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FILM COOLING HEAT TRANSFER ON A TURBINE AIRFOIL*

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INTRODUCTION

Recently considerable emphasis has been placed on developing more accurate analytical models for predicting hot gas side heat transfer rates to turbine airfoils. In order to achieve the durability and performance goals of new engines, cooling system designs must be carefully tailored to each application. This requires an accurate assessment of the hot gas side thermal loading. The development and verification of improved analytical models requires a systematic, closely coupled experimental and analytical program. The objectives of the current program are to develop an analytical approach, based on boundary layer theory, for predicting the effects of airfoil film cooling on downstream heat transfer rates and to verify the resulting analytical method by comparison of predictions with hot cascade data obtained under this program.

BACKGROUND

The overall approach to attaining the stated objective has involved a series of three programs as illustrated in figure 1. The initial program, performed under Contract NAS3-22761, assessed the capability of available modeling techniques to predict non-film cooled airfoil surface heat transfer distributions, acquired experimental data as needed for model verification, and provided verified improvements in the analytical models. This effort resulted in a baseline predictive capability and was reported in CR 168015 (ref. 1) published in May 1983.

The problem of heat transfer predictions with film cooling was broken into sequential efforts with the effect of leading edge showerhead film cooling being investigated first, followed by a program to study the effects of the addition of discrete site suction and pressure surface injection. The effort on showerhead film cooling was performed under Contract NAS3-23695 and was reported in CR 174827 (ref.2) published in July 1985. As part of that program, a five-row, simulated common plenum showerhead geometry was tested to determine differences between film and non-film cooled heat transfer coefficient distributions downstream of a leading edge, multiple hole film cooling array. Building on non-film cooling modeling improvements incorporated in a modified version of the STAN5 boundary layer code developed under Contract NAS3-22761, a program was developed to analytically model and predict differences resulting from leading edge mass injection.

A summary of the program results including experimental data and corresponding analytical predictions are shown in figures 2-5. Rather than simply form the film cooled Stanton number to non-film cooled Stanton number ratio (St_{FC}/St_{NFC}) to isolate the effects of film cooling on downstream heat transfer, an alternate parameter referred to as Stanton number reduction (SNR) is used. SNR is defined as

$$SNR = 1 - St_{FC}/St_{NFC} \qquad (1)$$

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Therefore, SNR=0 implies "no difference" and positive or negative values imply reduced or increased heat transfer levels respectively. Forming SNR values along the entire test surface gives the actual SNR distribution for the airfoil. In addition, St_{FC}/St_{NFC} is determined using data obtained at equivalent M_2 and Re_2 conditions, so SNR is approximately equal to the actual heat transfer coefficient reduction. Figures 2 and 3 illustrate the formation and type of information given by vane surface SNR distributions. All data shown in these figures were obtained at fixed operating conditions; i.e., $Ma_2 = 0.90$, $Re_2 = 2.0 \times 10^6$, $T_c/T_g = 0.82$. Variable blowing strengths (P_c/P_t = 1.0, 1.30, 1.52, and 1.72) were set at these conditions and heat transfer data were taken. The four different surface heat transfer coefficient distributions determined from the cascade data at the four coolant to free-stream pressure ratio conditions are shown in figure 2. A value of $P_c/P_t = 1.0$ signifies that no coolant is being ejected and $P_c/P_t > 1.0$ signifies that coolant is being ejected. Using the results of figure 2 and the SNR definition, surface SNR distributions can be constructed. These distributions are shown in figure 3. Because each SNR distribution shows only the difference between a given film-cooled and baseline nonfilm-cooled condition, an SNR data presentation is useful for discussing phenomena unique to the film-cooled problem.

The characteristic effect of blowing strength variation is illustrated by the SNR differences shown in figure 3. These results indicate that the most significant differences occur on the suction surface between 20% and 40% of the surface distance. This region corresponds to the suction surface boundary layer transition zone. From figure 2, it can be observed that the suction surface transition origin moves forward on the airfoil as the blowing strength is increased. This results in increases in heat transfer levels (negative SNR) with increasing blowing strength as illustrated in figure 3. Smaller, but significant, increases in heat transfer occur on the pressure surface. These preturbulent increases are similar in character to the increases that would be expected to be caused by increasing the free-stream turbulence intensity from a baseline state. The discrete injection process apparently acts as a turbulence promoter.

