NASA/CR—1999-209314

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

R. Craig Robinson
Dynacs Engineering Company, Inc., Cleveland, Ohio

Kenneth S. Hatton
AlliedSignal Composites, Inc., Newark, Delaware

September 1999

The NASA STI Program Office . . . in Profile

Since its founding, NASA has been dedicated to the advancement of aeronautics and space science. The NASA Scientific and Technical Information (STI) Program Office plays a key part in helping NASA maintain this important role.

The NASA STI Program Office is operated by Langley Research Center, the Lead Center for NASA's scientific and technical information. The NASA STI Program Office provides access to the NASA STI Database, the largest collection of aeronautical and space science STI in the world. The Program Office is also NASA's institutional mechanism for disseminating the results of its research and development activities. These results are published by NASA in the NASA STI Report Series, which includes the following report types:

- TECHNICAL PUBLICATION. Reports of completed research or a major significant phase of research that present the results of NASA programs and include extensive data or theoretical analysis. Includes compilations of significant scientific and technical data and information deemed to be of continuing reference value. NASA's counterpart of peer-reviewed formal professional papers but has less stringent limitations on manuscript length and extent of graphic presentations.

- TECHNICAL MEMORANDUM. Scientific and technical findings that are preliminary or of specialized interest, e.g., quick release reports, working papers, and bibliographies that contain minimal annotation. Does not contain extensive analysis.

- CONTRACTOR REPORT. Scientific and technical findings by NASA-sponsored contractors and grantees.

- CONFERENCE PUBLICATION. Collected papers from scientific and technical conferences, symposia, seminars, or other meetings sponsored or cosponsored by NASA.

- SPECIAL PUBLICATION. Scientific, technical, or historical information from NASA programs, projects, and missions, often concerned with subjects having substantial public interest.

- TECHNICAL TRANSLATION. English-language translations of foreign scientific and technical material pertinent to NASA's mission.

Specialized services that complement the STI Program Office's diverse offerings include creating custom thesauri, building customized data bases, organizing and publishing research results . . . even providing videos.

For more information about the NASA STI Program Office, see the following:

- Access the NASA STI Program Home Page at ***http://www.sti.nasa.gov***

- E-mail your question via the Internet to help@sti.nasa.gov

- Fax your question to the NASA Access Help Desk at (301) 621-0134

- Telephone the NASA Access Help Desk at (301) 621-0390

- Write to:
  NASA Access Help Desk
  NASA Center for AeroSpace Information
  7121 Standard Drive
  Hanover, MD 21076

NASA/CR—1999-209314

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

R. Craig Robinson
Dynacs Engineering Company, Inc., Cleveland, Ohio

Kenneth S. Hatton
AlliedSignal Composites, Inc., Newark, Delaware

Prepared for the
23rd Annual Conference on Composites, Materials, and Structures
sponsored by the American Ceramic Society
Cocoa Beach, Florida, January 25–29, 1999

Prepared under Cooperative Agreement SAA3–145

National Aeronautics and
Space Administration

Glenn Research Center

September 1999

Acknowledgments

AlliedSignal Composites Inc. and NASA Glenn Research Center at Lewis Field would like to thank the Office of Naval Research for their support of this program through ONR Contract #N00014-96-C-0149 and under the guidance of Dr. Steven Fishman. In addition, we acknowledge the help of the NASA Glenn Aeropropulsion Program, under the HiTEMP Project Office managed by Ms. Carol Ginty.

Available from

NASA Center for Aerospace Information
7121 Standard Drive
Hanover, MD 21076
Price Code: A03

National Technical Information Service
5285 Port Royal Road
Springfield, VA 22100
Price Code: A03

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

R. Craig Robinson
Dynacs Engineering Company, Inc.
Brook Park, OH 44142

Kenneth S. Hatton
AlliedSignal Composites, Inc.
Newark, DE 19714

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.

