NASA Technical Paper 1425

Industry Tests of NASA Ceramic Thermal Barrier Coating

Curt H. Liebert and Francis S. Stepka

JUNE 1979

NASA

NASA Technical Paper 1425

Industry Tests of NASA Ceramic Thermal Barrier Coating

Curt H. Liebert and Francis S. Stepka
Lewis Research Center
Cleveland, Ohio

National Aeronautics
and Space Administration

**Scientific and Technical
Information Office**

1979

SUMMARY

A NASA Lewis Research Center ceramic thermal barrier coating (TBC) system was tested by industrial and governmental organizations for a variety of aeronautical, marine, and ground-based gas-turbine engine applications. This TBC is a two-layer system with a bond coating of nickel-chromium-aluminum-yttrium (Ni-16Cr-6Al-0.6Y, in wt. %) and a ceramic coating of yttria-stabilized zirconia (ZrO_2-12Y_2O_3, in wt. %).

Twelve tests of the coating system were conducted. Seven of these tests evaluated the thermal protection and durability. The other five tests determined the thermal conductivity, high cycle fatigue characteristics, and corrosion protection of the system.

The TBC lowered metal temperatures, increased metal part life, and eliminated the burning, melting, and warping that occurred with some of the uncoated metal parts tested in this cooperative effort. For example, an uncoated turbine vane of a military aircraft engine and uncoated combustor dome, scroll, and turbine nozzle shroud parts of a military ground-based engine locally burned and melted in short testing times. However, a coated turbine vane of identical design satisfactorily completed longer testing times of over 300 hours in accelerated cyclic endurance tests (10 800 cycles), and the coated combustor dome, scroll, and turbine nozzle shroud parts of the same design satisfactorily completed 500 hours of testing. In another test on a gas-turbine plenum for a vehicular engine, the coating eliminated base metal warping and reduced hot-spot temperatures by 30 to 100 K.

During the thermal protection and durability tests, local loss of the ceramic layer occurred on some coated parts in regions where uncoated metal parts had previously melted. However, in these areas of coating loss there was no melting, burning, or other deterioration of the base metal. Also, during the corrosion protection and durability tests, the coating on some specimens cracked and spalled. However, metallographic examination showed no base metal deterioration. This demonstrated that local coating loss will not necessarily result in damage to metal parts.

The results suggest that parts should be subjected to a vacuum heat treatment after TBC deposition to increase the durability of the TBC. In addition, sharp metal edges should be rounded before coating to reduce stress and minimize ceramic spallation.

INTRODUCTION

Extensive testing is needed to determine the industrial potential of the ceramic thermal barrier coating (TBC) system recently developed at the NASA Lewis Research

Center (refs. 1 and 2). A cooperative program for this purpose was initiated with industry. Tests were conducted to determine the thermal conductivity, high cycle fatigue characteristics, thermal protection and durability, and corrosion protection of the system. This thermal-barrier coating is a two-layer system with a bond coating of nickel-chromium-aluminum-yttrium (Ni-16Cr-6Al-0.6Y, in wt. %) and a ceramic coating of yttria-stabilized zirconia (ZrO_2-$12Y_2O_3$, in wt. %). The technology which helped develop and apply this coating system is summarized in reference 3.

Thermal steady-state and cyclic tests (refs. 4 to 7) and mechanical cyclic tests (ref. 8) on the TBC were previously conducted at Lewis. The TBC was used on cooled blades and vanes in a gas-turbine engine, on turbine combustor liners, on cooled turbine blades in the exhaust gases of a burner rig, on uncooled metal specimens in a furnace, and in a fatigue testing machine using resistance heating of the specimens. These tests demonstrated that the coating has good durability and provides large temperature reductions of the metal parts. During the burner rig tests, described in references 5 and 7, the TBC did not spall at metal wall temperatures up to 1200 K, but erosion of the coating did occur. This erosion was attributed to carbon particles in the jet A fuel combustion gases. Tests in a "clean" combustion gas environment (using natural gas as fuel) gave no evidence of coating erosion (ref. 7). Tests in a research gas-turbine engine (ref. 4) demonstrated that the TBC reduced vane metal temperatures by as much 190 K. Also, the thermal protection provided by the TBC could decrease coolant flow requirements or increase gas temperatures. Increasing gas temperatures or decreasing coolant flow can decrease the fuel consumption and/or increase the gas-turbine engine performance.

Tests in a can-annular combustor (ref. 6) and in an annular combustor (ref. 9) showed that the TBC reduced the film-cooled liner metal temperatures by 200 and 100 K, respectively. The metal temperatures were reduced because the yttria-stabilized zirconia is highly reflective and thermally insulating when compared to the metal combustor liner materials. The radiation reflected from the ceramic-coated walls back to the combustor flame also reduces soot pollution by reducing the amount of soot initially formed or by burning the soot formed (ref. 6). More recent tests on coated, impingement-cooled combustor walls (refs. 10 to 12) show that the TBC can also reduce pollutant emissions because the hotter ceramic surface, compared to a bare metal wall surface, can reduce the quenching of carbon monoxide and hydrocarbon consumption reactions.

The potential benefits of the TBC, for example, are described in references 2 and 13. Interest in this TBC has been expressed by manufacturers of aircraft, missile, automotive, and military vehicle gas turbine engines, diesel and spark-ignition reciprocating engines, and electric power gas turbines and by other diverse industries. Because of this interest and the need to further evaluate and extend the use of the coating, the Lewis Research Center made an agreement (contract) with several commercial and governmental organizations to apply the TBC to their products. In return for applying this coating or consulting on how to apply the coating, this Center was furnished with the conditions and test results.

This report presents the results of twelve tests of the TBC conducted as part of the previously mentioned contracts. Two of these tests determined the thermal conductivity of the TBC and another determined the vibratory fatigue characteristics of a coated turbine blade. Seven tests determined the thermal protection and durability provided by the TBC. The remaining two tests evaluated the corrosion protection provided by the TBC. The seven thermal protection and durability tests were on (1) coated turbine vanes operating in a full-scale military aircraft engine, (2) coated turbine blades operating in a full-scale military aircraft engine, (3) coated turbine blades operating in a commercial aircraft engine, (4) coated combustor dome, scroll, and turbine nozzle shroud parts tested in a full-scale, military, ground-based gas-turbine engine, (5) rig tests of a combustor liner for a gas-turbine auxiliary power unit, (6) a partially coated turbine plenum for a vehicular gas-turbine engine, and (7) coated valves operating in a full-scale diesel engine.

To protect proprietary interests, test conditions are not always described in great detail. The information presented, in addition to a discussion of test results, includes (when available) photographs of coated parts before and after tests, measurements of metal temperatures with and without the TBC, measurements of coating loss, and extent of bond and base metal corrosion. Based on the results of these tests, recommendations are made for improving the coating procedures.

ARC-PLASMA-SPRAY PROCEDURES AND MATERIALS

Figure 1 shows a ceramic-coated turbine blade, and figure 2 shows the blade being arc plasma sprayed in air with the TBC. All metal surfaces to be coated were vapor degreased and grit blasted with 99.51 percent pure (white) aluminum oxide. The grit blasting nominally removes 0.001 centimeter of the material. Using the white alumina minimized the contamination that might occur with less pure grit. The inlet supply pressure to the commercial grit-blasting equipment was 0.55 megapascal. The alumina grit size was 250 micrometers. The measured roughness of the metal surface after grit blasting was about 6 micrometers rms.

Within 15 minutes after grit blasting, a bond coating of NiCrAlY (nominally Ni-16Cr-6Al-0.6Y) was arc plasma sprayed with a commercial hand-held gun (Plasmadyne SG-1B gun) onto the roughened metal surfaces to the thicknesses given in table I; the thicknesses ranged from 0.006 to 0.15 centimeter depending on the specimen and its location. The particle size of the bond powder fed into the rear port of the plasma-spray gun was 44 to 74 micrometers. The roughness of the bond coating was 5 micrometers rms. For this bond coating, the gun was held at about 13 to 15 centimeters from the metal surface. Power input to the electric (plasma) arc was maintained at 31 volts and 350 amperes. Argon was used for the arc gas and the powder gas at flow rates of 1.6 and 0.31 cubic meter per hour, respectively. The dial on the feed mechanism (Plasmadyne Roto Feed Hopper) which supplies the NiCrAlY powder to the arc-plasma-spray gun was set at 1.8 units.

