TURBINE AIRFOIL EXTERNAL HEAT TRANSFER MEASUREMENT
IN A HOT-CASCADE

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ABSTRACT

Detailed knowledge of local external heat transfer around a turbine airfoil is critical for an accurate prediction of metal temperature and component life. This paper discusses the results of measurements of the local gas side heat transfer coefficient distribution which were performed in an annular gas turbine vane hot-cascade test facility and provides comparisons with analytical predictions. The steady state tests were performed at simulated engine operating conditions. The tests were conducted on an internally cooled airfoil with the intent to provide close to uniform coolant temperature at the mid-span of the airfoil and also to obtain a high heat flux through the wall, minimizing uncertainty. Internal coolant side heat transfer coefficients were measured through calibration tests outside the cascade by a wire heater. A calibrated infrared pyrometer was used to provide detailed airfoil surface local temperatures and the corresponding external heat transfer coefficients. The airfoil was instrumented with a number of thermocouples for both calibration and temperature measurement. A 12% freestream turbulence was generated by a turbulence grid upstream of the test cascade. The static pressure measured over the test vane agreed well with inviscid predictions. Heat transfer coefficients were measured and compared to a TEXSTAN boundary layer code prediction. The influence of Reynolds number and Mach number on the airfoil external heat transfer coefficient were also studied and is presented in the paper.

NOMENCLATURE

Symbols

T  temperature
Tu  turbulence intensity
μ  dynamic viscosity
C  true chord
V  velocity

Subscripts

w  wall
o  total
g  free-stream condition
c  coolant
gen  generated
mix  mixing length turbulence model
2-eqn  two equation turbulence model
1  cascade inlet
2  cascade outlet
loss  heat loss

INTRODUCTION

The design of a cooling system for turbine nozzle vane represents one of the challenging engineering tasks in the design of modern industrial gas turbines. As part of the aerodynamic and thermal analyses, an accurate prediction of the mainstream heat transfer coefficient is required to establish cooling flow requirements. Experimental determination of heat transfer coefficients is complicated by the need to simulate actual engine conditions and has proved to be difficult. Due to the complexity of the flow field, numerical prediction alone usually cannot satisfactorily predict the external heat transfer coefficient. A number of experimental turbine vane heat transfer studies has been reported over the last 30 years. Because of the difficulties in performing tests on full size engine airfoils, most of the experimental work has been performed in wind tunnels on upscale models, which only simulate some of the key engine operating parameters (Mach number (Ma), Reynolds number (Re), wall-to-gas temperature ratio (TWTg), and turbulence intensity (Tu)).

Presented at the International Gas Turbine & Aeroengine Congress & Exhibition
Orlando, Florida — June 2–June 5, 1997
Some of the cascade work has been performed in linear cascade, where the flow is three dimensional but the geometry is two dimensional. Among the highly contributory works performed in a linear cascade (Ames, 1994, 1995; Hoff et al., 1995; Sato and Takeishi, 1987), there is the work by Arts et al. (1990). The turbine vane external heat transfer data obtained from the von Karman Institute Isentropic Light Piston Compression Tube facility, allowing a correct simulation of engine operating parameters, provides valuable information not only to turbine designers, but to numerical simulators also. Most of the works emphasized the effects of freestream turbulence and the Reynolds number on turbine airfoil external heat transfer coefficient (Ames, 1994, 9115; Blair et al., 1998a, 1988b).

It is very desirable to get information from annular cascade experiments at representative engine conditions (Dunn et al., 1994; Martinez-Botas et al., 1993; Blair et al., 1998a, 1988b). The work by Harasgama and Wedlake (1991) is an excellent example. The detailed aerodynamic and heat transfer measurements were reported on both airfoil surfaces and endwalls from a full ring annular cascade at the correct engine operating conditions.

It should be noted that the majority of the data have been acquired at or near room temperature. Very few "high temperature" or "engine environment" data have been reported. One such work is reported by Hyton et al. (1983) in which turbine vane external heat transfer coefficients were obtained by using thermocouples and a finite element analysis from a linear cascade at design and off-design engine operating conditions.

