D12

GAS SIDE HEAT TRANSFER

Larry D. Hylton
Allison Gas Turbine Operations
General Motors Corporation

Improvements in methods for predicting heat transfer rates on the hot gas side of turbine airfoils are necessary for improved turbine durability and performance. The development and verification of improved analytical models requires a systematic, closely coupled experimental and analytical program.

Work in this area has been performed under two contracts. The first, NAS 3-22761, addressed the problem of the prediction of hot gas side heat transfer rates to internally cooled (non-film cooled) airfoils. This effort was completed this year. The second contract, NAS 3-23695, is currently underway and will investigate the effect of leading edge "showerhead" film cooling on downstream heat transfer rates.

The objectives of the first contract were to assess the capability of currently available modeling techniques to predict airfoil surface heat transfer distributions in a 2-D flow field, acquire experimental data as required for model verification, and to make and verify improvements in the analytical models.

Two data sets, Turner (ref. 1) and Lander (ref. 2), were selected from the literature for use in evaluating existing models. Two additional airfoils were chosen for cascade testing under this contract. These airfoils, designated the Mark II and C3X, are representative of highly loaded, low solidity airfoils currently being designed. The aerodynamic configurations of the two vanes were carefully selected to emphasize fundamental differences in the character of the suction surface pressure distributions and the consequent effect on surface heat transfer distributions. Cross sections of the four airfoils and the grid used to make inviscid flow predictions for each airfoil are shown in figure 1. Note the significant variation in airfoil geometry. This variation was intended to provide a significant test of the analytical models. Predicted surface pressure distributions for the four airfoils are shown in figure 2.

The two heat transfer cascades tested were run in the Allison Aerothermodynamic Cascade Facility (ACF). The facility, described in figure 3, provides the capability of obtaining both heat transfer and aerodynamic measurements at simulated engine conditions. The experimental measurements were made in moderate-temperature, three-vane cascades under steady-state conditions. The principal independent parameters (Mach number, Reynolds number, turbulence intensity, and wall-to-gas temperature ratio) were varied over ranges consistent with actual engine operation, and the test matrix was structured to provide an assessment of the independent influence of each parameter. The test matrix over which both cascades were operated is shown in figure 4. Data was obtained at two exit Mach numbers, 0.9 and 1.05, and over a range of exit Reynolds numbers from $1.5\text{X}10^6$ to $2.5\text{X}10^6$. The inlet turbulence intensity, Tu, and wall-to-gas temperature ratio, T_w/T_g, were also varied.

The method employed in the facility to obtain airfoil surface heat transfer measurements is shown schematically in figure 5. Basically, the exterior of the airfoil is instrumented with grooved surface thermocouples with this data serving as the exterior boundary condition input to a finite element analysis. The internal boundary conditions are calculated heat transfer coefficients in the internal cooling holes. These values are calculated from measurements of the coolant temperature, pressure and flowrate. The instrumented airfoil also contains static pressure taps, thus permitting simultaneous measurement of the surface pressure and heat transfer distributions. A typical set of data for the C3X cascade illustrating the effects of exit Reynolds number on the heat transfer distribution is shown in figure 6. Note that the heat transfer measurement technique provides sufficient detail to clearly see the effect of Reynolds number on transition on the suction surface. Data from the two cascades, coupled with that from the literature cases, provide a data base covering a wide range of operating conditions and geometries and thus present a significant test for the predictive capabilities of the analytical methods.

The analytical methods development program consisted of two separate phases. In the first phase, the literature was reviewed to establish general candidate methods that were characteristic of current methodology incorporated within actual gas turbine preliminary design systems. As a result of this survey, three 2-D boundary layer methods were chosen: an integral method, a finite difference (differential) method with a zero-equation mixing length hypothesis turbulence model, and the same differential method with a two-equation turbulence model. The literature was thoroughly reviewed to obtain relevant airfoil heat transfer experimental data to use in a general evaluation of the three selected boundary layer methods. The data sets were selected based on relevance to realistic gas turbine environments (i.e., Reynolds number effects, free-stream turbulence effects, strong pressure gradient effects, etc.). Analytical/numerical solutions were compared with experimental results. Based on the findings of this first phase general methods evaluation process, the differential method with zeroth order turbulence modeling was selected for the second phase of the analytical program. The literature was further reviewed for models that had the potential of treating the airfoil heat transfer problem more realistically. A number of transition process models, free-stream turbulence augmentation models, and a single explicit longitudinal surface curvature correction model were selected for evaluation. Various single and/or combined model solutions were evaluated using data from four different airfoil experiments. This evaluation process eventually led to a final "gas turbine airfoil specific" modeling effort which resulted in an effective viscosity formulation that, when implemented, gave better overall solutions than any literature modeling approach tested previously.

Figures 7a and 7b respectively show the unmodified and modified suction surface heat transfer predictions for three different operating conditions using the STAN5 boundary layer code for the Lander airfoil. Increasing run numbers correspond to increased inlet or exit Reynolds number and free-stream turbulence intensity. The experimental data are represented as symbols. Lander's data are important in that they illustrated nominally laminar heat transfer augmentation attributed to free-stream turbulence effects, as well as Reynolds number effects related to transition origin. As shown in figure 7, the augmentation phenomenon is predicted significantly better by the final model.

Figures 8a and 8b show the unmodified and modified solutions compared with the data of Turner. The significance given to Turner's data was that they isolated the effects of free-stream turbulence. Figure 8a shows only one solution because the original unmodified method did not account for the effects of free-stream turbulence. As can be seen in figure 8b, the modified solutions give a very good representation of the pressure surface experimental data. The modified suction surface solutions give reasonable trends up to the point where a transition process is indicated by the experimental data. Overall, the modified solutions are a significant improvement over the unmodified solution, represent the pressure surface data very well, and provide qualitatively good trends for the suction surface.

Unmodified and modified predictions of the characteristic Reynolds number effects in the Mark II cascade are compared with the data in figures 9a and 9b, respectively. It should be pointed out that the analytically predicted stagnation point was displaced approximately 5% (0.05) of pressure surface distance toward the pressure surface away from the extreme forward point on the airfoil, which was used as the datum (0) in these figures. The stagnation point corresponds to the predicted inviscid flow solution zero velocity location on the pressure surface. Note that this does not correspond to the highest local value of measured heat transfer in the leading edge region. Both the modified and unmodified solutions reflect the proper trends moving away from the stagnation point. The absence of solutions beyond 0.2 normalized surface distance on the suction surface indicates that all solutions encountered separation due to the presence of a suction surface shock at the location. No attempt was made to restart the solutions downstream of the shock. Overall, the modified solutions are able to qualitatively and quantitatively predict the pressure surface data reasonably well and yield much better predictions than the unmodified solutions, which predicted pressure surface transition.

