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Comparison of NAE Porous Wall and NASA Adaptive Wall Test Results Using the NAE CAST-10 Airfoil Model

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Wind tunnels can now simulate flows over airfoils at high Reynolds numbers and high subsonic speeds. Methods to correct for (or reduce) test section wall interference at these test conditions must be validated. The National Aeronautical Establishment (NAE) of Canada and NASA have a cooperative agreement to study this area. NAE designed, built, and tested a CAST-10 airfoil model in its conventional Two-Dimensional High Reynolds Number Facility. The results were corrected using classical correction techniques. NASA then tested the same model in its 0.3-meter Transonic Cryogenic Tunnel with the adaptive wall test section. The adaptive wall test section reduced the wall interference to what was expected to be an acceptable level.

This paper will compare the corrected NAE results with the uncorrected NASA results. It will also compared the NAE results with NASA results after residual corrections for top and bottom wall interference. Finally, a comparison of both sets of results corrected for interference from all four walls will be presented.

TASK:

- Study wall interference for 2-D airfoil tests at high Reynold's numbers and high subsonic speeds.

APPROACH:

- Test a supercritical airfoil model in a traditional porous test section and apply classical corrections.

- Test the same airfoil model in an adaptive wall test section.

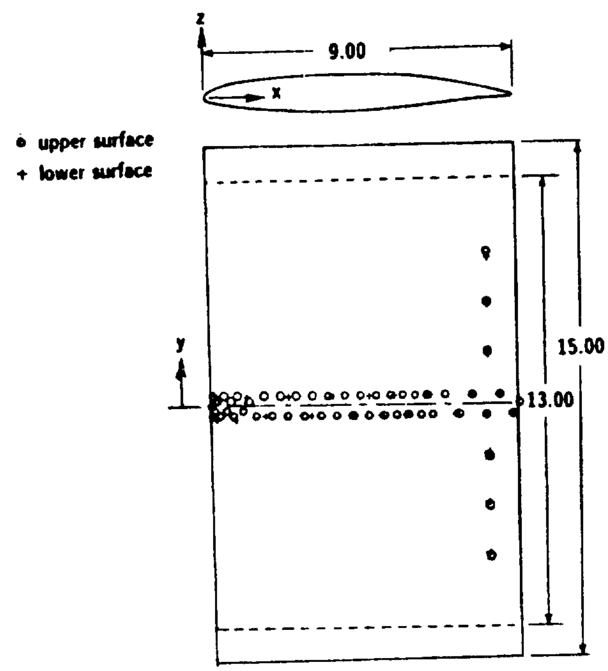
- Compare the results. Correct the results for any residual interference to try to improve the correlation.

The model has a 9.00-inch chord and a 15.00-inch span for testing in the NAE tunnel. The close manufacturing tolerances led to a very accurate representation of the airfoil contour. The largest deviation from the design ordinates was .0001 $\frac{z}{c}$. A chordwise row of orifices was centered at the mid-span with 45 orifices on the upper surface and 23 on the lower surface. Six orifices were arranged in a spanwise row at the 90-percent chord location on each surface. The diameter of the orifices from the leading edge to the 22-percent chord location was 0.010 inches. The diameter of all other orifices was .014 inches. A strip of carborundum grit #320 (average grit size of 0.0011 inches) was used to trip the boundary layer on each surface. The strip started at the 5-percent chord location and was 0.1-inches wide.

The chord line was defined as the line from the leading edge through the center of the trailing edge. This line is 0.88° nose up from the $z = 0.0$ reference line used to define the airfoil. The angle of attack was measured from this chord line.

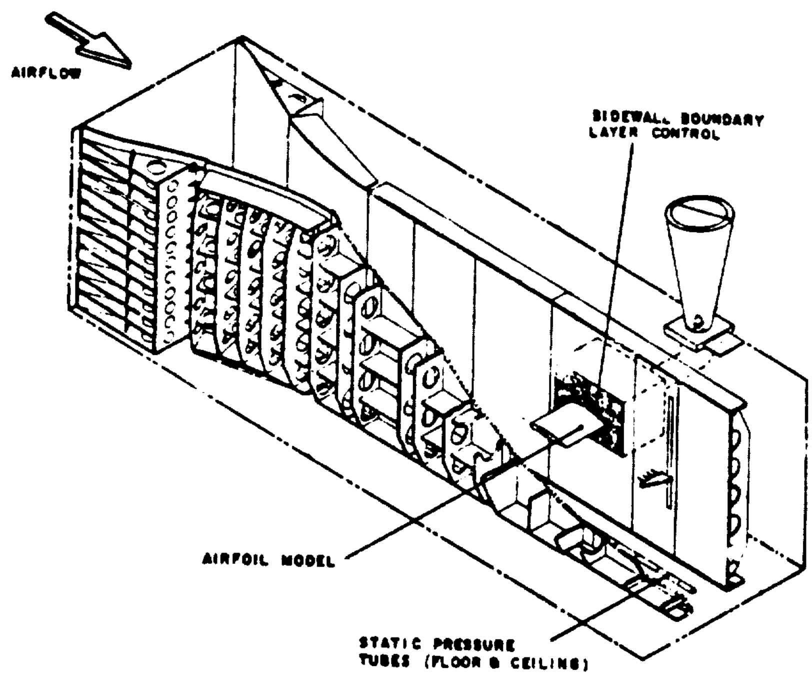
Tests were conducted at Mach numbers from 0.3 to 0.8 at chord Reynolds numbers of 10, 15, and 20×10^6. At each test condition, the angle of attack was varied from near zero lift through stall. The NASA test angles of attack were chosen such that the section normal force coefficients were nearly the same for both tests. This paper will present results for 10×10^6 chord Reynolds numbers only.