The blowing strengths represented in figures 2 and 3 are higher than would be expected in an actual engine design, but were run to better understand the physics of the cooling process. Reducing blowing strengths to levels of interest to the turbine designer ($<$ 1.10) provides the results shown in figure 4. Here the area of increased heat transfer (negative SNR) is limited to the transition zone on the suction surface.

One goal of this effort was to determine whether there were any benefits to be extracted from leading edge injection in terms of recovery region surface protection. Data shown in figure 5 were obtained at variable plenum coolant to mainstream total temperature ratios (T_c/T_g = 0.69, 0.82, and 0.89) and at fixed Ma_2, Re_2, and P_c/P_t conditions. The overall increase in SNR (i.e., decreased heat transfer) as the coolant to gas absolute temperature ratio decreased indicates the positive effect that results from diluting the hot free-stream fluid with the relatively cooler leading edge ejectant. However, as the pressure surface results indicate, the favorable thermal dilution phenomenon is offset by the adverse turbulence generation mechanism associated with the discrete injection process. The net result is that even for T_c/T_g = 0.69, SNR is still negative immediately downstream of the showerhead on the pressure surface. Figure 5 also indicates that the thermal dilution and turbulence generation mechanisms interact on the suction surface in the preturbulent zones although in the fully turbulent zones the SNR result is determined by thermal dilution strength only. These results indicate that leading edge film cooling by itself cannot be used to always offset high near recovery region heat loads even though far recovery region loads are reduced.

Utilizing the modeling improvements made as part of this program SNR distributions were computed for the six blowing condition data sets represented in figures 3 and 4 and are shown in the figures. The comparisons shown in figure 4 indicate that with the exception of the suction surface transition zone, there is little measured and/or predicted effect due to the leading edge injection. This small effect result is significant, because the blowing levels shown ($P_c/P_t \leq 1.10$) are more representative of actual design conditions than the higher blowing cases ($P_c/P_t > 1.10$) shown in figure 3. For the strong blowing condition SNR predictions shown in figure 3, the proposed two parameter method predicts trends reasonably well but quantitative discrepancies exist.

SNR predictions for the variable cooling temperature blowing conditions are shown in figure 5. As illustrated in figure 5, the analytical method does a reasonable job in predicting all of the trends indicated in the data. The detailed results of the technical effort under Contract NAS3-23695 are reported in NASA CR 174827 (ref. 2) which was published in July 1985.

CURRENT PROGRAM

Work under NASA Contract NAS3-24619 was started in August 1985. The objectives of this program are to extend the analytical airfoil film cooling code development to include discrete site pressure and suction surface injection, with and without leading edge blowing, and to obtain relevant hot cascade data to verify the model improvements.

Analytical Approach

The overall approach will be to extend a base 2-D boundary layer code containing the leading edge showerhead cooling model reported in reference 2 and the multiple row film cooling model implemented in the STANCOOL code (ref. 3). Three phases have been defined to accomplish the objective of producing a tool that is acceptable in terms of qualitative/quantitative accuracy and relatively easy to incorporate within a present day turbine airfoil design system. The three phases consist of a design mode analysis phase, a method characterization phase, and a method refinement/verification phase.

The design mode analysis phase is intended to demonstrate the use of the base boundary layer method in a film-cooled turbine airfoil design system environment. This initial study will address details involved with method set-up procedures (e.g., defining initial and boundary conditions) and the qualitative behavior of the film cooling models for a relevant film-cooled airfoil design. As part of this analysis, the heat transfer distributions on the film-cooled airfoil to be tested in the hot cascade will be predicted. This initial design mode analysis phase will be followed by a detailed method characterization study. This study will determine the qualitative/quantitative attributes and deficiencies of the proposed method using measured aerodynamic and heat transfer data obtained in the experimental program. Comparisons of the data with the predictions from the design mode analysis will be made at several operating conditions. The final effort, method refinement/verification, will address modeling deficiences discovered in the first two phases. At present, the method being proposed contains four modeling parameters that control predicted film cooling recovery region heat transfer phenomena. Two parameters are associated with the leading edge model described in Turner et al (ref. 2) and two with the full coverage STANCOOL model (ref. 3). It is anticipated that the majority of effort in the final phase will be aimed at determining proper formulations for these parameters for incorporation in a design code. While parameter formulation for the STANCOOL code applied within a gas turbine environment has received recent attention (ref. 4-6), the two parameter leading edge model of Turner et al (ref. 2) will be further tested to demonstrate its range of applicability. Finally, when the models are combined for the case of airfoil geometries with both leading edge and downstream injection, the overall formulation must still perform satisfactorily. Using available film cooled airfoil data, it is anticipated that appropriate formulations for the four parameters can be developed to cover the range of operating conditions of interest to turbine designers.

