Tunnel. NASA TM X-71625, 1974.

APPENDIX - WIND TUNNEL PRESSURE INSTRUMENTATION CALIBRATION EQUATIONS

The tunnel free-stream static pressure is a function of the tunnel reference static pressure, the normalized reference Q, which is expressed as K, and the derived constants A_1, A_2, and A_3. The free-stream static pressure equation is

$$P_{s,o} = (A_1 + KA_2 + K^2A_3)P_{s,ref}$$

where

K normalized reference Q, $(P_{t,ref} - P_{s,ref})/P_{t,ref}$

A_1 0.99993903

A_2 0.03885731

A_3 0.14837505

The tunnel free-stream total pressure is a function of the tunnel reference total pressure, the normalized reference Q, which is K, and the derived equation constants B_1 and B_2. The equation for free-stream total pressure is

$$P_{t,o} = (B_1 + KB_2)P_{t,ref}$$

where

K normalized reference Q, $(P_{t,ref} - P_{s,ref})/P_{t,ref}$

B_1 1.0000372

B_2 -0.05294221

The tunnel free-stream equations for Mach number, static temperature, and velocity are derived from the standard isentropic relations and are as follows

$$M_o = \left\{5\left[\left(\frac{P_{t,o}}{P_{s,o}}\right)^{2/7} - 1\right]\right\}^{1/2}$$

$$T_{s,o} = \frac{T_{t,o}}{1 + \frac{1}{2}M_o^2}$$

$$V_o = 49.021\ M_o(T_{s,o})^{1/2}$$

TABLE I. TEST CONDITIONS SUMMARY

Mach number, M_{ref}	Pressure ratio, $\frac{P_{t,ref} - P_{s,ref}}{P_{t,ref}}$	Altitude, ft; pressure, $P_{t,ref}$, psfa		
		2000; 1960	35 000; 500	50 000; 245
0.10	0.0070	a	a,b	a
.15	.0156	a,b,c	↓	↓
.20	.0275	a	↓	↓
.25	.0425	a,c	↓	↓
.30	.0605	a	↓	↓
.35	.0812	a,b,c	↓	↓

[a]Test section, 180° (vertical) flow survey rake position.
[b]Test section, 270° (horizontal) flow survey rake position.
[c]Bellmouth section.

TABLE II. - TEST SECTION PRESSURE DIFFERENCE SUMMARY AT 1960-psfa REFERENCE TOTAL PRESSURE

[Pressure and velocity ratio values represent 180° (vertical) flow survey rake position results, and values in parentheses represent 270° (horizontal) flow survey rake position results.]

Mach number, M_{ref}	Maximum total pressure ratio, $\frac{P_{t,max} - P_{t,cl}}{P_{t,cl}}$ 100, percent	Maximum static, pressure ratio, $\frac{P_{s,cl} - P_{s,min}}{P_{s,cl}}$ 100, percent	Maximum velocity ratio, $\frac{V_{max} - V_{cl}}{V_{cl}}$ 100, percent
0.35	0.37 (0.30)	0.27 (0.36)	3.4 (3.0)
.30	.28	.15	3.1
.25	.27	.10	3.7
.20	.18	.09	3.8
.15	.12 (0.09)	.04 (0.02)	3.6 (4.0)
.10	.08	-0.03	2.7

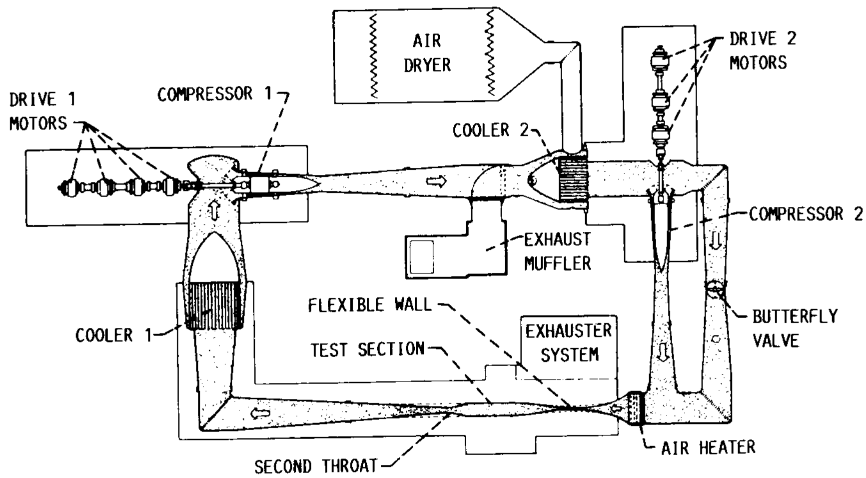

FIGURE 1. - NASA LEWIS 10- BY 10-FOOT SUPERSONIC WIND TUNNEL CIRCUIT SCHEMATIC, AERODYNAMIC CYCLE.

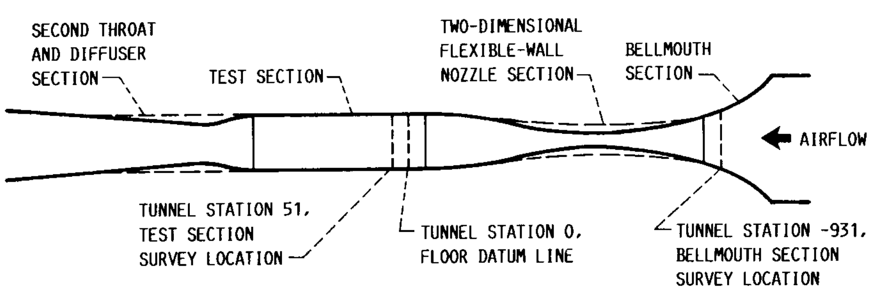

FIGURE 2. - TEST SECTION AND BELLMOUTH SECTION FLOW SURVEY LOCATIONS (TUNNEL STATION IN INCHES).

