SUMMARY OF NASA RESEARCH ON THERMAL-BARRIER COATINGS

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ABSTRACT

This paper summarizes the work conducted at the NASA Lewis Research Center to evolve and evaluate a two-layer, thermal-barrier coating system. A durable, two-layer, plasma-sprayed coating consisting of a ceramic layer over a metallic layer was developed that has the potential of insulating hot engine parts and thereby reducing metal temperatures and coolant flow requirements and/or permitting use of less costly and complex cooling configurations and materials. The investigations evaluated the reflective and insulative capability, microstructure, and durability of several coating materials on flat metal specimens, a combustor liner, and turbine vanes and blades. In addition, the effect on the aerodynamic performance of a coated turbine vane was measured. The tests were conducted in furnaces, cascades, hot-gas rigs, an engine combustor, and a research turbojet engine. Included also are summaries of current research related to the coating and potential applications for the coating.

STAR category 07

IT HAS BEEN RECOGNIZED for a long time that ceramic coatings could insulate engine parts from hot combustion gases. About 25 years ago, ceramic coatings were tried (1, 2)* as a means for reducing metal temperatures of, and providing corrosion protection for, turbine blades. These early tests, however, were made at relatively low turbine inlet temperatures and heat fluxes, and the use of the coatings showed no significant benefits. However, the higher gas temperatures and pressures of current and future gas turbine engines provide environments where benefits may be realized by the use of ceramic thermal-barrier coatings.

Attainment of the higher gas temperatures requires the use of larger amounts of compressor bleed air to cool the hot parts and requires more costly and complex alloys, cooling methods, and cooling configurations. The potential benefits of thermal-barrier coatings include reducing component cooling requirements and metal temperatures and thereby improving engine performance and durability. These considerations plus recent improvements in coating deposition techniques have stimulated a renewed interest in the coating of hot parts.

To be practical, a thermal-barrier coating must have a low thermal conductivity. It should be lightweight and have a higher flame-reflectance than the material it covers. It must also be durable. It should withstand several thousand hours of engine operation at gas temperatures as high as 2200 K without spalling, cracking, or excessively eroding. In addition, component aerodynamics must not be excessively degraded by the presence of the coating. A two-layer, thermal-barrier coating consisting of a stabilized zirconia coating over a metal bond coating, which potentially has these desirable features, has been developed at the NASA Lewis Research Center and is under investigation.

*Numbers in parentheses designate References at end of paper.

This paper summarizes the results of analytical and experimental investigations of thermal-barrier coatings on flat metal specimens, turbine vanes and blades, and a combustor liner. The thermal insulating effect and the durability of the coatings were determined in furnaces, hot-gas rigs, and a research gas turbine engine. The effect of the coating on the aerodynamic performance of a turbine vane was determined in a two-dimensional, cold-air cascade. The results include measured vane and combustor metal temperatures with and without the coating, calculated turbine metal temperatures and coolant flow reductions potentially possible with the coating, and comparisons of predicted metal temperatures of coated and uncoated turbine vanes with those measured in a research engine. Included also are summaries of current research related to the coating and potential applications for the coating.

ANALYTICAL AND EXPERIMENTAL RESULTS

COATED SPECIMENS AND TURBINE BLADES - An air-cooled turbine blade covered with a ceramic thermal-barrier coating is shown in figure 1. The procedure used for depositing the ceramic coating onto the metal substrate was to prepare the substrate surface by grit blasting, to plasma spray on a bond coating, and then to plasma spray on the ceramic coating. Details of the coating procedure and composition are given in references 3 to 5. The thickness of the bond coating was held between 0.003 and 0.010 cm. Flat metal test specimens and turbine blades were used, and the majority of these were coated with stabilized zirconia with thicknesses between 0.025 and 0.064 cm. The zirconia coatings investigated were stabilized with either calcia, magnesia, or yttria. A limited amount of testing was conducted initially with alumina-, hafnia-, and calcia-stabilized zirconia. Bond coatings used were Nichrome, molybdenum, and nickel aluminum. The substrate materials on which the coatings were applied included nickel-base alloys, Inconel alloy 718 HT, directionally solidified MAR-M-200 with Hf, MAR-M-200 with Hf, Hastelloy X and B-1900 with and without Hf. The cobalt alloys MAR-M-509 and directionally solidified (DS) MAR-M-302 were also coated.

Initial tests of the coating adherence were conducted in a commercial electric furnace. In the initial tests (3), coated specimens were placed in the furnace for 10 minutes and then forced-air cooled to 300 K. The furnace temperature was 1367 K for tests with coated flat specimens and 1088 to 1394 K for tests with coated turbine vanes and blades. The flat sheet material was Inconel alloy 718 HT and the vane and blade materials were MAR-M-302 and B-1900, respectively. The results of these tests, although run for a relatively small number of cycles (up to 40), gave encouragement on the adherence capability of a coating of calcia-stabilized zirconia ceramic and Nichrome bond. The tests helped establish the surface cleaning and coating procedure and disclosed that the ceramic-bond interface has a limiting temperature of about 1367 K for good adherence.

Analyses were made and tests conducted (3) to determine the benefits attainable with coated turbine vanes and blades. A simple one-dimensional, steady-state heat balance through a composite wall consisting of the ceramic, the bond coating, and the metal wall was used. It was applied at the gas and coolant conditions of turbines of an existing research engine and of a core turbine of an advanced turbofan engine. The results of the analysis for the existing research engine are shown in figure 2. The figure shows the predicted metal temperature at the leading edge of vanes with and without a thermal-barrier coating over a range of coolant-to-gas flow ratios. Also shown in the figure are the leading-edge metal temperatures of coated and uncoated vanes as measured in the research engine. The thickness of the zirconia coating was 0.028 cm. The result show good agreement (within 3.5 percent) between prediction and measurement and that large reductions in metal temperatures can be obtained with the coating. At a coolant-to-gas flow ratio of 0.06, for example, the metal temperature was reduced by 190 K, from 1055 K for the uncoated vane to 865 K for the coated vane.