With the development of computational methods and computer technology, the numerical prediction of the airfoil external heat transfer is becoming more accurate. Most of the experimental work were reported with some kind of numerical comparison, for example, a boundary layer code prediction. The different versions of TEXSTAN (STANS) have been refined using the experimental data (Crawford, 1986; Gaugler, 1981; Harasgama et al., 1993). To expand the experimental data base on airfoil external heat transfer coefficients in a high temperature environment, representative for engine operating conditions (simulation of Ma, Re, Tw/Tg and Tu) and provide validation for numerical simulation, a study was performed by the authors of this paper to obtain the airfoil external heat transfer coefficient data in an annular hot-cascade using a full size air-cooled nozzle.

The main objectives of this study were:
1. To verify the aerodynamic and heat transfer simulation in the hot-cascade environment.
2. To obtain external heat transfer coefficients and compare them with TEXSTAN predictions.
3. To study the effect of inlet and exit Re and Mach number on airfoil heat transfer.

TEST FACILITY AND INSTRUMENTATION

The tests were performed in the hot-cascade which was featured in earlier papers (Glezer et al., 1994; Zhang and Glezer, 1995; Moon and Glezer, 1996). The realistic production configuration nozzle was selected for the heat transfer measurement. The nozzle test section was modified for this study with the middle airfoil being replaced by one with a static pressure tap or an air-cooled and thermocouple instrumented nozzle airfoil.

A schematic of the test rig is shown in Figure 1. A configuration featuring a partial annular cascade with full-scale engine hardware was selected for the experimental rig to provide geometric similarity and the relation between convection and conduction through the wall of the test airfoil.

A natural gas fueled can combustor (actual engine employs a larger size annular combustor) supports the automated hot cascade rig operation for a wide range of pressures, flows and temperatures. Compressed air and natural gas supplied at about room temperature from the central factory source allow air flows of up to 1.5 kg/s and pressures of up to 2000 kPa.

Air flow is measured by a turbine type flow meter and is established by a remote-controlled regulator. A radially inserted electric igniter ensures the combustor light off and supports ignition even at relatively low fuel/air ratios. A flame indicator controls the automatic fuel shut-off system.

A circular combustor outlet directs the mainstream flow through a smooth transition duct to the four-vane annular hot cascade section. Four full scale vanes provide protection at the mid passage test surfaces from radiation error and aerodynamic disturbances imposed by the wall boundary layer.

Combustor exit temperature is automatically controlled at a predefined level within 1°K (average at three locations). An additional temperature traversing device is used periodically to verify the prescribed temperature profile. Combustor pressure is controlled by a throttle valve positioned downstream of the hot cascade. Water injection upstream of the turbine vane provides for safe operation at relatively high hot cascade temperatures. A maximum operating temperature of 1153°K is allowed for pressures up to 820 kPa to ensure the structural integrity of the rig. Somewhat limited "firing" temperatures are required for the uncooled test section which does not have a traditional outer housing which usually complicates assembly and requires bellows compensators for internal piping and instrumentation.

Figure 2 shows the air cooled nozzle airfoil used for heat transfer coefficient measurement along with the airfoil internal cooling passage. A full-size engine nozzle was used in this study and the standard nozzle configuration was modified. The chord length of the nozzle is 50.3 mm and the inlet span is 55.9 mm. The pin fins and ribs in the internal surface of the standard nozzle design were removed to form a smooth surface inside the airfoil. The film cooling holes and trailing edge slots were blocked to change the cooling flow from streamwise to spanwise. The insert was also modified to remove the impingement holes and to open the top endwall.
providing that the middle pocket of the insert could be used as a back flow passage. The coolant was introduced into a lower plenum and then distributed along the gap between the insert and the inside surface of the airfoil. After cooling the inside of the airfoil, the coolant was collected in the upper plenum (turbine inner shroud) where it flowed back through the middle of the insert and finally was discharged through an exit tube. Radial flow between the vane and insert (versus standard streamwise cooled vane) in the modified nozzle was chosen to provide uniform coolant temperature and internal heat transfer coefficient distributions at the mid-span of the airfoil. Ten K type thermocouples were installed at the mid-span of the airfoil to measure the wall temperature both in the heat transfer measurements and in the calibration tests. Figure 3 shows the locations of the thermocouples which form a row located at the mid-span along both the suction and pressure surfaces. Two thermocouples were located in the lower plenum and another two in the upper plenum to measure the coolant inlet and exit temperatures. The wide range infrared pyrometer (Fig. 4) developed at Solar (Moon et al., 1995) was used to map surface temperature and to obtain the heat transfer coefficient distribution along the airfoil surfaces (Fig. 4). Figure 5 shows the hot-cascade test section with the pyrometer being installed and Figure 6 shows the three pyrometer positions and viewing angles providing optical access to the full airfoil surfaces inside the hot-cascade.