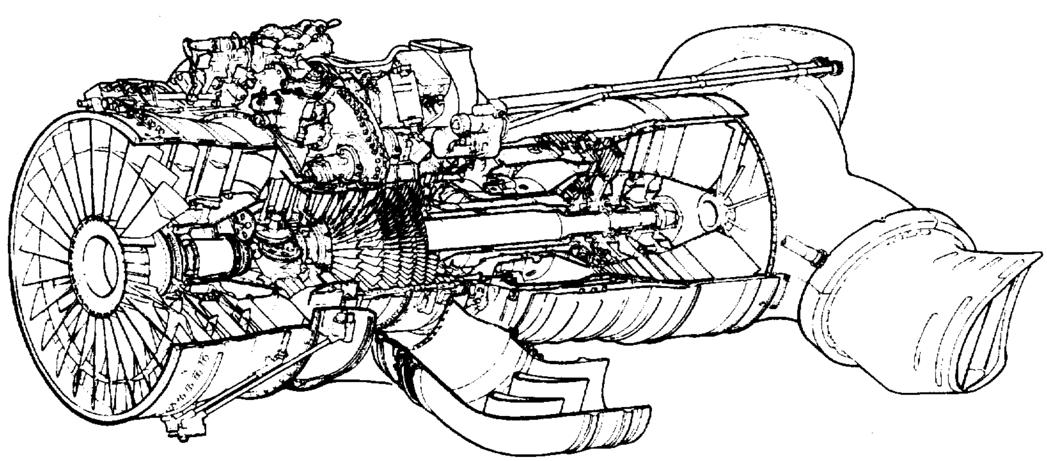

Figure 1. Cross-Section of Pegasus F402 Engine

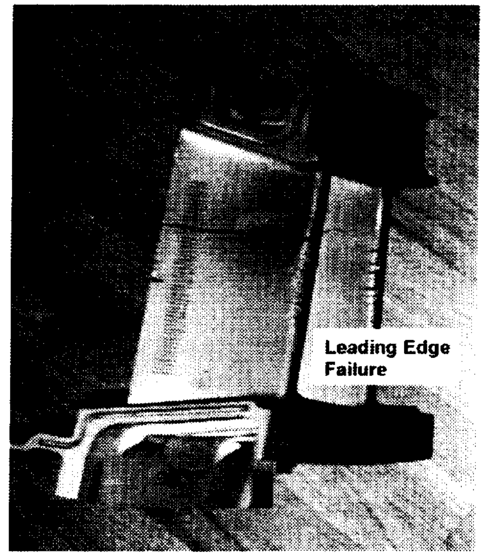

Figure 2. Pegasus F402-RR-406 2nd stage NGV.

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°F and a liquidus temperature of 2500°F. During operation, leading edge (LE) temperatures can approach 2100°F, while trailing edge (TE) temperatures can approach 2138°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°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).

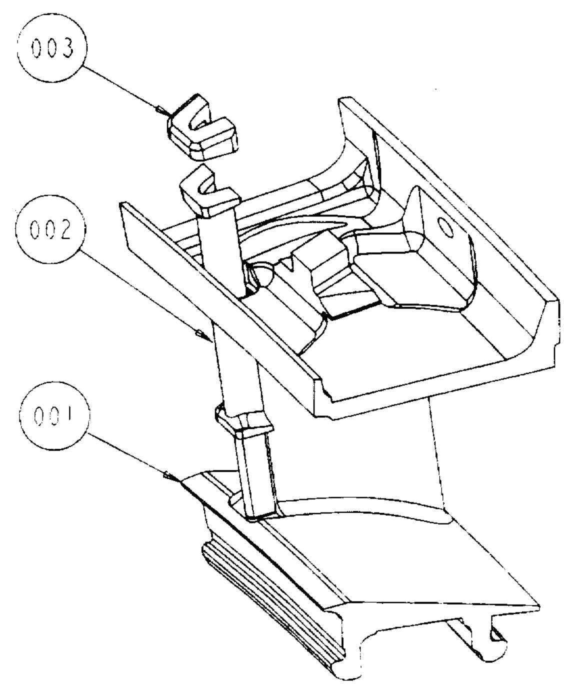

Figure 3. CMC LE Vane Schematic

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.

