Ceramic Thermal-Barrier Coatings for Cooled Turbines

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Ceramic thermal-barrier coatings on hot engine parts have the potential to reduce metal temperatures, coolant requirements, cost, and complexity of the cooling configuration, and to increase life, turbine efficiency, and gas temperature. Coating systems consisting of a plasma-sprayed layer of zirconia stabilized with either yttria, magnesia, or calcia over a thin alloy bond coat have been developed, their potential analyzed, and their durability and benefits evaluated in a turbojet engine. The coatings on air-cooled rotating blades were in good condition after completing as many as 500 two-min cycles of engine operation between full power at a gas temperature of 1644 K and flameout, or as much as 150 hr of steady-state operation on cooled vanes and blades at gas temperatures as high as 1644 K with 35 start and stop cycles. On the basis of durability and processing cost, the yttria-stabilized zirconia was considered the best of the three coatings investigated.

Introduction

RECENT work (1973) with cooled rocket engines operating at high gas temperatures and heat fluxes[1] shows that ceramic coatings are good heat insulators and can withstand large temperature differences through the coating thickness. In a much earlier work (1953) ceramic coatings were tried as a means for reducing the metal temperature of uncooled turbine blades in a turbojet engine during transient operation.[2] These uses of heat-barrier coatings, however, were for short time periods of a minute or less. Also in 1953, techniques resembling enameling or glazing were used for coating ceramics onto air-cooled turbine blades, and steady-state durability tests in a jet engine were conducted on these coatings.[3] The engine tests reported in Refs. 2 and 3, however, were made at relatively low turbine inlet temperatures and heat fluxes.

The operating conditions of current and future gas turbine engines – long-time steady-state operation at high pressure, temperature, heat flux, and stress levels – impose more severe strains on the coating. Also, the coating may have to withstand several thousand hours of cyclic engine operation to gas temperatures as high as 2200 K without cracking, spalling, or eroding. In addition, to be useful, the airfoil ceramic coating should have a low thermal conductivity and a low density and must not degrade turbine aerodynamic performance. Stabilized zirconia appears to have many of the desired properties.

The purposes of this report are to summarize the work conducted to: 1) demonstrate the insulation capability of stabilized zirconia in steady-state engine operation and compare the experimental results with analysis; 2) analyze the potential benefits of using coatings for air-cooled turbines; 3) evaluate aerodynamic performance of coated airfoils; 4) evaluate the durability of the coatings on turbine vanes and blades in steady-state and cyclic engine operation; and 5) evaluate the relative material and processing costs for several coatings.

The results include comparisons of measured and calculated vane metal temperatures with and without thermal-barrier coatings at steady-state engine conditions over a range of coolant-to-gas flow ratios. The potential benefits to be achieved by using a thermal-barrier coating are presented in terms of coolant flow and metal temperature reduction for both an advanced core engine turbine and a turbine in an existing research engine. A simple steady-state one-dimensional heat transfer analysis was used on composite walls that consisted of a metal wall, an alloy bond coat, and various thicknesses of zirconia coating. The gas temperatures and pressures for the analysis were taken at 1644 and 2200 K and 3 and 40 atm, respectively. The aerodynamic results are presented in terms of kinetic energy loss coefficients and were obtained by pressure and angle surveys made in a two-dimensional, cold-air cascade. The coating durability was evaluated in the turbine of a research engine at steady-state and cyclic conditions and the results are presented in terms of the coating condition as determined by visual and metallographic examinations. The maximum gas temperature and pressure conditions for the engine tests were 1644 K and 3.0 atm, respectively.

Apparatus and Procedure

An air-cooled turbine blade covered with a ceramic thermal barrier coating is shown in Fig. 1. The procedure used for depositing the ceramic coating onto the blade metal substrate was to prepare the substrate surface, plasma-spray on a bond coat, and then plasma-spray on the ceramic coating. The most current application process is described herein.

Coating Description and Application Process

Prior to coating, all airfoil surfaces and base platforms were first grit-blasted with commercial, pure (white) alumina. Use of the "white" alumina minimized contamination that might occur with less pure grit. The inlet supply pressure to the equipment was 70 N/cm^2. The grit impingement was nearly normal to the surface. The alumina grit size was 250μm and the surface roughness after grit-blasting was 6 μm, rms. All airfoils used, except for those in the aerodynamic tests, had prior usage and had an aluminide coating. Grit-blasting removed about one-tenth or about 10^{-3} cm of this coating.

For the blades to be used in the cyclic tests, a bond coat of NiCrALY (Ni-16Cr-6Al – 0.5Y) was sprayed onto the substrate to a thickness of 0.010 ± 0.005 cm. The particle size of the bond powders fed into the plasma spray gun was 74 to 44 μm. The measured roughness of the bond coat was 5 μm, rms.

