MATERIALS ISSUES FOR USERS OF GAS TURBINES

by
Henry L. Bernstein
President
Gas Turbine Materials Associates
San Antonio, Texas

ABSTRACT

Material issues are of paramount importance for the reliable and cost effective operation of gas turbine engines. Because of the high turbine temperatures, the materials degrade over time, which requires their repair or replacement. Rapid degradation or the inability to repair these complex materials results in excessively high operating costs.

This paper concentrates upon the hot section components, because this is where the majority of turbine distress occurs. Superalloys, both nickel base and cobalt base, and high temperature coatings are discussed from a user's perspective. The degradation of the hot section components — combustor hardware, blades and vanes — is described. This degradation includes that due to creep, thermal mechanical fatigue, high temperature oxidation and hot corrosion. Approaches to determining the remaining life of these components is discussed, including the data that the user must supply. A computer program for determining the remaining life of the first stage blades and vanes is presented. Finally, the repair and rejuvenation of the materials and components are described, including heat treatment, welding, brazing and recoating.

INTRODUCTION

Material issues are of paramount importance for the reliable and cost effective operation of gas turbine engines. Because of the high turbine temperatures, the materials degrade over time. This degradation must be reversed during overhaul, either by the repair or replacement of the component. It has been estimated that hot section component costs comprise from 50 to 70% of the maintenance costs for gas turbine operation. Furthermore, the timing of the overhaul of the engine is governed by durability of the hot section components, and the majority of engine failures are caused by failure of the hot section components

By contrast with the hot section components, other material issues in industrial gas turbines are of less importance. While still important when they occur, the durability of compressors, casings, shafts and disks (or wheels) is sufficiently great that material issues are less frequent. Thus, the focus of this tutorial will be on the hot section components.

Several excellent books about gas turbine materials are available. Two widely used books are The Superalloys (Sims and Hagel, 1972) and Superalloys II (Sims et al., 1987). While the second book is intended to succeed the first book, there is still valuable material in the first book that was not included in the second. Other excellent books are a collection of articles entitled Superalloys Source Book (Donachie, 1984) and Superalloys: A Technical Guide, 2nd Edition (Donachie and Donachie, 2002). A detailed discussion of high temperature coatings can be found in several articles (Committee on Coatings for High-Temperature Structural Materials, 1996, Bernstein and Allen, 1993, Bernstein, 1991, and Bernstein and McAlpin, 2006).

Objective

The objective of this tutorial is to acquaint gas turbine users with the various materials, components and degradation mechanisms that occur in industrial gas turbines. The repair of these components and materials is discussed, as is the prediction of their lifetimes. It is not possible to treat each of these subjects in the detail that they deserve in a tutorial paper. Such a treatment is the subject of at least one book, if not several. Instead, the objective is to provide a framework and understanding that the readers can build upon as they learn more about materials and encounter different material issues in the operation of their gas turbines. Such an understanding is especially important for the types of materials and coatings. If one can grasp the basic principals behind these materials, then one can sort through the myriad number of different materials and their variations that exist in the marketplace.

The readers should bear in mind that high temperature materials is an area that we are continually learning more about, encountering situations that have not occurred before, developing new materials, and applying older materials in new ways and in more demanding applications. As such, it can be both an exciting area, and one of great frustration, especially when one is seeking solutions to specific problems. Confusion in the materials area occurs partly because we have incomplete knowledge, and partly because of the desire to provide simple explanations to complex phenomena.

HOT SECTION COMPONENTS

In this section, the major hot section components are briefly reviewed, including the principal modes of degradation and the various terminology used to describe them. Also, included is a description of the various terms and definitions of the temperatures in the gas turbine.

As in many technical (and non-technical) areas, there is no single terminology to describe the components of the gas turbine. Most of the different terms in use are described in this section, along with the nomenclature that will be used in this tutorial. In addition, nicknames are often used in place of formal names, especially for more cumbersome terms. Examples of these nicknames are “rat ears” for the floating seal stops and “bullhorn brackets” for the transition piece support brackets.

Combustion Hardware

The combustion hardware consists of the area in which the fuel is burned and the adjacent area where the hot gases are guided into the turbine. The area in which the gases are burned is called the combustion liner or the combustion basket (liners and baskets for short). The area which guides the hot gases has a variety of names. Common names are transitions and transition pieces for United States manufacturers and hot gas casings for European manufactures. A schematic of a combustion liner is shown in Figure 1, and a transition piece is shown in Figure 2.

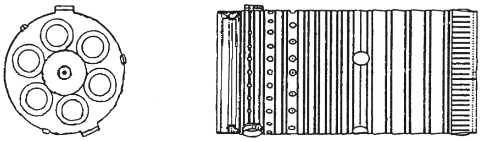

Figure 1. Combustion Liner With Six Fuel Nozzles. The Fuel Nozzles Are In The Left End, and the Combustion Gases Exit the Right End.

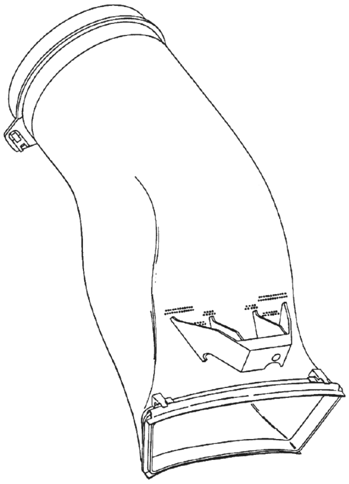

Figure 2. Transition Piece. The Combustion Liner Fits in the Top and the First Stage Vanes at the Bottom.

Nozzles and Vanes

The stationary airfoils in the turbine section are typically called either vanes or nozzles. Vanes is a more generic term and will be used in this tutorial. The term nozzle describes the function of the vanes to accelerate the flow of the hot gases. The term stationary blades is used by some European manufacturers. A schematic of some vanes is shown in Figure 3.

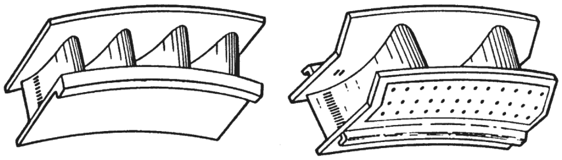

Figure 3. Vanes or Nozzles. Vanes are also made with one airfoil per segment.

Blades and Buckets

The rotating airfoils in the turbine section are typically called either blades or buckets. Blades is a more generic term and will be used in this tutorial. The term buckets is analogous to the function of buckets on water wheels. The term moving blades is used by some European manufacturers. A schematic of an industrial blade is shown in Figure 4, and that of an aeroderivative blade is shown in Figure 5.

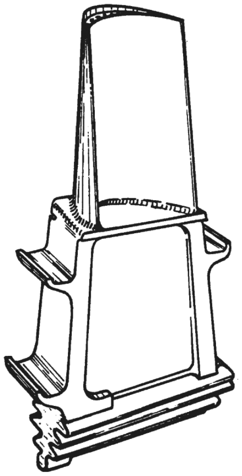

Figure 4. Heavy Duty Industrial Blade or Bucket.

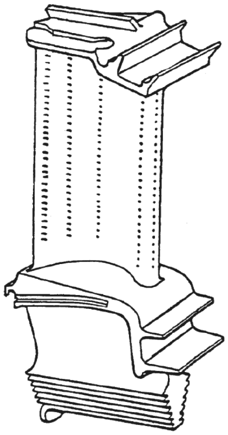

Figure 5. Aeroderivative Blade or Bucket.

Stages

Turbines are composed of a series of blades and vanes. Each set of blades and vanes is called a stage, or a row, as in 1st Stage or Row 1. In some engines, there are two shafts instead of a single shaft. In these engines, the blades and vanes closest to the combustor are often called the high pressure (HP) blades and vanes. The HP blades are on the high pressure shaft (or spool). The blades and vanes further away are called the low pressure (LP) blades and vanes. The LP blades are on the low pressure shaft (or spool).

Some engines have a separate turbine to provide the output energy, which is called a power turbine, or PT. For these engines, the gas turbine that drives the PT is called a gas generator, or GG.

Temperatures

There is a fair amount of confusion in the terminology for gas turbine temperatures. The same term may mean two different things, depending upon the speaker or type of gas turbine. When in doubt, it is best to obtain a clear definition of the term used.

The most common temperatures used in industrial gas turbines are compressor inlet temperature, compressor discharge temperature, firing temperature and exhaust temperature. The definition of these terms, as well as other terms, are given in Table I.

Table 1. Definition of Temperatures.

Name	Definition
Total Temperature	The temperature of the air at zero velocity. This temperature is also called the stagnation temperature. The kinetic energy of the moving air is converted into a higher temperature.
Static Temperature	The temperature of the moving air as measured by a stationary, or static, observer.
Relative Temperature	The total temperature relative to the rotor. Since the rotor is moving, the total temperature relative to it is lower than it is relative to the stator.
Ambient Temperature	The temperature of the air outside of the air intake.
Compressor Inlet Temperature	The temperature of the air as it enters the compressor. Since the velocity is low, the static and total temperatures are approximately the same.
Compressor Discharge Temperature	The total temperature of the air as it leaves the compressor.
Burner Discharge (or Outlet) Temperature	The total temperature of the air as it enters the first stage vanes.
Combustor Discharge Temperature	Same as the burner discharge temperature.
Firing Temperature	The total temperature of the air as it enters the first stage blades.
Turbine Inlet Temperature (TIT)	Either the firing temperature or the burner discharge temperature depending upon who is speaking. Need to obtain a definition of TIT.
Rotor Inlet Temperature (RIT)	Same as the firing temperature.
Exhaust Temperature	The total temperature in the exhaust of the turbine, typically measured away from the last stage of blades.
Temperature After Turbine (TAT)	Same as the exhaust temperature.
Blade Path Temperature	The total temperature measured just after the last stage of blades.

SUPERALLOYS

The hot section components are primarily constructed of superalloys, both nickel base and cobalt base. Iron base superalloys are still in use in older engines, and sometimes for colder components on newer engines. They may also be used for discs (or wheels). However, this tutorial will not address these iron base superalloys because of both their limited use for gas turbine blading, and a lower frequency of material issues associated with these alloys.

Superalloys derive their name for two reasons. First, they contain so many different alloying elements, they can be truly said to be a "super" alloy. Second, they have such good mechanical properties at high temperatures (and at low temperatures as well) that one could describe them as "super."

Cobalt Base Superalloys

Cobalt base superalloys are primarily used for vanes because of their good weldability, and resistance to hot corrosion. Because they cannot be made as strong as nickel base superalloys, they are not commonly used for blades.

