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AN ADVERSE EFFECT OF FILM COOLING ON THE SUCTION SURFACE OF A TURBINE VANE

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This information is being published in preliminary form in order to expedite its early release.

ABSTRACT

A J-75 size turbine vane with film cooling holes on the suction surface near the leading edge was tested with and without film cooling flow in a four vane cascade. Results show that the cooling effectiveness on the aft portion of the vane suction surface can decrease with the addition of film cooling near the leading edge. Apparently the film cooling air flow caused a laminar or transitional boundary layer to become a transitional or turbulent boundary layer. The vane was tested at a gas temperature and pressure of 1260 K (1800° F) and 22.7 newtons per square centimeter (33 psia), a coolant temperature of 280 K (50° F), film cooling flow ratios from 0.0 to 0.026, and backside midchord cooling flow ratios of 0.007 and 0.035.

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SUMMARY

A J-75 turbine vane with film cooling holes on the suction surface near the leading edge was tested in a four vane cascade. Results show that the cooling effectiveness on the aft portion of the suction surface can decrease with the addition of film cooling near the leading edge, that is, the introduction of the film cooling air flow near the leading edge can result in vane wall temperatures in the trailing edge region which are higher than those temperatures which would exist without film cooling. The effective gas-side heat transfer coefficient with film cooling apparently increased over that without film cooling. The increase in the heat transfer coefficient was caused by a laminar or transitional boundary layer being made to go transitional or turbulent by the injection of film cooling air. The vane was tested at a gas temperature and pressure of 1260 K (1800° F) and 22.7 newtons per square centimeter (33 psia), a coolant temperature of 280 K (50° F), film cooling flow-to-gas flow ratios from 0.0 to 0.026, and midchord convection cooling flow-to-gas flow ratios of 0.007 and 0.035.

INTRODUCTION

The film cooling effectiveness on the suction surface of a J-75 size turbine vane was investigated in a four vane cascade.

For certain combinations of turbine inlet temperature and pressure, laminar or transitional flow may exist over all or a portion of the suction surface of turbine vanes. The injection of film cooling air into a laminar or transitional boundary layer may result in a transitional or turbulent boundary layer with an accompanying increased gas side heat transfer coefficient at downstream locations on the vane surface. Therefore, in certain instances the benefits derived from film cooling are not sufficient to counter balance the effect of increased gas side heat transfer coefficients. Experimental results presented in reference 1 indicate that under certain conditions film cooling air ejection near the leading edge of a vane resulted in decreased cooling effectiveness near the trailing edge.

In order to investigate this effect in more detail a control experiment concerning film cooling on the suction surface was performed. The test vane had a row of film cooling holes near the leading edge. Vane metal temperatures were measured both with and without film cooling flow. For each case the aft portion of the vane had a separate air supply. The test conditions investigated were: gas temperature and pressure of 1260 K (1800° F) and 22.7 newtons per square centimeter (33 psia), a coolant temperature of 280 K (50° F), film cooling-to-gas flow ratios from 0.0 to 0.026, and midchord coolant-to-gas flow ratios of 0.007 and 0.035. The above gas condition Reynolds number is representative of 1650 K (2500° F) and 3 atmospheres.

SYMBOLS

h_c coolant side heat transfer coefficients

h_g gas side heat transfer coefficients

T_{Ti} gas total inlet temperature

T_{ci} coolant inlet temperature

T_w local vane metal temperature

$\dot{w}_{mc}$ coolant flow rate through the vane midchord

$\dot{w}_{fc}$ coolant flow rate through the film cooling holes

$\dot{w}_g$ gas flow rate per channel

ϕ temperature difference ratio, $(T_{Ti} - T_w)/T_{Ti} - T_{ci})$

APPARATUS

A four-vane cascade, designed for J-75 sized vanes, was utilized to obtain the results presented in this report. This facility is discussed in detail in reference 2. The cascade test section was an annular sector of a vane row and contained four vanes and five flow channels.

One of the central two vanes served as the test vane. Cooling air for this test vane was metered by two independent systems; one supplying the film cooling holes through the leading edge plenum and the other supplying the midchord and split trailing edge through the midchord impingement insert.

The vane configuration discussed in this report had a span of 10.2 centimeters (4 in.) and a midspan chord length of 6.5 centimeters (2.5 in.). A cross sectional schematic of the vane including the relative thermocouple locations is shown in figure 1.

After the coolant airflow entered the leading edge plenum through the hub of the vane, it exited through 58 film cooling holes located

1.78 centimeters (0.7 in.) downstream of the leading edge stagnation point. These film cooling holes had centers equally spaced 0.157 centimeters (0.062 in.) apart and had a diameter of 0.064 centimeter (0.025 in.). They were angled at 28° with respect to a line tangent to the suction surface at the exit region.

