NASA TECHNICAL MEMORANDUM

NASA TM X-3536

NASA TM X-3536

FILM COOLING ON THE PRESSURE SURFACE OF A TURBINE VANE

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION · WASHINGTON, D. C. · MAY 1977

1. Report No. NASA TM X-3536	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle FILM COOLING ON THE PRESSURE SURFACE OF A TURBINE VANE		5. Report Date May 1977
		6. Performing Organization Code
7. Author(s) James W. Gauntner and Herbert J. Gladden		8. Performing Organization Report No. E-8764
		10. Work Unit No. 505-04
9. Performing Organization Name and Address Lewis Research Center National Aeronautics and Space Administration Cleveland, Ohio 44135		11. Contract or Grant No.
		13. Type of Report and Period Covered Technical Memorandum
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D. C. 20546		14. Sponsoring Agency Code
15. Supplementary Notes		

16. Abstract

Film-cooling-air ejection from the pressure surface of a turbine vane was investigated, and experimental data are presented. This investigation was conducted in a four-vane cascade on a J75-size turbine vane that had a double row of staggered holes in line with the primary flow and located downstream of the leading-edge region. The results showed that (1) the average effectiveness of film-convection cooling was higher than that of either film cooling or convection cooling separately; (2) the addition of small quantities of film-cooling air always increased the cooling effectiveness relative to the zero-injection case; however, (3) the injected film must exceed a certain threshold value to obtain a beneficial effect of film cooling relative to convection cooling alone.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Heat transfer Film cooling Turbine vane	Unclassified - unlimited STAR category 34

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
Unclassified	Unclassified	17	A02

* For sale by the National Technical Information Service, Springfield, Virginia 22161

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SUMMARY

Effects of film-cooling-air ejection from the pressure surface of a turbine vane were measured in a four-vane cascade on a J75-size turbine vane. The tested vane had a double row of holes along the span inclined at 40° to the vane wall and in line with the primary flow. The test conditions investigated were a gas temperature of 1255 K, pressures of 22.7 to 45.5 N/cm^2, a coolant temperature of 283 K, midchord convection-coolant to primary flow ratios from 0 to 0.06, and film-coolant to primary mass flux ratios from 0 to 1.2 (film-coolant to primary flow ratios from 0 to 0.028).

On the average, a turbine vane was more effectively cooled by a combination of film and convection cooling than by either film or convection cooling separately for mass flux ratios greater than a threshold value of about 0.23. The addition of small quantities of film-cooling air always increased the cooling effectiveness on the pressure surface of a turbine vane relative to the zero-injection case. This is in contrast to what had been formerly observed on the suction surface of the same vane. The presence of film-cooling holes at 1/4 chord length from the leading edge was sufficient to decrease the cooling effectiveness along the entire pressure surface as compared with the same vane without film-cooling holes. Therefore, to obtain a beneficial effect of film cooling relative to convection cooling, the injected film must exceed a certain threshold value below which the cooling effect of the film is insufficient to overcome the roughness effect of the holes.

INTRODUCTION

The high heat-flux conditions expected in the hot-component sections of advanced gas turbine engines dictate the use of the most effective cooling schemes practical to maintain reasonable wall temperatures. Appearing in reference 1 is an analytical comparison of various cooling schemes for the walls of gas turbine engine components. These comparisons show that a combination of convection and film cooling results in a considerable reduction in wall temperatures relative to convection cooling alone. However, in references 2 to 5, the addition of film cooling to a convection-cooled turbine vane resulted, under certain conditions, in increased wall temperatures at downstream locations. These references conclude that a laminar or transitional boundary layer on the suction (convex) surface was tripped to a transitional or turbulent boundary layer both by the film-cooling holes and by the introduction of small amounts of film injection. As a result, both the heat transfer coefficient and the vane wall temperatures increased.

The boundary layer on the pressure (concave) surface of a turbine vane is usually turbulent over the entire surface aft of the leading-edge region. Consequently, the large increases in the heat transfer coefficient and in the vane wall temperatures encountered when a boundary layer is tripped turbulent should not occur with film injection on the pressure surface. However, an increase in boundary layer turbulence due to either the presence of holes or film injection might result in higher wall temperatures by augmenting the heat transfer coefficient associated with the turbulent boundary layer. Thus, a question arises as to the effectiveness of film cooling on the pressure surface of a turbine vane for small values of film injection.

An experimental film-cooling investigation was performed in a turbine vane cascade in a hot-gas environment. The purpose of the investigation was to determine the effect of small amounts of injected film on vane wall temperatures and to verify that the cooling effectiveness of a combined convection-film cooling scheme is superior to that of convection or film cooling separately at the same coolant flow rate.

The rates of coolant flow to the film-cooling holes and to the convection cooling passages in the midchord region were individually measured. Vane wall temperatures corresponding to these flow rates were measured (1) without film-cooling holes, (2) with holes but without film injection, and (3) with film injection. The data are presented as chordwise distributions of cooling effectiveness for vanes with and without film-cooling holes and with varying degrees of film injection. The mainstream environment is similar to that of reference 2, which discusses an adverse effect of film cooling on the suction-surface temperatures of a turbine vane. Included in the present report is a figure that relates the test conditions to engine conditions representative of a supersonic aircraft (SST) at cruise, that is, a gas pressure of 47.6 N/cm^2 and a gas temperature of 1783 K.