In a manner similar to the Mark II comparative studies, the experimental results for the C3X cascade were also simulated numerically and the predictions are shown in figure 10. Figures 10a and 10b show both unmodified and modified solutions at three different Reynolds number conditions. Qualitatively, the modified pressure surface solutions represent a substantial improvement over the original (unmodified) approach. However the quantitative predictions (using the modified procedure) begin to deviate significantly from the data along the aft portions of the surface. The suction surface predictions of both the unmodified, figure 10a, and the modified, figure 10b approaches yield quantitatively acceptable results for some of the cases, but the indicated suction surface transition process (i.e., gradual transition) is better represented by the modified solutions.

Finally, in response to the objectives of this program, a recommended procedure was developed for constructing a viable, 2-D airfoil external convective heat transfer method for gas turbine design systems, including the specification of boundary conditions, initial conditions, and preferred definitions of effective viscosity determined here to be most suitable for gas turbine preliminary design applications.

The analytical and experimental work performed under contract NAS 3-22761, including the recommended design procedure, are reported in detail in NASA CR 168015 (ref. 3) which was published in May, 1983.

Work on the second contract, NAS 3-23695, began earlier this year. This effort is intended to extend the work performed under the first contract in two respects. First, the analytical boundary layer analysis and experimental cascade studies of the first contract will be extended to include a leading edge showerhead film cooling array. Secondly, recognizing the long term limitations of the boundary layer approach, an analytical effort was added to investigate the application of a Navier-Stokes solver to the turbine cascade problem. This effort was subcontracted to Scientific Research Associates, Inc. (SRA) and will utilize their MINT code.

The boundary layer efforts are structured similar to the first contract. Namely the STAN5/STANCOOL type approach to making heat transfer predictions for airfoils with leading edge film cooling will be evaluated and modified as required to improve heat transfer predictions. These modifications will then be validated by comparisons with experimental data acquired under the contract.

The experimental program utilizes the C3X cascade from the original contract. The center, instrumented airfoil was replaced with one containing five rows of film cooling holes in the leading edge region. Figure 11 illustrates the original airfoil with the radial cooling holes and finite element grid shown. The new film cooled airfoil was modified in the leading edge region as shown in figure 12. Downstream of the region shown in figure 12 the cooling hole geometry and instrumentation is identical to the non-film cooled airfoil tested under the first contract. Heat transfer measurements will be made downstream of the film cooling array. No heat transfer measurements will be made within the array. By using a modified airfoil of the same profile as one of the two tested under the original contract, a good non-film cooled baseline will have been established, and fully qualified cascade hardware and experimental techniques will be employed to maximum advantage.

The analytical efforts on the Navier-Stokes MINT code are intended to provide a major step toward developing an analytical tool capable of predicting the flow and heat transfer in a full 3-D turbine cascade. Initial efforts on the MINT code will be performed on a 2-D version. The "C" grid generator previously used in the program will be replaced with an "O" grid generator. Due to its construction, the "C" grid requires a cusped trailing edge. In solving the transonic turbine problem, the trailing edge geometry appears to be very important. In preliminary calculations with the "C" grid, discrepancies between predictions and experimental data in the trailing edge region are thought to be a result of the cusped trailing edge approximation. The addition of the "O" grid generator to the program should enhance its capabilities.

The 2-D version of the code with the "O" grid generator will be used to make predictions for the cascades tested under the first contract. The comparison of the predictions with the data will serve as verification of the code modifications. Following these comparisons the code will be extended to handle the full 3-D case and a sample calculation for a 3-D cascade will be made. The final modification scheduled to be made to the code is the incorporation of a film cooling-trailing edge blowing capability into the code. Following completion of all modifications to the code, SRA will assist NASA personnel in running demonstration cases with the code on the NASA Lewis computer.

Progress on the second contract has been made on both analytical efforts and on the experimental effort. The boundary layer analytical efforts have concentrated on evaluating the STAN5/STANCOOL programs abilities to predict heat transfer to circular cylinders. This is the initial step in predicting the airfoil leading edge film cooling problem. Comparisons of the predictions for both solid cylinders and film cooled cylinders were made with the data in reference 4. Based on these results efforts are currently underway to make modifications to the models to improve their capabilities.

Progress on the MINT code includes installation and checkout of the program on the NASA Lewis computer. In addition, the "O" grid generator has been developed and sample "O" grids have been constructed. Efforts on verifying the capabilities of the program with the "O" grid generator by comparing predictions with cascade data sets are currently underway.

Under the experimental phase of the contract the new C3X airfoil containing the leading edge "showerhead" film cooling array, has been designed, fabricated, instrumented and installed in the cascade. Preliminary testing of the cascade has started. Technical efforts on this program are scheduled to be completed by May, 1984.

REFERENCES

1. Turner, A. B.: Local Heat Transfer Measurements on a Gas Turbine Blade, Jour. of Mechanical Engng. Sciences, Vol. 13, 1971, pp. 1-12.

2. Lander, R. D.: Effect of Free-Stream Turbulence on the Heat Transfer to Turbine Airfoils, Tech. Rept. AFAPL-TR-69-70, Air Force Systems Command, Sept. 1969.

3. Hylton, L. D.; Mihelc, M. S.; Turner, E. R.; Nealy, D. A.; and York, R. E.: Analytical and Experimental Evaluation of the Heat Transfer Distribution Over the Surfaces of Turbine Vanes, Detroit Diesel Allison Report EDR 11209, NASA CR-168015, May 1983.

4. Luckey, D. W.: Stagnation Region Gas Film Cooling: Spanwise Angled Injection from Multiple Rows of Holes, PhD Thesis, Purdue University, 1980.

