SiC/SiC Leading Edge Turbine Airfoil Tested Under Simulated Gas Turbine Conditions

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ABSTRACT

Silicon-based ceramics have been proposed as component materials for use in gas turbine engine hot-sections. A high pressure burner rig was used to expose both a baseline metal airfoil and ceramic matrix composite leading edge airfoil to typical gas turbine conditions to comparatively evaluate the material response at high temperatures. To eliminate many of the concerns related to an entirely ceramic, rotating airfoil, this study has focused on equipping a stationary metal airfoil with a ceramic leading edge insert to demonstrate the feasibility and benefits of such a configuration. Here, the idea was to allow the SiC/SiC composite to be integrated as the airfoil's leading edge, operating in a "free-floating" or unrestrained manner, and provide temperature relief to the metal blade underneath. The test included cycling the airfoils between simulated idle, lift, and cruise flight conditions. In addition, the airfoils were air-cooled, uniquely instrumented, and exposed to the same internal and external conditions, which included gas temperatures in excess of 1370°C (2500°F). Results show the leading edge insert remained structurally intact after 200 simulated flight cycles with only a slightly oxidized surface. The instrumentation clearly suggested a significant reduction (~600°F) in internal metal temperatures as a result of the ceramic leading edge. The object of this testing was to validate the design and analysis done by Materials Research and Design of Rosemont, PA and to determine the feasibility of this design for the intended application.
INTRODUCTION

A number of engine components have now been successfully manufactured in ceramic matrix composite (CMC) materials and tested on rigs and engines. There is a growing confidence in the use of these materials as more experience is accumulated and the design rules and criteria become validated. The need for increased component life in a number of military engines drives users and manufacturers to consider the application of high temperature, CMC materials to the hot section of in-service engines to reduce their operating costs and to improve readiness. The F402-RR-406 Pegasus second stage high pressure turbine (HPT2) vane is an ideal candidate for such a material. The Pegasus powers the Harrier, a McDonnell Douglas and British Aerospace produced V/STOL, which is projected to remain in operation well into the next century. The Pegasus engine is a two-spool design with a low and high pressure compressor and turbine based on materials technology available in the late 1950’s, Figure 1. Due to the high engine loads experienced during hovering at high ambient temperatures, second stage high pressure turbine stator vanes in the F402-RR-406 can exceed their temperature capability and fail by cracking or melting at the leading edge, followed by failure at the trailing edge, Figure 2. Once cooling air is lost at the airfoil due to failure, cooling air is also lost to downstream components, e.g., low pressure turbine vane, resulting in reduced life. Catastrophic vane failure can also create component failures downstream as fragments pass through low pressure turbine sections.

Figure 1. Cross-Section of Pegasus F402 Engine
The second stage high pressure turbine stator vanes are produced as-cast from the Ni-based NIMOCAST alloy PD21 (International Nickel Ltd.). This alloy has exceptionally high stress-rupture properties up to $1922^\circ F$ and a liquidus temperature of $2500^\circ F$. During operation, leading edge (LE) temperatures can approach $2100^\circ F$, while trailing edge (TE) temperatures can approach $2138^\circ F$. These temperatures occur during 30% of the life of the vane. For the remaining life of the vane, maximum LE temperatures are approximately $1600^\circ F$. Backside cooling air is provided to extend life with holes located on the LE and TE of a manifold passing through the center of the hollow vane. A thermal barrier coating is applied to the LE edge surface to further extend blade life, while water injection is provided to increase performance by cooling surfaces exposed to increased combustion temperatures. Finally, an aluminide coating is applied to the vane to improve hot corrosion resistance.

There are a number of complex issues that must be addressed in selecting a CMC material for high temperature engine applications. In a gas turbine engine, components in and along the gas flow path may be exposed to high temperature, mechanical and thermal stresses, moisture, corrosion, airborne solids, and oxidizing atmospheres. In marine environments experienced by many Naval aircraft, sea salt can accumulate in the compressor section and ultimately pass through the engine where it can form chloride or sulfate rich salts as it enters the hot sections of the combustor, turbine and augmentor. Moisture and salt laden moisture can also react with the materials during storage in marine environments. Due to difficulties in simulating gas turbine operating conditions, especially those operated in marine environments, engine test rigs continue to provide the best form of materials evaluation.
The vane design was established in a previous study, in which a CMC LE insert was determined to be the best candidate for this application. The insert consists of an AlliedSignal Composites, Inc. (ACI) enhanced HiNicalon™ fiber reinforced silicon carbide composite material fabricated via chemical vapor infiltration. The insert was 45 mils thick and occupied a recessed area in the leading edge of the vane and was designed to be free floating.

EXPERIMENTAL

Vane Fabrication

The materials and design analysis indicated that the vane should use an end cap design, as shown in Figure 3, as the lowest risk option. The metal vane’s LE was modified using electron discharge machining (EDM) to reduce the wall thickness from 0.060" to 0.015". The shrouds were machined to allow the ceramic LE (002) to be inserted through the top platform and into a recession in the bottom platform (001) to hold the insert in place. The CMC LE was installed and locked into place using a welded metal cap (003).