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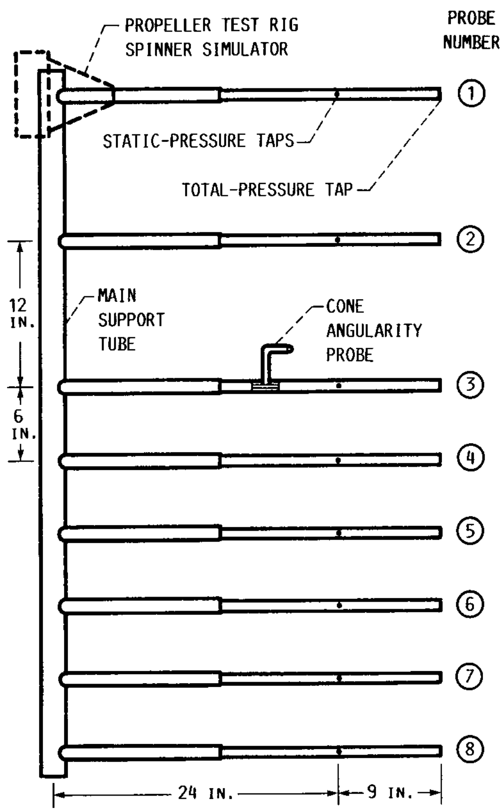

FIGURE 3. - PITOT-STATIC FLOW SURVEY RAKE.

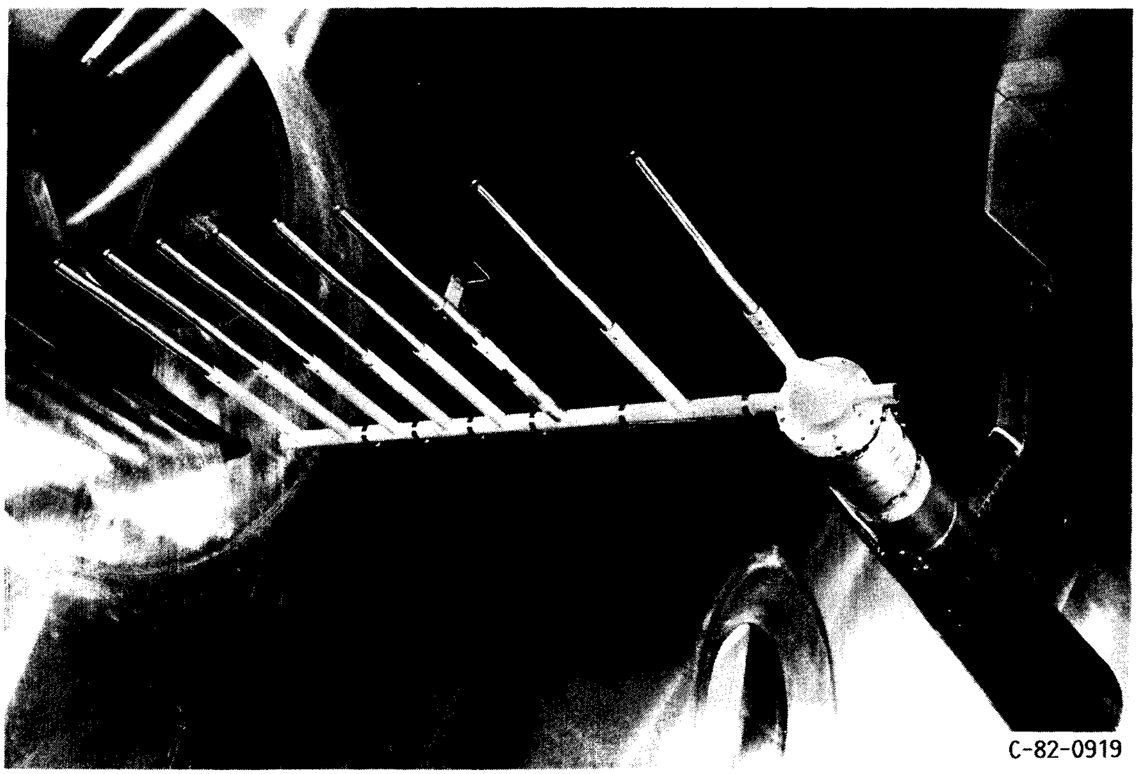

FIGURE 4. - PITOT-STATIC FLOW SURVEY RAKE MOUNTED TO THE ADVANCED TURBOPROP PROJECT SINGLE-ROTATION PROPELLER TEST RIG IN THE WIND TUNNEL TEST SECTION.

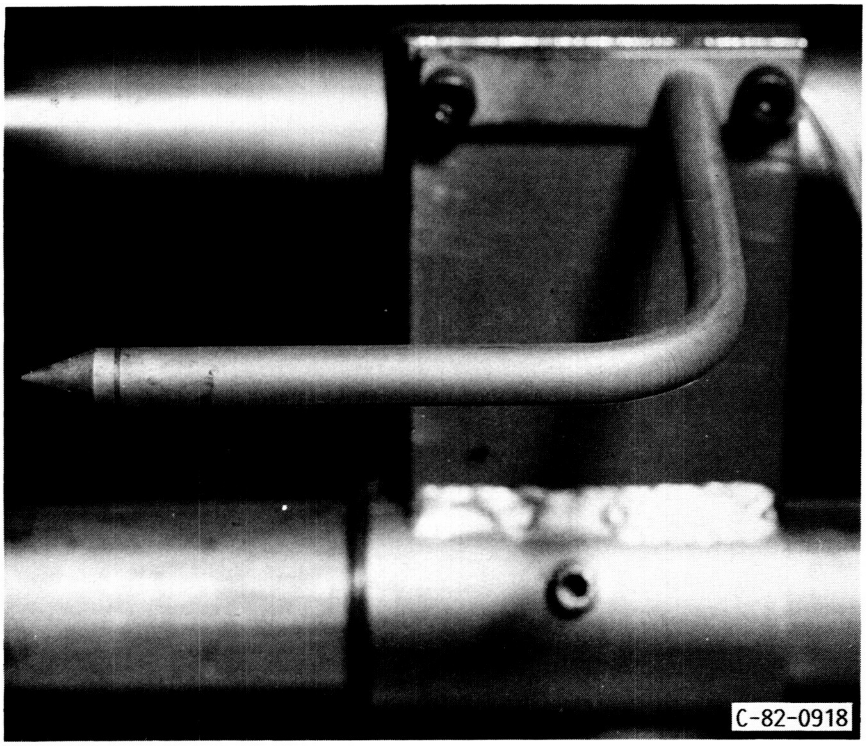

FIGURE 5. - CONE ANGULARITY PROBE.

