JTTEE5 6:99-104

Thermal Barrier Coating Experience in Gas Turbine Engines at Pratt & Whitney

S. Bose and J. DeMasi-Marcin

Pratt & Whitney has accumulated more than three decades of experience with thermal barrier coatings (TBCs). These coatings were originally developed to reduce surface temperatures of combustors of JT8D gas turbine engines to increase the thermal fatigue life of the components. Continual improvements in design, processing, and properties of TBCs have extended their applications to other turbine components, such as vanes, vane platforms, and blades, with attendant increases in performance and component durability. Plasma-spray-based generation I (Gen I) combustor TBCs with 7 wt% yttria partially stabilized zirconia deposited by air plasma spray (APS) on an APS NiCoCrAlY bond coat continues to perform extremely well in all product line engines. Durability of this TBC has been further improved in Gen II TBCs for vanes by incorporating low-pressure chamber plasma-sprayed NiCoCrAlY as a bond coat. The modification has improved TBC durability by a factor of 2.5 and altered the failure mode from a "black failure" within the bond coat to a "white failure" within the ceramic.

Further improvements have been accomplished by instituting a more strain-tolerant ceramic top layer with electron beam/physical vapor deposition (EB-PVD) processing. This Gen III TBC has demonstrated exceptional performance on rotating airfoils in high-thrust-rated engines, improving blade durability by three times through elimination of blade creep, fracture, and rumpling of metallic coatings used for oxidation protection of the airfoil surfaces. A TBC durability model for plasma-sprayed as well as EB-PVD systems is proposed that involves the accumulation of compressive stresses during cyclic thermal exposure. The model attempts to correlate failure of the various TBCs with elements of their structure and its degradation with thermocyclic exposure.

Keywords EB-PVD, gas turbine engine, plasma spray, TBC, thermal barrier

1. Introduction

GAS TURBINE engines use nickel- and cobalt-base superalloys in the turbine components, such as airfoils, combustors, transition ducts, and seals. Increased thermal efficiency demands of the newer engines dictate that the turbine inlet temperature be significantly increased. This must be accomplished without structural failure of components from melting, creep, oxidation, thermal fatigue, or other degradation mechanisms. Therefore, the surface temperature of these components must be maintained low enough to retain materials properties within acceptable bounds.

The demand is partially met by innovative component cooling schemes using compressor discharge air. However, cooling air is only available at the expense of loss of thrust and added fuel consumption. The major element in meeting the demand without significant performance loss comes from two parallel materials innovations: (1) improvement in the temperature capability of superalloys on the order of 80 to 85 °C and (2) development of thermal barrier coatings (TBCs) capable of providing thermal insulation equivalent to about 165 to 170 °C. The relative magnitude of the thermal benefit provided by TBCs highlights their extreme importance as enablers of superhigh-thrust engines such as Pratt & Whitney's PW4084.

Typically, TBCs are a bilayer system, consisting of a metallic layer (the bond coat) on the substrate on which a ceramic layer is deposited. In addition to providing oxidation protection to the substrate, the metallic bond coat layer provides an adequately prepared surface to which the ceramic layer adheres. The ceramic layer composition is selected based on thermal conductivity, high-temperature stability, and thermal expansion compatibility with the substrate. Due to its low thermal conductivity and good thermocyclic durability when stabilized against phase transformation, zirconia has become the ceramic of choice. The current bilayer design and composition of TBC have evolved from a multitude of designs and a number of compositions tested by Pratt & Whitney over a period of several decades.

2. Engine Experience and TBC Evolution

2.1 *Historical Background*

Pratt & Whitney first introduced TBCs on burner cans in JT8D engines in 1963. This TBC consisted of zirconia stabilized by the addition of 22 wt% MgO (22MSZ) to avoid the detrimental tetragonal (high-temperature phase) to monoclinic (low-temperature phase) phase transformation. The ceramic was deposited on a flame-sprayed Ni-Al bond coat. Subsequent evolution of TBCs at Pratt & Whitney, incorporating refinements that allowed incremental increases in combustor exit temperature, is summarized in Table 1.