Figure 3. CMC LE Vane Schematic

The CMC leading edge vanes were fabricated for component evaluation from ACI’s enhanced SiCf/SiC using HiNicalon™ fiber. The leading edge was preformed into a blank, placed in a graphite tool for boron nitride coating, removed from the tool and densified with silicon carbide via CVI. The parts were machined to their final dimensions.

Apparatus

The High Pressure Burner Rig (HPBR) at NASA GRC is a high-velocity, pressurized combustion test rig used for high-temperature environmental durability studies of advanced materials. The facility burns jet fuel and air in controlled ratios to produce combustion gas chemistries and temperatures that are comparable to those in gas turbine engines. In addition, the test section is capable of simulating the pressures and gas velocities representative of today’s aircraft. A schematic of the HPBR is shown in Figure 4. Combustion air enters the combustor and flows over the outside of the liner to provide cooling and to be preheated for more efficient combustion. The air is directed through a swirler in the combustor’s dome, mixed with jet fuel supplied by an air-blast fuel nozzle,
and ignited by a spark plug with hydrogen. The combustion products flow downstream through a water-cooled turbulator orifice and transition section before passing into the test section.

![Schematic of the High Pressure Burner Rig (HPBR).](image)

**Figure 4 – Schematic of the High Pressure Burner Rig (HPBR).**

The hot gas mixture passes through the test section and over the test sample held within a fixture. The sample holder is mounted on a shaft and inserted into the gas path through the “T-section” at an angle normal to the gas flow, also illustrated in Figure 4. The mass flow, gas chemistry, velocity, and pressure are controlled in the test section, and temperatures are measured both optically and using thermocouple technology. The combustion gases are then quenched downstream by a water spray before passing through an exit valve that maintains system pressure

**Test Setup**

A metal airfoil was machined to accommodate the ceramic leading edge, and instrumentation channels were machined into the blade to house thermocouples installed beneath a cover plate. The thermocouples were routed to the ceramic-metal interface at midspan and near both outer platforms. For the baseline metal blade, the thermocouples were terminated in the metal wall at a depth equal to that of the ceramic-metal interface. Shown assembled in Figure 5, water-cooled fixtures were used to secure the vane, shield the platforms from the combustion gases, and manifold air cooling to the platforms and internal vane. Each vane was centered in the rig with the leading edge directly toward the gas stream. This allowed optical temperature measurement of the vanes’ leading edge at midspan in close proximity to the thermocouple position.
Test Conditions

To simulate gas turbine conditions for test purposes, the primary variables of interest are temperature, pressure, and cooling flow requirements. Standard operation is to control the fuel-to-air ratio, test pressure, and mass airflow (1.0 lbm/sec) while providing airfoil cooling as required. The resultant temperatures and gas velocity are thus dependent variables fixed by the airflow, fuel-to-air ratio, and pressure. The experimental test cycle was based on the best possible match between rig capabilities and Pegasus F402 idle and lift conditions, using the fastest achievable transients.

Table I shows both the engine and resulting test cycles. By comparison, the most significant differences were in "Idle" pressures (gas and cooling air) and cooling air temperature. The reduced pressures were due to rig limitations, while a single level of cooling air temperature was chosen, due in part to preheater limitations, below the engine specification to maximize the temperature gradient. In general, the HPBR provided an adequate simulation of the F402 cycle. The gas composition (fuel-to-air ratio) was preset during calibrations based on temperature requirements and was nearly stoichiometric at lift conditions. Initially, the metal blade was calibrated to define the cycle then exposed to the resulting test conditions to try to duplicate the blade temperatures and failures seen in service by the F402. A lower peak metal temperature at take-off (lift) is also notable, most likely due to differences in heat flux and cooling air temperature between the rig and engine. Vane temperatures will be discussed further in the next section. Ultimately, the airfoil fitted with the CMC leading edge was installed and exposed to the same cycle.
Table I. Test Cycle Conditions

<table>
<thead>
<tr>
<th></th>
<th>Idle Pegasus / HPBR</th>
<th>Take-off Pegasus / HPBR</th>
</tr>
</thead>
<tbody>
<tr>
<td>Time</td>
<td>n/a</td>
<td>n/a</td>
</tr>
<tr>
<td>Gas Temperature</td>
<td>1004°F 1100°F</td>
<td>2521°F 2500°F</td>
</tr>
<tr>
<td>Gas Pressure</td>
<td>15 psia 60 psia</td>
<td>104 psia 104 psia</td>
</tr>
<tr>
<td>Max Blade Temp</td>
<td>900°F 900°F</td>
<td>2304°F 2100°F</td>
</tr>
<tr>
<td>Cooling Air Conditions</td>
<td>196°F 120°F</td>
<td>900°F 120°F</td>
</tr>
<tr>
<td></td>
<td>16.4 psia 135 psia</td>
<td>119 psia 135 psia</td>
</tr>
<tr>
<td></td>
<td>1.8 #/min 1.0 #/min</td>
<td>1.8 #/min 1.0 #/min</td>
</tr>
</tbody>
</table>

