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PROCEDURE FOR SCALING OF EXPERIMENTAL TURBINE VANE AIRFOIL TEMPERATURES FROM LOW TO HIGH GAS TEMPERATURES

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16. Abstract

A procedure was developed based on conditions of similarity whereby cooled turbine vane airfoil temperatures measured at low levels of gas temperature can be used to accurately represent both the local and average airfoil temperatures that would be expected at a much higher level of gas temperature. The feasibility of this scaling procedure was verified experimentally in a four-vane cascade capable of operation at gas temperatures and pressures up to 1645 K (2500° F) and 103.4 newtons per square centimeter (150 psia), respectively. A comparison of these scaled airfoil temperatures with experimentally measured airfoil temperatures for the same high temperature conditions is presented.

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SUMMARY

An experimental investigation of an air-cooled turbine vane was made in a four-vane cascade to determine the feasibility of scaling low level turbine and coolant inlet temperature data to higher levels of turbine and coolant inlet temperatures. The procedure developed herein was based on conditions of similarity which resulted in the following requirements: a Reynolds number and a pressure coefficient equality for the gas and coolant systems for the two levels of gas temperature considered, a constant ratio of coolant to gas flow, a constant ratio of coolant inlet to turbine inlet viscosities, and a constant ratio of temperature differences (turbine inlet minus airfoil wall to turbine inlet minus coolant inlet). Experimental data taken at a turbine inlet temperature and pressure of approximately 811 K (1000° F) and 12.6 newtons per square centimeter (18.3 psia), respectively, were scaled to turbine inlet temperatures and pressures of 1145 K (1600° F) and 37.2 newtons per square centimeter (54 psia), and 1645 K (2500° F) and 27.9 newtons per square centimeter (40.5 psia). These scaled data were compared with experimental data taken at the higher temperature conditions. The respective coolant temperatures for the previous turbine inlet temperatures were 319, 409, and 589 K (115°, 275°, and 600° F). A comparison of the scaled low gas temperature data with the equivalent high gas temperature experimental data demonstrated that the scaling procedure can be used with confidence and that the scaled results will accurately represent actual data to a gas temperature as high as 1645 K (2500° F).

INTRODUCTION

A procedure was developed whereby vane airfoil temperatures can be obtained for high levels of turbine and coolant inlet conditions by scaling experimental vane airfoil temperatures obtained at low turbine and coolant inlet conditions. This procedure is of importance because it reduces high turbine temperature testing time to a minimum. Testing at high turbine temperatures is costly and time consuming because of the deteriorating effects of the test environment on the test apparatus and the associated instrumentation. For example, thermocouples used on thin walled vanes and blades are necessarily small (about 0.0076 cm (0.003 in.) diameter wire). Since small diameter thermocouples are highly susceptible to failure when subjected to hot gas streams and thermal cycling, costly reinstrumentation may be necessary if a large amount of high gas temperature testing is required.

The scaling procedure developed herein was based on conditions of similarity. These conditions require the pressure coefficients, Reynolds numbers, and Prandtl numbers to be maintained constant between the low and the high temperature conditions for both the coolant and hot gas systems. This scaling also requires the ratio of the turbine inlet minus airfoil (wall or metal) temperatures to turbine inlet minus coolant inlet temperatures to be maintained constant between the low and the high temperature conditions. This ratio is used as the scaling parameter.

Low temperature gas and coolant data were obtained from tests of an air-cooled turbine vane operated in a four-vane cascade. The cascade was capable of operation at an average gas temperature as high as 1645 K (2500° F) and at pressures up to 103.4 newtons per square centimeter (150 psia). A description of the cascade is given in reference 1. For this investigation the turbine inlet and coolant inlet conditions of 1145 K (1600° F), 37.2 newtons per square centimeter (54 psia), and 409 K (275° F), and 1645 K (2500° F), 27.9 newtons per square centimeter (40.5 psia), and 589 K (600° F) were selected as typical high temperature applications. These points scaled down to 811 K (1000° F), 12.6 newtons per square centimeter (18.3 psia), and 319 K (115° F) for turbine inlet conditions and coolant inlet temperature, respectively (see table I). The scaled low gas temperature airfoil data are compared with the corresponding experimental airfoil data taken at the equivalent higher level of gas temperatures.

TABLE I. - SCALED TEST CONDITIONS INVESTIGATED

Gas conditions				Coolant conditions						Normalized properties	
Inlet temperature		Inlet pressure		Inlet temperature		Percent coolant flow range	Coolant to gas temperature ratio	Coolant pressure ratio, $P_{c,L}/P_{c,H}$		Thermal conductivity,[a] k_H^*/k_L^*	Specific heat ratio, $\gamma^* = \gamma_{g,H}/\gamma_{h,L}$
K	°F	N/cm²	psia	K	°F			scaled	(actual)		
811	1000	12.6	18.3	319	115	2 to 15	0.373	----	----	1.0	1.0
1145	1600	[b]37.2	54	409	275	↓	.357	0.69	0.75	1.013	.98
1645	2500	27.9	40.5	589	600		.359	.51	.49	1.026	.96
[c]811	1000	12.6	18.3	455	360		.561	----	----	1.0	1.0

[a] $k^* \equiv k_g/k_c$.
[b] Pressure level is double the scaled pressure level.
[c] These data were not used in the scaling procedure.

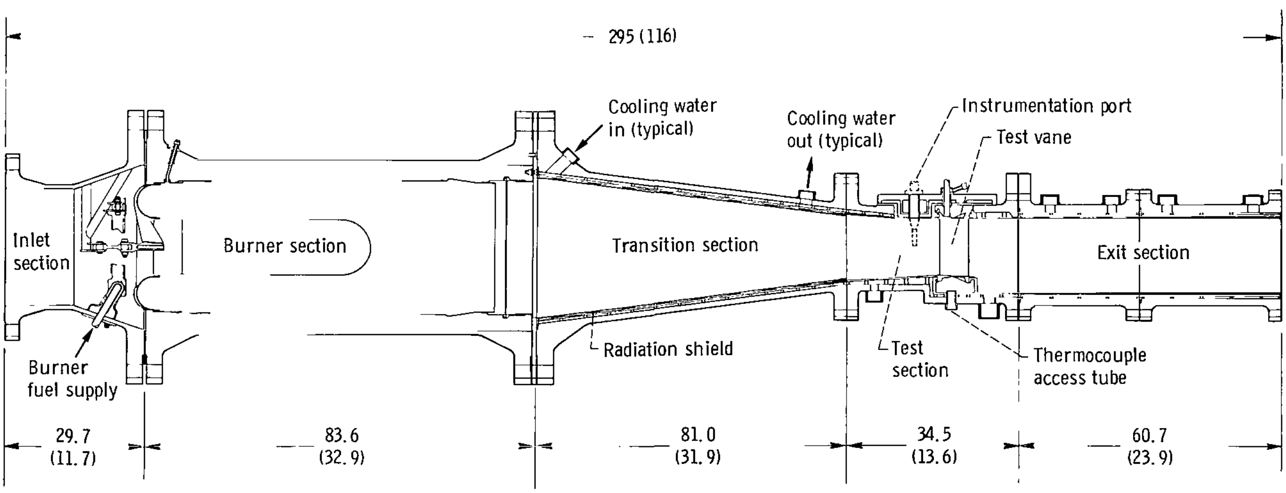


Figure 1. - Schematic cross-sectional view of cascade facility. All dimensions are in centimeters (in.)

APPARATUS

Facility Description

A schematic cross-sectional view of the cascade is shown in figure 1. The facility consists of the following components: (1) an inlet section, (2) a burner section, (3) a circular to annular transition section, (4) the test section, and (5) an exit section. The last three sections were water cooled to achieve structural durability during high temperature operation. More details about the facility are presented in reference 1.