Cobalt base superalloys are composed of cobalt alloyed primarily with chromium, tungsten, carbon and nickel. Chromium provides protection against hot corrosion and oxidation. Tungsten strengthens the metal by solid solution hardening. Carbon forms carbides with chromium. These carbides are the primary means of strengthening the alloy. Nickel maintains the cobalt atoms in an FCC crystal structure. (Note: FCC refers to a face centered cubic crystal structure. The normal crystal structure for cobalt is HCP (hexagonal close packed), but the HCP structure is not as ductile as the FCC structure.) The alloys X-40, X-45 and FSX-414 are based upon this group of alloying elements. The microstructure of FSX-414 is shown in Figure 6.

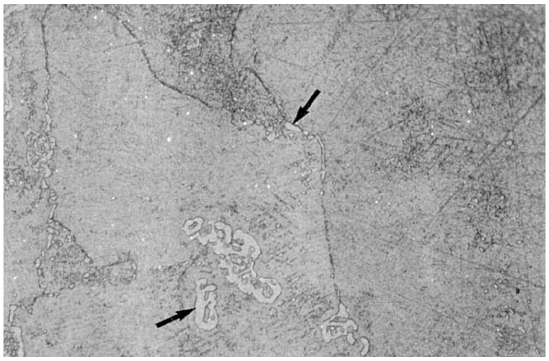

Figure 6. Microstructure of a Cobalt Base Superalloy, FSX-414. Arrows Show the Larger Carbides, Original at 500X.

In addition to the alloying elements above, titanium, zirconium, columbium and/or tantalum are sometimes also added to cobalt base superalloys, such as in MarM-509 and ECY-768. These elements form additional carbides, further strengthening the metal.

Nickel Base Superalloys

Most of the gas turbine alloy development has been with the nickel base alloy system because of the ability to achieve greater strength with this system. These alloys form gamma-prime second phase particles during heat treatment, which impart very high strengths to the alloy. Gamma-prime has the general composition of X_3Z, where X is primarily Ni, and Z is primarily Al and Ti. (Gamma-prime is generally written as $Ni_3(Al,Ti)$). Ta and Cb can substitute for Al and Ti, and Co can substitute for Ni. Therefore, a more accurate formula would be $(Ni, Co)_3$ (Al, Ti, Ta, Cb).)

Nickel base superalloys can be classified into solid solution alloys, and gamma-prime (or precipitation hardened) alloys. The solid solution alloys, which can be either cast or wrought, contain few elements that form gamma-prime particles. Instead, they are solid solution strengthened by refractory elements, such as tungsten and molybdenum, and by the formation of carbides. They also contain chromium for protection from hot corrosion and oxidation, and cobalt for microstructural stability. Because these alloys are not precipitation hardened, they are readily weldable. Common examples of these alloys are Hastelloy X, Nimonic 263, IN-617, and Haynes 230. The microstructure of IN-617 is shown in Figure 7.

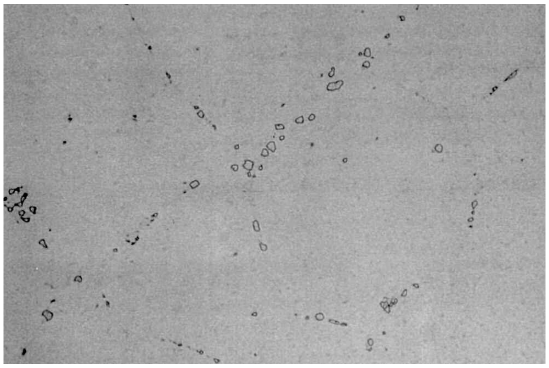

Figure 7. Microstructure of a Solid Solution, Nickel Base Superalloy, IN-617, Original at 500X.

The gamma-prime (or precipitation hardenable) alloys can be either cast or wrought. The cast forms are more common because of the economies of casting complex shapes, the ability to maintain very high mechanical properties by vacuum casting, and the difficulties encountered when forging metals having excellent mechanical properties at high temperatures. In addition to the formation of gamma-prime particles, which is the principal strengthening mechanism, these alloys also incorporate strengthening by solid solution hardening and carbide formation.

The gamma-prime superalloys are composed of many alloying elements. Chromium is used for resistance to environmental attack. Aluminum and tantalum assist in the resistance to environmental attack. Cobalt is used to stabilize the microstructure. Aluminum, titanium, tantalum and columbium are elements that form gamma-prime. Refractory elements, such as tungsten, molybdenum, tantalum and columbium are used for solid solution hardening. (Note: Chromium and cobalt also contribute to solid solution hardening.) These same elements, along with chromium, form carbides with the carbon that is added to the alloy. These carbides primarily strengthen the grain boundaries. In addition to these major elements, there are several elements added in minute quantities (sometimes called fairy dust) that strengthen the grain boundaries. These elements include boron, hafnium and zirconium. The microstructure of a common gamma-prime alloy, IN-738, is shown at two different magnifications in Figures 8 and 9.

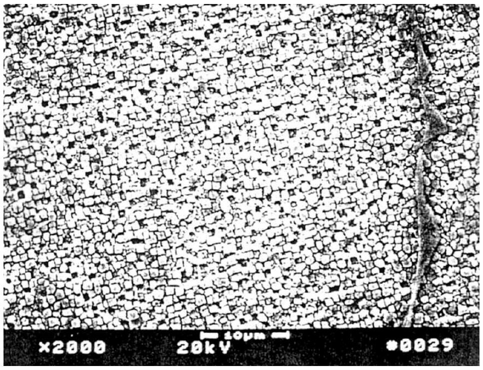

Figure 8. Microstructure of a Gamma Prime Nickel Base Superalloy, IN-738, Original at 2,000X. A grain boundary is on the right hand side of the micrograph.

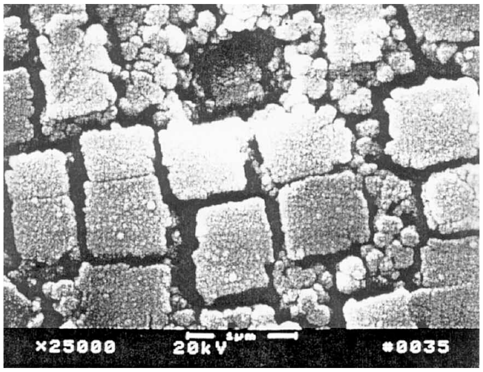

Figure 9. Microstructure of a Gamma Prime Nickel Base Superalloy, IN-738, Original at 25,000X.

Gamma-prime (or precipitation hardenable) alloys can be cast in three forms — conventionally cast, directionally solidified and single crystal. A conventionally cast form is where there are multiple grains having no preferred orientation, and is the type of casting provided unless one of the other two forms is specifically identified. Conventionally cast alloys (sometime abbreviated as CC) are sometimes called equiaxed alloys (abbreviated as EA), which means that the dimensions of the grains are equal in all directions, or axes. (Equiaxed is a misnomer because the grains at the surface are typically much longer in the direction perpendicular to the surface.) Directionally cast alloys (abbreviated as DS) have the grains oriented in one direction. These produce long grains that have a preferred orientation in this long direction. Single crystal alloys (abbreviated as SX or SC) consist of a single grain, or crystal, having a preferred orientation along one direction. For both the DS and SX alloys the preferred orientation is chosen so as to given the maximum mechanical properties in this orientation (which is the <001> crystallographic direction).

DS alloys typically are stronger than their counterpart CC alloys. The grain boundaries of DS alloys are aligned with the major stress in the blade (which is along the length of the blade). CC alloys have grain boundaries perpendicular to this major stress. Since grain boundaries are frequently the location for creep and fatigue crack formation, placing them perpendicular to the major stress reduces the strength of the CC alloy as compared to the DS version of the alloy. (This relation will not hold if the crack location is within the grain.) In addition, the DS crystals are oriented in a direction with a lower elastic modulus, which increases the thermal fatigue life of a DS blade.

SX alloys are stronger than both CC and DS alloys. SX alloys have the advantage of no grain boundaries, so that crack initiation must occur within the grain. More importantly, the lack of grain boundaries means that alloying elements required to strengthen the grain boundaries are not needed, which increases the incipient melting temperature of the alloy. (The incipient melting temperature is when melting first begins, which occurs at the grain boundaries.) This higher melting temperature allows the alloy to be heat treated at higher temperatures, so more gamma-prime can be formed. The greater amount of gamma-prime gives the SX alloys higher strength levels.

The strengthening mechanisms for the DS and SX alloys are the same as for the conventionally cast alloys. The chemical composition of the DS alloys may be the same as the conventionally cast alloys. The SX alloys are optimized to take advantage of the lack of grain boundaries, by removing elements, including carbon, whose main purpose is to strengthen the grain boundaries.

Examples of the more commonly used CC alloys for industrial gas turbines are IN-738, GTD-111, MarM-247 and Udimet 520. Each of these alloys can be made in a DS version, such as GTD-111DS and DS MarM-247. The SX alloys are usually different from the CC and DS alloys. Examples of these alloys include CMSX-4, SC-16, R-6, and PWA 1484. SX alloys are used in advanced, and higher temperature, aeroderivative and industrial gas turbines. They have been used in aircraft engines for a much longer time than in aeroderivative and industrial gas turbines.

COATINGS

Two basic types of coatings are used on the hot section blading for protection from environmental attack — diffusion coatings and overlay coatings. These coatings are sacrificial because they are attacked instead of the base metal. They are more resistant to high temperature attack than the base metal because they contain large amounts of aluminum and chromium, which form aluminum-oxide and chromium-oxide scales. These scales act as a physical barrier that reduce the rate of high temperature attack to very low values. When these scales are lost due to spallation, erosion, or other damage, a new scale forms from the aluminum and chromium in the coating. When the aluminum and chromium in the coating have been consumed below a minimum level for protection, then the coating ceases to protect the base metal and the airfoil must be refurbished.

The protective elements in coatings are aluminum and chromium because they form the protective oxide scale. Nickel and/or cobalt form the matrix of the coatings. Other elements used in coatings, but in smaller quantities, are silicon, platinum, rhodium, palladium, yttrium, hafnium, and tantalum. Silicon improves the hot corrosion resistance. The other elements improve both the oxidation and hot corrosion resistance of the coating.