In current systems, the ceramic consists of 7 wt% yttria partially stabilized zirconia (7YSZ). Depending on the application requirements, the bond coat is a variation of the NiCoCrAlY composition applied either by air plasma spray (APS) or low-

S. Bose and **J. DeMasi-Marcin,** Pratt & Whitney, East Hartford, CT 06108, USA.

pressure chamber plasma spray (LPPS). The ceramic deposition is accomplished either by APS or the electron beam/physical vapor deposition (EB-PVD) technique.

In the present generation of high-thrust engines, the major turbine components protected by TBCs and the TBC systems used are summarized in Table 2. A review of general experience on the various TBC systems in Pratt & Whitney engines has been presented by Meier and Gupta (Ref 1). However, some details beyond those covered by the review, with specific observations, are discussed in the sections that follow.

2.2 *Combustor Plasma-Sprayed TBC*

Magnesia-stabilized zirconia (MSZ) used to be the ceramic top layer for combustor application. However, due to its susceptibility to destabilization and relatively low temperature capability (~982 °C), MSZ has been replaced with 7YSZ as the ceramic coat. This modification has resulted in a fourfold improvement in durability.

The 7YSZ-based combustor TBC system, henceforth called generation I (Gen I) TBC, has performed extremely well in all engines. Failure of Gen I TBC usually occurs within the bond coat with the appearance of so-called "black failure." The bond coat is NiCoCrAlY deposited by APS, since the large component size does not allow for processing in a low-pressure chamber. X-ray diffraction analyses indicate that the bond coat oxidizes to a structure akin to NiO. It is speculated that kinetics favor the growth of this oxide because of the preexistence of the oxide nuclei formed during APS. This oxide is voluminous and is relatively weak in comparison with the thin oxide that forms on alumina formers. With increased thermocyclic exposure, biaxial in-plane compressive stresses develop within the oxide. When these stresses exceed the compressive strength of the oxidized bond coat, failure seems to occur within the bond coat, giving the black appearance of the oxide. The compressive stresses are exacerbated in the presence of sharp changes of radius of curvature. In combustor designs, therefore, such abrupt changes of radius are carefully avoided.

2.3 *Airfoil Plasma-Sprayed TBC*

Experience on Gen I combustor TBC showed that the weak link in the system was the bond coat. Therefore, the next improvement focused on strengthening of the bond coat, particularly its oxidation resistance. This was accomplished in the Gen II TBC system by incorporating a LPPS NiCoCrAlY bond coat in place of the APS bond coat of Gen I. The absence of oxygen during deposition of the bond layer eliminated the preexisting oxide nuclei to a great extent and altered the oxidation mode, eliminating the black failure observed previously.

Vane airfoil surfaces and platforms, with an operating environment structurally and thermally more severe than that experienced by the combustor, are usually protected by Gen II TBCs. This system has demonstrated outstanding performance in all engines, with a durability improvement of 2.5 times compared to Gen I. The failure mode of Gen II, extensively investigated under NASA's HOST program (Ref 2), is characterized by spallation of the ceramic, with resulting "white failure" in which major cracks develop and propagate parallel to and near the bond coat/ceramic interface but always remain within the ceramic. These major cracks are a result of linking of microcracks that form early in the ceramic due to cyclic thermal exposure. Bond coat oxidation results in the formation of a slow-growing thin alumina layer. Due to the strengthening of the bond coat and the resulting oxide, the weak link in Gen II TBC, relative to Gen I, is

Table 1 Development of TBCs at Pratt & Whitney

TBC system	Year of introduction	Bond coat	Ceramic coat	Design of layers
Early combustor TBC	1963	Flame-sprayed Ni-Al	APS 22MSZ	Ceramic/bond coat
	1973	APS Ni-Cr/Al	APS 22MSZ	Ceramic/cermet bond coat
	1974	APS CoCrAlY	APS 22MSZ	Graded
	1980	APS NiCoCrAlY	APS 22MSZ	Ceramic/bond coat
Gen I	1984	APS NiCoCrAlY	APS 7YSZ	Ceramic/bond coat
Gen II	1982	LPPS NiCoCrAlY	APS 7YSZ	Ceramic/bond coat
Gen III	1987	LPPS NiCoCrAlY	EB-PVD 7YSZ	Ceramic/bond coat