Experimental Approach

The experimental phases will be an extension of the previous contract work. The hot cascade tests will utilize the same facility and cascade used in the previous contract, with the instrumented airfoil in the cascade replaced with one containing suction surface and pressure surface film cooling arrays in addition to a leading edge showerhead film cooling array. A schematic of the airfoil is shown in figure 6. The film cooled region of the airfoil will be thermally isolated from the remainder of the airfoil. Surface heat transfer measurements downstream of the suction and pressure surface hole arrays will be made using the same technique utilized in the previous contract tests. This technique, illustrated in figure 7, uses experimentally measured steady-state aerothermal boundary conditions as input for numerically solving the heat conduction equation in order to determine the airfoil internal temperature distribution. Once the internal temperature distribution is determined, a local heat transfer coefficient can be determined using the local calculated surface normal temperature gradient, measured wall and gas temperatures, and material conductivity. In addition to heat transfer measurements, the airfoil will be instrumented to obtain the surface static pressure distribution. Also as part of the experimental program, aerodynamic losses for the cascade will be measured at the exit plane by traversing a five hole cone probe across one passage at the airfoil midspan. The cascade will be operated at three levels of exit Reynolds number and two levels of exit Mach number (expansion ratio). Blowing strength and cooling strength for selected blowing configurations will be varied at these operating conditions. The tests will be conducted at constant turbulence intensity and vane-surface-to-gas-absolute temperature ratio (T_w/T_g) levels. The test matrix should provide a significant data base for verifying the analytical models at relevant gas turbine conditions.

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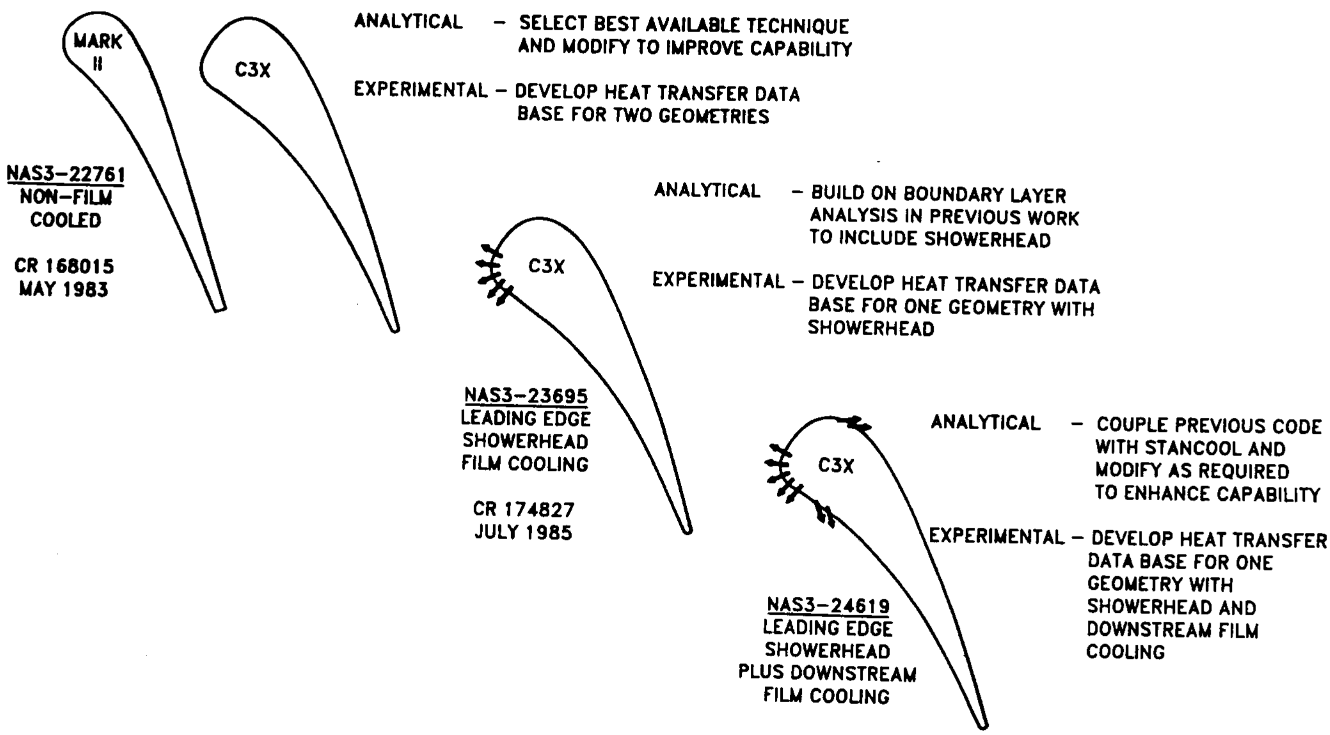