Within 30 minutes of applying the bond coating, nominally 12-weight-percent yttria-stabilized zirconia ceramic was arc plasma sprayed onto the bond coating. The same gun was used for deposition of the ceramic and bond coatings. The particle size of the ceramic powder fed into the rear port of the gun was 44 to 74 micrometers. The ceramic coating thicknesses after application are given in table I; the thicknesses ranged from 0.013 to 0.18 centimeter depending on the test objectives and conditions. The gun was held at 5 to 10 centimeters from the surface. The plasma arc was maintained at 33 volts and 550 amperes. Argon was used for the arc gas and powder gas at flow rates of 1.7 and 0.34 cubic meters per hour, respectively. The dial on the powder (zirconia) feed mechanism was set at 3.0 units.

To improve the coating adherence, the impingement of the grit, bond, and ceramic powder materials was nearly normal to the surface, sharp corners were rounded, and the ceramic and bond coating thicknesses were made as uniform as possible.

Recent tests at the NASA Lewis Research Center (ref. 14) show that a better coating system durability can be obtained on burner rig and furnace specimens when the percent by weight of yttria was reduced to 6.2 or 7.9 in the zirconia ceramic coating and when the percent by weight of yttrium was reduced to 0.15 or 0.35 in the bond coating. These ceramic and bond coatings were not used herein.

TEST OBJECTIVES, SPECIMENS, AND CONDITIONS

The details of the experimental objectives, the preparation of specimens, and the conditions for industrial testing of the TBC can be obtained from table I. Data for this table were obtained from references 15 to 20 and figures 3 to 11. Because the tests were so diverse, the details are described for clarity and coherence in table form. The first objective (table I) was to measure the thermal conductivity of the NiCrAlY bond coating and of the yttria-stabilized zirconia materials in the NASA TBC. The second objective was to determine the high cycle fatigue characteristics of the TBC system. The third objective was to determine the extent of thermal protection and durability provided by the TBC. This third objective was accomplished with cooled, coated hardware operating in full-scale gas-turbine engines, in diesel engines, and in rigs. The fourth objective was to determine the extent of corrosion protection obtained with the

TBC on solid, high-temperature alloy pins tested in marine diesel fuel products of combustion produced in a burner rig, and to test the corrosion protection in a device which simulated a method for drilling into rock material.

RESULTS AND DISCUSSION

THERMAL CONDUCTIVITY OF BOND AND CERAMIC COATINGS

The thermal conductivity values obtained by two experimental techniques (conditions 1 and 2, table I) were compared in corresponding temperature measurement regions. Values obtained with the two techniques agreed satisfactorily within 10 percent.

The measured thermal conductivity values of the yttria-stabilized zirconia are low when compared to most white, plasma-sprayed ceramic oxides. Because the thermal conductivity and density are low, thin layers of this material are an effective, lightweight insulator for reducing metal temperatures. Lower metal temperatures and thinner coatings reduce the probability of ceramic spallation.

The measured thermal conductivity values of the arc-plasma-sprayed zirconia varied from 0.50 to 1.4 watts per meter-K. This conductivity is represented by

$$k=-1.597+0.11T-1.896\times10^{-5}T^2+1.324\times10^{-8}T^3-3.077\times10^{-12}T^4 \text{ for } 400\text{ K}\le T\le1900\text{ K} \qquad (1)$$

The measured thermal conductivity values of the arc-plasma sprayed NiCrAlY bond coating varied linearly from about 6.5 to 16.0 watts per meter-K. This conductivity is represented by

$$k = 9.41\times10^{-3}\,T + 2.89 \text{ for } 367\text{ K} \le T \le 1367\text{ K} \qquad (2)$$

Overall thermal conductivity measurements were also made on the coated disks in tests (fig. 3) that simulated heat flow in diesel engine pistons and cylinder heads (condition 2). A diesel engine company concluded that the TBC has good insulating capability at practical thicknesses.

HIGH CYCLE FATIGUE CHARACTERISTICS

The fatigue tests (objective II, table I) showed that the fatigue strength of a single blade coated with the TBC was significantly better than blades without the TBC. The mean fatigue life of the uncoated aircraft gas-turbine blades and of the TBC-coated blade tested in the fatigue machine (figs. 4(a), (b), and (c)) at 1033 K was 16 500 and 220 000 cycles, respectively.

Figures 4(d) and (e) show a cross section of the coated blade. Fatigue failure occurred in the blade metal wall, not in the TBC, even though both bond and ceramic coating thicknesses were at their maximum in this airfoil location. The TBC did not spall in any area of the airfoil during this test. These results provided the confidence to install and test four blades coated with the TBC in a military aircraft engine. The test results are discussed later (see p. 7 and objective III-A-2, table I).

THERMAL PROTECTION AND DURABILITY

Gas-Turbine Applications

Turbine vanes - military aircraft engine. - An uncoated vane located in a hot-gas region (about 1700 K) melted and burned during accelerated cyclic endurance tests in a military aircraft engine (objective III-A-1, table I). This occurred during the first 60 hours of bimodal cyclic operation which included 240 cycles between the average gas temperature (1550 K) and shutdown and 1920 cycles between the average gas temperature and idle. This vane was replaced with a coated vane of identical design (fig. 5(a)) and subjected to the same test conditions. After 60 hours of these cyclic tests, the metal wall of the coated vane was not distressed, but loss of the TBC was observed at the trailing-edge pressure surface aft of the cooling-air dump holes (fig. 5(b)) and in small areas at the leading edge. Because the metal wall was very thin at the trailing-edge region, it could only be lightly grit blasted (about half the normal blasting pressure). This could be partly responsible for the premature coating deterioration in this region.

To improve durability, another vane of identical design was coated (after only a light grit blast), vacuum heat treated, and then tested at the same engine conditions. Visual inspection of this vane after 100 hours of testing showed no deterioration of the TBC. This test time included 400 and 3200 cycles of the two types just described. Testing continued with this vane for a total of 300 hours; this included 1200 and 9600 cycles at each of the two types of cycles. Visual inspection after 300 hours showed minute coating loss at the leading- and trailing-edge regions, but no deterioration of the cooled metal wall. Iron oxide particles were deposited on the TBC during the test (fig. 5(b)). Iron oxide (Fe_2O_3) can react with yttria-stabilized zirconia in the TBC (when monoclinic ZrO_2 is present) over times of 200 to 800 hours and temperatures of 1473 to 1673 K (ref. 21). Since the coating operated in this temperature range and is presumably partially monoclinic, such a reaction may be partly responsible for the deterioration of the TBC after 300 hours of cyclic operation.

The better durability of the coating on this vane is attributed to the vacuum heat treatment which sintered and densified the bond coating to improve its oxidation resistance and durability at the bond-coat - substrate interface (ref. 19). These tests also demonstrated that despite the local loss of the TBC deterioration of the cooled metal walls did not occur. These tests demonstrated the protective capability of the TBC and the associated improvement in vane life.

The vacuum heat treatment changed the color of the coating from pale yellow to medium grey. The color changed back to the original pale yellow after engine testing. Subsequent furnace tests of coated and heat-treated solid specimens in air indicated that the pale yellow color returns after several seconds of soaking at temperatures higher than about 900 K. This did not appear to effect coating durability.

The coated, unheat-treated test vane was sectioned and areas were found, primarily at the leading edge, where the bond coating thickness was 0.005 centimeter or less. This could have been responsible for some of the coating deterioration in this region. The bond layer should have been applied to a greater thickness at the leading edge since the bond coating thickness recommended for this TBC system is about 0.010 centimeter. This thin bond coating was necessary to minimize gas-path blockage when a uniform coating thickness is applied over airfoils.