CAST-10 Airfoil Model



The model was first tested in the NAE 5 ft x 5 ft Blowdown Wind Tunnel. This tunnel achieves the high Reynolds numbers by testing at elevated stagnation pressures (up to 310 psi). The stagnation temperature is about room temperature. The tunnel has two interchangeable test sections: one for 3-D testing (either full or semi-span models) and the other for 2-D testing (airfoil models). The 2-D testing configuration, referred to as the NAE Two-Dimensional High Reynolds Number Facility, was used for these tests. The empty test section Mach number range is from 0.10 to 0.95. This combination of test conditions yields Reynolds numbers up to 50 x 10^6/ft.

NAE 5 ft x 5 ft Blowdown Wind Tunnel

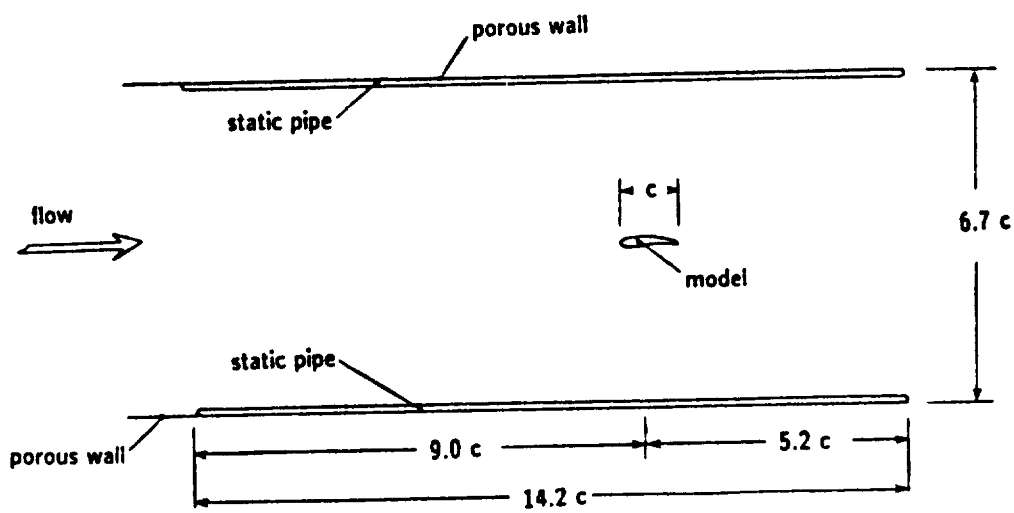


The NAE Two-Dimensional High Reynolds Number Facility test section is 141-inches long, 60-inches high, and 15-inches wide. It has solid, parallel sidewalls and porous top and bottom walls. The top and bottom walls are covered with a 30 mesh screen to reduce the edgetone noise in the test section. A 1-inch-diameter, 128-inch-long static pipe is attached to the top and bottom walls. Each pipe has 40 static pressure orifices. For the 9.00-inch airfoil used in these studies, the test section was 14.2 chords long and 6.7 chords high. The model aspect ratio was 1.7.

The airfoil model was mounted on a porous turntable within an 18-inch by 24-inch porous panel on each sidewall. Moderate suction was applied to the porous region to prevent the sidewall boundary layer from prematurely separating. A four-tube, total pressure rake was mounted 1.8 chords downstream from the trailing edge. The rake was traversed through the model wake to obtain the drag.

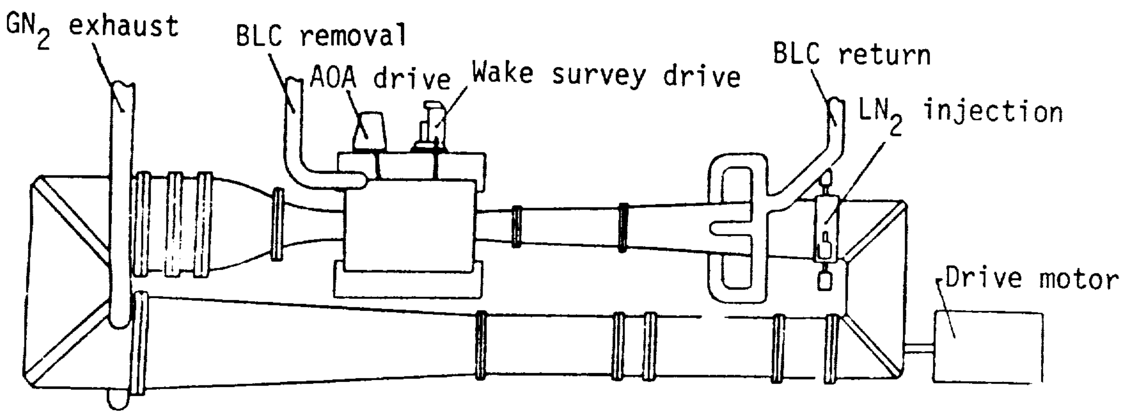
The measured data were corrected for top and bottom wall interference using the method of Mokry and Ohman. The large size of the test section relative to the model insured that the assumptions used to develop the correction technique would not be violated. The corrections to the Mach number and angle of attack were moderate.

NAE Two-Dimensional High Reynolds Number Facility



The NASA 0.3-meter Transonic Cryogenic Tunnel achieves high Reynolds numbers through a combination of elevated stagnation pressures and cryogenic stagnation temperatures. It is a fan-driven, cryogenic pressure tunnel. Nitrogen, rather than air, is used as a test gas. The range of stagnation temperature is from 80 K to 327 K and the range of stagnation pressure is from about 17 psi to 88 psi. The empty test section Mach number range is from about 0.20 to 0.95. This combination of test conditions yields Reynolds numbers up to 100 x 10^6/ft.