SYMBOLS

a_* critical velocity, cm/hr

C_D discharge coefficient

d film-cooling-hole diameter, cm

M Mach number

N number of film-cooling holes

P pressure, N/cm^2

R gas constant for mixture of air and combustion products used in this investigation, 291 J/kg-K

T temperature, K

V velocity, cm/hr

W mass flow rate, kg/hr

x_* dimensionless surface distance

Γ specific-heat function defined in eq. (4)

γ specific-heat ratio

μ viscosity, kg/cm-sec

ρ density, kg/cm^3

φ cooling effectiveness, $(T_g - T_w)/(T_g - T_c)$

Subscripts:

c coolant

g hot gas

t total conditions

w hot-gas-side surface

Superscripts:

e engine condition

t test condition

$-$ average

APPARATUS

Cascade Description

A detailed description of the cascade facility is given in reference 6. The test section was a 23° annular sector of a stator row and contained four vanes and five flow channels. A schematic of the test section is shown in figure 1. The central flow channel was formed by the suction surface of vane 2 and the pressure surface of vane 3. The two outer vanes in the cascade completed the flow channels for the two central vanes and also served as radiation shields between these vanes and the water-cooled cascade walls.

The test vane (vane 3) had two separate cooling-air systems; one for the film-cooling flow, and the second for the convection flow. The cooling-airflow rates were metered by turbine-type flowmeters.

Vane Description

A J75-size turbine vane with a span of 9.78 centimeters and a chord of 6.27 centimeters was used in this investigation. The vane material was MAR M-302. The internal cooling configuration, shown in figure 2, consisted of an impingement-cooled midchord region and a pin-fin-augmented, convection-cooled trailing-edge region. This vane was designed to have an impingement-cooled leading edge. However, for this investigation, the leading-edge impingement tube was removed, and the chamber was blocked at the tip end. Therefore, the cooling air entering the vane from the vane tip plenum was restricted to convection cool the midchord and trailing-edge regions only. The leading-edge chamber served as a plenum for the film-cooling holes, which were added to the vane after the initial series of tests. The film-cooling air entered the plenum at the vane hub and was measured separately.

Two rows of staggered holes, 0.071 centimeter in diameter, were positioned on the pressure surface, as shown in figure 2. The distance of the holes from the leading edge is given in table I. The spacing between rows was 3.5 diameters, and the holes in each row were spaced 2.2 diameters apart in the spanwise direction. There were 58 holes in the first row and 59 holes in the second row. All holes were inclined at an angle of approximately 40° to the airfoil surface and in line with the mainstream flow.

The midchord supply tube contained a staggered array of 0.038-centimeter-diameter holes. There were 481 and 334 holes, respectively, on the suction and pressure surfaces. The midspan chordwise center-to-center hole spacings were 0.24 and 0.28 centimeter, and the spanwise center-to-center hole spacings were 0.20 and 0.23 centimeter on the suction and pressure surfaces, respectively. The hole-to-impingement-surface spacing was 0.076 centimeter, or two hole diameters.

The split trailing edge contained four rows of oblong pin fins and a single row of round pin fins. The oblong pin fins were 0.38 centimeter by 0.25 centimeter and varied in height from 0.18 to 0.094 centimeter. The round pin fins had a diameter of 0.20 centimeter and a height of 0.064 centimeter.

INSTRUMENTATION

Seven Chromel-Alumel thermocouples were located at the midspan on the pressure surface of vane 3, with six of the seven located downstream of the film-cooling holes (fig. 1). The locations of the thermocouples and the thicknesses of the vane wall are given in table I. Figure 2 also shows the thermocouple locations. The construction and installation of the thermocouples are discussed in reference 7. The cooling-air temperature and pressure were measured at the inlet to the vanes. The primary-flow temperature and pressure were measured at the inlet to the vane row by means of spanwise traversing probes. These and other operational instrumentation are discussed in reference 6.

ANALYTICAL METHODS

The local cooling effectiveness is defined as the ratio of the difference between the gas and vane temperatures to the difference between the gas and coolant temperatures.

$$\varphi = \left(\frac{T_g - T_w}{T_g - T_c}\right) \tag{1}$$

Cooling effectiveness data are presented as a function of the dimensionless surface distance, x_*, as measured from the leading edge of the vane, with the mass flux ratio $(\rho V)_c/(\rho V)_g$ as the parameter.

The mass flux ratio is defined as

$$\frac{(\rho V)_c}{(\rho V)_g} = \frac{W(\rho_t/\rho)(a_*/V)}{(NC_D \pi d^2/4)\rho_t a_*} \tag{2}$$

where

$$\frac{\rho}{\rho_t} = \left(1 + \frac{\gamma - 1}{2} M^2\right)^{-1/(\gamma-1)}$$

$$\rho_t = \frac{1}{100} \frac{P_t}{RT_t}$$

$$\frac{V}{a_*} = \left[\frac{\gamma + 1}{2} M^2 \left(1 + \frac{\gamma - 1}{2} M^2\right)^{-1}\right]^{1/2}$$

$$a_* = (3.6\times10^5)\left(\frac{2\gamma}{\gamma + 1} RT_t\right)^{1/2}$$

For the data of this report the local Mach number of the gas stream was 0.425, γ was 1.305, and C_D was assumed to be unity. For these values the mass flux ratio of equation (2) reduces to

$$\frac{(\rho V)_c}{(\rho V)_g} = 0.023 \frac{\sqrt{T_t}}{P_t} w_g\left(\frac{W}{W_g}\right) \tag{3}$$

where T_t, P_t, W_g, and W/W_g can be found in table II.