Greater benefits of the thermal-barrier coating occur at the high heat fluxes of highly cooled hardware that exist in the high-gas-temperature-and-pressure environments of core turbines of advanced turbofan engines. Predicted bulk turbine-vane metal temperatures (integrated average temperature over the entire vane) as affected by coolant-to-gas flow ratios and ceramic coating thickness are shown in figure 3. At constant coolant-to-gas flow ratios, bulk wall metal temperatures were substantially reduced as the ceramic coating thickness was increased. The figure shows that the metal temperatures could be reduced by as much as 390 K (point A to point B) at a coolant-to-gas flow ratio 0.10 when the vanes were coated with a 0.051-cm thickness of zirconia. Alternatively, vanes coated with a 0.051-cm thickness of zirconia could have both an eightfold decrease in coolant-to-gas flow ratio (point B to point C) and a 110 K reduction in metal temperature (point A to point C) as compared with the uncoated vane. The coolant-to-gas flow ratio was reduced from 0.10 for an uncoated vane to 0.0125 for a coated vane with a corresponding vane metal temperature reduction of 1390 K to 1280 K.

The dashed portions of the curves in the figure illustrate a limitation associated with using the current ceramic composite 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 approaches 1367 K.

Another calculation was made to compare flow requirements for cooling a turbine to a given metal temperature with three different cooling configurations: a convection-cooled turbine; a full-coverage, film-cooled turbine; and a thermal-barrier-coated, convection-cooled turbine. The results obtained from (6) are shown in figure 4. Comparison shows that the thermal-barrier-coated turbine has the potential for operating at coolant flows similar to those of a full-coverage, film-cooled turbine. A ceramic thickness of 0.038 cm was assumed in the analysis.

Aerodynamic Performance Tests - Early in the coating development program, tests were conducted (7) to determine the effect of the coating on the aerodynamic performance of turbine vanes. The tests were conducted in a two-dimensional, cold-air cascade on an uncoated, an as-spray-coated, and a smoothed coated vane. The results of these tests showed that the kinetic energy loss coefficient for the as-spray-coated vane was about 4 points larger than that for the uncoated vane at a design exit velocity ratio of about 0.8. Smoothing the coating with aluminum oxide reduced its surface roughness (from 8.9 to 2.3 μm) and eliminated most of the loss. The smoothed coated vane had a kinetic energy loss coefficient that was still 0.7 larger than that for the uncoated vane. This loss, however, was attributed to the 38-percent-thicker trailing edge caused by the presence of the coating.

Furnace Tests - Long-duration cyclic durability tests were run with coated, flat metal specimens in a furnace. In these tests (5) the coated specimens were heated in air to 1248 K for 1 hour (specimens reached temperature in about 4 min) followed by furnace cooling to 553 K within 1 hour. The substrate materials were turbine-blade superalloys in the form of small thin-sheet specimens. The materials included B-1900 with Hf, directionally solidified (DS) MAR-M-200 with Hf, conventionally cast MAR-M-200 with Hf, and MAR-M-509. The coatings were calcia-, magnesia-, or yttria-stabilized zirconia (ZrO_2-5.4CaO, ZrO_2-3.4MgO, and ZrO_2-12Y_2O_3, respectively) over a bond coating of Ni-16Cr-6Al-0.6Y. The results of these tests are summarized in table 1. The data in the table show that ZrO_2-$12Y_2O_3$ was the most adherent, followed by ZrO_2-3.4MgO, partially stabilized ZrO_2-5.4CaO, and zirconia totally stabilized with CaO. The data in table 1 show that the ZrO_2-$12Y_2O_3$ thermal-barrier coatings completed 558 to 673 cycles between 1248 K and 553 K without failure. The ZrO_2-3.4MgO coating failed in 460 cycles or less, the partially stabilized ZrO_2-5.4CaO coating in less than 255 cycles, and the totally stabilized ZrO_2-5.4CaO coating in less than 87 cycles. Furthermore, it is evident from the data in this table that the properties of the substrates (such as coefficient of thermal expansion) apparently had little effect on the adherence or performance of the coatings. The ZrO_2-MgO and ZrO_2-CaO thermal-barrier coatings generally failed within the oxide layer very close to the bond coating. The failures always started with the formation of a small, visible crack at one of the corners (a region of high stress) of the specimens, and the crack propagated along the edges.

Examination of the microstructures of ZrO_2-$12Y_2O_3$ thermal-barrier coatings disclosed no failures but revealed that the thickness of the coating decreased during the 673-hour exposure. Since the coating varied in thickness from sample to sample, no definite estimate was made on the extent of the degradation. Examinations of micrographs showed that even after 673 hours at 1248 K the NiCrAlY bond coating was still intact.

Engine Tests - A research turbojet engine was used to evaluate the durability of the coatings in a gas turbine environment. The vanes and blades were air cooled and were made from cast MAR-M-302 and cast B-1900, respectively.

Steady-state durability tests: Tests of the durability of a 0.028-cm-thick calcia-stabilized zirconia coating on two vanes and two blades were conducted (3) 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 3.039×10^5 pascals. 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. At the completion of 150 hours of test time, including 35 start-and-stop cycles, inspection of the coatings showed no evidence of deterioration.

Cyclic durability tests: Tests of the durability of three zirconia coatings that were stabilized by either calcia, magnesia, or yttria were conducted (8) on 74 air-cooled turbine blades in a research turbojet engine. The calcia-stabilized zirconia was on 38 blades, the magnesia on 12 blades, and the yttria on 24 blades. The ceramic coatings were about 0.038 cm thick, and the NiCrAlY bond coatings were about 0.010 cm thick. Control of the desired cyclic conditions was accomplished primarily by controlling the combustor fuel supply. Adjustments were made so that the gas temperature, which was held at 1644 K (full power) for about 70 seconds, would be decreased in 20 seconds to 1000 K (flameout). The combustor was then reignited, and the engine reached full-power conditions in 30 seconds.

