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NOMENCLATURE

c	chord of airfoil
C_p	local pressure coefficient $C_p = \dfrac{p - p_\infty}{q_\infty}$
M_{chan}	local Mach number in the channel formed by the splitter plates
M_{set}	tunnel set Mach number
M_∞	free-stream approach Mach number
OAR	tunnel open area ratio, $OAR = \dfrac{\text{wall open area}}{\text{total wall area}}$
p	local static pressure
p_∞	free-stream static pressure
q_∞	free-stream dynamic pressure
t	maximum thickness of airfoil
TSDT	transonic small-disturbance theory
TSFOIL	transonic airfoil code
γ	ratio of specific heats $\gamma = 1.4$
δ	airfoil thickness ratio $\delta = \dfrac{t}{c}$

TWO-DIMENSIONAL TRANSONIC TESTING WITH SPLITTER PLATES

Sanford Davis and Bodapati Satyanarayana

Ames Research Center

SUMMARY

The use of splitter plates for two-dimensional transonic testing in wind tunnels is investigated on a 12% biconvex airfoil section over the Mach number range 0.6 to 1.0. Measured pressure distributions are compared to transonic theory and to other experiments, including an investigation in the same facility without splitter plates. The results of the present experiment show the best agreement with theory over the entire transonic Mach number range.

INTRODUCTION

After more than a decade of relative inactivity, transonic aerodynamics is again in the forefront of research. Modern aircraft now fly routinely at transonic speeds, and the search for more efficient systems and components is more critical than ever. The revolution in computational aerodynamics has contributed greatly to this difficult area, but experimental transonic aerodynamics has not kept pace. Much more needs to be learned about ventilated wind tunnels, wall interference, and the flow over aircraft components at Mach numbers near 1. The first real breakthrough in experimental transonic aerodynamics was the development of ventilated wind tunnels in the late 40's and early 50's. A large number of these wind tunnels were designed for measuring forces on wing-bodies and other three-dimensional configurations rather than two-dimensional testing of airfoils. This resulted in many facilities which are ventilated on all four walls.

Modern research, however, is focussing in new directions. Active walls are being developed to counteract wall interference and experimental studies of two-dimensional airfoils are underway to measure flow-field details that were glossed over 20 years ago. Due to the interest in supercritical airfoils, it would be very useful to utilize the existing transonic wind tunnels for two-dimensional testing of reduced span models. One approach is to rebuild the test section to more appropriate dimensions, and this has been done in some cases. Another approach is to utilize splitter plates to guide the flow into a two-dimensional channel, forming a test section within a test section. This is the approach to be explored in this paper.

The concept of splitter plates in wind tunnels is not new; it has been considered for low speed, high lift applications by Van den Berg (ref. 1) and Anscombe and Williams (ref. 2). The concept has not been developed because the flow in the various passages formed by the plates does not correspond to the main wind-tunnel flow. As far as the authors are aware, no investigation of this problem in the transonic speed range has been attempted. In this report it is shown that a good quality transonic flow can be established by using splitter plates. In fact, for Mach numbers near unity, excellent agreement between transonic theory and experiment is shown. The problem is that the effective Mach number in the channel cannot be "calibrated" in a channel empty configuration, but must be measured along with the airfoil's pressure distribution.

Results of the current tests are compared with an experiment by Riddle[1] in the same facility. Riddle's earlier experiment used the same wings, but his configuration used a full span model instead of the splitter plates and all four walls were ventilated. Comparisons are also made with other experimental investigations of biconvex airfoils and with solutions to the Euler equation, the full potential equation, and two independent methods for solving the transonic small-disturbance equation.

APPARATUS AND MODELS

The tests were conducted in the Ames 2- by 2-Foot Transonic Wind Tunnel (ref. 3). The tunnel is a continuous flow, variable-pressure facility with a slotted test section. The data presented in this paper were taken at a total pressure of 67,000 N/m^2 (9.72 psi) and a total temperature of approximately 270 K. The Reynolds number was approximately 10 million/m.

A perspective view of the splitter plate model is shown in figure 1. The model consists of a biconvex airfoil rigidly attached to two steel end plates. The end plates are 0.517 m long × 0.610 m high × 0.012 m thick. The two end plates are positioned in the 0.610 m square test section to form a center channel and two side channels. Each plate is diverged 0.1° to account for boundary-layer growth. One of the end plates has a row of twenty-five 0.034 cm (0.0135 in.) static pressure orifices displaced 6.12 cm from the plane of the wing.

Two models with nominal chords of 5.08 cm (2 in.) and 7.62 cm (3 in.) were tested. Each had the same shape with a thickness ratio of 12%. After the test the airfoil ordinates were measured and best fit to a parabolic arc. Based on these best fit curves, a parabolic arc fit the measured contour to 0.001 cm (0.0004 in.).

Since the chord-based Reynolds numbers are relatively small, boundary-layer transition was artificially induced with a "trip" in order to avoid any problems due to an unknown or erratic transition location. Sieved glass beads in the 0.0089 to 0.0104 cm (0.0035 to 0.0041 in.) size range located 0.813 cm (0.32 in.) back from the leading edge in a band 0.102 cm (0.04 in.) wide were used to form the boundary-layer trip.

MEASURING TECHNIQUE

The primary data for this test were obtained by measuring static pressure distributions on the end plates, on the wing models, and on an axial survey tube. Tunnel stagnation conditions were monitored with a pitot probe and thermocouple in the settling chamber.

The tunnel and pressure data were analyzed with a data acquisition and reduction system consisting of a small (9K memory) digital computer and associated peripheral equipment. The data

[1] Unpublished data obtained in the Ames 2- by 2-Foot Transonic Wind Tunnel, April 1966.

acquisition software was designed so that pressure calibrations and tunnel conditions were checked before and after each data sequence to ensure that consistent, accurate data were obtained.

FLOW IN THE CHANNEL

It was apparent that the Mach number in the channel between the two splitter plates could not be calculated by measuring the static pressure in the main tunnel. The flow in the channel was affected by many parameters. Among the most relevant were the blockage and the open area ratio (*OAR*) of the wind tunnel. Since it was not feasible to calculate the channel Mach number analytically, the Mach number in the channel was determined from the static pressures measured on the splitter plate and from the tunnel stagnation pressure.

The Mach number distributions in the channel as affected by blockage and *OAR* are shown in figures 2 and 3. Distributions are shown for four values of the tunnel set Mach number. This Mach number represents the speed of the main flow in the region upstream of the splitter plates. In the absence of a wing model, the channel Mach number, M_{chan}, is always less than the set Mach number. The deviation increases with M_{set} and is sensitive to model blockage and *OAR*. Figure 2 shows this behavior as it is affected by the two different size models. The set Mach number is indicated by tic marks on the Mach number axis. At $M_{set} = 0.60$, the channel Mach number is somewhat less from M_{set}, and the effect of the larger wing is small. At $M_{set} = 0.95$, M_{chan} is quite a bit less than M_{set}, and the sensitivity to wing size is greater. Figure 3 shows the same trends, but the *OAR* of the entire wind tunnel is varied rather than the size of the test model. It was found that for Mach numbers near unity M_{chan} was very insensitive to changes in M_{set}. Thus, depending on the model and tunnel *OAR*, it may be difficult to realize certain predetermined Mach numbers near $M_\infty = 1$.

Even though the Mach numbers are not the same as the tunnel set Mach numbers, there are no apparent gradients or other pathological characteristics in the channel Mach number distribution. Thus, M_{chan} in the region upstream of the wing model is a good approximation to the approach Mach number M_∞. In the data analysis, M_∞ was calculated by taking the average of three values of M_{chan} near the 6.0 cm station. In most cases the distribution was flat to within 0.002 Mach number. It should be noted that there is no way to "calibrate" the channel in a tunnel empty configuration. The approach Mach number will change with each change in model or tunnel configuration.

AIRFOIL PRESSURE DISTRIBUTIONS

The major purpose of this test was to measure the pressure distribution on a 12% biconvex airfoil and to compare the results with Riddle's experiment and with theory. It was hoped that the data obtained with splitter plates would be at least as valid as the previous full span data using the same airfoil. As will be shown presently, the use of splitter plates actually yields a better quality two-dimensional flow than the full span test.