The airflow which entered the midchord impingement insert through the tip of the vane impinged on the internal surfaces of the vane suction and pressure sides by flowing through an array of 0.038 centimeter (0.015 in.) diameter holes. There were 481 and 334 holes, respectively, on the suction and pressure sides of the midchord impingement insert. This flow then exited through a split trailing edge containing pin fins. There were four rows of oblong pin fins which were 0.38 centimeter (0.15 in.) by 0.26 centimeter (0.10 in.) and varied in height from 0.18 to 0.094 centimeter (0.070 to 0.037 in.). A single row of round pin fins at the trailing edge had a dimaeter of 0.20 centimeter (0.08 in.) and a height of 0.064 centimeter (0.025 in.).

RESULTS AND DISCUSSION

Figure 2 shows the local temperature difference ratio ϕ as a function of the film cooling-to-gas flow ratio $\dot{w}_{fc}/\dot{w}_g$ on the suction surface. These ϕ data were for thermocouple position 2 shown in figure 1, where

$$\phi = (T_{Ti} - T_w)/(T_{Ti} - T_{ci}) \tag{1}$$

The parameter in the figure is the midchord coolant flow ratio $\dot{w}_{mc}/\dot{w}_g$. The midchord coolant flow supplies the convection cooling and exits from the vane through the split trailing edge. The upper curve on the figure is for a midchord coolant flow ratio of 0.035 while the lower curve is for a midchord coolant flow ratio of 0.007.

The figure shows that cooling effectiveness at the location of thermocouple 2 at first decreases with the addition of film cooling flow, that is, the addition of film cooling flow causes the temperature of the suction surface of the vane to increase. Although not shown herein, this same phenomenon was observed at thermocouple locations 1 to 5 shown in figure 1. As the film cooling flow ratio increases above 0.0065 the cooling effectiveness increases. For the range of film cooling flow ratios shown, the cooling effectiveness eventually exceeds that with no film cooling flow for the lower midchord cooling flow ratio of 0.007. However, for the higher midchord cooling flow ratio of 0.035 the cooling effectiveness is always less than that with no film cooling flow for the range of film cooling air used.

A film cooled turbine vane such as that of reference 1 or this report could be operated at total coolant-to-gas flow ratios around 0.05 with approximately 0.03 exiting through the split trailing edge and 0.01 through the film cooling holes on both the suction and pressure surfaces of the vane. For these coolant flow ratio values the data presented in figure 2 show that film cooling can have an adverse effect on vane wall temperatures downstream of the film cooling holes.

For the test conditions being considered here, apparently either a laminar or transitional boundary layer exists on the suction surface of the vane before the injection of the film cooling air flow. After the injection the boundary layer becomes transitional or turbulent and thus the gas side heat transfer coefficient increases. The combination of a high gas side heat transfer coefficient and a moderate film cooling effectiveness results in a higher vane wall temperature than is obtained with a lower heat transfer coefficient and no film cooling.

Thermocouple locations 6 and 7 which are immediately downstream of the film cooling holes did not show any adverse film cooling effect. The combination of a high film cooling effectiveness and a high gas side heat transfer coefficient yields a lower vane wall temperature than does the lower heat transfer coefficient without film cooling.

CONCLUSIONS

When gas turbine operating conditions are such that laminar or transitional flow exists on the suction surface of a turbine vane, ejection of local film cooling air near the leading edge can result in a boundary layer transition and increased vane metal temperatures at positions downstream from the point of ejection.

REFERENCES

1. Yeh, Frederick C.; Gladden, Herbert J.; and Gauntner, James W.: Comparison of Heat Transfer Characteristics of Three Cooling Configurations for Air-Cooled Turbine Vanes Tested in a Turbojet Engine. NASA TM X-2580, 1972.

2. Calvert, Howard F.; Cochran, Reeves P.; Dengler, Robert P.; Hickel, Robert O.; and Norris, James W.: Turbine Cooling Research Facility. NASA TM X-1927, 1970.