Within 30 min after bond coat application, stabilized zirconia was applied to a thickness of 0.038 ± 0.008 cm. Thirty-one blades were prepared with 12 wt% of yttria-stabilized zirconia, 13 with 3.2 wt% of magnesia-stabilized zirconia, and 39 with 5 wt% calcia-stabilized zirconia. The yttria- and magnesia-stabilized zirconia particle size was 74 to 44 μm and the calcia-stabilized zirconia powder size was 105 to 10 μm. The roughness of the applied ceramic coatings was

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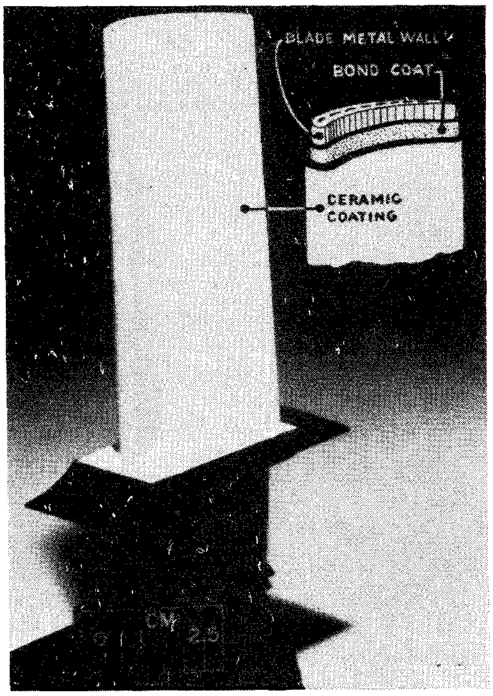

Fig. 1 Ceramic coated turbine blade.

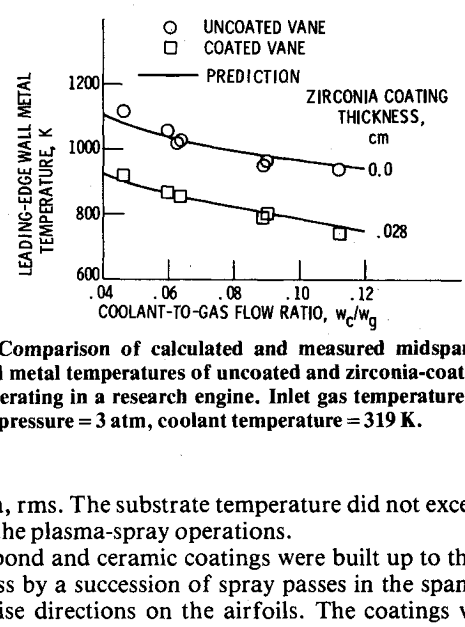

Fig. 2 Comparison of calculated and measured midspan leading-edge wall metal temperatures of uncoated and zirconia-coated turbine vanes operating in a research engine. Inlet gas temperature = 1644 K, inlet gas pressure = 3 atm, coolant temperature = 319 K.

8-10 μm, rms. The substrate temperature did not exceed 420 K during the plasma-spray operations.

The bond and ceramic coatings were built up to the desired thickness by a succession of spray passes in the spanwise and chordwise directions on the airfoils. The coatings were first applied to the blade leading edge, then to the trailing edge, and finally to the pressure and suction surfaces. In this way, overlapping coating seams were joined on the flatter surfaces. This was important because furnace tests of the coating have shown that seams along small radii such as the leading and trailing edges can lead to coating cracking.

The coated surface area on each blade was 110cm^2. The coating thickness was measured during the coating process by checking the overall thickness of the airfoil at points at the midspan and midchord. The measurements were made with micrometer calipers. The powder needed to coat a blade was 113 g for the yttria-stabilized zirconia, 255 g for the magnesia-stabilized zirconia, and 56 g for the calcia-stabilized zirconia. The plasma-spray gun was held nearly perpendicular to the surface at distances of 15 and 10 cm for bond and ceramic coat applications, respectively. The processing time for a blade with yttria-, magnesia-, and calcia-stabilized zirconia was about 20, 35, and 15 min, respectively.

For the vanes used in the aerodynamic tests, a bond coat of nichrome (80Ni-20Cr) was used and covered with zirconium orthosilicate. For the vanes and blades used in the steady-state durability and insulating capability tests, a bond coat of nichrome of a nominal thickness of 0.010 cm was used and covered with calcia-stabilized zirconia applied to a nominal thickness of 0.028 cm.

Coating Equipment

Commercial grit-blasting equipment was used to clean and roughen the vane and blade surfaces. A hand-held plasma-spray gun was used to apply powders of bond and ceramic materials. In the gun[4] an electric arc is contained within a watercooled nozzle. Argon gas passes through the arc and is excited to temperatures of about 17,000 K. The bond and ceramic powders were mechanically fed into the nozzle and were almost instantaneously melted.