Turbine blades - military aircraft engine. - The TBC on four convection-cooled core-turbine blades installed in the rotor of a military aircraft engine are shown in figure 6 before the start of "piggy-back" tests (objective III-A-2, table I). Boroscope inspection of the leading edges of the four coated turbine blades showed no evidence of deterioration after 10 hours of testing (ref. 22). These tests were at a gas temperature and pressure of 1640 K and 2.5 megapascals, respectively. After 15 hours of operation, an aluminide coating on the blades without the TBC showed some evidence of wear, but the TBC showed no signs of wear. Testing of the coated blades is continuing.

Turbine blades - commercial aircraft engine. - The coated air-cooled turbine blades in the commercial aircraft engine (objective III-A-3, table I) were first tested at accelerated cyclic endurance conditions for 39 hours (327 cycles). Inspection after this time showed coating loss at locations of highest metal temperatures at the leading edge. The maximum gas temperature for the tests was 1700 K and the gas pressure was 2.2 megapascals. A typical cycle was 2 minutes at takeoff power and 5 minutes at idle power. Because there was no blade metal wall deterioration, the cyclic tests on these coated blades were continued for an additional 225 hours (1097 cycles) in a second engine of the same type. This additional testing resulted in the loss of about one-third of the coating on the pressure (concave) side of the airfoil from about midspan to tip.

Although the results were not good at the leading-edge and pressure-surface regions of the coated blade, a visual examination indicated that the thermal barrier coatings were not damaged at other locations of the airfoils nor on the platforms after 264 hours (1424 cycles) of engine testing.

Combustor dome and scroll and turbine nozzle shroud - military ground-based vehicle. - Early deterioration of uncoated metal parts occurred during accelerated endurance tests of a military ground-based-vehicle gas-turbine engine at a gas temperature of 1476 K and pressure of 1.5 megapascals (objective III-A-4, table I). Early deterioration was stopped when the TBC was applied to new parts. A visual inspection of the engine and the coated parts after 500 hours of testing showed no deterioration of the coating. These tests again demonstrated the protective capability of the TBC and the associated improvement in the life of the metal parts.

Combustor liner - auxiliary power unit. - Heat-transfer calculations showed that the experimental combustor liner designed for use in the auxiliary power unit (objective III-A-5, table I) would probably distort during planned performance tests unless a TBC was used. The TBC was plasma-sprayed onto the gas-side surfaces of two combustor liners. Figure 7(a) shows the first coated liner before performance testing. Inspection of this liner after 15 hours of testing at measured hot-spot, coated metal temperatures of 1300 to 1370 K showed no liner warpage and no deterioration of the TBC. The gas conditions for these tests were more severe than required for the current gas-turbine engine. Tests were conducted on the second liner which had a more complex geometrical design. After 6 hours of testing, the TBC flaked off in the region where rolled offsets were fabricated into the sheet metal of this liner (fig. 7(b)); however, there was no liner warpage. The metal temperatures in this region were about 1140 K. These tests again showed the protective capability of the TBC, and they showed that the adherence of the coating is reduced when placed in regions where there are rapid changes in metal shape (the rolled offsets).

Turbine plenum - vehicular engine. - Figures 8(a) and (b) show patches of the TBC plasma sprayed onto each half of the turbine plenum before testing was started (objective III-A-6, table I). Figure 8(c) shows the good condition of the patches after 50 hours of accelerated endurance testing in a burner rig. These tests consisted of 120 cycles at five cycle variations (described in table I). The highest gas temperature for these cycles was 1310 K and the lowest was 530 K. The cycle consisted of 5 minutes of heating and 20 minutes of cooling. Some uncoated metal areas were badly warped after the test; however, no warping or coating deterioration occurred where patches of TBC were applied on the plenum metal wall. Because these cyclic burner tests were so promising, two other plenums with fully coated internal surfaces are undergoing tests in full-scale vehicular engines. One coated plenum has satisfactorily accumulated 220 hours with no visible deterioration, and the TBC has lowered hot-spot metal wall temperatures from about 1300 to 1200 K. Testing of the TBC on these plenums is continuing.

Diesel Engine Valves

There was no measurable thermal-barrier benefit of the TBC on the valves (fig. 9) as determined from the tests of the supercharged diesel engine (objective III-B, table I). The cooling effect of the scavenging air on this two-cycle engine is apparently sufficient to nullify the insulation provided by the ceramic coating. Both uncoated and coated metal temperatures were about 920 K. Examining the ceramic coating after the 1-hour engine test showed that (1) the thick (0.18 cm) layer of ceramic on the thick (0.25 cm) substrate adhered well and (2) there was no deterioration in this environment.

CORRISION PROTECTION AND DURABILITY

Solid, High-Temperature Alloy Pins

Coated pin-type specimens similar to the one shown in figure 10 were used to investigate the corrosion protection and durability of the TBC and its ability to protect base metal parts from corrosion in marine diesel fuel products of combustion (objective IV-A, table I). The tests with the burner rig simulated coating operation on marine gas-turbine engines. One specimen withstood 1000 hours (40 cycles) of testing. A cycle consisted of 23 hours at a gas (and metal) temperature of 978 K with a 2-hour cooldown period to 300 K. Nominal, polished zirconia thickness before testing was 0.038 centimeter, and metallographic examination showed that about 0.008 centimeter of yttria-stabilized zirconia was lost during the test. The TBC on the second specimen cracked and spalled after 322 hours or 14 of these cycles. The other two specimens were heated with this rig from about 300 to 1070 K and then cooled to 300 K. One of these specimens ran for 450 hours, and the other specimen was tested for 800 hours before cracking and spallation occurred. Metallographic examination showed no base metal deterioration on any of the four specimens. The results of these corrosion protection and durability tests indicated that the TBC is as good as some other coating systems also tested at these cyclic conditions.

Rock Drill Specimens

Coated solid metal disks (shown in fig. 11) were used to investigate the corrosion protection and durability of the TBC and its ability to protect the base metal from the corrosive attack of melted lava (objective IV-B, table I). This rock drilling technique incorporates electrically heated lava which jets from the drill head against the rock. The drill head scrapes the residual melt layer from the solid strata. In these laboratory tests, lead borosilicate glass was used to simulate the rock material, and Incoloy disks were used to simulate the drill base metal material.

The results of the tests at 1256 K indicated that molten glass diffused into the TBC, acted as a flux, and melted the ceramic. Therefore, the TBC cannot be used in this application.

CONCLUDING REMARKS

The results of this cooperative research on the NASA Lewis Research Center thermal barrier coating (TBC) with industrial and governmental organizations have demonstrated the benefits of the coating for aeronautical, marine, and ground-based gas-turbine - engine applications. This TBC is a two-layer system with a bond coating of nickel-chromium-aluminum-yttrium (Ni-16Cr-6Al-0.6Y, in wt. %) and a ceramic coating of yttria-stabilized zirconia (ZrO_2-$12Y_2O_3$, in wt. %).

In general, the results support the prior laboratory test results and heat-transfer analyses which showed that the TBC is highly adherent and substantially reduces the temperature of the cooled metal parts. Loss of the TBC which occurred on some parts reaffirmed prior research which showed that the ceramic layer spalled if ceramic-bond interface temperatures are excessively high (above 1250 K). The results also re affirmed the susceptibility of the TBC to erosion by gas products of combustion.

The TBC lowered metal temperatures, protected the coated parts, increased part life, and eliminated the burning, melting, and warping that occurred with some of the uncoated metal parts tested in this cooperative effort. For example, an uncoated turbine vane of a military aircraft engine and the uncoated combustor dome, scroll, and turbine nozzle parts of a military ground vehicle engine burned and melted at local spots after short test times, but a coated turbine vane of identical design satisfactorily completed over 300 hours (10 800 cycles) of testing and the coated combustor dome, scroll, and turbine nozzle shroud parts satisfactorily completed 500 hours of testing. In another test on a coated gas-turbine plenum for a vehicular engine, the TBC eliminated base metal warping and reduced hot-spot temperatures by 30 to 100 K.

During the thermal protection and durability tests, the local loss of the ceramic layer occurred on some coated parts in regions where uncoated parts had previously melted. However, in these areas of ceramic coating loss, there was no melting, burning, or other deterioration of the base metal. Also, during corrosion resistance and durability tests, the ceramic on some coated specimens cracked and spalled. However, metallographic examination showed no base metal deterioration. This demonstrated that local coating loss will not necessarily result in damage to metal parts.