To relate the test performance of this cooled turbine vane to actual engine performance, both must have similar distributions of their momentum-thickness Reynolds numbers and critical Mach numbers. To achieve these similar distributions, the test temperature and pressure are related to the actual engine temperature and pressure. Equation (5) of reference 8 gives the functional relation between gas pressure and temperature for test and engine conditions that will provide the same Reynolds number and critical Mach number distributions around the vane:

$$\frac{P_g^t}{\mu_g^t} \frac{\Gamma_g^t}{\sqrt{(RT)_g^t}} = \frac{P_g^e}{\mu_g^e} \frac{\Gamma_g^e}{\sqrt{(RT)_g^e}} \tag{4}$$

where

$$\Gamma = \sqrt{\gamma}\left(\frac{2}{\gamma + 1}\right)^{(\gamma+1)/2(\gamma-1)}$$

Figure 3 displays this functional relation for the test conditions of this report. For example, a supersonic aircraft at cruise has a turbine inlet pressure of 47.6 N/cm^2 and a turbine inlet temperature of 1783 K. These engine conditions are simulated by some of the data of this report, that is, a gas pressure of 31 N/cm^2 and a gas temperature of 1255 K. These test and engine conditions are denoted by the open circular and triangular symbols, respectively, in figure 3. The remaining three test points of this investigation are denoted in figure 3 by the solid circular symbols. A similar relation for an SST aircraft at takeoff is included for reference, with the engine operating condition denoted by the solid triangular symbol in figure 3.

TEST PROCEDURE

The investigation was conducted at the following nominal conditions:

Gas inlet total temperature, K	1255
Gas inlet total pressure, N/cm^2	22.7, 25.5, 31.0, 45.5
Coolant inlet temperature, K	283
Convection-coolant to primary flow ratios	0 to 0.06
Film-coolant to primary flow ratios	0 to 0.028
Film-coolant to primary mass flux ratios	0 to 1.2

The first series of tests were made without film-cooling holes on the vane. The convection-coolant to primary flow ratio for these tests was varied from 0 to about 0.06. A double row of film-cooling holes was added on the pressure surface of the test vane, and the previous test points were repeated over a range of film-coolant to primary flow ratios of 0 to about 0.028. Over this range, the film-coolant to primary mass flux ratio varied from 0 to 1.2.

RESULTS AND DISCUSSION

The effects of film-cooling airflow from a double row of holes in the pressure surface on a turbine vane were investigated. The data from this investigation are presented in table II. However, only the results for the 31-N/cm^2 data set are discussed in the text and presented in the figures; these results are similar to those obtained with the other data.

Relative Effectiveness of Convection and Film-Convection Cooling

Figure 4 presents the cooling effectiveness averaged over the suction and pressure surfaces for both convection-cooled and film-convection-cooled turbine vanes. The data for the cooling effectiveness on the pressure surface were obtained from table II of this report; the data for the cooling effectiveness on the suction surface were obtained from table II of reference 2. The average cooling effectiveness is presented as a function of the ratio of the sum of the convection-coolant and film-coolant flows to the primary flow. Mass flux ratio is the parameter.

The data in figure 4 show the combination of film and convection cooling to be more effective than convection cooling alone for mass flux ratios greater than about 0.23. For mass flux ratios less than 0.23 a threshold value of film cooling exists below which convection by itself is more effective. This effect is discussed in the section Adverse Effect of Holes.

For small total-coolant to primary flow ratios where most of the total coolant flow is used for film cooling, the comparison shows that convection cooling is again more effective. For example, in figure 4 see the curve for a mass flux ratio of 0.23 for small values of total coolant flow. This curve shows that film cooling without adequate convection is ineffective relative to convection cooling alone. This effect is discussed in the following section.

The main conclusion that can be drawn from figure 4 is that, on the average, a turbine vane is more effectively cooled by a combination of film and convection cooling than by convection cooling alone for mass flux ratios greater than a threshold value of 0.23.

Relative Effectiveness of Film and Film-Convection Cooling

Figure 5 shows the cooling effectiveness distributions for both film cooling by itself and combined film-convection cooling, each for the same total coolant flow. The film-cooled vane has a film-coolant to primary flow ratio of 0.02 (mass flux ratio of 0.88) but no convection cooling. As expected, the vane cooling effectiveness decreases with the distance from the film-cooling holes. The other cooling effectiveness distribution is for a combination of a film-coolant to primary flow ratio of 0.007 (mass flux ratio of 0.29) and a convection-coolant to primary flow ratio of 0.013. This effectiveness distribution is uniform over the entire surface. Near the leading edge, the vane is film cooled; near the trailing edge, the vane is convection cooled. For this comparison the average effectiveness for the film-convection-cooled vane is 0.38, compared with 0.35 for the convection-cooled vane. Thus, the film-convection-cooled vane is the more effectively cooled vane in spite of the fact that the comparison is biased in favor of the film-cooled vane. That is, the coolant that was diverted from film cooling the pressure surface must convection cool both the pressure and suction surfaces. The conclusion that can be drawn from figure 5 is that a combination of film and convection cooling is more effective, on the average, than film cooling alone for the same total coolant flow.