Test Equipment and Procedure

Aerodynamic Cascade Tests

Solid core-vanes were tested in a simple two-dimensional cascade tunnel described in Ref. 5. This cascade tunnel has ten vanes. Uncoated, rough-coated, and smooth-coated vanes were tested. Only one coated vane was used. This single coated vane was inserted in place of one of the uncoated vanes near the center of the ten-vane cascade.

In operation, cold atmospheric air was drawn through the cascade tunnel, the vanes, and the exhaust control valve into the laboratory exhaust system. The aerodynamic performance of all three configurations was determined in terms of kinetic energy loss coefficient over a range of pressure ratios corresponding to ideal exit critical velocity ratios of about 0.6 to 0.95 using pressure and flow-angle survey rakes.

Engine Tests

An existing research turbojet engine modified to investigate air-cooled turbine vane and blade configurations was used to evaluate the insulating capability and durability of the coatings. The vane and blade walls were made from cast MAR-M-302 and cast B-1900,[6] respectively. The turbine vanes had impingement-cooled and chordwise-finned leading edges, impingement-cooled pressure and suction-surfaces, and convention- and film-cooled trailing edges. The blade was convection-cooled with air flowing radially outward from hub to tip over internal spanwise fins. The turbine wheel diameter was 81.8 cm and the blade length was 10.5 cm. Instrumentation was provided for measurements of turbine inlet gas temperatures and gas pressure; fuel/air ratio; vane and blade cooling-air inlet temperature; and flow rate, vane, and blade average metal wall temperature and blade trailing-edge ceramic coating temperature. The thermocouples were mounted halfway into the metal wall thickness to measure the average wall temperature. Details of the thermocouple installation are described in Ref. 7.

Steady-State Durability Tests

Prior to thermal insulation testing, the coolant flow to each of five test vanes was measured in a bench test at room temperature over a range of inlet pressures. The flow rates between vanes were found to be uniform to within 2%. Two of these five vanes were coated. One of the coated vanes and one of the uncoated vanes was instrumented with a Chromel-Alumel thermocouple at the midspan of the leading edge. These two vanes, along with the three other vanes, were fitted into a segment of the engine vane ring where the coolant flow to the vane group could be independently controlled and measured. The engine was operated at a turbine inlet gas temperature of 1644 K, a gas pressure of 3 atm, and coolant-to-gas flow ratios of about 0.045-0.11.

Steady-state durability of the coating on vanes and blades in the engine was evaluated as part of another research test. The operating conditions and number of starts and shutdowns were, as a consequence, partially influenced by the other test.

The coated vanes and blades were usually operated at turbine inlet gas temperatures of 1367 to 1644 K and a gas pressure of 3 atm. The resulting coated vane and blade leading-edge metal temperatures generally did not exceed 920 K. On several occasions, hot starts resulted in transient metal temperatures of 1200 K.

Cyclic Durability Tests

Thermal-barrier coatings were applied to 83 blades of the type just discussed. All but six had been previously operated in the engine for 200 to 500 hr. About 10% of the blades were dented at the leading-edge tips because of foreign object damage.

Control of the desired cyclic conditions was accomplished primarily by controlling the combustor fuel (ASTM A-1) supply. Adjustments were made for this control system so that maximum turbine inlet temperature and pressure were maintained at about 1644 K and 3 atm, respectively. These maximum temperature and pressure conditions will be called "full power." At full power condition the rotor speed was 8300 rpm. After about 70 sec at these conditions turbine inlet temperature and speed were reduced to about 1000 K and 3300 rpm in about 20 sec by reducing and then shutting off the fuel supply. This condition is designated "flameout." Fuel was supplied and reignited and the engine reached maximum power condition in about 30 sec. Cooling-air flow was adjusted to limit the leading- and trailing-edge metal wall temperatures to about 1200 K at the maximum temperature condition. During flameout the metal wall temperatures reached about 800 K. Other details of the procedure for automatic cycling and fuel flow control are described in detail in Ref. 8.

A total of 500 two-min cycles were run. The engine was stopped for visual inspection of the coating at 100, 300, and 500 cycles. The cycle duration was 2 min. Seventy-four blades were tested in the engine at any one time. After 100 cycles, eight blades were removed from the wheel for detailed examination and reference purposes. After termination of tests at 500 cycles, one of each of the three different types of stabilized coated blades which had been run for the full duration of the tests was sectioned and the coating and blade microstructure were examined at the midspan, leading-edge region with light optical photomicrographs, at 150×. The microstructure of one untested yttria-stabilized zirconia coated blade was also examined. The ceramic coating thickness and roughness were also obtained on one of each of the three different stabilized coated blades which had been tested for 500 cycles. The ceramic coating thickness was measured after it was purposely spalled from the surface. This was done by exceeding the allowable interface temperature of 1376 K by heating the blade in a furnace to 1600 K and then instantly cooling it by plunging it into water at a temperature of 300 K.