The results of a test on a vane suggest that consideration should be given to vacuum heat treatment after TBC deposition to increase the durability of the TBC. In addition, sharp metal edges should be rounded before coating to reduce stress and minimize ceramic spallation.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, February 12, 1979,
505-04.

REFERENCES

1. Stecura, Stephan; and Liebert, Curt H.: Thermal Barrier Coating System. U.S. Patent 4,055,705, Oct. 1977.
2. Liebert, Curt H.; Stecura, Stephen; and Brown, Jack E.: Ceramic Thermal Barrier Coating - 1976, IR-100 Winner. Ind. Res., vol. 18, no. 10, Oct. 1976, p. 22.
3. Plunkett, Jerry D.: NASA Contributions to the Technology of Inorganic Coatings. NASA SP-5014, 1964.
4. Liebert, Curt H.; and Stepka, Francis S.: Ceramic Thermal-Barrier Coatings for Cooled Turbines. J. Aircr., vol. 14, no. 5, May 1977, pp. 487-493.
5. Stecura, S.: Two-Layer Thermal Barrier Coating for Turbine Airfoils - Furnace and Burner Rig Test Results. NASA TM X-3425, 1976.
6. Butze, Helmut F.; and Liebert, Curt H.: Effect of Ceramic Coating of JT8D Combustor Liner on Maximum Liner Temperatures and Other Combustor Performance Parameters. NASA TM X-73581, 1976.
7. Levine, Stanley R.: High Temperature Surface Protection. NASA TM-73877, 1978.
8. Kaufman, Albert; Liebert, Curt H.; and Nachtigal, Alfred J.: Low Cycle Fatigue of Thermal-Barrier Coatings at 982° C. NASA TP-1322, 1978.
9. Claus, Russell W.; Wear, Jerrold D.; and Liebert, Curt H.: Ceramic Coating Effect on Liner Metal Temperatures of Film-Cooled Annular Combustor. NASA TP-1323, 1979.
10. Mularz, E. J.; Gleason, C. C.; and Dodds, W. J.: Combustor Concepts for Aircraft Gas Turbine Low-Power Emissions Reduction. NASA TM-78875, 1978.
11. Diehl, Larry A.: Reduction of Aircraft Gas Turbine Engine Pollutant Emissions - A Status Report. NASA TM-78870, 1978.

12. Dodds, W. J.; Gleason, C. C.; and Bahr, D. W.: Aircraft Gas Turbine Low-Power Emissions Reduction Technology Program. (DOC-R78AEG408, General Electric Co., NASA Contract NAS3-20580.) NASA CR-135434, 1978.

13. Amos, D. J.: Analytical Investigation of Thermal Barrier Coatings on Advanced Power Generation Gas Turbines. (EM-1636, Westinghouse Electric Corp.; NASA Contract NAS3-19407.) NASA CR-135146, 1977.

14. Stecura, Stephan: Effects of Compositional Changes on the Performance of a Thermal Barrier Coating System. NASA TM-78976, 1978.

15. Shiembob, L. T.: Development of a Plasma Sprayed Ceramic Gas Path Seal for High Pressure Turbine Applications. (PWA-5521, Pratt & Whitney Aircraft; NASA Contract NAS3-19759.) NASA CR-135183, 1978.

16. Wilkes, K. E.; and Lagedrost, J. F.: Thermophysical Properties of Plasma Sprayed Coatings. (Battelle Columbus Labs.; NASA Contract NAS3-13329.) NASA CR-121144, 1973.

17. Sevcik, William R.; and Stoner, Barry L.: An Analytical Study of Thermal Barrier Coated First Stage Blades in a JT9D Engine. (PWA-5590, Pratt & Whitney Aircraft Group; NASA Contract NAS3-21033.) NASA CR-135360, 1978.

18. Materials Reference Card, Cast Nickel Base Alloys. Pratt & Whitney Aircraft, July 31, 1972.

19. Tucker, R. C.; Taylor, T. A.; and Weatherly, M. H.: Plasma Deposited MCrAlY Airfoil and Zirconia/MCrAlY Thermal Barrier Coatings. Union Carbide Corp., Indianopolis, Ind., 1976.

20. Unified Numbering System for Metals and Alloys. SAE J-1086 and ASTM DS-56. Soc. Automot. Eng., Inc., 1975, p. 6.2.

21. Zaplatynsky, Isidor: Reactions of Yttria-Stabilized Zirconia with Oxides and Sulfates of Various Elements. DOE/NASA/2593-78/1, NASA TM-78942, 1978.

22. Andress, D. E.: An Analytical Study of Thermal Barrier Coated First Stage Blades in an F-100 Engine. (FR-9609, Pratt & Whitney Aircraft Group; NASA Contract NAS3-21032.) NASA CR-135359, 1978.

TABLE I. - INDUSTRY TESTS OF NASA LEWIS RESEARCH CENTER CERAMIC THERMAL BARRIER COATING

Objectives	Specimens[a]	Conditions
I. Thermal conductivity of bond and ceramic coatings	(1) Zirconia specimens used in a cut-bar thermal conductivity measurement apparatus were fabricated as follows. Mar-M-509 was cut to a thickness of 0.635 cm and a diameter of 2.54 cm. Then NiCrAl was plasma sprayed onto the Mar-M-509 to a thickness of 0.008 to 0.013 cm. Finally, ZrO_2-$12Y_2O_3$ (wt. %) was plasma sprayed to a thickness of 0.15 to 0.20 cm over the NiCrAl. Other ZrO_2 specimens used in a laser thermal conductivity apparatus (ref. 16) were shaped into disks about 2.5 cm in diameter and 0.015 to 0.3 cm thick. These disks were formed using the plasma-spray process.	(1) Measurements were made on both the yttria-stabilized zirconia and NiCrAlY plasma-sprayed materials using a comparative cut-bar method (ref. 15) and a laser thermal diffusivity technique (ref. 16). Measurements on the zirconia were made at temperatures of 400 to 1900 K, and measurements on the NiCrAlY were made at 400 to 1400 K (ref. 17).
	(2) TBC was plasma sprayed onto solid disks of gray cast iron material (fig. 3). NiCrAlY and yttria-stabilized zirconia thickness was about 0.013 and 0.064 cm, respectively. The stabilized zirconia was not polished.	(2) Thermal conductivity measurements of coated cast-iron disks were made at 530 K in an apparatus which simulates heat flow in diesel engine cylinder heads, cylinder liners, and pistons.
II. High cycle fatigue characteristics	One coated and three uncoated first-stage turbine blades of a military aircraft engine were investigated. Blades were made from DS-MAR-M200 + HF material (ref. 18). Total measured coating thickness of an airfoil section ranged from 0.013 to 0.033 cm: 0.006 to 0.015 cm was NiCrAlY bond coating, and 0.013 to 0.033 cm was yttria-stabilized zirconia ceramic. Before coating with the TBC, blades were stripped of the original aluminide coating.	Fatigue testing was done in a hydraulic actuated fatigue machine (figs. 4(a), (b) and (c)) at a frequency of 30 Hz. Blades were fixed at the root, restrained at the shroud, and loaded in bending toward the concave side at the airfoil tip. Each blade was strain gaged at the maximum airfoil stress location (maximum root thickness near platform radius). Calibration of tip load against strain was established for each blade. Alternating stress of ±172 MN/m^2 superimposed on a steady stress of 448 MN/m^2 resulted in maximum stress of 620 MN/m^2 and minimum stress of 276 MN/m^2. Test temperature of 1033 K was maintained at maximum stress location.
III. Thermal protection and durability		
A. Gas-turbine applications 1. Turbine vanes - military aircraft engine	TBC was plasma-sprayed onto three convection cooled first-stage core-turbine vanes (fig. 5(a)) of X-40 (ref. 18) material. One coated vane was also vacuum heat treated for 4 hours at 1.3×10^{-3} Pa and 1450 K (ref. 19). Bond and ceramic coating thicknesses were about 0.005 and 0.02 cm, respectively. Third vane was sectioned for metallographic examination after plasma spraying.	Accelerated, full-scale engine mission tests were made at two cycle variations. The first was at an average gas temperature and pressure of 1550 K and 2 MPa to shutoff conditions, and the second was at 1550 K and 2.2 MPa to idle power. Each vane was separately tested in the hot-spot region of the vane ring where gas temperatures were about 1700 K.
2. Turbine blades - military aircraft engine	After removing the aluminide coating, the TBC was plasma sprayed onto four convection cooled, first-stage, core-turbine blades (fig. 6). Blades were made from DS-MAR-M200 + HF (ref. 18), and ceramic thicknesses were about 0.01 and 0.025 cm, respectively.	Full-scale engine tests were made on a "piggy-back" basis as part of other research. Included was a checkout run. Test conditions were at average gas temperatures and pressures of 1640 K and 2.5 MPa, respectively. Coolant temperature was about 800 K.
3. Turbine blades - commercial aircraft engine	TBC was plasma-sprayed onto six convection-cooled first-stage core-turbine blades made from DS-MAR-M200 + HF (ref. 18). TBC was deposited over a NiCoCrAlY oxidation resistant coating. This coating was applied by an electron beam evaporation - physical vapor deposition process. Bond and ceramic coating thicknesses were about 0.008 and 0.018 cm, respectively.	Accelerated, full-scale engine cyclic endurance tests were made at an average gas temperature of 1700 K and gas pressure of 2.2 MPa. A typical thermal cycle was 2 minutes at takeoff and 5 minutes at idle power (ref. 17).