A total of 500 of these two-minute cycles were run. The engine was stopped for visual inspection of the coatings at 100, 300, and 500 cycles. At the completion of 100 cycles, eight coated blades were removed from the wheel for inspection and replaced with other calcia-stabilized-zirconia-coated blades. The coatings on 66 blades completed 500 cycles, and those on the other 8 blades completed 400 cycles. All coated blades (except for minor foreign object damage incurred during the first 100 cycles) showed no unmagnified visual evidence of deterioration.

A trailing-edge view of the rotor assembly of the coated blades after conclusion of the tests is shown in figure 5. The two uncoated blades shown in the figure were used as reference blades for other tests. The black spots on the blade tips are soot deposits that accumulated during engine shutdown. The black lines along the span near the root on the suction surface were also caused by soot deposition.

The microstructures of the bond coating and the ceramic coatings were metallographically examined on several blades at the leading-edge region, where durability problems are most likely to occur. The ceramic microstructure as seen in figure 6 consisted of solid material connected with a network of fine voids interspersed with larger voids. The photomicrographs also show that the aluminide that was originally present on all blades tested in the engine was not completely removed by grit blasting. Also, the NiCrAlY bond coating adhered well to this aluminide coating. Cracks in the calcia-stabilized zirconia coating tested for 500 cycles were observed on some of the photomicrographs (e.g., fig. 6(a)). These cracks generally were located parallel and adjacent to the bond coating. 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. Cracks were not found in the yttria-stabilized (fig. 6(b)) and magnesia-stabilized zirconia coatings. The microstructures of all these composites were similar and showed that the NiCrAlY bond coating adhered well to the aluminide coating.

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.

Hot-Gas-Rig Tests - In these tests (5), hot gas at a Mach number of 0.3 impinged on the leading edge of a coated test blade held in a fixture as shown in figure 7. Air was used to cool the blade wall to desired metal temperature levels. The blade inner wall, outer wall (or substrate), and coating surface temperatures were measured during the tests. The details of the instrumentation are described in reference 5. In these tests the coated blades were subjected to cyclic heating by inserting them into the gas stream and then withdrawing them into ambient air. Two cycle times were investigated. In the short cycle the hold time in the hot-gas stream was 80 seconds, and in the long cycle the hold time was 1 hour. The time for heatup or cooloff in both these cycles was about 30 seconds.

Short-cycle tests: Calcia-, magnesia-, and yttria-stabilized coatings on eight air-cooled turbine blades were evaluated in these tests. The results in figure 8 show that the yttria-stabilized zirconia was superior to the other two coatings. The 0.028-cm thick yttria-stabilized zirconia satisfactorily completed 2000 cycles at a surface temperature of 1463 K when testing was stopped. The magnesia-stabilized coating of the same thickness completed 1010 cycles, as shown in figure 8(a), when testing was stopped due to excessive erosion. The results with the 0.038-cm-thick coatings at a coating surface temperature of 1553 K (fig. 8(b)) show that the yttria-stabilized coating completed 3200 cycles before testing was stopped because of excessive erosion. The tests of the calcia-stabilized zirconia were stopped after only 200 cycles because of failure near the ceramic-bond material interface.

Long-cycle tests: Based on the superior performance of the yttria-stabilized zirconia coating in the short-cycle tests, additional testing was limited to this coating. The results of the long-cycle tests of 0.051-cm-thick coatings at surface temperatures of 1683, 1713, 1753, 1813 K (and corresponding substrate (blade outside wall) temperatures of 1173, 1188, 1198, and 1233 K) are shown in figure 9. These results show that the coating had completed 246 1-hour cycles at 1683 K before the test was stopped because of excessive erosion. The erosion was considered excessive when visual observation indicated a 40- to 50-percent loss of the coating thickness. The figure shows that coating life is significantly reduced with increased surface or interface temperatures. Increasing the surface temperature by 130 K (substrate temperature by 60 K) resulted in a fivefold decrease in cyclic life (from about 250 to 50).

High Cycle Fatigue Tests - Tests were run to determine the effect of a thermal-barrier coating on the high cycle fatigue life of turbine blades. These tests were run as a part of a study to determine the advisability of installing several coated turbine blades in an advanced high-gas-temperature-and-pressure turbofan test engine. One blade with a 0.036-cm-thick, yttria-stabilized zirconia thermal-barrier coating and 3 uncoated blades were subjected to high cycle stress loading while being heated in an electric furnace to 1033 K. The results of these tests, wherein the blades were run until failure, showed that the coated blade had about 10 times greater fatigue life than the uncoated blades. The failure origin of the coated blade was in the blade metal wall material, and the coating did not appear to assist in the initiation or propagation of the failure. Also, the coating did not spall in any area of the blade. The reason for the substantial improvement in the fatigue life of the coated blade as compared with the uncoated blades has not been determined.

COATED COMBUSTOR LINER - The effect of the thermal-barrier coating on combustor liner performance was also investigated (9). The tests were run in a laboratory rig and were of short duration (about 6 hr). The tests were conducted with a single-can combustor from the JT8D engine over a range of gas temperatures and pressures that included engine idle, cruise, and takeoff. Tests were run with two fuels of widely different aromatic content. One fuel was ASTM A-1 (Jet A), and the other was a blend of Jet A and a mixture of single-ring aromatic compounds. The aromatic content of the Jet A was 17 percent and that of the mixture was 60 percent. The thermal-barrier-coated combustor liner is shown in figure 10. The liner material was Hastelloy X (about 0.097 cm thick), the bond coating was NiCrAlY (about 0.010 cm thick), and the ceramic coating was yttria-stabilized zirconia (about 0.025 cm thick).

