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SIMILARITY TESTS OF TURBINE VANES - EFFECTS OF CERAMIC THERMAL BARRIER COATINGS

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NOMENCLATURE

A  heat transfer area
C1 constant
h  heat transfer coefficient
k  thermal conductivity
Nu Nusselt number
P  pressure
Pr Prandtl number
q  heat flux
R  gas constant
Re Reynolds number
T  temperature
w  mass flow rate
x  surface distance
\( f \) function of specific heat ratios
Y  specific heat ratio
\( \mu \) viscosity

\( \tau \) wall or coating thickness; characteristic dimension
\( \varphi \) \( (T_g - T_{\omega 0})/(T_g - T_{c1}) \)

SUBSCRIPTS:
c  coolant
ci coolant inlet
e  engine
g  gas inlet
g_e effective gas
t  test
u  uncoated condition
w  wall
wi coolant side of metal wall
wo gas side of metal wall
zi metal-ceramic interface
zo gas side of ceramic
z  ceramic

SUPERSCRIPTS:
(e) engine conditions
(t) test conditions
- average
INTRODUCTION

Ceramic thermal barrier coatings are being considered for the hot section components of gas turbine engines to supplement the thermal protection provided by various air-cooling schemes. The thermal performance of these coated components is often evaluated at reduced gas/coolant temperatures and pressures to avoid the complexity and expense of testing at actual engine gas/coolant conditions. However, extrapolation (scaling) of these data results to engine conditions can lead to erroneous conclusions unless the thermal effects of variations in ceramic, metal, gas, and coolant thermal conductivities are considered.

Several techniques exist which can be used to establish the relationship between ceramic coated and uncoated turbine vanes tested at reduced gas/coolant temperatures and pressures (Refs. 1 to 4) necessary for testing turbine components at similarity conditions. The approach used in each of these references is somewhat different but the results and conclusions are basically the same. That is, kinetic, dynamic, and thermal similarity of the gas and coolant can be maintained between test conditions and engine conditions by maintaining an equality of various dimensionless parameters. References 2 and 3 also discuss the need to maintain similarity of the material thermal conductivity to achieve a similar thermal performance of the hardware at both test and engine conditions. Reference 4, however, concludes that the thermal conductivity effect is not significant for the conditions and materials considered. Reference 2, however, shows that gas, coolant, and metal thermal conductivity variations between test and engine conditions result in errors when data corrections are made between ceramic coated and uncoated turbine vanes and when these data are extrapolated to engine conditions. A correction technique developed in reference 5 resulted in correction factors which were of opposite sign between the uncoated vane data and the coated vane data. The analysis of reference 5 is used herein to predict the data corrections required for ceramic coated and uncoated turbine vanes tested at reduced gas/coolant temperatures and pressures. These corrected data are then compared to actual data taken at engine conditions. The results are also presented as an error between engine and test conditions for turbine vane cooling effectiveness parameters from 0.3 to 0.6.

APPARATUS AND EXPERIMENTAL PROCEDURE

Cascade Facility

The cascade facility was designed for continuous operation at gas temperatures and pressures up to 1600 K and 100 N/cm² (absolute). A schematic of the cascade facility is shown in Fig. 1(a). The cascade facility consisted of five major components shown in Fig. 1(b): an inlet section, a high temperature combustor section, a circular-annular transition section, the test section, and an exit section. The transition, test, and exit sections were water-cooled to achieve structural durability during high-temperature operation. A more detailed description is contained in reference 6.

The high temperature combustor section was removed and replaced by a spoil piece for low-temperature tests in the facility. Hot combustion air was then supplied to the test section by the low-temperature combustor shown in Fig. 1(a). The low-temperature combustor was capable of supplying combustion air to the test section at temperatures up to 900 K.

The test section was a 25° annular sector of a turbine row and contained four vanes and five flow channels. A plan view of the test section, showing the test vane (vane number 2) and selected instrumentation, is presented in Fig. 2. The slave vanes complete the flow channels for the test vane and serve as radiation shields between the test vane and the water-cooled walls. The test section walls were coated with yttria stabilized zirconia to increase the surface temperature and minimize net thermal radiation from the test vane.