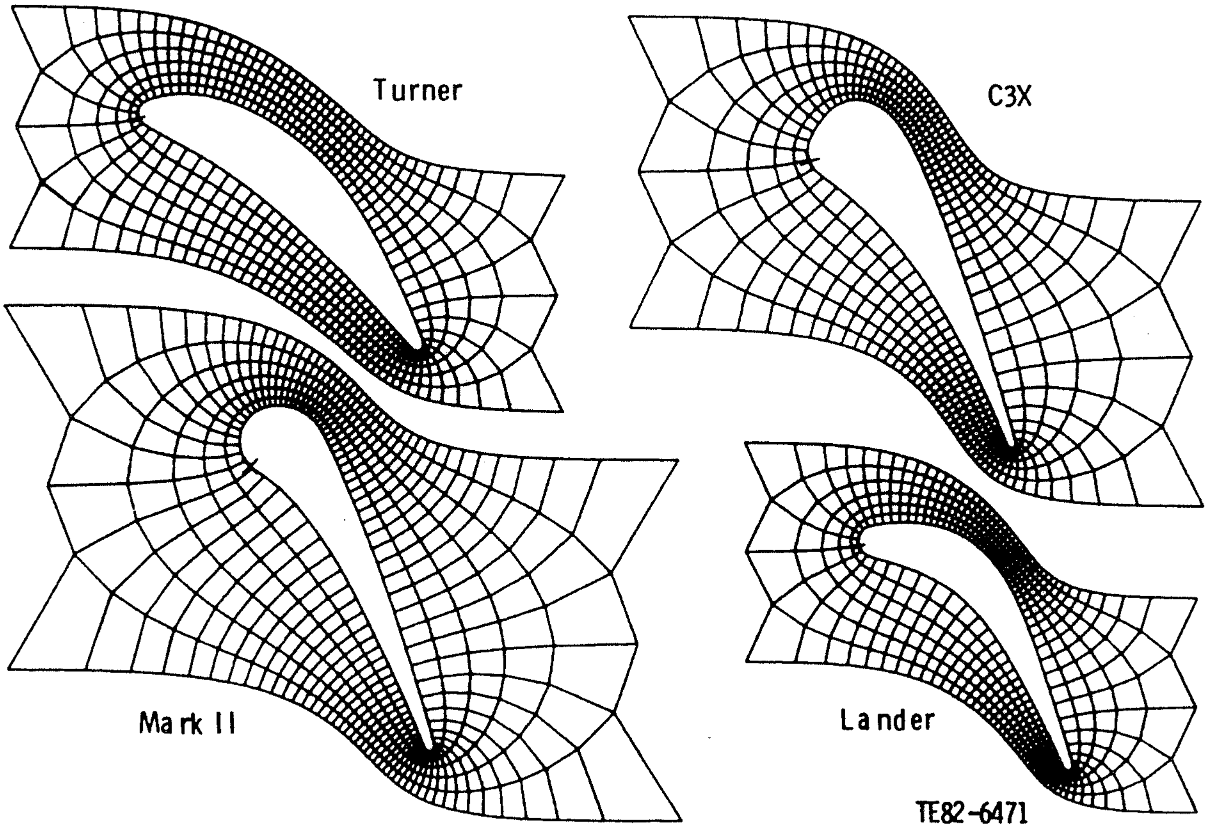


Figure 1. Airfoil profiles with inviscid flow analysis grid.

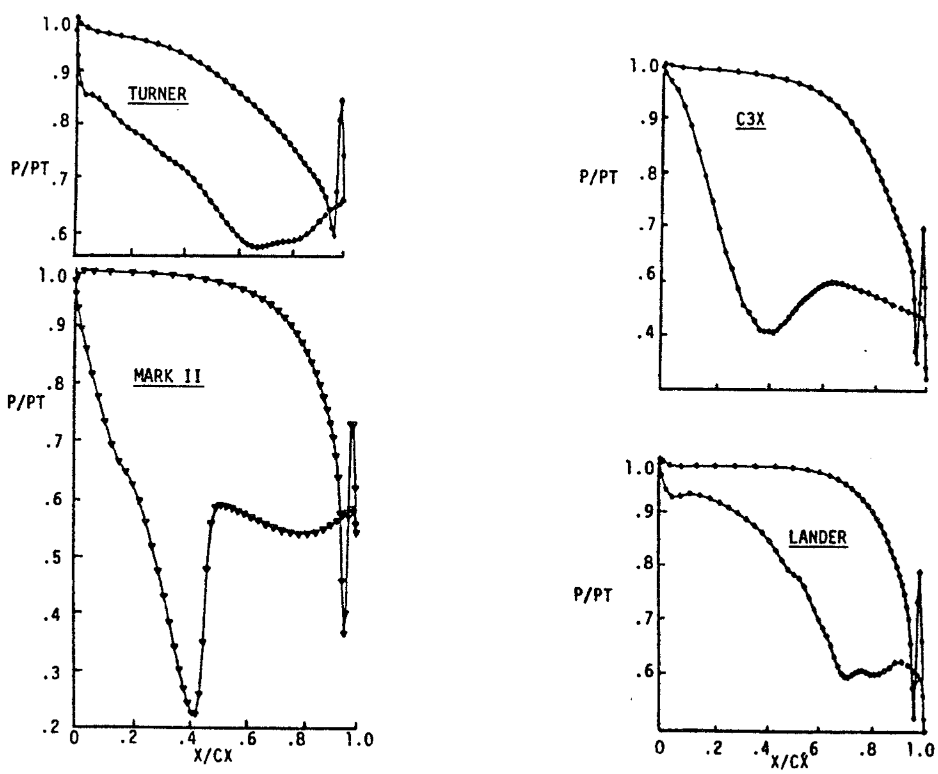
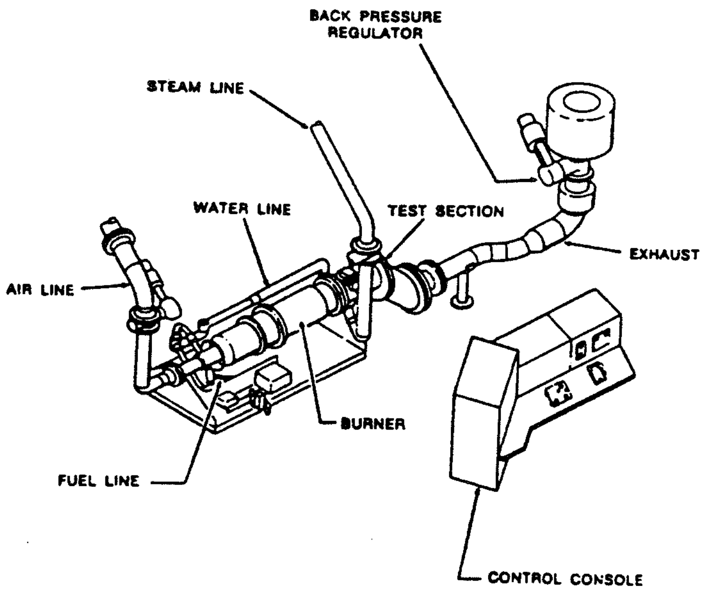


Figure 2. Airfoil predicted pressure distribution.