Favorable Effect of Film Cooling

Figure 6 presents a chordwise distribution of cooling effectiveness for a convection-cooled vane with a convection-coolant to primary flow ratio of about 0.037. It also presents similar distributions for a film-convection-cooled turbine vane with mass flux ratios of 0, 0.33, 0.49, and 0.73, all having a convection-coolant to primary flow ratio of about 0.037. The data show increasing cooling effectiveness with increasing mass flux ratio; that is, the vane cooling effectiveness did not decrease below that of the convection-cooled vane when small quantities of air were injected into the boundary layer as had occurred on the suction surface of this vane (refs. 2 and 5).

In reference 2, a small amount of film-cooling airflow caused a laminar or transitional boundary layer to become transitional or turbulent, with an accompanying increase in the gas-side heat transfer coefficient. The effect of the increase in the heat transfer coefficient was greater than the increased benefits of film cooling. As a result, the local temperatures increased when small amounts of coolant were injected into the boundary layer. For the present study, the boundary layer on the pressure surface is already turbulent. As a result, the cool film in the boundary layer more than compensates for any increase in the heat transfer coefficient that accompanies a small amount of film injection from discrete holes. Therefore, at small values of mass flux ratio the cooling effectiveness on the pressure surface of a turbine vane always increased relative to the zero-injection case, in contrast to what had been formerly observed on the suction surface of the same vane.

Adverse Effect of Holes

Data were also taken both without film-cooling holes and with film-cooling holes but without film injection. These results, some of which are included in figure 6, show that the presence of film-cooling holes at 1/4 chord length from the leading edge, but without film injection, resulted in decreased cooling effectiveness along the entire pressure surface when compared with the same vane without holes. Therefore, to obtain a beneficial effect of film cooling relative to convection cooling, the injected film must exceed a certain threshold value. Below this value the cooling effect of the film is insufficient to overcome the roughness effect of the holes.

It is possible, though unlikely, that this adverse effect of holes may be the result of the recirculation of hot gas in through the holes at the hub and out through the holes at the tip. However, the test facility used for this investigation performed more as a two-dimensional cascade than as a three-dimensional cascade. As a result, very little spanwise static pressure gradient exists to cause recirculation of the hot gas. Surface static pressure distributions for this vane are presented in reference 9. They show less than a 2-percent difference between the hub and tip static pressures.

In any event, a threshold value of injected film is required to cool the vane to the condition that existed before holes were added. As shown in figure 4, the threshold mass flux ratio was about 0.23.

SUMMARY OF RESULTS

An investigation into the effects of film cooling on wall temperatures on the pressure surface of a turbine vane gave the following results:

1. On the average, a turbine vane was more effectively cooled by a combination of film and convection cooling than by either film or convection cooling separately for mass flux ratios greater than a threshold value of about 0.23.

2. Adding small quantities of film-cooling air always increased the cooling effectiveness on the pressure surface of a turbine vane relative to the zero-injection case. This is in contrast to what had been formerly observed on the suction surface of the same vane.

3. The presence of film-cooling holes at 1/4 chord length from the leading edge, but without film injection, resulted in decreased cooling effectiveness along the entire pressure surface when compared with the same vane without the film-cooling holes. Therefore, to obtain a beneficial effect of film cooling relative to convection cooling, the injected film must exceed a certain threshold value. Below this value the cooling effect of the film is insufficient to overcome the roughness effect of the holes.

Lewis Research Center,
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TABLE I. - THERMOCOUPLE LOCATION

Thermocouple	Distance from leading edge, cm	Dimensionless location	Vane wall thickness, cm
Pressure surface, L = 6.53 cm			
[a]9	1.07	0.163	0.152
10	2.10	.321	.254
11	2.77	.424	.254
12	3.62	.554	.254
13	4.33	.663	.254
14	4.93	.755	.191
15	5.58	.854	.127

[a]Film-cooling holes were located between thermocouples 9 and 10, at 1.60 and 1.85 cm from leading edge (0.245 and 0.284 dimensionless location).

TABLE II. - Concluded.