NASA 0.3-meter Transonic Cryogenic Tunnel

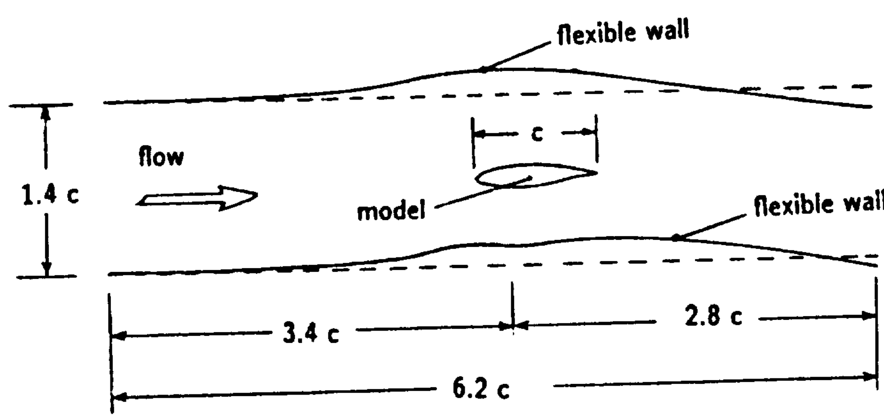


The adaptive wall test section is 13 inches high and 13 inches wide at the entrance of the test section. The solid sidewalls are fixed and parallel. The top and bottom walls are solid and flexible. The forward 55.8-inches of each flexible wall form the test section. The wall shape is controlled by 18 independent jacks. For the 9-inch airfoil used in this test, the test section was 6.2 chords long and 1.4 chords wide. The model aspect ratio was 1.4.

The 15-inch span model was positioned in special turntables so that the chordwise pressure row was aligned with the centerline of the test section. A six-tube, total pressure rake was mounted 1.2 chords downstream of the trailing edge.

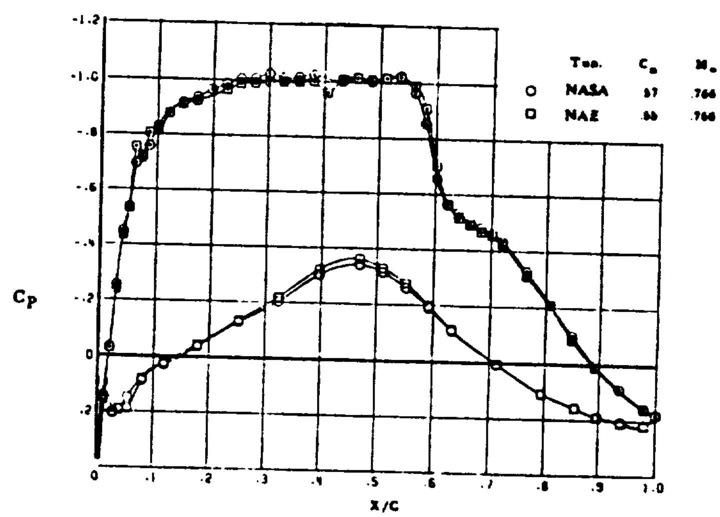
The flexible wall position was determined iteratively using the measured wall shape and static pressures. The algorithm is based on the work of Goodyer and Wolf. The small size and short length of the test section relative to the model should lead to significant wall interference if the walls are not properly positioned.

0.3-m TCT Adaptive Wall Test Section



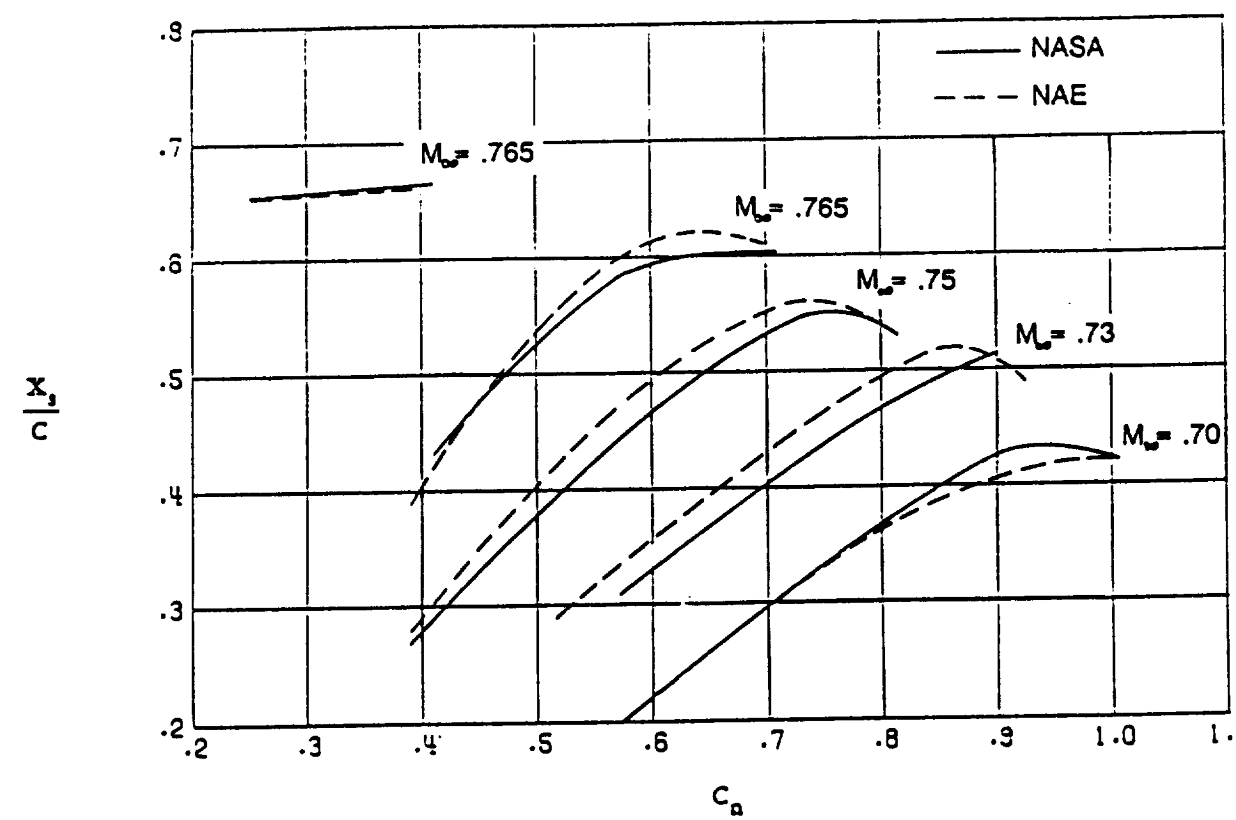
The results from the NASA tests were expected to be practically interference free. Therefore, the uncorrected NASA results are compared to the corrected NAE results. The airfoil chordwise pressure distributions are compared first. In general, the chordwise pressure distributions were in good agreement. The shock locations and trailing edge pressure coefficients agreed well for angles of attack below stall. However, the NASA pressure coefficients were less negative upstream of the beginning of the pressure recovery region. For this case, the peak local Mach number on the lower surface is about 0.009 smaller for the NASA results. Assuming that there is a residual error in the NASA results, the actual Mach number for the NASA tests could be 0.009 smaller than measured.