[a]Unless indicated otherwise, the yttria-stabilized zirconia was polished with a silicon carbide cloth to a surface roughness averaging 1.6 to 3.0 μm rms as measured with a commercial surface roughness indicator.

TABLE I. - Concluded.

Objectives	Specimens[a]	Conditions
4. Combustor dome and scroll and turbine nozzle shroud - military ground based vehicle	Bond and ceramic material thicknesses could not be accurately measured on these geometrically complex surfaces. Estimates of these thicknesses are 0.013 to 0.025 cm for the bond material and 0.038 to 0.127 cm for the ceramic, which was not polished.	Full-scale engine tests were made in the laboratory and in the field on the coated parts at nominal average turbine inlet temperatures and pressures of about 1470 K and 1.5 MPa, respectively.
5. Combustor liner - auxiliary power unit	TBC was plasma sprayed onto the gas-side metal surfaces of two liners (fig. 7). Base material was Hastelloy X (ref. 18).	Eight rig tests were performed with the two coated combustors. Five tests were performed on the first liner and three on the second. Base metal temperatures were measured with temperature sensitive paints. Combustor exit temperatures were varied from 1300 to 1400 K at a nominal combustor discharge pressure of 0.8 kPa. Each of the eight tests lasted about 3 hours.
6. Turbine plenum - vehicular engine	Gas-turbine engine plenum was processed for application of the TBC in the areas shown in figures 8(a) and (b). Bond and ceramic coating thicknesses were 0.015 and 0.050 cm, respectively.	One hundred and twenty accelerated cyclic tests were made at five cycle variations in a burner rig test facility: 75 cycles from 530 to 1310 K, 15 cycles from 590 to 1390 K, and 15 cycles from 760 to 1310 K. Total time for one cycle was 25 minutes. There was a rapid (2 sec) temperature rise from the lower to the higher temperature (1310 K). The higher temperature was held for 5 minutes. It was followed by a 20-minute cooldown period.
B. Diesel engine valves	TBC was plasma sprayed onto four exhaust valves (fig. 9) made from Silicrome I (i.e., SAE J 775, unified number code 65007, ref. 20). Two valves were coated on the underside only, and two were coated on the underside and in the neck area. Bond coating was 0.10±0.005 cm thick and caramic coating was 0.04 to 0.18 cm thick. Ceramic was not polished.	Tests were made for 1 hour in a turbocharged diesel engine operating under full load at 2100 rpm. Metal temperatures were obtained by converting Rockwell C indentations to temperatures by using an experimentally obtained 1-hour temperature curve plot of C hardness against temperature of Silicrome I alloy.
IV. Corrosion protection and durability		
A. Solid, high-temperature alloy pins	TBC was deposited on three solid Rene 80 (ref. 20) pin-type substrates which had a 0.318-cm diameter and a 3.8-cm length (fig. 10). Bond and ceramic coating thicknesses were nominally 0.010 and 0.038 cm, respectively.	Tests were performed in gases produced by marine diesel fuel burning in a burner rig. The fuel contained 1 wt. % sulphur and 10 ppm sea salt. Specimens were heated from about 300 to 978 K and held at the elevated temperature for about 23 hours. The specimens were then cooled in still air for 2 hours. Hot gases impinge the front of the specimen and circulate over the rear. Specimens were sectioned for metallographic examination after testing.
B. Rock drill specimens	TBC was plasma sprayed onto four annealed Incoloy 802 HR Ann (ref. 20) disks (fig. 11). Stabilized zirconia ceramic was not polished. Bond and ceramic coating thicknesses were nominally 0.010 and 0.048 cm, respectively.	Disks coated with the TBC were tested in a device which simulated lava drilling at 1265 K. Glass was used in place of the lava material to simulate the lave drilling.

[a]Unless indicated otherwise, the yttria-stabilized zirconia was polished with a silicon carbide cloth to a surface roughness averaging 1.6 to 3.0 μm rms as measured with a commercial surface roughness indicator.

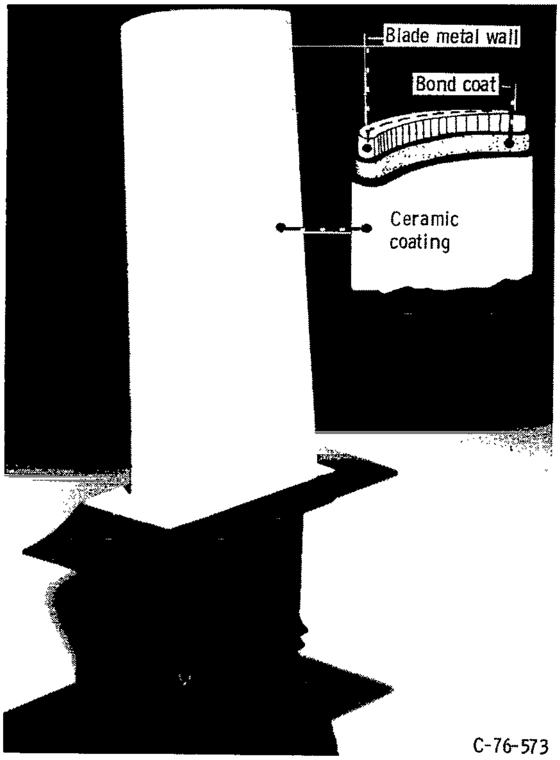


Figure 1. - Ceramic-coated turbine blade.

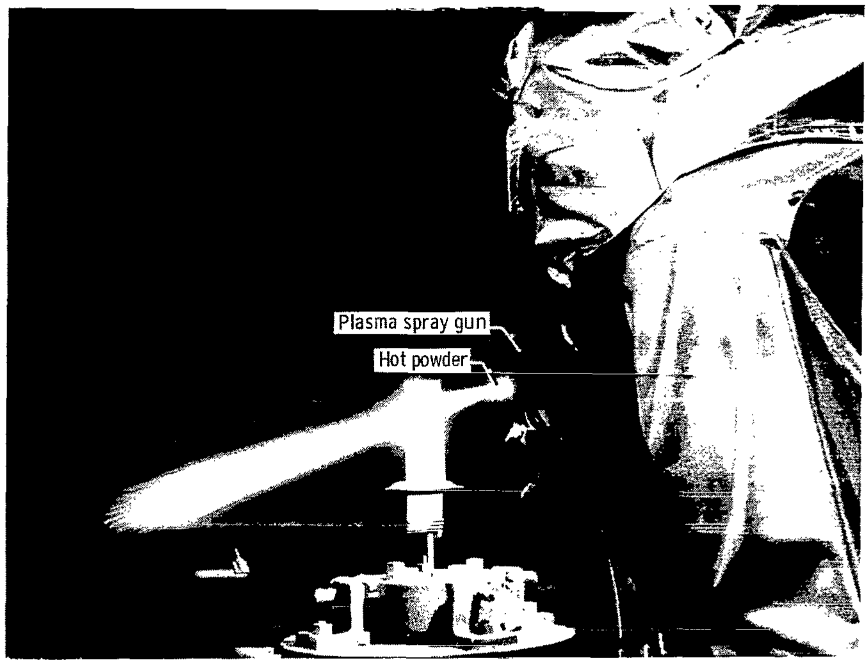


Figure 2. - Arc plasma spraying of turbine blade.