For low temperature tests, the burner section was removed and replaced by a spool piece. Combustion air was supplied to the test section by the auxiliary system shown in figure 2. The burner in the auxiliary system was capable of supplying combustion gas at

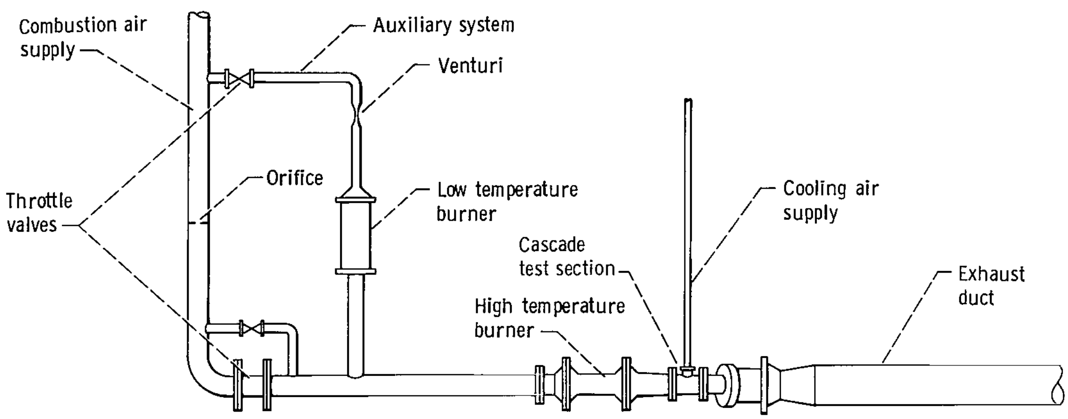


Figure 2. - Schematic showing low temperature burner.

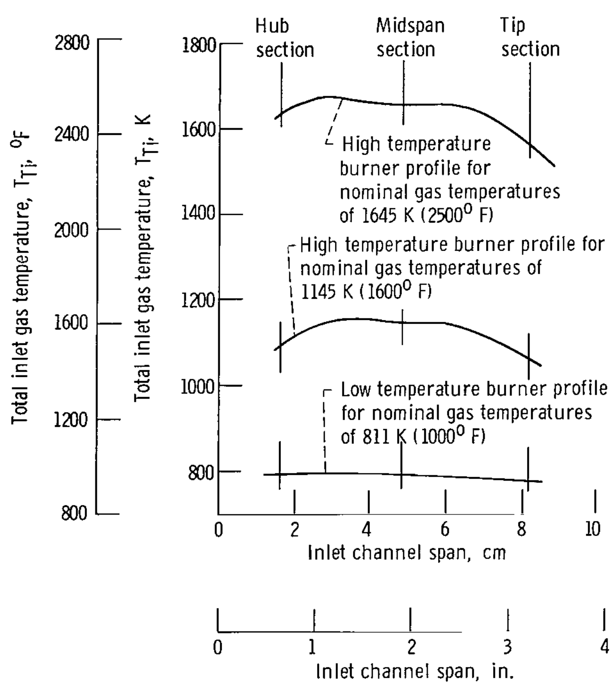


Figure 3. - Typical turbine inlet temperature profiles for high and low temperature burners.

temperatures up to 922 K (1200° F). The gas temperature profile provided to the vane row by the low temperature burner had a maximum to average temperature ratio of 1.011 or less. The high temperature burner had a maximum to average temperature ratio of 1.025 or less at 1645 K (2500° F) and slightly higher at 1145 K (1600° F). The nominal 1645, 1145, and 811 K (2500°, 1600°, and 1000° F) gas temperature profiles shown in figure 3 represent typical patterns experienced by use of the two burners. The low temperature burner profile was more symmetric and of more uniform temperature than the high temperature burner, due in part to its being located farther from the test vanes.

The test section represented an annular sector of a vane row and contained four vanes and five flow channels. A top view of the test section with the access cover removed is shown in figure 4. A vane pack assembly is shown in place. Extending above the vane pack are the cooling air supply tubes for each vane. The central two vanes are

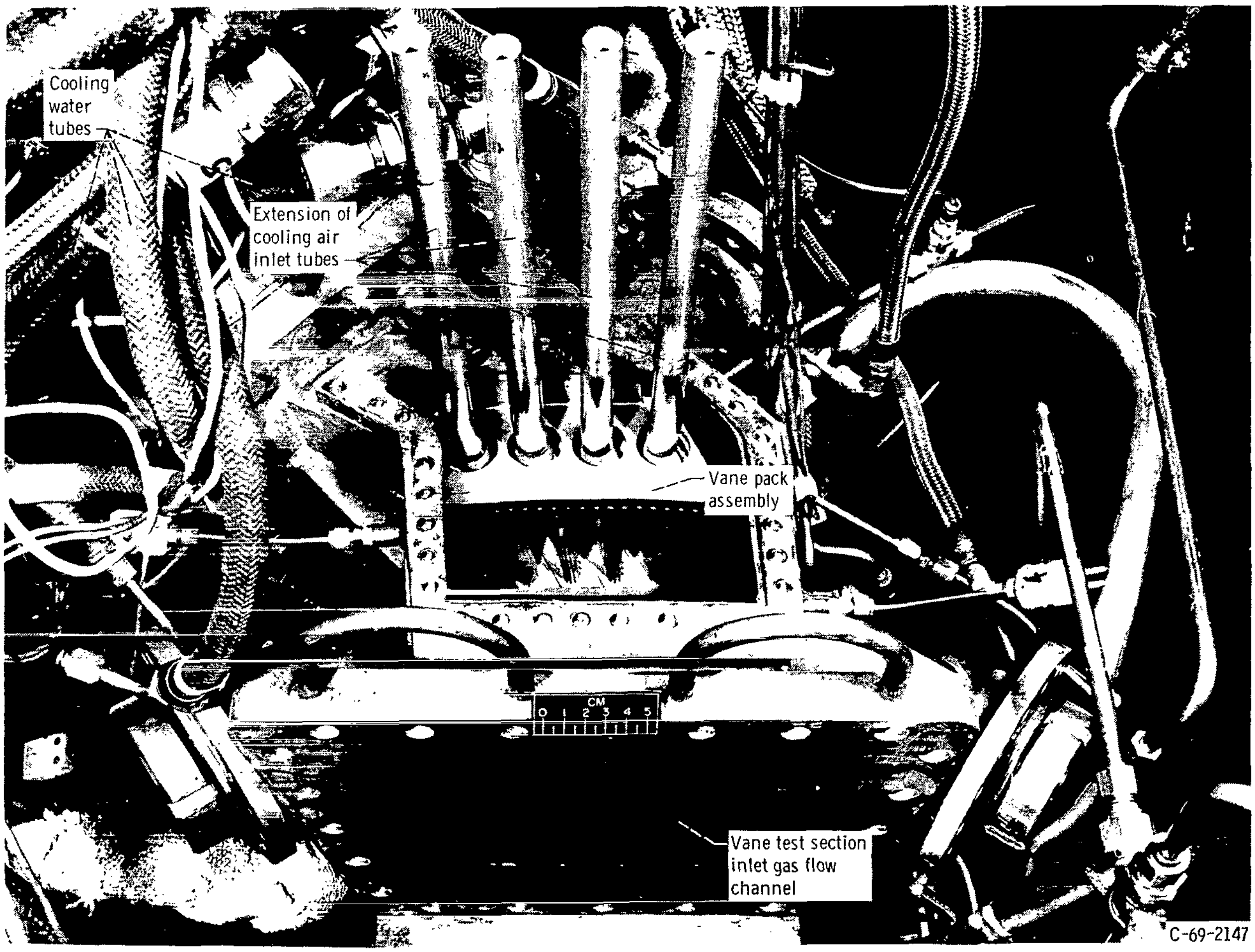


Figure 4. - Top view of vanes in test section with access cover removed.