Figure 1. Overall approach

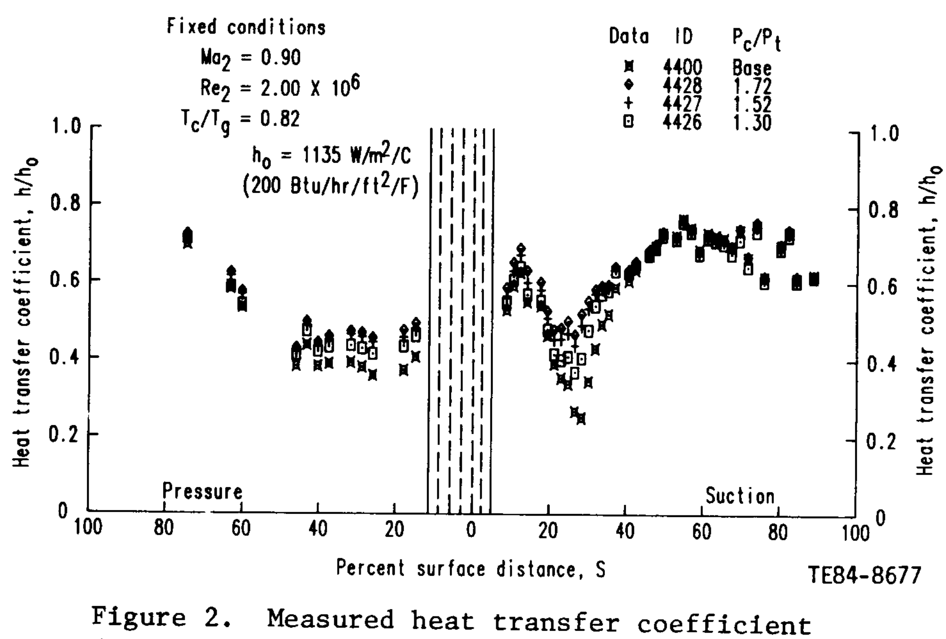

Figure 2. Measured heat transfer coefficient distribution for varying blowing strength.