One of the first problems to be encountered was that the set Mach numbers, which were chosen to correspond to Riddle's 1966 test, were not the actual approach Mach numbers (see fig. 2, for example). Thus, the two tests could not be compared on a one-to-one basis. A useful basis for comparison, however, is the degree to which the measured pressures can be predicted from the small disturbance transonic equation. Such a comparison among the present results and the small disturbance equation is shown in figures 4 through 7.

Each of the figures shows comparisons between experiment and theory for five Mach numbers on an offset scale. The theoretical curves were obtained from the computer program *TSFOIL* (ref. 4). *TSFOIL* solves the small-disturbance transonic equation by utilizing a mixed finite difference scheme which evaluates field variables at selected nodal points by a line relaxation technique. The results presented in this paper correspond to free-air solutions on a 77X56 mesh (mesh limits: 1 chord upstream, 1.9 chords downstream, 5.2 chords above the airfoil, and 48 mesh points on the airfoil slit). Each computation took approximately 16 sec on a CDC 7600 computer.

Figure 4 shows a comparison for the case of the 7.6 cm (3 in.) chord wing in a well ventilated wind tunnel (*OAR* = 18.5%). Comparison between theory and experiment is quite good over the indicated Mach number range. The results can be described by considering three distinct regions, the subcritical, supercritical, and high supercritical. In the subcritical range, the peak pressure is predicted quite well, but there is a slight systematic overexpansion in the theoretical curves fore and aft of the midchord point. If this overexpansion were not so symmetrical with respect to the experimental data, it would be easy to attribute it to viscous effects, transition strip effects, etc. Although these effects are certainly present to some degree, the primary cause is that thin airfoil theory, upon which *TSFOIL* is based, is starting to exceed its domain of validity. In the forward region of the airfoil this overexpansion persists at all test Mach numbers.

A more serious defect in the theory is evident in the supercritical comparisons. As is well known, inviscid transonic theory fails miserably in regions where strong viscous effects occur. The strong shock-wave boundary-layer interaction causes the shock wave to move well upstream compared to the inviscid (unseparated) calculation. Many investigations are now underway concerning this effect. Approaches ranging from approximating boundary-layer effects in inviscid calculations, to local patching of viscous solutions, to attempts at the full Navier-Stokes equations are under active investigation. Upstream of the region where viscous effects dominate, the theory is actually quite good aside from the slight overexpansion.

In the strong supercritical region, the experimental technique used here excells. Aside from the slight overexpansion in the forward part of the airfoil (which was attributed to defects in thin airfoil theory), the agreement between theory and experiment is excellent. The comparisons show vividly that the two-dimensional flow has not been compromised at Mach numbers near one.

Figure 5 shows similar results for the 7.6 cm (3 in.) wing, but for a tunnel whose *OAR* = 6.3%. Figures 6 and 7 show similar data for the 5.1 cm (2 in.) wing. Overall, the agreement is satisfactory in the subcritical range, good in the supercritical range upstream of the shock wave, and excellent in the highly supercritical range. In each of these four figures (except for $M_\infty = 0.903$ in fig. 5), pressure distributions are presented for the same five tunnel set Mach numbers of 0.6, 0.75, 0.85, 0.95, and 1.0.

Some explanations can be given for the good agreement between theory and experiment. As was pointed out by Spreiter (ref. 5) and Busemann (ref. 6) the transonic small-disturbance equations are obtained from the full potential equations by retaining only one of the five second-order terms. It was shown by Busemann that this choice is only justified at sonic speed. Away from $M_\infty = 1$, there is no justification for retaining only one of the second-order terms. This freedom of choice in the second-order terms when M_∞ is not very close to one was used to advantage by some investigators who chose the parameters to force better agreement between theory and experiment for certain configurations. In the present calculations, the equations advocated by Busemann and Spreiter were used. Thus, although the solution may not be a uniformly valid approximation to the same order over the entire Mach number range, it is certainly a valid approximation to $O(\delta^2)$ as $M_\infty \to 1$. Another reason for good agreement near $M_\infty = 1.0$ can be traced to the "Mach number freeze" phenomenon. Assuming a good two-dimensional flow exists, the sensitivity of the pressure distributions to varying M_∞ is very small at Mach numbers near unity. Hence, a small error in measuring M_∞ will not affect the pressure coefficients to a great extent. These two effects, the ambiguity in the small-disturbance equations and the "Mach number freeze" could explain the degree of success which was obtained in comparing experiment and theory.

Limited calculations were also attempted with more accurate theories. A computer code developed by Carlson (ref. 7) for solving the exact potential equation for two-dimensional compressible flow was used. Carlson's algorithm is based on a mixed finite difference scheme on a stretched Cartesian grid. Calculations were made on a 48X24 grid and took about 19 sec on a CDC 7600 computer. Figure 8 shows calculated pressure distributions for the 7.6 cm wing compared with experiment and transonic small disturbance theory. There is not much difference between the two theories, but the more exact theory shows even better agreement with experiment. The exact potential equations are still an inviscid approximation to the Navier-Stokes equations so the shock-wave boundary-layer interaction is not predicted correctly.

A third computer code, an Euler equation solver developed by MacCormack (private communication from R. W. MacCormack, NASA Ames Research Center, Moffett Field, CA) was used to compute some supercritical pressure distributions. This program solves the Euler equations by means of an explicit finite difference scheme on a 32X24 mesh. The program converged to a stable solution in approximately 32 sec on the CDC 7600. The results are shown in figure 9 compared with transonic small disturbance theory and experiment. A comparison of figures 8 and 9 shows that the Euler equation and the exact potential equation are about equal in their predictions. The Euler equations are expected to be more accurate near strong shock waves, but this increased accuracy is compromised by the strong viscous effects in the experimental data.

These comparisons between theory and experiment show that transonic small disturbance theory is actually quite good in predicting inviscid pressure distributions on nonlifting configurations. In the absence of viscous effects, the exact potential equation and the Euler equation predict similar pressures, both being slightly better than the small disturbance theory. The validity of the splitter plate concept for 2-D testing is demonstrated by the convergence of the more exact theories to the measured experimental data. In the remainder of this report, only transonic small disturbance theory will be used as the theoretical basis of comparison.

Extensive calculations to assess the effect of wall interference were not attempted in this investigation. The wall geometry of the 2- by 2-foot transonic wind tunnel is a complicated amalgam of slats and corrugated slots which is not easily transferred into a homogenous wall boundary condition. Some calculations were made, however, for the configuration and Mach numbers presented in figure 5. Two-dimensional computations were made for closed-throat (solid wall) and open-throat (free jet) boundary condition. For the ratio of chord to height used in this test, no significant differences between these two cases and the free-air calculations were observed (excepting the case $M_\infty = 0.903$, closed throat where the flow choked).

A comparison between theory and experiment also was made for Riddle's data. The experimental setup is shown in figure 10. The 7.6 cm (3 in.) wing was mounted horizontally in the 2- by 2-foot wind tunnel and all four walls were ventilated ($OAR = 18.5\%$). The experimental data is shown in figure 11. The overexpansion of the computed curves persists in this comparison. The supercritical case ($M_\infty = 0.804$) predicts the shock-wave location quite well because the shock-wave boundary-layer interaction is weak, as indicated by the flow recompression aft of the shock. The highly supercritical flow is consistently under-expanded with respect to the theory. As mentioned before, the small-disturbance theory is expected to be quite good in this region. The lack of agreement is attributed mainly to three-dimensional contamination of the flow. A primary reason is that all four walls of the tunnel were ventilated in this test. Another cause was probably due to the thick boundary layer on the side wall and its interaction with the shock wave. The configuration with splitter plates seems to give better results in the highly supercritical region.