Thermal Barrier Coating Analysis

A simple one-dimensional steady-state heat balance through a composite wall was used to evaluate the potential benefits of using a ceramic coating as a thermal-barrier for impingement-cooled turbine vanes. Thermal radiation was neglected in the analysis. The gas and coolant conditions presented in Table 1 were those of an advanced core engine turbine and those of an existing research engine. The former engine was chosen to evaluate the benefits of a ceramic thermal insulating coating on cooled turbines that are subjected to conditions of high heat flux. The latter engine was used to evaluate the benefits of the coating at lower heat flux conditions and to demonstrate the insulating capability of the stabilized zirconia. The bulk temperature (integrated average temperature over the entire vane) was the primary variable used in evaluating the benefits of the coating. The average metal wall temperature at the leading edge region was also used in comparing predictions with experimental data. Further details are given in Tables 1 and 2 and Ref. 9.

Table 1 Analytical conditions

Parameter	Turbine vane type	
	Advanced core engine	Research engine
Turbine-inlet gas temperature, K	2200	1644
Turbine-inlet gas pressure, atm	40	3
Gas-side heat transfer coefficient, $W/(m^2)(K)$:		
Leading edge	----	2326
Bulk	8994	1186
Coolant-side heat transfer coefficient, $W/(m^2)(K)$:		
Leading edge	------------------	$7.6\times10^3\ (w_c/w_g)^{0.49}$
Bulk	[a] $5.6\times10^4\ (w_c/w_g)^{0.62}$	$9.6\times10^3\ (w_c/w_g)^{0.70}$
Coolant temperature, K	811	319
Metal wall thickness, cm	0.127	0.102
Bond coating thickness, cm	0.0102	0.0152
Ceramic coating thickness, cm	0 - 0.051	0 - 0.051

[a] w_c/w_g is the coolant-to-gas flow ratio.

Table 2 Thermal conductivity of ceramic, bond, and metal wall materials.

Material	Use	Conductivity, W/(m)(K)	Temperature range, $\overline{T}$, K	Reference
MAR-M-302	Metal wall (research engine)	$4.9\times10^{-3}\ \overline{T} + 18.0$	300 - 1260	6
MAR-M-509	Metal wall (advanced core engine)	$3.0\times10^{-2}\ \overline{T} + 3.7$	590 - 1370	10
Nichrome	Bond	$8.3\times10^{-3}\ \overline{T} + 6.7$	400 - 1400	11
Calcia-stabilized zirconia	Ceramic	$4.1\times10^{-4}\ \overline{T} + 0.46$	400 - 2400	11

Results and Discussion

Comparison of Measured and Predicted Thermal Performance

Figure 2 compares measured and predicted wall metal temperatures at the midspan of the leading edge of an uncoated and coated turbine vane operating in the research engine. The comparison is shown over a range of calculated coolant-to-gas flow ratios from 0.04 to 0.12 and includes measurements at ratios of about 0.045, 0.06, 0.09, and 0.11. The results show good agreement between prediction and measurement. The predicted and measured reductions in leading-edge metal temperature for the coated vane agreed within 25 K. The results showed large reductions in leading-edge metal temperature with the 0.028 cm thick zirconia coating. At a coolant-to-gas flow ratio of 0.06 the metal temperature was reduced by 190 K – from 1055 K for the uncoated vane to 865 for the coated vane.

Aerodynamic Performance

The kinetic energy loss coefficient[5] for the three test vanes is shown in Fig. 3. The loss for the rough-coated vane was much larger than the loss for either the smooth-coated or uncoated vane. In the as-sprayed condition the ceramic coating had a surface roughness averaging about 8-10 μm. Polishing with solid aluminum oxide smoothed the ceramic coating to a surface roughness averaging 1.6 to 3.0 μm. This smoothing reduced the loss to about one-half of that obtained with the rough coating. The loss for the smooth-coated vane was higher than the loss for the uncoated vane. At an ideal exit critical velocity of 0.8, which is near design, the kinetic energy loss coefficients were 0.062, 0.031, and 0.023 for the rough-coated, smooth-coated, and uncoated vanes, respectively.

Much of the difference in loss between the smooth-coated vane and the uncoated vane was attributed to the difference in

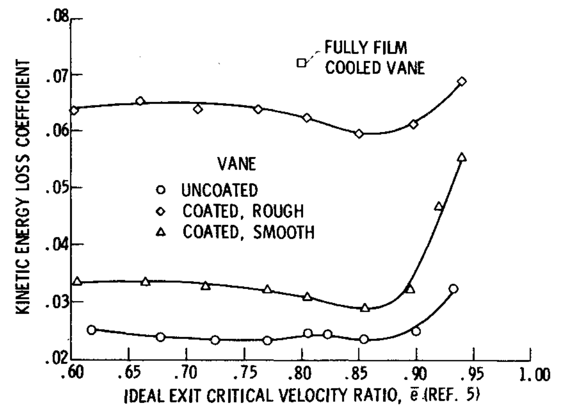

Fig. 3 Overall aerodynamic performance of a core turbine vane.