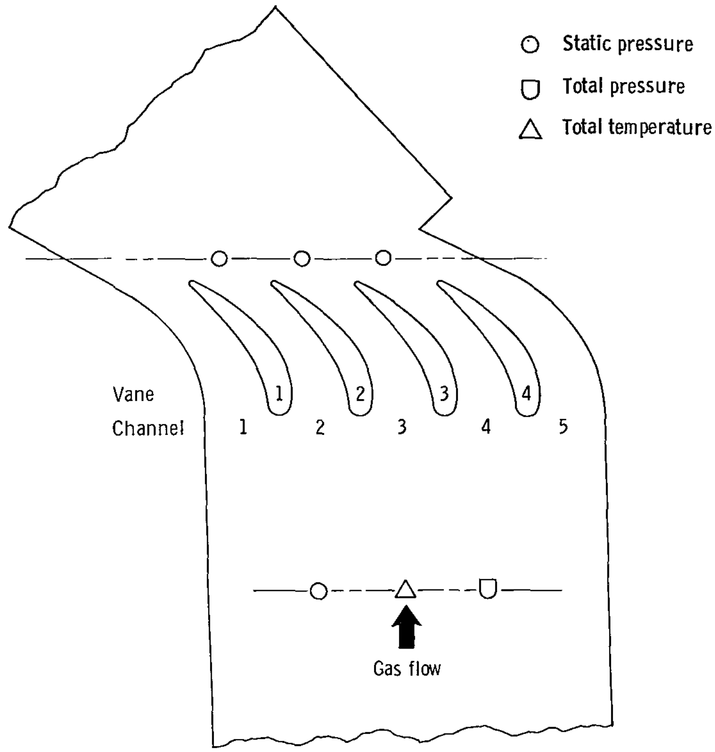


Figure 5. - Schematic top view of test section.

the test vanes and the outer two vanes are called slave vanes (see fig. 5). The two test vanes had a common air supply system and each slave vane had its own air supply system.

Ambient cooling air was individually supplied and metered to the test vanes, the slave vanes, and the inner and outer diameter platforms. The test vanes could also be supplied by a vitiated air heater capable of supplying air at temperatures up to 922 K (1200° F).

Vane Description

A schematic of the cooling configuration of the vane considered herein is shown in figure 6. The vane span was 9.78 centimeters (3.85 in.) and the midspan chord was 6.28 centimeters (2.47 in.). Cooling air entered the vane from the supply tube at the tip or outer diameter of the vane. From a tip plenum chamber the airflow divided into two parts with approximately 20 to 30 percent entering the leading edge impingement tube while the remainder entered the midchord supply tube. The airflow which entered the leading edge flowed through 16 slots to impinge on the internal surface of the leading

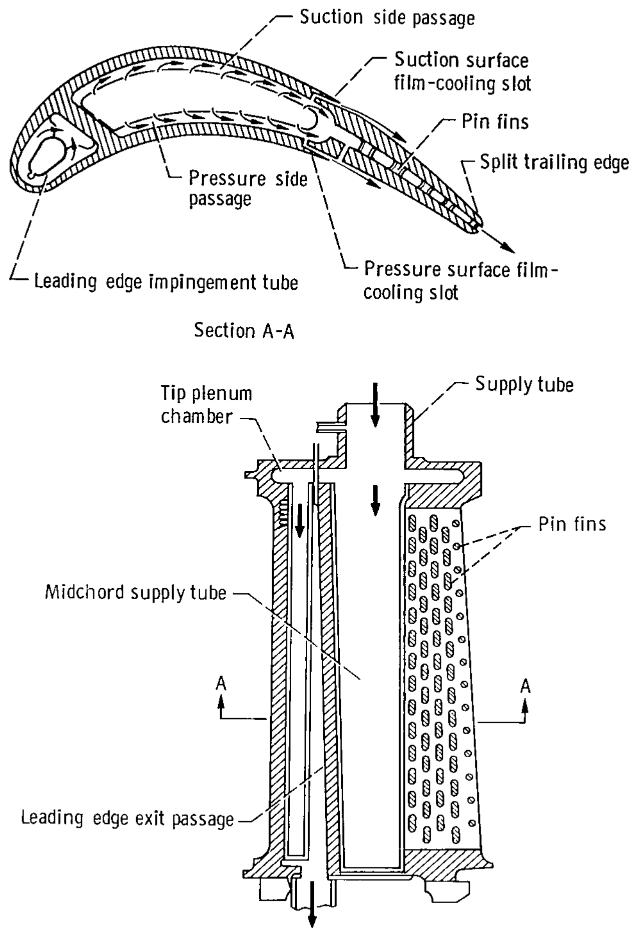


Figure 6. - Vane internal flow configuration.

edge. The airflow passed around the impingement tube in a chordwise direction and into an exit collector passage. This air then flowed to the vane hub and exited into a chamber which in turn vented downstream of the test section.

The airflow which entered the midchord supply tube impinged on the internal surfaces of the vane suction and pressure sides by flowing through an array of holes. There were 481 and 334 holes on the suction and pressure sides, respectively. This flow then exited from the vane through film-cooling slots on the suction and pressure surfaces and through a split trailing edge containing pin fins. There were four rows of oblong pin fins and a single row of round pin fins in the trailing edge. The vane flow characteristics and heat-transfer data are reported in references 2 and 3, respectively.

INSTRUMENTATION

The instrumentation is separated into two categories: general operational instrumentation and research instrumentation. The operational instrumentation was used to set data points and to monitor the general condition of the cascade and its supporting systems. Most of this instrumentation was connected to visual readouts in the control room. The research instrumentation was concentrated on or around the test vanes. These data were recorded by a central data recording system. A more detailed description of the cascade instrumentation and data recording systems is presented in reference 1.

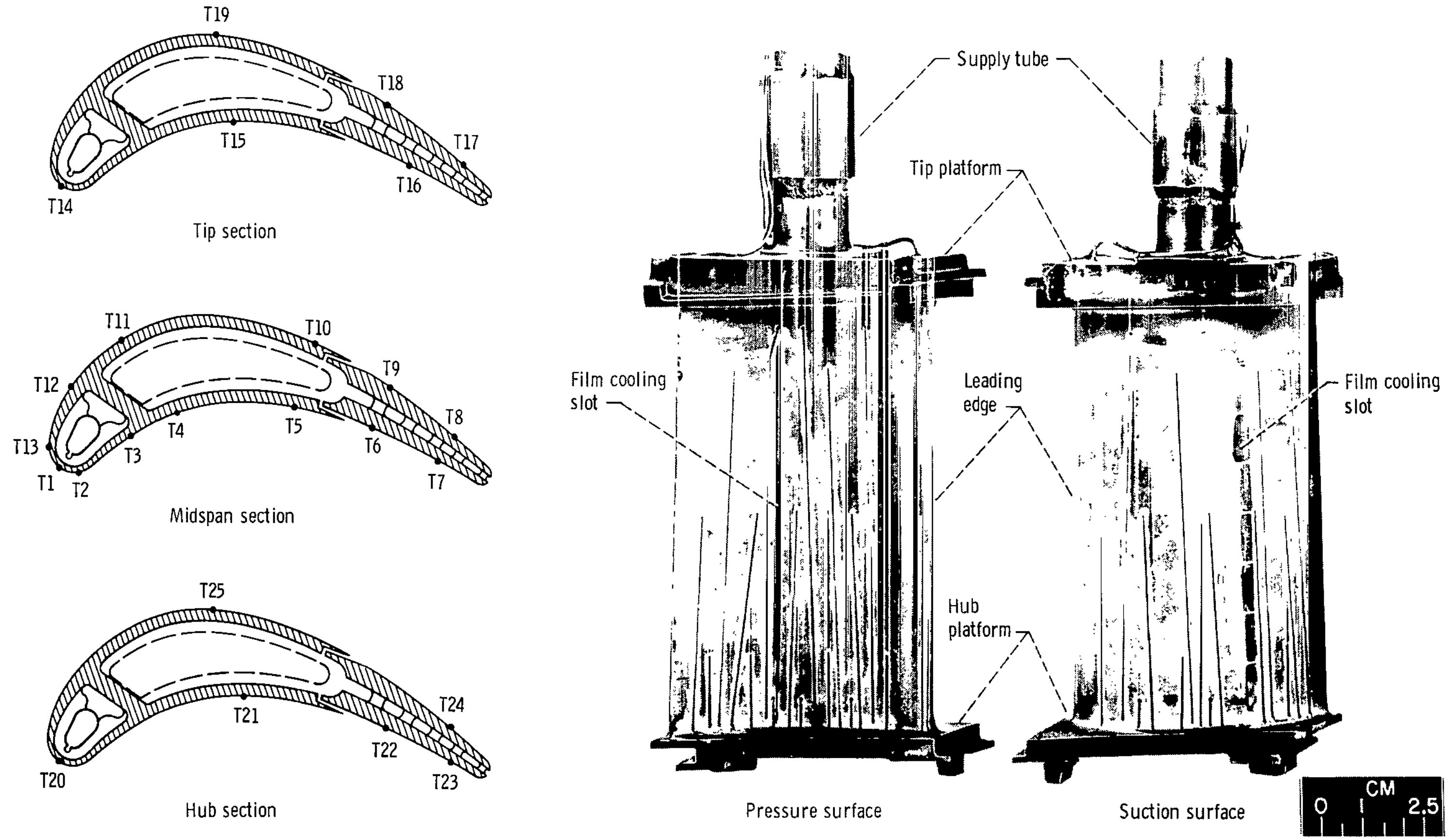
Operational Instrumentation

The cascade was equipped with the general instrumentation required to monitor quantities such as combustion gas total inlet temperature and pressure, vane row exit static pressures, fuel, cooling water and cooling air flow rates, temperatures, and pressures.