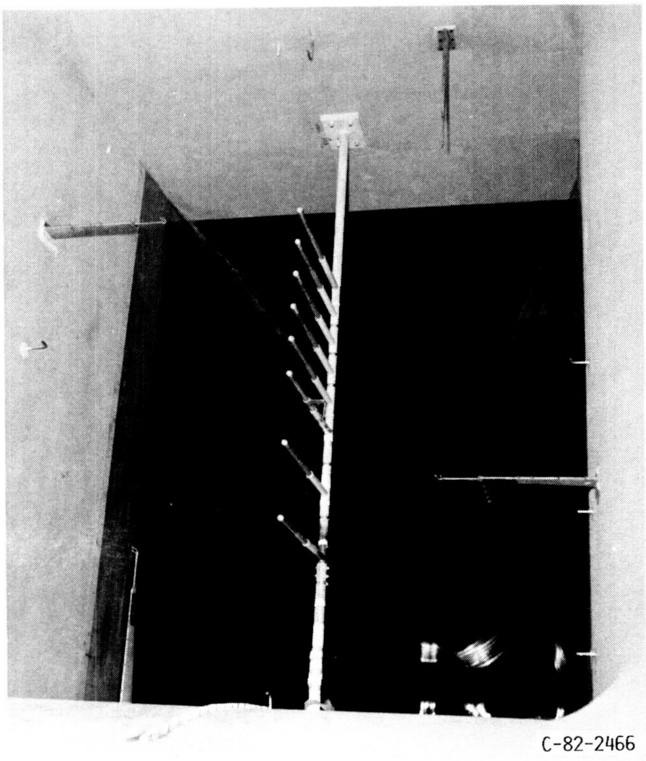

FIGURE 6. - PITOT-STATIC FLOW SURVEY RAKE MOUNTED IN THE WIND TUNNEL BELLMOUTH SECTION.

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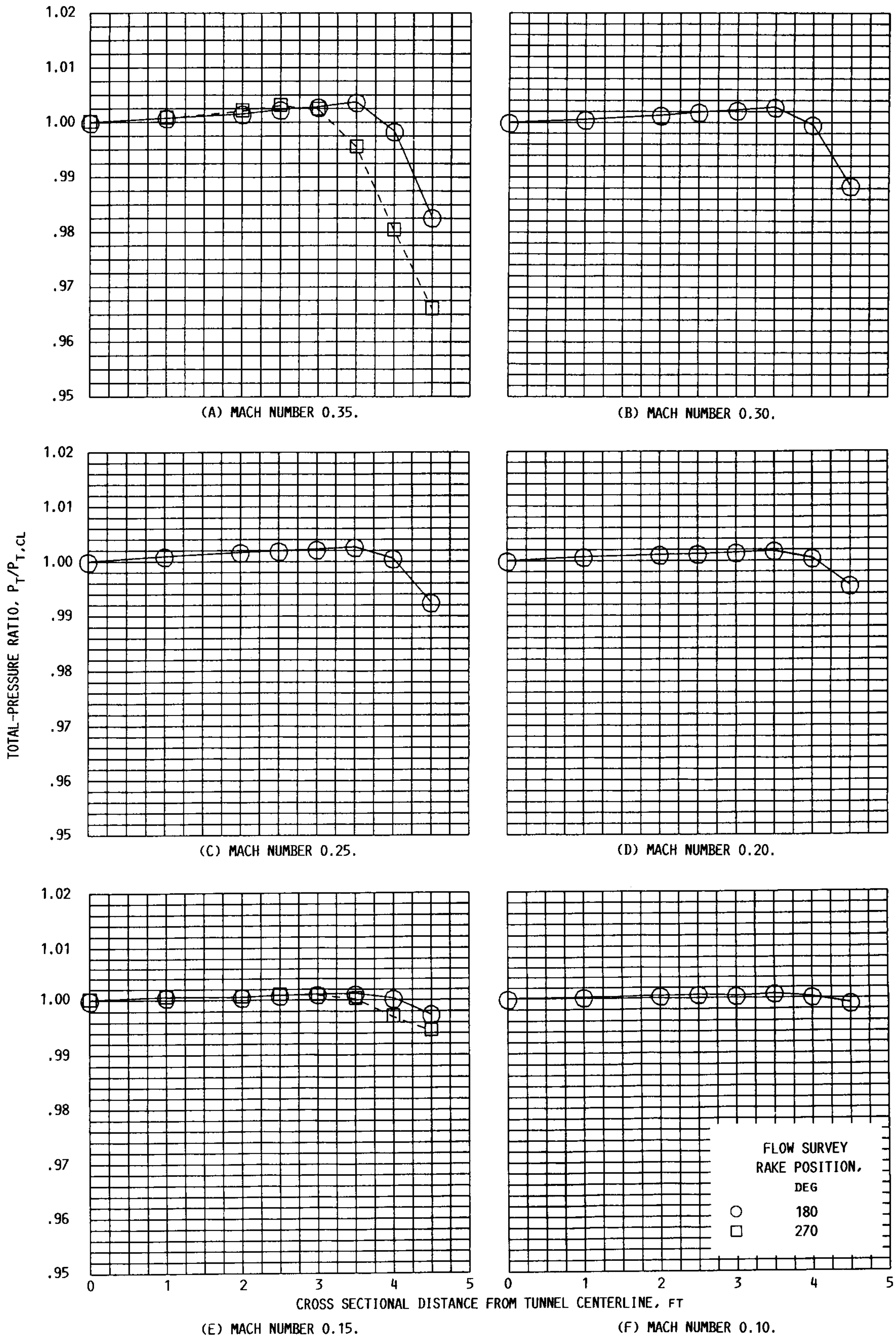

FIGURE 7. - TEST SECTION TOTAL-PRESSURE DISTRIBUTION AT 2000 FT (1960 PSFA).