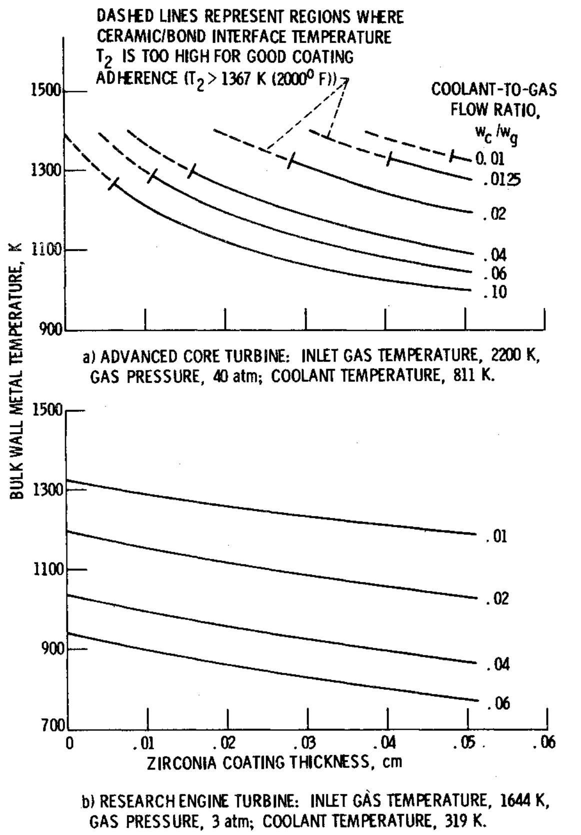

Fig. 4 Reductions in metal temperatures and coolant flows for vanes coated with various thicknesses of zirconia.

trailing-edge thickness. The trailing-edge thickness of the uncoated vane was 0.203 cm and the ceramic coating increased the trailing-edge thickness to 0.280 cm. The figure also shows a data point for a full-coverage film cooled vane.[5] The loss with this vane, though not necessarily an optimum aerodynamic design, is greater than the rough-coated blade.

Benefits of Thermal Barrier Coating

Predicted reductions in bulk turbine-vane metal temperatures and coolant-to-gas flow ratios with increases in ceramic coating thickness on vanes in the advanced core engine turbine and in the research engine turbine are shown in Figs. 4a and b, respectively. Bulk wall metal temperature (integrated average temperature over entire vane) was substantially reduced as ceramic coating thickness was increased. The reductions in metal temperature with increasing coating thickness were greater for the core engine than for the research engune. The reason is the higher heat fluxes associated with the conditions of the core engine.

The bulk wall metal temperature of an impingement-cooled advanced core turbine vane could be reduced by as much as 390 K at a coolant-to-gas flow ratio of 0.10 when the vanes were coated with a 0.051-cm thickness of zirconia (Fig. 4). Alternatively, when both coolant flow and wall metal temperature were allowed to vary, large reductions in both metal temperature and coolant flow were predicted. Vanes coated with a 0.051-cm thickness of zirconia could have both an eightfold decrease in coolant flow and 110 K reduction in metal temperature compared to the uncoated vane. The coolant flow was reduced from 0.100 for an uncoated vane to 0.0125 for a coated vane with a corresponding vane metal temperature reduction from 1390 to 1280 K, respectively.

The dashed portions of the curves in Fig. 4a illustrate a limitation associated with using the current ceramic coatings in applications such as the core engine with high gas temperature and pressure. The limitation is the ability of the ceramic coating to adhere to the bond coating when the temperature at the interface between these two layers exceeds 1367 K. This limiting temperature was determined with furnace tests described in Ref. 9.

As shown in Fig. 4a at the tick marks, thicker layers of the coating were required to give acceptable interface temperatures (1367 K) as coolant-to-gas flow ratio was decreased. This is because the heat flux will decrease as flow ratio is reduced. Reducing the heat flux will also reduce the thermal gradient through the metal wall. This together with the constant limiting interface temperature resulted in slightly increased average vane metal wall temperatures (Fig. 4a).

Although not shown in Fig. 4, calculations indicate large (bulk) temperature drops through the coating. The largest drop of 923 K occurred through a 0.051-cm thick coating on the turbine vane of the core engine at a coolant-to-gas flow ratio of 0.10. At these conditions, the coating outer temperature was 1980 K and the ceramic/bond interface temperature was 1048 K. The temperature drop through the same coating thickness on the turbine vane of the research engine was calculated to be 351 K at a coolant flow ratio of 0.06. The coating outer temperature for this condition was 1147 K and the ceramic/bond interface temperature was 796 K. The resulting average ceramic coating temperature was considerably higher in the core engine (1514 K) than in the research engine (972 K). In general, the differences in gas temperature levels and heat fluxes for the two engine conditions resulted in the temperature gradients through the coating on the advanced core engine vanes being about 2.7 times those of the research engine. The larger temperature gradient, coating exterior surface temperature, and ceramic/bond interface temperature would impose more severe strains on the coating in the advanced core engine.