Another nozzle airfoil was instrumented with 24 pressure taps on the airfoil surfaces (Fig. 7) to measure the mid-span surface static pressure distribution and to calculate the isentropic Mach number distribution. The 0.762 mm O.D. diameter tubing was installed inside the metal wall and 0.254 mm holes were drilled along the mid-span of the airfoil. Additional pressure taps were installed at the inlet and exit of the cascade endwalls for static pressure measurement and for controlling the inlet and back pressures. In addition to the existing instrumentation, a modified kiel probe and a heat flux probe were used for turbulence and flow measurements. A turbulence grid was installed in the hot-cascade to generate a turbulence intensity in the heat transfer tests of about 12% (Zhang and Giezer, 1995).

TEST CONDITIONS

Table 1 shows the test conditions at which both the aerodynamic and heat transfer tests were performed. The aerodynamic tests were performed at both high and low freestream turbulence conditions while the heat transfer tests were performed only at high freestream turbulence conditions. Three freestream temperatures were chosen to examine the possible effect of mainstream temperature and pressure on the airfoil external surface Nusselt number when similarity parameters of Reynolds and Mach number are satisfied. Three inlet (and exit) Reynolds numbers
<table>
<thead>
<tr>
<th>Test No.</th>
<th>Range of Freestream Turbulence (%)</th>
<th>Re&lt;sub&gt;t&lt;/sub&gt;</th>
<th>M&lt;sub&gt;a&lt;/sub&gt;</th>
<th>Inlet Total Pressure (mpa)</th>
<th>Exit Static Pressure (mpa)</th>
<th>Mass Flow (kg/s)</th>
<th>Temperature (°K)</th>
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<tbody>
<tr>
<td>1</td>
<td>1.0, 12.0</td>
<td>2.82 x 10&lt;sup&gt;5&lt;/sup&gt;</td>
<td>0.72</td>
<td>0.21</td>
<td>0.152</td>
<td>0.613</td>
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<tr>
<td>2</td>
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<td>2.82 x 10&lt;sup&gt;5&lt;/sup&gt;</td>
<td>0.72</td>
<td>0.62</td>
<td>0.449</td>
<td>1.171</td>
<td>699.7</td>
</tr>
<tr>
<td>3</td>
<td>1.0, 12.0</td>
<td>2.82 x 10&lt;sup&gt;5&lt;/sup&gt;</td>
<td>0.72</td>
<td>0.682</td>
<td>0.495</td>
<td>1.237</td>
<td>755.2</td>
</tr>
<tr>
<td>4</td>
<td>1.0, 12.0</td>
<td>2.82 x 10&lt;sup&gt;5&lt;/sup&gt;</td>
<td>0.72</td>
<td>0.745</td>
<td>0.542</td>
<td>1.301</td>
<td>810.8</td>
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<tr>
<td>5</td>
<td>1.0, 12.0</td>
<td>2.82 x 10&lt;sup&gt;5&lt;/sup&gt;</td>
<td>0.792</td>
<td>0.62</td>
<td>0.39</td>
<td>1.186</td>
<td>699.7</td>
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<tr>
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<td>2.82 x 10&lt;sup&gt;5&lt;/sup&gt;</td>
<td>0.648</td>
<td>0.699</td>
<td>0.538</td>
<td>1.191</td>
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<tr>
<td>7</td>
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<td>3.1 x 10&lt;sup&gt;5&lt;/sup&gt;</td>
<td>0.72</td>
<td>0.699</td>
<td>0.507</td>
<td>1.316</td>
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<tr>
<td>8</td>
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<td>2.55 x 10&lt;sup&gt;5&lt;/sup&gt;</td>
<td>0.72</td>
<td>0.572</td>
<td>0.415</td>
<td>1.077</td>
<td>699.7</td>
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</table>

* Re<sub>t</sub> - Reynolds number based on inlet average velocity and true chord.
* M<sub>a</sub> - Average Mach number at the cascade exit.