OPERATIONAL PHILOSOPHY

- o TWO-DIMENSIONAL LINEAR CASCADE
- o PROVIDES HEAT TRANSFER AND AERODYNAMIC DATA SIMULTANEOUSLY
- o OPERATED AT SIMULATED ENGINE CONDITIONS
  - o REDUCED TEMPERATURE
  - o REDUCED PRESSURE
  - o SCALED-UP AIRFOIL GEOMETRY
  - o HIGH FREESTREAM TURBULENCE
- o WIDE OPERATING RANGE
  - o REYNOLDS NUMBER CONTROL
  - o EXIT MACH NUMBER CONTROL
  - o WALL-TO-GAS TEMPERATURE RATIO CONTROL
  - o INLET TURBULENCE INTENSITY CONTROL
- o HIGH DENSITY INSTRUMENTATION
  - o UP TO 300 TEMPERATURES
  - o UP TO 288 PRESSURES
- o DEDICATED FACILITY COMPUTER
  - o COMPUTER CONTROLLED DATA ACQUISITION
  - o ONLINE DATA ANALYSIS

OPERATIONAL SPECIFICATIONS

AIR SUPPLY	9.5 LB/SEC AT 105 PSIA OR 5.0 LBM/SEC AT 245 PSIA
INLET PRESSURE	20 PSIA TO 245 PSIA
STAGNATION TEMPERATURE	400^{o}F TO 3200^{o}F
PRIMARY FUEL	NATURAL GAS

Figure 3. Allison Aerothermodynamic Cascade Facility

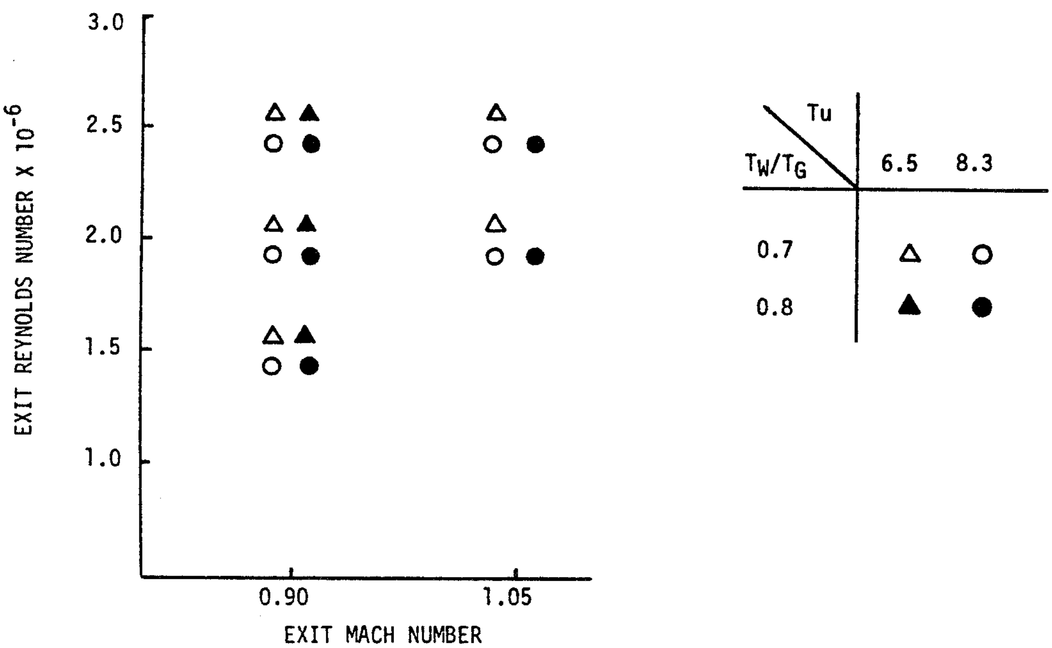


Figure 4. Cascade test matrix.