Research Instrumentation

Research instrumentation provided detailed information on the gas stream conditions, cooling air flow conditions, and the vane metal temperature distribution. Total temperature and total pressure probes were located upstream of the vane row and were used to radially traverse the gas stream. The total temperature distribution was measured in front of channel 3, and the total pressure distribution was measured in front of channel 4. Figure 5 indicates a static pressure measurement in front of channel 2; this, however, was only measured at the hub platform and assumed to be constant across the gas stream. Static pressures were also measured at the exit midchannel position at both the hub and tip platform. These pressures were used to establish the midspan exit Mach number.

Each test vane was instrumented with an array of 25 thermocouples. Figure 7(a) shows a schematic layout of the thermocouples. Photographs of the suction and pressure surface are shown in figure 7(b). Three spanwise locations (hub, midspan, and tip) were instrumented. The hub and tip sections were located approximately 1.63 centimeters (0.64 in.) from the respective platforms. The midspan of the test vanes contained 13 of the 25 thermocouples and the hub and tip sections contained 6 thermocouples each. The chordwise locations of the thermocouples are given in table II.



(a) Schematic showing location of airfoil thermocouples.

(b) View of suction and pressure surfaces showing instrumentation slots.

Figure 7. - Schematic and photograph of test vanes used in investigation.

TABLE II. - LOCATION OF THERMOCOUPLES SHOWN IN FIGURES 7(a) AND (b)

(a) Suction surfaces

Thermocouple	Suction distance, x		Dimensionless surface distance, x/L
	cm	in.	
Tip section			
14	0.0	0.0	0.0
19	3.10	1.22	.425
18	5.40	2.125	.74
17	6.45	2.54	.885
Midspan section			
1	0.0	0.0	0.0
2	.318	.125	.044
3	.81	.32	.112
4	2.29	.90	.315
5	3.91	1.54	.539
6	5.27	2.075	.727
7	6.18	2.435	.852
Hub section			
20	0.0	0.0	0.0
25	3.21	1.265	.443
24	7.20	2.835	.992

(b) Pressure surfaces

Thermocouple	Suction distance, x		Dimensionless surface distance, x/L
	cm	in.	
Tip section			
15	2.65	1.045	0.408
16	4.90	1.93	.753
Midspan section			
13	0.394	0.155	0.06
12	.97	.38	.148
11	2.08	.82	.319
10	3.42	1.345	.524
9	4.37	1.72	.67
8	5.69	2.24	.87
Hub section			
21	2.65	1.045	0.402
22	4.50	1.77	.683
23	5.47	2.155	.832

The construction of the thermocouple assemblies consisted of Chromel-Alumel wire with magnesium oxide insulation in an Inconel 600 sheath. These assemblies were drawn to two sizes, 0.051 centimeter (0.02 in.) outside diameter and 0.025 centimeter (0.01 in.) outside diameter, with a closed end grounded thermocouple junction formed at one end. Reference 4 presents a detailed description of the procedures used for thermocouple construction.

The five thermocouples nearest the leading edge had 0.025 centimeter (0.01 in.) outside diameters, while the remaining thermocouples had 0.051 centimeter (0.02 in.) outside diameters.

ANALYSIS

The scaling procedure consists of defining conditions of similarity for a system of two compressible flow paths with heat addition or subtraction. Four types of similarity are desired: geometric, kinematic, dynamic, and thermal. Since the same hardware is considered for scaling, geometric scaling is satisfied. Kinematic and dynamic similarity provide that similar flow patterns and similar force distributions prevail at all flow points for the two turbine inlet temperatures considered. Dynamic and thermal similarity provide similar velocity and temperature fields in the fluid. It is stated in reference 5 that kinematic and dynamic similarity will prevail if equality is maintained for the following dimensionless terms:

$$\frac{p}{\rho V^2} = P \tag{1}$$

$$\frac{\rho V l}{\mu} = Re \tag{2}$$

The pressure coefficient (P) is the ratio of pressure forces to inertia forces, while the Reynolds number (Re) is the ratio of inertia forces to friction forces. Consequently, kinematic and dynamic similarity is maintained for equality of Reynolds numbers and pressure coefficients. Thermal similarity is satisfied by an equality of the Prandtl numbers (Pr):

$$\frac{C_p \mu}{k} = Pr \tag{3}$$

The relation of controllable parameters due to similarity will be shown separately for the gas side and the coolant side. There are six parameters which can be varied, namely, gas pressure, temperature, and flow, and coolant pressure, temperature, and flow. However, since the two systems are interconnected at the coolant exit points, there are only five independent variables (the coolant pressure is dependent on the coolant flow, coolant temperature, and the gas side conditions).

Considering the gas side first, equalities of the Reynolds numbers and pressure coefficients are maintained for different gas temperatures by adjusting the gas side parameters to maintain the following ratios:

$$\left(\frac{\rho V x}{\mu}\right)_{g,L} = \left(\frac{\rho V x}{\mu}\right)_{g,H} \tag{4}$$

Substituting $\dot{w}/A$ for ρV in equation (4) and canceling like terms yield

$$\left(\frac{\dot{w}}{\mu}\right)_{g,L} = \left(\frac{\dot{w}}{\mu}\right)_{g,H} \tag{5}$$

Equating the pressure coefficients and making the substitution of

$$\left(\frac{\dot{w}}{A}\right)^2 \frac{RT}{p}$$

for ρV^2 and canceling like terms give equation (1) as

$$\left(\frac{p}{\dot{w}\sqrt{T}}\right)_{g,L} = \left(\frac{p}{\dot{w}\sqrt{T}}\right)_{g,H} \tag{6}$$

Equations (5) and (6) can be combined to relate the gas pressure ratio to the gas temperature and viscosity ratios. When the desired high temperature and pressure conditions and the lower temperature level desired for testing are selected, the lower pressure level required for scaling can be determined by

$$p_{g,L} = p_{g,H}\left[\frac{(\mu\sqrt{T})_L}{(\mu\sqrt{T})_H}\right]_g \tag{7}$$

The pressures and temperatures are turbine inlet static values. It can be shown that the local static pressure distribution similarity (between high gas temperature and low gas temperature conditions) around the periphery of a turbine vane will not vary by more than 1 percent for the turbine inlet static pressure similarity as defined in equation (7).

The pressure coefficient can be represented as a function of the Mach number and specific heat ratio:

$$P = \frac{1}{\gamma M^2} \tag{8}$$

Therefore, it can be seen that for pressure coefficient equality between two gas temperatures, the variation in Mach number is small. Table I shows a maximum variation in specific heat ratio of 4 percent, which would result in only a 2 percent variation in Mach number.

The coolant side equalities of Reynolds numbers and pressure coefficients can be handled in a similar manner as the gas side equalities to obtain the relationship of the controllable parameters. First, a Reynolds number equality yields an equation similar to equation (5):

$$\left(\frac{\dot{w}}{\mu}\right)_{c,L} = \left(\frac{\dot{w}}{\mu}\right)_{c,H} \tag{9}$$

Dividing equation (5) by equation (9) yields the following ratio of gas to coolant flow parameters:

$$\left[\frac{(\dot{w}/\mu)_g}{(\dot{w}/\mu)_c}\right]_L = \left[\frac{(\dot{w}/\mu)_g}{(\dot{w}/\mu)_c}\right]_H \tag{10}$$

Rearranging equation (10) and maintaining an equality of the coolant to gas flow ratio for the two temperature levels considered yield the following relationship of viscosities:

$$\mu_{c,L} = \mu_{c,H}\left(\frac{\mu_{g,L}}{\mu_{g,H}}\right) \tag{11}$$

The high and low gas temperature and the high coolant temperature are known or are selected. Equation (11) can then be used to determine the low coolant temperature required for similarity.