Vane

The turbine vane used in this investigation was a J-75 size airfoil with impingement cooling in the forward 2/3 of the airfoil and pin fin/film cooling in the aft 1/3 of the airfoil. A cross-sectional schematic of the airfoil and cooling configuration is shown in Fig. 3. The vane span was 9.78 cm and the midspan chord length was 6.28 cm. The wall thickness in the impingement cooled region was 0.152 cm. The vane airfoil shell material was MAR M-302.

The impingement insert had a staggered array of holes which were 0.051 cm in diameter. The spacing varied, depending on location, between 6.3 and 9 hole diameters span-wise and between 2.4 and 9 hole diameters chord-wise. The closely spaced holes were in the leading edge region (6.5 by 2.4) while the midchord region had larger spacings (9 by 9 on the pressure side and 8.5 by 8.5 on the suction side). The impingement holes-to-heat-transfer-surface-spacing ratio was approximately 1.5 hole diameters in the midchord region and approximately 2 hole diameters in the leading edge region. The impingement insert material was L-605.

There were 7 chord-wise rows of round pin fins in the split trailing edge. The three upstream rows had pin diameters of approximately 0.102 cm with a span-wise spacing of 0.406 cm and a chord-wise spacing of 0.333 cm. The last four rows had pin diameters of 0.076 cm with a span-wise spacing of 0.305 cm and a chord-wise spacing of 0.264 cm. The width of the split trailing edge channel at the point of discharge was 0.089 cm.

A single row of film cooling holes was located between pin fin rows 3 and 4 on the vane pressure surface and ejected air at an angle of 30° to the vane surface in the span-wise direction. The purpose of these holes was to provide a sufficient flow area to accommodate the design coolant flow requirements.

Thermal Barrier Coating

The procedure used for depositing ceramic coating (Ref. 7) onto the vane metal substrate was to prepare the substrate surface by grit-blasting, plasma-spray on a bond coat of HVOF, and plasma-spray on the ceramic coating of yttria stabilized zirconia. The measured surface roughness of the applied ceramic coating was 8 to 10 micrometers, rms. However, the coating surface was polished with silicon carbide paper to a surface finish of about 3 micrometers, rms.

The bond and ceramic coatings were built up to the desired thickness by a succession of spray appli-
locations in the span-wise and chord-wise directions on the airfoil. The coatings were first applied to the vane leading edge, then to the trailing edge, and finally to the suction and pressure surfaces. The final total coating thickness was determined, after the polishing operation, by comparing 10X profiles of the airfoil before and after coating at each of the thermocouple locations. The ceramic coating thickness was then assumed to be the total thickness less the approximately 0.010 cm thick bond coat.

The distribution of the coating thickness is given in Table I. The coating was tapered to negligible thickness at thermocouple location 12. This was necessary because of the film cooling holes aft of this location. The coating techniques were not sufficient to develop the time to permit cooling in the hole region without hole blockage.

Test Procedure

Thermal performance tests were made at the gas and coolant conditions given in Table II. The desired combustion gas temperature, pressure, and critical velocity ratio (0.85) were established and then the cooling-air flowrate was varied in a step-wise fashion from test point to test point. Steady state data were recorded at each of the cooling air and gas condition set-points. Ambient temperature cooling air was utilized for all test and engine conditions investigated. The data taken in the cascade at high gas temperatures and pressures are defined as engine data while the data taken at reduced gas temperatures and pressures are defined as test data.

INSTRUMENTATION

A radially traversing, sonic aspirated, type R (Platinum vs Platinum - 13% Rhodium) total temperature probe and a radially traversing total pressure probe provided the inlet gas conditions to the test vane (Fig. 2). The temperature distribution was measured upstream of channel 3 and the pressure distribution was measured upstream of channel 4. The inlet static pressure was measured only at the inner radius (hub) and was assumed to be constant across the gas stream. Static pressures were also measured at the exit midchannel position of channels 2, 3, and 4 at both the inner (hub) and outer (tip) radius platforms. These pressures were used to establish the midspan inlet and exit critical velocity ratios.