The effect of the coating was to significantly reduce metal liner temperature and flame radiation at cruise and takeoff conditions. Small reductions in exhaust smoke concentration were also observed. The differences in concentrations of gaseous pollutants and in combustion efficiency with and without the coating on the liner were generally small and often within the limits of the accuracy of measurements. Maximum liner temperatures as a function of average exhaust-gas temperature for Jet A fuel are shown in figure 11. At an exhaust-gas temperature of 1325 K, representative of takeoff conditions, the maximum liner temperature was reduced from about 1220 K for the uncoated liner to about 1060 K for the ceramic-coated liner. At an exhaust-gas temperature of 1126 K, representative of cruise, the maximum liner temperature was reduced from about 1050 K to about 920 K through the use of the ceramic coating. The maximum liner temperatures were higher with the high-aromatic-content Jet A fuel, but the liner metal temperature reductions with the coating was approximately the same with both fuels. The maximum uncoated liner temperature at takeoff with the blended Jet A was about 1265 K as compared with 1050 K with the coated liner.

The results of measurements of flame radiation with and without the thermal-barrier coating on the liner over a range of average exhaust-gas temperature for Jet A fuel are shown in figure 12. The flame radiation was measured with a commercially available radiometric microscope using an unimmersed bolometer thermal detector with a sensitivity range from 0.25 to 6 μm. The flame was viewed from a single port through an air-cooled sapphire window in the primary combustion zone. The results in figure 12 show substantial reductions in flame radiation with the ceramic coating. For example, at takeoff conditions (1325 K), flame radiation was reduced from 6.9 to 6.1 $W/(cm^2/sr)$. It is believed that the radiation reflected from the ceramic-coated liner walls back into the flame is effective in burning up a substantial portion of the soot particles in the flame. Since the hot soot particles account for most of the flame radiation, any reduction in soot concentration due to burnup should reduce flame radiation. Flame radiation values obtained with the high-aromatic-content Jet A and the uncoated liner were considerably higher than those obtained with the lower-aromatic-content Jet A fuel. The radiation obtained with the ceramic-coated liner was approximately the same for both fuels. The result again was a substantial reduction in flame radiation due to the higher reflectance of the ceramic coating as compared with the bare metal wall.

The effect on smoke concentration in the exhaust gases with coated and uncoated combustor liners was measured by using the SAE-recommended practice. This consists of passing metered volumes of exhaust gas through a filter paper. Then the darkness of the stain on the paper is measured by optical means. The measurement is an indication of the relative concentration of smoke (soot) in the gas sample. The results of these measurements showed that coated liners slightly reduced smoke concentration. For example, at conditions simulating takeoff with Jet A fuel, the SAE smoke number was reduced from 33.2 for the uncoated liner to 28.5 for the coat coated liner.

CURRENT RESEARCH

Each of two gas turbine manufacturers is currently performing analytical studies to evaluate the benefits of using thermal-barrier coatings on combustor and high-temperature gas turbine components and to identify a research and technology plan to develop thermal-barrier coatings for commercial engine demonstration. The studies are directed at engines used in simple and combined cycles for production of electric power. Initial results suggest that thermal-barrier coatings on turbine airfoils may provide the potential to eliminate the use of leading-edge film cooling and the holes that might become clogged if residual fuels are used. The results also indicate significant potential improvements in specific power output, heat rate, and thermal efficiency.

Studies are continuing to evaluate the endurance of the coatings in burner rigs for long periods of time with clean and residual fuels. Experiments are under way to determine changes, if any, of the thermal conductivity and emissivity of the coatings with test environment, fuel composition, and time. Studies are also under way to optimize the coating in terms of chemistry and deposition parameters and to characterize the mechanical properties of the coating composite and its effect on metal substrates.

Experimental tests are also being conducted in a cascade rig to measure local metal temperatures around the periphery of an all-impingement-cooled vane with the thermal-barrier coating as compared with its temperatures without the coating.

Lastly, in order to overcome such factors as the variability in deposited thickness and composition caused by manual coating application, a contractual study is currently under way to determine the feasibility of developing an automated system to control plasma spray deposition of the NASA, two-layer, thermal-barrier coating on gas turbine blades. The system to be evaluated will integrate a multiaxis blade-handling fixture, an optical instrument for coating thickness monitoring, the plasma spray equipment, and a microprocessor-based system controller.

POTENTIAL APPLICATIONS OF CERAMIC COATING

Substantial interest has been expressed by industry in the NASA-developed, thermal-barrier coating. Airline and aircraft engine companies have expressed interest in using the coating on turbines and combustors for reducing metal temperatures to improve engine durability and/or for reducing coolant flows to improve engine performance. Manufacturers of gas turbine engines for electric power generation have expressed interest for similar purposes. Manufacturers of reciprocating engines, particularly diesel engines, have expressed interest in the coating for use on piston heads, valves, and exhaust parts to reduce metal temperatures, heat flux, and pollution. Interest has also been expressed in the coating for use on the space shuttle, in a rapid-earth-drilling scheme, on turbine shrouds and seal surfaces, and in thin-film thermocouple applications.

In many of these areas, when the potential results have been assessed to be to the benefit of the government, NASA has coated a small quantity of parts that have been or will be tested under the various application environments. Initial data from completed tests indicate the results to be as good or better than expected. In some of these cases, where subsequent coating and testing or large parts or large numbers of parts were desired, a nonexclusive license on the coating process has been obtained by the companies from NASA. When the various tests are completed, a report summarizing the results will be published.

CONCLUDING REMARKS

The results of the investigation of the reflective and insulating capability, microstructure, and durability of a thermal-barrier coating and its small effect on aerodynamic performance have indicated a potential for its application in high-gas-temperature environments such as those of gas turbine engines. The best of the coatings investigated based on durability and material and processing costs was yttria-stabilized zirconia ceramic (0.028 to 0.064 cm thick) over a NiCrAlY bond coating (about 0.010 cm thick).

Analyses suggested that large reductions in metal temperatures and/or coolant flow requirements are possible with the coating. Measured turbine-vane metal temperatures in a research turbojet engine (which agreed closely with predicted values) indicated metal temperature reductions of about 190 K by using the coating. Analytically, the coolant flow requirement for a convection-cooled turbine with a thermal-barrier coating was similar to that required for a more complex full-coverage, film-cooled turbine without a coating.