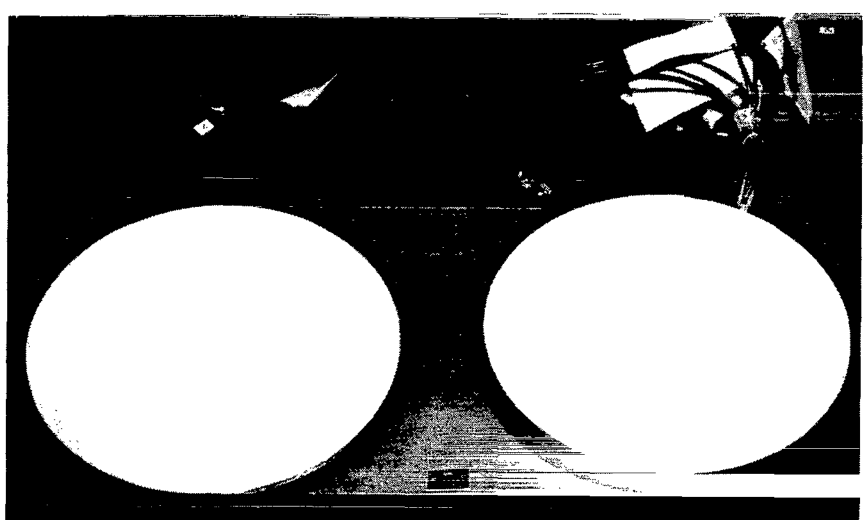
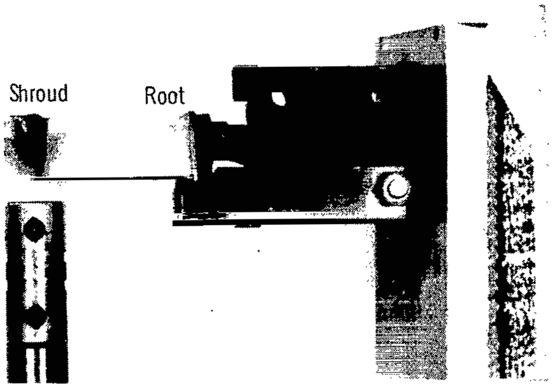
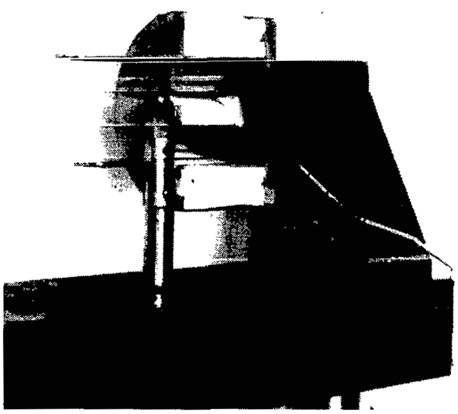


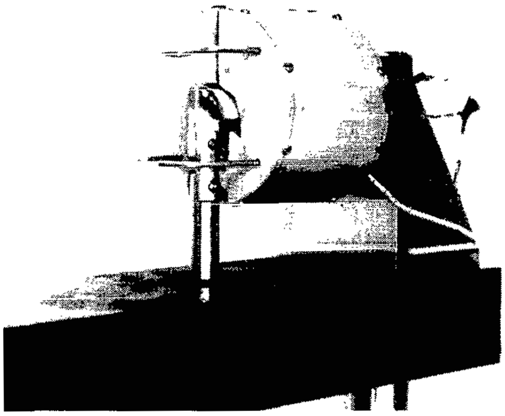
Figure 3. - Coating on disks for measuring thermal conductivity of NASA thermal barrier coating.



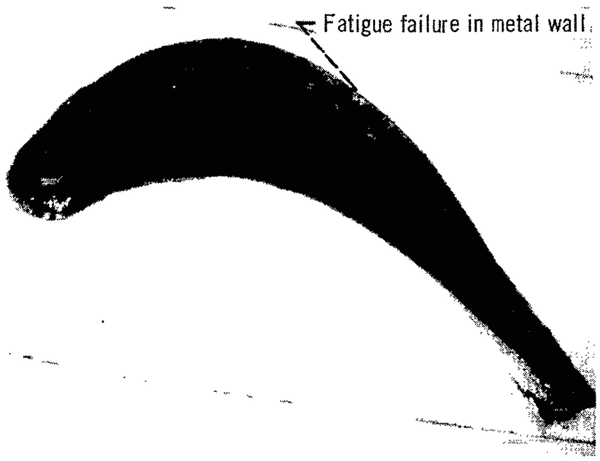
(a) Hydraulic actuated fatigue machine.



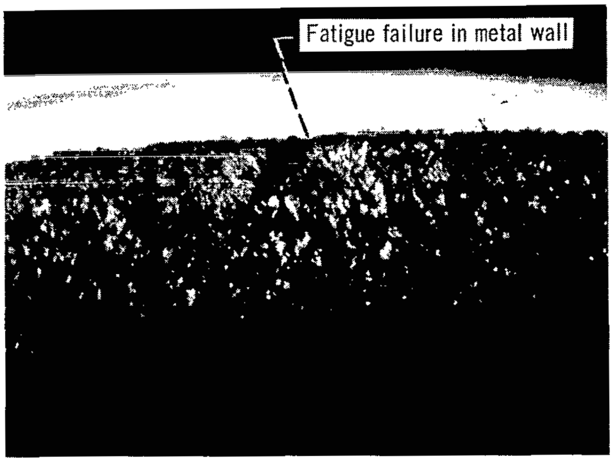
(b) Fatigue machine and furnace cutaway.



(c) Fatigue machine and furnace in place.

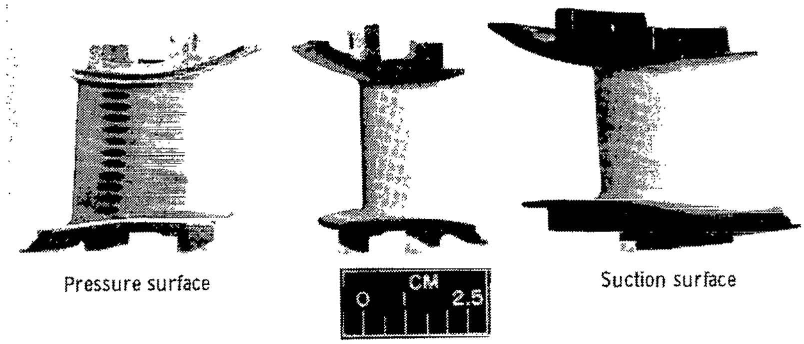


(d) Metal wall fracture of coated turbine blade after tests. ×20.

