LAMINATED TURBINE VANE DESIGN
AND FABRICATION

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UNITED TECHNOLOGIES CORPORATION
PRATT & WHITNEY AIRCRAFT GROUP
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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

NASA Lewis Research Center
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A turbine vane and associated endwalls were designed for advanced engine conditions supplied by NASA. The advanced vane design investigation resulted in an effectively cooled scheme that combined the cooling methods of convection and full coverage film cooling. The design incorporated the cooling benefits available through the utilization of Pratt & Whitney Aircraft's unique wafer fabrication concept. Upon completion of the design task, two vanes and associated endwalls of the final configuration were fabricated, cold flowed, and shipped to NASA for experimental evaluation in their high temperature one-vane test tunnel.
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I. SUMMARY

A design and fabrication effort was conducted using the Pratt & Whitney Aircraft Group (P&W) wafer fabrication concept for an advanced engine turbine vane and associated endwalls. The construction concept involves the fabrication of airfoils from a stack of laminates (wafers) which have the desired cooling passage geometry photo-etched on the surface. The wafers are then diffusion-bonded and the final airfoil shape is machined from the bonded block to produce the finished turbine vane with integral cooling passages.

The advanced vane design combined the methods of convection cooling and selective areas of full coverage film cooling. The full coverage film cooling technique was utilized on the leading edge, pressure side, and endwall regions. It was not employed on the suction side of the vane due to the adverse aerodynamic penalties associated with film cooling in that region because of the relatively high main stream velocity.

The predicted surface temperature profile at the design point for the midspan section of the vane indicated a maximum temperature of 1059°C (1939°F). This temperature is below the design life requirement upper limit of 1079°C (1975°F) and results in oxidation/erosion life of approximately 245 hr based on the life data for the vane material — MAR-M 200 + Hf (PWA 1422). The only thermal transient condition defined was ten thermal cycles of the vane between design conditions and ambient conditions. The predicted maximum strain range was 0.32% which corresponds to a pseudo-cyclic life in excess of \(10^5\) cycles based on the fatigue life data for PWA 1422.

Upon completion of the design task, two vanes and associated endwalls were fabricated to the final configuration. Before shipment to NASA both vanes and associated endwalls were cold air flow calibrated to determine the cold flow characteristics. The cold flow results for the two vane airfoils indicated excellent agreement between the prediction and the experimental data. The flow characteristics of the endwalls were also determined for both vanes; however, the experimental data indicated a severe underflowing condition when compared to the prediction. A portion of this is attributed to a number of endwall passages that did not open up during final machining operations. It is recommended that the endwalls be flow-checked again before hot cascade tests are conducted to define flow conditions required for the hot tests.
II. INTRODUCTION

Turbine inlet temperatures and pressures for advanced gas turbine engines have progressed to levels where convection cooling schemes are inadequate. To maintain reasonable wall temperatures in advanced turbines, more sophisticated cooling schemes are required. A method of approach, which is relatively efficient, is full-coverage film cooling combined with an effective convection cooling scheme. One method to accomplish this is to perforate a cast hollow airfoil with hundreds of small holes. This method is expensive and is hindered by fabrication and structural limitations on the minimum size and shape of the holes. This limits the effectiveness of these cooling schemes. A potential solution to this problem is to construct the airfoil of horizontally or radially stacked wafers with the desired cooling passage geometry photo-etched on the surface. This affords more flexibility in passage size and shape which can result in improved cooling effectiveness.

The vane designed and constructed for this program utilized this wafer fabrication concept. This concept involves the fabrication of airfoils from a stack of wafers. Each wafer has the desired cooling passage geometry photo-etched in its surface. The wafers are then diffusion-bonded together using Transient Liquid Phase (TLPTM*) bonding. The final airfoil endwalls, and internal plenum are machined from the bonded block to produce the finished turbine blade or vane with integral cooling passages of any desired complexity.

In turbine vane applications, the wafers can be oriented either radially or horizontally. For turbine blade applications, the wafers are oriented in a radial direction so the high centrifugal stresses are carried by the parent material wafers instead of the bond joints. A radial orientation was also selected for this vane design because it is more amenable to full (100%) coverage film cooling and corresponds to recent P&WA efforts. This orientation also results in a structurally stronger airfoil and simpler construction, since the endwall cooling design can be included in the same wafers that form the airfoil.