Using the coating on the liner of an engine combustor gave significant reductions in metal temperatures and flame radiation and small reductions in exhaust-gas smoke concentration. For simulated takeoff conditions with Jet A fuel, for example, the maximum liner temperature was reduced from 1223 to 1058 K, flame radiation from 6.9 to 6.1 W/(cm^2/sr), and the SAE smoke number from 33.2 to 28.5.

The coatings completed significant test times and cycles in hot-gas environments without spalling. No deterioration was noted in engine testing, and only erosion of the coating was observed in severe test conditions in a hot-gas rig. For example, the coatings completed 150 hours of steady-state operation in a research engine at gas temperatures as high as 1644 K and 500 2-minute cycles in the research engine between turbine inlet gas temperatures of 1644 and 1000 K. The coatings also completed 673 1-hour cycles at 1248 K in a furnace, and 3200 short cycles (80 sec at 1553 K coating surface temperature) and 240 long cycles (1 hr at 1683 K coating surface temperature) in and out of a hot-gas rig.

Metallographic examination of the coating after various tests indicated that the NiCrAlY bond coating adhered well to the blade wall surfaces. Cracks were not found in the yttria- or magnesia-stabilized zirconia but were detected in the calcia-stabilized zirconia coating. The yttria-stabilized zirconia also showed less erosion of the surface than did the magnesia-stabilized zirconia.

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FIGURE CAPTIONS

Fig. 1 - Thermal-barrier-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, 3.04×10^5 pascals; coolant temperature, 319 K.

Fig. 3 - Calculated metal temperatures for turbine vane coated with thermal-barrier ceramic. (Advanced core turbine: inlet gas temperature, 2200 K; gas pressure, 40.5×10^5 pascals; coolant temperature, 811 K.)

Fig. 4 - Cooling requirements for several cooling methods.

Fig. 5 - Thermal-barrier-coated turbine blades after 500 cycles of testing.

Fig. 6 - Microstructure of zirconia composites on turbine blade leading edge at midspan after cyclic engine tests. X150. (Ref. 12.)

Fig. 7 - Hot-gas test rig with thermal-barrier-coated air-cooled test blade.

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Fig. 8 - Cyclic performance of stabilized zirconia thermal-barrier coatings on air-cooled turbine blades in a hot-gas rig. Substrate temperature, 1188 K. Figure determined on basis of visual observation of coating thickness.

Fig. 9 - Yttria-stabilized zirconia thermal-barrier coating lives at high surface temperature. Oxide thickness, 0.051 centimeter; failures determined on basis of visual observation of coat thickness loss (40 to 50 percent) by erosion.

Fig. 10 - Thermal-barrier-coated combustor liner.

Fig. 11 - Effect of thermal-barrier coating on maximum liner temperatures; fuel, Jet A.

Fig. 12 - Effect of thermal-barrier coating on flame radiation; fuel, Jet A.

Table 1. - Cyclic furnace evaluation of various zirconia thermal-barrier coatings on Ni-16Cr-6Al-0.6Y bond coating

Alloy	Cycles to failure[a] - First visible crack, spall, etc.			
	ZrO_2-12Y_2O_3	ZrO_2-3.4MgO	ZrO_2-5.4CaO[b]	ZrO_2-5.4CaO[c]
DS MAR-M-200 + Hf	[d]673	460	255	78
MAR-M-200 + Hf	[d]650	450	255	87
MAR-M-509	[d]558	450	196	76
B-1900 + Hf	[d]628	438	226	--

[a]Cycle, 1 hr at 1248 K and 1 hr to cool to 553 K.
[b]Partially stabilized zirconia derived from ZrO_2 and $CaCO_3$ spray powders (cubic and monoclinic phases).
[c]Totally stabilized zirconia derived from stabilized spray powder (cubic phase).
[d]No failure observed.

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Fig. 1 - Thermal-barrier-coated turbine blade.

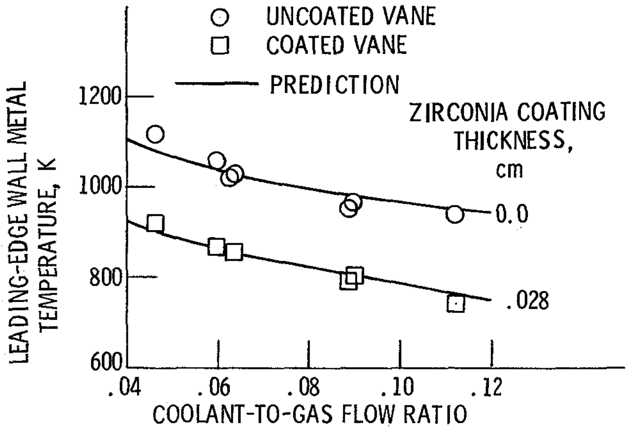

Fig. 2 - Comparison of calculated and measured midspan leading-edge wall metal temperatures of uncoated and zirconia-coated turbine vanes operating in a research engine. Inlet gas temperature, 1644 K; inlet gas pressure, 3.04×10^5 pascals; coolant temperature, 319 K.

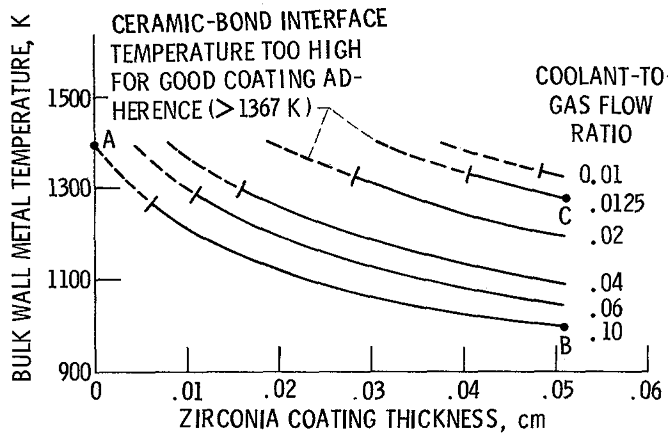

Fig. 3 - Calculated metal temperatures for turbine vane coated with thermal-barrier ceramic. (Advanced core turbine: inlet gas temperature, 2200 K; gas pressure, 40.5×10^5 pascals; coolant temperature, 811 K.)