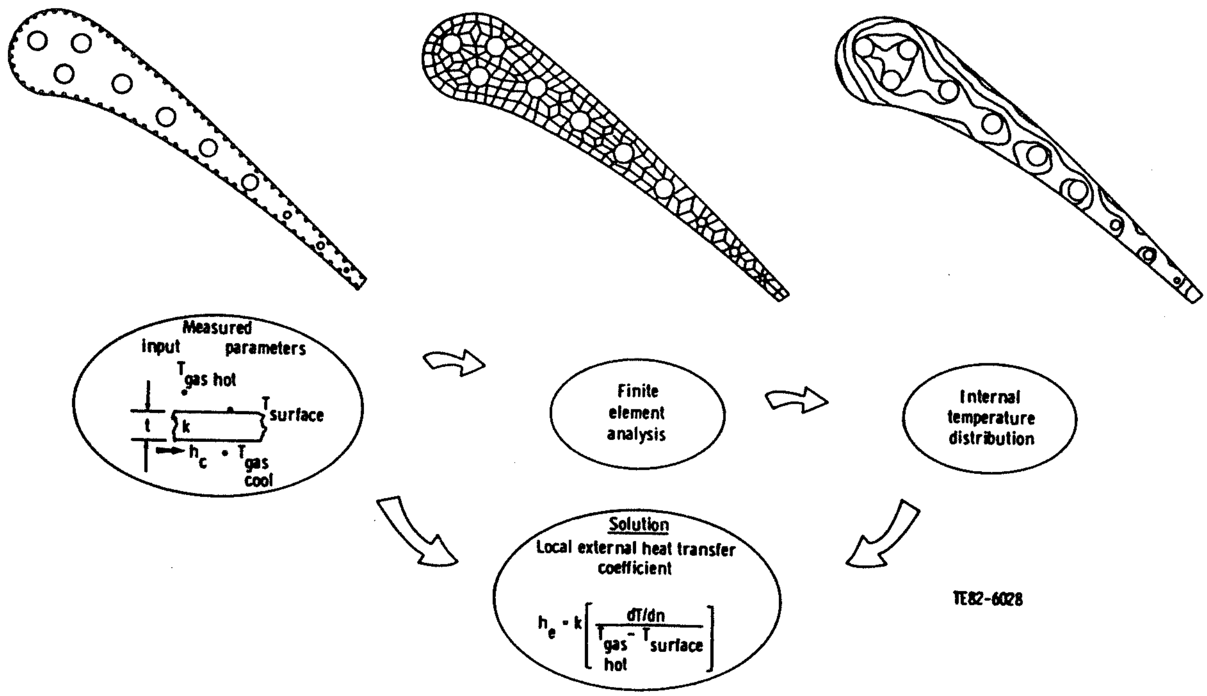


Figure 5. Heat transfer measurement technique.

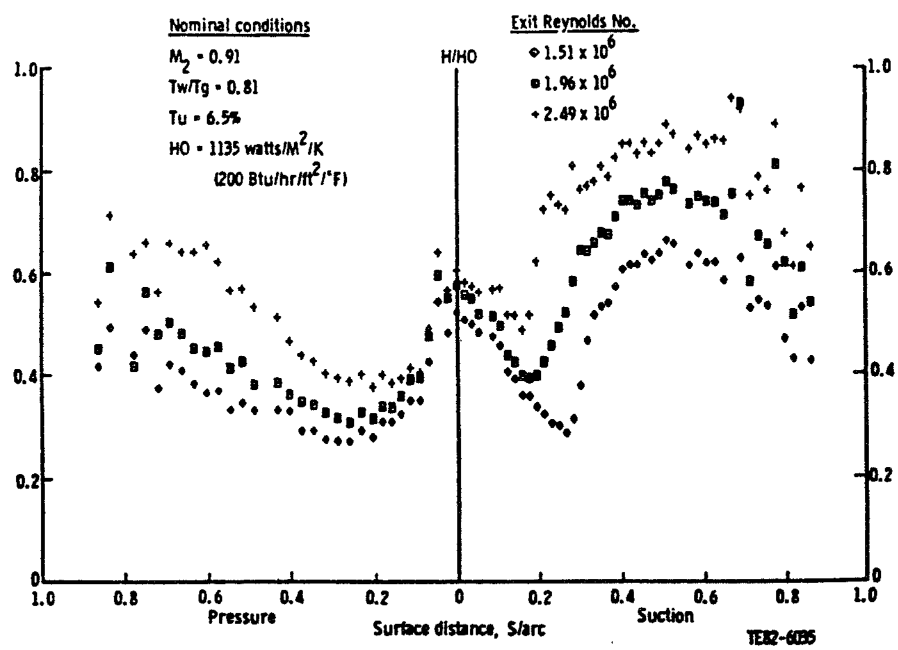
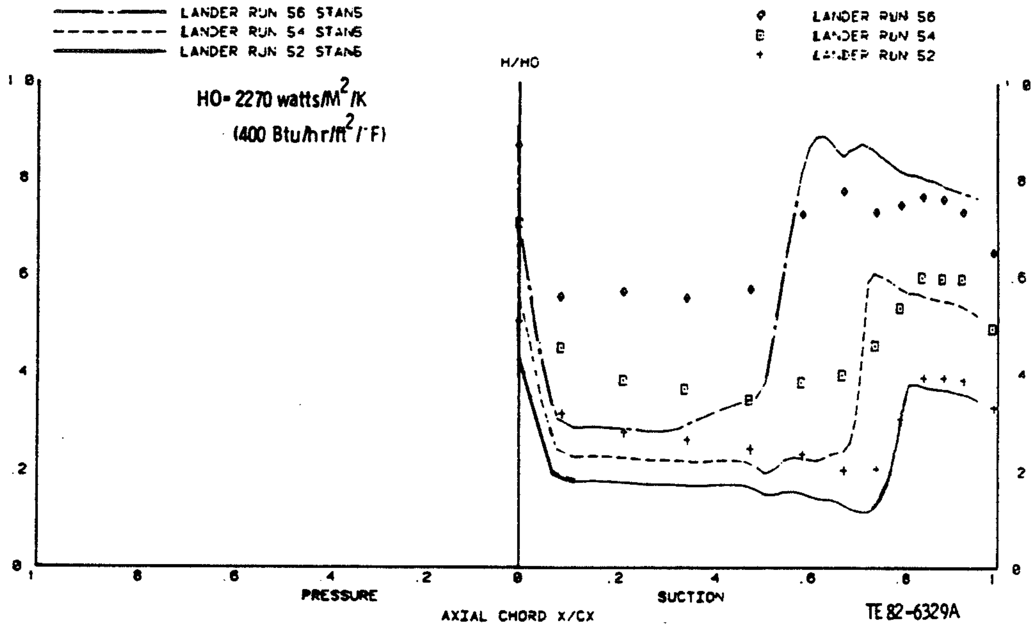
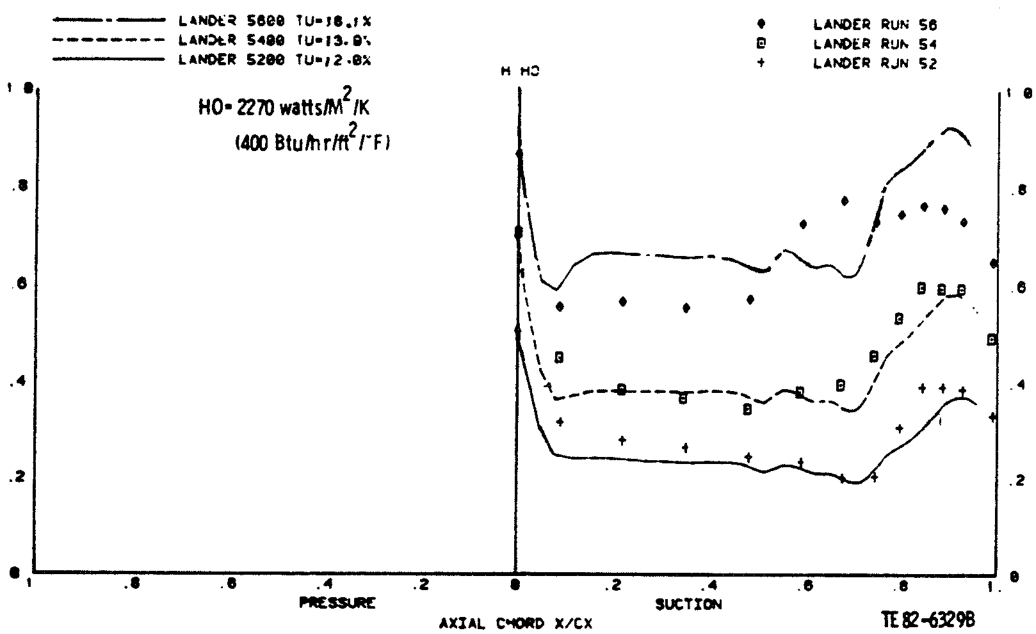