Comparison of Typical Chordwise Pressure Distributions

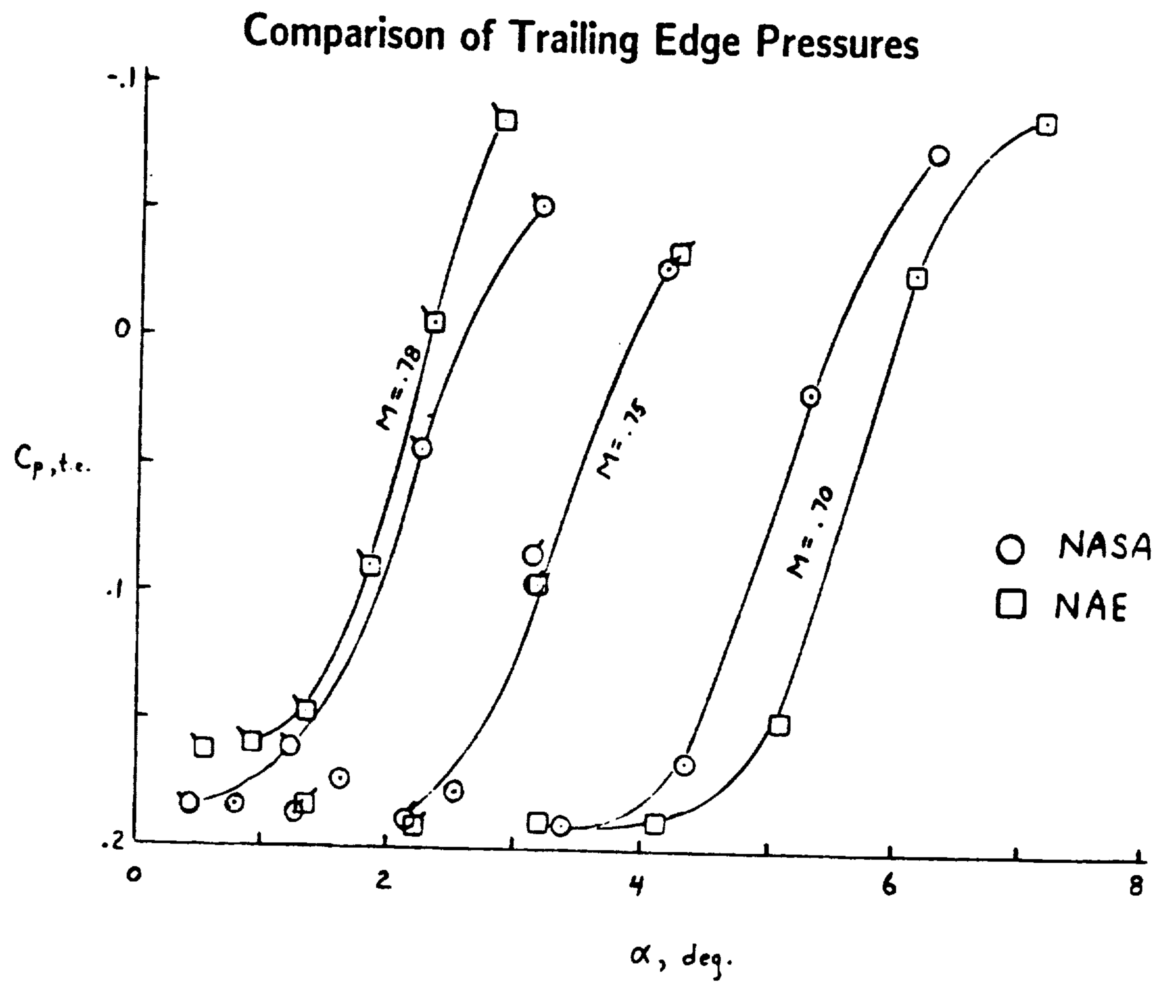


The shock locations were determined by fitting a straight line through the pressures just upstream and just downstream of the pressure rise associated with the shock. A third line was fitted through the pressure rise. The shock location was defined as the midpoint of the two intersections of the fitted lines. Because of the spacing of the orifices, the shock location could be determined with an accuracy of about 2 percent chord. Both sets of data show the same trends. Below the design Mach number, the shock moves aft with increasing normal force until stall. Above the design Mach number, the shock location tends to move slightly forward with increasing normal force. There is a small shift in the curves. When the results are cross-plotted at constant c_n, the maximum shift is equivalent to an error in Mach number of about 0.005. Again assuming there is a residual error in the NASA results, the corrected NASA Mach number could be 0.005 smaller than the measured Mach number.