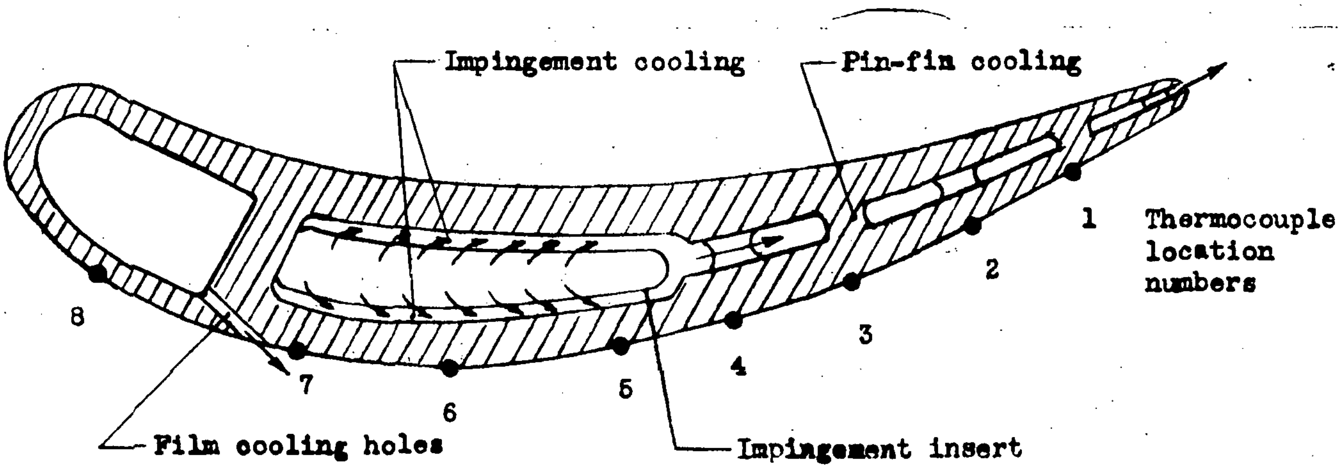

Figure 1 - Cross-sectional view of test vane showing relative location of thermocouples. Not to scale.

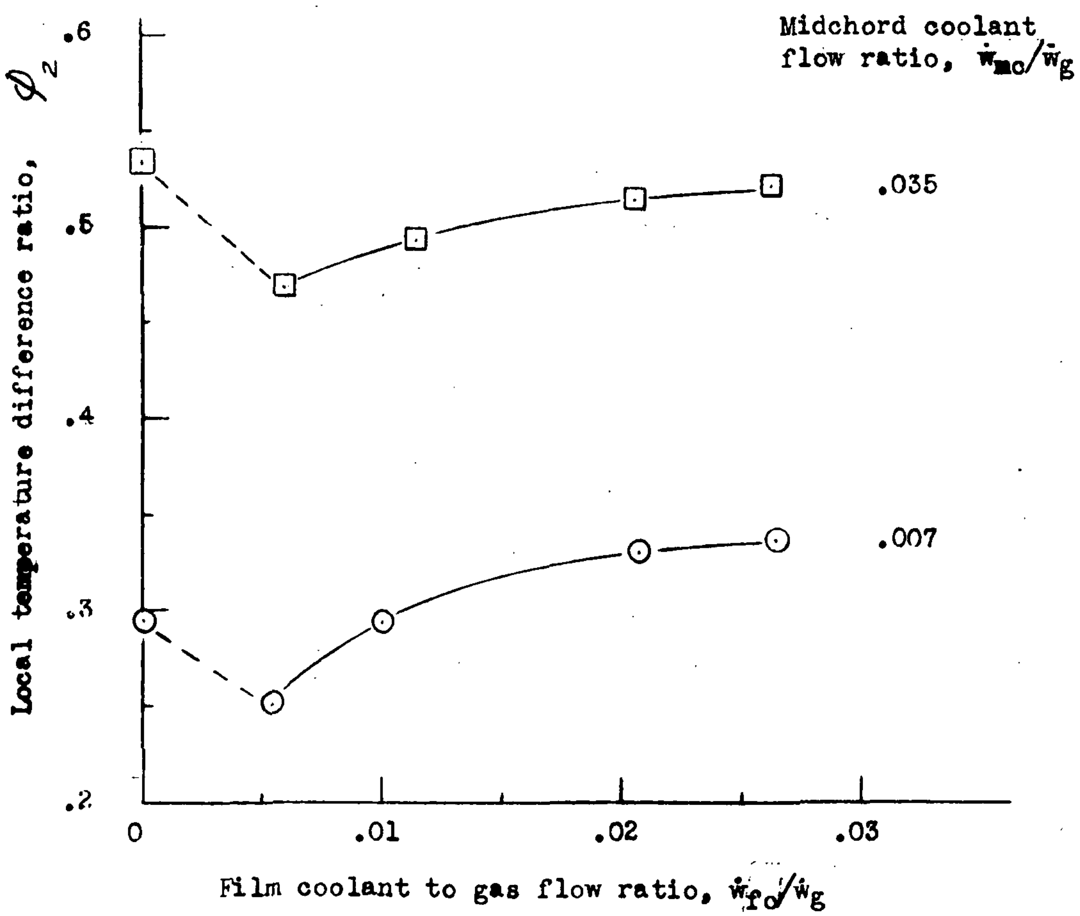

Figure 2 - Decrease in cooling effectiveness on the film-cooled suction surface of a turbine vane for two midchord coolant flow ratios.