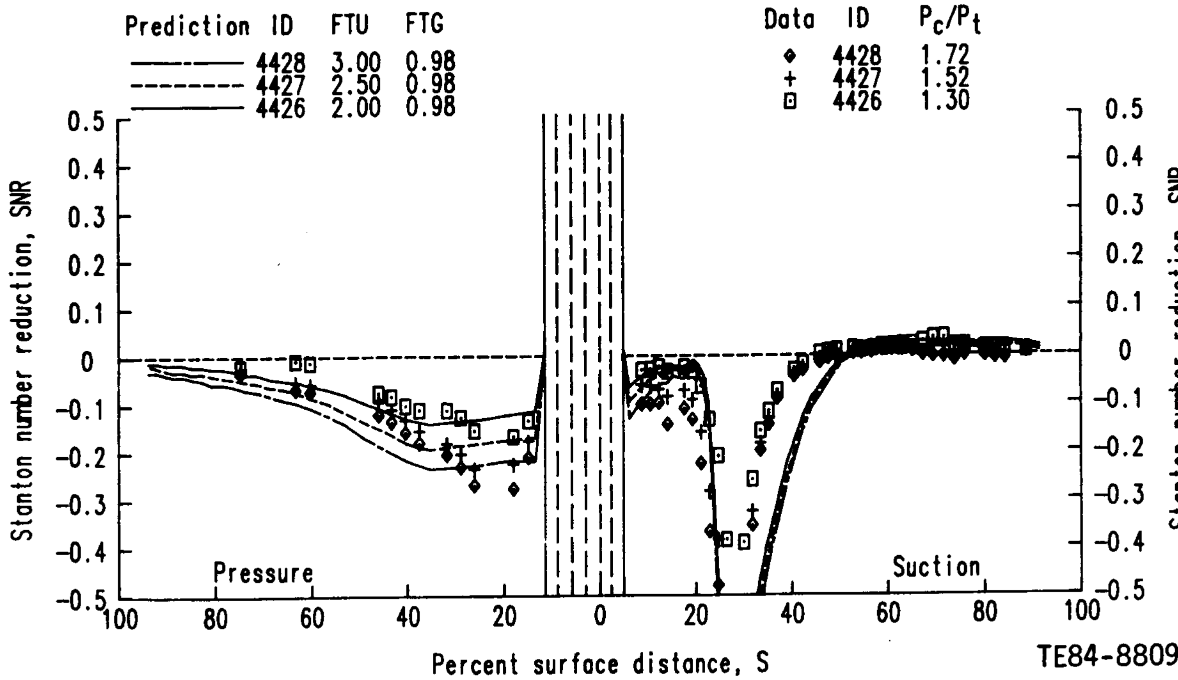

Figure 3. Measured and predicted SNR distributions for varying blowing strengths above design range.

