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MATERIALS AND STRUCTURAL ASPECTS OF ADVANCED GAS-TURBINE HELICOPTER ENGINES

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INTRODUCTION

The key to improved gas-turbine-engine performance, whether the ultimate application is a large transport aircraft or a helicopter or whether the planned usage is commercial or military, lies in the successful development of advanced materials as well as advanced structural and design concepts. Although each application calls for differences in emphasis, generalizations can be made concerning the needed advances in the major engine components. Considerations of the various engine components indicates both the nature of the improvements needed and the resulting benefits of those improvements (fig. 1). For example, modifying the low-temperature components of helicopter engines (such as the inlet particle separator) can improve erosion resistance and, thus, increase engine component life. The use of composite materials (with their greater stiffness and strength) in the compressor and front frame can improve performance, reduce engine weight, and lower cost. Increasing the strength of materials used for intermediate temperature components (such as turbine disks) may reduce both engine weight and cost. Similarly, increasing turbine blade and stator vane use temperature will permit engine operation with a higher gas inlet temperature or with less cooling air; the resultant benefits are increased power output, decreased fuel consumption, and/or decreased maintenance cost. Improved engine environmental protection is needed because increases in high-temperature metal strength usually go hand-in-hand with decreased oxidation resistance. Furthermore, the eventual use of cheaper, less clean cuts of petroleum, brought about by the increasing cost and scarcity of petroleum products, can cause even more severe corrosion and erosion of turbine hot-section components than are experienced currently. Greatly improved coatings for turbine vanes, combustor liners, and blades are therefore essential to allow these turbine components to tolerate such high temperatures and fuels.

An important area in advanced gas-turbine helicopter engine technology that must be considered to a far greater degree than heretofore is economic viability. Clearly, the improved performance will be costly, and it may have to be compromised to achieve better durability and lower operating and maintenance costs. As always, the unfortunate corollary to higher cycle temperature is higher maintenance cost due to shortened component life. Engine hot-section parts are particularly susceptible to wear and damage, and require frequent replacement. For example, higher temperatures and lighter structures often result in engine case distortion. Thermal gradients and flight maneuvers can cause rubbing and wear, thus, opening up blade tip seal clearances and reducing engine efficiency. High thermal gradients also increase the likelihood of engine component fatigue failures.

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To achieve durability and life assurance (i.e., the ability to design to meet a required lifetime) requires that the interrelated aspects of higher operating temperatures and pressures, cooling concepts, and environmental protection schemes be understood and integrated into component designs. New methodology must be developed and verified, through laboratory tests of simple specimens, engine environment simulation studies, and finally in ground and flight engine tests.

In this paper, the authors review advances in materials, coatings, turbine cooling technology, and structural and design concepts, as well as assess future trends using selected, helicopter gas-turbine-engine components as a framework for discussion. To a degree, this paper is also an update of reference 1, a general progress report on high-temperature materials for aircraft gas-turbine engines. However, in this paper the helicopter application is emphasized. Also, with helicopter applications in mind, additional discussions of the cool end of the engine, as well as advances in hot-section component-life prediction and design techniques have been included, and general observations concerning life assurance and life management are presented.

STATIONARY COMPONENTS

Several stationary components of helicopter engines have been selected for discussion. They include the inlet particle separator, the front frame, rotor tip seals, vanes, and combustors. Each represents areas of technology that have some commonality with large turbojets and turbofans; however, in the following discussion, special emphasis has been placed on turboshaft engine requirements for helicopter operations.

Inlet Particle Separator

Since the advent of turbine engines, there has been a continuing concern over the ability of high-speed rotating components to withstand the effects of ingesting foreign objects. Many lessons have been learned through painful experience after fielding qualified equipment The well-recognized problem of erosion from sand and dust, as differentiated from large objects, such as birds and ice balls, has presented formidable tasks to the designer. The result is that it now is accepted practice to provide some sort of protection, either dust separators or filters, at the inlet to minimize erosion of aerodynamic components. Several types of inertial separators or barrier filters are used today, and it is clear that many military helicopters cannot survive without them. An example of the situation is shown in figure 2. Here, a helicopter rotor blade is barely visible through the sand and dust cloud produced by rotor downwash. Repeated use in these circumstances has been known to deteriorate compressor performance so rapidly as to impede operation and mission effectiveness, thereby forcing early component replacement (ref. 2). It should be noted however, that erosion may not be as severe in commercial operations, which are usually conducted from prepared sites.