**FIG. 5 - HOT CASCADE TEST SECTION WITH TRAVERSING PYROMETER**

**FIG. 6 - LOCATIONS AND VIEWING ANGLES OF THE PYROMETER**

were selected to study the effect of Reynolds number on turbine aerodynamics and heat transfer, while three exit Mach numbers were chosen to study the effect of exit Mach number.

**CALIBRATION TEST SETUP AND RESULTS**

To reduce the uncertainty of the test results, the coolant side heat transfer coefficients were obtained through calibration tests on the air-cooled nozzle outside the hot-cascade before heat transfer measurements were made. Figure 8 shows the setup of the calibration tests. During the calibration tests, a uniform heat flux was generated by an electrical wire heater outside the airfoil surface. The heater was assembled of two pieces of MWS-75 #24 heating wires which were wound parallel around the air-cooled nozzle. Fiber glass and rubber sheet were used outside the airfoil as an insulator to reduce the heat losses from the outside surface of the wire heater. Two thermocouples were installed to measure the insulator external surface temperature required for heat loss calculations. The cooling air mass flow was measured by a turbo-meter and was varied from 0.00455 kg/s to 0.0864 kg/s. The electric load of the heater was measured by two
These calibrated coolant side heat transfer coefficients, obtained from curved conducting surfaces, account for the non-uniform distribution of the coolant and partially account for the effects of curvature and spanwise/streamwise conduction which are two main sources of uncertainty. The air-cooled calibration results at selected locations are shown in Figure 9. In these figures, TC 5 is a thermocouple located at the stagnation point on the nozzle airfoil leading edge which represents the highest curvature location. TC 6 is a thermocouple located in a flat portion of the airfoil surface and TC 7 is a thermocouple located in the curved midsection of the suction side where the radius of the surface curvature is intermediate between that of the stagnation point and the flat portion closer to the trailing edge. The low coolant side internal heat transfer coefficient at the stagnation point (TC 5) is mainly a result of locally high curvature (2D effect). The heat transfer coefficients calculated from the correlation for turbulent internal flow between two parallel plates are also plotted in Figure 9 as a comparison. The poor predictions by the turbulent internal flow correlation was partially due to the assumed simplified geometry of the cooling channel.

\[
h_{e} = \frac{1}{\frac{T_{e} - T_{w}}{\frac{q_{gen} - q_{loss}}{\Delta T}}} = \frac{1}{\frac{q_{gen} - q_{loss}}{2K}}
\]  

AERODYNAMIC MEASUREMENTS
For an accurate heat transfer measurement, it is essential that a correct simulation of engine operating conditions is achieved in the hot-cascade environment. The isentropic Mach number was calculated based on the static pressure measurements from the pressure tap instrumented nozzle. Tests parameters, particularly mass flow rate and back pressure, were used to establish a similarity in the heat transfer test conditions. The tests were performed at four different inlet gas total temperatures (300 K, 699.7 K, 755.2 K and 810.8 K), three inlet Reynolds numbers (255,000, 282,000 and 310,000) and three exit Mach numbers (0.648, 0.720 and 0.792). Among the key parameters, special attention was paid to the mass flow rate and the static (back) pressure at the exit of the cascade. The results show that the variation of the mass flow rate does not significantly affect the Mach number distribution which is in a linear relation with Reynolds. Also, as expected, variation of back pressure results in a significant change in Mach number distribution.

Figure 10 shows the isentropic Mach number distribution calculated from the pressure tap static pressure measurements. In the same figure, the
Mach number predictions using a two-dimensional inviscid code are also plotted for comparison. The Mach number distribution curves were obtained from the tests performed for the engine design inlet and exit Mach numbers at three levels of inlet Reynolds numbers (255,000, 282,000 and 310,000) and at four inlet total temperatures (300°K, 699.7°K, 755.2°K and 810.8°K). The results for different Reynolds number collapse into the same lines for both the pressure and suction surfaces (the curves of inlet total temperature 300°K, 755.2°K and 810.8°K were removed for clarity). This indicates that the Mach number distribution is not affected by the Reynolds number providing that the inlet and exit Mach numbers remain unchanged.