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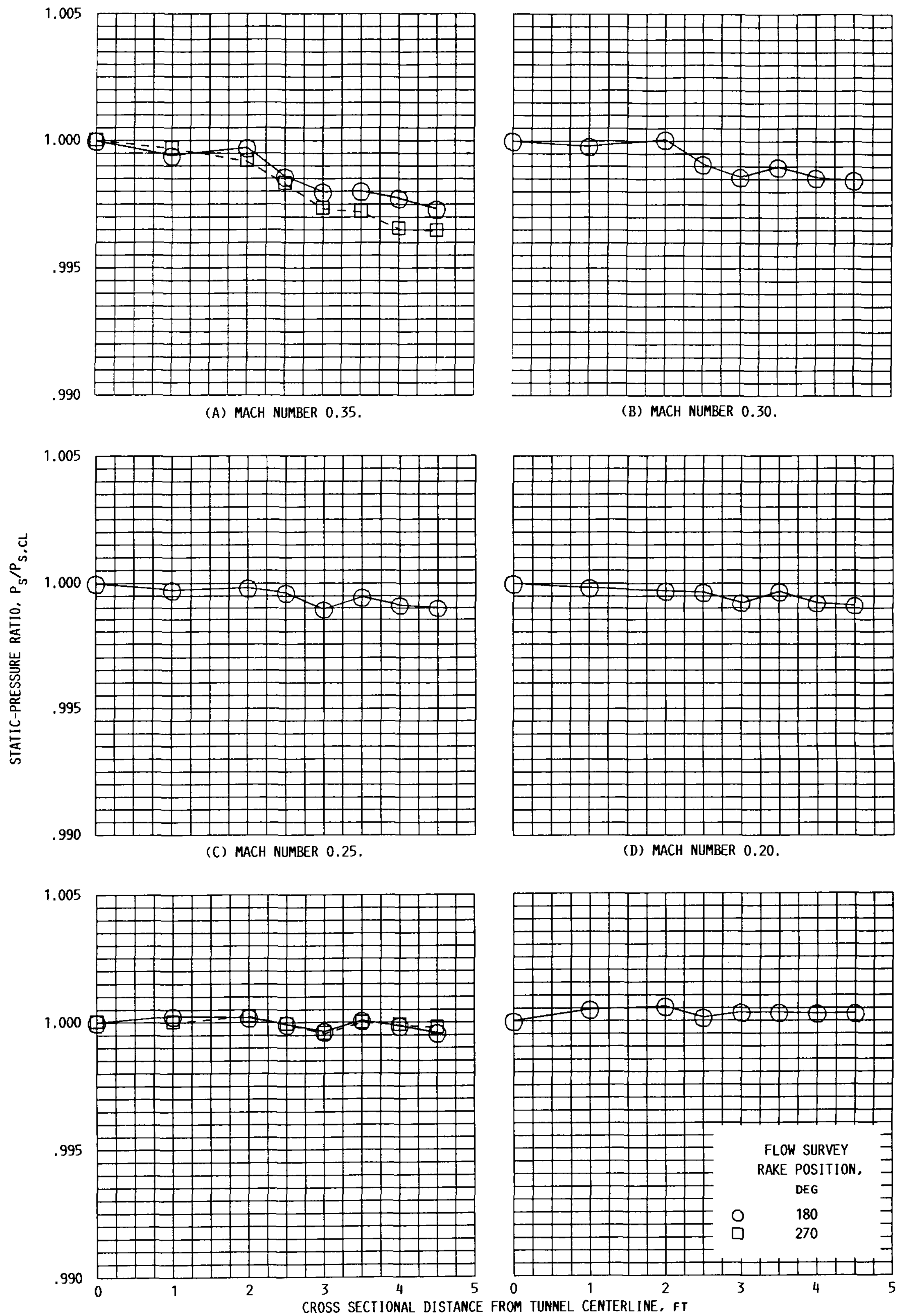

FIGURE 8. - TEST SECTION STATIC-PRESSURE DISTRIBUTION AT 2000 FT (1960 PSFA).