Reading	Gas total inlet temperature, K	Gas total inlet pressure, N/cm^2	Gas flow per vane channel, kg/hr	Coolant inlet temperature, K	Convection-coolant to primary flow ratio	Film-coolant to primary gas flow ratio	Thermocouple location (fig. 2)						
							9	10	11	12	13	14	15
							Vane wall temperature, K						
66	1258	31.0	1670	281	0.01365	No holes	1203	1074	1013	960	941	906	910
67	1264			281	.00199		1238	1224	1220	1215	1197	1185	1183
68	1261			293	.0628		1146	889	782	723	713	682	664
69	1274			293	.0534		1150	912	807	746	789	704	685
70	1274			292	.0521		1154	918	814	752	739	709	691
71	1277			292	.0458		1160	939	838	775	759	730	713
72	1275			291	.0308		1175	993	904	841	820	793	782
73	1281			290	.0234		1186	1029	950	889	865	842	833
74	1281			289	.0185		1192	1054	984	926	900	878	870
75	1282			289	.0147		1199	1078	1017	962	935	917	909
76	1286			288	.00942		1209	1113	1064	1015	988	971	964
77	1283				.00524		1214	1147	1112	1074	1047	1035	1030
78	1283				.00167		1227	1186	1168	1144	1120	1111	1108
79	1283				.00063		1233	1219	1220	1214	1191	1184	1179
80	1287			292	.0550		1169	923	815	754	743	713	697
81	1280			292	.0442		1172	945	843	782	766	737	722
82	1278			291	.0407		1172	955	855	793	777	749	735
83	1279			290	.0318		1185	993	900	839	819	793	783
84	1279			289	.0177		1200	1059	987	930	906	886	882
85	1283			289	.0105		1213	1106	1052	1002	977	960	957
86	1266			294	.0526	0	1150	921	819	755	744	709	679
87	1266			293	.0420		1157	952	856	791	776	742	715
88	1262			290	.0119		1178	1085	1030	977	952	932	921
89	1264			289	.00653		1192	1131	1091	1048	1023	1006	1000
90	1263			289	.00357		1194	1159	1132	1100	1077	1063	1059
91	1268			292	0	.0200	881	722	833	931	985	1021	1029
92	1264			291		.0144	940	800	910	995	1034	1053	1065
93	1262			290		.0101	992	888	983	1050	1074	1085	1093
94	1264			289		.00660	1098	1047	1093	1128	1129	1130	1133
95	1263			289		.00493	1177	1159	1163	1172	1158	1154	1153
96	1286			287		.0128	951	765	840	887	917	925	933
97	1287				.00490	.00698	1073	954	980	987	989	985	986
98	1284				.00547	.00979	1000	841	899	930	947	949	953
99	1283				.00541	.00605	1104	994	1004	1002	998	990	989
100	1283			286	.00559	.00398	1185	1100	1077	1047	1030	1017	1012
101	1284				.00747	.00528	1135	1026	1014	998	989	978	974
102	1286				.00762	.00765	1033	885	926	941	949	945	945
103	1288				.00759	.00935	998	829	884	910	927	928	929
104	1287	~31.0	~1670	286	.00761	.0106	976	794	857	890	912	915	917
105	1286			285	.0128	.0107	973	767	815	838	857	852	849
106	1288				.0129	.00892	1003	810	848	862	874	867	862
107	1273				.0130	.00645	1055	897	904	897	896	884	878
108	1274				.0129	.00517	1126	986	958	930	920	903	895
109	1272			290	.0131	.00285	1177	1067	1015	968	946	924	913
110	1260			289	.0131	.00416	1150	1034	986	948	930	910	900
111	1260				.0130	.00539	1107	970	948	926	916	899	891
112	1259				.0129	.00686	1036	877	893	891	892	880	874
113	1259				.0131	.00858	997	819	853	864	873	864	860
114	1272			284	.0384	.00519	1109	878	815	773	768	737	714
115	1274				.0381	.00757	1009	754	746	733	740	715	695
116	1273				.0381	.0111	952	673	689	695	712	692	673
117	1278				.0382	.0130	931	645	668	680	701	685	673
118	1278				.0381	.0165	896	604	631	652	680	669	660
119	1267			293	.0368	.00890	989	730	734	728	737	714	694
120	1265			293	.0369	.00744	1016	769	759	745	749	723	703
121	1270			292	.0377	.00595	1070	833	793	763	760	732	709
122	1270			292	.0397	.00428	1135	920	842	790	779	745	721
123	1268			292	.0399	.00283	1150	943	857	797	783	749	723
124	1263	45.5	2450	281	.01474	No holes	1207	1059	997	951	938	917	899
125	1264				.02053		1198	1019	946	898	887	864	843
126	1259				.03040		1195	973	891	840	833	808	781
127	1260				.03807		1185	939	852	800	798	770	735
128	1250				.00156		1237	1223	1220	1218	1204	1196	1200
129	1276			294	.0240	0.00651	1054	829	826	823	826	804	791
130	1277				.0241	.00784	1007	767	787	797	808	789	778
131	1282				.0240	.00962	970	710	748	769	788	773	763
132	1277				.0241	.0117	940	666	715	745	770	758	750
133	1275			295	.0242	.0137	914	635	688	725	755	746	740