The midspan of the test vane airfoil was instrumented with an array of 12 Chromel-Alumel thermocouples. Figure 3 shows the relative location of these thermocouples with respect to the important features of the vane. Chord-wise thermocouple locations are given in Table I. The thermocouples were installed in slots EDM'd in the exterior surface of the airfoil. The junction end of each thermocouple assembly was pierced into the slot which effectively located the measuring station a specified distance from the bottom of the slot. The remainder of the slot over the thermocouple junction was filled by spot-welding a nickel-chromium material in the void and fairing the resultant construction to the original airfoil profile.

The construction of the thermocouple assemblies consisted of Chromel-Alumel thermocouple elements with magnesium oxide insulation in an Inconel-600 sheath. These assemblies were drawn to two sheath sizes, 0.05 and 0.025 cm outside diameter, with a closed-end grounded junction formed at one end. The three thermocouples near the leading edge were 0.025 cm diameter while the remaining thermocouples were of 0.05 cm diameter. A detailed description of the procedures utilized for thermocouple construction is given in reference (8). The slots for the 0.05 cm diameter thermocouples were 0.06 cm square while the slots for the 0.025 cm diameter thermocouples were 0.03 cm square. The measuring stations were nominally located 0.047 and 0.022 cm, respectively, below the gas side surface of the airfoil.

ANALYSIS METHOD

Similarity

References (2) and (3) show that Reynolds, Prandtl, and Mach numbers are sufficient to ensure dynamic, kinematic, and thermal similarity of the gas and coolant. Geometric similarity is maintained by using prototype engine component hardware. The following equations from references (2) and (3) establish the relationship between the engine and test conditions for similarity of both the hot gas and the coolant. These equations are based on an equality of momentum thickness Reynolds number and Mach number between the gas conditions of the engine and the similarity tests. In addition, an equality of the coolant Reynolds number is assumed.

\[ \frac{\rho}{\rho} \frac{u_t}{u_r} \left( \frac{T_{\infty}}{T_r} \right)^{\gamma/2} = 1 \]

where

\[ \Gamma = \gamma \left( \frac{T_{\infty}}{T_r} \right) \left( \frac{T_{\infty}}{T_r} \right)^{\gamma/2} \]

\[ \left( \frac{u_t}{u_r} \right) = \left( \frac{u_t}{u_r} \right) \]

\[ \left( \frac{v_t}{v_r} \right) = \left( \frac{v_t}{v_r} \right) \]

Maintaining similarity between engine and test conditions is necessary for duplicating thermal performance of the test components. The vane airfoil temperature distribution is generally sought by these tests of air-cooled turbine vanes. Reference (2) has shown that airfoil temperatures at engine conditions can be predicted from test results by a dimensionless temperature difference ratio. This ratio \( T_d = T_{\infty} - T_r \), which is also called the cooling effectiveness \( \phi \), is based on a one-dimensional heat balance which assumes heat flow only from the gas to the coolant by convection and conduction.

Figure 4 is a representative cross-sectional schematic of a cooled turbine component with a layer of a ceramic coating on a metal substrate. The component temperature is assumed known at the metal-ceramic interface. The following one-dimensional equations can be written by neglecting lateral heat conduction in the component and radiation heat transfer between the component and the surrounding environment.

\[ q_g = q_z = q_w = q_c \]

where
Tzi is the metal-ceramic interface temperature, which is also the gas side metal temperature Twi. Two inlet total gas and coolant temperature values can be substituted for local values (Ref. (9)) and the above equations then combined to obtain the following dimensionless form. It is also assumed that the heat transfer areas through the airfoil are equal (no curvature).