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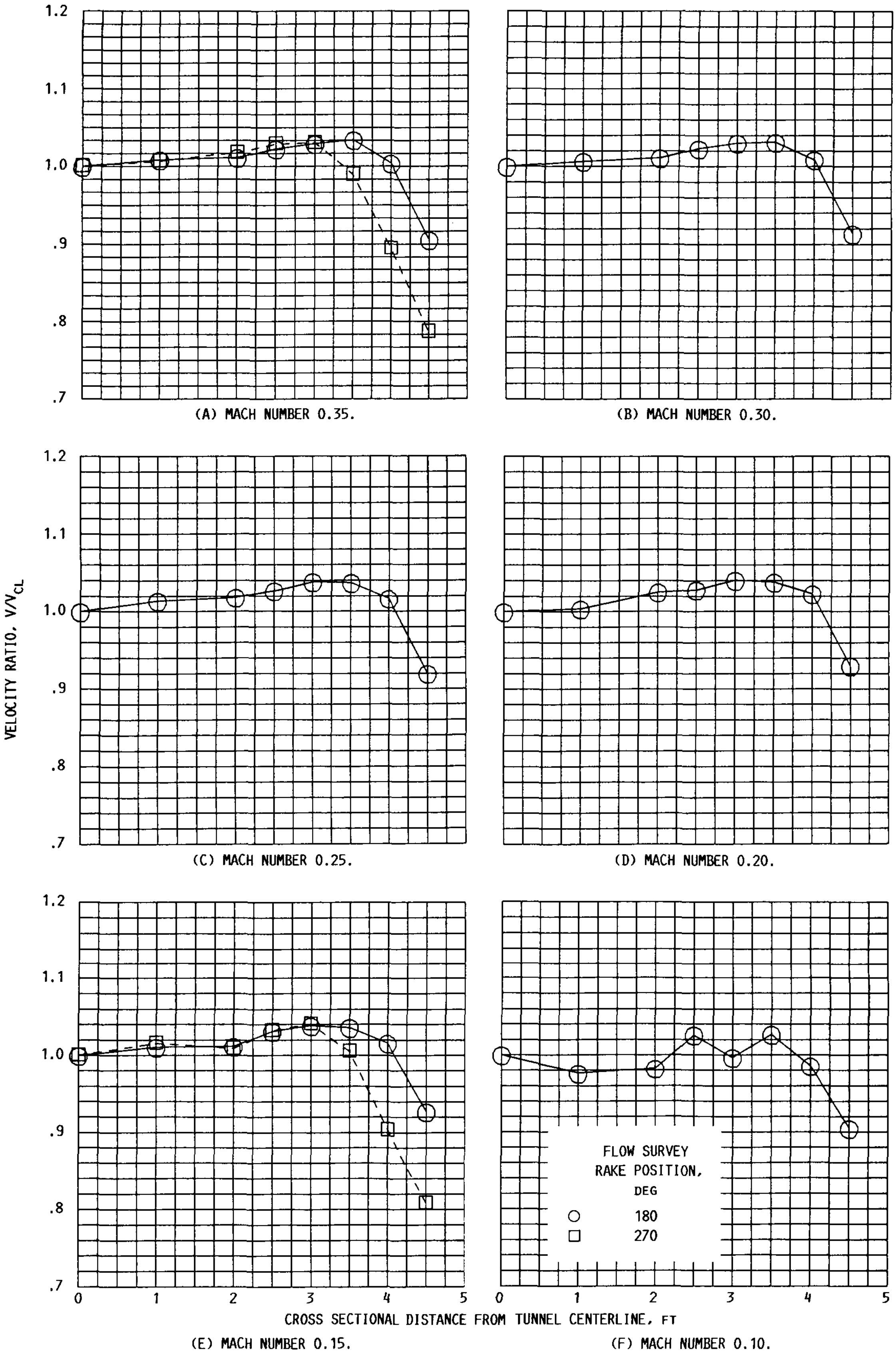

FIGURE 9. - TEST SECTION VELOCITY DISTRIBUTION AT 2000 FT (1960 PSFA).

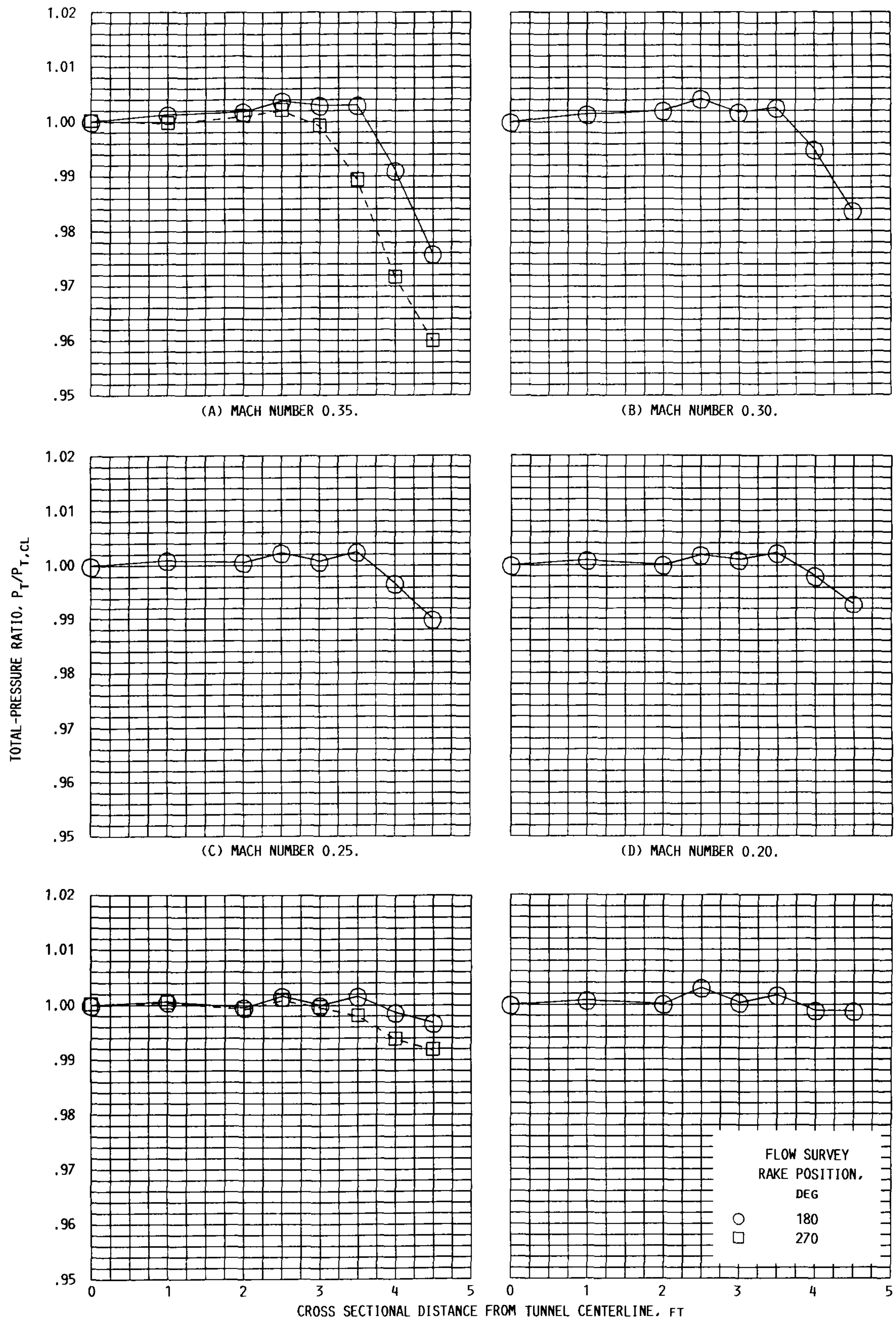

FIGURE 10. - TEST SECTION TOTAL-PRESSURE DISTRIBUTION AT 35 000 FT (500 PSFA).