TABLE II. - PRESSURE-SURFACE DATA

Reading	Gas total inlet temperature, K	Gas total inlet pressure, N/cm^2	Gas flow per vane channel, kg/hr	Coolant inlet temperature, K	Convection-coolant to primary flow ratio	Film-coolant to primary gas flow ratio	Thermocouple location (fig. 2)						
							9	10	11	12	13	14	15
							Vane wall temperature, K						
1	1248	22.7	1230	284	0.00813	No holes	1188	1092	1043	991	969	956	945
2	1252			284	.01536		1182	1052	988	928	906	890	875
3	1258			282	.03363		1165	974	884	815	795	774	752
4	1262			282	.04681		1155	927	827	755	738	715	686
5	1261			281	.04650		1155	927	826	753	736	713	685
6	1260			281	.04272		1157	940	842	769	751	729	701
7	1259			281	.03546		1160	961	869	798	778	756	731
8	1256			281	.02664		1173	1004	922	855	833	813	795
9	1285			283	.0461	0	1180	975	870	794	772	739	713
10	1283				.0377		1185	1000	903	828	804	772	748
11	1283				.0249		1199	1054	973	903	877	848	833
12	1284				.0169		1209	1017	1030	968	940	917	904
13	1284			284	.00976		1221	1142	1092	1039	1012	993	985
14	1266			290	0	.0276	849	717	811	905	962	992	1005
15	1265			289		.0186	924	800	900	985	1024	1044	1054
16	1266			288		.0141	969	959	955	1030	1059	1074	1082
17	1268					.00967	1034	958	1035	1090	1103	1111	1115
18	1267					.00663	1120	1080	1121	1150	1145	1146	1148
19	1267					.00554	1178	1155	1164	1178	1163	1160	1160
20	1268					.00434	1201	1190	1190	1194	1174	1168	1167
21	1283			284	.00979	.00756	1059	922	945	948	950	944	942
22	1284				.00976	.00928	1023	867	904	919	929	927	925
23	1283				.00916	.0156	938	749	804	844	873	880	882
24	1284				.00944	.0198	899	702	758	805	844	855	859
25	1267			289	.00954	.00326	1188	1106	1060	1019	993	977	972
26	1268			289	.00975	.00417	1155	1088	1043	1007	983	969	964
27	1267	~22.7	~1230	289	.00958	.00549	1138	1033	1008	988	972	961	957
28	1268			288	.00952	.00750	1055	925	943	947	944	939	937
29	1267			288	.00958	.00951	1020	872	905	920	925	923	922
30	1268			288	.00969	.0115	995	832	877	900	911	911	910
31	1268			289	.00951	.00454	1164	1069	1028	997	977	965	960
32	1275			288	.00954	.00531	1145	1040	1011	989	989	963	959
33	1276				.00964	.0102	1008	852	890	909	918	917	916
34	1275				.00962	.0135	965	790	839	870	889	894	894
35	1274				.00956	.0150	946	765	817	853	877	882	883
36	1266			290	.0126	.0176	918	719	767	804	832	838	837
37	1267				.0129	.0126	970	784	825	849	865	864	862
38	1262				.0132	.0095	999	837	863	873	879	874	870
39	1269				.0132	.00693	1063	923	925	918	913	904	899
40	1262				.0133	.00454	1150	1040	992	956	935	920	913
41	1283			283	.0337	.0198	893	634	654	675	703	697	687
42	1286			283	.0342	.0149	938	681	699	710	728	718	705
43	1287			284	.0341	.0111	984	735	744	743	753	737	722
44	1287			283	.0338	.00797	1032	807	795	778	777	756	740
45	1286			291	.0334	.00318	1159	983	898	837	814	785	767
46	1267			292	.0334	.00363	1154	976	892	834	812	784	766
47	1263				.0338	.00471	1138	950	874	824	804	778	760
48	1264				.0341	.00662	1054	849	816	789	782	760	744
49	1264				.0340	.00871	1011	788	779	768	768	750	735
50	1259	25.5	1380	281	.01393	No holes	1194	1075	1015	960	939	921	908
51	1260				.02218		1180	1021	944	882	862	843	826
52	1257				.03211		1170	974	884	817	799	777	754
53	1260				.04029		1161	942	844	776	761	738	710
54	1255				.00042		1226	1214	1212	1204	1185	1174	1170
55	1272			290	.0239	0.0175	911	669	703	730	754	750	744
56	1271			291	.0246	.0130	952	717	744	760	775	767	759
57	1273				.0241	.0107	983	763	782	788	797	785	777
58	1268				.0240	.00908	1003	797	804	804	807	794	784
59	1268				.0237	.00774	1025	834	828	819	817	801	791
60	1267				.0238	.00623	1083	905	872	847	837	817	806
61	1269				.0238	.00521	1137	974	911	869	850	829	817
62	1267			290	.0238	.00268	1160	1012	939	884	860	835	822
63	1265	31.0	1670	281	.02007	No holes	1197	1035	959	901	884	864	847
64	1262	31.0	1670	280	.02791	No holes	1189	993	907	845	830	807	785
65	1257	31.0	1670	281	.03633	No holes	1179	959	865	802	791	765	736

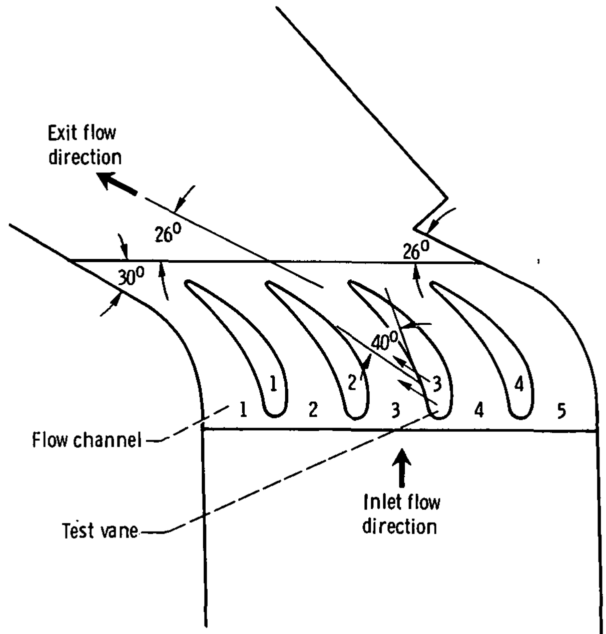

Figure 1. - Schematic of cascade test section.