Yttrium and hafnium improve the adherence of the oxide scale to the substrate by reducing the amount of the oxide that spalls off during a thermal cycle. (*Note:* Platinum, rhodium and palladium also reduce oxide spalling, due to reasons that are not understood.) These elements are sometimes called “active elements” because of their affinity for oxygen. It has been shown that yttrium (Smialek, 1991) forms sulfides with sulfur in the coating. If unreacted, this sulfur migrates to the oxide-coating interface, weakening this interface and increasing the amount of oxide spalling.

Diffusion Coatings

Diffusion coatings are nickel-aluminide, NiAl, and cobalt-aluminide, CoAl. NiAl forms on nickel-base superalloys and CoAl forms on cobalt-base superalloys. Diffusion chrome coatings are also used. A micrograph of a nickel-aluminide diffusion coating is shown in Figure 10.

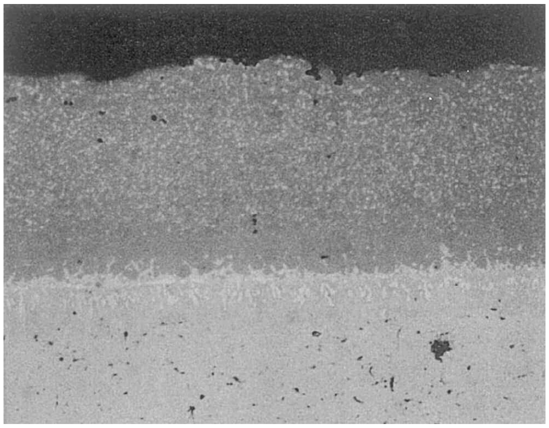

Figure 10. Micrograph of an Aluminide Diffusion Coating, Original at 800X.

Diffusion coatings are formed by depositing a layer of aluminum on the metal surface and heating the component in a furnace. During this heat treatment, the aluminum and metal atoms migrate, or diffuse, into each other, which is the reason these coatings are called diffusion coatings. This processing can be performed by pack cementation processes, chemical vapor deposition (CVD) processes, slurry processes, or other processes — all of which involve different methods of depositing the aluminum on the surface. Internal passages can be coated by filling them with an aluminum-rich powder, or by flowing an aluminum-rich gas through the passages.

Aluminide coatings for industrial gas turbines are often modified by the addition of chromium, silicon, platinum or other noble metals. Chromium and silicon impart resistance to hot corrosion and chromium can improve resistance to high temperature oxidation. Platinum significantly improves the resistance to high temperature oxidation and high temperature hot corrosion.

Chrome diffusion coatings are applied by a pack cementation process similar to aluminide coatings. The coating thickness is limited to the 0.001-0.002 inch (0.025-0.051 mm) range, and the chromium concentration is limited to about 35%. These coatings provide resistance to hot corrosion.

Overlay Coatings

Overlay coatings are a layer of a special alloy applied to the metal surface, and is selected for maximum environmental resistance. Overlay coatings can be made thicker than diffusion coatings, which may provide more protection to the base metal. The microstructure of an overlay coating is shown in Figure 11.

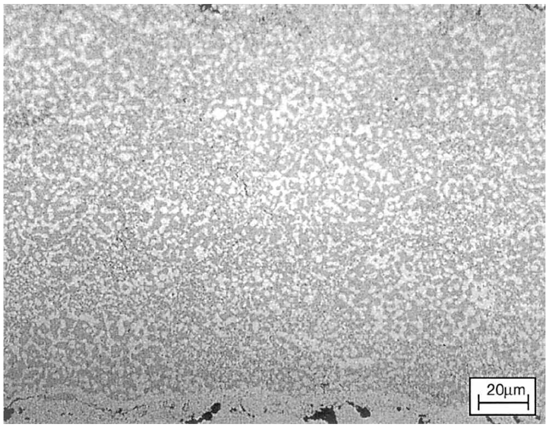

Figure 11. Overlay Coating Made By HVOF (Courtesy of Howmet).

There are a wide variety of overlay coating compositions. These coatings are usually called MCrAlY (pronounced M crawl-ee) coatings, since chromium, aluminum and yttrium are almost always present. 'M' is either Ni, Co, or a mixture of these elements. The concentrations of the coating elements depend on the intended use of the coating. Other elements such as silicon, hafnium, and tantalum are sometimes added to the coating to improve the performance. The aluminum forms an intermetallic compound with the nickel or cobalt, NiAl, Ni_3Al or CoAl. Overlay coatings, such as NiCrSi, also can be made without aluminum or yttrium and can protect airfoils up to about 1560°F (850°C) (Bauer, et al., 1985).

Chromium and aluminum provide the oxidation and hot corrosion resistance of the coating. Their levels are adjusted to provide maximum resistance to one or another form of attack, or are balanced to provide protection from several types of attack. Higher aluminum is used for oxidation protection, and higher chromium is used for protection from low temperature hot corrosion. In order to be fatigue resistant, the aluminum and chromium levels should be kept low because higher levels reduce the ductility of the coating. However, if these levels are too low, there will be inadequate protection from high temperature attack.

The majority of overlay coatings are composed of cobalt, nickel, or a combination of the two. Cobalt is better for hot corrosion, and nickel is better for high temperature oxidation. A mixture of both cobalt and nickel is a compromise for protection from both types of attack. NiCrAlY coatings are more ductile than CoCrAlY coatings, but NiCoCrAlY coatings can be more ductile than either (Hect, et al., 1975). CoCrAlY coatings containing high levels of aluminum are prone to cracking (Linask and Dierberger, 1975).

Overlay coatings are generally applied by high velocity oxygen flame (HVOF) and plasma spray, although other procedures such as electron beam physical vapor deposition and chemical vapor deposition are available. Frequently, a diffusion anneal treatment is carried out after coating to obtain diffusion between coating and substrate, which gives good bonding, and homogenizes the as-sprayed microstructure.

Plasma spraying is a process in which a powder of the coating material is heated and accelerated toward the part being coated by a plasma gas stream. Since it is a line-of-sight coating process, it is necessary to rotate and tilt the part and plasma gun during coating. Internal passages, such as cooling holes, cannot be coated by plasma spray.

Plasma spraying can be performed in air at atmospheric pressure (air plasma spraying), in air with an inert shielding gas such as argon (shielded plasma spraying), or in a vacuum chamber. The vacuum processes are called vacuum plasma spraying (VPS) or low pressure plasma spraying (LPPS). The LPPS process decreases contamination of the powder and base metal from oxidation, and, therefore, substantially improves coating quality. The process also produces high density coatings with virtually no unmelted particles, and it allows the substrate to be heated without oxidizing. Shielded plasma spray can produce excellent coatings, but requires special procedures and equipment to be used (Taylor, et al., 1985).

HVOF coatings are made by heating the powder and propelling it to the substrate by a high velocity flame. The high velocity allows denser coatings to be made than can be achieved by conventional plasma spray processes. The principal advantage to HVOF coatings is that they are less expensive than VPS and LPPS coatings because a vacuum chamber is not required.

Duplex Coatings

Duplex coatings are a combination of overlay and diffusion aluminide coatings. The overlay coating is applied first. Then a diffusion aluminide coating is applied on top of the overlay coating to provide increased protection by substantially increasing the aluminum content of the outer layer. However, this increased aluminum level makes these coatings more prone to cracking of this outer layer. A micrograph of a duplex coating is shown in Figure 12.

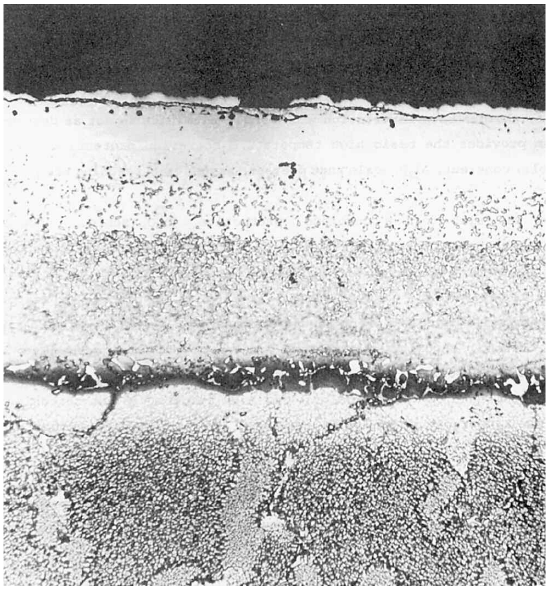

Figure 12. Micrograph of a Duplex Coating, GT-29+, (Bernstein and Allen, 1993).

Thermal Barrier Coatings

Thermal Barrier Coatings (TBC's) are ceramic coatings applied over metal substrates to insulate them from high temperatures. They consist of zirconium oxide, ZrO_2, stabilized by about 8 wt% yttria, Y_20_3, or magnesia, MgO. They are applied over a bond coat that is usually an MCrAlY overlay coating but can be a diffusion coating. The ceramic layer is typically 0.005 to 0.015 inch (0.13 to 0.38 mm) thick, and the overlay bond coat is usually 0.003 to 0.008 inch (0.08 to 0.20 mm) thick.

The bond coat improves the adhesion of the ceramic top coat by reducing the oxide buildup underneath the ceramic. (The ceramic is porous to oxygen, and will spall off when the oxide formed on the bond coat is sufficiently thick.) The bond coat is applied in the same manner as for diffusion or overlay coatings.

The ceramic layer is normally plasma sprayed in an air environment. This layer can be made by electron beam physical vapor deposition (EBPVD) and is used on aircraft engines and some aeroderivative and industrial engines. The ceramic layer made by EBPVD is columnar grained and is more resistant to spalling than the plasma sprayed TBC (Sheffler and Gupta, 1988). EBPVD TBC's are more expensive than air plasma sprayed TBC's. A plasma sprayed TBC is shown in Figure 13. An EBPVD TBC is shown in Figure 14.

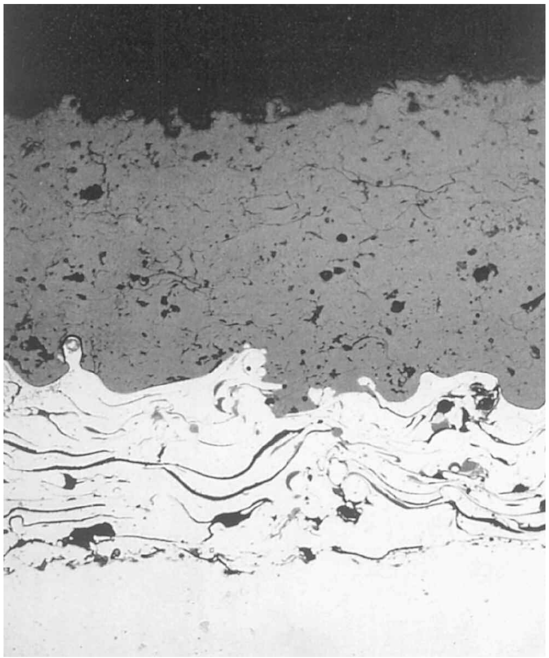

Figure 13. Plasma Sprayed Thermal Barrier Coating, Original at 250X.