Table 2 Application of current TBC systems in Pratt & Whitney gas turbine engines

Turbine component	TBC system	Size requirement	Structural/thermal environment	Specific engines
Combustor	Gen I	Large	Moderate thermal	JT8D, JT9D, PW2000, PW4000, and V2500
Vane airfoils	Gen II	Small	High thermal	First and second stages in JT9D, PW2000, PW4000, and V2500
Vane platforms	Gen II	Small	Moderate thermal	First and second stages in JT9D, PW2000, PW4000, and V2500
Blade airfoils and platforms	Gen III	Small	High mechanical and thermal	First in JT9D, PW2000, PW4000, and V2500

now within the ceramic layer itself, thus providing the white appearance.

The thickness of the TBC is an important parameter; although increased ceramic thickness provides additional thermal insulation, it adds to component weight and accelerates TBC spallation. The nominal thickness of TBC systems calls for a 0.125 mm bond coat and a 0.250 mm ceramic top coat. These thickness values have been determined from experience by balancing thermal protection against additional weight imposed by the TBC.

The durability of TBCs was found to be strongly dependent on thickness. In order to understand the ceramic thickness issue in greater detail, Pratt & Whitney used laboratory tests to determine the effects of ceramic thickness on cyclic durability as measured in burner rig tests. Burner rig bars were coated with Gen II TBCs of increasing ceramic thickness. These bars, in combination with baseline Gen II TBC with a 0.25 mm ceramic layer, were exposed to a thermocyclic endurance test at 1120 °C. The results, in the form of relative durability versus ceramic thickness, with data scatter, are shown in Fig. 1. The data show that increased thickness has a strong negative effect on durability. The failure mode of the ceramic also changed with increased thickness, moving from a compressive failure within the ceramic for a nominal 0.25 mm thickness to tensile cracking at higher thicknesses. It is speculated that with increased ceramic thickness, the temperature of the top surface increases, introducing sintering as a degradation mechanism. Sintering and associated shrinkage in turn introduce tensile stresses, which result in transverse cracking of the ceramic perpendicular to the interface. Such failure modes have indeed been observed with thicker TBCs in the laboratory and in engine testing.

2.4 *Airfoil EB-PVD TBC*

The successful application of TBCs on the vane airfoil surface and platform logically leads to the question of rotating airfoil applications. Earlier engine tests at Pratt & Whitney had indicated that improvement beyond Gen II TBC would be required because of the combination of rotational stresses and high temperatures that blades experience. Thermal barrier coating with a LPPS NiCoCrAlY bond coat and a ceramic layer consisting of the strain-tolerant EB-PVD 7YSZ structure (Ref 1), designated Gen III TBC, appeared very attractive based on burner rig tests in which a tenfold durability improvement relative to Gen II TBC was exhibited. This translates roughly into an anticipated threefold improvement in blade life.

Due to the potential importance of this TBC in high-thrust engines such as PW4000, early engine validation was conducted. The test was comparative in nature, conducted on first-stage turbine blades of the JT9D engine. Blades coated with patches of Gen III TBC on the concave side were run in the same engine, in a rainbow configuration, with two additional sets of blades—one coated with an equivalent patch of plasma-deposited Gen II TBC and the other with just the LPPS metallic bond coat, which only provides oxidation protection. As shown in Fig. 2, both the metallic coating and Gen II patches exhibited significant blade distress. However, the EB-PVD-based Gen III patch performed extremely well. This excellent performance has been verified on blades in JT9D revenue engines, with total run time on TBC exceeding 15,000 h on the same blade.