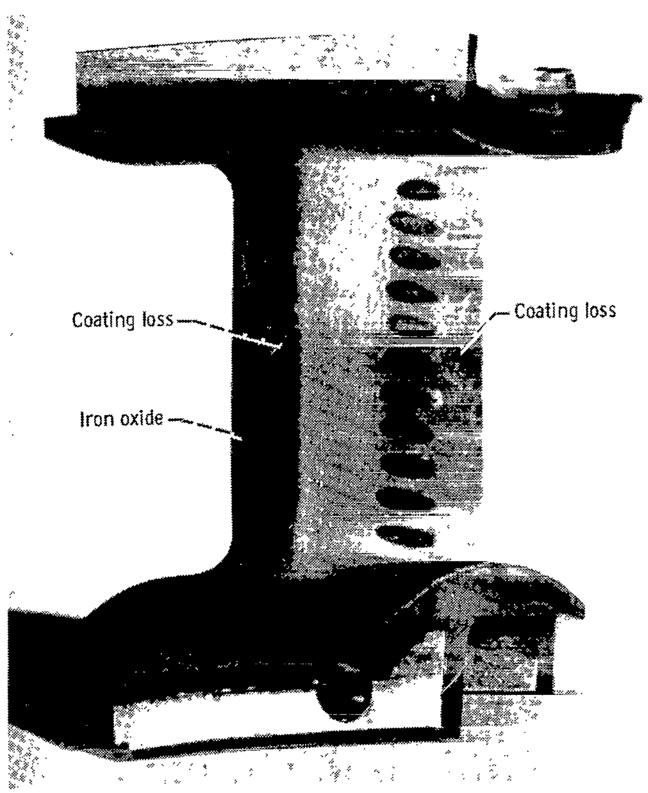


(e) Metal wall fracture of coated turbine blade after tests. ×20.

Figure 4. - High cycle fatigue testing of coated first-stage turbine blade.



(a) Before testing.



(b) After tests.

Figure 5. - Coated turbine vanes of a military aircraft gas-turbine engine.

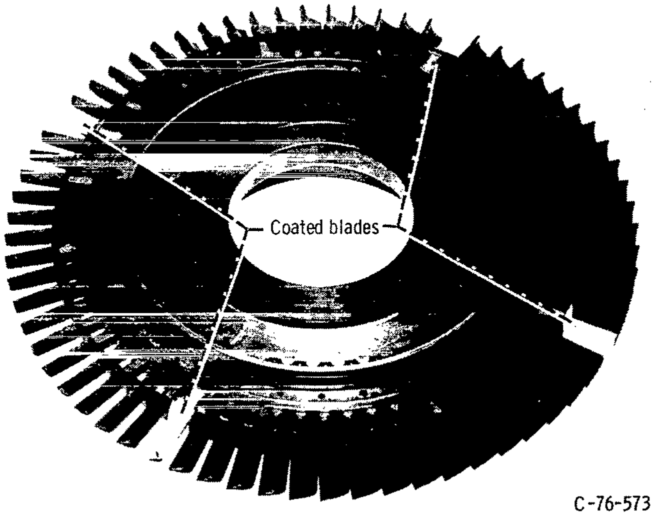
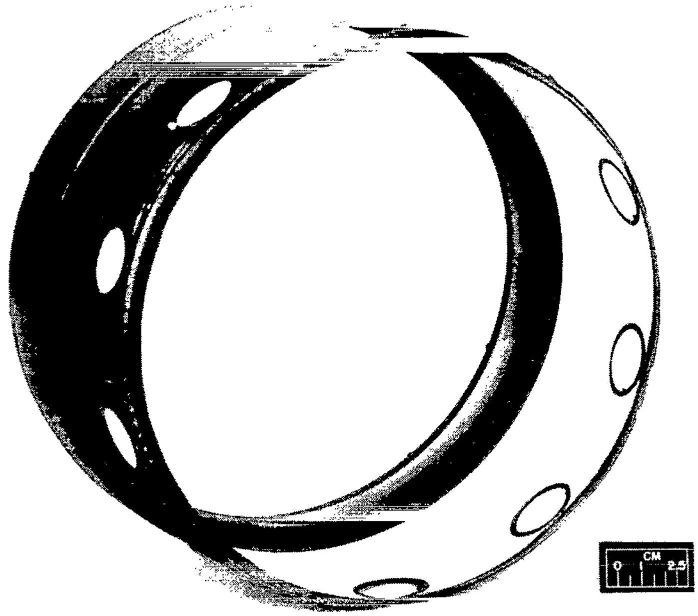
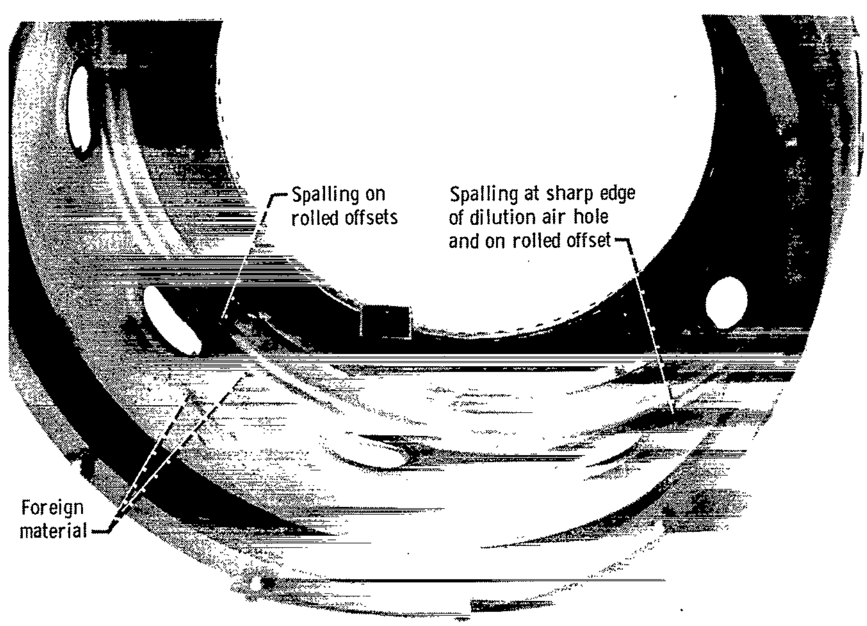


Figure 6. - Four coated military engine blades before tests.

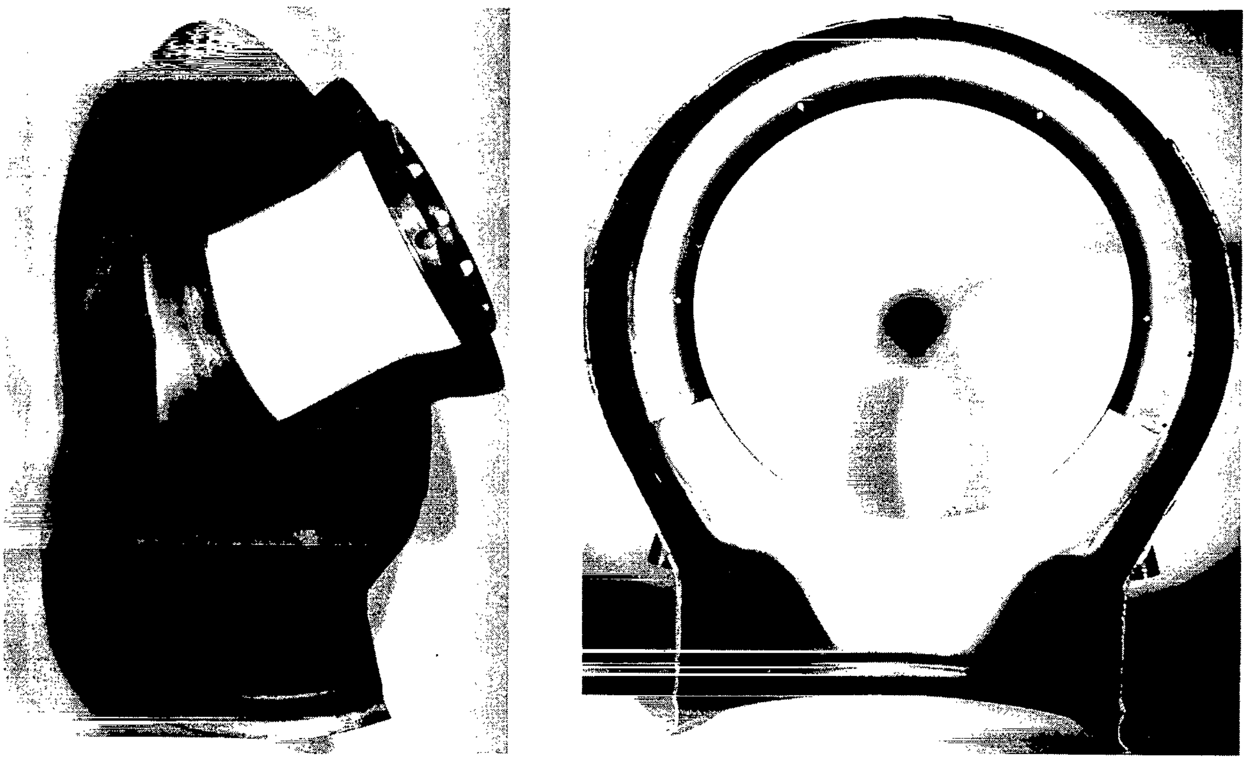


(a) First coated combustor liner before tests.