The pressure coefficient for the coolant can also be written in the same form as equation (6):

$$\left(\frac{p}{\dot{w}\sqrt{T}}\right)_{c,L} = \left(\frac{p}{\dot{w}\sqrt{T}}\right)_{c,H} \tag{12}$$

However, since the coolant pressure is a dependent variable, this parameter is fixed once the low temperature level coolant flow and coolant temperature are established. Kinematic similarity for the coolant is dependent on the other similarity parameters of the two systems. A comparison of the pressure required for kinematic scaling and the pressure utilized will be given in the RESULTS AND DISCUSSION section (see table I).

The temperature difference ratio is used as the scaling parameter since this ratio does not change between the two gas temperature levels considered:

$$\varphi = \frac{T_{Ti} - T_{w,g}}{T_{Ti} - T_{ci}} \tag{13a}$$

Also,

$$\varphi = \frac{1}{1 + \frac{(hA)_g}{(hA)_c} + \frac{t(hA)_g}{(kA)_w}} \tag{13b}$$

The convection heat-transfer coefficients (gas and coolant side) can be defined by the Nusselt equation. For constant Reynolds and Prandtl numbers, equation (13b) can then be written as

$$\varphi = \frac{1}{1 + C\left[\frac{(\dot{w}/\mu)_g}{(\dot{w}/\mu)_c}\right]^n + B\left[\frac{(\dot{w}/\mu)_g^n}{k_w}\right]} \tag{13c}$$

The results of reference 3 show that, for gas pressures up to 31 newtons per square centimeter (45 psia) and gas temperatures from 700 to 1645 K (800° to 2500° F), the airfoil temperature data can be correlated by equation (13d) which neglects the term in equation (13c) representing conduction through the airfoil wall:

$$\varphi = \frac{1}{1 + C_1\left[\frac{(\dot{w}/\mu)_g}{(\dot{w}/\mu)_c}\right]^n} \tag{13d}$$

This form of the equation assumes that the temperature gradient through the wall is zero. (For high heat flux conditions, where the temperature gradients through the airfoil are significant, equation (13b) should be used.) The temperature difference ratio (eq. (13d)) is constant for different gas temperature levels provided the ratio of coolant to gas flow parameter does not vary and the constant C_1 does not change. The ratio of the flow parameters does remain constant between two temperature levels as shown by equation (10). Therefore, based on the Nusselt equation, the constant C_1 can be shown to be a function of the ratio of gas-to-coolant thermal conductivities only:

$$C_1 = f\left(\frac{k_{g,L}}{k_{c,L}}\right) = f\left(\frac{k_{g,H}}{k_{c,H}}\right) \tag{14a}$$

Also,

$$C_1 = f(k_L^*) = f(k_H^*) \tag{14b}$$

Therefore, the temperature difference ratio equality depends on equality between k_L^* and k_H^*.

The error associated with using the temperature difference ratio as a scaling parameter can be demonstrated by differentiating equation (13b) with respect to h_g/h_c and with respect to h_g/k_w. The resulting definitions of the change in the temperature difference ratio are shown in equations (15) and (16):

$$\Delta\varphi = \frac{-\frac{A_g}{A_c}\,\Delta\left(\frac{h_g}{h_c}\right)}{1 + \frac{(hA)_g}{(hA)_c} + \frac{t(hA)_g}{(kA)_w}} \tag{15}$$

$$\Delta\varphi = \frac{-(t)\,\frac{A_g}{A_w}\,\Delta\left(\frac{h_g}{k_w}\right)}{1 + \frac{(hA)_g}{(hA)_c} + \frac{t(hA)_g}{(kA)_w}} \tag{16}$$

Based on the dimensionless ratio of thermal conductivities k_H^*/k_L^* shown in table I and defined by equations (14a) and (14b), the ratio of gas to coolant heat-transfer coefficients h_g/h_c will be less than about 3 percent between high and low temperature conditions. The airfoil conduction term h_g/k_w will vary by 20 percent providing the thermal conductivity of the airfoil is a linear function of temperature and does not change significantly over the range of temperatures investigated (this h_g/k_w term is strongly dependent on the airfoil material thermal conductivity and temperature range). However, combining the two errors just discussed and assuming the area ratios are approximately 1.0 will cause the temperature difference ratio to change by about 0.03 at gas temperatures and pressures up to 1920 K (300° F) and 410 newtons per square centimeter

(600 psia), respectively. This is generally within the expected experimental accuracy.

The hot gas Reynolds number and pressure coefficient equalities are satisfied by properly varying the turbine inlet temperature, pressure, and flow rate. For a given set of high temperature operating conditions, the turbine inlet temperature and pressure, the coolant inlet temperature, and the coolant and gas flow rates are known. To scale to a lower turbine inlet temperature condition, the lower gas temperature is selected and, then, by varying the turbine inlet pressure and gas flow rate, the required gas density and velocity that satisfy the hot gas Reynolds and Mach numbers equalities are determined. The Reynolds number equality for the coolant is satisfied by maintaining a similar coolant to gas flow rate ratio $\dot{w}_c/\dot{w}_g$ and viscosity ratio μ_c/μ_g.

The local temperature difference ratio φ_x is known based on the measured low temperature data. Using this value and substituting the higher temperature case values of turbine inlet temperature T_{Ti} and coolant inlet temperature T_{ci} give a new value of local airfoil temperature:

$$(T_{w,x})_H = (T_{Ti})_H - (T_{Ti} - T_{ci})_H \varphi_x \tag{17}$$

An average airfoil surface temperature is found by a weighted sum of the local airfoil temperatures:

$$\frac{\sum (T_{w,x})_L \, dx}{L_S + L_P} \equiv \overline{T}_{w,L} = T_{Ti,L} - (T_{Ti} - T_{ci})_L \frac{\sum \varphi_x \, dx}{L_S + L_P} \tag{18}$$

The term $\sum \varphi_x dx/(L_S + L_P)$ is the average temperature difference ratio $\overline{\varphi}$. Consequently, the average airfoil temperature can be calculated by a method similar to the local airfoil temperature calculation:

$$(\overline{T}_w)_H = (T_{Ti})_H - (T_{Ti} - T_{ci})_H \overline{\varphi} \tag{19}$$

In summary, the requirements and assumptions for scaling are the following:

(1) Geometric similarity is satisfied.

(2) Reynolds number equality exists for both the gas and coolant.

(3) Pressure coefficient equality exists for both the gas and coolant.

(4) The Prandtl number does not vary significantly over the range of temperatures covered.

(5) The thermal conductivity of the airfoil material is a well-behaved function of temperature and does not change radically over the range of temperature considered.

(6) The dilution $\dot{w}_c/\dot{w}_g$ is constant for the gas temperature levels considered.
(7) The perfect gas law applies.

TEST PROCEDURE

An experimental investigation was made to test the feasibility of scaling low gas temperature heat-transfer data to high gas temperature conditions. Low gas temperature data and high gas temperature data for correspondingly scaled conditions were taken over a range of coolant to gas flow ratios. The high gas temperature conditions investigated were turbine inlet total temperatures and pressures of 1145 K (1600° F) and 37.2 newtons per square centimeter (54 psia), and 1645 K (2500° F) and 27.9 newtons per square centimeter (40.5 psia), respectively. The coolant inlet temperatures actually tested were 409 and 589 K (275° and 600° F). The pressure level for the 1145 K (1600° F) turbine inlet temperature point was double the scaled pressure level. The low temperature test conditions considered were turbine inlet total temperature and pressure of 811 K (1000° F) and 12.6 newtons per square centimeter (18.3 psia), respectively, and an actual inlet coolant temperature of 303 K (85° F). It is noted here that low and intermediate coolant temperatures actually tested were 16 K (30° F) below the desired coolant temperatures.

It was necessary to utilize two burners to obtain, in the same facility, the conditions just described - a low temperature burner for an 811 K (1000° F) gas temperature and a high temperature burner for 1145 K (1600° F) and 1645 K (2500° F) gas temperatures (discussed in the APPARATUS section).