Repeated demonstrations of excellent performance have led to GEN III TBC becoming the bill-of-material for blade application on all PW high-thrust engines, with cumulative experience exceeding 4 million hours. Figure 3 shows the excellent condition of Gen III TBC on a cluster of first-stage blades after 1666 endurance cycles in a PW4084 experimental engine. Although the third-generation TBC performance is at least an order of magnitude improved in thermocyclic durability, failure still occurs by spallation. However, the failure mode of this TBC is unlike those for both Gen I and II. A very thin aluminum oxide layer grows on the bond coat at the ceramic/bond coat interface. This tenacious thermally grown oxide (TGO) plays a key role in the adhesion of the ceramic layer of the TBC. Failure usually occurs by spallation due to the propagation of cracks either within the TGO or at the TGO/bond coat interface. Frequently, the ceramic spalls at room temperature after the component has been out of test for a considerable length of time (from a few to tens of hours). Such delayed failures

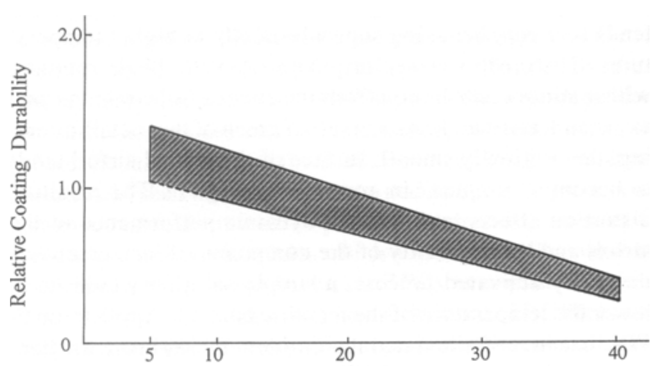

Fig. 1 Negative effect of increased TBC thickness on durability as demonstrated in burner rig testing

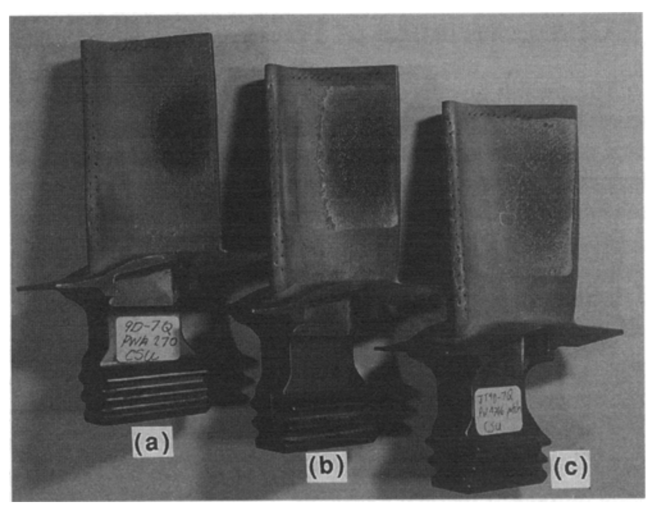

Fig. 2 Relative durability of (a) metallic coating for oxidation resistance, (b) plasma-sprayed TBC, and (c) EB-PVD TBC demonstrated by application of respective patches on the concave airfoil surface of a JT9D first-stage blade. Whereas the metallic coating plasma TBC exhibited distress, EB-PVD TBC performed extremely well. Each blade is approximately 13 cm long.

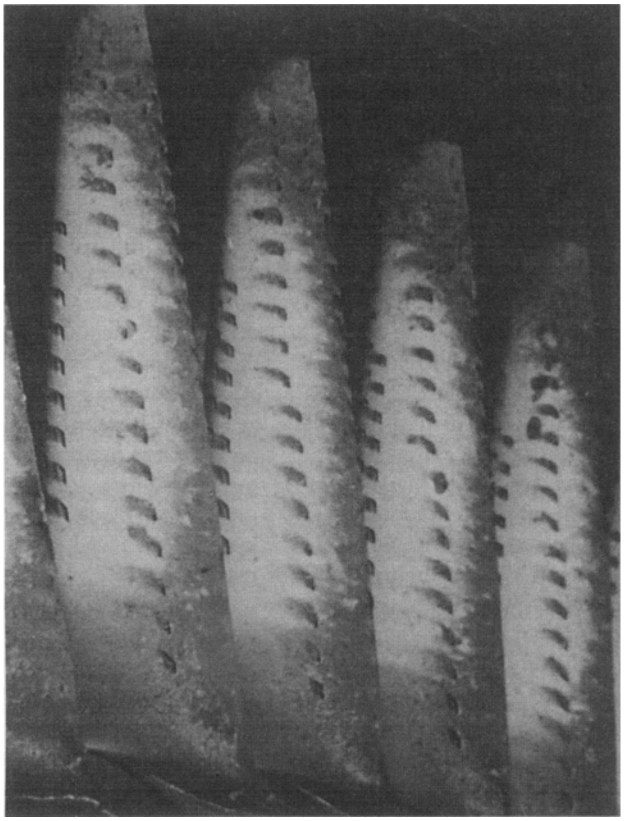

Fig. 3 Leading-edge cluster of first-stage blades tested in a PW4084 engine, demonstrating excellent performance of EB-PVD TBC. Each blade is approximately 13 cm long.