Comparison of Shock Locations

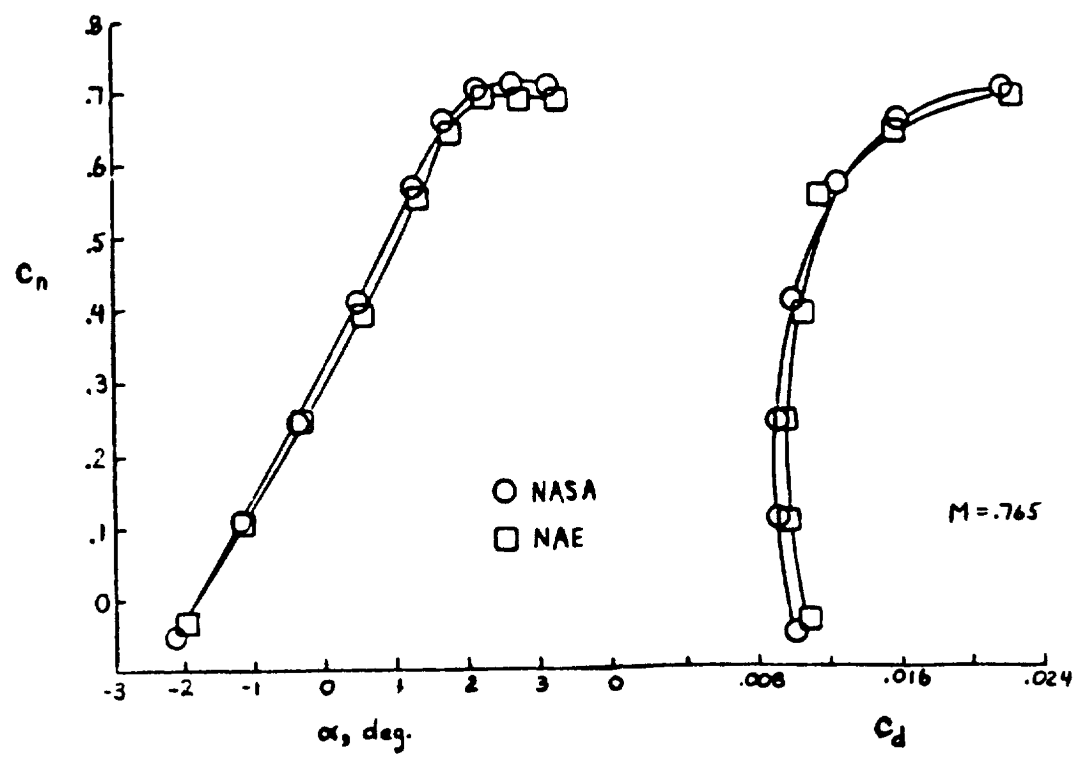


The trailing edge pressure coefficients are sensitive to the condition of the boundary layer. Since the boundary layer was tripped the same way and at the same location in both tests, comparing the trailing edge pressures can be used to check for residual interference. The trailing edge pressures are in reasonable agreement for those angles of attack below stall with $C_P \approx .2$. Near stall, the curves break. The angle of attack for this break does not follow a consistent trend. The reason is not known.



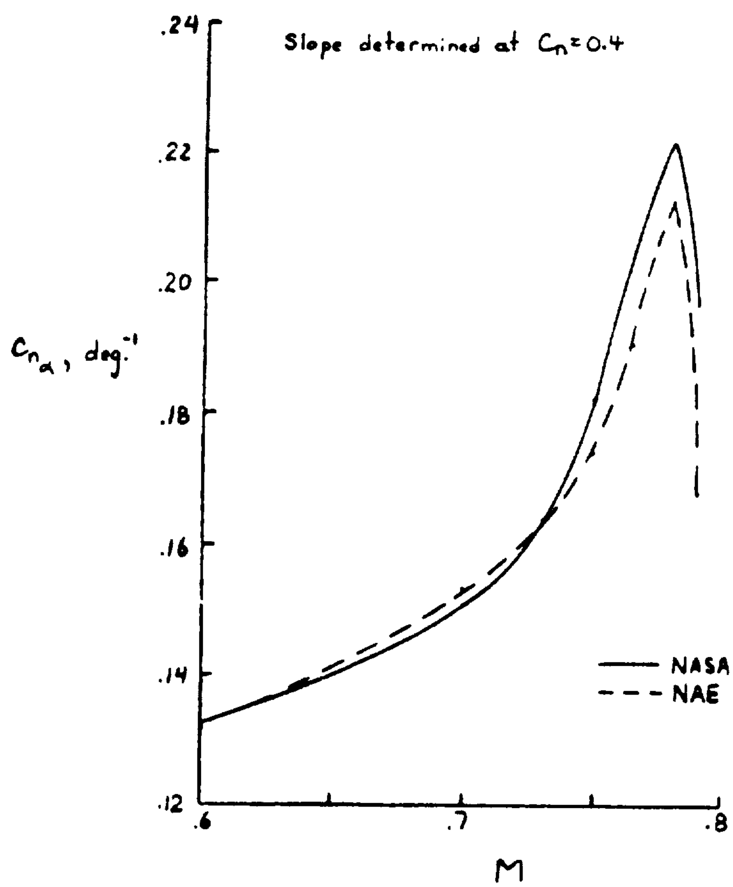
The uncorrected NASA results were expected to have a low level of residual interference from the top and bottom walls. Problems with the adaptive walls prevented data acquisition near zero lift and near stall for some of the test conditions. In general, the angle of attack for the NASA tests was less than the angle of attack for the same normal force for the NAE tests. If the problem was a simple misalignment, the difference would not show up in the normal force — drag polar. This is not the case. The drag for a given normal force is smaller for the NASA tests. The pitching moment data (not shown) was in good agreement. The slopes of the normal force curves and the drag rise characteristics can be examined to help understand the differences.