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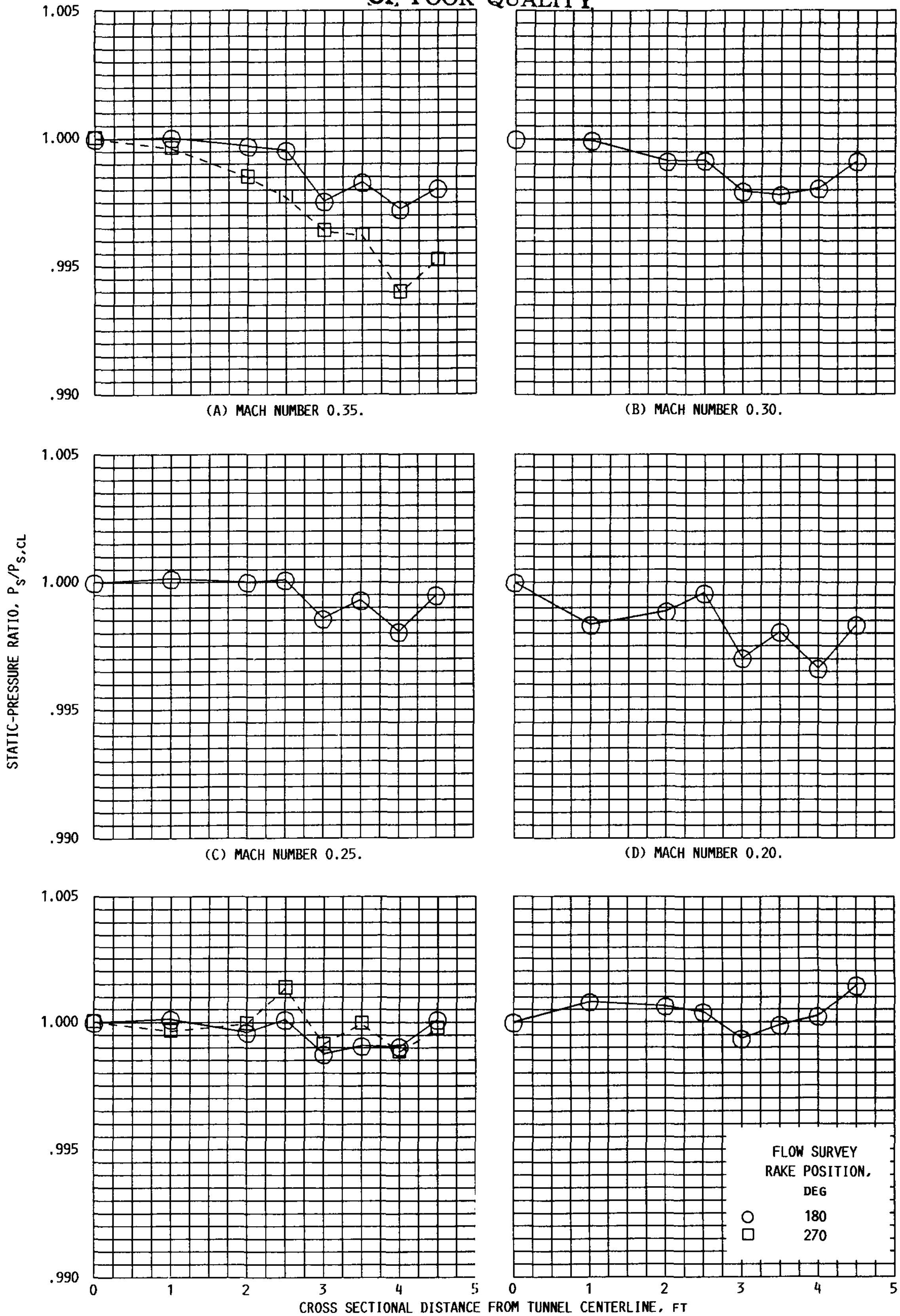

FIGURE 11. - TEST SECTION STATIC-PRESSURE DISTRIBUTION AT 35 000 FT (500 PSFA).