The vane design parameters and external aerodynamic profile were supplied by NASA Lewis Research Center (LeRC). The cooling design and analysis of the vane and endwalls were made by several computerized analytical procedures, programmed for use on an IBM 370 digital computer by the Pratt & Whitney Aircraft Group Government Products Division (P&WA/GPD). The analysis consists of programs for the computation of airfoil external heat transfer coefficients, adiabatic wall temperatures for both non-film cooled and film cooled conditions, internal convective heat transfer coefficients, internal coolant pressure loss and temperature rise, and metal temperature and stress-strain distributions for both transient and steady-state conditions.

Two vanes incorporating the final cooling design were fabricated by photo etching the design cooling configuration on radially oriented wafers, bonding the wafers to form a block, and then electro-discharge machining the block to the final airfoil-endwall configuration.

*TLP bonding is a P&WA process for joining superalloys. The process achieves near parent metal strength but requires only moderate bonding pressures of 15 to 20 psi, compared to conventional diffusion bonding pressures, which are generally in excess of 2000 psi.
III. DESIGN CONDITIONS

The design conditions selected were representative of the first-stage turbine stator of an advanced commercial, energy-efficient engine. In addition, the performance of the final vane design was evaluated at selected operating conditions of the NASA “One Vane Tunnel” which are representative of an advanced high temperature engine. These conditions are:

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<th>NASA “One Vane Tunnel” (Off-Design Conditions)</th>
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Figure 1 shows the vane external profile and coordinates which were defined by NASA LeRC. The critical velocity ratio distribution for this airfoil profile is shown in figure 2 and is the result of both an analytical and an experimental study at NASA LeRC (ref. 1). Because of the anticipated gas stream flow characteristics in the “One Vane Tunnel,” a flat gas temperature profile was assumed to exist at the inlet to the turbine vane.

During the initial phase of the design effort, a coolant inlet temperature of 704°C (1300°F) was specified. It became apparent, however, that this was too severe to result in an acceptable cooling scheme without utilizing film cooling in the high Mach number recovery region of the suction surface. Discussions with the NASA Program Manager indicated that the aerodynamic penalties of film cooling the suction surface were unacceptable and that a convection cooling scheme should be used. A relaxation of the coolant flowrate goal of 5% and a reduction in coolant inlet temperature to 649°C (1200°F) were permitted to facilitate an acceptable suction surface cooling scheme without film cooling. For both conditions the coolant supply pressure was assumed equal to approximately 1.05 times the inlet gas stream pressure.
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**Figure 1. NASA Vane External Profile**
Figure 2. NASA Vane Critical Velocity Ratio
IV. DESIGN DESCRIPTION

A. GENERAL

The objective of the thermal analysis was to establish an effectively cooled turbine vane airfoil and endwalls for the stated design conditions using the radial wafer concept. After a given design was established, a detailed thermal analysis of the entire structure was performed. The P&WA/GPD thermal and durability analyses are composed of several computerized analytical procedures programmed for use on an IBM 370 digital computer. These analyses consist of programs for the computation of airfoil external heat transfer coefficients, adiabatic wall temperatures for both film cooled and nonfilm cooled airfoils, internal convective heat transfer coefficients, internal coolant pressure loss and temperature rise, metal temperature distribution, stress-strain relationship, and the creep-rupture life analysis. Analyses are capable of handling both transient and steady-state conditions.

The design procedure utilized for the endwall analysis was somewhat less rigorous than the vane airfoil analysis. Because the flow field in the endwall region is highly three dimensional, a detailed design procedure equivalent to the two dimensional procedure for the airfoil has not presently been established. Several endwall investigations have been conducted and are currently being conducted both within P&WA and by other companies or government agencies. The empirical results of the P&WA investigations have been utilized to assist in defining endwall cooling design.

B. THERMAL ANALYSIS

The external heat transfer coefficients were obtained using the P&WA-developed computational procedure (ref. 2). In this boundary layer program a general finite-difference procedure for computing the behavior of compressible two-dimensional boundary layers is utilized together with a turbulence model which allows quantitative predictions of the location and extent of the transition region between laminar and turbulent flow as it is influenced by such disturbances as surface roughness and free-stream turbulence. Reverse transition, i.e., relaminarization, caused by large favorable streamwise accelerations, is also quantitatively predicted by this procedure. The solution procedure depends upon the calculation of the streamwise development of a turbulent mixing length whose magnitude is governed by the turbulence kinetic energy equation. A large number of comparisons between predictions and measurements have been made with this program and, in general, very good agreement is obtained.