The operating procedure for the cascade was essentially the same for the high and the low temperature burners. After burner ignition, the desired combustion gas flow, exit static to total pressure ratio, and pressure level were established by adjusting inlet and exhaust throttle valves while maintaining the desired total temperature by an automatic controller. For a given coolant temperature, the coolant flow was then varied in a step-wise fashion over a range of approximately 2 to 15 percent of the gas flow.

RESULTS AND DISCUSSION

Experimental verification of the scaling procedure is presented in the following discussion. The verification includes scaling the local airfoil temperatures defining different modes of coolant and, also, scaling the average airfoil temperature.

Figure 8 represents low airfoil temperatures that have been scaled to high airfoil temperatures and then compared with experimentally measured airfoil temperatures at

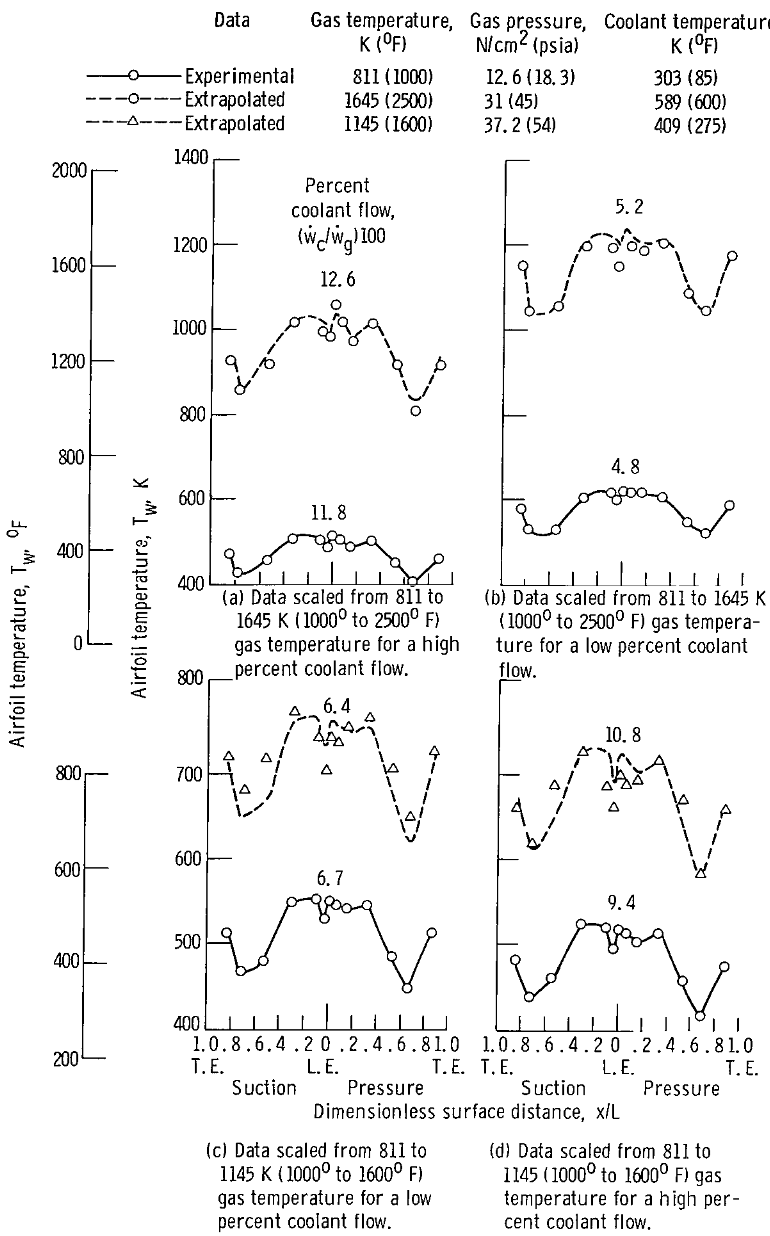


Figure 8. - Midspan temperature profile showing comparison of experimental and scaled temperatures for various coolant flows.

the higher temperature conditions. These measured airfoil temperatures have been corrected for radiation loss by the method discussed in reference 7. The comparison was made for two coolant flow rates and for turbine inlet temperatures of 1145 and 1645 K (1600° and 2500° F). The data in figure 8(a) are for approximately 12 percent coolant flow and the data in figure 8(b) are for approximately 5 percent coolant flow at turbine inlet temperatures of 811 and 1645 K (1000° and 2500° F). The low temperature experimental data are connected by a smoothly curved solid line. These data points were then scaled up to generate the dashed curves shown above the solid curves. Experimental data taken at similar high temperature conditions are superimposed on the dashed curves. The comparison of experimental and scaled temperatures is good with only a few exceptions. These exceptions are due in part to experimental error in measuring temperature and also in part to the slight difference in coolant flows. Figures 8(c) and (d) also show two coolant flows for turbine conditions of 811 and 1145 K (1000° and 1600° F). These data follow similar trends as those of figures 8(a) and (b). But since the 1145 K (1600° F) data were taken at the end of the test program (and after the thermocouples had deteriorated due to thermal cycling), there is a greater deviation between scaled and experimental values. Local data points fall on either side of the scaled value, however, suggesting the error involved is random in nature.

The way in which the ratio of gas to coolant thermal conductivities and the specific heat ratio vary from high to low temperature can affect the scaling procedure. Contained in table I are values of k_H^*/k_L^* and γ^* for the test conditions investigated. The parameter γ^* is a ratio of high to low gas temperature specific heat ratios. This parameter is important if an equality of Mach numbers is used to establish test conditions instead of an equality of the pressure coefficient. Ideally, these ratios should be unity; values other than unity are an indication of the error involved in the scaling procedure. As can be seen, the error increases for both ratios as the difference between high and low temperatures increases. The degree to which kinematic scaling is obtained on the coolant system is shown by the pressure ratios in table I. The scaled pressure ratio is found by using equation (10); the actual pressure ratio is a ratio of experimental coolant inlet pressure data.

A basis for the scaling procedure was that the values of φ for both high and low temperature conditions were identical. A check of this requirement is a good indication of how well this procedure will work in practice. Figure 9 is a plot of $\overline{\varphi}$ against $\dot{w}_c/\dot{w}_g$ for gas conditions of 811 K (1000° F) and 12.6 newtons per square centimeter (18.3 psia), 1145 K (1600° F) and 37.2 newtons per square centimeter (54 psia), and 1645 K (2500° F) and 31 newtons per square centimeter (45 psia) and for coolant temperatures of 303, 409, and 589 K (85°, 275°, and 600° F). A coolant temperature of 455 K (360° F) for the low turbine conditions is also shown. As indicated by figure 9, a good comparison exists for all conditions listed. The discussion of error in the ANALYSIS section in-

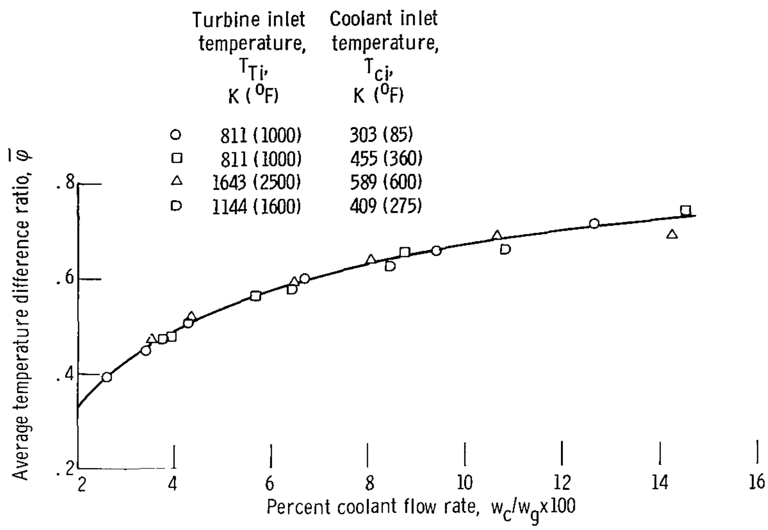


Figure 9. - Average temperature difference ratio for midspan as function of percent coolant flow rate.

dicated a possible variation of up to 0.03 in the value of φ between the high and low gas temperature data. It can be seen that all the data fall within this range. The data points representing turbine conditions of 1145 K (1600° F) fall somewhat below the curve formed by the other data. This is due to the deterioration of the thermocouples with some airfoil temperature readings much higher than expected while others are lower than expected.