COMPARISON WITH OTHER INVESTIGATIONS

Comparisons of the present experiment, Riddle's experiment, and various computer codes were shown in the previous section. In this section, a comparison between various experimental data and another method for solving the transonic equation will be discussed. In the 1950's, many approximate methods were developed for the transonic small-disturbance equations. One of the more successful was a procedure developed by Spreiter and his co-workers (refs. 8 and 9) which is based on iterative solutions of an integral equation which is equivalent to the transonic small-disturbance equations. Spreiter's solution for the pressure coefficient at the midchord station of a biconvex nonlifting airfoil is compared to the present experiments in figure 12. Results are plotted in transonic similarity variables to suppress the effect of thickness ratio. The comparisons shown in figure 12 include all of the relevant data from figures 4–7 and agreement is quite good, especially in the highly supercritical region. Figure 13 shows the same information as figure 12 but includes the results of four other experimental investigations with biconvex airfoils. As mentioned above, Riddle's data is quite good in the subcritical range, but drops below the theory in the supercritical range. Collins and Krupp (ref. 10) performed their tests in a closed wind tunnel and experienced severe wall interference which culminated in choking at a value of the similarity parameter of −0.64. Before choking, their results agreed well with the theory. Knechtel's data (ref. 11), which is perhaps the current standard for circular arc airfoils, shows a consistent overexpansion in the high subsonic Mach number regime. This behavior was mentioned by Knechtel in his technical note. Spreiter (ref. 10) presents calculations which show that deviations in this direction correspond to a closed throat type of interference, but the deviation may also be due to three-dimensional effects. The early data of Wood and Gooderum (ref. 12) is also presented in figure 13. This data is subject to some uncertainty for two reasons. The data was obtained with an interferometer which is less precise than pressure measurements and the model was mounted on the wind-tunnel floor. Even with these qualifications, the data does seem to follow the theoretical trend, but is displaced upwards. The overall impression of all this data is that the present experiment with splitter plates shows the best agreement with theory. It is believed that a major reason for this good agreement is the absence of three-dimensional contamination, but the effect of wall interference in the other data is an unknown quantity. It seems that a carefully controlled experiment is necessary to assess the relative effect of wall interference and three-dimensionality.

CONCLUSIONS

An experiment on nonlifting, circular arc airfoils at transonic speeds has demonstrated that splitter plates can be used to generate a high quality flow at Mach numbers near unity. The experimental data has been compared to other experiments and to three separate theories. Among the experimental data analyzed, the present experiments yield the best agreement with theory. The technique needs to be extended to supersonic flows and to lifting configurations before its full potential can be assessed.

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Moffett Field, California 94035, November 16, 1977

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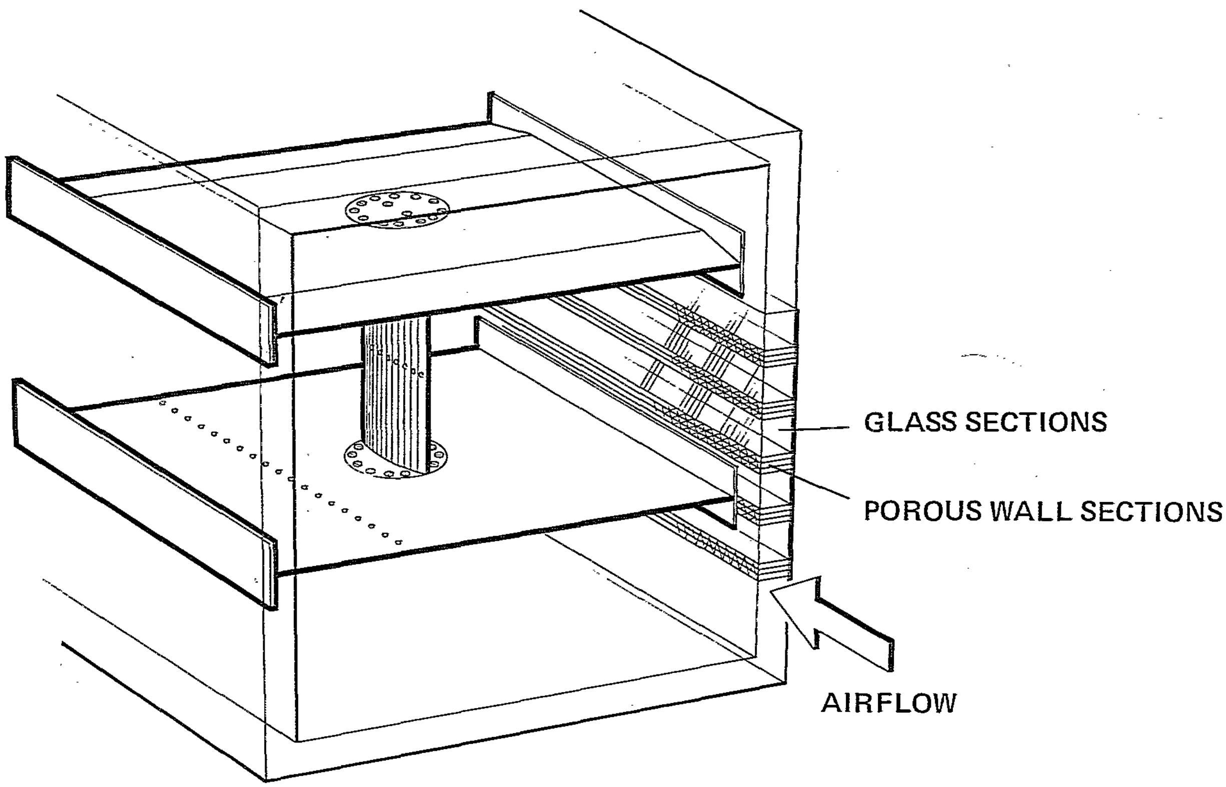

Figure 1.— Schematic of splitter plates and wing model installed in Ames 2- by 2-Foot Transonic Wind Tunnel.

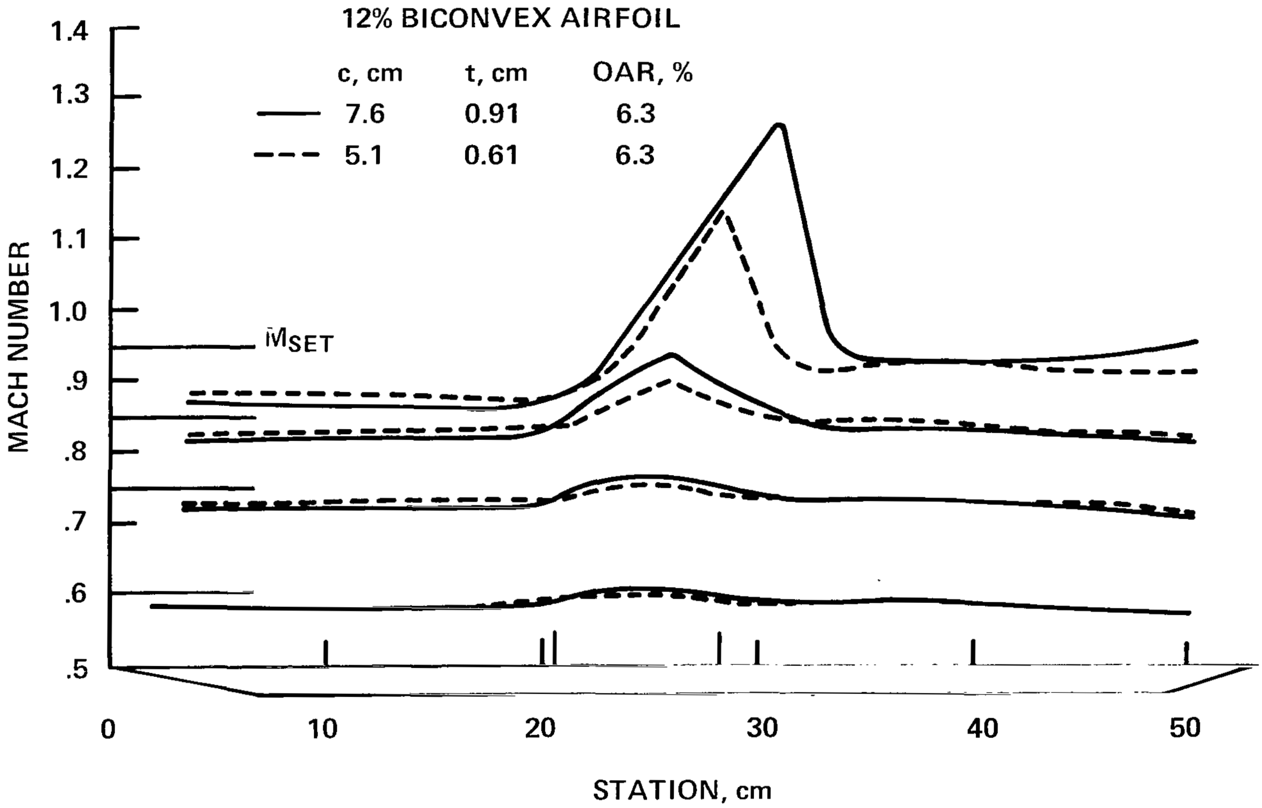

Figure 2.– Blockage effect on channel Mach number distribution.