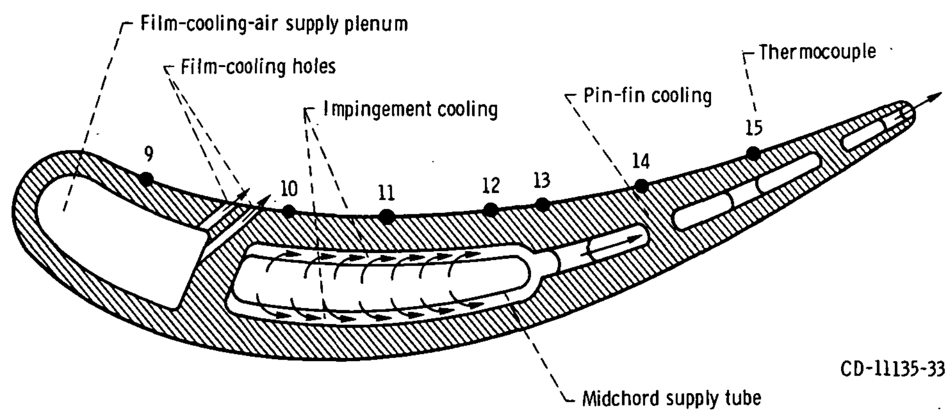

Figure 2. - Cross-sectional midspan view showing internal cooling scheme and thermocouple locations.

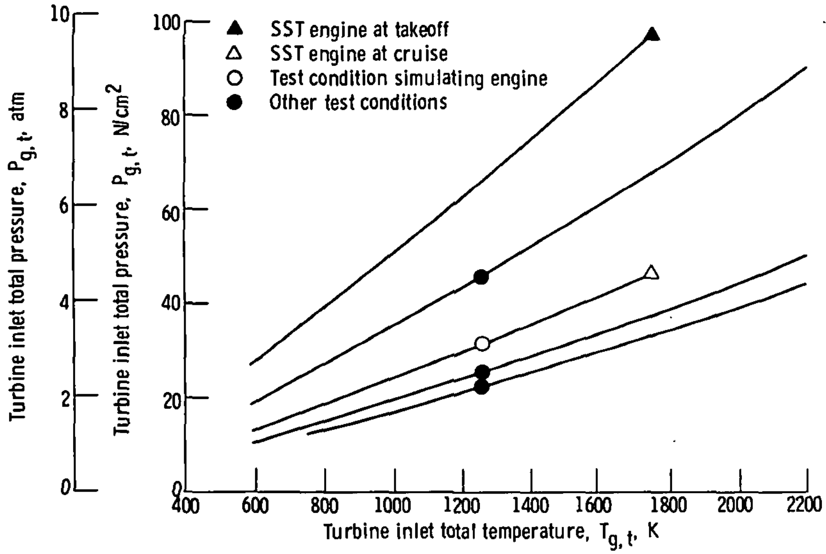

Figure 3. - Similarity curves of constant critical Mach number and momentum thickness Reynolds number distributions around turbine vane.