clearly indicate the role of accumulated residual stresses in the ceramic/bondcoat system.

3. Performance- and Failure-Related Characteristics of TBCs

The overall benefit of TBCs lies in lowering the substrate temperature, thereby retaining a significant portion of the structural properties of the substrate materials. Other specific performance advantages of TBC also derive from the lowered substrate temperature.

3.1 *Suppression of Thermal Transient Effects*

Figure 4 shows a generic strain/metal temperature cycle, the so-called "peanut curve" of a turbine blade. As illustrated, the particular operating points within the mission cycle of an engine can significantly affect both the component mechanical strain and the metal temperature, and every change in the operating point introduces thermal transients. The effect of these thermal transients is nonuniform temperature distribution and temperature gradients across the airfoil. These can result in distortion and thermomechanical fatigue cracking of the components. In addition to the transients, component design and associated cooling patterns generate nonuniform temperature distribution

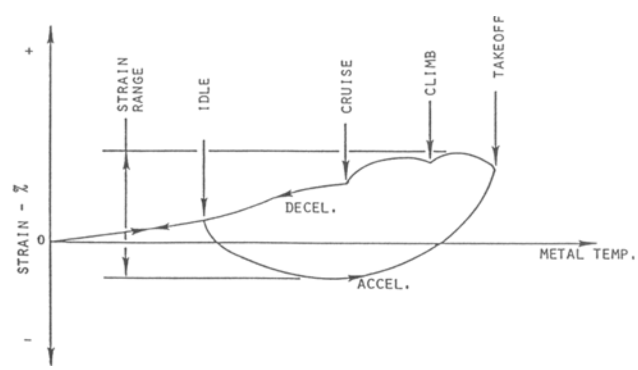

Fig. 4 Generic "peanut curve" for a turbine blade showing the transient nature of strain versus metal temperature cycle

and temperature gradients, which can contribute to distortion and cracking.

Distortion induced by nonuniform temperature distribution is illustrated in Fig. 5, which shows the cross section of the airfoil section of an engine-tested blade. Due to its operating environment, the blade experienced a temperature difference of approximately 85 °C between the concave pressure side and the convex suction side. In combination with the high operating temperature, the ΔT translated into a significant thermal stress to creep distort the pressure side, as indicated by the arrow in Fig. 5. The distortion has two effects. First, it disturbs the aerodynamic flow. Second, it reduces blade life. A solution to the distortion issue was the use of TBC on the airfoil surface. Application of Gen III TBC has alleviated the problem by lowering the overall temperature and reducing the temperature gradient across the airfoil surface. Figure 6 shows the cross section of an airfoil section of a blade tested with Gen III TBC under similar conditions. It shows no signs of the creep distortion seen in Fig. 5.

3.2 *Creep of Metallic Coatings*

To impart oxidation protection, turbine blades are coated with an oxidation-resistant metallic coating. The first and second turbine blades are usually coated with LPPS NiCoCrAlY overlay coating. Above the brittle-to-ductile transition temperature of the intermetallic NiCoCrAlY, the coating tends to creep, behaving superplastically at higher temperatures. The creep stresses originate from the blade rotation, with a minor contribution from the coating/substrate thermal expansion mismatch. As a result of creep of the metallic coating, the originally smooth surface of the coated airfoil tends to become "rumpled" in appearance (Fig. 7). The resulting distortion affects both the aerodynamic performance of the airfoil and the durability of the component. Since creep is a thermally activated process, a simple solution would be to lower the temperature of the metallic coating. Application of TBC has indeed alleviated the problem by lowering the temperature. As can be seen in Fig. 7, there are no signs of creep in areas protected by TBC, while adjacent areas with only metallic coating exhibit the creep "rumples."