Steady-State Durability

The evaluation in engine operation of 0.028-cm thick calcia-stabilized zirconia coating on each of the two vanes and blades showed no evidence of deterioration after 150 hr at gas temperatures as high as 1644 K and as many as 35 start-and-stop cycles including 4 hot starts. The measured leading-edge vane and blade steady-state metal wall temperatures were generally 920 K and the transient values were momentarily as high as 1200 K. Figure 5 shows one of the blades after completion of the tests. No deterioration was evident.

Cyclic Durability

Visual inspection of the coating after 100 cycles of testing in the research turbojet engine showed that about 90% of the blades had a metallic colored scuff mark at the tip of the leading edge. Also, about 40% of the blades had a minimal 1-cm^2 chipped area of ceramic in the vicinity of the scuff mark. The cause of the chipping was traced to the impingement of metallic pieces of thermocouple probes which broke during engine transient overheating and excessive vibration. About half of the yttria- and half of the calcia-stabilized zirconia-

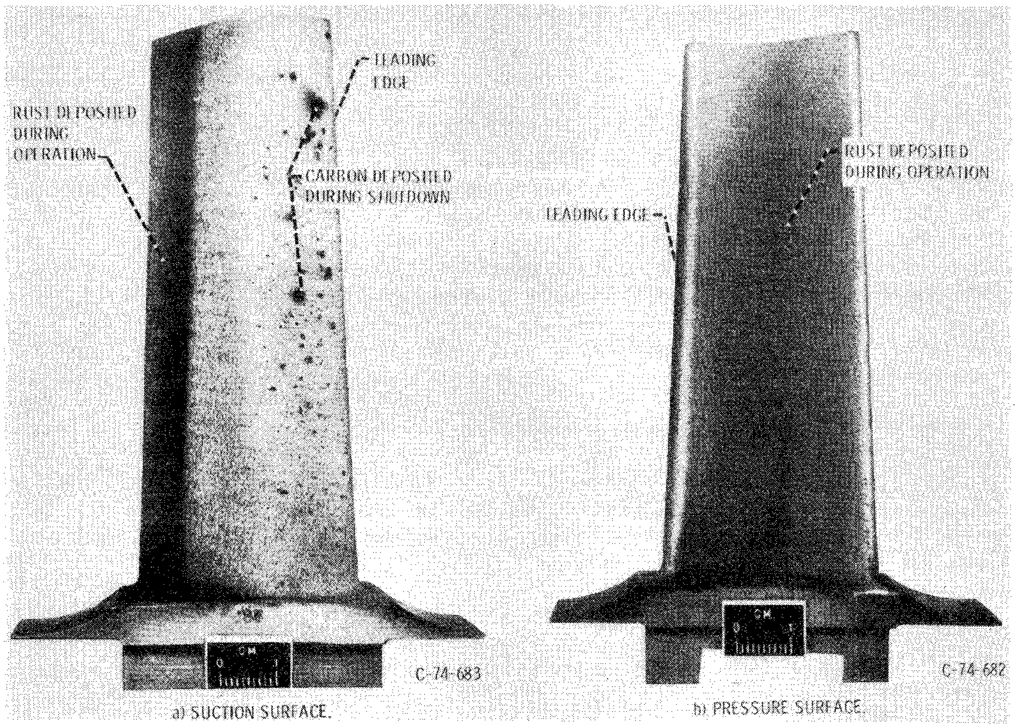

Fig. 5 Ceramic-coated blade after steady-state testing in research engine.

coated blades were chipped at the leading-edge tip, but only one of the 13 magnesia-stabilized zirconia coatings showed this damage. The reason for this apparent greater resistance to chipping is not known.

The inspection after 100 cycles showed that the yttria-stabilized zirconia was completely removed from three blades. The cause for this was the procedure in processing the first group of ten blades. The roughening and cleaning procedure was probably not adequate for the hard, dented, and oxidized surfaces. Also, during application of the bond coat to the first group of blades, particles were observed to intermittently spurt from the plasma gun. This anomaly could also have contributed to the poor coating adherence. The blades processed after the first group were more thoroughly cleaned and roughened; fresh, clean grit was used and more attention was given to impinging the grit normal to the surface and keeping the air pressure at or above 70 N/cm^2. Better control was also maintained on the performance of the plasma-spray feed apparatus.

