

# **Durability Challenges for Next Generation of Gas Turbine Engine Materials**

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## **Abstract**

Aggressive fuel burn and carbon dioxide emission reduction goals for future gas turbine engines will require higher overall pressure ratio, and a significant increase in turbine inlet temperature. These goals can be achieved by increasing temperature capability of turbine engine hot section materials and decreasing weight of fan section of the engine. NASA is currently developing several advanced hot section materials for increasing temperature capability of future gas turbine engines. The materials of interest include ceramic matrix composites with 1482 – 1648°C temperature capability, advanced disk alloys with 815°C capability, and low conductivity thermal barrier coatings with erosion resistance. The presentation will provide an overview of durability challenges with emphasis on the environmental factors affecting durability for the next generation of gas turbine engine materials. The environmental factors include gaseous atmosphere in gas turbine engines, molten salt and glass deposits from airborne contaminants, impact from foreign object damage, and erosion from ingestion of small particles.

# Durability Challenges for Next Generation of Gas Turbine Engine Materials

## Introduction

Significant research and development efforts are underway to reduce thrust specific fuel consumption (TSFC) for commercial and military aircraft propulsion. Approaches for reducing TSFC include increasing overall pressure ratio (OPR), turbine inlet temperature, and bypass ratio. Engines with high OPR and turbine inlet temperature require significant increases in temperature capability of engine materials. While significant advances in single crystal nickel-base superalloys along with thermal barrier coatings and cooling technologies have been responsible for increases in turbine operating temperatures over the last several years, the blade alloy temperatures are reaching close to their melting points. Advanced thermal barrier coatings (TBCs) with lower thermal conductivity (ref. 1) and improved bondcoats, will be required for increasing temperature capability of turbine engines using metallic blades and vanes.

Revolutionary increases in OPR and turbine inlet temperature can be achieved by the use of ceramic matrix composites (CMCs) in hot section of gas turbine engines. It is envisioned that CMCs will be introduced in static structures, such as shrouds, combustor liners, vanes, and then in rotating components, first with low pressure turbine (LPT) blade and then with high pressure turbine (HPT) blade. Besides offering the capability for higher operating temperatures, CMCs have densities  $1/3^{\text{rd}}$  of nickel-base superalloys, thus offering significant weight benefits as well. Silicon carbide fiber reinforced silicon carbide (SiC/SiC) CMCs is the material of choice for gas turbine engines (refs. 2 and 3). SiC/SiC CMCs are targeted to have temperature capability equal to or greater than  $1315^{\circ}\text{C}$ , offering greater than  $100^{\circ}\text{C}$  increase in temperature capability over single crystal nickel-base superalloys.

Because of high stresses in turbomachinery disk, it is unlikely that CMCs will be used for disks. High OPR engines will require compressor disk alloys with temperature capability greater than that for the current state-of-the-art alloys. With recent introduction of third generation of powder metallurgy nickel-base disk alloys, the disk temperature in advanced commercial engines today is on the order of  $704^{\circ}\text{C}$ . With further increases in the OPR beyond the current state-of-the-art, the disk temperature for last stage of compressor is expected to be greater than  $760^{\circ}\text{C}$ . It is envisioned that for OPR of 60, disks with  $815^{\circ}\text{C}$  temperature capability will be required.

In addition to the mechanical and thermal loads, the environmental factors and the combined effect of environment plus mechanical/thermal loading are expected to have a greater degree of influence on durability of advanced materials with further increase in hot section temperatures for gas turbine engines. With increase in turbine operating temperatures, new environmental effects, such as deposition of sulfate melts and glassy silicate melts along with erosion by small particles, are becoming major factors affecting durability of new high temperature materials.

The objective of this paper is to provide an overview of the durability challenges for advanced materials that are being considered for future gas turbine engines, along with strategies to

mitigate these challenges. While there are many durability challenges (such as creep, low cycle and thermomechanical fatigue, flutter), this paper will focus on environmental factors that will have significant effect on durability of next generation of gas turbine engine materials. The environmental challenges include reaction with gas turbine atmosphere, erosion and impact due to ingestion of foreign objects, and deposition of airborne contaminants. Material systems of interest are SiC/SiC CMC, advanced TBCs, and higher temperature disk alloy.

### Reaction of Advanced Hot Section Materials with Turbine Engine Atmosphere

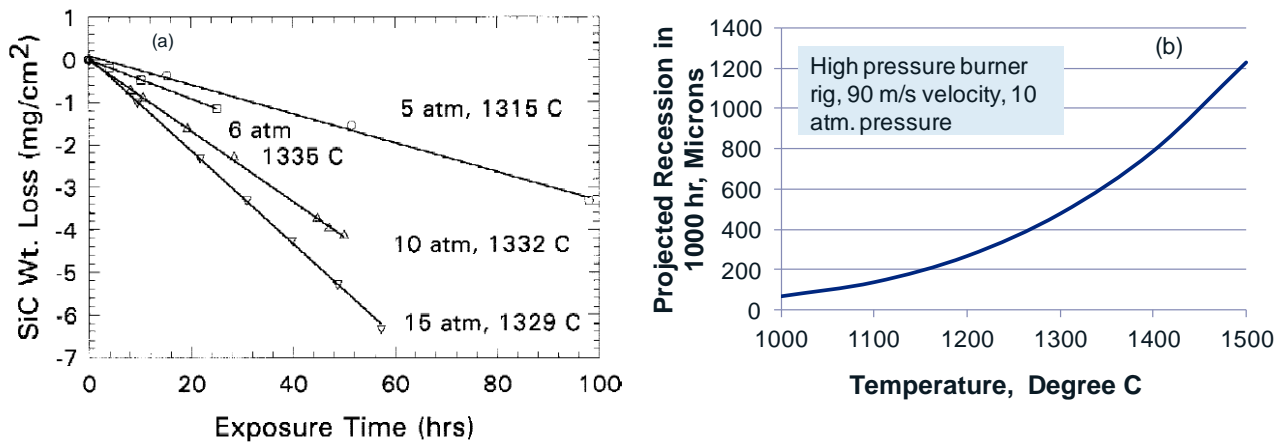
#### *Reaction of SiC/SiC CMC with Moisture in Gas Turbine Environment:*