Comparison of the Integrated Force Coefficients



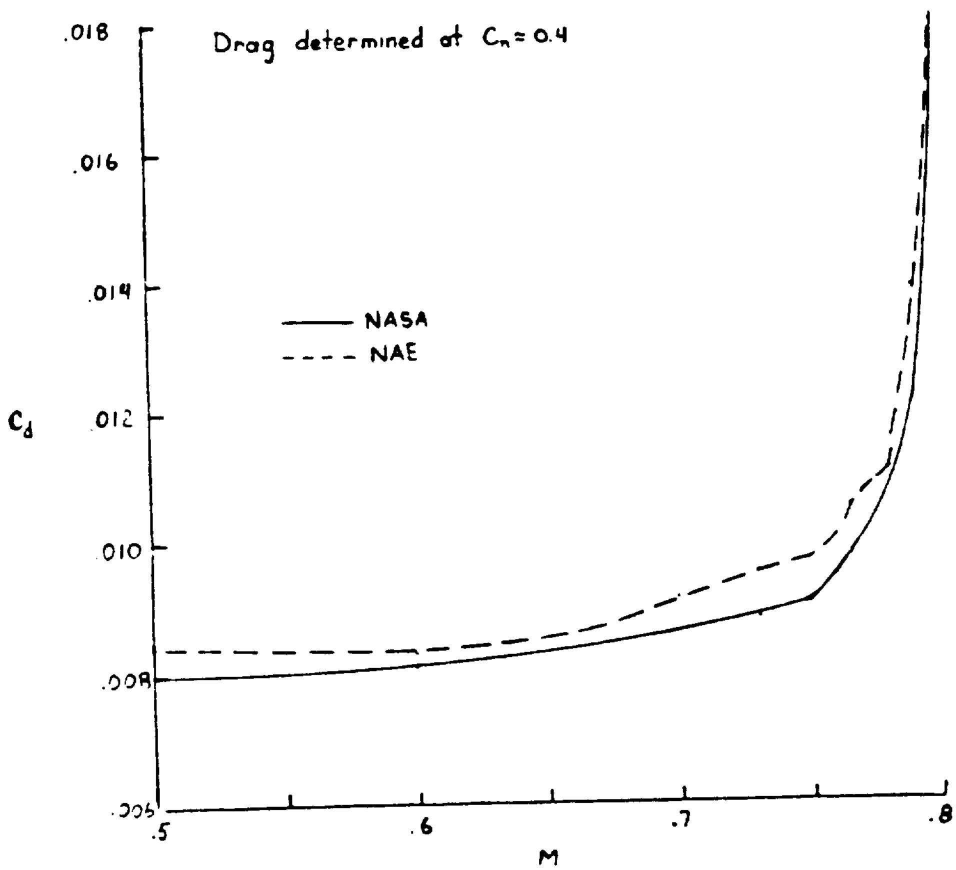
The slope of the normal force curve was measured from the faired data at $c_n \approx .4$. Both sets of data show the same trends. The maximum value of normal force curve slope occurs at $M \approx .78$. In general, the slopes are larger for the NASA results. The differences in the slopes are accentuated by the rapid change in slope with Mach number. Again, assume that there is a residual error in the NASA results. If the difference was attributed to a residual interference in the Mach number, the peaks would not line up. Also, it would indicate the NASA Mach number was higher than measured. This doesn't agree with the previous speculation. The difference could be attributable to an overcorrection of the lift interference which decreases with increasing lift.

Comparison of the Normal Force Curve Slopes



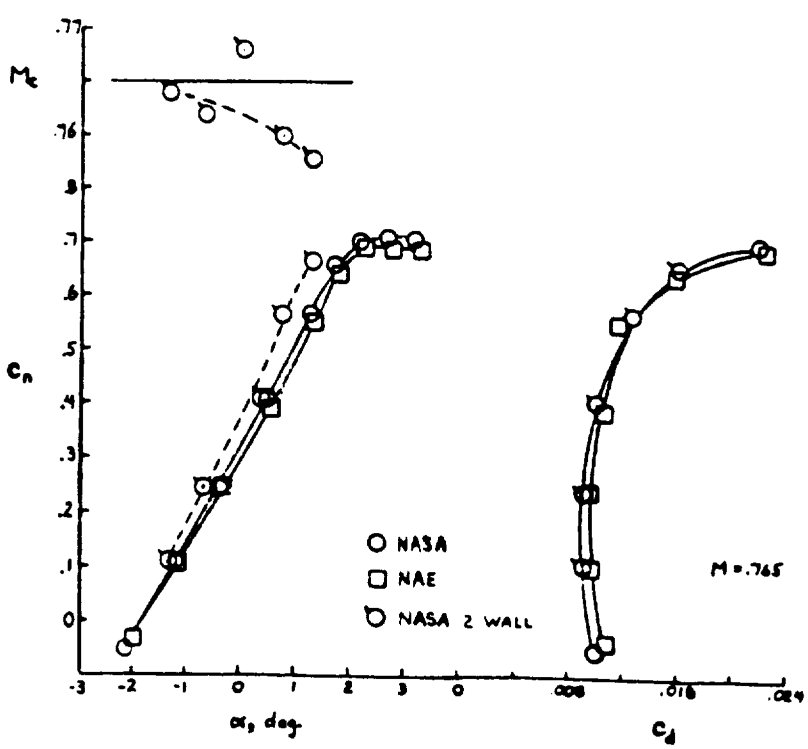
The drag rise was determined from the faired normal force — drag polars with c_d values determined at $c_n \approx 0.4$. The drag level was lower for the NASA tests. The trends around the design Mach number of 0.765 are different. If the NASA corrected Mach number were lower than the measured Mach number, the correlation at the higher Mach number would be improved.