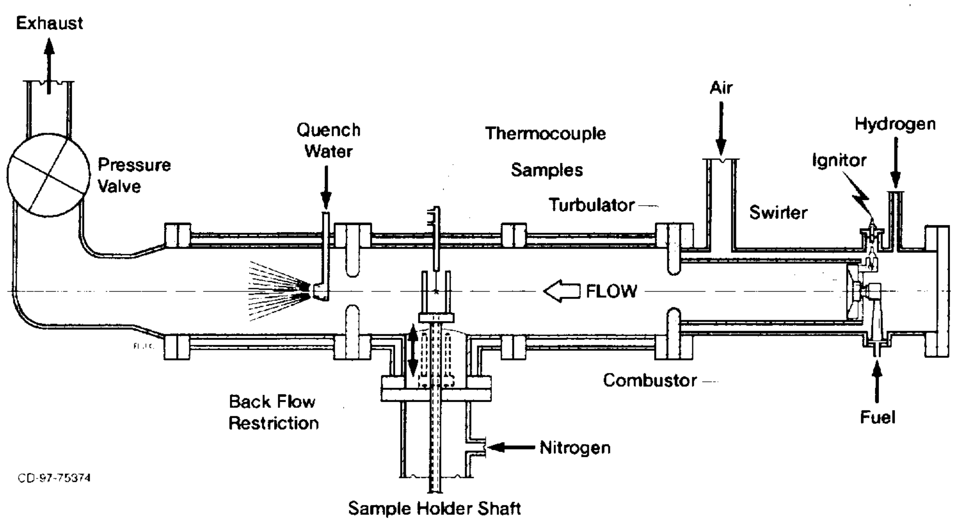

Figure 4 – Schematic of the High Pressure Burner Rig (HPBR).

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.

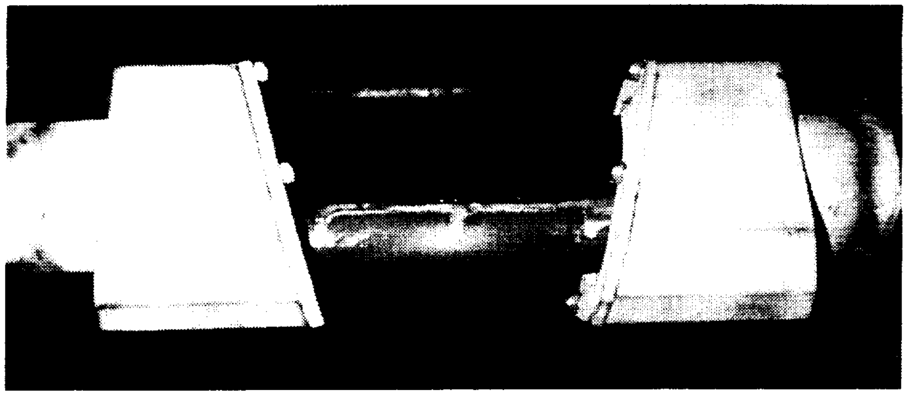

Figure 5. HPBR test holder.

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

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

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.

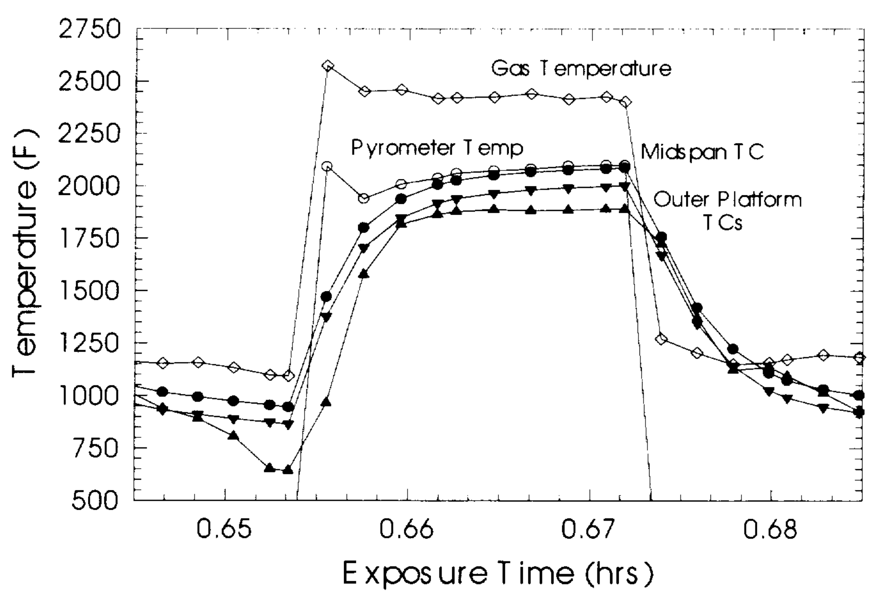

Figure 6. Thermocouple profile for metal vane during HPBR test.

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 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.

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.