\[
q_g = h_g A_g (T_g - T_{zo}) = Nu_g h_g A_g (T_g - T_{zo}) \frac{k_g}{T_g}
\]

\[
q_2 = \frac{k_A}{T_2} (T_{zo} - T_{ci})
\]

\[
q_w = \frac{k_A}{T_w} (T_{wi} - T_{wl}) = \frac{k_A}{T_w} (T_{wo} - T_{wi})
\]

\[
q_c = h_c A_c (T_{wi} - T_c) = \frac{k_A}{T_w} (T_{wi} - T_c) \frac{k_c}{T_c}
\]

\[
T_{ci} \text{ is the metal-ceramic interface temperature which is also the gas side metal temperature } Twi. \text{ The inlet total gas and coolant temperature values can be substituted for local values (Ref. (9)) and the above equations then combined to obtain the following dimensionless form. It is also assumed that the heat transfer areas through the airfoil are equal (no curvature).}
\]

\[
(T_g - T_{zo}) = \left( \frac{Nu_g}{k_g} \right) + \left( \frac{Nu_g}{k_g} \right)
\]

\[
(T_{wi} - T_{co}) = \left( \frac{Nu_g}{k_g} \right) + \left( \frac{Nu_g}{k_g} \right)
\]

\[
(T_{wi} - T_{wl}) = \left( \frac{Nu_g}{k_g} \right) + \left( \frac{Nu_g}{k_g} \right)
\]

\[
(T_{wi} - T_{wl}) = \left( \frac{Nu_g}{k_g} \right) + \left( \frac{Nu_g}{k_g} \right)
\]

A vane without the ceramic coating can be represented by the following equations (6), (8), and (9) where Tzo in equation (6) is replaced by Twi.

In order that the model cooling effectiveness, \( \varphi \), be directly applicable equations (12) and (13) show it is necessary to have Nusselt number similarity. Since heat transfer results follow the form \( Nu = c_1 Re^{m/Pr^n} \) Reynolds and Prandtl number similarity assures achieving the Nusselt number similarity required in equations (12) and (13). In addition, equality of the thermal conductivity ratios must be maintained if total similarity of the cooling effectiveness is to be maintained between test and engine conditions. This is, generally, not possible with most component materials.

Cooling Effectiveness Correction

The inability to maintain equality of the thermal conductivity ratios can lead to significant errors when using test data to predict component temperatures at engine conditions. The magnitude of this error can be determined by calculating the total derivative of equations (12) and (13) with respect to these variables.

Reference (4) has shown that, since the ratio \( b_{g}/h_{c} \) is proportional to \( h_{g}/k_{g} \), correcting for the gas-to-coolant thermal conductivity ratio between test and engine conditions is equivalent to correcting the heat transfer coefficient in a simplified model. The error model represented by equation (14) can also be used to calculate a correction factor between test data and engine data.

The first partial derivative in equation (14) becomes

\[
\frac{\partial \varphi_z}{\partial k_g} = \left( \frac{Nu_g k_g}{k_z} \right) \left( \frac{Nu_g k_g}{k_z} \right) \left( \frac{Nu_g k_g}{k_z} \right)
\]

The second partial derivative in equation (14) becomes

\[
\frac{\partial^2 \varphi_z}{\partial k_g^2} = \left( \frac{Nu_g k_g}{k_z} \right) \left( \frac{Nu_g k_g}{k_z} \right) \left( \frac{Nu_g k_g}{k_z} \right)
\]

The third partial derivative in equation (14) becomes

\[
\frac{\partial^3 \varphi_z}{\partial k_g^3} = \left( \frac{Nu_g k_g}{k_z} \right) \left( \frac{Nu_g k_g}{k_z} \right) \left( \frac{Nu_g k_g}{k_z} \right)
\]

Finally, combining equations (15) to (17) and simplifying, the correction factor for a ceramic coated turbine vane is

\[
\frac{\partial \varphi_{z,e-t}}{\partial k_g} = \left( \frac{1 - \varphi}{\varphi} \right) \left( \frac{Nu_g k_g}{k_z} \right) \left( \frac{Nu_g k_g}{k_z} \right) \left( \frac{Nu_g k_g}{k_z} \right)
\]

where
was ambient air at 300 K which was not the 180 K re-

bine inlet temperature and pressure of 1550 K and

whatever method the user has most

The gas and coolant thermodynamic and transport prop-

erties are taken from reference

300 K (see Table

fixed, the other, and also

All the terms in these equations for the correction

factors are known or can be calculated. The gas

side heat transfer coefficient can be calculated by

whichever method the user has most confidence in

modeling his experiment. The turbulent flat plate cor-

relation is used herein.

\[ b = 0.029 \frac{k_b}{k_c} Re^{0.8} Pr^{1/3} \]  

The gas and coolant thermodynamic and transport prop-

erties are taken from references (11) and (12), respectively.