Research at NASA in 1990's have identified a new environmental degradation problem (refs. 4 and 5) with SiC/SiC CMCs resulting from reaction of SiC with moisture component of gas turbine engine environment to form volatile  $\text{Si(OH)}_4$  reaction product. SiC oxidizes to form a  $\text{SiO}_2$  scale, which then reacts with  $\text{H}_2\text{O}$  to form volatile  $\text{Si(OH)}_4$  by the reaction:



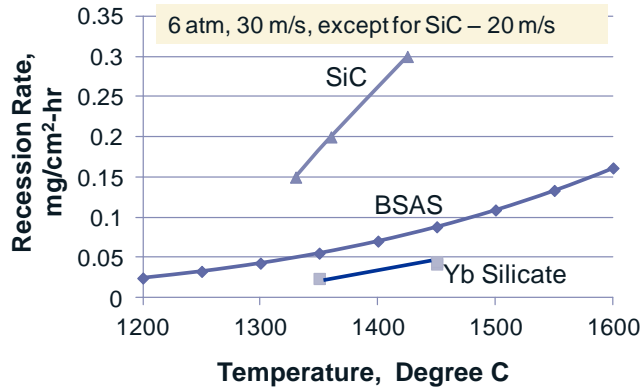
Formation of volatile  $\text{Si(OH)}_4$  reaction product results in continuous surface recession of SiC/SiC CMC, as shown in Figure 1.

The hot section components of gas turbine engines are expected to have durability on the order of thousands of hours, with time at the highest temperature on the order of 300 to 1000 hr, depending on the flight speed regime. Clearly, high surface recession rates shown in Figure 1b are not acceptable, and protection schemes must be developed to enable long-term durability of SiC/SiC CMC engine components. This led to the development of environmental barrier coatings or EBCs (refs. 6 and 7). The coating chemistries are based on complex silicates with significantly lower thermodynamic activity for  $\text{SiO}_2$ . The first generation of EBCs, developed for combustor liners, consisted of complex barium strontium aluminum silicate

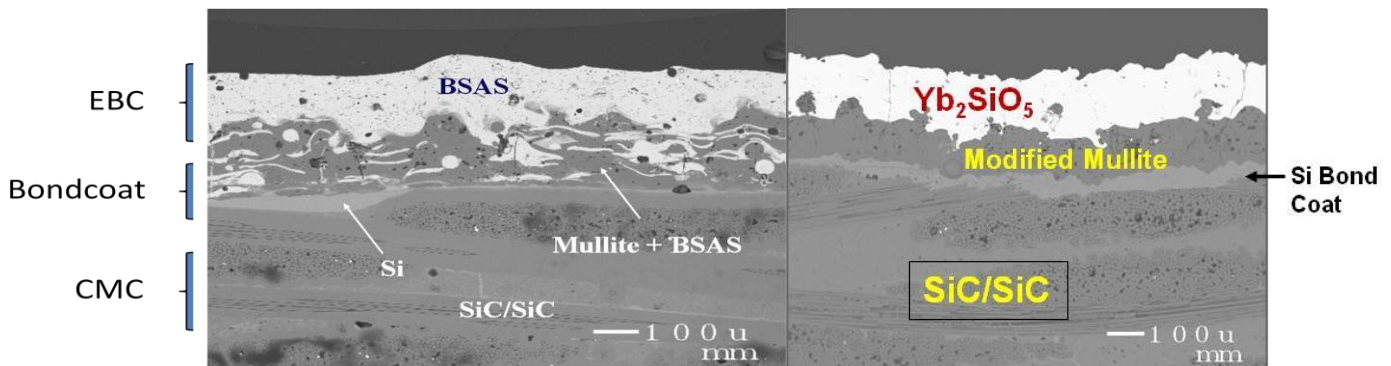


**Figure 1:** (a) Weight loss of SiC in high pressure burner rig, (b) projected surface recession of SiC after 1000 hr as a function of temperature

( $Ba_{1-x}Sr_xAl_2Si_2O_8$ ), also known as BSAS (refs. 6 and 7) that has matching coefficient of thermal expansion (CTE) with SiC/SiC CMC. Rare earth silicates, such as  $Yb_2SiO_5$  are promising EBC candidates because of their lower volatilization rates than BSAS (ref. 8), as shown in Fig. 2. The typical EBC is a multilayer coating, consisting of a bond coat and a top coat, as shown in Figure 3. The first generation (Gen 1) EBC had a Si bondcoat with an intermediate layer of mullite+BSAS or modified mullite between the Si bondcoat and topcoat (BSAS or  $Yb_2SiO_5$ ).

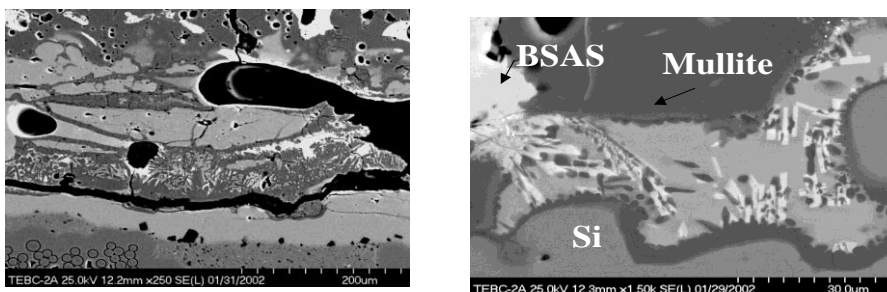


**Figure 2:** Recession rate of BSAS and Yb silicate compared to that of SiC



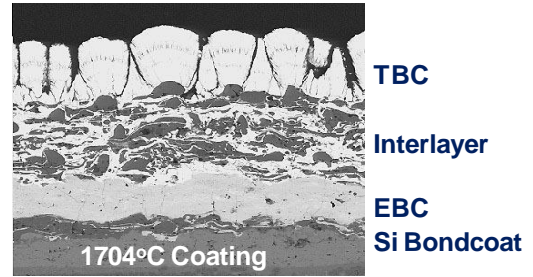
**Figure 3:** First generation of EBC for SiC/SiC CMCs

The upper temperature limit for BSAS is dictated by various chemical degradation processes that occur at high temperatures (ref. 9) and is limited to below 1315°C. For example, the BSAS coating suffers from interfacial pore formation at 1300°C due to reaction with water vapor and diffusion of Si (Fig. 4). The upper temperature limit of  $Yb_2SiO_5$  may not be significantly higher than that of BSAS because of possible decomposition (ref. 10) of  $Yb_2SiO_5$  at 1300°C.