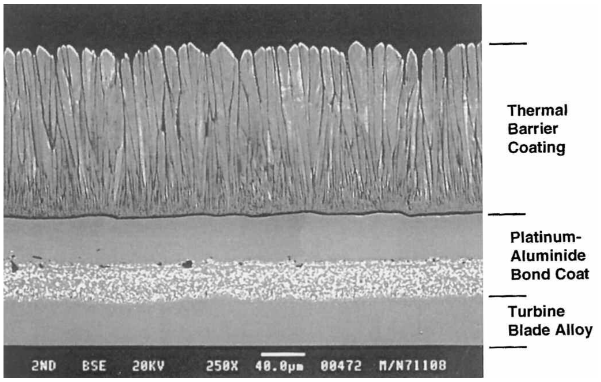

Figure 14. EB-PVD Thermal Barrier Coating on Top of a Platinum-Aluminide Bond Coat, (Courtesy of Howmet).

There is also a type of plasma sprayed ceramic layer that has a columnar structure. It is called a segmented TBC, and is made by a proprietary process. This segmented TBC can be made much thicker than a typical plasma sprayed TBC, and will provide more thermal insulation.

The TBC protects the metal by acting as an insulator between the metal and the hot gases. The thermal conductivity of the ceramic is one to two orders of magnitude lower than the metal. Although thin, the TBC can significantly reduce the metal temperature provided that the metal component is air cooled. (The air cooling provides a heat sink.) Furthermore, the ceramic has a higher reflectivity than the metal. This means that more of the radiative heat is reflected away, which is important for combustion hardware.

During startup and shutdown the TBC improves the thermal fatigue life by reducing the magnitude of the temperature transients the metal is exposed to. A 10 mil thick TBC on airfoils in experimental aircraft engines has achieved a 300°F reduction of metal temperature (Brindley and Miller, 1989). During steady state operation, the TBC lowers the temperature of the underlying metal, thereby improving its durability. It also reduces the severity of hot spots.

The principal failure mechanism of thermal barrier coatings is the spallation of the ceramic layer. Spallation is caused by the synergistic interaction of bond coat oxidation and thermal cycling. The oxidation occurs at the interface between the bond coat and the ceramic. As more oxide forms at this interface, the ceramic layer is more likely to spall.

DEGRADATION OF HOT SECTION COMPONENTS

The hot section components degrade by a number of different mechanisms. The most common mechanisms encountered in service are cracking, environmental attack, creep, wear and metallurgical embrittlement. Cracking is normally due to thermal mechanical fatigue and/or creep, and can be assisted by environmental attack and metallurgical embrittlement. Environmental attack is caused by oxidation and hot corrosion, and results in the loss of the surface profile. Environmental attack can lead to cracking of the metal in combination with fatigue and creep. (Coating degradation, which is a very important degradation mode, is considered as a form of environmental attack, since this degradation is caused by this attack.) Creep is the gradual distortion or elongation of the component, and can lead to cracking and/or wear. Metallurgical embrittlement is the loss of mechanical properties because of metallurgical changes within the metal. Wear processes, especially for combustor hardware, are important for surfaces in contact with each other.

High cycle fatigue is not considered a material degradation mechanism, although it certainly can lead to cracking and failure of the components. Instead, it is usually treated as a design issue, since both the cause and the solution to high cycle fatigue cracking is the vibrational response of the component, or the exciting forces that cause the vibration.

Creep

Creep is caused by the steady operating stress on the blades, vanes and combustion hardware in conjunction with the high temperatures that these components operate at. During creep, the metal elongates due to the tensile stresses acting on it. This elongation can result in tip rub of blades, loss of dimensions on vanes and bulges in combustion hardware. (On cooled blades, tip rub can close off exit holes for the cooling air, resulting in loss of cooling air flow and overheating of the blades.) Excessive creep strains will lead to cracking, and, at worse, fracture of the component.

Minimizing creep and preventing fracture by creep is one of the primary objectives during component design. Due to the conservative design practices typically used in industrial gas turbines, creep fractures are rare, but they do occur. When failures are due to a generic design deficiency, it is typically because the temperatures of the component are higher than predicted by the designers. In this case, the solution requires design or material modifications. When creep fractures occur due to operation, it is usually caused by excessive operating temperatures or the blockage of the cooling air channels.

Prediction of creep life, or the amount of creep life consumed, is difficult to perform by operators because they lack the requisite design data for this analysis, and because the calculation of the metal temperatures is quite difficult, even for the OEM. In the absence of creep cracking, there may be no means of measuring the amount of creep strain that has occurred, especially for blades.

Determination of the remaining creep life by testing metal samples from the component suffer from a number of methodological difficulties, including a potential change in creep mechanism between the test conditions and the operating conditions, an inability to test the creep critical location for many component designs, and a difference between the rupture life and the design criteria (which is usually a maximum creep strain).

Data from stress rupture tests on service-exposed buckets, new buckets, and cast specimens for IN-738LC has been collected (Bernstein, 1990) and is shown in the graph in Figure 15. The horizontal axis is the Larson-Miller Parameter (LMP), which is equal to $T'[20 + \log t]$. (*Note:* T is in absolute temperature, either °R or °K, and t is time, usually in hours. Factors other than 20 can be used, but 20 has been found to be adequate in most instances. There are other time-temperature parameters similar to LMP, but none has been shown to be consistently better than any other.) It can be seen that at low stresses, there is no loss in stress rupture life, but at higher stresses, there is a loss in life. The reason for this discrepancy is most likely due to the different metallurgical deformation mechanisms operating at the two stress levels as described by Koul and Castillo (1988) and McLean and Peck (1984). Thus, determining remaining life from stress rupture tests at high stresses can lead to conclusions inconsistent with those reached from tests at low stresses. At best, these tests tell one how the material compares to the original material, and if the material is in a very brittle condition.

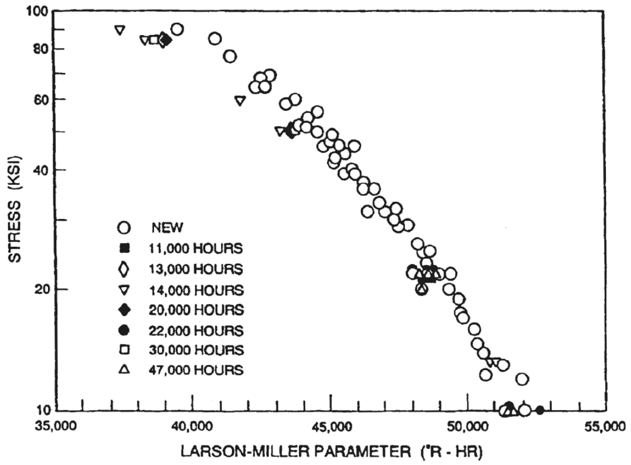

Figure 15. Stress Rupture Life of IN-738 Expressed in Terms of the Larson-Miller Parameter. Results Are Shown For New and Aged Material and For New and Service Run Blades (Bernstein, 1990).

Thermal Mechanical Fatigue

Thermal mechanical fatigue (TMF) occurs during the start-stop cycle of the engine. It is sometimes called thermal fatigue, or low cycle fatigue (LCF), although LCF usually describes constant temperature conditions. TMF derives its name because both the temperature and the mechanical load is cycled simultaneously.

During the start of the gas turbine, the cold metal parts are bathed in hot air from the combustor causing their exterior surfaces to heat up. Due to thermal expansion, these exterior surfaces try to elongate, but are restrained by the cooler metal interior. This restrained expansion causes the exterior surfaces to experience compressive stresses and strains, and the internal metal to have tensile stresses and strains (to counterbalance the compressive stresses on the external surfaces). When the engine is shut down, the reverse situation takes place. The flame is shut off, allowing much cooler compressor discharge air to bathe the hot metal parts, causing their exterior surfaces to cool down and attempt to contract. This contraction is resisted by the hot metal interior. This restrained contraction causes the exterior surfaces to experience tensile stresses and strains, and the internal metal to have compressive stresses and strains.

These alternating compressive/tensile stresses and strains on the external surface cause fatigue cracks to form and grow. These cracks will continue to grow, although at rates which may decrease. If the cracks become sufficiently long, creep can cause further crack growth and eventual fracture, or vibratory stresses can increase to the point that rapid crack growth by high cycle fatigue occurs, eventually leading to fracture.

As with creep, TMF cannot be detected until cracks have occurred. Prior to cracking, there is no way to determine the amount of TMF damage by inspection, nor is there any method to test for this damage.

Prediction of TMF life or the amount of TMF life consumed is difficult to perform by either operators or the OEM. Operators lack the requisite design data for this analysis. Because the calculation of the metal temperatures during starting and stopping is extremely difficult, even the OEM has great difficulty in predicting the TMF life in the absence of actual measurements of the components in the engine.

Metallurgical Embrittlement

Metallurgical embrittlement occurs when either new phases form or existing phases change their morphology, so that the material losses its strength and/or ductility. Nickel and cobalt base superalloys are meta-stable alloys, which means that they are not in their true equilibrium condition. These alloys are in a state of partial stability that has been created by cooling the alloy to freeze a relatively stable, but still non-equilibrium, condition. This condition is desirable because it has very favorable mechanical properties. At a high enough temperature and for a long enough time, these alloys will slowly change to their equilibrium condition, with an attendant loss of mechanical properties.

One of the metallurgical changes of most concern to superalloys are the formation of deleterious phases called topologically closed packed (TCP) phases or platelike phases. These phases, which are called sigma, nu and mu, are intermetallic compounds that form thin plates within the metal. (Eta can form a plate-like phase, but is not a TCP phase. It can be detrimental if it is large and numerous.) These TCP phases form because the local chemistry of the metal is favorable to the formation of these phases. Prevention of TCP phases is done by using a chemistry that will not form these phases, and by casting practices that minimize segregation (or enrichment) of TCP prone chemistries. Computer programs are available to determine which chemistries are likely to form TCP phases (Sims, et al., 1987, pp. 228-238).