Some deviations from the scaling conditions are noted. The first is the doubled pressure level associated with the 1145 K (1600° F) gas condition and another is the pressure associated with the 1645 K (2500° F) gas condition. The latter point was run at 31 newtons per square centimeter (45 psia) instead of 27.9 newtons per square centimeter (40.5 psia). However, the equality of pressure coefficients (Mach numbers) was maintained and only the equality of the Reynolds numbers was affected. Since at the low pressure levels investigated herein the gas side heat flux is low, the variation in pressure does not significantly affect the results. Also, the low and intermediate coolant temperatures were 16 K (30° F) below the desired values. The effect of these deviations on Reynolds numbers was approximately 6 percent and 2.5 percent, respectively. As noted in the discussion of figure 8, the coolant flows were not matched between the scaled airfoil temperatures and the equivalent high temperature data. However, reference to figure 9 shows that the coolant flow does not need to be matched as closely at high coolant flows as at lower coolant flows to obtain the same accuracy.

The following figures are included to demonstrate the ability to scale different cooling modes such as film cooling, impingement cooling, and convection cooling. Figures 10 and 11 were plotted for four thermocouple positions and show scaled tempera-

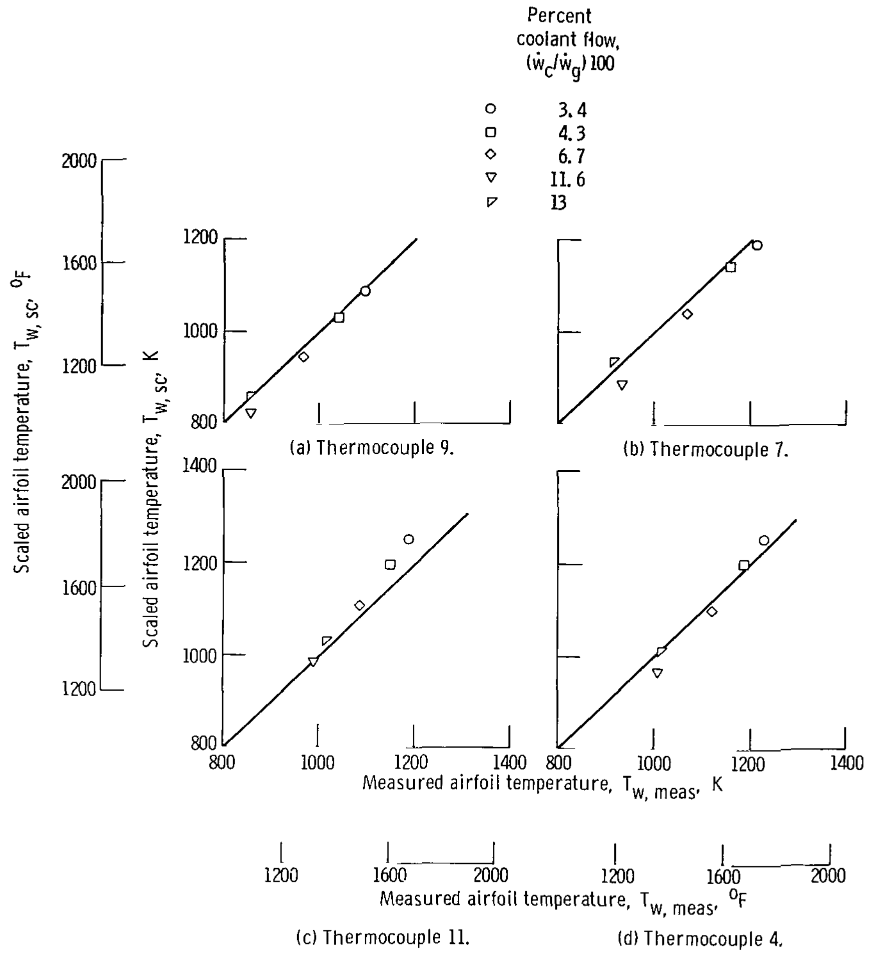
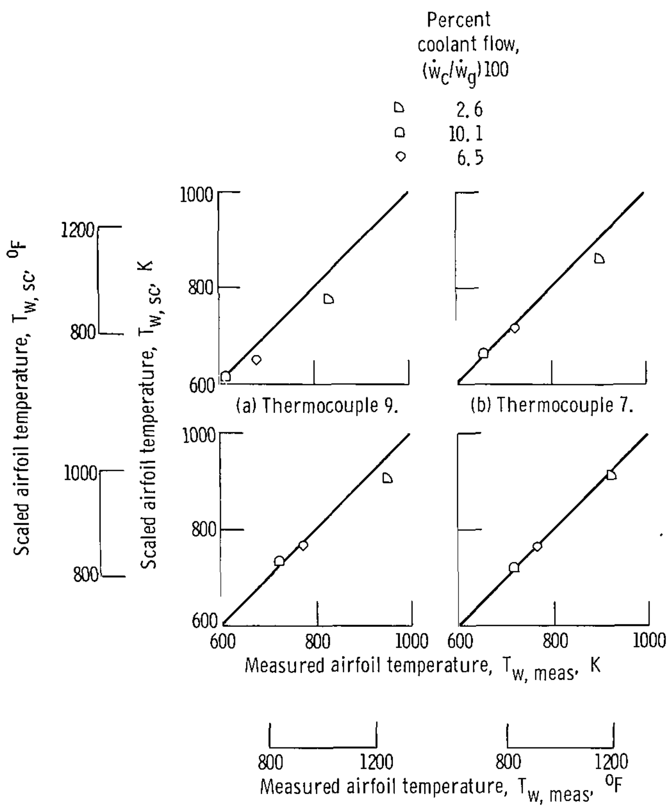


Figure 10. - Comparison of airfoil temperatures (scaled from low temperature data) with experimental airfoil temperatures at the high temperature conditions. Data are shown for 811 and 1645 K (1000° and 2500° F) gas temperatures.

tures as functions of measured temperatures for a range of coolant flows. Figures 10(a) to (d) compare 811 to 1645 K (1000° to 2500° F) turbine conditions while figures 11(a) to (d) compare 811 to 1145 K (1000° to 1600° F) turbine conditions. Thermocouple positions 9, 7, 11, and 4 are shown in figures 10 and 11.

These locations were cooled by impingement, convection, film convection, and impingement, respectively. All thermocouple position data presented in figures 10(a) to (d) show good agreement between measured and scaled temperature data except for number 11 (fig. 10(c)). Thermocouple 11 shows an increasing difference of temperatures with a decreasing coolant flow. This location is an impingement cooled point at the start of the midchord pressure side passage. However, the data of figures 11(a), (b), and (d) show a similar temperature difference for thermocouples 7, 9, and 11 as that shown for thermocouple 11 of figure 10(c). The maximum temperature difference shown in fig-



(c) Thermocouple 11. (d) Thermocouple 4.

Figure 11. - Comparison of airfoil temperatures (scaled from low temperature data) with experimental airfoil temperatures at high temperature conditions. Data are shown for 811 and 1145 K (1000° and 1600° F) gas temperatures.