A separate film cooling program is used to calculate film temperatures for airfoils with showerhead and/or aft section film cooling. Figure 3 presents the model used for the definition of heat transfer and film effectiveness. The film temperatures are based on correlations between cascade test data and an empirical model for multihole cooling. This is further supplemented by the basic wafer slot film effectiveness tests conducted at the United Technologies Research Center. Film temperatures along the airfoil surface are calculated using a driving potential defined as the difference between the coolant ejection temperature and the mainstream gas temperature. Figures 4 and 5 present the film effectiveness curves for the showerhead and pressure side sections, respectively.
Figure 3. Heat Transfer and Film Effectiveness Model
Figure 4. Leading Edge Film Effectiveness
Figure 5. Pressure Side Film Effectiveness
The airfoil coolant-side analysis provides detailed heat transfer, pressure drop, and flow distribution information. This is accomplished by the analysis of compressible flow through multiple channels with multiple exits and variable flow conditions. This analysis is a general purpose computer program that accounts for convective heat transfer, rotational effects on total temperature and pressure, pressure losses due to bends and sudden changes in cross-sectional area, and distribution of airflow among the multiple passages. The internal heat transfer coefficients for the leading edge, pressure side, and suction side were obtained using the Colburn turbulent flow equation, (ref. 3). For the trailing edge section, the coefficients were defined based on tests conducted by P&WA for the wavy crisscross pattern. The computer solution for the flow distribution was further adjusted by the experimental cold flow results from several P&WA radial wafer airfoil designs and the AFAPI radial wafer vane program (ref. 4). The modifications consisted of defining a friction factor for the etched passages based on the previous experimental results.

The predicted cold flow characteristics for the vane and the end-walls are also determined using the experimentally adjusted computer deck. For the cold flow study, however, a static discharge pressure of ambient is assumed, heat transfer is eliminated, and a coolant temperature of 21°C (70°F) is utilized.

A generalized heat transfer program is used to determine the airfoil transient or steady-state metal temperature distribution after defining the flow and heat transfer characteristics for both the external and internal surfaces. Initially the airfoil is broken up into sufficient elements or nodes to define the temperature distribution. In addition to conduction and convection calculations, this program accounts for radiation, internal heat generation and heat storage. Provisions are available for specifying thermal variations in properties such as specific heat, conductivity, and heat transfer coefficients, and time dependent variations in film coefficients and fluid temperatures which result in a direct solution for temperatures. This analysis is readily applicable for evaluation of convection, film, and transpiration cooling, or any combination of the three.

C. STRESS ANALYSIS

A transient cycle was not defined for the vane design other than the capability of the vane to withstand cascade testing and ten cycles from the design condition to shutdown where the vane will reach ambient temperatures. Assuming the NASA cascade shutdown cycle is a gradual decrease in conditions and not an abort type of shutdown, steady-state conditions were utilized to determine the durability of the vane.

The object of the durability analysis is to calculate elastic thermal stresses in a body or irregular shape such as turbine vane or blade which is subject to a non-uniform temperature distribution. The analysis is programmed into a computer deck which operates in conjunction with the generalized heat transfer computer deck. For durability analysis the identical nodal breakup defined for the heat transfer solution was utilized with the main input, temperature distribution being supplied from the heat transfer deck. Assuming no external restraint will prevent the body from expanding freely, a system of simultaneous equations is developed that represents the internal interference of one fiber or node of the
body with another. These equations must satisfy the following conditions: (1) the sum of the internal interference forces must be zero; (2) the moment of internal stresses about any two arbitrarily chosen axes that are mutually perpendicular must also be zero. The following assumptions are also inherent to the analysis:

a. Elastic behavior of the body is assumed

b. The body cross section remains plane even after heating and subsequent elongation and bending

c. The cross section is not near the ends of the body so that boundary effects need not be included

d. The elements of the body are considered thin-walled so that stresses perpendicular to the wall of the body can be neglected (i.e., Poisson Ratio effects are non-existent).

The durability analysis included: (1) low-cycle fatigue life, (2) creep-stress rupture life, (3) vane suction side bulging study, and (4) oxidation/erosion life.

D. MATERIAL SECTION

The material selected for the vane and endwalls was Mar-M 200 +IIf (PWA 1422). This same material is currently used by P&WA in the first stage turbines for both the government (F100) and commercial (JT9D) engine applications. Although other advanced materials such as single crystal and Ti-nickel were considered, lack of etching and bonding experience for the single crystal material and unacceptably high oxidation characteristics of the Ti-nickel eliminated these two materials. Also due to the high utilization of PWA 1422 in P&WA advanced engines, detailed material characterization work, and the fact that one radial wafer airfoil had been fabricated with this material, it was deemed the best material available for the design.
V. RESULTS OF DESIGN

A. VANE DESIGN THERMAL RESULTS

The design study was initiated with the determination of the external heat transfer coefficients as discussed previously in Section IV. The coefficients for both the design and off-design points are shown in figure 6. The vane external contour and pressure profile utilized to define the coefficients were presented previously in Section III.