TCP phases are very detrimental to mechanical properties, especially creep strength, when these TCP phases become numerous enough to form a semi-continuous path in the metal. Some superalloys are prone to form TCP phases, and most superalloys can tolerate some amount of TCP phases without loss of mechanical properties. However, the amount that can be tolerated is not well quantified. An example of sigma phase and its effect on creep life is shown in Figure 16.

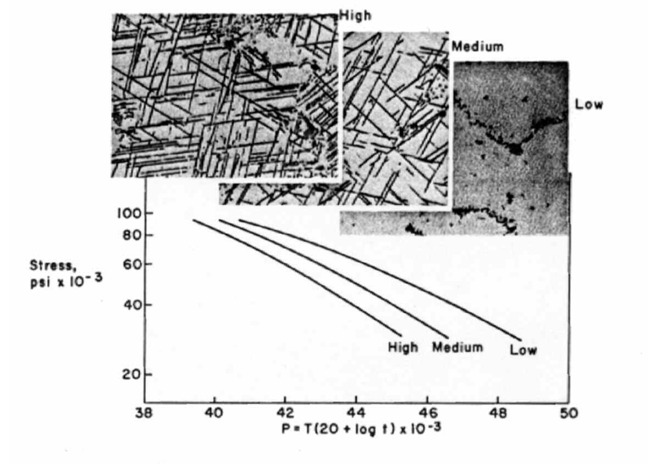

Figure 16. Micrographs Showing Different Levels of Sigma Phase Formation in IN-100 and the Effect Upon Creep Rupture Life (Expressed in Terms of the Larson-Miller Parameter) (Dreshfield and Ashbrook, 1969).

Another type of metallurgical embrittlement that can occur is the formation of a continuous grain boundary carbide. Since carbides are brittle, if they can form a continuous film around the grain boundary, the metal will be rendered brittle. However, if these carbides are not continuous, the metal will still retain some ductility.

Over time, the gamma-prime particles change their size and their morphology. Initially, there is a bi-modal distribution of gamma-prime size. (*Note:* Some nickel base superalloys have only a single size gamma-prime. The size of the particles is adjusted to give the desired mechanical properties.) The larger size particles are cuboidal shaped and have a side length of approximately 1 micron. The smaller size particles are spherical and have a diameter of approximately 0.1 micron. These particles can be seen in Figure 9. The large gamma-prime particles provide high temperature creep strength, and the fine gamma-prime particles provide low temperature tensile strength. Over time at high temperatures, the very fine gamma-prime are absorbed by the larger particles, which causes the larger particles to grow and to lose their cuboidal shapes. Eventually, these larger particles will agglomerate to form larger masses of gamma-prime. Micrographs of these changes in gamma-prime morphology are shown in Figure 17 through Figure 20. Under stress, there can be a directionality to the gamma-prime agglomeration, such that plates, or rafts, of gamma-prime particles can form.

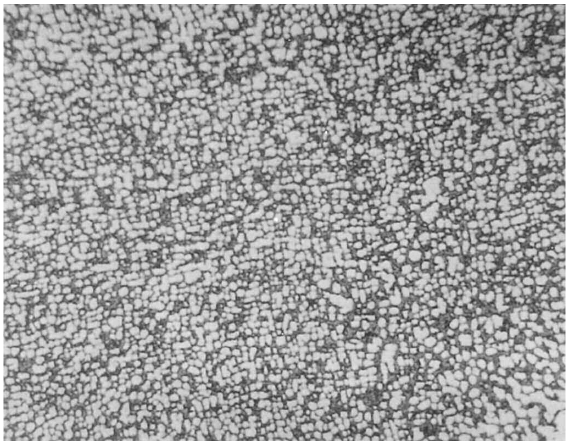

Figure 17. Change in Gamma-prime Morphology, 1000X. Original Structure.

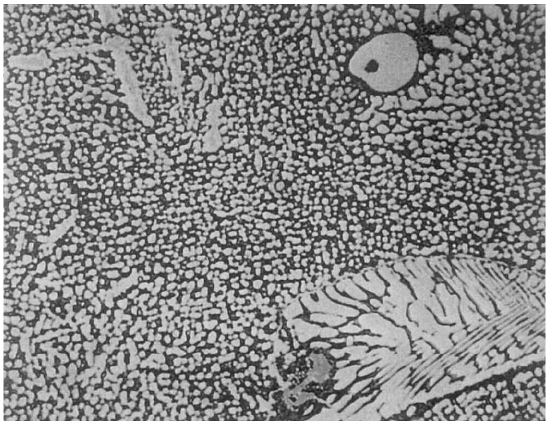

Figure 18. Change in Gamma-prime Morphology, 1000X. Small Amount of Change.

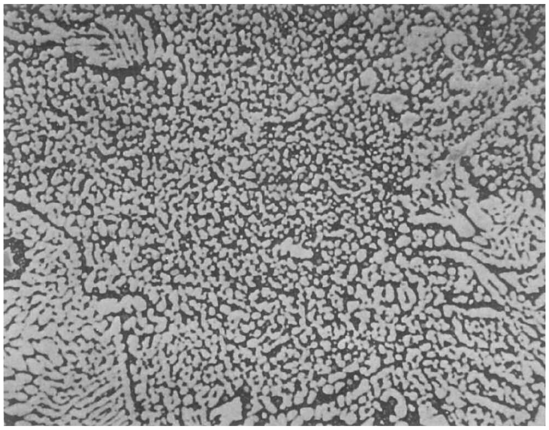

Figure 19. Change in Gamma-prime Morphology, 1000X. Moderate Amount of Agglomeration.

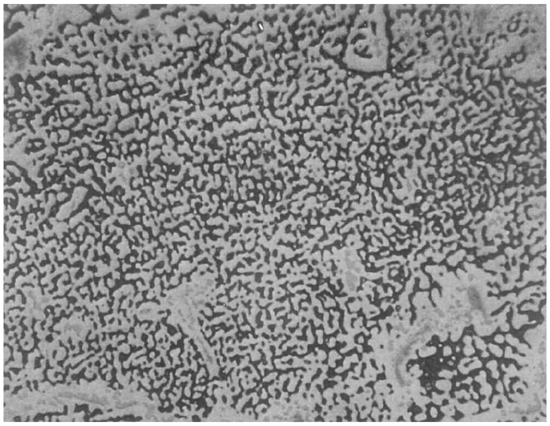

Figure 20. Change in Gamma-prime Morphology, 1000X. Large Amount of Agglomeration.

The effect of these changes in gamma-prime size and morphology on the serviceability of the hot section components is not clearly established. It is well accepted that if large agglomerations of gamma-prime particles form, then the component should be removed from service because of loss of mechanical properties. It is also known that if the fine gamma-prime particles disappear from the airfoil section, the component can continue to operate as the large gamma-prime particles still provide adequate creep strength. However, between these two extremes — where the large gamma-prime particles are growing, becoming spherical and agglomerating — it is not clear when the component should be removed from service and refurbished. Part of this difficulty is that the continued serviceability depends on the stress level in the component design. Low stress components will have more margin for microstructural change than high stress components.

Wear

Wear is the process of two surfaces under load moving with respect to each other. Wear can result in the removal of the surface metal and in the formation of cracks in the surfaces. It is a widely studied phenomenon, with many forms and sub-forms identified, and with its own terminology. The subject of wear is sometimes called tribology.

For hot section components, wear is most important for combustion hardware. The contacting surfaces of combustion liners and transitions are continually rubbing against each other due to combustor pulsations. And, during the start-stop operation of the engine, these surfaces can undergo large relative motion. Depending upon the fitup of the components, binding can be created which is detrimental to the wear of the components. An example of the wear of combustion hardware is shown in Figure 21.

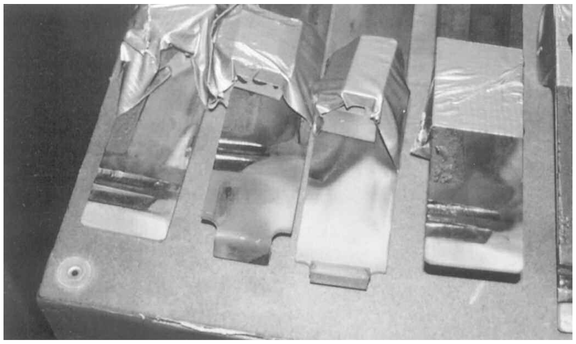

Figure 21. Wear of Sealing Plates Used Between Transition Pieces.

The ability to predict the wear of gas turbine components does not currently exist because of the inability to know, or predict, the dimensional fitup of the components, the difficulty of determining the relative movements during startup and shutdown, and the lack of measurements, or predictions, of combustion pulsations.

Environmental Attack and Coating Degradation

There are two basic forms of high temperature attack — oxidation and hot corrosion. Hot corrosion can be subdivided into two types — high temperature hot corrosion (Type I) and low temperature hot corrosion (Type II), although additional types are sometimes used. The approximate temperature regimes over which these forms of attack occur, and their severity, is shown in Figure 22.

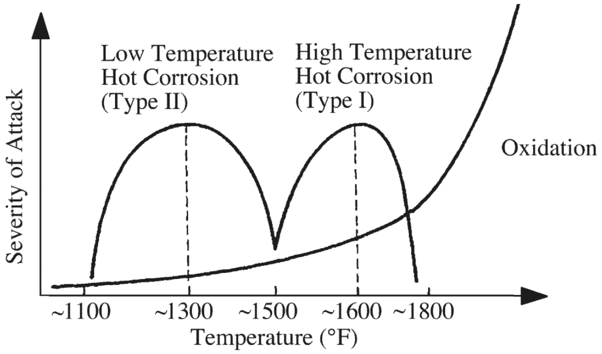

Figure 22. Approximate Regimes of High Temperature Environmental Attack (Committee on Coatings for High-Temperature Structural Materials, 1996).

Oxidation

High temperature oxidation is the oxidation of the metal and its alloying elements. It results in the formation of an external oxide scale that may or may not be protective. A protective scale grows slowly (due to a slow rate of diffusion through the scale) and is adherent to the metal substrate. For nonprotective scales, aninternally oxidized region occurs beneath the metal surface. Micrographs of external and internal oxidation are shown in Figure 23. High temperature oxidation accelerates with higher temperatures.

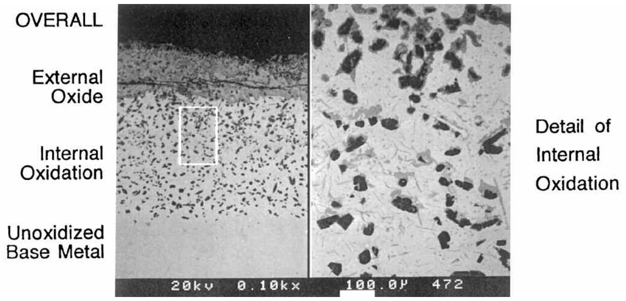

Figure 23. High Temperature Oxidation of MarM 421, (Courtesy of Solar).