At completion of 100 cycles, eight blades (including the three blades discussed previously) were removed from the wheel and replaced with calcia-stabilized zirconia coatings processed using the more consistent and carefully controlled coating procedure developed after the coating of the first group of blades. Cyclic testing was then continued for another 200 cycles. Inspection with the unaided eye disclosed no change in coating appearance. The tests were then continued for another 200 cycles and then terminated. The thermal barrier coatings on 66 blades (24 with yttria, 12 with magnesia, and 30 with calcia-stabilized zirconia) completed 500 two-min cycles between full power and flameout without external visual evidence of deterioration except for foreign object damage incurred during the first 100 cycles. The other 8 calcia-stabilized zirconia coatings completed 400 cycles of testing and all coated blades, except for minor foreign object damage, were in as good a condition as the blades run during the steady-state durability tests. During the full-power portion of the cycle the measured leading edge blade metal wall temperatures were about 1200 K and the ceramic external surface temperatures were about 1350 K.

A trailing-edge view of the rotor assembly of the coated blades after conclusion of the cyclic tests is shown in Fig. 6. (The two uncoated blades shown in the figure were used as reference blades for other tests.) The black spots on the blade tips in Fig. 6 are soot deposits that occurred during engine shutdown. The black lines along the span near the root on the suction surface were also caused by soot deposition.

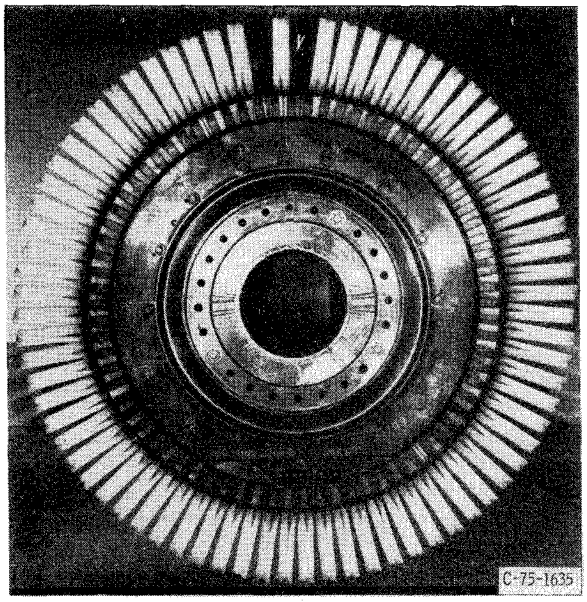

Fig. 6 Ceramic-coated blades after 500 cycles of testing.

Despite foreign object damage, which caused minor chipping at the blade leading edges, the coatings were in very good condition. The chipped areas did not further deteriorate and the exposed bond coat in the chipped regions remained intact for at least 400 cycles of testing. Ceramic coating roughness measurements showed no roughness change over the duration of the tests. In actual usage the roughness should be reduced to improve the durability and aerodynamic performance.

The microstructure (Fig. 7)[12] of the bond coat and the ceramic coatings was metallographically examined on several blades at the leading edge region where durability problems are most likely to occur. The ceramic microstructure consisted

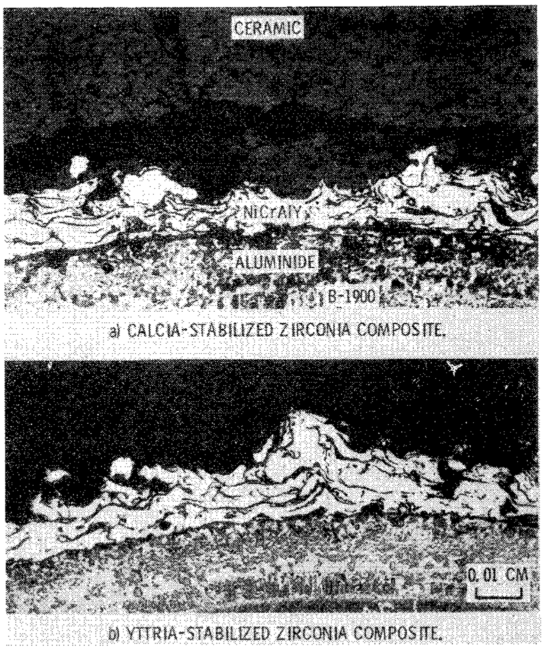

Fig. 7 Microstructure of zirconia composites on turbine blade edge at midspan after cyclic engine tests (× 100).

of solid material connected with a network of fine voids interspersed with larger voids. The photomicrographs (Fig. 7) also showed that the aluminide that was originally present on all of the blades tested in the engine was not completely removed by grit-blasting. Also, the NiCrALY bond adhered well to this aluminide coating. Cracks in the calcia-stabilized zirconia coating tested for 500 cycles were observed on some of the photomicrographs (Fig. 7a). These cracks generally were located parallel and adjacent to the bond coat. In some cases these cracks penetrated to the outer surface of the coating. The formation of such cracks can weaken the coating adherence. These cracks, however, did not cause spalling of the coating.

The microstructure of the magnesia-stabilized zirconia composite was similar to the other composites and the NiCrALY bond coat adhered well to the aluminide coat.