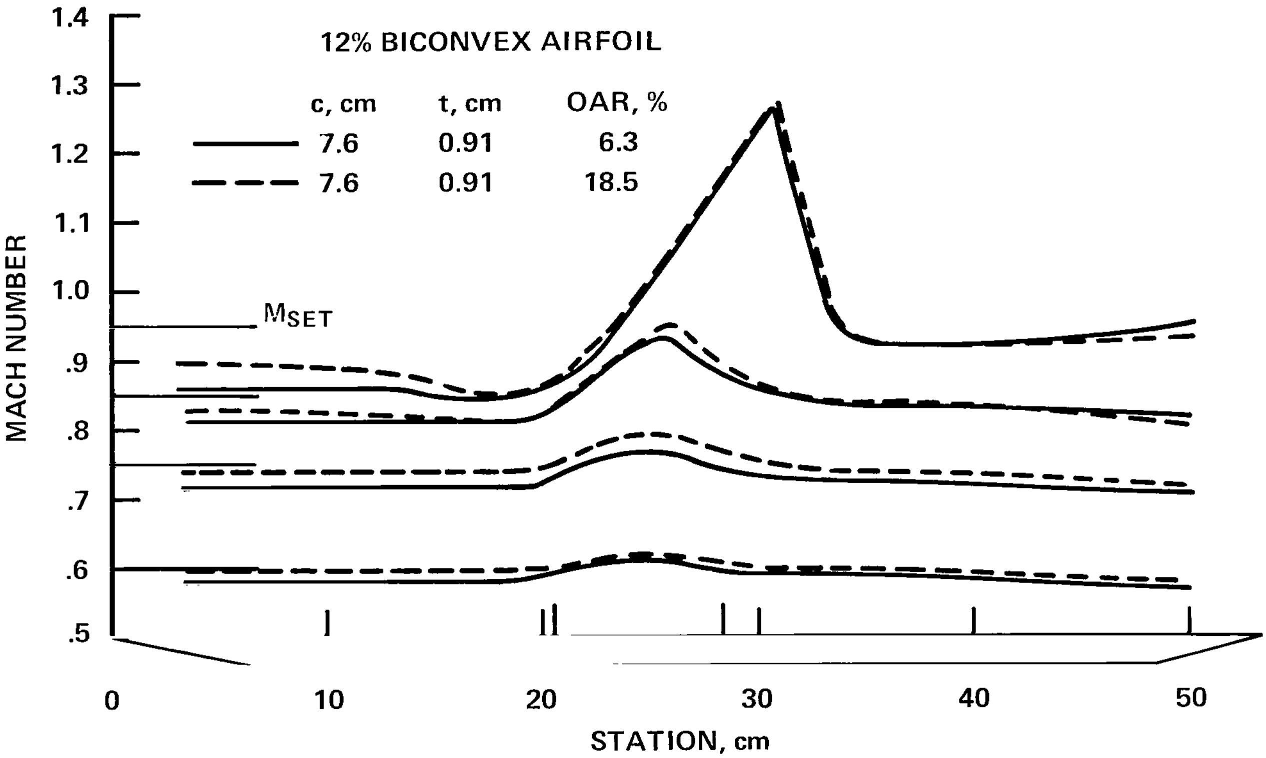

Figure 3.– Effect of wind-tunnel ventilation on channel Mach number distribution.

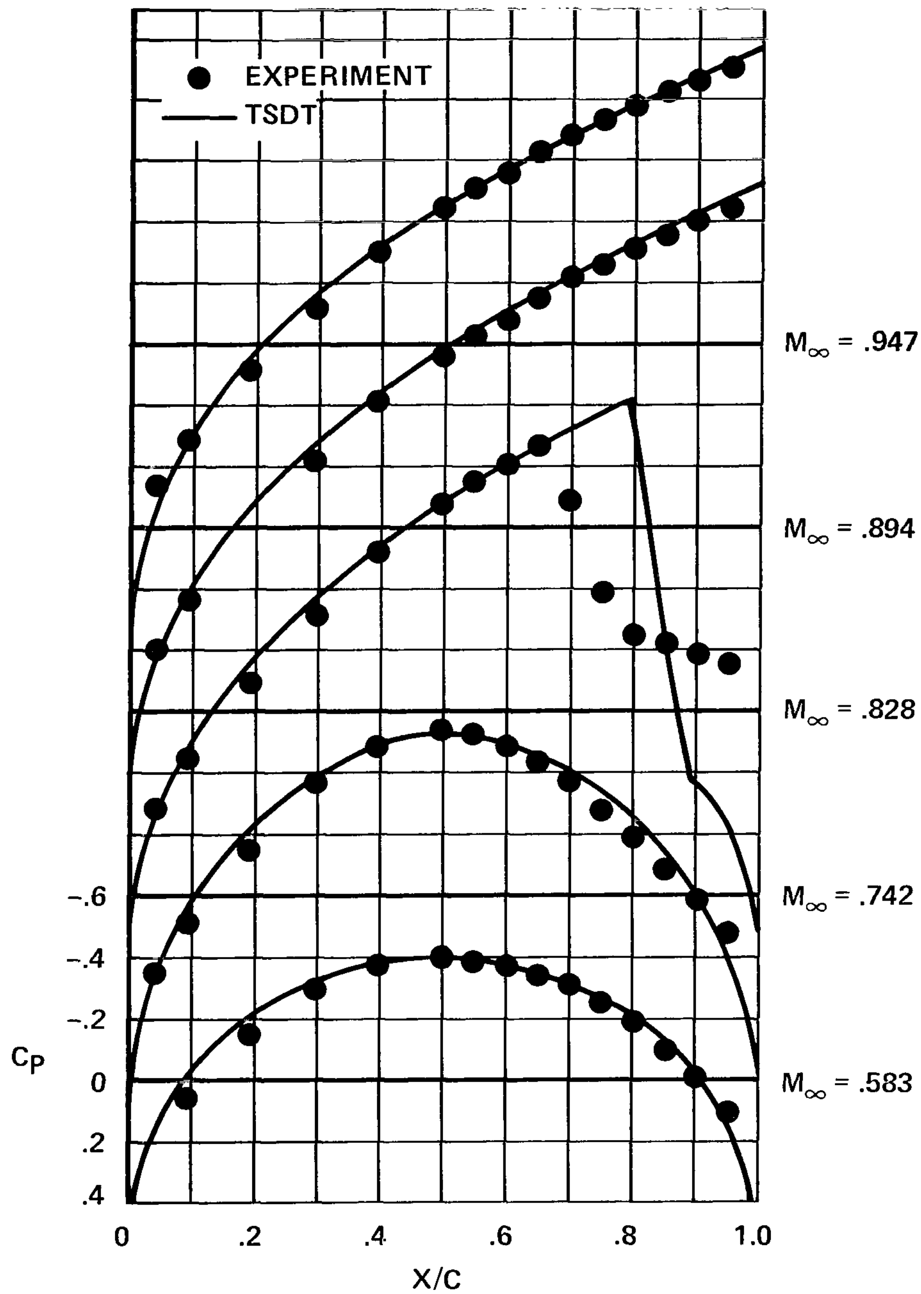

Figure 4.– Pressure distributions on 7.6 cm (3 in.), 12% biconvex airfoil. OAR = 18.5%.