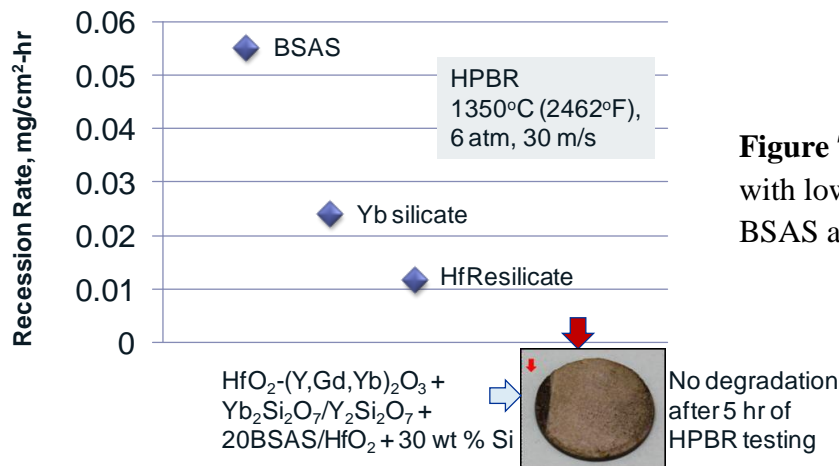
**Figure 4:** Degradation BSAS-based environmental barrier coating system after long-term exposure at 1300°C

Other potential candidates with significantly lower recession rates include (1) earth silicates with various combinations of  $\text{Re}_x\text{O}_y$  and  $\text{SiO}_2$ , e.g.,  $\text{Re}_2\text{O}_3\text{-}2\text{SiO}_2$ ,  $2\text{Re}_2\text{O}_3\text{-}3\text{SiO}_2$ ,  $\text{Re}_2\text{O}_3\text{-SiO}_2$  (where  $\text{Re} = \text{Sc, Dy, Er, Tm, Lu, Sc, Lu, Gd, and Eu}$ ), and (2)  $\text{HfO}_2$  and  $\text{ZrO}_2$ -based composite oxides (ref. 11). A successful strategy for increasing temperature capability of EBCs is to use a combination of thermal barrier coating (TBC) with EBC (ref. 12), where the TBC top layer reduces the temperature of the EBC and the bondcoat. An example of such a coating is shown in Fig. 5, which shows four layers,  $\text{HfO}_2\text{-}18 \text{ mol}\%(\text{Y,Gd,Yb})_2\text{O}_3$  TBC/Gd, Yb, and Hf modified mullite interlayer, mullite + 20% BSAS EBC layer/Si bondcoat. The multilayer coating system has demonstrated thermomechanical stability after 100, 1-hr cycles in laser heat flux testing at  $1704^\circ\text{C}$  surface temperature. The coating system has also demonstrated excellent water vapor stability and durability in high pressure burner rig test (ref. 12). The combined TBC/EBC coatings are thick and would find application as coatings for static CMC components, such as combustor liners and turbine vanes; however, such thick coatings would not be desirable for blades, which require thinner coatings for aerodynamic performance.



**Figure 5:** Combined TBC/EBC multilayer coating

Research is underway (ref. 13) at NASA to identify coatings with higher temperature capability, with temperature capability in the range of  $1482 - 1648^\circ\text{C}$ . Two complex coatings, Hf Resilicate ( $\text{Re} = \text{rare earth element}$ ) and  $\text{HfO}_2\text{-}(\text{Y,Gd,Yb})_2\text{O}_3 + \text{Yb}_2\text{Si}_2\text{O}_7/\text{Y}_2\text{Si}_2\text{O}_7 + 20\text{BSAS}/\text{HfO}_2 + 30 \text{ wt}\% \text{ Si}$ , show promise based on initial feasibility studies (Fig. 7).

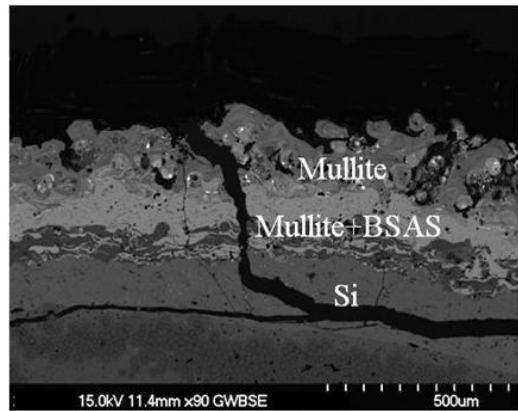


**Figure 7:** Complex silicate coatings with low recession rates compared to BSAS and Yb silicate

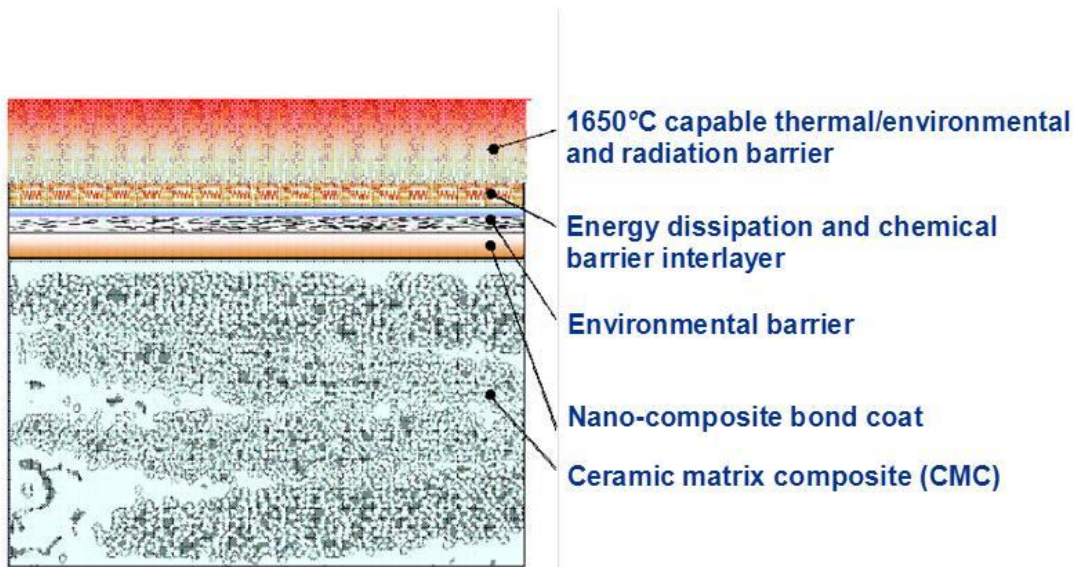
**Challenges for Increasing Temperature Capability of EBCs:** The state-of-the-art EBCs have upper temperature limit of  $1315^\circ\text{C}$ . Use of CMCs in hot section components of high OPR gas turbine engines will require EBCs with temperature capability greater than  $1482^\circ\text{C}$ . Based on the recession data so far, it appears that coatings will consist of complex silicates as one constituent of the coating. While complex silicates show initial promise in terms of chemical stability in moisture environment, long-term durability needs to be demonstrated. Chemical