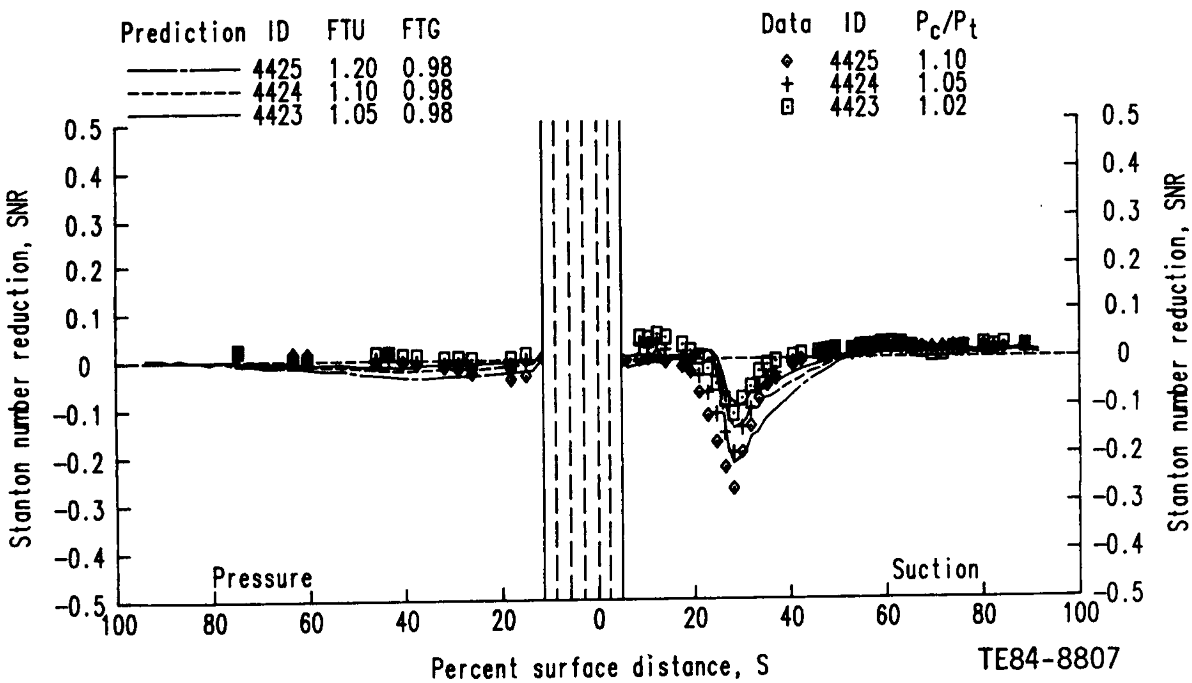

Figure 4. Measured and predicted SNR distributions for varying blowing strengths in design range.

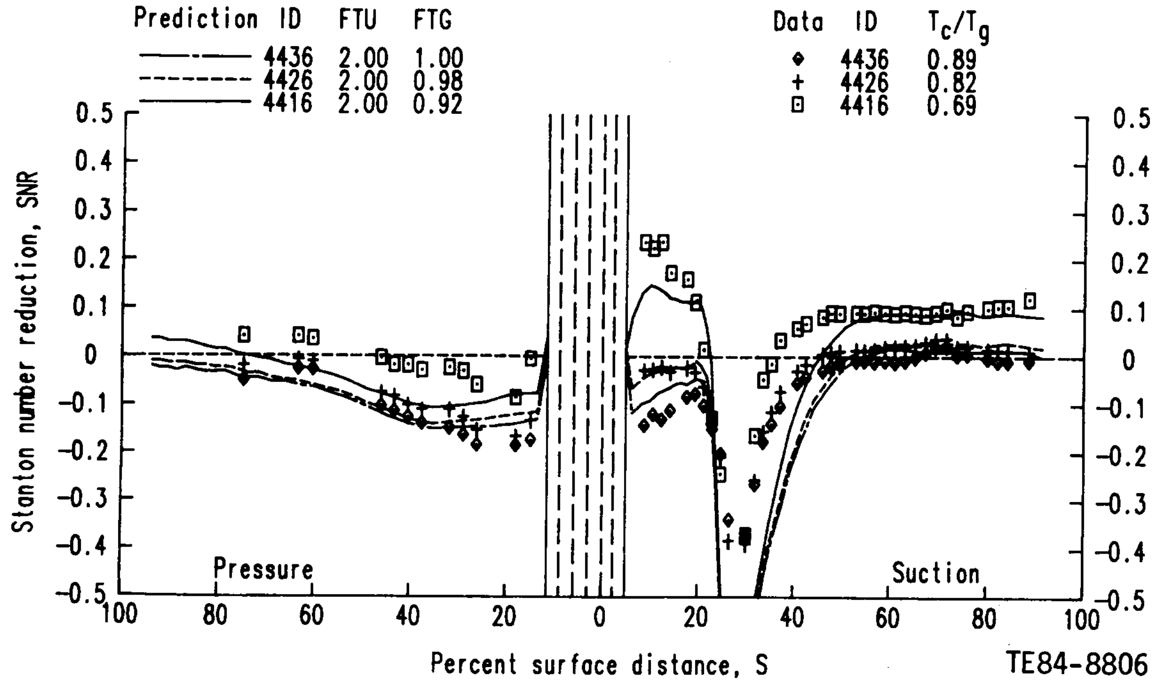

Figure 5. Measured and predicted SNR distributions for varying coolant-to-gas temperature ratio.