Figure 6. Reynolds number effect on surface heat transfer distribution for C3X airfoil.

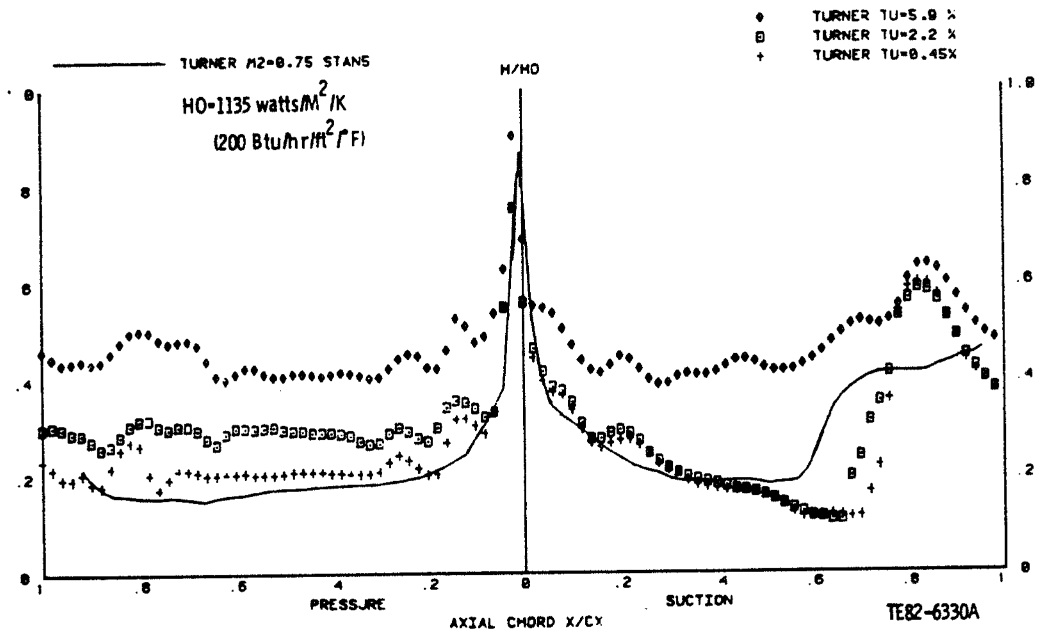


(a) Unmodified STAN5 results

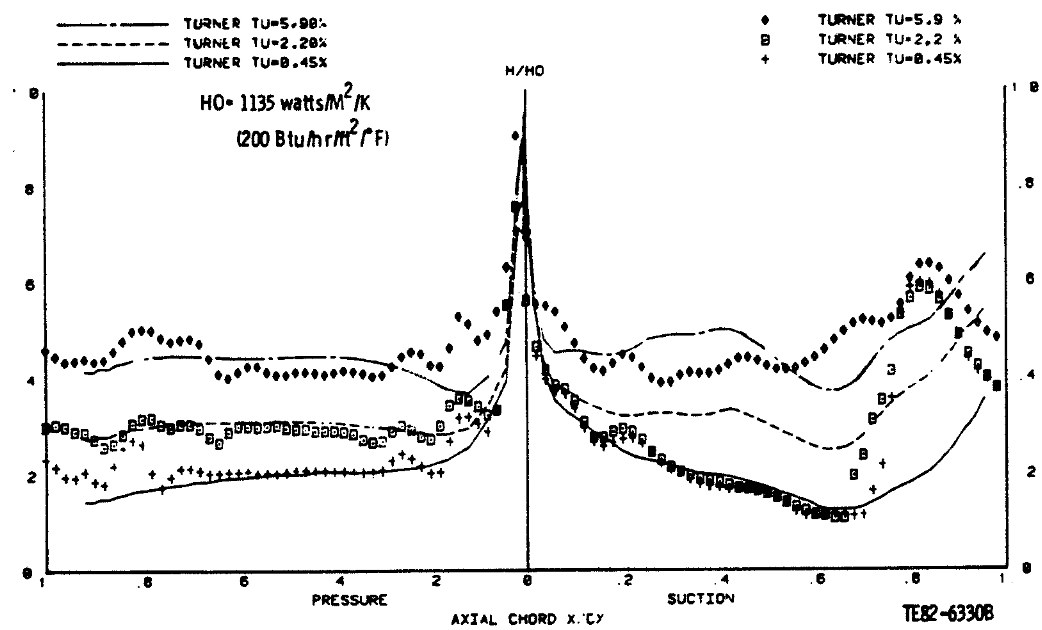


(b) Modified STAN5 results

Figure 7. STAN5 solutions compared with Lander airfoil suction surface experimental heat transfer coefficient data illustrating the combined effects of varying Reynolds number and free-stream turbulence intensity.