reaction of coatings with moisture at high gas velocities encountered in turbine section of the gas turbine engine is of concern as reaction rates increase with gas velocity (ref. 4). Experiments in high pressure burner rig has shown eight fold increase in recession rate for SiC/SiC with increase in velocity from 30 m/s to 200 m/s. Maintaining long-term phase stability of complex oxides at temperatures on the order of 1482 – 1648°C is a challenge as one component of the oxide preferentially reacts with moisture to form a volatile product. A suitable non-Si containing bondcoat will be required for such high temperatures as Si melts at 1410°C. The thermochemical stability challenges will even be greater for thin coatings that will be required for rotating components, such as turbine blades.

In addition to thermochemical stability, the EBCs must also have thermomechanical stability for long periods of time. One mode of degradation is delamination of coatings under high heat flux thermal gradient (ref. 13), as shown in Figure 8. Coatings with higher fracture toughness will be required to reduce the extent of delamination. Since coatings will be under thermal and mechanical load, creep of oxide silicates will be of concern and composite approaches to reduce creep may be required. It is expected that under mechanical loading, the EBC is likely to crack before the CMC, and this might limit the load bearing capability of the CMC. Multilayer coating approach is currently under development at NASA (Fig. 9).



**Figure 8:** Delamination of layered EBC under high heat flux as tested in high heat flux laser experiment

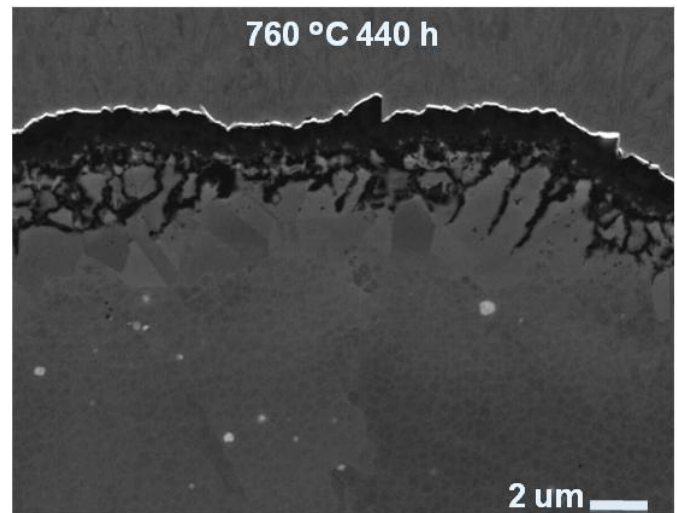
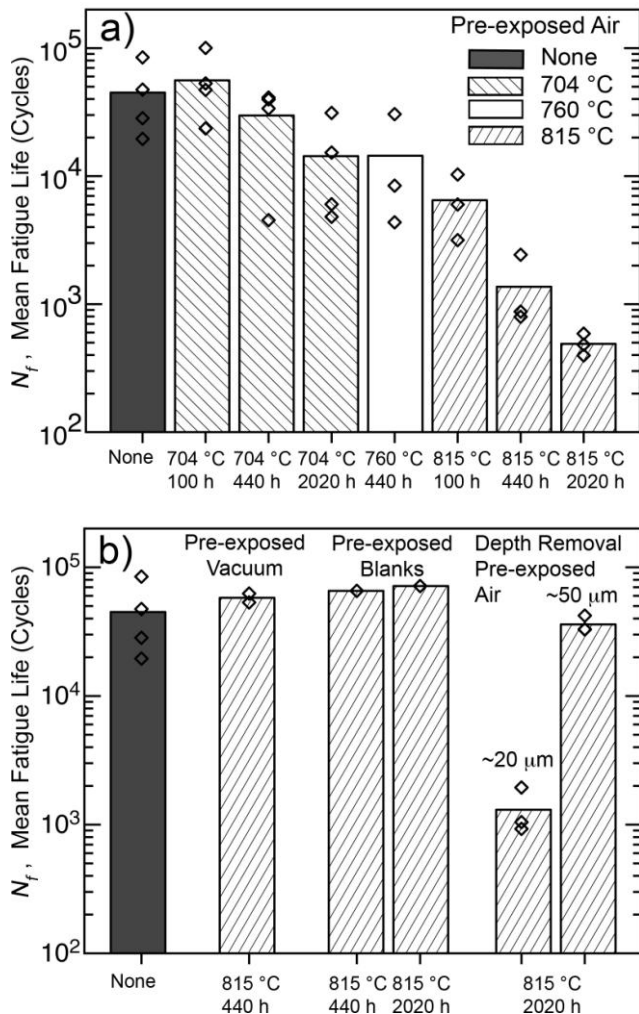


**Figure 9:** Multilayer EBC under development at NASA



### Oxidation of Turbomachinery Disk Alloys:

It is well established that oxidation can reduce fatigue life of disk alloys (ref. 14). Reduction in fatigue life poses significant durability challenges as temperature is increased to 704°C and higher. Recent studies at NASA (ref. 15) for an advanced powder metallurgy disk superalloy, ME3, has shown significant reduction in notch fatigue life after long-term exposure to air at high temperatures, as shown in Fig. 10. Oxidation at 704°C for 440 hr results in 2X reduction in notch fatigue life at the same temperature. Oxidation at 815°C results in 80-150 X reduction in notch fatigue life at 704°C. This reduction in fatigue life can be attributed to oxidation of the ME3 alloy (Fig.11), since no reduction in fatigue life is observed after exposure to vacuum at 815°C. Removal of the oxide layer restores the notch fatigue properties, as shown in Figure 10. Current research efforts to mitigate the effect of oxidation include development of oxidation-barrier coatings.