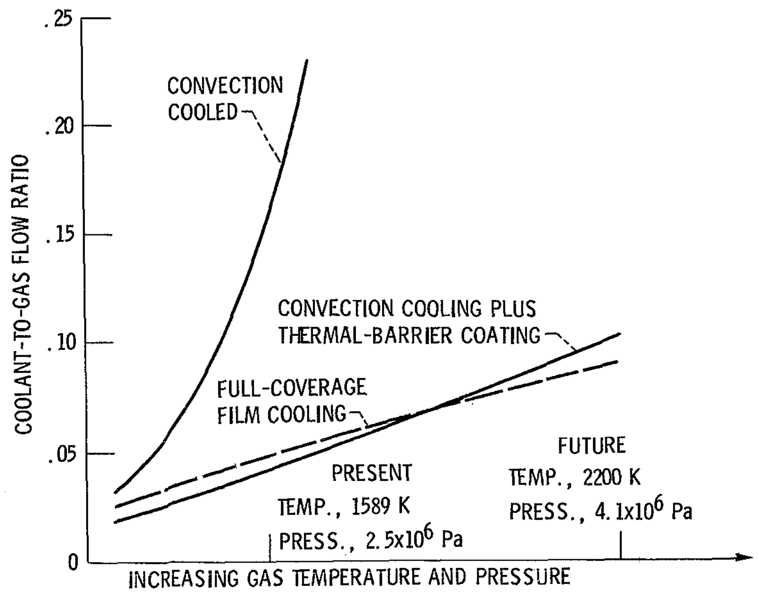

Fig. 4 - Cooling requirements for several cooling methods.

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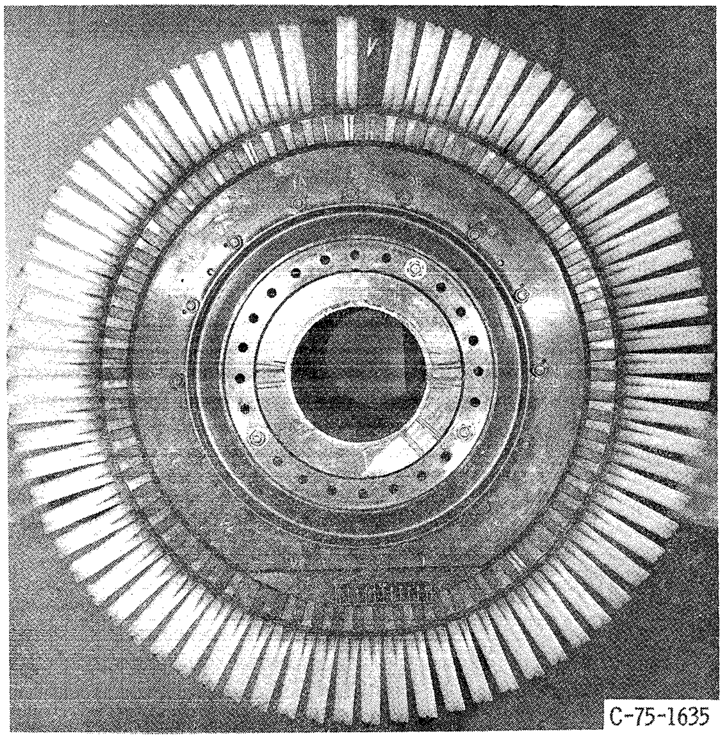
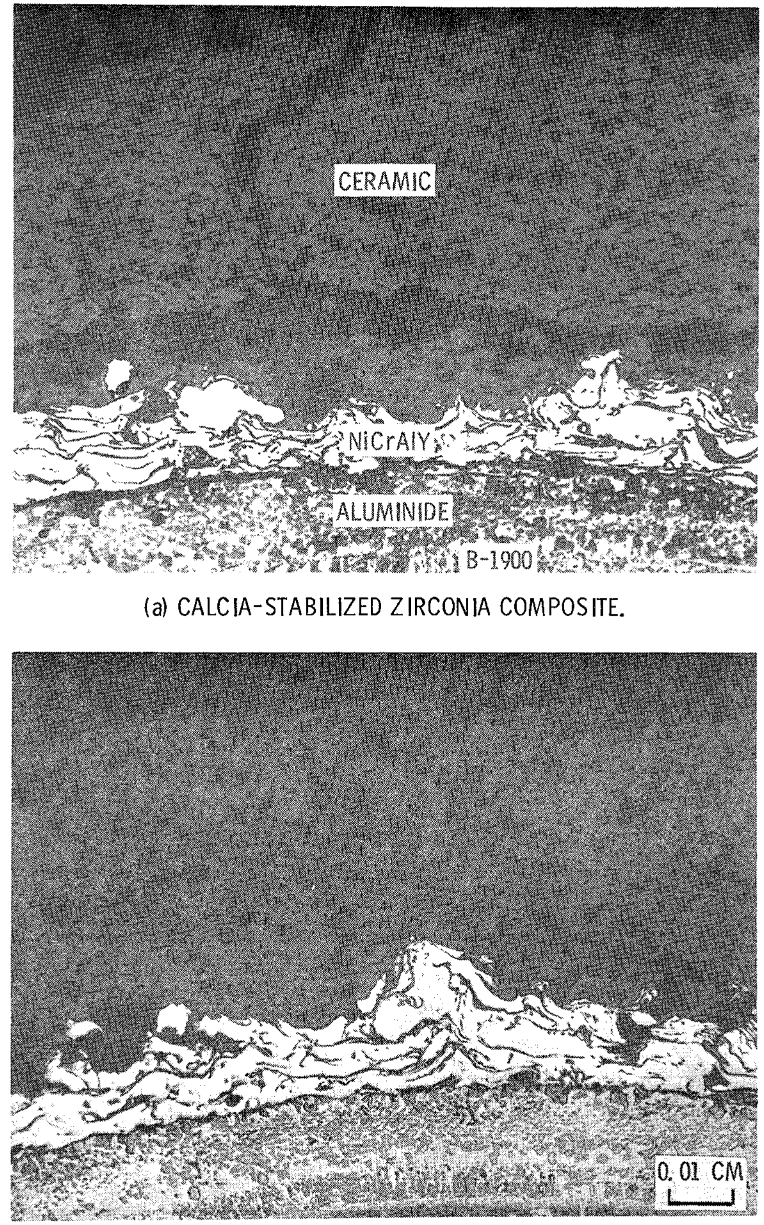


Fig. 5 - Thermal-barrier-coated turbine blades after 500 cycles of testing.