The calculated Mach number distribution on the pressure surface match well with predictions of an inviscid panel code. On the suction surface, the Mach number distributions are not very smooth but follow the trend of the predictions. A maximum Mach number was located downstream of the predicted maximum Mach number location. This is considered to be the result of the three dimensional effect caused by the contoured endwall shroud. Mach number distributions at higher and lower exit Mach numbers (M2 = 0.648 and M2 = 0.792) were also obtained and plotted in the same figure. On the pressure surface, an increase (or decrease) in exit Mach number corresponds to a gradual increase (or decrease) in Mach number from the leading edge towards the trailing edge. On the suction surface, an increase (or decrease) in exit Mach number results in a significant increase (or decrease) in Mach number along almost the entire suction surface downstream from the leading edge. A Mach number of 1.0 was measured at the throat on the suction surface when the exit Mach number was 0.792.

HEAT TRANSFER MEASUREMENT

The heat transfer tests were conducted using air as the coolant for test conditions 2 through 8, as shown in Table 1. At each of these test conditions, the tests were repeated three times with the pyrometer positioned at three different locations (Fig. 4). Special care was taken to repeat the identical test condition each time. Temperature readings from the thermocouples located on the nozzle surface were used as reference temperatures to correct the pyrometer readings for emissivity. Test conditions (inlet pressure, back pressure and flow rate, etc.) were also carefully controlled to repeat the test conditions as in the aerodynamic tests. The freestream turbulence intensity in all the heat transfer tests was preset at 12% using an indirect measurement method with a heat flux probe (Zhang and Glezer, 1995). In all the air-cooled tests, the cooling air flow rate was 0.0082 kg/s which is within the range of the calibration tests. The internal coolant heat transfer coefficient was based on the calibration and modified for the temperature difference between calibration and hot-cascade tests. Two- and three-dimensional finite element analyses (using ANSYS) were performed before the test and during the data reduction. The uncertainty caused by spanwise and streamwise conduction is within 4% after correction was made based on the finite element analyses.

Figure 11 shows the measured vane external Nu number distributions obtained at the simulated design inlet Reynolds number and the design exit Mach number. The freestream total temperatures were set at 699.7°K, 755.2°K and 810.8°K. The inlet and exit pressures and the cascade mass flow rate were varied based on the corresponding simulation requirements. The tests were conducted to observe any possible effect of freestream temperature and pressure on the airfoil external Nu number when the Reynolds number, Mach number and the turbulence condition are correctly simulated. Two predictions using the boundary layer code TEXSTAN employing the two equation Lam/Bremhorst model and the zero equation mixing length model are also plotted in Figure 11 as a comparison. The turbulence intensity input to TEXSTAN was 12%. On the pressure surface, the Nu number obtained from the three inlet total temperature conditions form a single line, with considerable scatter, and lay below the two equation model prediction and above the zero equation model prediction. In the vicinity of the trailing edge on the pressure surface, the measured Nu numbers were lower than both predictions. On the suction surface, the difference between the predicted Nu number obtained with the two turbulence models was smaller than that on the pressure surface. The measured Nu numbers also form a single line and is closer to the mixing length prediction. The data scatter, particularly downstream of the throat, may be due to the secondary flow effect (the aspect ratio of the stage-1 nozzle is about 0.63 at trailing edge). The measured Nu number distributions show a transition from laminar to turbulent boundary layer at the same location as that predicted by TEXSTAN. The effect of inlet total temperature on the Nu number distributions for three inlet temperatures was found to be insignificant for both the pressure and suction surfaces. This indicates that the Nu number distribution does not change with the inlet total temperature.

![FIG. 10 - ISENTROPIC MACH NUMBER DISTRIBUTIONS AROUND THE NOZZLE AIRFOIL SURFACE](image1)

**FIG. 10 - ISENTROPIC MACH NUMBER DISTRIBUTIONS AROUND THE NOZZLE AIRFOIL SURFACE**

![FIG. 11 - Nu NUMBER DISTRIBUTIONS - EFFECT OF THE INLET TOTAL TEMPERATURE](image2)

**FIG. 11 - Nu NUMBER DISTRIBUTIONS - EFFECT OF THE INLET TOTAL TEMPERATURE**
total temperature when a good simulation of test parameters is achieved. This also suggests that the component tests (Moon and Glezer, 1996) performed in the hot-cascade at a reduced inlet temperature are valid for the engine operating conditions.