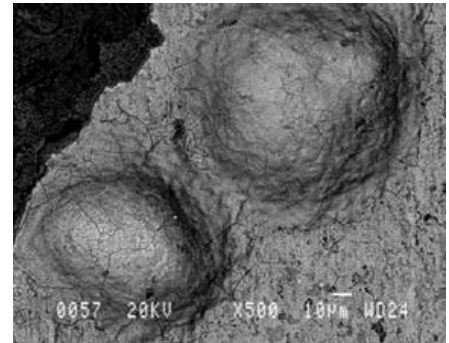
**Figure 11:** oxidation of ME3 at 760°C for 440 hr

**Figure 10:** Comparison of the mean notched fatigue lives at 704 °C of the unexposed ME3 specimens and ME3 specimens with a) prior exposures in air and b) alternate conditions as marked.

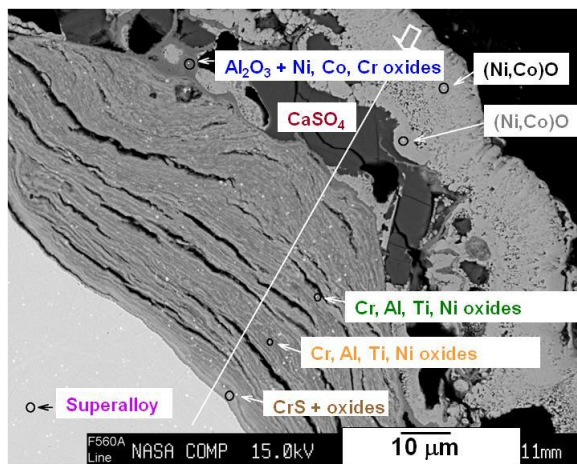
## Reaction of Hot Section Components with Airborne Contaminants

### *Molten Sulfate Induced Hot Corrosion:*

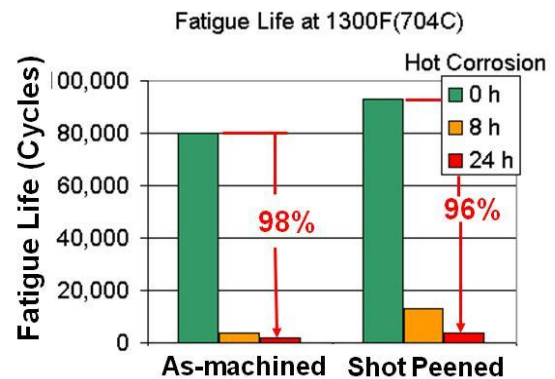
Molten  $\text{Na}_2\text{SO}_4$ -induced hot corrosion of turbine blade alloys was a major durability challenge in the 1970's and early 80's. However, hot corrosion has not been a major problem for turbine blade components since late 1980's. This may be attributed to (1) better alloys and coatings, and (2) higher temperatures in hot section of turbine engines, which prevents deposition of molten  $\text{Na}_2\text{SO}_4$  as the surface temperature of turbine components is greater than the dew point for molten sulfate deposition. Recently, as disk temperatures have increased from  $\sim 648^\circ\text{C}$  to  $\sim 704^\circ\text{C}$ , hot corrosion has surfaced as a new environmental degradation mode (ref. 16). Hot corrosion is due to deposition of salts on the disk alloy surface. Figure 12 shows a hot corrosion pit that has been observed in service. The pitting morphology (Figure 13) resembles typical Type II hot corrosion, which was identified as a new corrosion phenomena in 1970's for marine gas turbine engines (ref. 17). There is a significant reduction in fatigue life due to the formation of hot corrosion pits, as shown in Figure 14.



**Figure 12:** Pitting observed on higher temperature disk alloy



**Figure 13:** Typical Type II hot corrosion morphology observed for corrosion of disk alloys

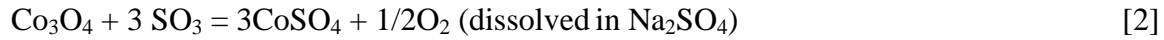


**Figure 14:** Fatigue life degradation due to hot corrosion of ME3 alloy

Hot corrosion mechanisms for higher temperature disk alloys are yet to be understood. It is not clear whether the salt deposit itself is molten or a melt forms by reaction of the salt with transient

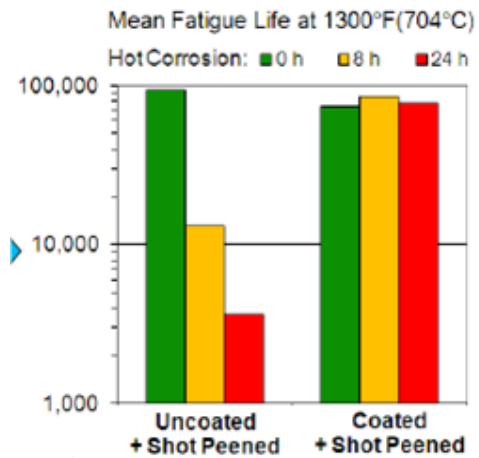


oxides that form during oxidation of disk alloys. In Type II hot corrosion encountered in marine gas turbines where  $\text{Na}_2\text{SO}_4$  is the primary deposit, a molten salt forms by the following reaction (ref. 17) involving transient metal oxides and  $\text{SO}_3$  gas in the turbine atmosphere.