RESULTS AND DISCUSSION

The similarity relationship of turbine inlet
gas temperature and pressure expressed by equa-
tion (1) is shown in Fig. 5. One point on each curve
represents a typical gas turbine engine with a tur-
bine inlet temperature and pressure of 1550 K and
8.3 atm for the uncoated vane and 1440 K and 8.6 atm
for the ceramic coated vane. The inlet coolant tem-
perature assumed for these engine conditions was
300 K (see Table II). Choosing either a test gas
temperature or pressure fixes the other, and also
fixes all other parameters which satisfy the simi-
larity relationships. The reduced gas temperature
and pressure test conditions were 890 K and 4.5 atm
for the uncoated vane and 920 K and 5.2 atm for the
coated vane. The test condition coolant temperature
was ambient air at 300 K which was not the 180 K re-
quired by similarity constraints. The discussion that
follows will refer to data taken at high gas
temperature and pressure as "engine data" and will
refer to data taken at reduced gas temperature and
pressure as "test data."

Cooling effectiveness correction factors were
calculated by the procedure discussed in the ANALYSIS
METHOD section and the test and engine conditions
selected for study. Since the coolant inlet tempera-
ture for the test data was greater than that required
for similarity a correction procedure similar to that
developed in reference (1) was used to evaluate its
effect on the cooling effectiveness. This correction
factor is combined with the corrections for the
ceramic and metal thermal conductivity effects.
These net total correction factors are shown in
Fig. 6 as a percent correction versus the cooling
effectiveness. The net correction for the ceramic
coated vane, at the conditions investigated, is es-
entially zero near a \( \varphi \) of 0.6 and minus 3 percent
at a \( \varphi \) of 0.57. In analysis of the various terms
of the correction equation show that a positive cor-
rrection for the ceramic is off-set by negative cor-
rrections for both the metal and the coolant thereby
reducing the net effect to nearly zero. In constrast,
the net correction for the uncoated vane is strictly
negative, composed only of the metal and the coolant
correction factor terms, and is a function of the
cooling effectiveness. The correction factor term
for the coolant was about the same for both the
ceramic and uncoated vane.

The effects of these correction factors, when
applied to the average cooling effectiveness test
data of the uncoated vane, are shown in Fig. 7. The
average cooling effectiveness was based on the area
weighted average airfoil temperature and thermal
gas and coolant temperatures. The uncorrected test
data are shown to be considerable higher than the
engine data. However, corrected test data (using
correction factors in Fig. 6) are shown to compare
quite well with the engine data.

The correction factors for the ceramic coated
vane test data were nearly zero at cooling effective-
ness values of about 0.4 and decreased to minus 3
percent at a cooling effectiveness of about 0.57.
A comparison of corrected test and engine cooling
effectiveness data in Fig. 8 shows good agreement.

The importance of correcting the test data is
shown in Fig. 9 which is a cross-plot of Figs. 7 and
8. A comparison of the uncorrected test data of
ceramic coated and uncoated turbine vanes would erro-
neously show the thermal barrier coating to be in-
effective. A comparison of test data corrected for
thermal conductivity and engine data of ceramic
coated and uncoated vanes show that the coating actu-
al increases the cooling effectiveness by an aver-
age of about 12.5 percent.