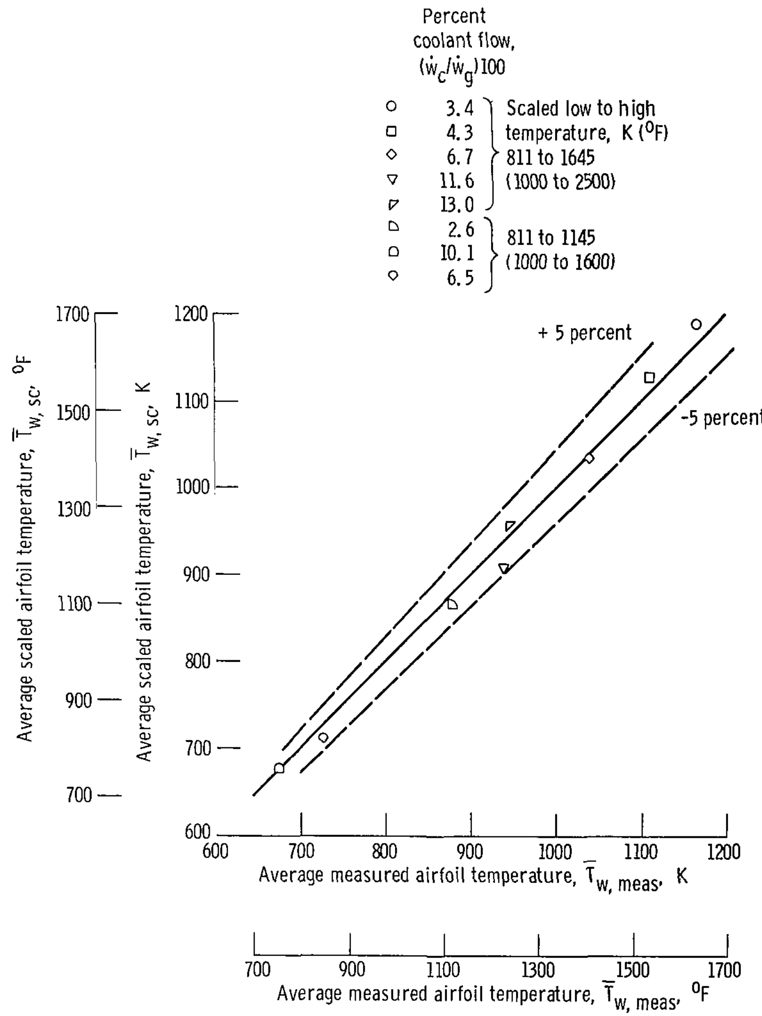


Figure 12. - Comparison of scaled and measured average midspan airfoil temperature for a range of coolant flows.

ures 10 and 11 is 60 K (108° F). The maximum error occurs at about 811 K (1000° F) for thermocouple 9 of figure 11(a) and is approximately 7.5 percent. The error for the other data shown is less than 5 percent.

How well the average temperature level is scaled is also of importance. A comparison of the scaled and experimentally determined average airfoil temperatures is shown in figure 12. These data agree quite well (within ±5 percent). However, as mentioned before, the average experimental temperatures for the 1145 K (1600° F) turbine condition were higher than the scaled values primarily due to the deterioration of the thermocouples over the duration of the test program.

CONCLUSIONS

The following conclusions can be made based on the scaling analysis and the results of the experimental verification:

1. Airfoil temperatures can be scaled from low to high gas temperature conditions provided the low temperature conditions are obtained from the high temperature conditions by methods of similarity.

2. Geometric, kinematic, dynamic, and thermal similarity are requirements which must be satisfied both in determining temperature level test conditions and in subsequent scaling of low temperature level experiment airfoil temperatures to the higher levels of temperature desired.

3. The scaling procedure developed herein was demonstrated to a gas temperature and pressure of 1645 K (2500° F) and 27.9 newtons per square centimeter (40.5 psia).

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, July 28, 1971,
720-03.

APPENDIX A

SYMBOLS

A	area
B, B_1, C, C_1	constants
C_p	specific heat at constant pressure
h	convection coefficient
k	thermal conductivity
L	length of vane suction or pressure surface
l	characteristic length
M	Mach number
n, m_1, m_2	exponents
P	pressure coefficient
Pr	Prandtl number
p	pressure
R	gas constant
Re	Reynolds number
T	temperature
t	wall thickness
V	velocity
$\dot{w}$	flow rate
x	distance along vane from stagnation point
γ	specific heat ratio
φ	temperature difference ratio
ρ	density
μ	viscosity

Subscripts:

c	coolant
ci	coolant inlet
g	gas
H	high temperature case
L	low temperature case
meas	measured
P	pressure surface
S	suction surface
sc	scaled
Ti	turbine inlet
w	airfoil
1-25	local thermocouple locations on the airfoil

Superscripts:

-	average
*	dimensionless ratio

APPENDIX B

NUMERICAL EXAMPLE

The following numerical example is developed by first selecting the high gas temperature level of interest and then scaling down to the low gas temperature at which data were taken. The high gas and coolant temperature conditions are:

Turbine inlet total temperature, K (^{o}F)	1645 (2500)
Turbine inlet total pressure, N/cm^2 (psia)	27.9 (40.5)
Coolant inlet temperature, K (^{o}F)	589 (600)
Coolant flow, $\dot{w}_c/\dot{w}_g$, percent	5
Mean trailing edge Mach number	0.85

The desired low gas temperature level at which data will be taken is 811 K (1000^{o} F). Equation (7) can be used to select the pressure which will satisfy kinematic and dynamic similarity (see ref. 8 for the properties):

$$p_{g,L} = 27.9\ \frac{N}{cm^2}\left(\frac{3.64(10^{-4})\ \dfrac{g}{(cm)(sec)}}{5.66(10^{-4})\ \dfrac{g}{(cm)(sec)}}\right)\sqrt{\frac{811\ K}{1645\ K}} = 12.6\ N/cm^2$$

The gas flow can also be scaled by equation (5), however, by setting low gas temperature and pressure. When the static to total pressure ratio at the trailing edge is maintained for a Mach number of 0.85, the gas flow becomes a dependent variable. The scaled coolant temperature can be determined by using equation (16) and by assuming the dilution ratio is constant between the two temperature levels:

$$\mu_{c,L} = 2.93(10^{-4})\ \frac{g}{(cm)(sec)}\ \frac{3.64(10^{-4})\ \dfrac{g}{(cm)(sec)}}{5.66(10^{-4})\ \dfrac{g}{(cm)(sec)}} = 1.88(10^{-4})\ g/(cm)(sec)$$

For a viscosity of 1.88×10^{-4} g/(cm)(sec) the coolant temperature is 319 K (115^{o} F). The coolant pressure can be determined by equation (10); however, since five of the six variables have been determined, the coolant pressure becomes a dependent variable. The low gas and coolant temperature conditions are the following:

Turbine inlet temperature, K (^{o}F)	811 (1000)
Turbine inlet pressure, N/cm^2 (psia)	12.6 (18.3)
Coolant inlet temperature, K (^{o}F)	319 (115)
Coolant flow, $\dot{w}_c/\dot{w}_g$, percent	5
Mean trailing edge Mach number	0.85

Local airfoil temperature data taken at the previous condition can be scaled up to the higher gas temperature level by using equation (17).

REFERENCES

1. 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.

2. Clark, John S.; Richards, Hadley T.; Poferl, David J.; and Livingood, John J. B.: Coolant Pressure and Flow Distribution Through an Air-Cooled Vane for a High-Temperature Gas Turbine. NASA TM X-2028, 1970.

3. Gladden, Herbert J.; Gauntner, Daniel J.; and Livingood, John N. B.: Analysis of Heat-Transfer Tests of an Impingement-Convection- and Film-Cooled Vane in a Cascade. NASA TM X-2376, 1971.

4. Crowl, Robert J.; and Gladden, Herbert J.: Methods and Procedures for Evaluating, Forming, and Installing Small-Diameter Sheathed Thermocouple Wire and Sheathed Thermocouples. NASA TM X-2377, 1971.

5. Schlichting, Hermann (J. Kestin, trans.): Boundary Layer Theory. Sixth ed., McGraw-Hill Book Co., Inc., 1968, p. 67.

6. Gladden, Herbert J.; Dengler, Robert P.; Evans, David G.; Hippensteele, Steven A.: Aerodynamic Investigation of Four-Vane Cascade Designed for Turbine Cooling Studies. NASA TM X-1954, 1970.

7. Gladden, Herbert J.; Hippensteele, Steven A.; Hickel, Robert O.; and Dengler, Robert P.: Radiation Heat Transfer Characteristics of Turbine Vane Airfoils in a Water-Cooled Cascade. NASA TM X-2203, 1971.

8. Poferl, David J.; Svehla, Robert A.; and Lewandowski, Kenneth: Thermodynamic and Transport Properties of Air and the Combustion Products of Natural Gas and of ASTM-A1 Fuel with Air. NASA TN D-5452, 1969.

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