Typically, the high temperature alloys form either a protective  $\text{Cr}_2\text{O}_3$  or  $\text{Al}_2\text{O}_3$  scale; however at low temperatures such as  $700^\circ\text{C}$ , it takes significantly longer times for protective scale to form and transient oxides such as  $\text{NiO}$  and  $\text{Co}_3\text{O}_4$  form in the initial period. It takes only 50 ppm  $\text{SO}_3$  at  $750^\circ\text{C}$  for  $\text{Na}_2\text{SO}_4$ -  $\text{CoSO}_4$  melt to form, whereas, 400 ppm  $\text{SO}_3$  would be required for  $\text{Na}_2\text{SO}_4$ -  $\text{NiSO}_4$  melt formation. Since disk alloys such as ME3 contain 20.6% Co, a  $\text{Na}_2\text{SO}_4$ -  $\text{CoSO}_4$  melt can form at  $700^\circ\text{C}$ . The other possible scenario is that the deposit is molten to begin with. For example,  $\text{Na}_2\text{SO}_4$ -  $\text{MgSO}_4$  system has a eutectic, which melts at  $\sim 680^\circ\text{C}$ . Once a melt forms, the corrosion is due to transport of  $\text{SO}_3$  through the melt (ref. 18), resulting in formation of oxide plus sulfide mixture.

Clearly there are a lot of unanswered questions related to the hot corrosion of higher temperature disk alloys: (1) What is the composition of the deposits? (2) How does a melt form? (3) What is the temperature dependence for corrosion? (4) How do the corrosion pits initiate and what is the critical pit size for adversely affecting fatigue life? Fundamental study is needed to answer the questions. Since the deposits are likely to be multi-component, complex phase relationships need to be studied and conditions for melt formation need to be identified. Hot corrosion experiments under controlled  $\text{SO}_2/\text{SO}_3$  partial pressures need to be conducted to define the corrosion regime as a function of temperature.



**Figure 15:** Effectiveness of coating in providing hot corrosion protection for ME3 disk alloy

One approach to mitigate the effect of hot corrosion is to apply a protective coating. Initial coating approaches appear promising (ref. 19), as shown in Fig. 15. The key challenge for any coating is that it must be thin and ductile so that the fatigue life of disk is not adversely affected by the coating.

NASA is currently developing advanced hybrid disk system with  $815^\circ\text{C}$  temperature capability. In this concept the rim region of the disk will be a single crystal nickel base superalloy or cast polycrystalline (PX) nickel base alloy that will be bonded to the powder metallurgy (PM) disk alloy with  $760 - 787^\circ\text{C}$  temperature capability. The hybrid design will overcome the temperature limitations of the PM disk alloys. Whether hot corrosion degradation of the  $815^\circ\text{C}$

rim region will pose a challenge is not known. It will be a function of the deposit chemistry and whether the deposit is molten to begin with. It is known that Type II hot corrosion is most prevalent in the temperature regime 700 – 760°C, and Type I hot corrosion occurs at temperatures in the range of 900 – 950°C. Typically there is a hot corrosion minimum at 800 – 850°C. However, the hot corrosion minimum is based on the assumption that the deposit is primarily Na<sub>2</sub>SO<sub>4</sub>, which is molten at 884°C and forms a molten Na<sub>2</sub>SO<sub>4</sub>-MSO<sub>4</sub> (M = Co or Ni) by reaction of Na<sub>2</sub>SO<sub>4</sub> with transient oxides and SO<sub>3</sub> in the gas turbine atmosphere. This may not be the case if a complex, multi-component molten sulfate deposit forms with low melting points. Studies are needed to understand the salt deposition processes, nature of the deposits, and melting point of the deposits to determine if the hot corrosion will be a problem for the 815°C rim region. It is likely that the hot corrosion affected zone will shift to lower temperature portion of the disk.

### ***Degradation Due to Molten Glassy Silicate Deposit:***

A new concern regarding environmental degradation has arisen from the persistent drive to increase the operating temperature of gas turbines and the presence of siliceous minerals (dust, sand, volcanic ash, runway debris) ingested with the intake air. These contaminants deposit onto the coated surfaces of the components, yielding glassy melts of calcium-magnesium aluminosilicate (CMAS) when the surface temperatures exceed ~1200°C (refs. 20, 21). Thermal barrier coatings (TBCs) are susceptible to degradation by CMAS, resulting in delamination of the TBC. Several mechanisms have been postulated for degradation of TBCs due to interaction with CMAS (refs. 22-25). The mechanisms include (1) chemical reaction leading to loss of strain tolerance in the coating and increase in stiffness of the coating, (2) residual stresses induced by presence of CMAS layer resulting in initiation of in-plane and vertical cracks leading to partial removal of TBC layer, (2) penetration of CMAS liquid and interaction with the thermally grown oxide (TGO) on bondcoat, resulting in delamination of coating along the TGO, and (3) CMAS promoting creep cavitation of the bond coat resulting in propagation of delamination crack path within the metal. It is likely that new degradation mechanisms will emerge with further increases in temperature.

Several potential solutions are being considered (refs. 25 -27) to improve resistance of TBCs to CMAS-induced degradation. Potential solutions include: (1) creating an impermeable layer on the top of the TBC that prevents infiltration of CMAS, (2) Addition of a sacrificial coating that reacts with CMAS to increase the melting temperature or viscosity so that the melt cannot infiltrate the TBC, (3) Addition of a non-wetting layer that minimizes contact between the coating and the molten deposit, (4) Manipulating the chemical reaction between the TBC and the melt to immobilize the melt by capturing the main constituents into crystalline phases and generating enough volume of reprecipitated products to fill the pore spaces and block access of any residual melt to the remaining TBC. So far, there have been some successes in improving

the resistance of TBCs to CMAS attack; however, much more remains to be done to improve the durability of TBCs in the presence of CMAS.