CONCLUSIONS

The thermal performance of a turbine vane can be
evaluated reliably at reduced gas and coolant condi-
tions. However, thermal conductivity corrections
are required for the data at reduced conditions. These
corrections for a ceramic thermal barrier coated vane
are significantly different than the corrections for
an uncoated vane. Comparison of uncorrected test
data, therefore, would show erroneously that the ther-
mal barrier coating was ineffective. When thermal
conductivity corrections are applied to the test data
these data are then shown to be representative of
engine data and also show that the thermal barrier
cooling increases the vane cooling effectiveness by
an average of 12.5 percent.

\[ \Delta \left( \frac{k}{k} \right) = \left( \frac{k}{k} \right)_{t} - \left( \frac{k}{k} \right)_{a} \]

\[ \Delta \left( \frac{k}{k} \right) = \left( \frac{k}{k} \right)_{t} - \left( \frac{k}{k} \right)_{u} \]

The correction factor for an uncoated turbine
vane is written as follows

\[ \Delta \varphi_{u,t} = - \varphi_{u,t} \left( \frac{Nu}{b} \right) \Delta \left( \frac{k}{k} \right)_{t} + \left( \frac{Nu}{b} \right) \Delta \left( \frac{k}{k} \right)_{a} \]  

Finally, the cooling effectiveness at engine
conditions can be defined as the cooling effective-
ness at test conditions plus the correction factor.

\[ \varphi = \varphi_{t} + \Delta \varphi_{e-t} \]  

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an average of 12.5 percent.
REFERENCES


TABLE I. - THERMOCOUPLE LOCATIONS AND COATING THICKNESS

<table>
<thead>
<tr>
<th>Thermocouple</th>
<th>Distance from leading edge, cm</th>
<th>Dimensionless distance</th>
<th>Ceramic coating thickness, cm</th>
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</thead>
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<tr>
<td>Suction surface, L = 7.42 cm</td>
<td>Vane 2</td>
<td></td>
<td></td>
</tr>
<tr>
<td>1</td>
<td>6.42</td>
<td>0.866</td>
<td>0.033</td>
</tr>
<tr>
<td>2</td>
<td>5.07</td>
<td>0.684</td>
<td>0.030</td>
</tr>
<tr>
<td>3</td>
<td>3.88</td>
<td>0.523</td>
<td>0.028</td>
</tr>
<tr>
<td>4</td>
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<td>0.021</td>
</tr>
<tr>
<td>5</td>
<td>1.32</td>
<td>0.178</td>
<td>0.020</td>
</tr>
<tr>
<td>6</td>
<td>0.30</td>
<td>0.067</td>
<td>0.013</td>
</tr>
<tr>
<td>7</td>
<td>0</td>
<td>0</td>
<td>---</td>
</tr>
</tbody>
</table>

| Pressure surface, L = 6.45 cm |                     |                          |
| 8            | 0.39                          | 0.061                  | 0.013                         |
| 9            | 1.25                          | 0.196                  | 0.023                         |
| 10           | 2.52                          | 0.391                  | 0.018                         |
| 11           | 3.78                          | 0.586                  | 0.015                         |
| 12           | 4.85                          | 0.752                  | ---                           |

Note: Tolerance ±0.002 cm


TABLE II. - TEST AND ENGINE CONDITIONS

<table>
<thead>
<tr>
<th></th>
<th>Test conditions</th>
<th>Engine conditions</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Gas temperature, K</td>
<td>Gas pressure, atm</td>
</tr>
<tr>
<td>None</td>
<td>890</td>
<td>4.5</td>
</tr>
<tr>
<td>ZrO2</td>
<td>920</td>
<td>5.2</td>
</tr>
</tbody>
</table>
(a) Overall view of facility.

(b) Cross-sectional view of main high-temperature components. (Dimensions are in cm unless noted.)

Figure 1. - Schematic of cascade facility.
Figure 2. - Vane row and location of instrumentation stations in static cascade test section.
Figure 3. - Schematic cross-sectional midspan view of test vane, showing the internal cooling configuration and the thermocouple locations.

Figure 4. - One-dimensional heat transfer model of airfoil wall with ceramic coating. Thermocouple junction assumed to be at the metal-ceramic interface.
Figure 5. - Similarity relationships for the test and engine conditions investigated.

Figure 6. - Test data correction factors for ceramic coated and uncoated vanes.
Figure 7. - Average cooling effectiveness test and engine data for an uncoated turbine vane.

Figure 8. - Average cooling effectiveness test and engine data for a ceramic coated turbine vane.
Figure 9. Comparison of test and engine cooling effectiveness data of ceramic coated and uncoated turbine vanes.