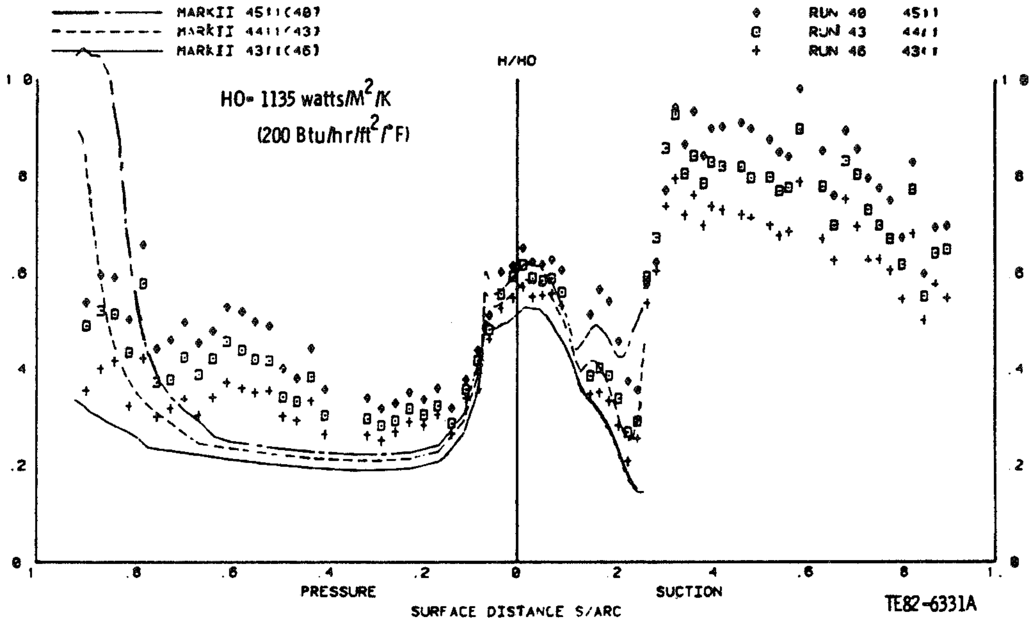


(a) Unmodified STAN5 results

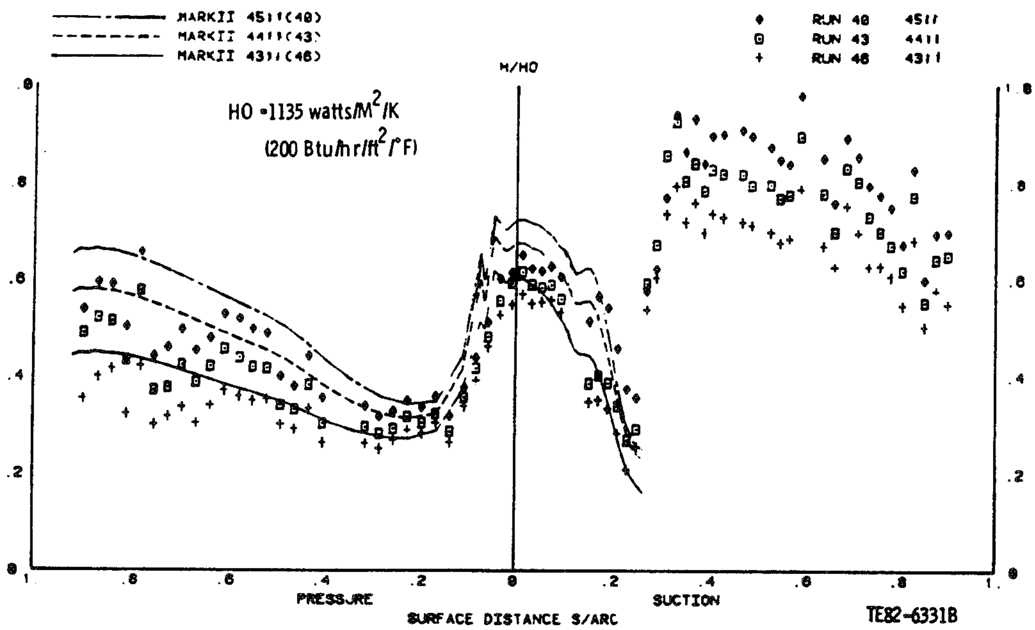


(b) Modified STAN5 results

Figure 9. STAN5 solutions compared with Turner airfoil experimental heat transfer coefficient data illustrating the effects of varying free-stream turbulence intensity.