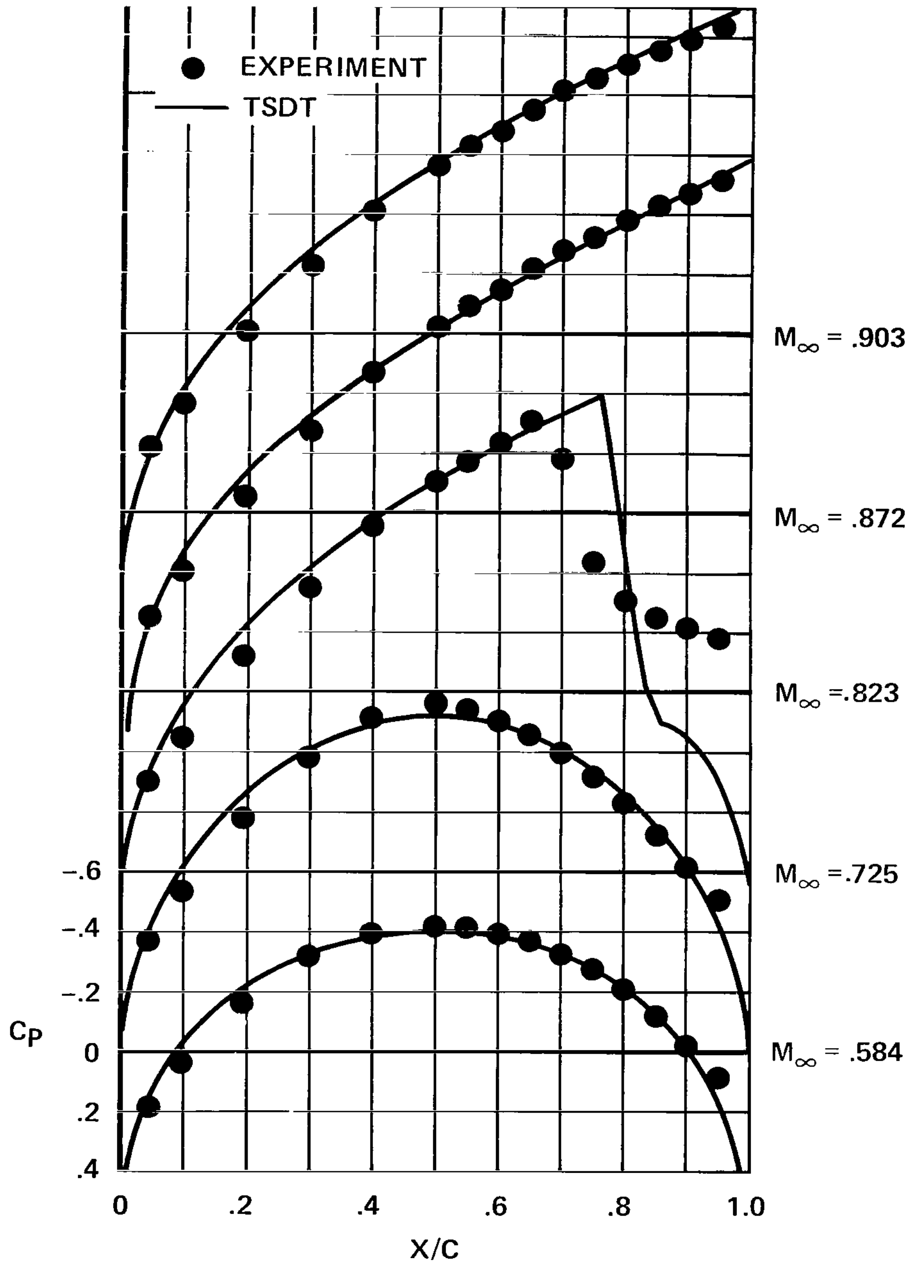

Figure 5.– Pressure distributions on 7.6 cm (3 in.), 12% biconvex airfoil. *OAR* = 6.3%.

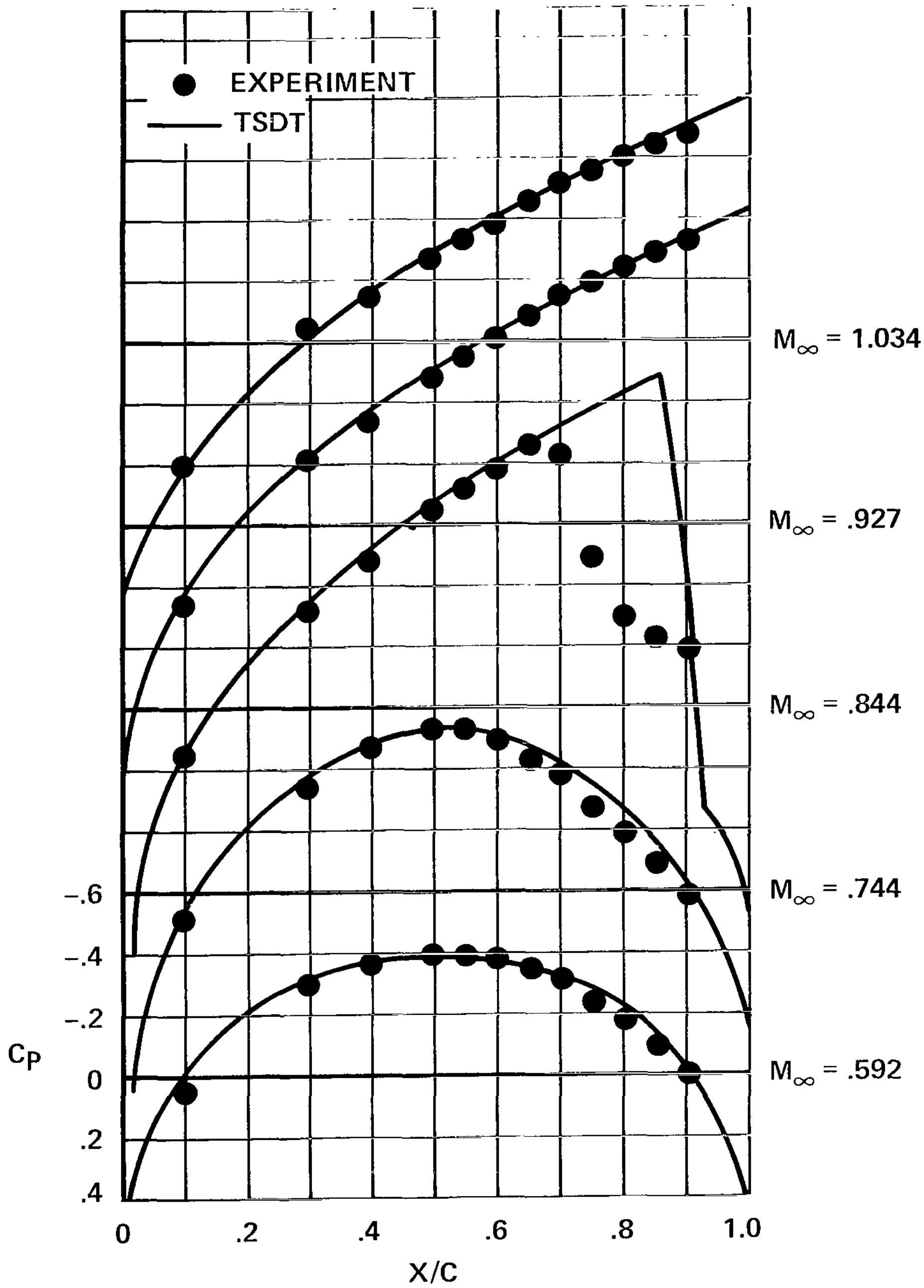

Figure 6.– Pressure distributions on 5.1 cm (2 in.), 12% biconvex airfoil. *OAR* = 18.5%.