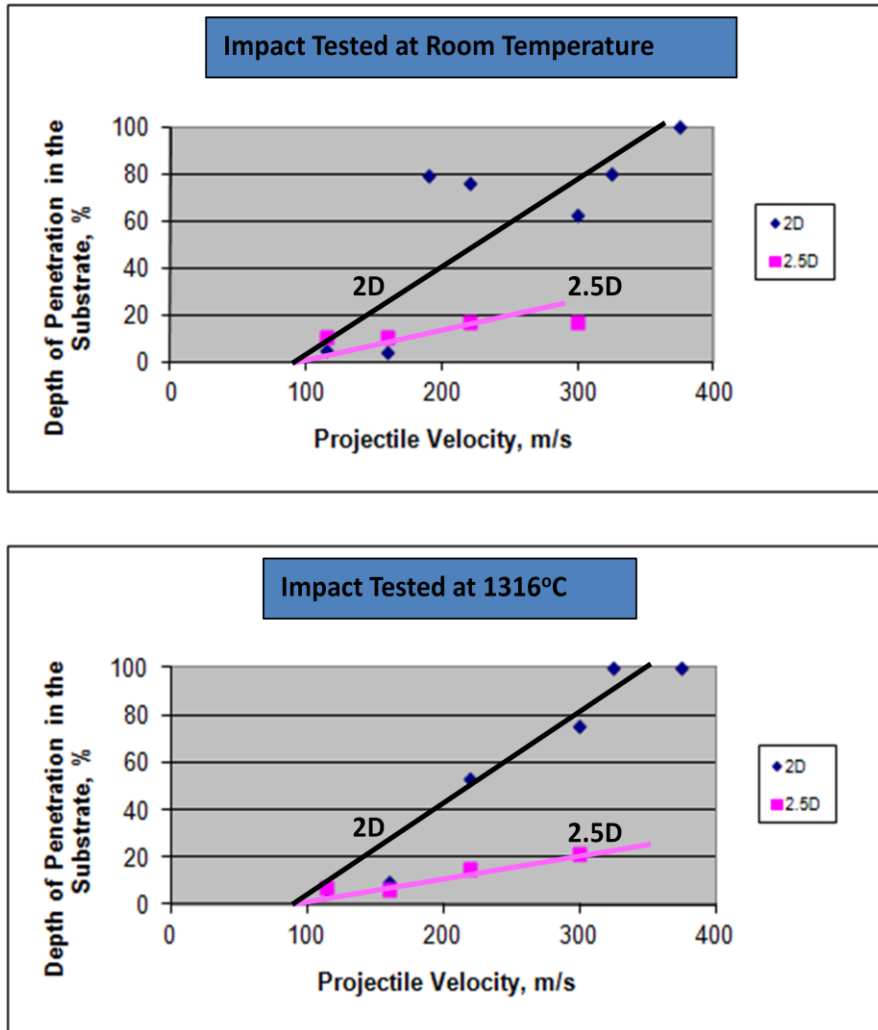
With CMC hot section components coated with EBCs, the temperature of the EBC surface is expected to be greater than 1315°C, and CMAS-induced degradation of EBCs will pose a major durability challenge for EBCs. The degradation of EBCs due to molten CMAS deposit is an emerging area of research (ref. 28) and will continue to be studied in the near future. Studies (ref. 28) have shown dissolution of BSAS EBC into CMAS and re-precipitation as a modified Celsian phase incorporating Ca, as well as secondary crystalline silicates that may degrade the durability and efficiency of the EBC. The process is aggravated by grain boundary penetration of CMAS into the polycrystalline BSAS. Recent studies at NASA (ref. 29) indicate degradation of EBC layer ( $\text{Yb}_2\text{Si}_2\text{O}_7$ ) resulting in formation of low melting grain boundary phase formation and loss of thermal conductivity with time due to delamination of the coating.

Fundamental research is needed to understand the degradation mechanisms associated with interaction of CMAS with EBCs. As future EBCs are being designed to operate at temperatures in the range of 1482 – 1650°F, the degradation of coating due to reaction with CMAS will pose significant durability challenges. Fundamental studies should be directed toward (1) understanding phase relationship for the complex EBC-CMAS system, (2) microstructure and morphology of the reaction product, (3) rate of evaporation as a function of temperature for the reaction product in the presence of moisture, (4) mode of penetration of the CMAS (e.g., through grain boundaries), (5) interaction of CMAS with the bond coat. All of the degradation mechanisms need to be studied as a function of temperature to define the temperature limits of potential EBC systems that are being developed. Mitigation strategies for reducing the extent of degradation of EBCs due to CMAS are expected to be similar to that being considered for the TBCs.

### **Degradation of Hot Section Components Due to Foreign Object Impact and Erosion**

Gas turbine engine components are subject to foreign object damage (FOD), which is induced by impact of foreign objects, which could be due to ingestion of sand, debris, rivet mandrels, and other substances. First stage high pressure CMC turbine blades would be prone to impact damage. The sizes of the hard foreign objects could be in the micron to millimeter range, which can damage the CMC turbine blades. Impact studies (ref. 30) at NASA with 1.59 mm diameter steel-ball projectiles at velocities ranging from 115 to 400 m/s have shown the extent of damage for uncoated 2-D woven SiC/SiC CMC increasing with increase in projectile velocity. At 115 m/s projectile velocity, the composite showed no noticeable surface or internal damage and retained its as-fabricated mechanical properties. Beyond 115 m/s projectile velocity, the extent of material degradation increased with increase in projectile velocity, with degradation of mechanical properties. At velocities > 300 m/s, the projectile penetrated the composite;

however, catastrophic failure of the composite was not observed. Predominant internal damage was observed to be delamination of fiber plies, followed by fiber fracture and matrix shearing. The delamination between the fiber ply and the matrix is due to low through-the-thickness tensile strength for 2-D SiC/SiC CMC. Impact studies for different fiber architectures (ref. 31) have shown the benefit of 2.5 D fiber architectures in improving the impact resistance, as shown in Fig. 17. Optimization of the fiber architecture will be a pathway for improving impact resistance of SiC/SiC CMCs.