Only in the 1960's, after several quick-fix approaches with fielded systems, have effective steps been taken; that is, making compressor airfoils more tolerant to the environment. Fewer blades with longer chords are one answer. Another is selecting blade materials that are stiff enough to withstand vibratory excitations, yet hard enough to resist erosion. Such materials as AM 355 and AM 350 are used, and, in some cases, hard coatings of titanium carbonitride have provided excellent blade leading-edge protection (ref. 3). Further improvements are needed to meet extended life requirements.

Engine placement cannot be ignored as a factor in reducing the amount of sand and dust ingested, but the need for an inlet particle separator still remains. As a result, attention must also be directed toward improving the life of the protection systems themselves. Most of these (fig. 3) are inertial protection systems that use blowers or fans to discharge the contaminated air overboard. As might be expected, in such systems blower life is low. In addition, surfaces of the separator are subject to particle impingement, and local wear can be a problem. This, of course, is deemed far more acceptable than parts changes within the compressor rotating section.

Composite Front Frame

Composite materials offer significant advantages over monolithic metals from the multiple standpoints of modulus, strength, and density. These factors are illustrated in figure 4, in which strength and modulus of various fiber-reinforced, polymer-matrix composites that have been normalized for density are compared with similarly normalized metals. The maximum values shown for each type of composite represent unidirectional reinforcement of the composite. The range of values shown for each type of composite indicates crossply effects, which account for the pseudoisotropic properties that occur in most practical applications of composites. It is apparent from the figure that all of these composite systems can operate in a regime that is inaccessible to monolithic metals and that varying degrees of improvement over metallic components are possible, depending on the particular ply layup selected. Although composites have so far seen little use in turbine engines, considerable research is underway in this area. Their potential application to helicopter gas-turbine engines is discussed herein in terms of several key components. In this section we deal with a primary potential application, the front frame structure.

It is extremely likely that a substantial saving in helicopter engine weight can be realized by substituting composites for metals in the cold-end structure such as the front frame. An outstanding example of weight saving in primary engine structure, although for a different type of aircraft, is noted in the Quiet, Clean, Short-Haul Experimental Engine (QCSEE) program being conducted at the NASA Lewis Research Center (refs. 4 and 5). Figure 5 is a cutaway view of the composite fan frame for this engine. The use of composites is expected to reduce frame weight by about one-third when compared with that of the usual metal construction; it also is expected to significantly reduce frame cost.

The QCSEE engine frame was built of a graphite-fiber-reinforced epoxy-resin composite in which the individual pieces were bonded into one continuous structure. Although composites are already used in gas-turbine aircraft engines as nonload carrying structures (e.g., sound absorption material), their use for primary structures presents a fertile field for further research.

It should be noted that major advances are being made in the development of higher use-temperature (315° C (600° F)) composite systems, such as the PMR (polymerization of monomer reactants) polyimide composites (ref. 6). The higher use temperature of the polyimide systems can provide an added dimension insofar as structural usefulness is concerned. In somewhat the same vein, the development of advanced computerized analysis techniques, such as NASTRAN, and design methods (ref. 7) afford the designer with the necessary tools to use composites to a much greater extent than was heretofore possible.

Tip Seals

An important way of enhancing helicopter gas-turbine-engine performance is to reduce and preserve turbine airfoil tip clearances. Increases in these clearances with engine operating time decrease turbine efficiency, as a result of gas leakage past the blade tips with attendant loss of thrust and increased fuel consumption. It has been estimated that in a large, high-bypass turbofan engine the turbine efficiency penalty could be as much as 1 percent for a 0.025-cm (0.010-in.) increase in first-stage turbine blade tip clearance. For helicopter engines, with their smaller blades, the penalty for such an increase is even more dramatic. To deal with this situation significant advances have been made (ref. 8) in the development of new shroud materials which act to preserve the airfoil tip clearance during engine operation. Shroud materials must have improved resistance to oxidation and erosion. At the same time they must have the quality of abradability.

Their response to rubbing by the airfoil tip must be such that rubbing can occur without material removal from the blade. An abradable seal minimizes the clearance increase caused by rubbing of the airfoil tips against the shroud so that rub clearances could be reduced by as much as 70 percent. Figure 6 schematically illustrates the effect of airfoil and seal interaction on tip clearance.

Figure 7 (from ref. 8) dramatically illustrates the results of a CF6-50C engine ground test of shroud segments of a newly developed General Electric and NASA seal material (Genaseal) and currently used materials. The Genaseal shroud segments (porous NiCrAlY alloy optimized for resistance to the engine environment) are numbered 11 and 14. The formerly used shroud segments are numbered 9, 10, 12, and 13. All were subjected to 1000 engine test cycles, each equivalent to a typical takeoff, cruise, and landing condition. The superiority of the Genaseal shroud segments is evident. These results indicate that marked improvement is possible in this critical engine performance area by alloy development designed to achieve improved resistance to severe engine environments.