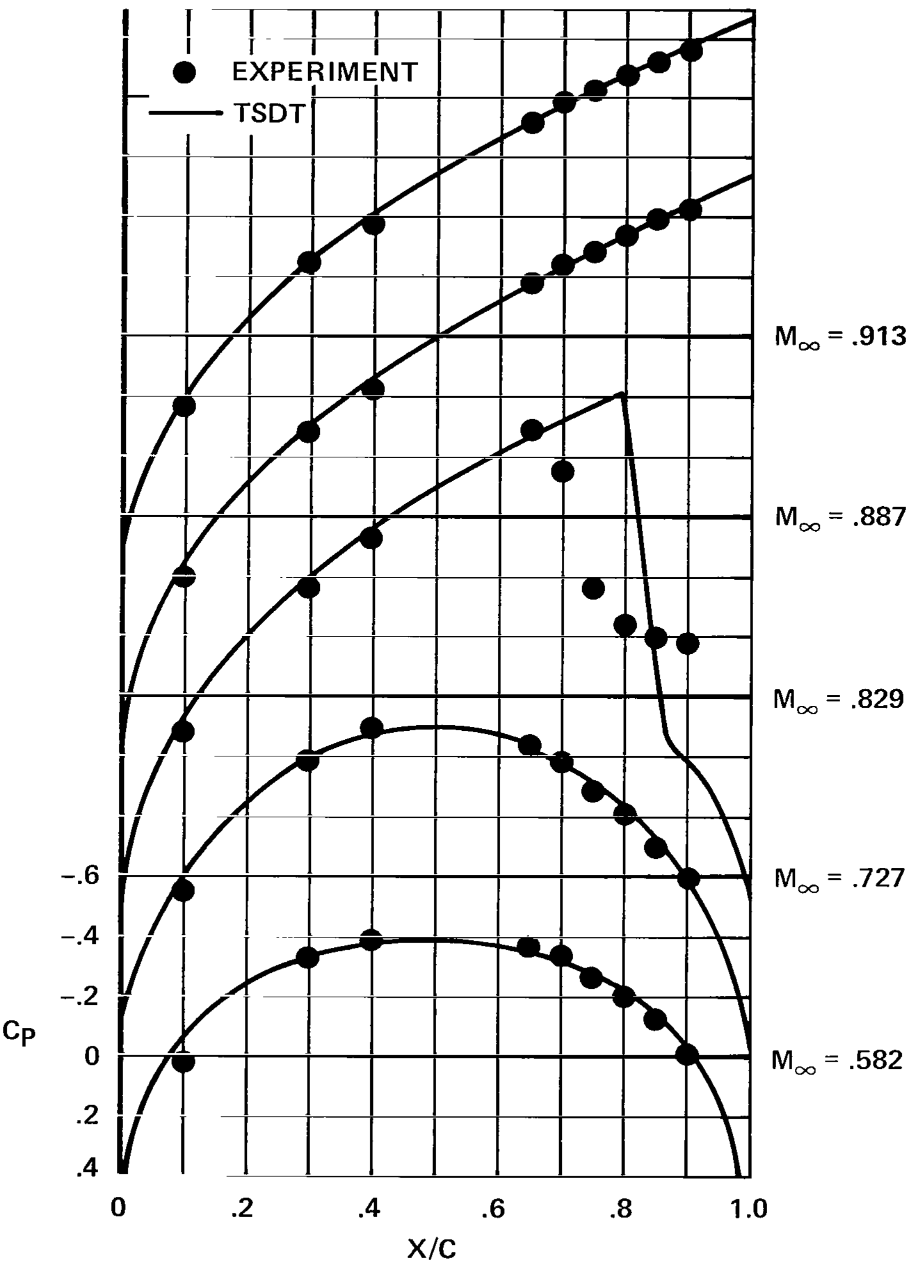

Figure 7.– Pressure distributions on 5.1 cm (2 in.), 12% biconvex airfoil. OAR = 6.3%.

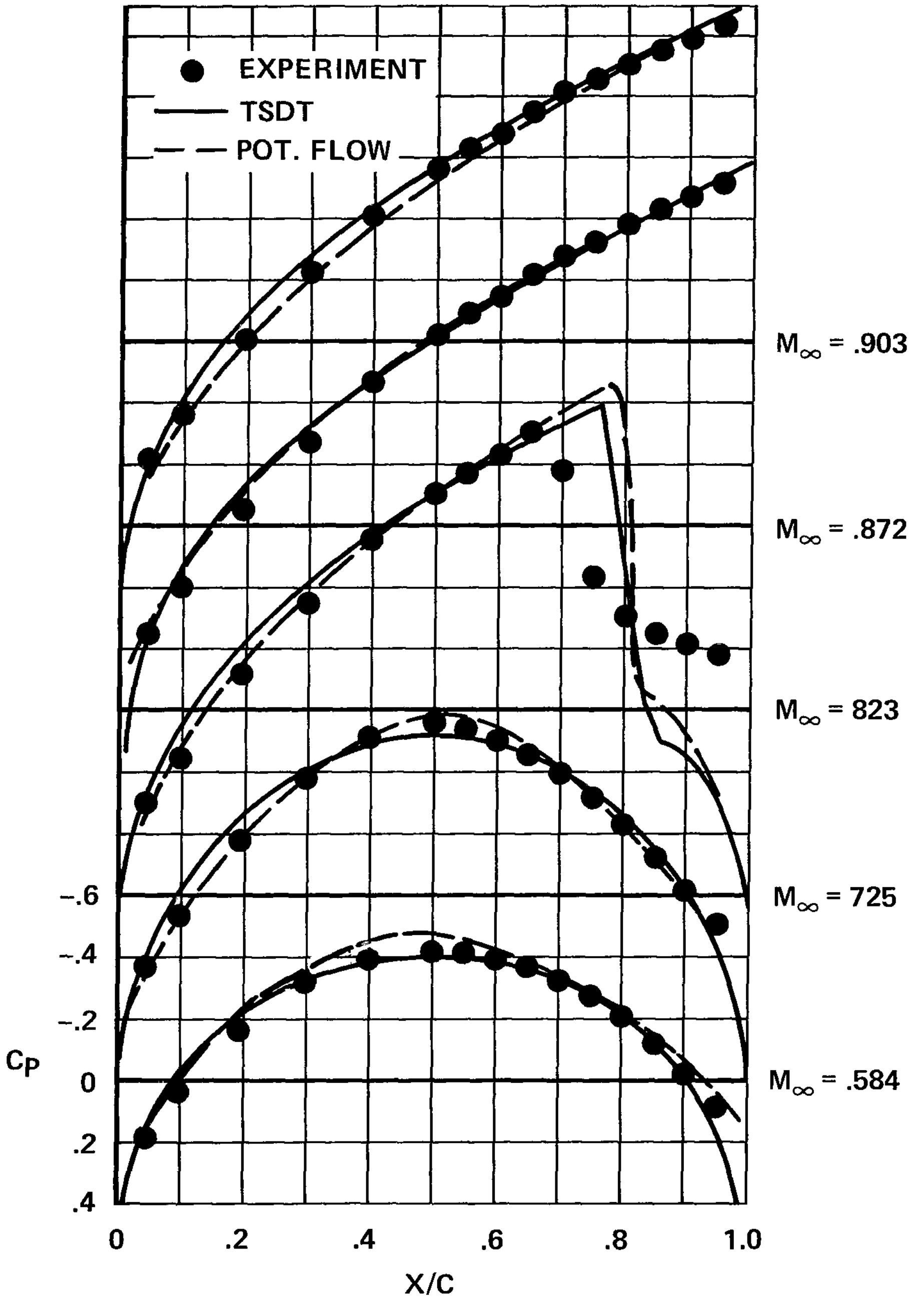

Figure 8.– Pressure distribution on 7.6 cm (3 in.), 12% biconvex airfoil compared to full exact potential theory. OAR = 6.3%.