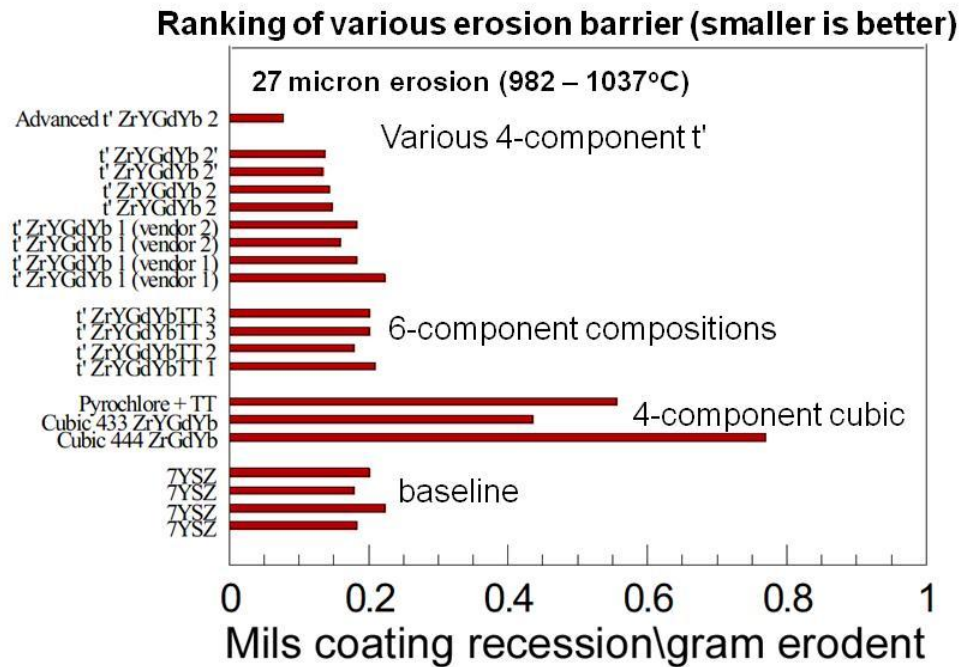


**Figure 17:** Effect of fiber architecture on impact resistance of SiC/SiC CMC at room temperature and 1316<sup>0</sup>C.

Studies on impact damage of SiC/SiC CMC with EBCs (ref. 32) have shown impact damage primarily confined within the EBC for velocities less than 160 m/s. As the projectile velocity increases, a part of the EBC begins to spall off and the transverse fiber ply close to the impact site delaminates. The weakest link in the EBC is the interface between the bond coat-EBC interfaces. These initial studies on impact of coated and uncoated SiC/SiC CMCs suggest that impact resistance can be increased by optimizing the fiber architecture of composite and increasing the interfacial strength for various layers in the EBC.

Micron size foreign object particles can cause erosion of turbine blades and coatings. Erosion of first stage turbine blade coatings due to sand ingestion is of major durability concern for rotorcraft engines. A major challenge for studying erosion of materials at high temperatures has been replicating gas turbine engine operating conditions in a laboratory rig. Recently, a Mach 0.3 – 1 high velocity burner rig capability (ref. 33)

has been established at NASA. This rig is designed to study erosion rates of TBCs as a function of particle size in 20 – 560 micron size range. Erosion related research using the new test capability at NASA for high temperature materials has so far been focused on developing advanced TBCs. Earlier NASA research (ref. 32) have demonstrated that multicomponent, oxide defect-cluster based low conductivity thermal barrier coatings, with 1/2 - 1/3 thermal conductivity of state-of-the-art YSZ TBCS, can increase the gas turbine engine blade temperature. However, such low thermal conductivity TBCs has low erosion resistance. Through modification of composition and structure of low thermal conductivity TBCS, new erosion-resistant, low thermal conductivity TBCs have been developed (ref. 34), as shown in Fig. 18. Advanced low conductivity TBCs with nano-toughened (t<sup>2</sup>) phase toughening mechanism have erosion resistance equal to the baseline YSZ TBC. Similar strategies are being developed for increasing erosion resistance of EBCs.



**Figure 18:** Low thermal conductivity TBCs with erosion resistance developed through coating composition and structure modification



## Summary

With increase in gas turbine engine operating temperatures, environmental factors will pose significant durability challenges for the next generation of high temperature engine materials. The environmental factors include reaction with oxygen and moisture in gas turbine engines, degradation due to ingestion of airborne contaminants that result in molten sulfate and glassy CMAS silicate deposits, impact and erosion due to ingestion of foreign objects. New materials for next generation of gas turbine engines include SiC/SiC CMCs, low conductivity TBCs, and higher temperature disk alloy system. The environmental factors have significant impact on durability of these new materials.

With the introduction of SiC/SiC CMC hot section components in gas turbine engines, surface recession due to reaction with moisture component of gas turbine environment will pose major durability challenge. Durable environmental barrier coatings (EBCs) to protect SiC/SiC CMCs from surface recession have been demonstrated for temperatures up to 1315°C. However, SiC/SC CMCs with temperature capability in the range 1482 - 1650°C will be required for future gas turbine engines. Development of EBCs with chemical stability both in moisture environment and with CMAS deposits, mechanical integrity, and chemical compatibility with the substrate CMC material will be a challenge for such high temperatures. Multidisciplinary fundamental research will be required to identify such coatings. NASA continues to advance the EBC technology and has aggressive goal for developing EBCs with 1482 - 1650°C capability. Research at NASA has demonstrated the initial feasibility of integrated, thick EBC/TBC system with 1650°C temperature capability. While such thick coatings are suitable for static CMC components, thinner coatings are required for rotating components for aerodynamic performance. The other environmental challenge for CMCs is protection from impact and erosion due to foreign object damage. Innovative fiber architectures and engineering of the EBC microstructures through alloying and reinforcement phases are potential solutions for protecting CMC components from foreign object damage.

Advanced thermal barrier coatings with 1/2 – 1/3 thermal conductivity of state-of-the-art YSZ TBCs have been developed. However, the advanced coatings suffer from surface erosion due to ingestion of micron-sized particles. Erosion resistant, low thermal conductivity TBCs have been developed through modification of microstructure and composition of low thermal conductivity coatings. The EBCs will also be subject to erosion damage. Based on the success in developing erosion resistant thermal barrier coatings, it can be envisioned that erosion resistant EBCs can be developed through new chemistries and microstructural engineering.

The environmental factors also have significant effect on durability of disk system with continued push for disk temperatures as high as 815°C. The key durability challenges are oxidation and hot corrosion due to molten salt deposit, both of which reduces the fatigue life significantly. Better understanding of the environment-induced degradation of disk alloys as a function of temperature is required. While coatings are potential solutions to minimize

environmental degradation of disk alloys, coatings need to be very thin so that fatigue properties are not adversely affected by the coating. The effectiveness of very thin coatings in providing protection against molten-salt induced hot corrosion is not a proven concept yet.

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