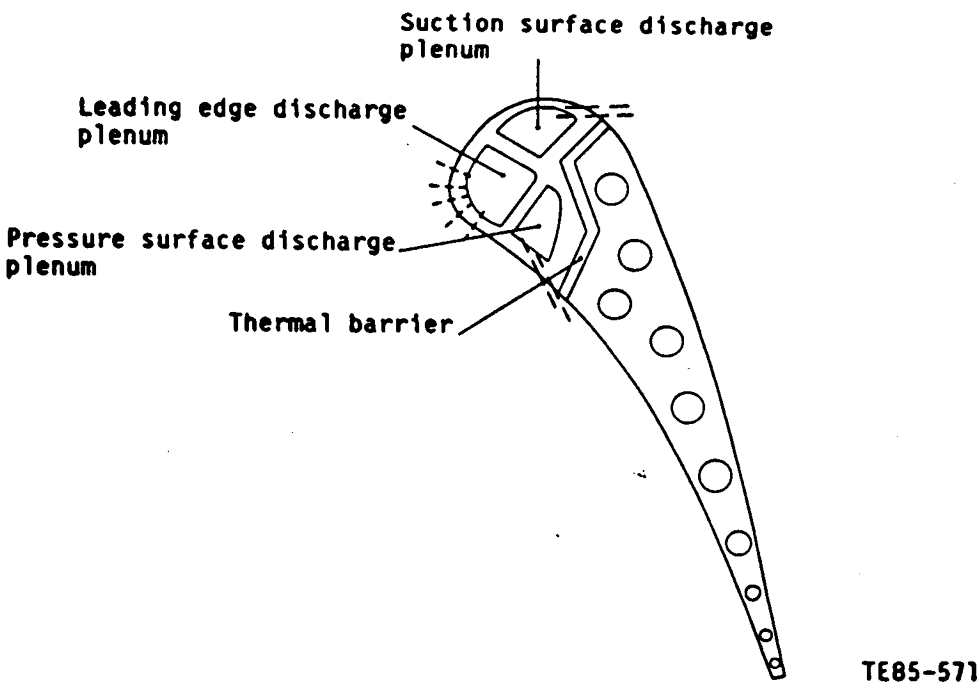

Figure 6. Schematic of airfoil with leading edge showerhead and downstream film cooling for cascade tests.

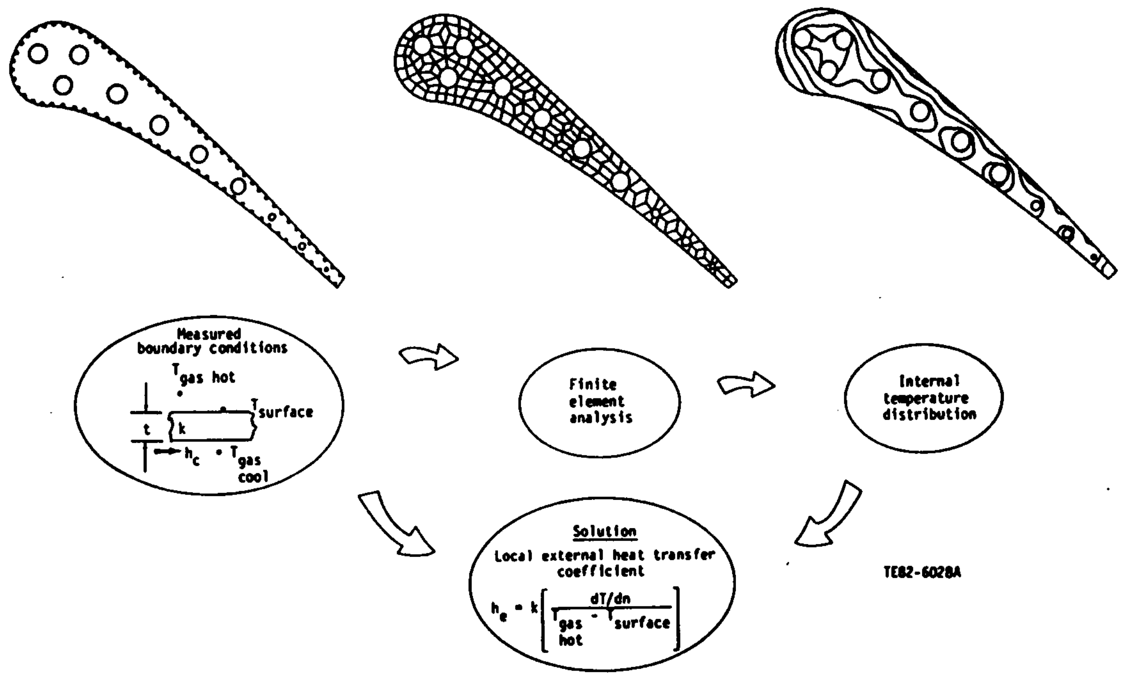

Figure 7. Heat transfer measurement technique.