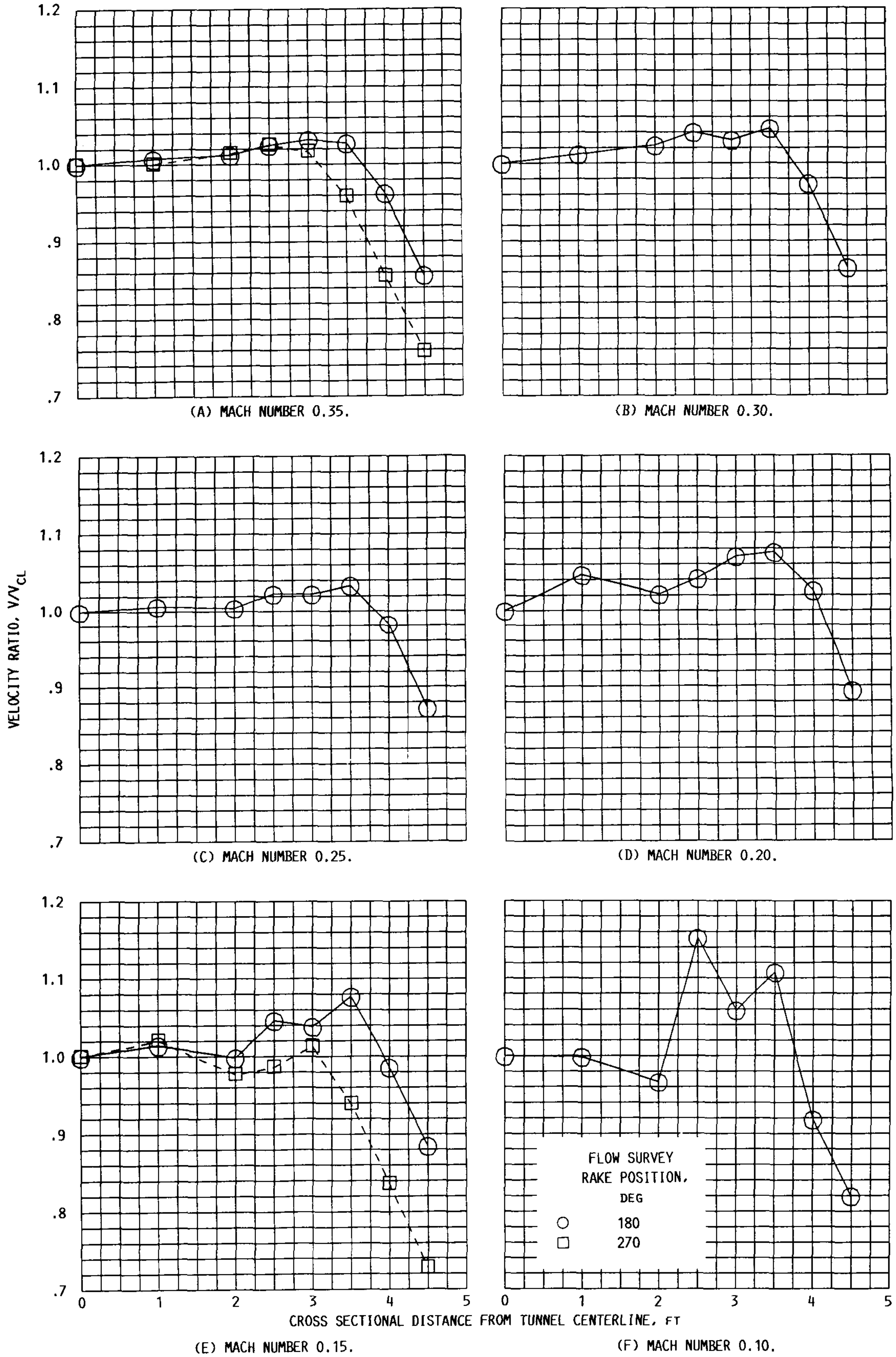

FIGURE 12. - TEST SECTION VELOCITY DISTRIBUTION AT 35 000 FT (500 PSFA).

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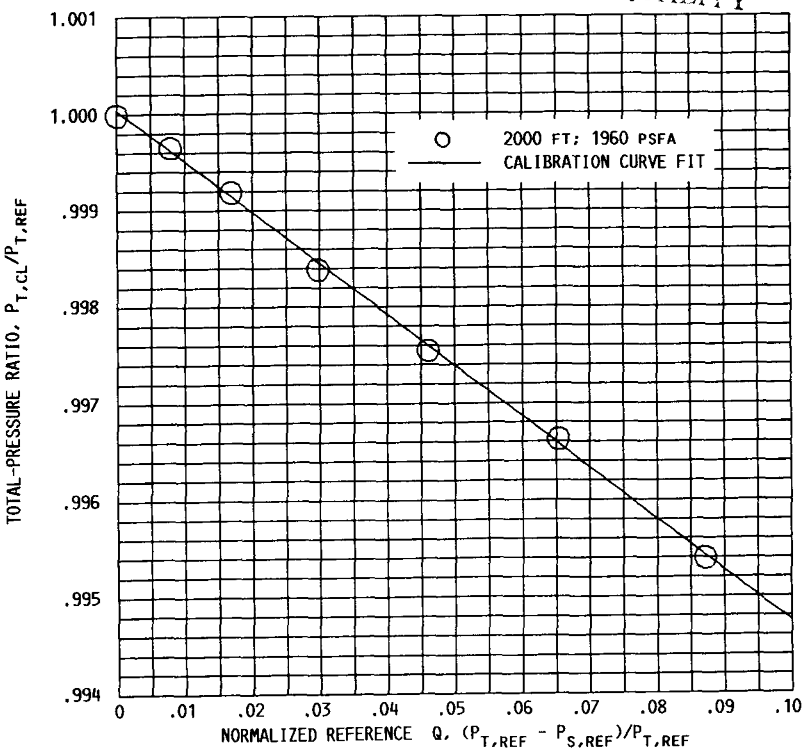


FIGURE 13. - WIND TUNNEL TOTAL-PRESSURE CALIBRATION CURVE.