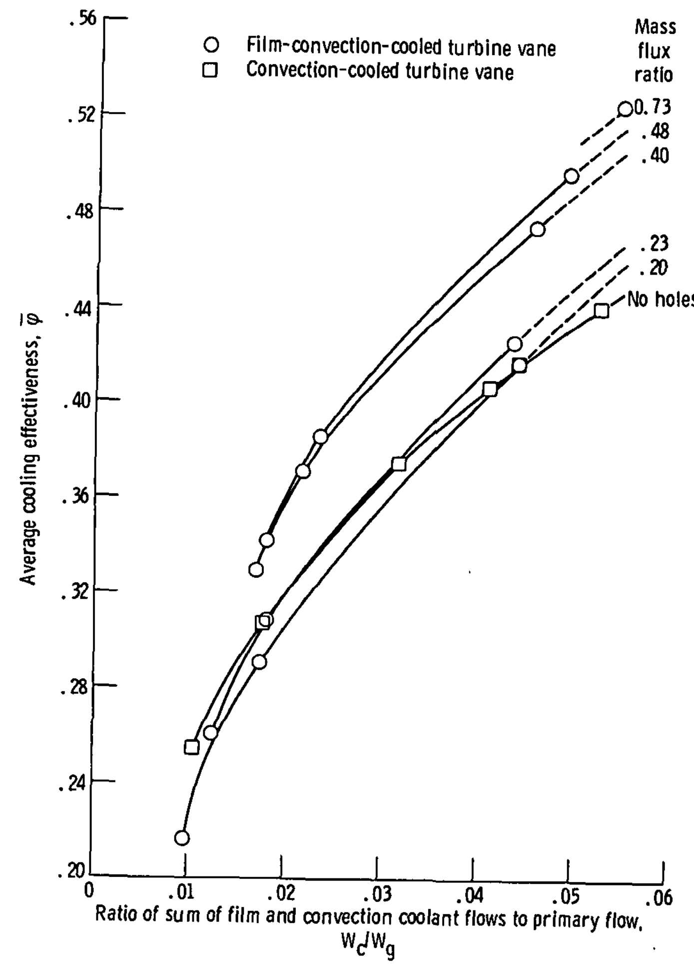

Figure 4. - Comparison of average cooling effectiveness for convection-cooled and film-convection-cooled turbine vanes. Gas inlet total pressure, 31 N/cm^2; gas inlet total temperature, 1276 K; coolant inlet temperature, 283 K.

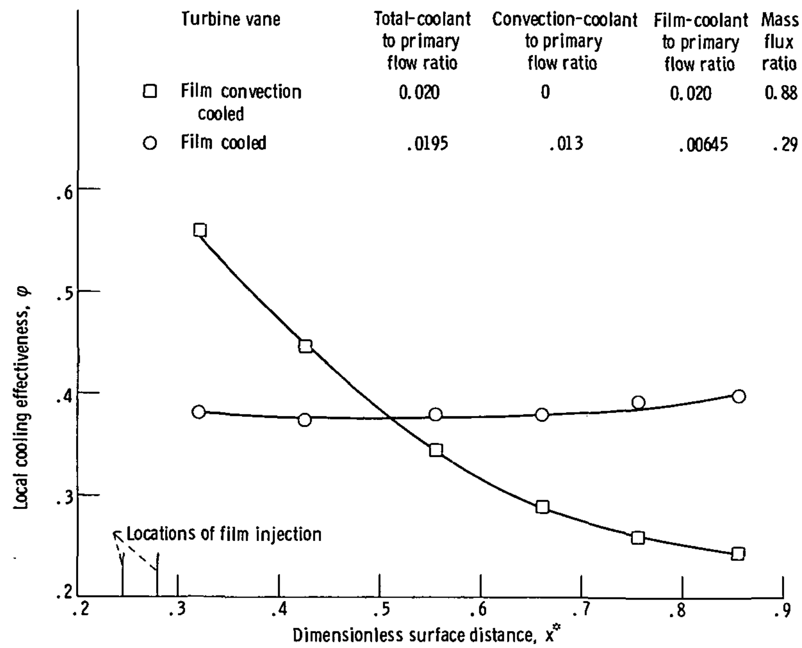

Figure 5. - Comparison of cooling effectiveness distributions for film-cooled and film-convection-cooled turbine vanes. Gas inlet total pressure, 31 N/cm^2; gas inlet total temperature, 1255 K; coolant inlet temperature, 283 K.

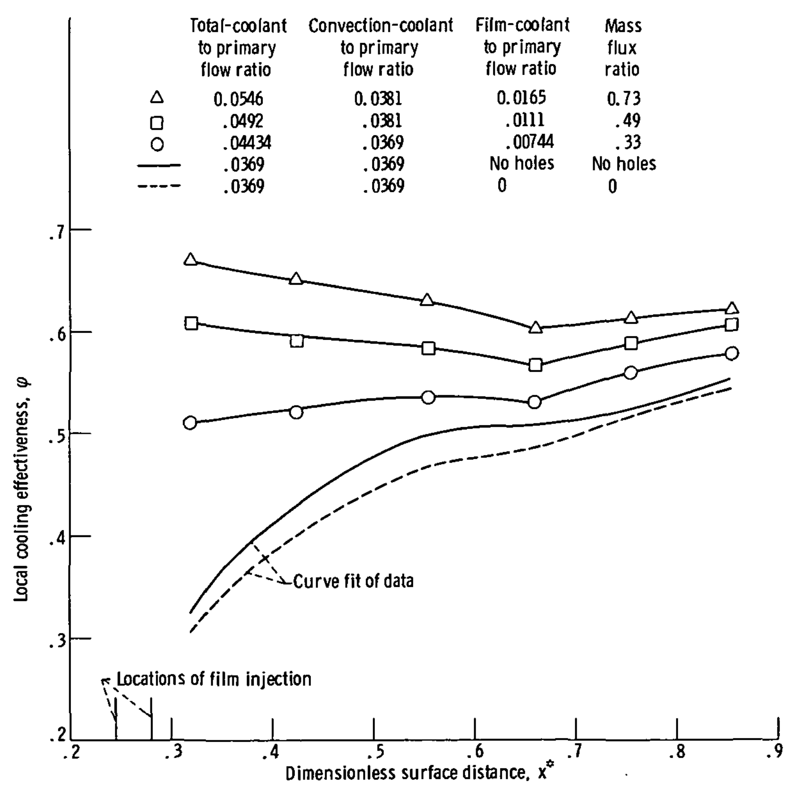

Figure 6. - Effect of mass flux ratio and holes on turbine vane cooling effectiveness. Convection-coolant flow ratio, 0.037; gas inlet total pressure, 31 N/cm^2; gas inlet total temperature, 1255 K; coolant inlet temperature, 283 K.

NASA-Langley, 1977 E-8764

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