Other Considerations

Control of the coating thickness during deposition of the magnesia-stabilized zirconia was more difficult and the quantity of powder used was about 2 and 4.5 times more than for the calcia- and yttria-stabilized zirconia, respectively. Also, the total processing time for the magnesia-stabilized zirconia (about 35 min) was almost twice as long as for calcia- or yttria-stabilized zirconia. Since the current cost of the magnesia- and yttria-stabilized zirconia powders is about twice that of the calcia-stabilized zirconia, the processing cost per blade for the magnesia-stabilized coating is the highest of the three coatings investigated herein. Based on these considerations and the results of the cyclic tests, the yttria-stabilized zirconia coating was considered the best of the three coatings investigated.

Thermal radiation heat flux was neglected to simplify the analysis used herein. The effects are negligible for the low-pressure conditions of the research engine. However, as gas pressures and temperatures increase, the absorbed radiation heat flux increases, particularly when the part directly views the combustion gases. The higher reflectance of the clean zirconia coating, 0.8 compared to 0.2 for the oxidized metal, provides the additional potential benefit of reducing the radiative heat flux absorbed by the hot parts.

The higher reflectivity of the coating could be particularly beneficial for reducing the metal temperature of such parts as combustor liner walls. The benefit would be in addition to the insulating effect of the coating. Alternatively, maintaining metal temperature could permit the surface of the coating exposed to the gas to operate at higher temperatures than without the coating. This could provide a potential benefit of reducing the amount of unburned hydrocarbons and pollutants by reducing the quenching of the combustion gases at the wall. The ability of the coating to maintain a high reflectivity with prolonged engine operation is not known and needs to be investigated.

It is important to emphasize that the benefits of a thermal barrier coating are directly related to the level of heat flux through the uncoated hardware. As a consequence, hardware or portions of it that are poorly cooled will not show large benefits with a coating. Trailing-edge regions of turbine vanes and blades of small engines, for example, have a physical limitation on the use of effective cooling geometries and thus may not show large benefits with a thermal barrier coating.

Application of thermal barrier coatings to existing hardware could limit the potential benefits and could impose aerodynamic losses because of increased trailing-edge thickness. Coating benefits can best be maximized when the coating is integrated into an original design. The added weight of the coating increases the stress level in rotating parts, which may diminish some of the potential benefits of the coating.

Although the results obtained are encouraging, more testing is required at the high-gas-pressure and high-temperature conditions of advanced core engines, where the coating may be especially susceptible to particle erosion, corrosion, vaporization, thermal fatigue, and thermal shock.

Summary of Results

The following are the results of tests to evaluate the coating insulating capability, aerodynamic performance, durability, and cost, and of calculations made to show the potential benefits of the coating.

1) The zirconia coating reduced the measured midspan leading-edge vane metal temperature in the research engine by 190 K, from 1055 K for the uncoated vane to 865 K for the coated vane. This reduction compared well with analysis. Engine turbine inlet temperature, pressure, and coolant-to-gas flow ratio were 1644 K, 3.0 atm, and 0.6, respectively.

2) Smoothing the surface of the ceramic coating markedly reduced the aerodynamic loss. At a design exit critical velocity ratio of about 0.8 the kinetic energy loss coefficients were 0.062, 0.031, and 0.023 for the rough-coated, smooth coated, and uncoated vanes, respectively.

3) Reductions in metal temperatures of an impingement-cooled vane of as much as 390 K at a constant coolant-to-gas flow ratio of 0.10 were predicted for an advanced core turbine when the vanes were assumed to be coated with a 0.051-cm thickness of zirconia. Turbine inlet temperature and pressure were 2200 K and 40 atm, respectively.

4) Alternatively, large reductions in both coolant flow and metal wall temperature were predicted for coated vanes operating in the advanced core turbine. Vanes coated with a 0.051-cm thickness of zirconia could have both an eightfold decrease in coolant flow and a 110 K reduction in metal temperature compared to the uncoated vane.

5) A calcia-stabilized zirconia coating on cooled turbine vanes and blades withstood 150 hr of steady-state operation including 35 stop and start cycles in a research engine at gas temperatures as high as 1644 K without deteriorating.

6) The coatings on 66 blades (24 with yttria, 12 with magnesia, and 30 with calcia-stabilized zirconia) completed 500 two-min cycles between full power (turbine inlet temperature of 1644 K) and flameout (turbine inlet temperature of 1000 K) in a research turbojet engine.

7) Metallographic examination of the coatings after cyclic testing showed that the NiCrALY bond coat adhered well to the blade wall surfaces. However, cracks were detected in the calcia-stabilized zirconia coating. These cracks did not cause spalling of the coating.

8) Based on material and processing cost and on the results of the cyclic tests, the yttria-stabilized zirconia coating was considered the best of the ceramic coatings investigated herein.

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