The final design is separated into four sections that consist of the leading edge, pressure side, suction side, and trailing edge. The predicted internal coolant flow distribution for each section is shown in figure 7 for the design conditions. The total coolant flow to the vane is 6.17% of the gas stream flow.

The predicted cold flow parameters for the total vane, endwalls, and each of the four sections are presented in figures 8 through 13. These cold flow parameters for the vane, endwalls, and each of the four sections of the vane were determined analytically utilizing the airfoil coolant-side analysis deck. The cooling flow circuits used in the analysis for the vane and endwall are presented in figure 14. The curves were generated by varying the coolant flowrate over a range consistent with the particular section in question and solving for the corresponding supply pressure while assuming a coolant temperature of 21°C (70°F) and discharging to a constant ambient pressure of 10.1 X 10^5 N/M^2 (14.7 psia). The design point pressure ratio included on figures 10 through 13 refer to the coolant supply total pressure divided by the average discharge static pressure at the design point conditions. This parameter is omitted from figures 8 and 9 due to the wide range in discharge static pressures associated with the entire vane and endwalls.

The final passage dimensions required to obtain the desired coolant flow split are shown in figure 15. Included on the figure are the suction side and pressure side wall thicknesses. The transition in thickness from suction side to pressure side occurs gradually through the leading edge showerhead section. A detailed description of each of the four sections is presented in the following paragraphs.

The leading-edge section consists of a ten row showerhead array with nine etched rows and one electrical discharge machined (EDM) row. The EDM row is located near the stagnation region and is required to provide film cooling where no wafer interface exists. The etched rows are split into three rows on the pressure side and six rows on the suction side. The increased number of rows on the leading-edge suction side is required to provide sufficient cooling effectiveness in the downstream high Mach number region to assist in cooling the suction side. As stated previously, ejection of film cooling air in this region was eliminated because of the high aerodynamic penalties.
Figure 6. NASA Radial Wafer Vane Heat Transfer Coefficient Distribution
Figure 7. NASA Radial Wafer Vane Cooling Design
Figure 8. Predicted Cold Flow for NASA Radial Wafer Vane
Figure 9. Predicted Cold Flow for NASA Radial Wafer Vane - Endwall
Figure 10. Predicted Cold Flow for NASA Radial Wafer Vane - Leading Edge Section
Figure 11. Predicted Cold Flow for NASA Radial Wafer Vane - Pressure Side
Figure 12. Predicted Cold Flow for NASA Radial Wafer Vane - Suction Side
Figure 13. Predicted Cold Flow for NASA Radial Wafer Vane - Trailing Edge
Figure 14. Coolant Flow Circuits for Vane and Endwall
Film Cooling Holes
0.038 (0.015) dia Hole
at 30° 21 or 22/Row
Typ 6 Rows

Radial Convective Passages
0.030 (0.012) X 0.094 (0.037)
Passage

Film Cooling Holes
0.061 (0.024) dia
at 6.3° 3 or 4/Row
Typ 3 Rows

Film Cooling Holes
0.061 (0.024) dia Semi-circle
at 6.3° 3 or 4/Row
Typ 18 Rows

Film Cooling Holes
0.061 (0.024) dia Semi-circle
at 16.3° or 10/Row
Typ 3 Rows

Radial Convective Passages
0.061 (0.024) X 0.094 (0.037)
Passages Typ 23 Rows

Convective Passage
0.152 (0.060) X 0.051 (0.020)
Wavy Crisscross Typ 15 Places

Film Cooling Holes
0.061 (0.024) dia Semi-circle
at 6.3° 3/Row

Note: Dimensions - cm (in.)

Figure 15. NASA Radial Wafer Vane Cooling Geometry
Radial film cooling injection angles rather than normal injection angles were used on the leading edge for two reasons. The first reason is to increase the coolant side convective heat transfer. A radially angled film cooling hole will increase the passage length through the wall relative to film cooling hole with a normal angle. This increase in length will increase the coolant side convective heat transfer area and thereby increase the convective heat transfer. The second reason for angling the hole is to increase the film effectiveness. An angled passage results in less penetration and mixing of the coolant jet into the mainstream than a normal passage based on tests conducted by P&WA. In addition, angling the hole also results in the breakout area of the passage on the surface of the airfoil being greater than a normal passage. Both effects result in an increase in film effectiveness when compared to a normal passage. The film passages are also staggered from row to row to provide full coverage film cooling.