Hot Corrosion

There are two types of hot corrosion, high temperature hot corrosion and low temperature hot corrosion. High temperature hot corrosion, also called Type I hot corrosion, occurs at metal temperatures between 1500°F to 1750°F (815°C to 955°C) when sulfur is present along with potassium or sodium. It is most severe at temperatures between 1550°F to 1650°F (845°C to 900°C). High temperature hot corrosion is caused by the formation of alkali metal salts, such as sodium sulfate (Na_2SO_4) and potassium sulfate (K_2SO_4), which melt and form a liquid on the surface. This molten salt attacks the oxide scale, destroying the protection normally afforded by the scale. Once the oxide scale is breached, sulfidation and oxidation of the metal occurs. High temperature hot corrosion results in the rapid destruction of the metal surface. A micrograph of high temperature hot corrosion is shown in Figure 24.

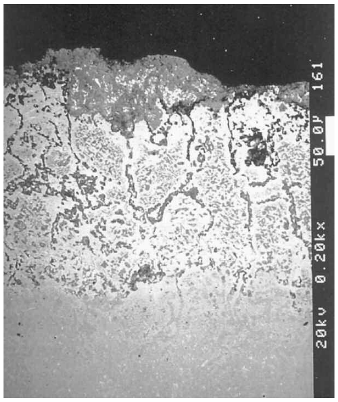

Figure 24. Micrograph of High Temperature Hot Corrosion (Type I) of FSX-414 Coated With RT-44, (Courtesy of Solar).

Low temperature hot corrosion, also called Type II hot corrosion, occurs at temperatures between 1100°F to 1500°F (595°C to 815°C) with a maximum rate of attack between 1250°F to 1350°F (675°C to 730°C). It is caused by the mixture of alkali sulfates with alloy metal sulfates, such as $CoSO_4$ or $NiSO_4$, and requires a significant partial pressure of SO_3. This sulfate mixture destroys the oxide scale and its protective ability. Low temperature hot corrosion results in pitting of the metal surface, a porous layered scale, little base metal depletion, and few internal sulfides.

A micrograph of low temperature hot corrosion is shown in Figure 25. The rate of pitting can be as great as the rate of attack by high temperature hot corrosion.

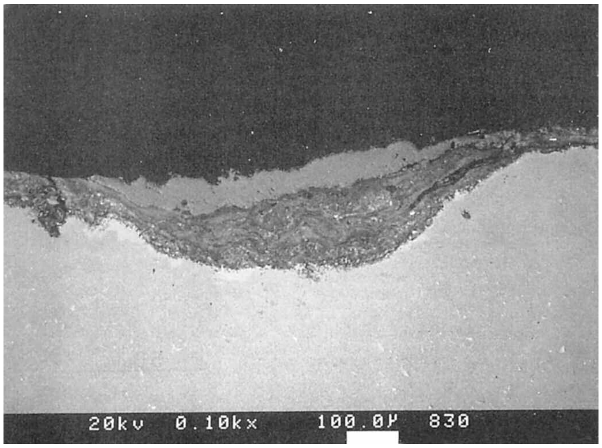

Figure 25. Micrograph of Low Temperature Hot Corrosion (Type II) of MarM 421, (Courtesy of Solar).

Coating Degradation

High temperature coatings are sacrificial in nature. They are designed to be consumed during the operation of the engine and then replaced during the refurbishment of the component. High temperature coatings provide their protection by forming a thin oxide layer, either aluminum oxide or chromium oxide, which acts as a barrier between the environment and the base metal and coating, as shown in Figure 26. This thin oxide is removed during engine operation by spallation and/or erosion. The coating then supplies sufficient aluminum or chromium to reform this protective oxide layer. The coating is simply a reservoir of the elements that form the protective oxide layer. When this reservoir is depleted by the successive loss and reformation of the oxide layer, then the coating is no longer protective. The coating is then attacked by oxidation and/or hot corrosion. After attacking the coating, this environmental attack continues through the coating and into the base metal.

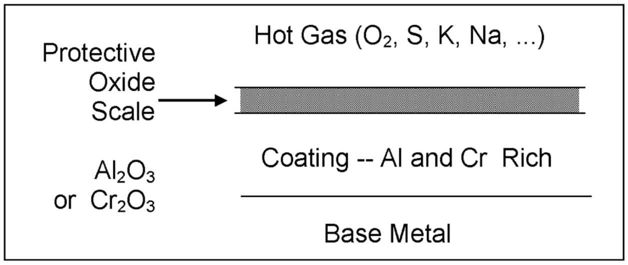

Figure 26. High temperature coatings protect the base metal by forming a protective oxide, which acts as a barrier to the corrosive gases.

The important aspect to note about coating degradation is that it is controlled by the chemistry of the coating, and not by the thickness of the coating. While the thickness can be an important variable, and cannot be overlooked, it is the aluminum and chromium content of the coating that determines when the coating can no longer provide adequate protection to the component.

The prediction of coating degradation, as well as base metal attack, is not well developed. Most OEM's rely upon empirical methods. These methods are based upon laboratory testing that is heavily supported by engine experience (Strangman, 1990). Methods have been developed for predicting coating degradation by high temperature oxidation (Chan, et al., 1998; Smialek, J. L., 2003), which is more amenable to prediction than is hot corrosion. Prediction of coating degradation by hot corrosion is complicated by the difficulty of determining what the corrodants are and their rate of deposition upon the surface.

Coatings can also degrade by cracking if the imposed tensile strains become too great. Most coatings undergo a ductile-to-brittle transition with temperature, as shown in Figure 27.Below a certain temperature, the coatings show very little ductility, but above this temperature, their ductility increases rapidly with increasing temperature. Components must be designed and operated such that the tensile strain in the coating does not exceed the strain to cause cracking in the coating. As shown in Figure 28, this cracking is most likely during the shutdown of the engine, when the surface of the component goes into tension. The most severe situation occurs during an emergency shutdown (or trip).

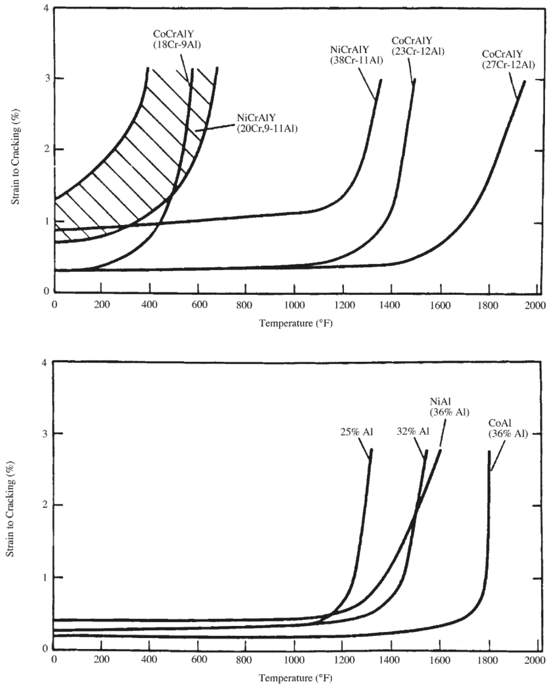

Figure 27. Coating Transition From Ductile to Brittle as a Function of Temperature (Bernstein and Allen, 1993).

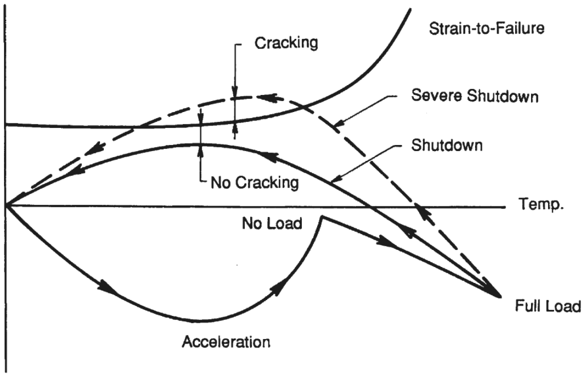

Figure 28. Schematic of the Strain and Temperature at the Leading Edge of a Blade Showing the Effect of Different Starts and Stops (Bernstein and Allen, 1993).

REPAIR OF HOT SECTION COMPONENTS

Because of their high cost, every effort is made to repair hot section components prior to scrapping them. Most components can be repaired once, some more than once and some an undetermined number of times. As repair technologies continue to improve, and new repair methods are developed, components previously thought to be non-repairable have now become repairable. Because of this possibility, some users save their scrapped hardware to await the development of better repair techniques.

In general, blades can be repaired at least once, and are usually designed to accommodate one repair cycle. Blades have been repaired more than once and operated successfully, but this requires a careful evaluation of the set of blades and their potential for refurbishment.

Vanes can be repaired several times, but with each repair cycle, the condition of the vanes deteriorates, and the subsequent repair becomes more costly. At some point, the full economic life of the vanes has been obtained, and it is more cost effective to replace the vanes with a new set.

Combustion hardware has the largest variability in the number of times it can be repaired. Some components on some engines appear to have a limitless ability to be repaired. Other components can be repaired only once. Because combustion components are fabricated structures, damaged areas can be cut out and new pieces welded into place.

The ability of a repair to return the component to its original condition is, in general, not well quantified. The typical repair is stated to bring the component to a “like new” or “almost new” condition. In some cases, this is a reasonable statement, in others, less so. The ability of the repair process to return the component to some known condition varies with the component, the repair process and the condition of the component when it was received. One of the limitations to quantifying the effectiveness of the repair process is a lack of knowledge of the properties of the repair metal and the repaired base metal.

Heat Treatment

During manufacturing, heat treatment of superalloys establishes the proper mechanical properties in the component. The heat treatment schedule is chosen so as to form a certain desired microstructure which will have optimum mechanical properties. In addition, the heat treatment schedule must not be so long or complicated that it is uneconomical to perform.

During service at high temperature, the microstructure changes over time, resulting in lower mechanical properties, the enlargement of certain microstructural phases, and the formation of new microstructural phases, some of which can be deleterious. During repair, the role of heat treatment is to restore the microstructure to its original condition as much as possible.

The microstructure is restored by first heating the metal to a high temperature to dissolve as much of the microstructural phases into the parent metal as possible. Then the metal is cooled to a lower temperature to either freeze this annealed microstructure, or to form a desired amount of certain microstructural phases so that the proper mechanical properties are obtained.