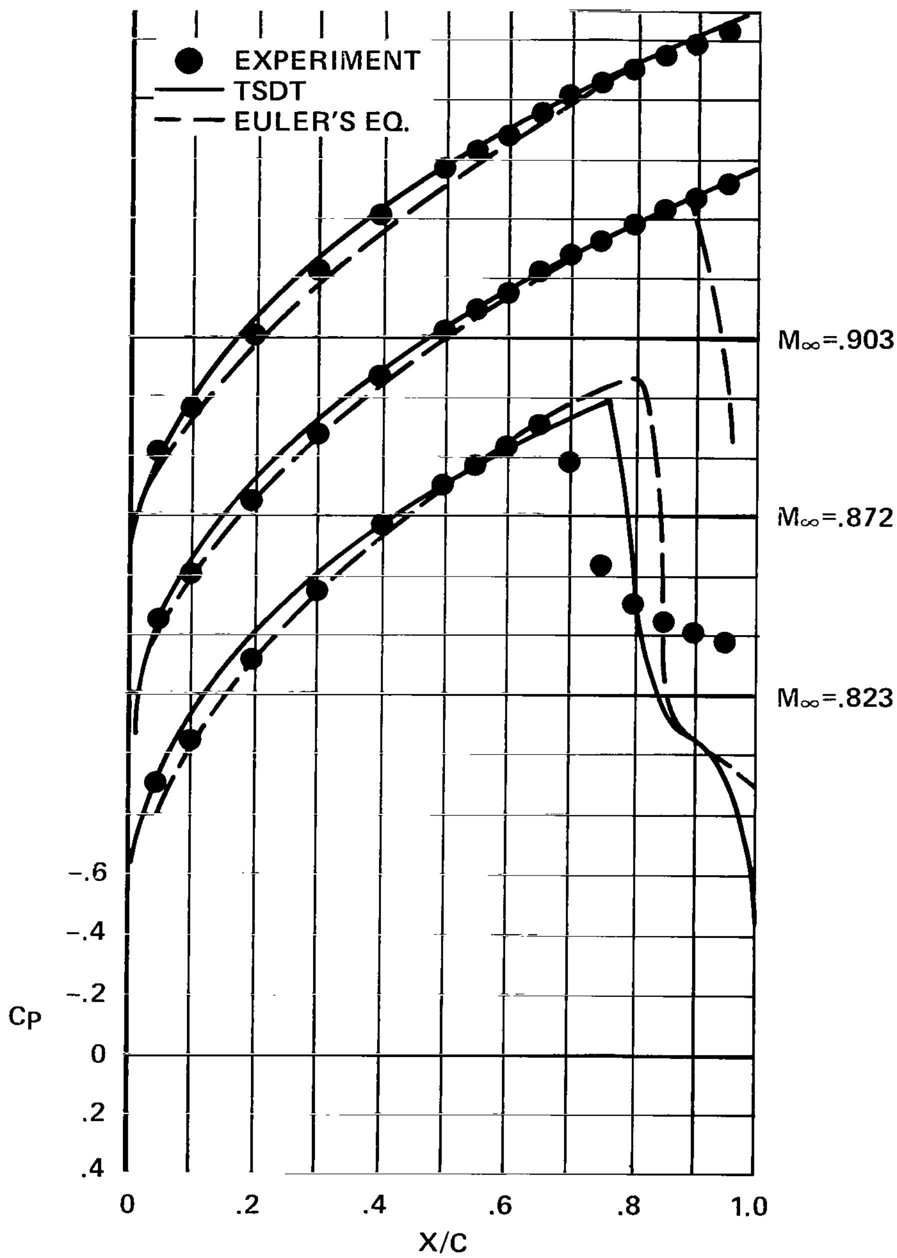

Figure 9.— Pressure distribution on 7.6 cm (3 in.), 12% biconvex airfoil compared to solutions of Euler's equation. *OAR* = 6.3%.

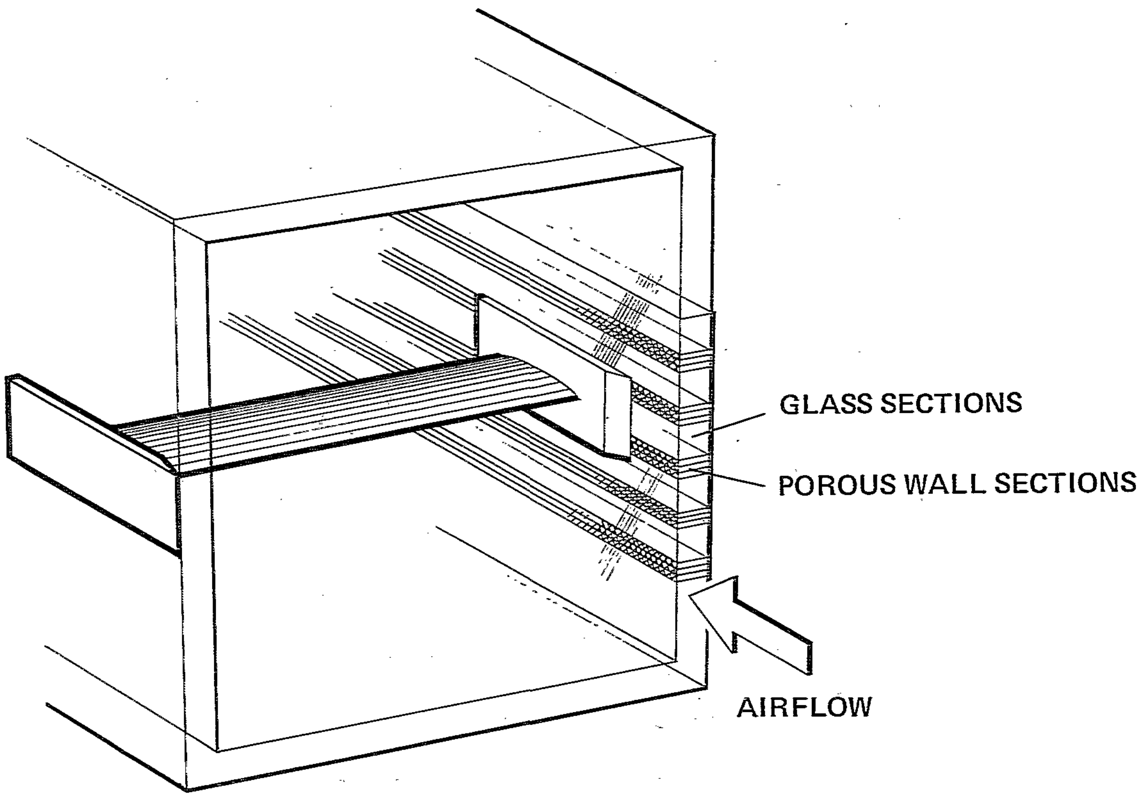

Figure 10.— Schematic of full-span wing model installed in Ames 2- by 2-Foot Transonic Wind Tunnel.

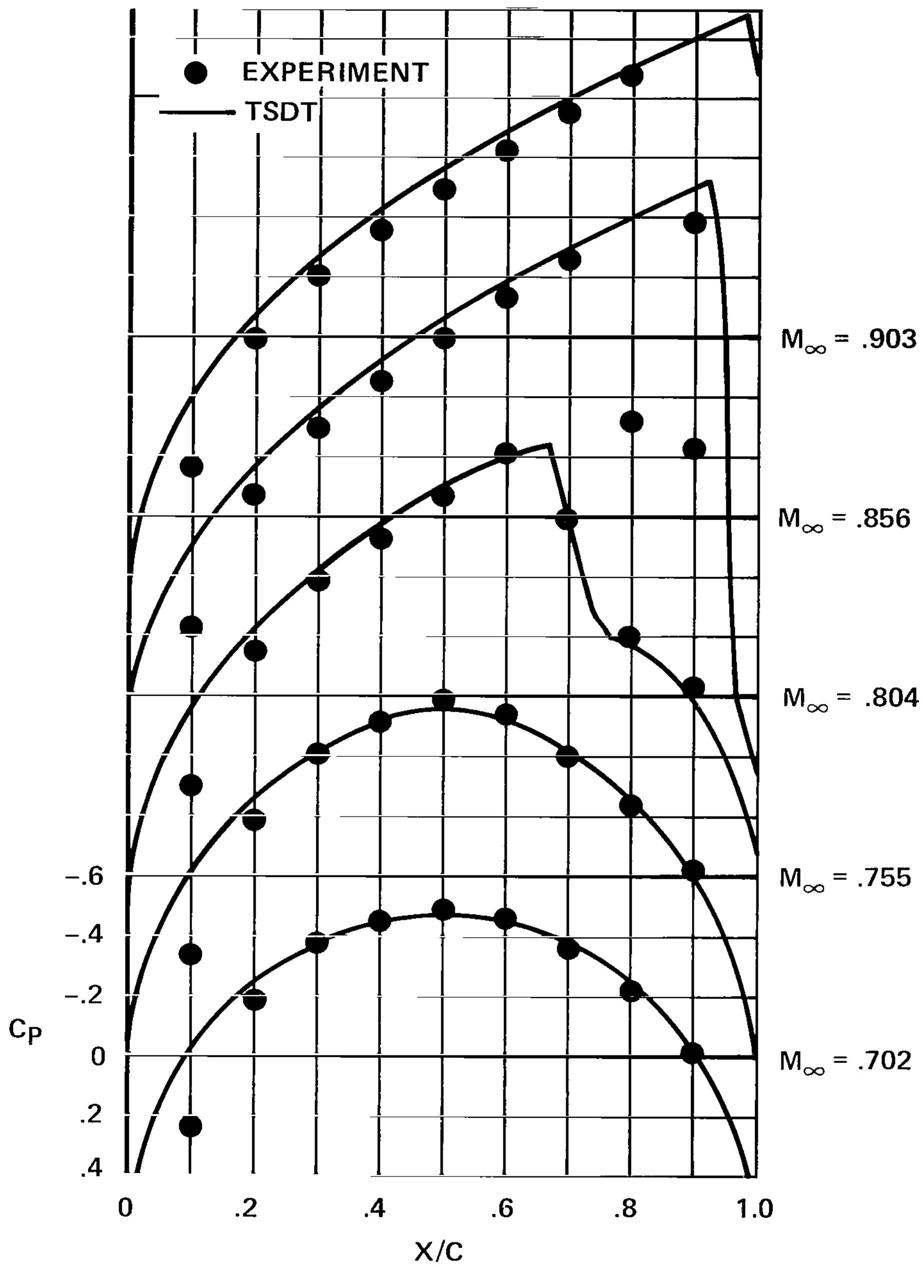

Figure 11.– Pressure distributions on 7.6 cm (3 in.), 12% biconvex airfoil. Full-span model. *OAR* = 18.5%.