The pressure side film cooling design is also a radially angled passage - staggered row cooling scheme. As illustrated in figure 15, the passages are etched into one side of the wafer except for the three rows preceding the last row. In this region the wafers are etched on both sides to double the flow area which will increase the flowrate in that region and increase the film effectiveness. This was done to assist in cooling the convectively cooled trailing edge section.

The suction side utilizes a simple oval shape radial convective passage. The coolant is used first to cool the suction side then collected in a tip plenum for additional cooling of the blade rub strip or the platform. Double use of the coolant in this manner is a potential means to increase engine cycle efficiency by reducing the total amount of coolant required.

The cooling design for the trailing edge section is the wavy crisscross slot geometry. The design is an extremely efficient convective cooling technique which was used on the AFAPL radial wafer vane design (ref. 4). Because the wafers which form the trailing edge passage are bonded in the plane of the trailing edge slot, each wafer has half the passage geometry. The passage for each half is sinusoidal and the two sine waves are 180 deg out of phase with respect to each other. Therefore, they are continually crossing one another along the trailing edge slot.

The vane nodal breakup is shown in figure 16. This breakup is relatively detailed in that four temperatures are defined to supply the temperature gradient through the wall. Figure 17 shows the predicted film temperature distribution for the design and off-design points.

The predicted vane outside wall temperature profiles for the design point and the off-design point are shown in figures 18 and 19, respectively. A tabulation of the temperature and surface distances for the two points is included in Appendix A. For the design point the maximum (hot spot) temperature was 1059°C (1939°F) on the suction side near the trailing edge. The temperature is below the design life requirement of 1079°C (1975°F). The figures also indicate an overcooled section of the suction side near the leading edge. This was caused by discharging a relatively large amount of cooling air in this region to assist in cooling the suction side of the vane. This extra cooling was required because film cooling along the suction side was not desired due to the aerodynamic penalties associated with film cooling in a relatively high velocity area.
Figure 16. Detailed Nodal Breakup of the Radial Wafer Vane

Note:
Nodes 1138 Internal Nodes
Nodes 139-207 External Surface Nodes
Nodes 208-276 Internal Surface Nodes
Figure 17. NASA Radial Wafer Vane External Film Temperature Distribution
Figure 18. NASA Radial Wafer Vane - Temperature vs Surface Distribution for Design Point
Figure 19. NASA Radial Wafer Vane - Temperature vs Surface Distribution for Off-Design Point
B. VANE DESIGN DURABILITY RESULTS

1 Low Cycle Fatigue Life

Low cycle fatigue (LCF) life is normally determined based on the percent strain range developed during a transient cycle such as idle to sea level take-off. For this program, however, no transient cycle was defined since the design will be evaluated in a steady-state heat transfer cascade rig. It was therefore decided to base the LCF life on the percent strain range developed at the two design points.

Although no LCF tests have been performed on etched and bonded wafer passages with PWA 1422 material, tests have been conducted with PWA 1401 material and astrology. Results from these tests have indicated the wafer samples exhibited LCF characteristics equal to that of the solid material. Because of the results obtained with the other materials and since LCF testing of wafered PWA 1422 material was beyond the scope of this program, the LCF life curve for solid PWA 1422 material was utilized.

Maximum strain ranges of 0.32 and 0.46% were developed for the design and off-design points respectively. This results in predicted cyclic life of approximately $10^5$ and $10^4$ cycles as shown by the LCF life curve for PWA 1422 presented in figure 20. The location of the maximum strain range node for both design points is on the pressure side near the trailing edge as shown in figure 21 as well as some typical values around the vane contour.

2. Creep-Stress Rupture Life

The creep-stress rupture life analysis was conducted with the steady state relaxation computer program. Due to the extremely large amount of computer time required to run this deck to completion, the deck was run to determine approximately one-third of the design life. The remaining life was defined utilizing a logarithmic extrapolation method. Analysis resulted in a total life prediction of greater than 4000 hr at the design point and 500 hr at the off-design point.