Heat treatment is also usually needed when welding superalloys. Most of the superalloys must first be annealed, or softened, prior to welding. After welding, they must be again heat treated to removed the effects of the high heat of welding, and to restore the proper microstructure and mechanical properties.

HIPPING

HIPPING refers to Hot Isostatic Pressing. It is a process in which the metal is heated to a high temperature under a high pressure. This method is used to repair creep damage by closing the voids in the metal created by creep. It is also used to close, or reduce, the amount of casting porosity. By applying a high external pressure at a high temperature, the metal will flow, causing these internal pores to both shrink in size and fuse together. It should be noted that surface connected porosity and creep damage cannot be removed by HIPPING because the oxidized surfaces of the pore or crack will not fuse together.

As a repair process, HIPPING is useful to remove porosity present in the original casting, although it can be argued that if the original casting could tolerate this porosity, then there is no need to remove it. It can also be used to remove creep damage, although for industrial gas turbines, the presence of creep damage is cause for concern and investigation because it usually should not be present in the first place.

HIPPING has another use in repair. It allows higher temperatures to be obtained during heat treatment because it raises the incipient melting temperature. These higher temperatures allow a more uniform structure to be obtained and most of the precipitates to be dissolved. The need for HIPPING depends upon the alloy and its condition. In general, neither heat treatment nor hipping can restore badly overaged microstructures.

Welding

Welding is widely used to repair combustion components. These components crack and wear. The cracks are welded up, and the worn areas are removed and new metal, or sections, are welded on.

Vanes and nozzles are typically welded if they are made of cobalt base alloys or nickel base alloys with a low to moderate amount of gamma-prime.

Blades are only welded at low stress areas, such as the tip and sealing surfaces. Because of the propensity of blade alloys to crack and the lower mechanical properties of the weld metal, welds are not allowed in higher stress areas.

Nickel base alloys containing large amounts of gamma-prime particles are difficult to impossible to weld by conventional methods. This is because these alloys will crack, either during welding, or during post weld heat treatment. The cracking occurs by a complex interaction of the formation of gamma-prime during welding or heat treatment, the change in the flow properties of the metal and the concentration of strain near the weld. Success in welding these alloys has been achieved by preheating the metal to high temperatures before welding, and/or by welding with very low heat input. The weldability of nickel base alloys as a function of aluminum and titanium content (the major gamma-prime formers) is shown in Figure 29.

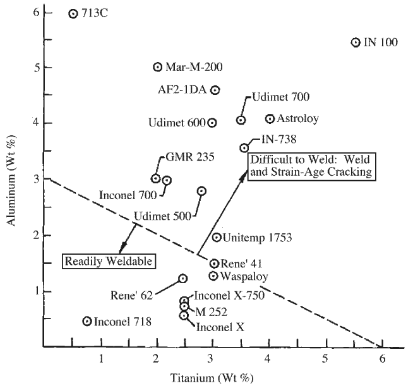

Figure 29. Weldability of Nickel Base Superalloys as a Function of Aluminum and Titanium Contents (After Prager and Shira, 1968).

All welding is done in protective environments to prevent oxidation of the weld metal or the base metal. This is most often accomplished by a protective gas sheath, as in TIG welding. Sometimes wielding of blades is done in a box filled with argon.

Brazing

Brazing is widely used in aircraft gas turbines to repair stationary components. It has been used less in industrial gas turbines, but is finding growing application.

Brazing consists of using a filler metal of a lower melting temperature than the parent metal to fill in missing metal and cracks. Usually boron and silicon are added to the braze metal to lower its melting temperature. Boron and silicon form brittle phases within the braze, and diffuse into the parent metal for a short distance, also reducing the properties of the parent metal.

Braze alloys have been developed that produce very good strength, sometimes equivalent to that of the parent metal, but braze alloys still usually suffer from lower ductility. This has been accomplished by minimizing the amount of boron and silicon used, and in the selection of high strength filler alloys.

Recoating

Recoating restores the surface protection to the base metal when the coating has either been breached or its protective ability exhausted. Recoating is widely done. The process of recoating is the same as that for a new coating, except that the old coating has to be removed, and any surface damage, or attack, removed as well. These is no limit to the number of times that a metal can be recoated, although most manufacturers recommend only one recoating cycle. This recommendation is based upon the removal of surface metal that occurs as part of removing the coating. When the thickness of the metal becomes too thin, especially adjacent to interior cooling surfaces, the component can no longer be repaired, or recoated.

LIFE PREDICTION OF HOT SECTION COMPONENTS

The prediction of the life of hot section components has been discussed to some extent in the sections on the different degradation modes. In this section, it is the intent to describe a specific example of life prediction that has been developed (Bernstein, 1990) for a large industrial gas turbine, the General Electric MS7001E (Frame 7E). The methods described herein are applicable for any life prediction scheme based upon engineering analysis. (*Note:* Life prediction methods based upon historical experience, including statistical approaches, use a different methodology than that described in this section.) This work was sponsored by the Electric Power Research Institute (EPRI) and was performed by the author. (*Note:* Portions of the work were conducted by Mr. Ted Russell, and others of the General Electric Company, and by staff at the Southwest Research Institute, who were all very helpful in the successful outcome of the project. Mr. James Allen was the EPRI project manager.)

There are two major sections in a life prediction approach, as shown in Figure 30. The first section is the development of the algorithms to predict component lives. The second is the actual prediction of these lives that usually involves the use of a computer program to perform these calculations.

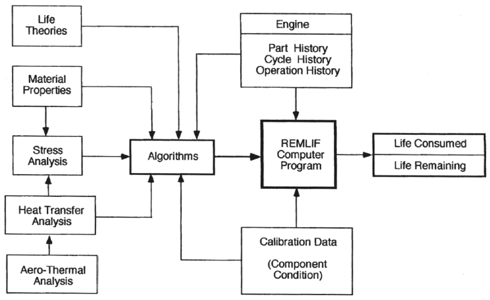

Figure 30. Diagram for the Life Management of Gas Turbine Engines.

There are five major elements that go into developing the algorithms. These are life theories (sometimes called failure theories), materials properties, stress analysis, heat transfer analysis and aerothermal analysis. The algorithms are calibrated and verified by means of field data whenever possible.

The life theories are the equations and theories that describe how the material degrades over time. These theories address creep, fatigue, environmental attack, etc. Material properties, which include tensile, creep and fatigue properties, are used in the life theories. These properties are determined by laboratory testing of specimens.

The stress analysis calculates the stresses and strains in the component, which are then used to determine the component life. The heat transfer analysis computes the temperatures in the component. These temperatures are necessary to determine the stress and strains in the component, since the majority of the stresses and strains are caused by thermal gradients in the component. Component temperatures are also critical because material properties and life can be strongly dependent upon temperature. The aerothermal analysis calculates the temperatures, velocities and pressures in the gas stream. These values are used in the heat transfer analysis to determine component temperatures. The air pressures may also be used in the stress analysis.

The algorithms for predicting the life are theoretical, semi-empirical or empirical. They need to be calibrated to actual field experience, when available, or to measurements of temperature and strain on the components. These measurements are performed by the manufacturer during the engine development, which is an involved and expensive undertaking. Operators normally do not make these measurements, although methods, such as optical pyrometry, are available. Instead, operators can make use of actual field data to calibrate the algorithms, especially when a known amount of degradation has occurred.

The second section in the life prediction approach is the actual prediction of the life. In the example described in this paper, this prediction is done by a computer program. The algorithms are incorporated into this computer program. The user then must tell the program the history of the component and the engine. With this information, the program is then able to calculate the amount of life consumed and the amount of life remaining for a given operating profile.

The component information required is typically the installation date (and, if applicable, removal date), and refurbishment type. The engine history is divided into information on starts and stops, and information on running hours. Start/stop information consists of the number and type of starts and stops, and especially any abnormal events, such as emergency stops (also called trips). Running hours information consists of number of hours, hours at any overtemperature condition, and, if available, load levels or engine temperatures and pressures. The more detailed the information that can be provided, the better the predictions will be.

The development of a life prediction system must be done for a specific engine model and component design, and not as a single, general system applicable to all engines. This requirement is because the design requirements and the results of the analyses differ among components, models, and manufacturers.

The specific components addressed in the example in this tutorial are the first stage nozzle and the first stage bucket from the GE Frame 7E. (Since the General Electric engine is addressed in this section, the GE terminology of bucket and nozzle will be used.) These components experience some of the highest temperatures and stresses within the engine, and are the cause for major maintenance activities. They are also two of the most expensive components within the engine. The degradation modes predicted are cracking for the first stage nozzle, thermal mechanical fatigue for the first stage bucket, and creep for the first stage bucket.

Cracking of the First Stage Nozzle

The life of the first stage nozzle is controlled by the amount of cracking that occurs. This cracking is caused by a combination of thermal mechanical fatigue, creep, environmental attack and metallurgical changes. Since these forms of degradation occur simultaneously and synergistically, they are treated as a single phenomenon. Furthermore, the heat transfer and stress analysis of the nozzle is extremely difficult to accurately perform because of the sidewalls, whose effects cannot be accurately modeled theoretically. As such, a semi-empirical approach was taken to predict the cracking of the first stage nozzle. Such an approach was feasible because of the availability of field data showing the cracking of the nozzle. A detailed discussion of the prediction of nozzle cracking is available in the literature (Bernstein, et al., 1994).

Cracking can occur on all portions of the nozzle. Cracking is geometrically complex and there are numerous cracks with various crack lengths. While no nozzle has completely fractured because of this cracking, excessive cracking leads to high repair costs and shortens the total life of the nozzle. What constitutes excessive cracking is determined by the operator of the gas turbine, because it primarily involves economic costs.

The first task in developing a model to predict the cracking on the nozzle is to select a parameter to describe the cracking. After investigating several different parameters, it was decided to choose the cracking that occurs on the pressure side of the airfoil because this is where the most extensive cracking occurs on the Frame 7E nozzle. (On the Frame 7B nozzle, this parameter is the cracking on the suction side.) The length of all cracks on the pressure side of each vane are added to compute the amount of cracking for each vane. The cracking on the worst vane is then found and is used as the parameter for predicting the life. The worst cracked vane will determine when to refurbish the nozzle, and is a guide to the refurbishment cost.