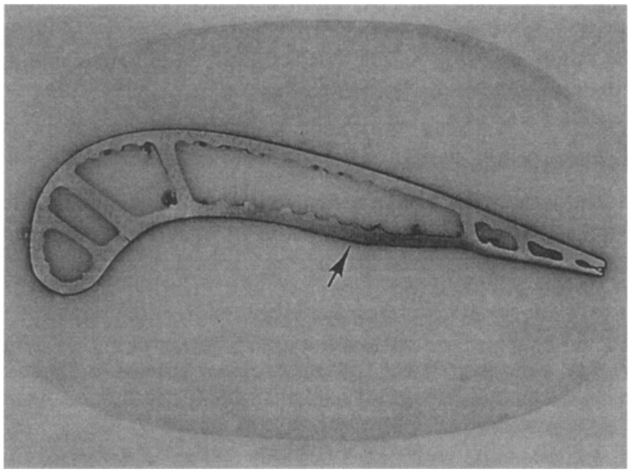

Fig. 5 Creep distortion of an airfoil due to nonuniform temperature distribution. The blade is about 4 cm wide.

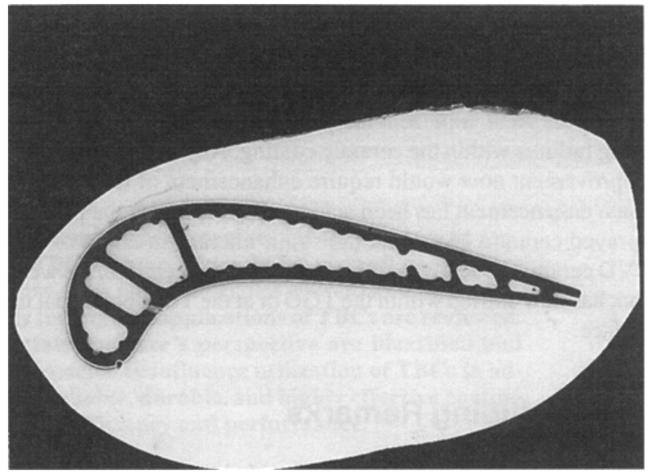

Fig. 6 Creep distortion of an airfoil eliminated by application of TBC. The blade is about 4 cm wide.

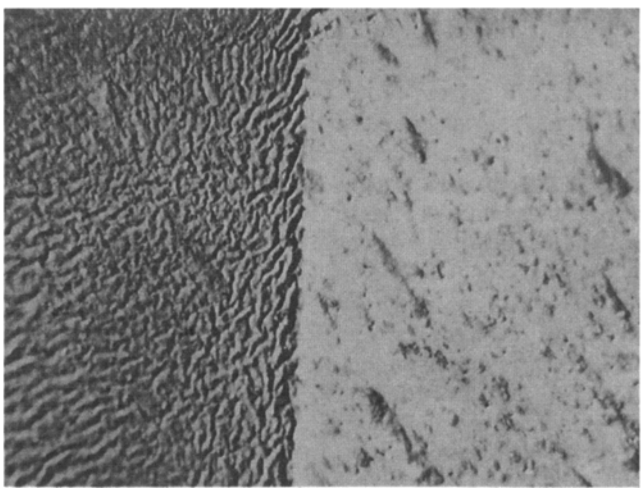

Fig. 7 Bond coat creep eliminated by TBC application. The field of view is approximately 1.3 by 0.9 cm.

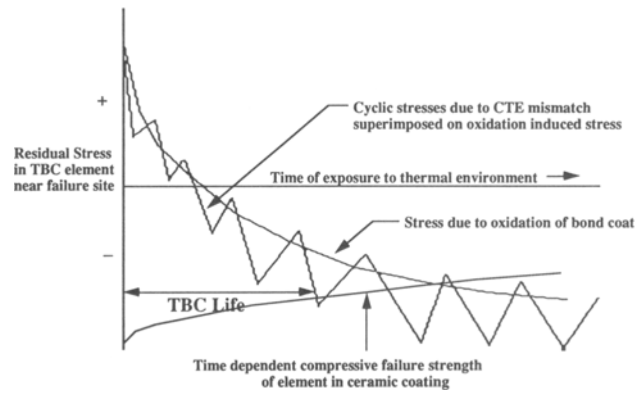

Fig. 8 "Weak link" model of TBC failure. In-plane compressive residual stresses in the TBC increase with thermocyclic exposure time. Failure occurs when these stresses exceed the compressive strength of elements at the failure location.