Comparison of the Drag Rise



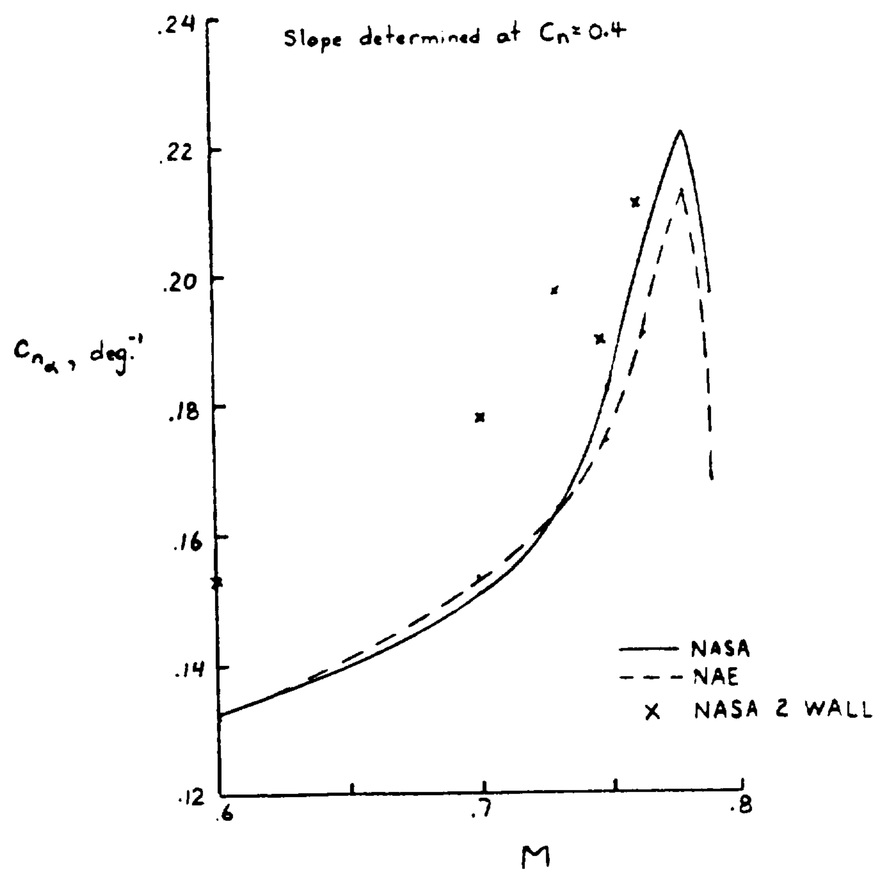
The above comparisons speculated that a residual error remained in the NASA results. When these results were published, there were no production correction techniques which would treat the non-planar boundary condition at the flexible walls. Green of NASA Langley has modified the non-linear correction technique developed by Kemp. This modified code will treat top and bottom walls only or all four walls. The NASA results were corrected using the top and bottom wall (2 wall) option. The code predicted a wall induced downwash which decreased the angle of attack. The correction increased with increasing normal force coefficient. The code also predicted that the actual Mach number was less than the measured Mach number. The correction also increased with increasing normal force coefficient. The effect of the 2-D wall correction on the correlation is mixed as shown on the next page.

Effect of Correcting the NASA Integrated Force Coefficients for Top and Bottom Wall Interference



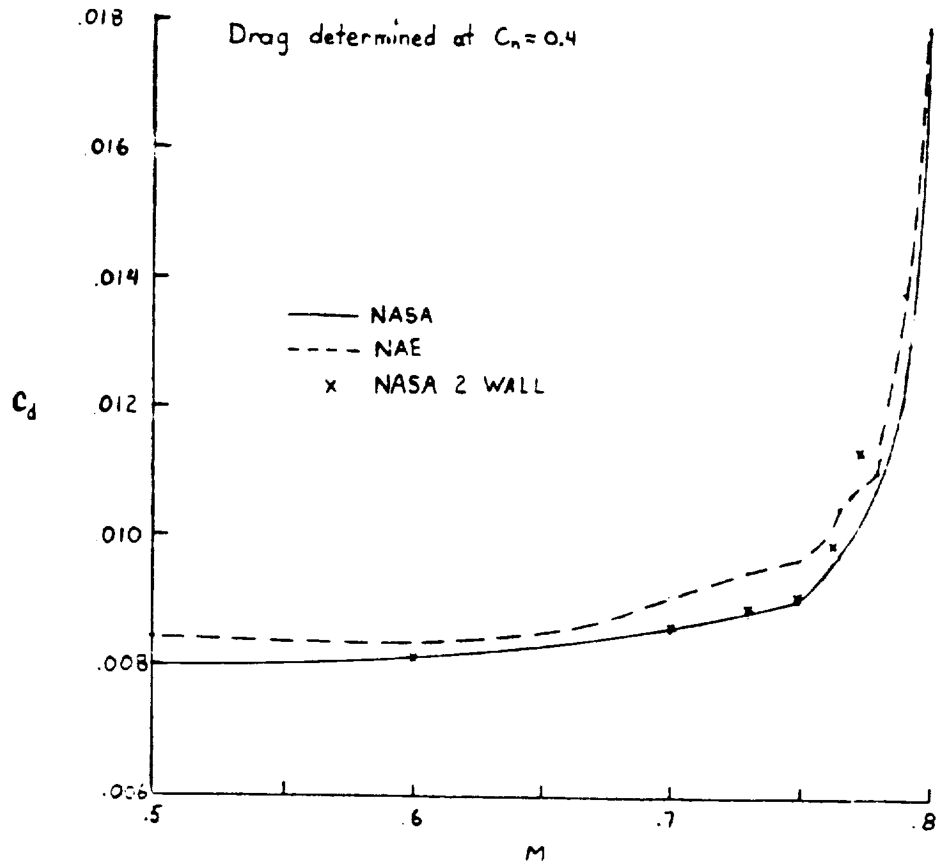
The corrected NASA results were faired and the corrected normal force curve slopes computed. In all cases, the correction drastically degrades the correlation. It is possible that the correction technique is being used incorrectly or that the test section is too short. It is also possible that some of the assumptions used to develop the code are being violated. These results are undergoing further study. They should not be used to form any conclusions of the validity of the NASA results or the correction code at this time.