(a) CALCIA-STABILIZED ZIRCONIA COMPOSITE.

(b) YTTRIA-STABILIZED ZIRCONIA COMPOSITE.

Fig. 6 - Microstructure of zirconia composites on turbine-blade leading edge at midspan after cyclic engine tests. X150.

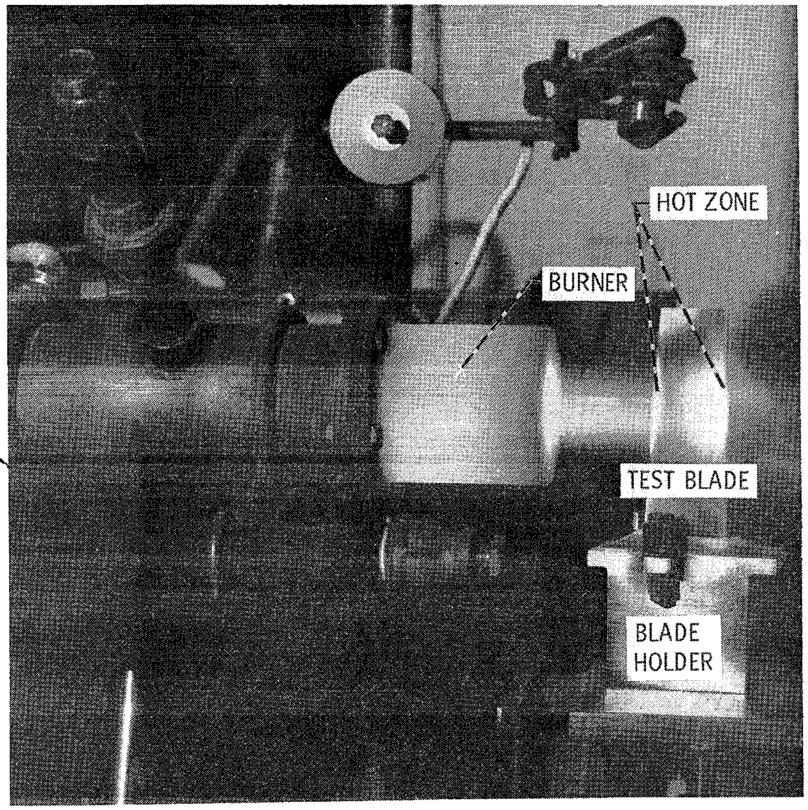
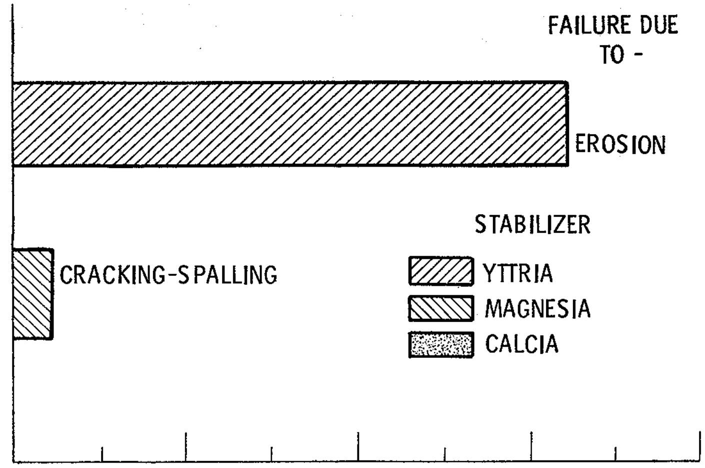
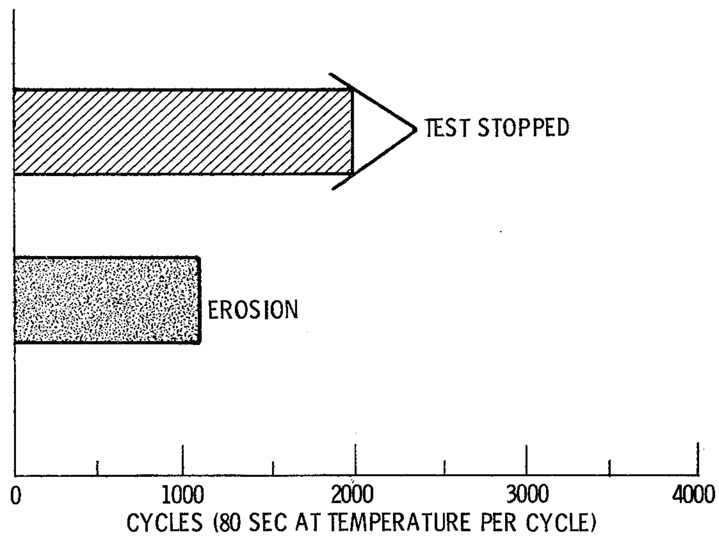


Fig. 7 - Hot-gas test rig with thermal barrier-coated air-cooled test blade.



(a) SURFACE TEMPERATURE, 1553 K; MAXIMUM OXIDE LAYER THICKNESS, 0.038 CENTIMETER.



(b) SURFACE TEMPERATURE, 1463 K; MAXIMUM OXIDE LAYER THICKNESS, 0.028 CENTIMETER.

Fig. 8 - Cyclic performance of stabilized zirconia thermal-barrier coatings on air-cooled turbine blades in a hot-gas rig. Substrate temperature, 1188 K. Failure determined on basis of visual observation of coating thickness.