Another recent, and promising, approach is the addition of abrasive particles to the blade tip. The abrasive permits the blades to abrade the outer shroud, while blade wear, which would otherwise result from such blade-tip to shroud contact, is reduced. Continued efforts in this area should be pursued to further enhance helicopter turbine-engine performance.

In addition to the hot section of the engine (obviously subjected to the most severe environment), extensive research is also needed to provide improved tip seals for the lower temperature engine sections. These include the high-pressure compressor stages, as well as stages of the low-pressure or output turbines. A variety of candidate shroud seal materials are being screened for abradability, chemical stability, and erosion resistance. Some promising materials are porous, high-temperature alloys and superalloy foils in the form of honeycombs. Advanced techniques such as plasma spraying of thin layers of superalloys or ceramics also show promise. Due to the importance of preventing engine performance deterioration during operation, as well as increasing durability, continued efforts in the entire area of tip seal technology cannot be overemphasized.

Hot-Section, Low-Stress Applications - Vanes and Combustors

Several key helicopter turbine-engine components must withstand very high temperatures, but the stresses to which they are subjected are relatively low. The most promising materials for two of these applications, vanes and combustors, are discussed in the next sections.

<u>ODS alloys</u>. - For use as turbine stator vanes, the oxide dispersion strengthened (ODS) alloys offer a significant improvement over both currently used and the most advanced conventional cast superalloys (fig. 8). These alloys have use temperatures to 1230° C (2250° F), and they should see service as stator vanes within the next 5 years. One of the more promising is MA 754 (a NiCr alloy with Y_2O_3). Relatively low-cost manufacturing processes (ref. 9) have been demonstrated for ODS alloys. The aluminum-containing versions of such alloys have not been as strong and environmentally resistant as was originally expected (refs. 1 and 10).

A major advantage of ODS materials is their high melting point, about 1370° C (2500° F). At the same time, they have good microstructural stability, providing another plus over conventionally cast superalloys for vane use. The advantage of greater overtemperature capability may be seen in figure 9, in which a conventionally cast MAR M-509 and a TD-NiCr vane are compared (ref. 11). These vanes were subjected to overtemperature while being tested in the same nozzle assembly of an experimental engine. The cast vanes, although cooled, melted. The uncooled TD-NiCr vanes were essentially undamaged. Another advantage of ODS alloys is their greater thermal fatigue resistance. Some ODS alloys have shown up to 10 times greater resistance to cracking under simulated vane operating conditions than conventionally cast superalloys (ref. 11), and there is evidence that structural orientation plays a major role in the degree of thermal fatigue resistance obtainable with ODS materials.

Combustors also fall into the category of high-temperature, low-stress applications where ODS alloys may be used. Combustor components require formable, weldable, as well as oxidation-, thermal fatigue-, and distortion-resistant, sheet alloys. For use temperatures near 980° C (1800° F), conventionally wrought alloys, such as Haynes Alloy 188 and Inconel 617, show good potential (refs. 12 and 13). However, the advanced ODS alloys have a potential for a 90° C (160° F) higher use temperature than conventional sheet materials. ODS alloys under consideration or currently in the development phase for combustor applications include the nickel-base alloy HDA 8077 and the iron-base alloy MA 956. Manufacture of an ODS-NiCr sheet product has been demonstrated, and there now are commercial sources for ODS sheet products. In fact, high alloy cost has been associated with ODS materials (about five times that of conventional superalloys). However, improved powder manufacturing techniques, such as mechanical alloying (ref. 14), and current activities to scale-up for large production quantities promise substantial reductions in the cost of these materials.

<u>Directionally solidified superalloys</u>. - The promising nature of the ODS alloys notwithstanding, another approach, one that employs directionally solidified superalloys for vanes, also shows considerable potential. As shown in figure 10, significantly improved thermal fatigue resistance can be achieved by directional

solidification of superalloys. These results (taken from ref. 15) show an order of magnitude increase in the number of cycles to the first observable crack for a typical, high-temperature, γ'-strengthened, directionally solidified, nickel-base superalloy (Mar-M 247) as compared with its random polycrystalline form. This burner heating and forced-air cooling study confirmed results obtained previously with a wide variety of superalloys in fluidized bed tests. The clearly excellent thermal-fatigue resistance of directionally solidified superalloys suggests that such materials afford great potential for relatively early application to stator vanes for helicopter as well as large turbofan engines.