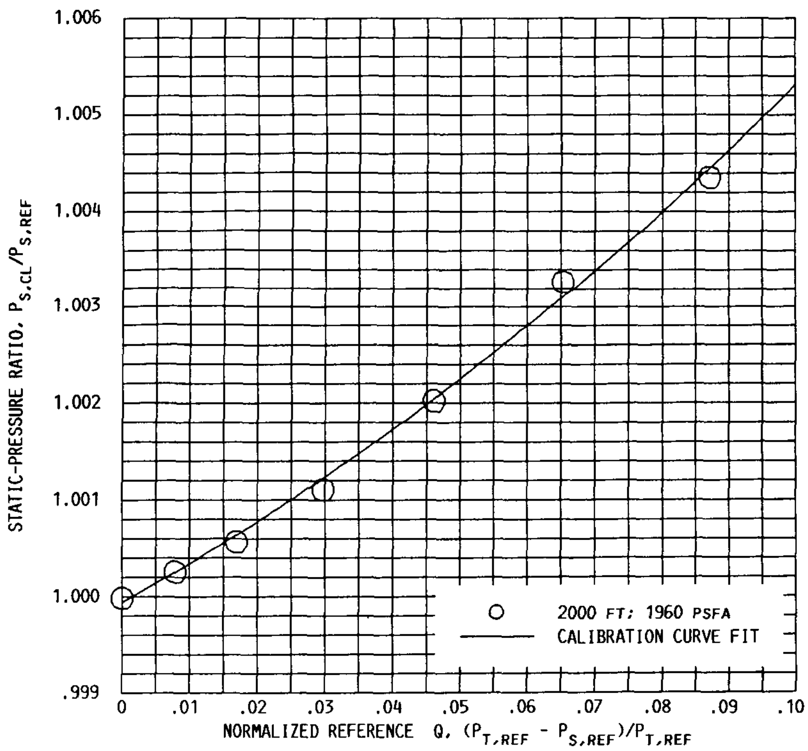
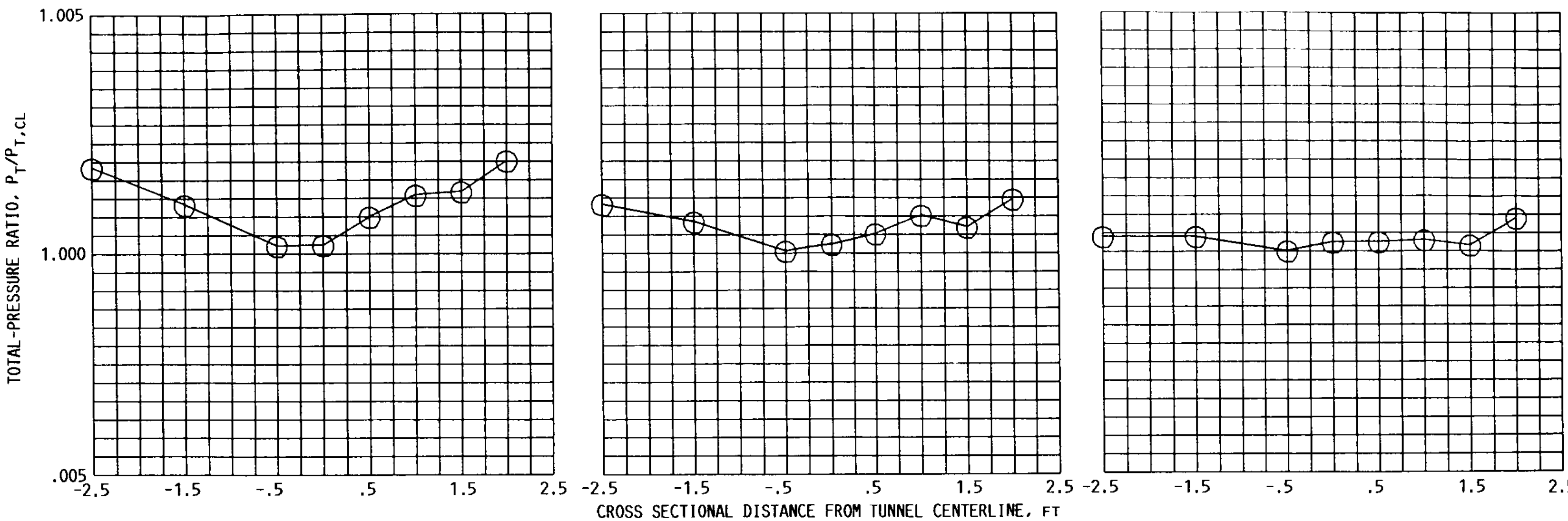


FIGURE 14. - WIND TUNNEL STATIC-PRESSURE CALIBRATION CURVE.



(A) MACH NUMBER 0.35.

(B) MACH NUMBER 0.25.

(C) MACH NUMBER 0.15.

FIGURE 15. - BELLMOUTH SECTION TOTAL-PRESSURE DISTRIBUTION AT 2000 FT (1960 PSFA).

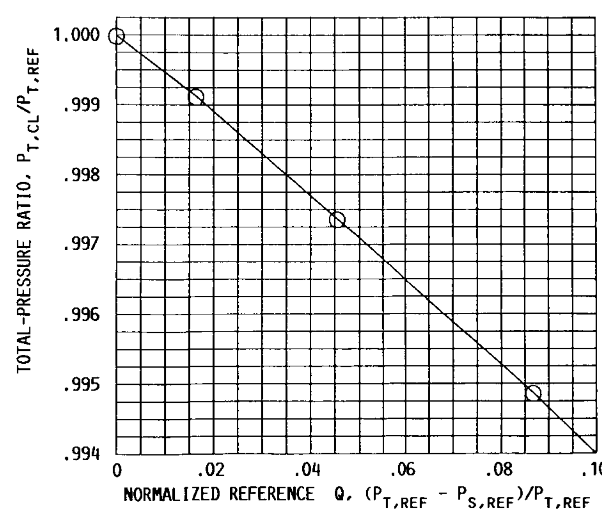


FIGURE 16. - VARIATION OF BELLMOUTH SECTION TUNNEL-CENTERLINE-TO-REFERENCE-TOTAL-PRESSURE RATIO AT 1960-PSFA REFERENCE TOTAL PRESSURE.

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15. Supplementary Notes

16. Abstract

An experimental investigation was conducted in the NASA Lewis 10- by 10-Foot Supersonic Wind Tunnel during subsonic tunnel operation in the aerodynamic cycle to determine the test section flow characteristics near the Advanced Turboprop Project propeller model plane of rotation. The investigation used an eight-probe pitot-static flow survey rake to measure total and static pressures at two locations in the wind tunnel: the test section and the bellmouth section (upstream of the two-dimensional flexible-wall nozzle). A cone angularity probe was used to measure any flow angularity in the test section. The evaluation was conducted at tunnel Mach numbers from 0.10 to 0.35 and at three operating altitudes from 2 000 to 50 000 ft, which corresponded to tunnel reference total pressures from 1960 to 245 psfa, respectively. The results of this experimental investigation indicate a total-pressure loss area in the center of the test section and a static-pressure gradient from the test section centerline to the wall. These total and static pressure differences were observed at all tunnel operating altitudes and diminished at lower tunnel velocities. The total-pressure loss area was also found in the bellmouth section, which indicates the loss mechanism is not the tunnel flexible-wall nozzle. The flow in the test section is essentially axial since very small flow angles were measured. The results also indicate that a correction to the tunnel total and static pressures must be applied in order to determine accurate freestream conditions at the test section centerline.

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