(b) Second coated combustor liner with a more complex geometrical design after tests.

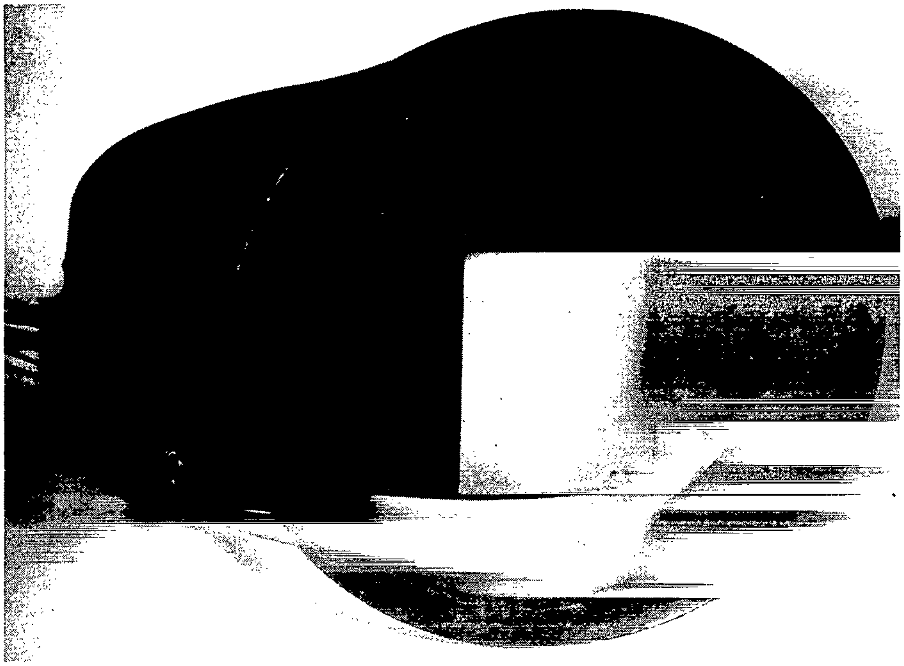
Figure 7. - Coated combustor liners of auxiliary power unit.



(a) Coating patch on one-half of plenum, before tests.

(b) Coating patch on other half of plenum, before tests.

Figure 8. - Partially coated turbine plenum of vehicular gas-turbine engine.



(c) Coated patch on assembled plenum, after tests.

Figure 8. - Concluded.

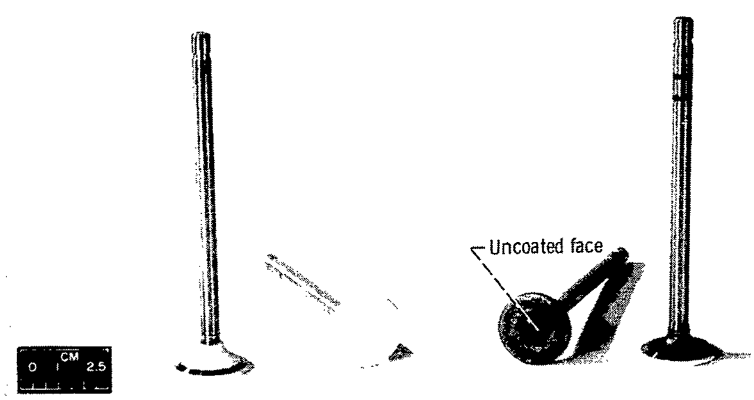


Figure 9. - Coated and uncoated diesel engine valves before tests.

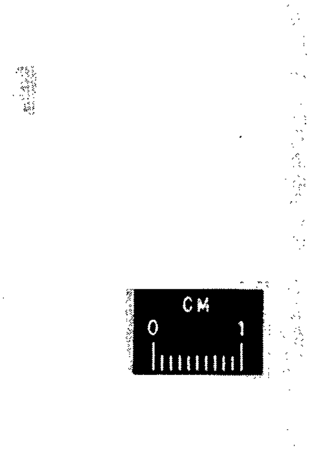


Figure 10. - Coated solid pin specimen for corrosion protection and durability tests in marine diesel fuel products of combustion.

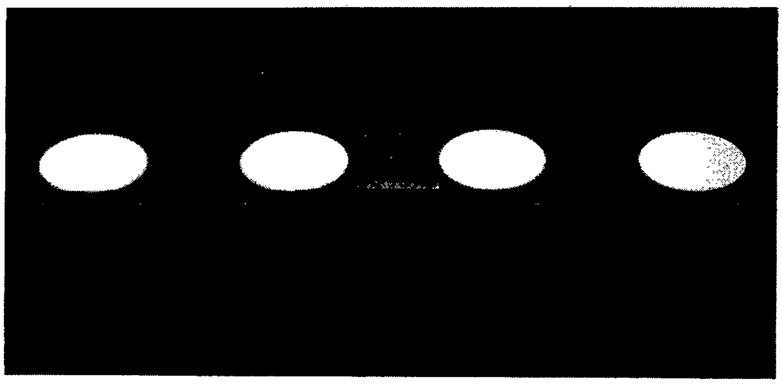


Figure 11. - Coated disks for tests to simulate lava drilling.

1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
NASA TP-1425		

4. Title and Subtitle	5. Report Date
INDUSTRY TESTS OF NASA CERAMIC THERMAL BARRIER COATING	June 1979
	6. Performing Organization Code
7. Author(s) Curt H. Liebert and Francis S. Stepka	8. Performing Organization Report No. E-9846
	10. Work Unit No. 505-04
9. Performing Organization Name and Address Lewis Research Center National Aeronautics and Space Administration Cleveland, Ohio 44135	11. Contract or Grant No.
	13. Type of Report and Period Covered Technical Paper
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D.C. 20546	14. Sponsoring Agency Code

15. Supplementary Notes

Presented at Sixth International Vacuum Metallurgy Conference sponsored by the American Vacuum Society, San Diego, Calif., April 23-27, 1979.

16. Abstract

A NASA Lewis Research Center ceramic thermal barrier coating (TBC) system was tested by industrial and governmental organizations for a variety of aeronautical, marine, and ground-based gas-turbine engine applications. This TBC is a two-layer system with a bond coating of nickel-chromium-aluminum-yttrium (Ni-16Cr-6Al-0.6Y, in wt. %) and a ceramic coating of yttria-stabilized zirconia (ZrO_2-12Y_2O_3, in wt. %). Seven tests evaluated the system's thermal protection and durability. Five other tests determined thermal conductivity, vibratory fatigue characteristics, and corrosion resistance of the system. The information presented includes test results and photographs of the coated parts. Recommendations are made for improving the coating procedures.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Thermal-barrier coatings Gas-turbine engines Arc-plasma spray	Unclassified - unlimited STAR Category 07

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
Unclassified	Unclassified	25	A02

* For sale by the National Technical Information Service, Springfield, Virginia 22161

National Aeronautics and Space Administration

Washington, D.C.
20546

Official Business
Penalty for Private Use, $300

SPECIAL FOURTH CLASS MAIL
BOOK

Postage and Fees Paid
National Aeronautics and
Space Administration
NASA-451

U.S.MAIL

2 1 1U,A, 060179 S00903DS
DEPT OF THE AIR FORCE
AF WEAPONS LABORATORY
ATTN: TECHNICAL LIBRARY (SUL)
KIRTLAND AFB NM 87117

NASA

POSTMASTER: If Undeliverable (Section 158 Postal Manual) Do Not Return