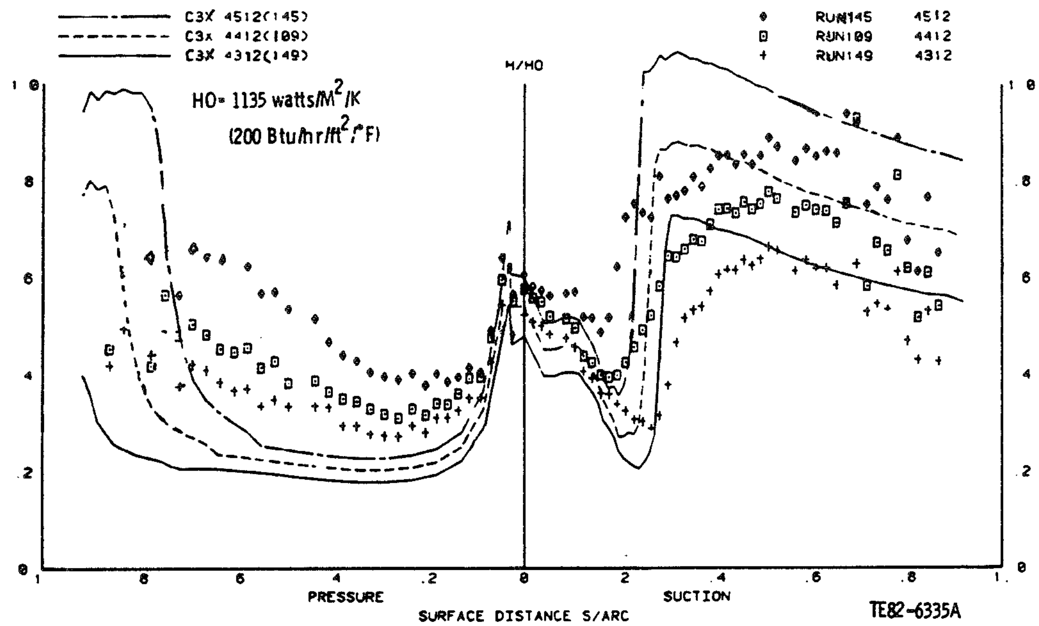


(a) Unmodified STAN5 results

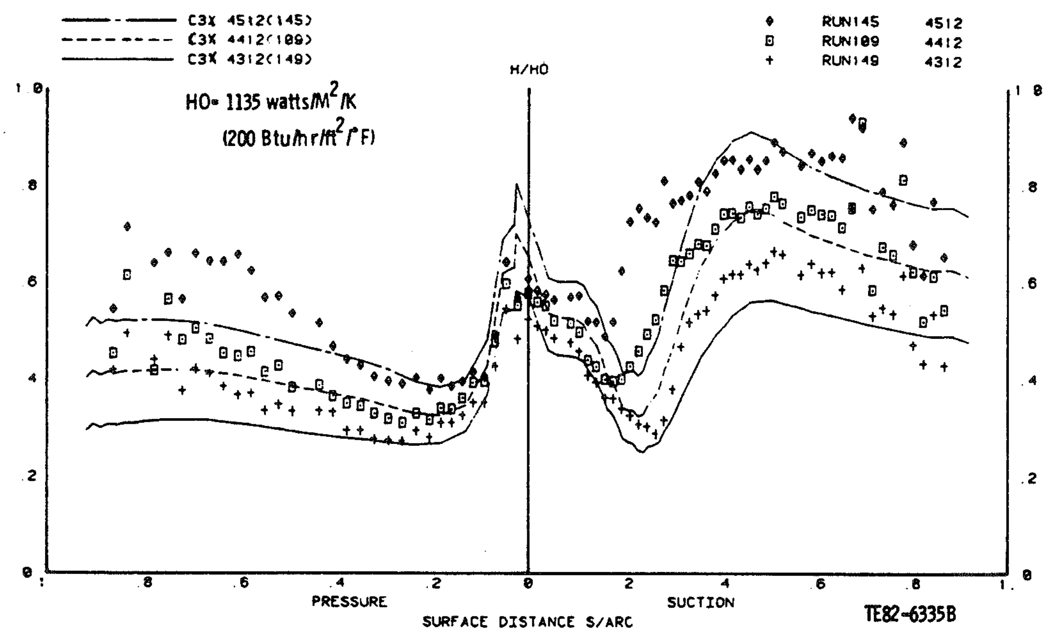


(b) Modified STAN5 results

Figure 9. STAN5 solutions compared with Mark II airfoil experimental heat transfer coefficient data illustrating the effects of varying exit Reynolds number.



(a) Unmodified STAN5 results



(b) Modified STAN5 results

Figure 10. STAN5 solutions compared with C3X airfoil experimental heat transfer coefficient data illustrating the effects of varying Reynolds number.

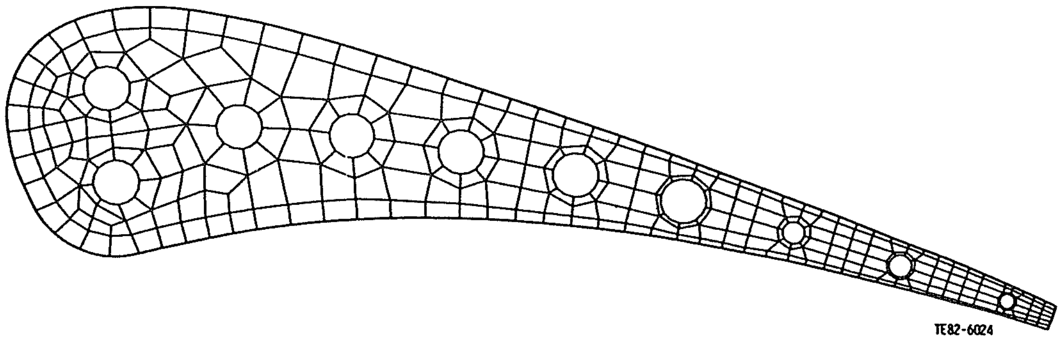


Figure 11. C3X airfoil cross section with finite element grid and cooling hole locations for contract NAS 3-22761.

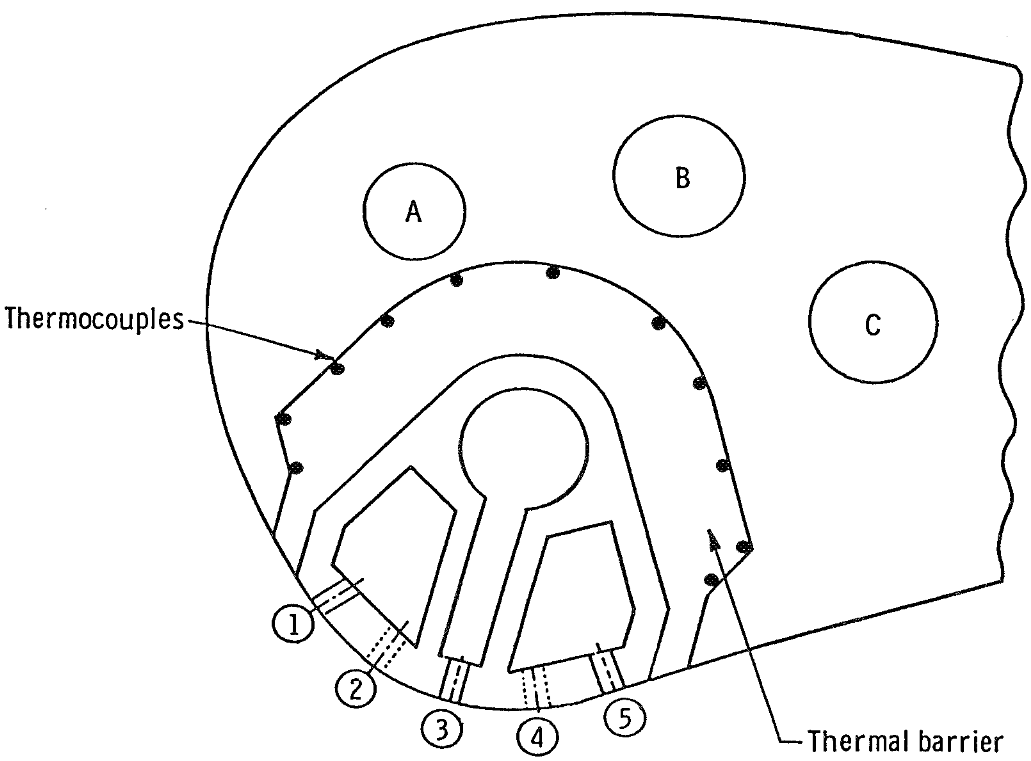


Figure 12. Modified leading edge region of C3X airfoil for film cooling studies in contract NAS 3-23695.