3. Vane Suction Side Bulging Study

The bulging study was conducted on the vane suction side near the trailing edge which was the location where rupture failure was predicted to occur based on the creep-stress rupture analysis. The study was conducted assuming a wall thickness of 0.05 cm (0.02 in.), which is the minimum thickness between the outside wall and the convective passage. The maximum wall thickness is approximately 0.19 cm (0.077 in.); therefore, assuming an effective wall thickness of 0.05 cm should be a conservative approach. The 1.0% creep material property was utilized and the bulging life was predicted based on an empirical correlation resulting from a P&WA experimental investigation. The calculation indicated a creep-stress rupture life of 70 and 20 hr for the design and off-design points, respectively. The off-design life of 20 hr is relatively short which is a result of the high temperature, 1124°C (2056°F), and thin wall assumption associated with that location on the vane.
Figure 20. Low Cycle Fatigue Life for PWA 1422
Figure 21. NASA Wafer Vane Design Point Local Stress/Strain Distribution
4. Oxidation/Erosion Life

The oxidation/erosion life evaluation was based on the bare Mar-M-200 + Hf material properties shown in figure 22. A design point life of 245 hr and an off-design point life of 78 hr were determined for the predicted hot spot metal temperature of 1060 and 1124°C (1940 and 2056°F). These predicted lives are sufficient to satisfy a typical cascade test program; however, for engine application a coating is recommended to prolong the airfoil life.

C. ENDWALL DESIGN RESULTS

The endwall cooling analysis was less detailed than that conducted for the vane, as explained previously in Section IV. For this area high coverage film cooling was utilized as the cooling scheme due to the severe gas environment conditions which were identical to the vane design conditions. Normally the endwall gas stream temperature is reduced from the vane hot spot temperature due to an inlet gas stream profile consisting of a hot temperature near midspan and cooler temperature at the endwalls. For the NASA cascade, however, the profile was defined as flat.

The cooling design geometry for the endwall is presented in figure 23. The total amount of coolant flowrate is 1.72% for each end-wall. The cooling scheme is a mixture of etched film holes and EDM film holes. The reason for the EDM holes is that in the areas where wafers were not required for the vane cooling a solid end block was used. Wafers could be used in these areas but for this program it was not necessary and endblocks were used to reduce the total number of parts and costs.

The endwall ISO-BAR distribution, shown in figure 24, was obtained through the use of the given endwall geometry and the inlet and exit conditions. Knowing the average static pressure distribution and streamline length, the external heat transfer coefficients were calculated using the turbulent flat plate correlation. This calculated value was adjusted based on the endwall experimental investigation results obtained from tests conducted within P&WA. The endwall nodal breakup, external heat transfer coefficients, and the hand calculated one-dimensional steady-state temperature distributions for the design and off-design points are presented in figures 25 and 26, respectively. The design point maximum metal temperature is 1070°C (1959°F). This temperature is approximately equal to the maximum metal temperature predicted for the vane surface. Since no durability analysis was conducted on the endwall, the life characteristics were not defined; however, the life is assumed approximately equal to that predicted with the vane since the maximum metal temperature is approximately the same.
Figure 22. Oxidation/Erosion Life Prediction for PWA 1422
Figure 23. NASA Wafer Vane Endwall Cooling Geometry
Figure 24. Endwall Iso-bar Line
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<th>$T_{surface}$ (°C)</th>
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$T_g = 1538°C (2800°F)$

$T_c = 649°C (1200°F)$

$P_g = 4.136 \times 10^6$ N/M$^2$ (600 psia)

$C/A = 1.72\%$

Figure 25. Endwall Steady State Temperature Distribution for Design Point
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Figure 26. Endwall Steady State Temperature Distribution for Off-Design Point
VI. FABRICATION DESCRIPTION

The cooling design was incorporated into two cascade test vanes and associated endwalls using the radial wafer fabrication technique. The vanes are a constant cross-section design with a 3.81 cm (1.50 in.) span height. The wafers were etched utilizing an electrochemical process. The etching process was developed satisfactorily at P&WA/GPD Materials Laboratory; however, during the sample wafer etching work at the selected etching vendor facility, certain deviations were observed. The etching results were not acceptable due to resist coating (protective layer which prohibits etching) breakdown and excessive groove width and depth variation as indicated in figures 27 and 28.

The coating breakdown results in excessive pitting around the coolant passages. In wafer airfoil construction this may result in passages being exposed to the mainstream in regions not desired after the final machining operation. The excessive groove width and depth variation is not acceptable since an accurate definition of flow distribution in varying passages is impractical.