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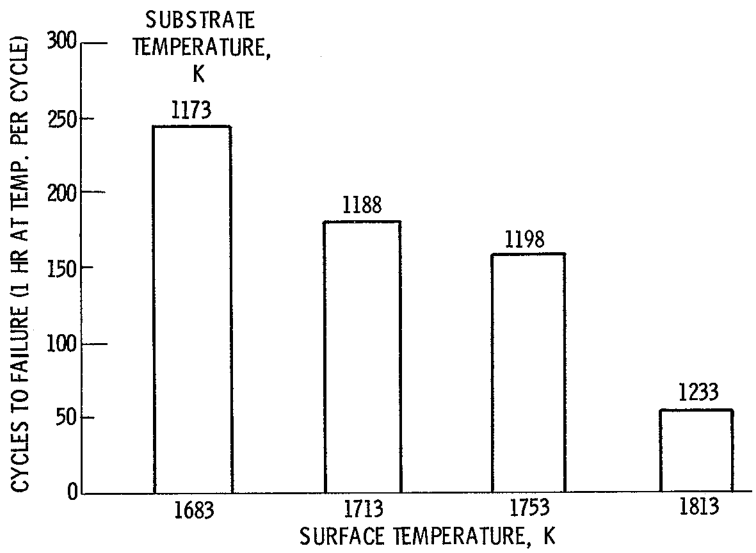

Fig. 9 - Yttria-stabilized zirconia thermal-barrier coating lives at high surface temperatures. Oxide thickness, 0.051 centimeter; failures determined on basis of visual observation of coat thickness loss (40 to 50 percent) by erosion.

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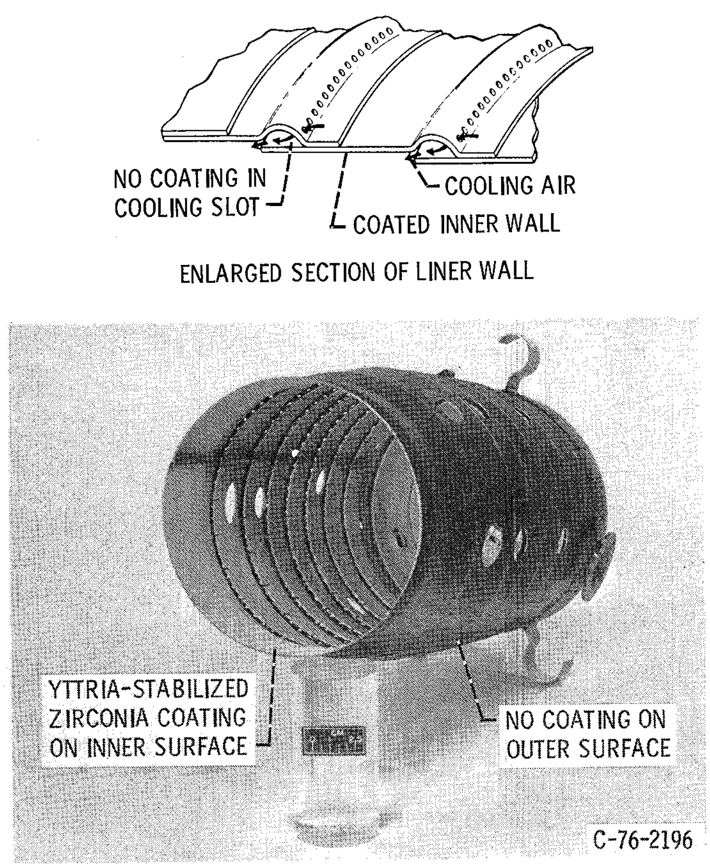

Fig. 10 - Thermal-barrier-coated combustor liner.

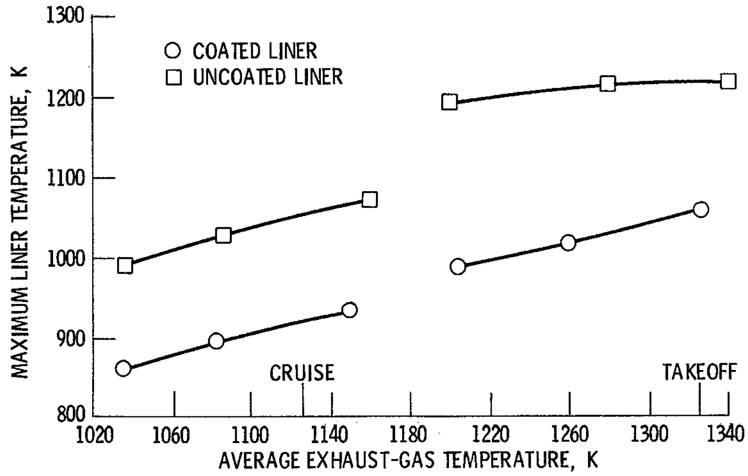

Fig. 11 - Effect of thermal-barrier coating on maximum liner temperatures; fuel, Jet A.

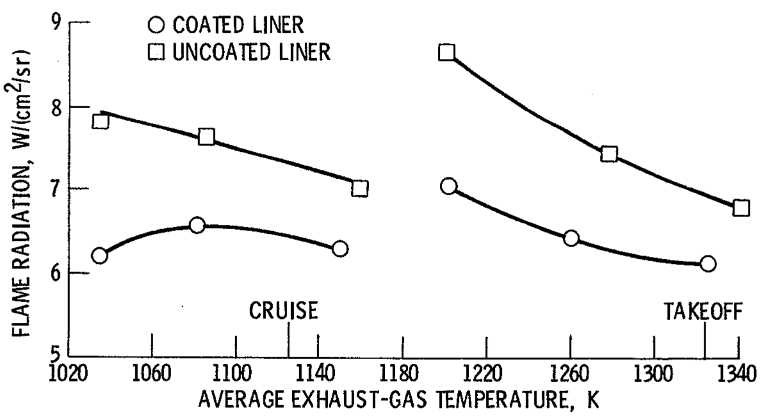

Fig. 12 - Effect of thermal-barrier coating on flame radiation; fuel, Jet A.

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