RESULTS

Metal

The metal vane was tested for a total of 150 cycles. Figure 6 illustrates the typical temperature response of the blade from one of those cycles. Here, the cycle begins in the idle condition (30 sec), where the gas temperature is fairly stable near 1100°F. This is followed by a step-change in fuel to simulate lift conditions during acceleration and steady-state portions of the transient which reaches temperatures near 2500°F. The fuel is then throttled back during deceleration (30 sec) before returning to idle after a 2.0 minute cycle. During the transient the airfoil heats up from approximately 900°F to nearly 2100°F at the midspan location, and approximately 1900°F at the outer platforms. In an attempt to reach the F402 maximum blade temperature of 2300°F, the cooling air was turned off for cycles 136-150. Here, temperatures were indeed increased to nearly 2300°F.

Figure 6. Thermocouple profile for metal vane during HPBR test.
In addition to documenting the temperature response of the all metal airfoil for comparison to the ceramic, duplicating the airfoil damage as seen in service by the F402 engine, such as fatigue cracking, was desired. Figure 7 shows a photo of the blade after the 150 cycles. Both the leading edge and trailing edge of the blade show signs of cracking very similar to that experienced in the engine. The leading edge also shows preliminary signs of “burn-through” and the instrumentation cover has been burned way. A leak in the cooling jacket of the transition section caused the corrosion visible in the photograph as stabilizers in the water reacted with the metal. However, this most likely did not contribute to the fatigue cracking because the leaks occurred during cycles after visible damage was already indicated.

![Figure 7. Metal vane after 150 cycles in HP burner rig test.](image)

**Ceramic**

Next, the airfoil fitted with the ceramic leading edge insert was exposed to the very same test cycle to document the temperature relief provided to the metal and overall durability of the insert. In all, the blade was tested for a total of 200 cycles. Figure 8 shows the temperature response of one of those cycles and indicates reduced internal metal temperatures by as much as 600°F at the midspan location given the same surface temperature (2100°F) as indicated by the pyrometer.
After testing, the CMC structure appeared intact with no signs of cracking on either the leading edge or trailing edge as shown in Figure 9. Although an additional 50 cycles were completed, the instrumentation cover exhibited little or no damage, supporting that a lower surface temperature was also experienced just downstream of the insert. Only a slight oxide scale, as would be expected, was noted on the insert.

The metal temperatures in the CMC LE vane were approximately 300-600°F cooler than in the metal vane during lift conditions. This was cooler than an analysis that predicted a temperature of 1835°F at the metal-ceramic interface. Table II lists the actual measured temperatures and pressures for the CMC LE vane, along with targeted or predicted values, for the lift conditions.
Overall, it is believed that the CMC acted as a thick thermal barrier coating (TBC). With a small air gap between the metal and the CMC leading edge, heat transfer from the CMC to the metal alloy was low, effectively lowering the predicted temperatures.

Table II. NASA HP Burner Rig Test Conditions – Pegasus 406 Uncooled Leading Edge

<table>
<thead>
<tr>
<th>Conditions (SLW)</th>
<th>Units</th>
<th>Goal</th>
<th>Metal</th>
<th>CMC</th>
</tr>
</thead>
<tbody>
<tr>
<td>Gas $T_{Probe}$</td>
<td>°F</td>
<td>2521</td>
<td>2500</td>
<td>2500</td>
</tr>
<tr>
<td>Vane $T_3$</td>
<td>°F</td>
<td>1835</td>
<td>2050</td>
<td>1500</td>
</tr>
<tr>
<td>Vane $T_{Pyro}$</td>
<td>°F</td>
<td>2304</td>
<td>2100</td>
<td>2100</td>
</tr>
<tr>
<td>Vane $T_{Cooling Air}$</td>
<td>°F</td>
<td>900</td>
<td>120</td>
<td>120</td>
</tr>
<tr>
<td>Vane $P_{Cooling Air}$</td>
<td>psi</td>
<td>120</td>
<td>135</td>
<td>135</td>
</tr>
<tr>
<td>Gas $P_{HP2}$</td>
<td>psi</td>
<td>400</td>
<td>104</td>
<td>104</td>
</tr>
<tr>
<td>Cycles</td>
<td></td>
<td>300/25 sec</td>
<td>150/120 sec</td>
<td>200/120 sec</td>
</tr>
</tbody>
</table>

SUMMARY

The testing of the CMC leading edge is considered to have successfully demonstrated the goals of this program. The CMC LE has proven that an uncooled CMC can be engineered and designed to withstand the thermal upshock during short wet lift conditions for the Pegasus engine (the most severe conditions for this engine). The design of the leading edge insert, which minimized thermal stresses in the CMC, showed that the CMC/metal assembly can be engineered to be a functioning component.
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