The etching system parameters between the laboratory and the vendor were checked for deviations and two parameters, electrolyte temperature and resist coating application, were found to vary. A study was conducted in the P&WA/GPD laboratory varying the electrolyte temperature between 7 and 41°C (45 and 106°F). Figure 29 presents the results of the study, indicating the cooler the electrolyte solution the better the etched groove definition. Therefore, an ice cooled bath to control electrolyte solution temperature was installed at the vendor. The method of applying the resist coating was switched from dipping the wafers in liquid to a thin-film form. The thin-film form of resist application was obtained by passing the wafer between two rollers that deposited a thin sheet of the resist material on each side of the wafer. This thin-film form of application resulted in an even resist coating and satisfactory etching results.

The vane airfoil and associated endwalls cooling scheme was fabricated utilizing thirty-three wafers and four endblocks. The wafer and endblock orientation is shown in figure 30. The wafers and endblocks that formed the leading edge showerhead and the trailing edge wavy crisscross slot were fabricated first. These were bonded into subassemblies and machined before being bonded with the remaining wafers to form the complete assemblies from which the cascade vanes were machined. The wafers and end blocks that form the leading edge and trailing subassemblies for one of the cascade vanes are shown in figures 31 and 32, respectively. The wafers which made up the main body of the vane, in addition to the leading and trailing edge bonded subassemblies, are shown in figure 33. One of the complete bonded block assemblies is shown in figure 34.

During the first bond cycle for the complete assembly, the load became misaligned and a slight fanning of the wafers was observed on the suction side. This produced some unbonded areas which were not removed during the final machining operation. Before initiating the second bond cycle, the bonding fixture was modified to eliminate the misalignment problem observed with the first block. The fixture modification was successful, as the second bonded block did not exhibit the fanning phenomenon observed with the first block. The completed cascade vane was obtained by electrical discharge machining the internal cavities and external contour.
Figure 27. Etched Midchord Wafer Showing Resist Coating Breakdown and Passage Taper
Figure 28. Etched Midchord Wafer Showing Passage Size Variation
Figure 29. Effects of Solution Temperature On Line Width During Electrochemical Photoengraving
Figure 30. NASA Vane Wafer and Endblock Orientation
Figure 31. Etched Wafers and Blocks for NASA Wafer Vane Leading Edge Section
Figure 32. Etched Wafers and Blocks for NASA Wafer Vane Trailing Edge Section
Figure 33. Leading Edge Block, Trailing Edge Block and Midchord Wafers for NASA Wafer Vane
Leading Edge Subassembly
Midchord Wafers
Trailing Edge Subassembly
Locating Pin

Figure 34. NASA Wafer Vane Bonded Assembly
During final machining of the suction side external contour, it was discovered that the etching pattern for the first wafer aft of the leading edge section was mislocated. This resulted in a slot running the full span length as shown in figure 35. Since this was located in a cool portion of the vane and was only one-half a passage, the result of the reduction in cooling air was assumed insignificant. In addition, since one of the four thermocouple slots was originally planned near that position, it was decided to utilize the slot for a thermocouple location. In addition, some of the endwall film cooling holes did not open fully when the airfoil was EDM'ed to the specified span of 3.81 cm (1.500 in.). An additional 0.076 cm (0.030 in.) is required to open these holes. The 3.81 cm (1.500 in.) span was retained since the additional EDM work could be done at a later date if fully opened film cooling holes are required.

The final steps in the fabrication process were the machining of the showerhead and platform film cooling holes and the thermocouple instrumentation slots. The two completed vanes and associated endwalls are shown in figure 36. Figure 37 shows the external contour inspection results of the second vane which is similar to that obtained with the first vane, indicating good agreement with the vane engineering drawing. A maximum of 0.013 and 0.015 cm (0.005 and 0.006 in.) oversize was obtained on the pressure side of the vane near the leading edge and on the platform near midchord, respectively.

46
Figure 35. NASA Wafer Vane Showing Mislocated Suction Side Slot
Figure 36. NASA Wafer Vanes and Associated Endwalls
Figure 37. Radial Wafer Vane External Contour Inspection Results
VII. COLD FLOW CALIBRATION