Effect of Correcting the NASA Normal Force Curve Slopes for Top and Bottom Wall Interference



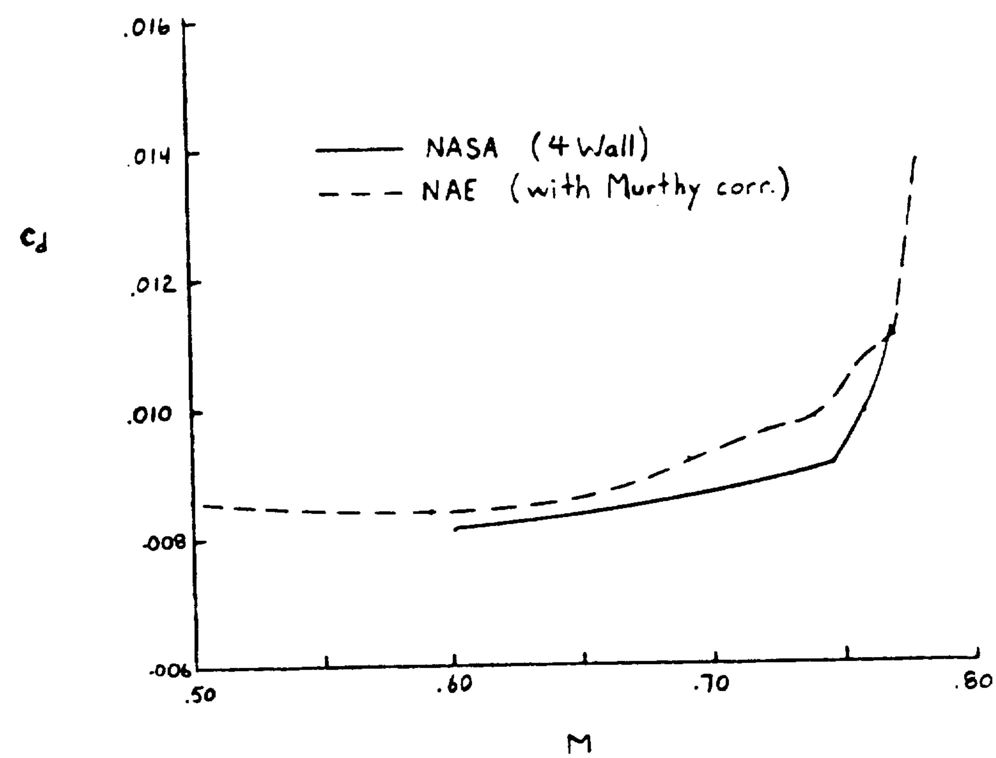
The corrected NASA normal force — drag polar was faired and the drag determined as before. The correction does not have much of an impact on the correlation.

Effect of Correcting the NASA Drag Rise for Top and Bottom Wall Interference



Both tunnels have similar sidewall boundary-layer layer characteristics and the model, as tested, has similar aspect ratios. Neither the NAE wall correction technique for the NASA wall adaption technique directly accounts for the changes in the sidewall boundary layer. The blockage changes will be sensed by the wall static pressures. The effect measured at each wall will be different since the test section heights are so different. If the measured effect is small and can be neglected, the analytical approach by Murthy can be used to correct both sets of data. Since this is only a blockage correction, only the Mach number (and dynamic pressure) will be corrected. The sidewall correction will not affect the angle of attack. Correcting the results for the sidewall interference improves the correlation at the highest Mach numbers where there is a large gradient. It does little to improve the correlation elsewhere.

Effect of Applying the Murthy Sidewall Boundary Layer Correction to the Mach Number



I wish to express my appreciation to the National Aeronautical Establishment of Canada for all its efforts in support of this cooperative agreement. I would like to thank Mr. Lars Ohman and his staff for their assistance and cooperation. Their efforts helped to make this cooperative program a success. I wish to also thank the staff of the 0.3-m TCT for their help in preparing and testing the model and reducing the data. Finally, I would like to express my deepest appreciation to Dr. Y. Y. Chan for his patience, cooperation, insight, and understanding.

CONCLUSIONS

- The adaptive wall test section reduced the wall interference
- Uncorrected adaptive wall and corrected porous wall results:
  - Showed similar pressure profiles and shock locations
  - Showed similar trends of normal force curve slope and drag rise
  - Differences suggest a residual Mach number and angle of attack interference remains
- Correcting adaptive wall results for top and bottom wall residual interference does not improve the correlation
- Correcting results for sidewall interference has a small effect on the correlation