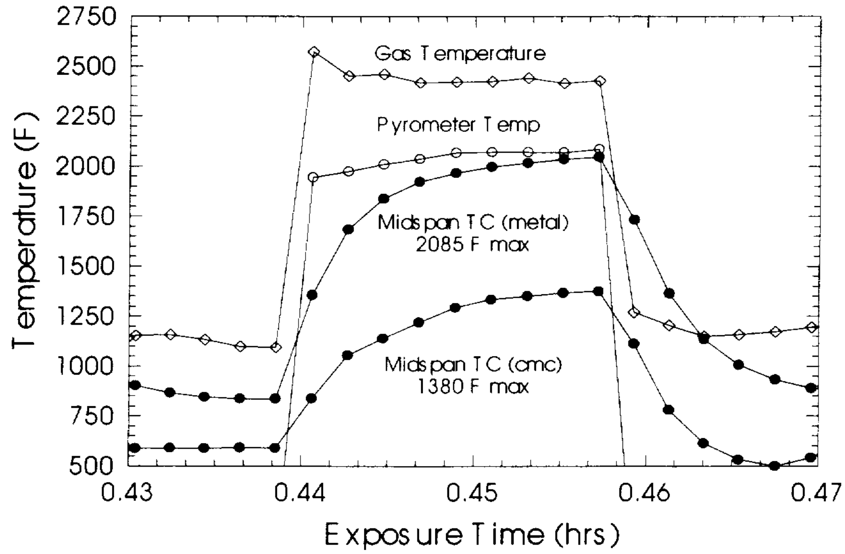

Figure 8. Thermocouple profile for CMC LE vane during HPBR test.

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.

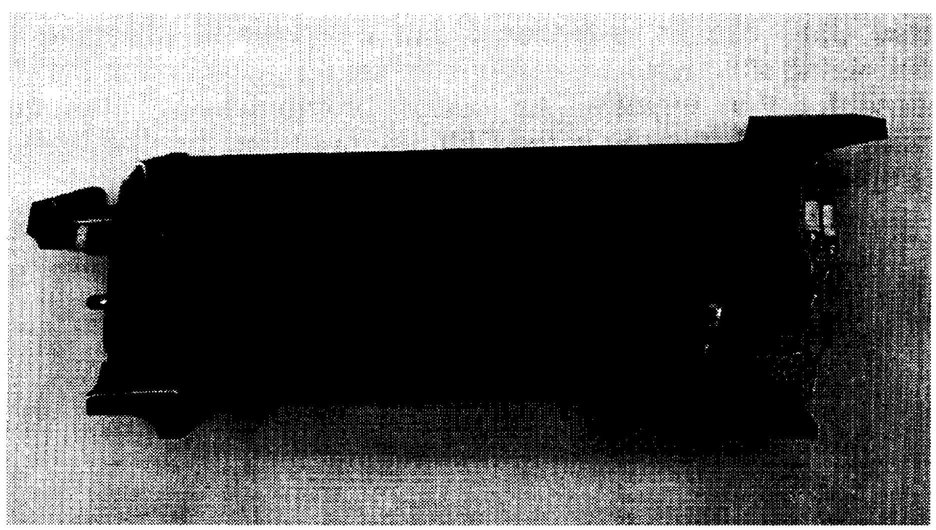

Figure 9. CMC LE vane after testing with instrumentation showing.

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

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

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.

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY (*Leave blank*)	2. REPORT DATE	3. REPORT TYPE AND DATES COVERED
	September 1999	Final Contractor Report

4. TITLE AND SUBTITLE

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

5. FUNDING NUMBERS

WU-505-23-2F-00
SAA3-145

6. AUTHOR(S)

R. Craig Robinson and Kenneth S. Hatton

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

Dynacs Engineering Company, Inc.
Brook Park, Ohio 44142

8. PERFORMING ORGANIZATION REPORT NUMBER

E-11919

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)

National Aeronautics and Space Administration
John H. Glenn Research Center at Lewis Field
Cleveland, Ohio 44135-3191

10. SPONSORING/MONITORING AGENCY REPORT NUMBER

NASA CR—1999-209314

11. SUPPLEMENTARY NOTES

Prepared for the 23rd Annual Conference on Composites, Materials, and Structures sponsored by the American Ceramic Society, Cocoa Beach, Florida, January 25–29, 1999. R. Craig Robinson, Dynacs Engineering Company, Inc., Brook Park, Ohio, and Kenneth S. Hatton, AlliedSignal Composites, Inc., 1300 Marrows Road, P.O. Box 9559, Newark, Delaware 19714–9559. Project Manager, Leslie Greenbauer-Seng, Materials Division, NASA Glenn Research Center, organization code 5160, (216) 433–6781.

12a. DISTRIBUTION/AVAILABILITY STATEMENT

Unclassified - Unlimited
Subject Categories: 01 and 24 Distribution: Nonstandard

This publication is available from the NASA Center for AeroSpace Information, (301) 621–0390.

12b. DISTRIBUTION CODE

13. ABSTRACT *(Maximum 200 words)*

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.

14. SUBJECT TERMS

Airfoils; Ceramic composites; Combustion

15. NUMBER OF PAGES 16

16. PRICE CODE A03

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	