The amount of cracking is predicted as a function of the damage induced by fatigue cycling and the damage induced by steady operation at temperature,

$$\text{cracking} = A(\text{fired hours})^{B}(\text{cyclic damage})^{C}$$

Values for the constants A, B, and C were determined by a multiple linear regression with available 7E cracking data. The cyclic damage factor is a weighting of the amount of fatigue damage produced by the different cycle types. This factor was provided by General Electric and is considered proprietary to them.

The success of this method in describing cracking is illustrated in Figure 31, which compares actual cracking with the predicted cracking. The central diagonal line indicates perfect agreement, while the two outer lines indicate factors-of-two in error. There is no segregation of the data. The quality of the fit, in view of the normal 2x-3x scatter in laboratory fatigue life testing, is excellent. The data is correlated to within a factor of 2.0, except for two points which are correlated to factors of 2.1 and 2.2.

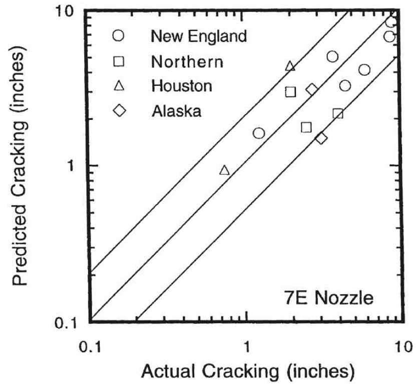

Figure 31. Predicted vs. Actual Cracking for Gas Turbine Vanes (Bernstein, et al., 1994).

Thermal Mechanical Fatigue Of The First Stage Bucket

Thermal mechanical fatigue (TMF) of the first stage bucket results in cracks that can lead to the fracture of the bucket, the premature removal of the bucket from the engine, or the scrapping of the bucket due to the inability to repair it. TMF cracking can occur at the leading and trailing edges of the bucket, on the suction side, or on the pressure side. Cracks normally start on the exterior surface and grow inwards, but they can also start on interior cooling surfaces and grow to the exterior. A detailed discussion of bucket TMF and its prediction can be found in the literature (Bernstein, et al., 1993, and Bernstein and Allen, 1992).

The life theory, or model, used for bucket fatigue was developed from laboratory test data. The parameters used are strain range, strain amplitude ratio, and dwell time. (Note: The dwell time is the time of operation during a start-stop cycle. The strain A-ratio is the ratio of the strain amplitude to the mean strain.) This combination of parameters is reasonable, since the strain amplitude ratio can account for mean strain effects, and the dwell time can account for both environmental effects and mean stress effects caused by steady-state stress relaxation, both of which can affect the fatigue life. The life model is,

$$N_f = C_0(\Delta\varepsilon)^{C_1}(t_h)^{C_2}\exp\left(\frac{C_3}{A}\right)$$

where N_f is the fatigue life, A is the strain A-ratio, De is the total strain range (expressed in percent strain), and t_h is the dwell time (in minutes). The values of the material constants for IN-738 coated with GT-29 are: $C_0 = 125$, $C_1 = -3.40$, $C_2 = -0.217$, and $C_3 = 0.247$.

A linear damage rule was used to account for the different cycle types

$$D = \sum \frac{N_i}{N_{fi}}$$

where the damage ratio D is equivalent to the amount of life consumed, and is 100% when a crack initiates. N_i is the number of cycles of type *i*, and N_{fi} is the number of cycles to failure of cycle type *i*.

Based upon a finite element analysis of the bucket and in conjunction with estimates of startup and shutdown strains, the TMF life at the leading edge of the bucket was predicted. (These predictions are for the bucket made from IN-738 and having the sharp leading edge design with 13 cooling holes.) The results of the predictions are shown in Figure 32 as the percentage of life consumed versus the number of starts. Below 100% life, cracks were not expected to be present. (Note: Cracks are defined for the purposes of life prediction as cracks of the order of 0.005 to 0.010 inch (0.2 to 0.4 mm) in depth.) Above 100% life, cracks are expected in the leading edge. As the percentage of life consumed becomes greater than 100%, the extent of cracking is expected to be greater, but the model does not make a quantitative prediction of the crack length. At about 100% life, cracks are expected to be present. Shown in Figure 32 is the cracking observed. It should be noted that a metallurgical analysis is required to determine if cracks are just beginning or not.

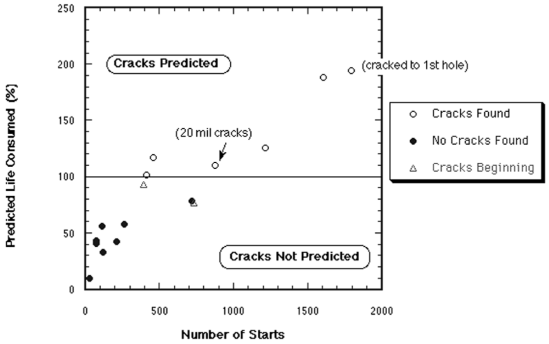

Figure 32. Thermal Mechanical Cracking of a Blade Leading Edge. Predicted Life Consumed as a Function of the Number of Starts. Field Results Are Shown (Bernstein, et al., 1993).

As can be seen in Figure 32, the agreement of the field data with the predictions was remarkably good. All points below the 100% line were not predicted to have cracking, and no cracking was found. All points above the 100% line were predicted to have cracking, and cracks were found. The points at which cracks were just beginning to form were predicted at 77% and 93% life consumed.

Some of the buckets for which predictions were made were metallurgically analyzed to verify the predictions. Figure 33 shows a crack in the leading edge of a bucket that has grown to the first cooling hole, a distance of 5 mm (0.020 inch), which is very severe. It was predicted that 194% of the life of this bucket was consumed, which is in agreement with the condition of the bucket. Figure 34 shows a crack beginning in the suction side of another bucket from a sister engine. Other cracks on the leading edge were found as deep as 0.5 mm (0.020 inch), which is in reasonable agreement with the predicted life consumed of 110%. Additional details about the metallurgical analysis of these two buckets have been presented by Bernstein and Allen (1992).

Figure 33. Thermal Mechanical Fatigue Cracking of the Leading Edge of a Blade From the Exterior (Top) to the Cooling Hole (Bottom), (Bernstein, et al, 1993).

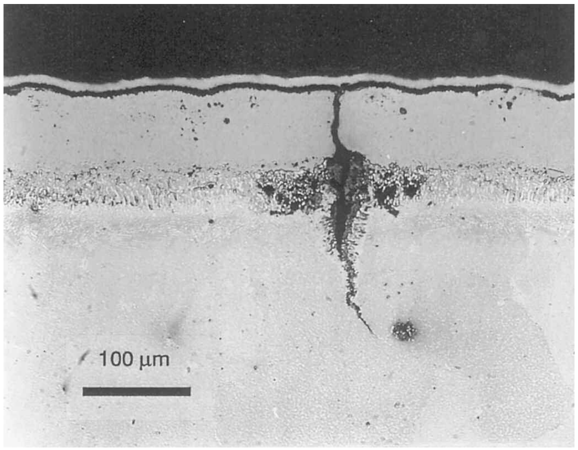

Figure 34. Thermal Mechanical Fatigue Cracking of the Convex Side (Suction Side) of a Blade After 11,000 Hours of Service (Bernstein, et al., 1993).

Creep of the First Stage Bucket

A second important determinant of the life of the first stage bucket is creep. The importance of creep depends upon how conservative the design of the bucket is. If sufficiently conservative, then creep will not be a significant operational concern. If less conservative, then creep can be the leading cause of bucket refurbishment.

The General Electric design practice for IN-738 is to design to 0.5% creep strain, instead of to rupture (or fracture). This criteria is conservative, and is based upon the observation that at greater than 0.5% creep strain, grain boundary cracks begin to form (Embley and Kallianpur, 1986).

The model used in the life prediction for creep of the bucket was provided by GE based upon their design practice, (*Note:* These predictions are for the bucket made from IN-738 and having the sharp leading edge design with 13 cooling holes)

$$t_f = f(T_{firing}, T_{cool})$$

where t_f, the time to reach 0.5% creep strain, is a function, *f*, of T_{firing}, the firing temperature, and T_{cool}, the temperature of the cooling air flowing through the bucket cooling holes. The firing temperature and cooling air temperature are determined by the computer program from measurements of the compressor discharge and exhaust conditions, or from the power level during operation. The amount of creep life consumed is,

$$\text{Creep Life Consumed} = \sum \frac{t_i}{t_{fi}}$$

where t_i is the time at condition i, and t_{fi} is the time to reach 0.5% creep strain at condition i. Each condition i represents a different combination of firing and cooling temperatures, or power levels.

The predictions of the creep life are shown in Figure 35, in which the predicted life consumed by creep versus the number of fired hours is plotted. It should be noted that the maximum coordinate shown is 50% life consumed. The Gulf Coast site is a baseload operation which is accumulating creep damage continuously. The New England site operates at a lower firing temperature than the Gulf Coast site and, consequently, does not accumulate as much creep damage. The Alaska site is operating in a load following mode in which much lower temperatures are involved for most of the operating time. It can be seen in Figure 35 that there is essentially no creep damage for the Alaska site.

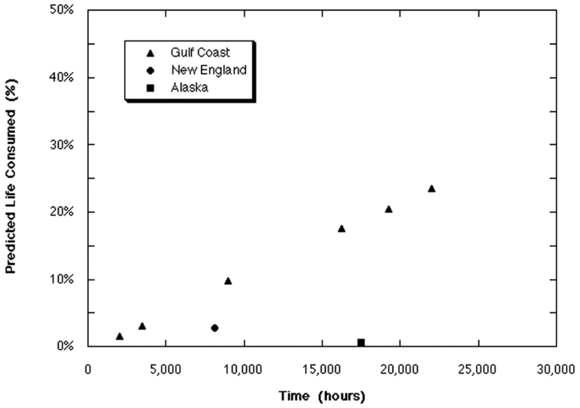

Figure 35. Prediction of Creep Life (Bernstein, 1990).

As described in an earlier section, it is not possible to determine the amount of creep damage in an air cooled bucket prior to the formation of cracks. Since no creep cracks were found, it is only possible to say that the creep damage observed was less than 100%.

CLOSING REMARKS

Gas turbines are a materials driven technology. The ability to achieve higher output and increased maintenance intervals is limited by the materials of construction of the hot section components. These materials and components are operated close to their maximum potential and are expected to be repaired or replaced over time.

This tutorial is designed to provide the reader with an overview of materials technology with an emphasis on the concepts behind this technology. Once the concepts are grasped, then the reader can better understand the myriad number of materials, varieties of material degradation, and repair processes that exist in the commercial world. The reader should also bear in mind that we are continuing to learn more about materials, especially as we are using them in new and more demanding ways.

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