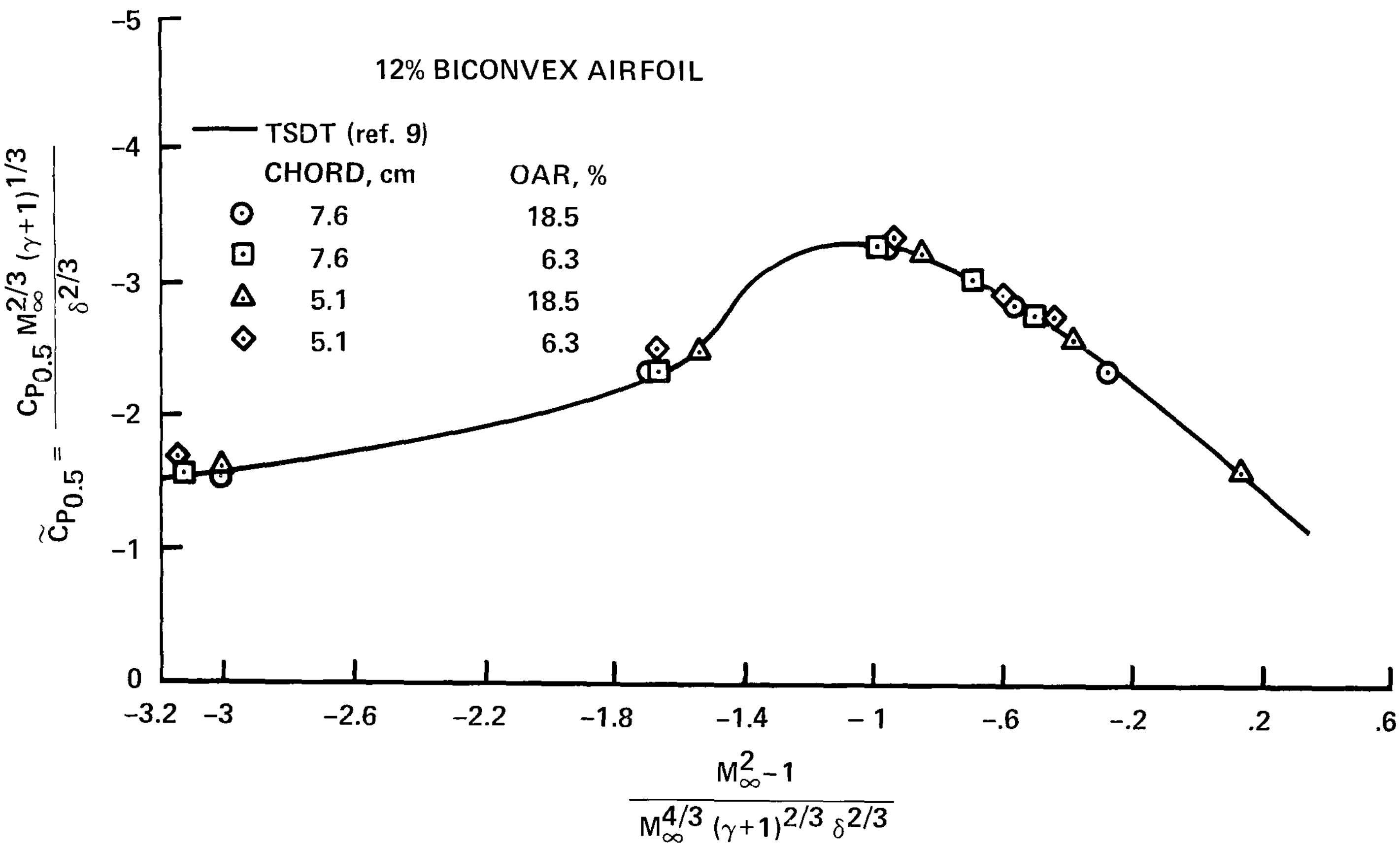

Figure 12.– Variation of midchord pressure coefficient with transonic similarity parameter.

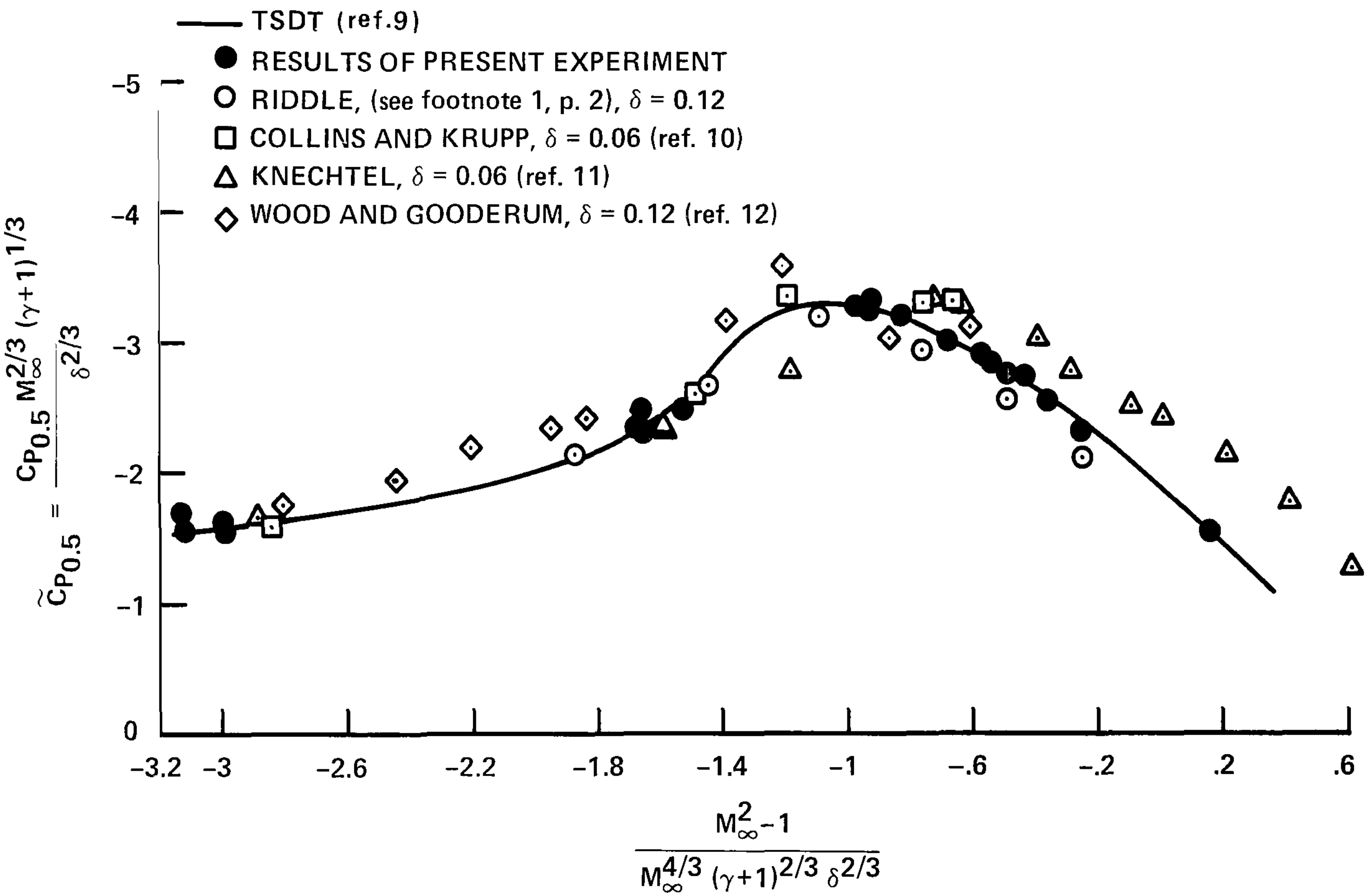

Figure 13.– Variation of midchord pressure coefficient with transonic similarity parameter for various experiments.

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15. Supplementary Notes

16. Abstract

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