3.3 *Generic Failure Mechanism*

Based on the experimental engine experience and laboratory data generated to date, model emerges that explains the thermocyclic behavior of TBCs, including plasma-sprayed on Gen I, Gen II, and EB-PVD-based Gen III systems. The details of the model depend on individual TBC systems, the time-dependent stress-strain response, and the materials degradation mechanism of individual elements of the TBC. However, in all cases, it has been established that coating failure occurs due to in-plane compressive residual stresses. In-plane tensile residual stresses result only in transverse cracks, which do not generally contribute to coating spalls except in thicker TBCs as explained before. The influence of oxidation on plasma-deposited TBC life is well demonstrated (Ref 3). A significant portion of the residual stress develops due to the constrained oxide growth on the bond coat at the ceramic/bond coat interface. Depending on the kinetics of the oxide growth in the presence of the ceramic top layer, the compressive residual stresses increase with time, as depicted schematically in Fig. 8. Superimposed on this increasing compression are the cyclic stresses generated by the thermal expansion mismatch between the ceramic and the underlying layer, the cycles being the heating and cooling associated with engine operation. Failure occurs when the cumulative compressive stress exceeds the compressive strength of the elements at the failure sites.

The failure locations are coating specific. In MSZ-based TBC, failure takes place in the ceramic due to destabilization of MSZ. In Gen I TBC, failure occurs in the bond coat. In Gen II TBC, failure takes place within the ceramic but near the ceramic/bond coat interface. The EB-PVD-based Gen III TBC, on the other hand, fails either within the TGO or at the TGO/substrate interface. The failure sites are the weak links in the TBC structure. In Gen I with an APS bond layer, the weakest link lies within the bond coat. The oxidized bond coat, with predominantly NiO structure, has a relatively low compressive strength.

In Gen II, replacement of the bond layer with LPPS-deposited NiCoCrAlY favorably altered the oxidation characteristics. Instead of a weak and voluminous NiO structure, a thin and strong alumina forms. The strengthening of the bond layer, therefore, moved the weak link from the bond coat into the ceramic, initiating failures within the ceramic coating. Any further durability improvement now would require enhancement of the ceramic. Such enhancement has been achieved by replacing the plasma-sprayed ceramic layer with the strain-tolerant structure of EB-PVD ceramic. Due to the strengthening of the ceramic, the weak link has now shifted within the TGO or at the TGO/bond coat interface.

4. Concluding Remarks

Thermal insulation benefits provided by TBCs and the resulting impact on component creep and thermomechanical fatigue life have made them enablers of high-thrust gas turbine engines. Of particular importance is the EB-PVD-based TBC, which is anticipated to improve blade life by a factor of three. This new generation of TBC has evolved from an understanding of and improvements to the structure, properties, processing, and failure modes of earlier TBC systems. Such understanding will continue to improve TBC performance.

Acknowledgments

The authors would like to thank Dr. D.N. Duhl and Dr. D.K. Gupta of Pratt & Whitney for reading the manuscript and offering suggestions.

References

1. S.M. Meier and D.K. Gupta, The Evolution of Thermal Barrier Coatings in Gas Turbine Engine Applications, *J. Eng. Gas Turbines Power (Trans. ASME)*, Vol 116, 1994, p 250-257
2. J.T. DeMasi, M. Ortiz, and K.D. Sheffler, "Thermal Barrier Coating Life Prediction Model Development, Phase I Final Report", NASA Cr-182230, NASA–Lewis Research Center, 1989
3. R.A. Miller and C.E. Lowell, Failure Mechanisms of Thermal Barrier Coatings Exposed to Elevated Temperature, *Thin Solid Films*, Vol 95, 1982, p 265-273