Figure 12 shows the effect of Reynolds number on the Nu number distribution along the airfoil surface. The freestream total temperature of the tests was set at 699.7°C while the inlet and exit Mach numbers were at the design values. The inlet Reynolds numbers of the cascade were 255,000, 282,000, and 310,000 and were obtained by varying the inlet (and exit) pressure and the mass flow rate. The exit Mach numbers were maintained as close as possible to the design exit Mach number (0.72).

Special consideration was given to repeating the test conditions of the airfoil surface static pressure measurements. Predictions using the TEXSTAN Lam/Bremhorst two equation turbulence model for the three inlet Re numbers are plotted in the figure. The inlet pressure input to TEXSTAN was varied such that the appropriate density was used to generate the three inlet Re numbers. Compared with the predicted curves, the measured Nu number distributions were lower, except that on the pressure surface at a Re number of 310,000. On the pressure surface, the measured Nu number increases consistently with the inlet Re number. The Nu number distribution curves were smooth and followed the trend predicted by TEXSTAN. The measured Nu values were about 70% to 80% of the predicted values. On the suction surface, the Nu number distribution also shows a strong dependency on the Re number but are not as smooth as that on the pressure surface. At a Re number of 282,000, the curve does not lay smoothly between that of a Re number 255,000 and 310,000, which again may be the result of the secondary flow. Close to the leading edge, where the boundary layer is considered laminar due to high acceleration, the measured Nu numbers show a significant increase with Re number. Downstream of that region, the point of transition from laminar to turbulent boundary layer moves forward toward the leading edge with an increasing Re number as expected. Downstream from the transition point to the turbulent region, the Nu number increases considerably with the Re number showing a stronger dependency then the TEXSTAN predictions.

0.792 and were established by changing the inlet and exit static pressures while the mass flow rate was held constant. On the pressure surface near the trailing edge, considerable data scatter can be observed. The effect of exit Mach number on the airfoil external Nu number is within the uncertainty of the data. This less significant effect on the Nu number can be explained by the Mach number distribution in Figure 10 where the Mach number on the pressure surface does not significantly change with exit Mach number until near the trailing edge. On the suction surface, in the laminar region close to the leading edge, there is no well defined relationship between the exit Mach number and the Nu number. In the transition region, the increase in exit Mach number results in a reduced Nu number which indicates a delayed transition from laminar to turbulent boundary layer due to the higher acceleration. At the turbulent region toward the trailing edge, it is clearly seen that an increase in exit Mach number results in a significant reduction in measured Nu number, which agrees with similar studies in the open literature (Arts et al., 1994, Hylton et al., 1983).

The uncertainty analysis was carried out based on Kline and McClintock (1953). The uncertainties due to $T_{gas}$, $T_{wall}$, $T_c$ and internal cooling air side heat transfer coefficient are 6, 3, 2 and 1.5 percent, respectively. The uncertainty of the infrared pyrometer temperature measurement was 2%. The maximum uncertainty in Nu number is less than 15 percent for all the Reynolds number studied.

**CONCLUSIONS**

Distribution of external heat transfer coefficient along the turbine nozzle vane surface was obtained in the hot-cascade test facility using an instrumented air-cooled nozzle. Several conclusions can be made from this study:

- Application of hot cascade facility for turbine airfoil external heat transfer measurement presents an attractive alternative to the engine testing provided simulation of critical heat transfer parameters at lower operating temperatures.
- The experimental approach based on application of the radial passages for the cooling provided a relatively small variation of the coolant temperature along the mid-span vane section leading to a reduced uncertainty in the local heat transfer measurement.
The Nusselt number increases with inlet Reynolds number on both the pressure and suction surface while it decreases with exit Mach number on the suction surface particularly near the trailing edge.

Measured Nusselt numbers are consistently below the TEXSTAN two-equation model predictions. This indicates a need to improve current analytical methods.

REFERENCES


Personal communication with M.E. Crawford, University of Texas at Austin.