A cold flow program was conducted with the two vanes upon completion of the fabrication process. The experimental results for the two vane airfoils along with the predicted curve are shown in figure 38 indicating good agreement. Endwall data are shown in figure 39 along with the predicted curve, and poor agreement was obtained. One reason contributing to the underflowing situation is that some of the coolant passages on the platform did not open up during the final machining operation. However, the number of passages not opening up was not sufficient to account for all of the discrepancy. Another probable source of error in the cold flow data was in measurement of coolant supply pressure. This parameter was measured outside of the platform cavity, and if a pressure drop exists between the location of measurement and the platform cavity, this would result in an indicated underflowing condition. The problem may have existed because in the endwall region a different supply manifold arrangement was used than with the airfoil. This arrangement was required to seal off airflow to the airfoil and only supply airflow to the platform cavity. In doing this a restriction may have been introduced between the platform cavity and the section where the coolant supply pressure was measured. Since measurement of this orifice area (if it did exist) was not practical, a correction could not be determined. It is recommended that once the coolant supply tubes are connected to the platform, this section be rechecked to determine if during actual cascade testing the supply pressure must be increased to obtain the design flowrate. A tabulation of the cold flow data and pressure ratio for the vanes and endwalls is included in Appendix A.
Figure 38. Cold Flow for NASA Radial Wafer Vane
Figure 39. Cold Flow for NASA Radial Wafer Vane Endwall
VIII. CONCLUSIONS AND RECOMMENDATIONS

A. CONCLUSIONS

1. Use of an efficient convective cooling scheme on the suction side and in the trailing edge region of the NASA vane eliminated the need for film cooling in those regions and minimized the amount of convective coolant required.

2. Elimination of film cooling on the suction side will result in a vane design that is more desirable from an aerodynamic performance standpoint than one with film cooling on the suction side.

3. The effective cooling scheme was possible through the use of the radial wafer fabrication techniques which permit designs with small intricate connective passages not attainable in cast and drilled airfoils.

4. Use of the radial wafer fabrication method permitted full coverage film cooling in the areas desired and allowed incorporation of the endwall cooling design into the wafers that formed the vane.

B. RECOMMENDATIONS

The cold flow calibration test for the endwalls should be repeated before conducting the hot cascade tests.
IX. APPENDIX A – REQUIRED DATA
Table 1. Pressure Side Chordwise Metal Temperatures for Design and Off-Design Points

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### Greek Symbols

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<td>Cooling Effectiveness Parameter = ( \frac{T_g - T_m}{T_g - T_c} )</td>
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**FINAL REPORT, CR-159655**

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<td>NASA Scientific and Technical Information Facility</td>
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<td>Attn: F. E. Bailey</td>
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<td>Attn: Dr. Norman Tallan</td>
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<td>AiResearch Manufacturing Company</td>
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<td>Attn: Dr. F. B. Wallace</td>
<td>C. E. Corrigan</td>
<td>F. E. Faulkner</td>
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7. AVCO Corporation
   Lycoming Division
   550 South Main Street
   Stratford, CT 06417
   Attn: A. F. DeFerrari

8. Curtiss-Wright Corporation
   One Passaic Street
   Woodridge, NJ 07075
   Attn: Dr. Sam Wolisin

9. Electric Power Research Institute
   3412 Hillview Avenue
   PO Box 10412
   Palo Alto, CA 94303
   Attn: Dr. Arthur Cohn

10. General Electric Company
    Advanced Product Planning
    Schenectady, NY 12301
    Attn: Dr. William H. Day

11. General Electric Company
    Gas Turbine Products Division
    Schenectady, NY 12301
    Attn: Dr. J. E. Palko

12. General Electric Company
    Flight Propulsion Division
    Cincinnati, OH 45215
    Attn: M. Zipkin
    H. Maclin

13. General Electric Company
    1000 Western Avenue
    Lynn, MA 01910
    Attn: Arne Brook
    Mail Drop 240 6B

14. General Motors Corporation
    Detroit Diesel Allison
    PO Box 894
    Indianapolis, IN 36206
    Attn: D. A. Nealy
    D. Klingman
    W. C. Davis
15. International Harvester Corporation
Solar Division
PO Box 80966
San Diego, CA 92138
Attn: Dr. Arthur Metcalfe
D. M. Evans

16. National Aeronautics and Space Administration
Ames Research Center
Moffett Field, CA 94035
Attn: Library

17. National Aeronautics and Space Administration
NASA Headquarters
Washington, D.C. 20546
Attn: G. Banerian M.S.
L. Harris RTM-6
R. Rudey RTP-6

18. National Aeronautics and Space Administration
Flight Research Center
PO Box 273
Edwards, CA 93533
Attn: Library

19. National Aeronautics and Space Administration
Langley Research Center
Hampton, VA 23665
Attn: Library

20. U.S. Energy Research and Development Administration
Division of Conservation Research and Technology
Washington, D.C. 20545
Attn: John Neal
Erick Lister

21. Chief, Materials Engineering Department
Department 93-03-503-4
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