Ceramics. - As shown in figure 8, ceramics offer the highest use-temperature potential of all materials for stator vanes, on the order of 1400° C (2600° F).

Currently, the most promising ceramics appear to be Si_3N_4 and SiC. Extensive screening studies of some 35 ceramics in the NASA Lewis Mach 1 burner (refs. 16 to 18) have shown them to have the most favorable thermal shock resistance. Other investigations (refs. 19 to 22) have also shown the superiority of Si_3N_4 and SiC. These ceramics also have excellent high-temperature stress-rupture properties as may be seen from figure 11. Commercial hot-pressed Si_3N_4 (ref. 23) has a substantially higher use temperature (approximately 1320° C (2400° F)) at typical vane stresses of approximately 50 MPa (7 ksi), than the strongest known conventionally cast vane alloy, WAZ-16 (ref. 24). It also has about a 100° C (180° F) higher use temperature than the ODS alloys.

Both Si_3N_4 and SiC are under continuing development and improvements in high-temperature strength are being achieved as was shown in figure 11. Work is continuing under NASA sponsorship (ref. 25) to improve stress-rupture strength by using only high purity Si_3N_4 powder (impurities such as alkali metal content less than 400 ppm and O_2 and Fe contents less than 1 and 0.18 percent, respectively). The intent is to limit the amount of strength-reducing second-phase formation at the grain boundaries which results from the reaction of such impurities with densification additives such as MgO and Y_2O_3. Further strength improvements are possible with these ceramics by improving processing procedures such as powder handling, sintering, and hot pressing. Although limited creep-rupture data are available, a promising Si_3N_4 base material is the class known as Sialons (ref. 26). These are solid solutions of Al_2O_3 in Si_3N_4. Demonstrated advantages are that the Sialons do not require hot pressing to reach high density and that they appear to have attractive strength, oxidation, and thermal shock characteristics (ref. 27).

It should be noted that ceramics for turbine application have become the center of much effort in recent years. For example, the U.S. Energy Research and Development Administration (ERDA) program (ref. 28), which began in July 1971, is intended to demonstrate that ceramics can be applied successfully as stator vanes, turbine blades, disks, combustors, and other components in a 1370° C (2500° F) inlet gas temperature, automotive turbine powerplant. One phase of this program has already demonstrated the effectiveness of ceramic first-stage stator vanes for a 30-megawatt ground power turbine installation (ref. 29). It is expected that the vehicular unit combustors, ducts, and stators will meet the goal of 200 hours of operation at the maximum turbine-inlet temperature of 1370° C (2500° F). Major effort is directed at the most demanding application of ceramics, the automotive turbine rotor.

Although the ERDA program is intended to demonstrate the successful application of ceramics in automotive turbines, much of the experience will be of great value in the development of ceramics for helicopter gas-turbine applications as well. Typically, component sizes of helicopter engines are not far different from those of the automotive gas turbine, and the transition to helicopter engines should be much more straightforward than to large turbojets and turbofans. At the same time, a qualitative assessment indicates that the smaller sizes lend themselves far better to uses of ceramics. This assessment is based on the belief that the processing and parameter control will be easier to achieve in the small sizes. This aspect will be covered in some depth at the forthcoming Specialists's Meeting of AGARD (jointly held by the Structures and Materials Panel and the Propulsion and Energetics Panel) in October 1979 at Cologne, West Germany (Porz Wahn, DFVLR).

ROTATING COMPONENTS

In this section, three of the major rotating elements of helicopter gas-turbine engines will be considered - compressor blades, disks, and turbine blades. Each of these components can be significantly improved by the introduction of advanced technology.

Compressor Blades

An area of considerable interest and potential involves the use of composites for compressor blades. Thus far, much of the work dealing with composites for such applications has been directed to their use as large fan blades in high-bypass turbofan engines (ref. 30). The major problem to be overcome for this application is that of large-foreign-object impact damage, such as that caused by birds and ice balls. Composite compressor blades for helicopter engines involve much smaller blade configurations. A major concern when monolithic metal blades are used is fatigue failure due to induced vibrations. The greater stiffness and strength of composites as compared with metals (described more fully in a previous section) could permit significant improvements in this regard as well as allow greater flexibility in aerodynamic design so as to achieve better engine performance. Because many helicopter engines use an inlet particle separator or some type of filtration device, large-foreign-object impact may not be a significant problem. However, it is clear that the engine must be protected from the effects of erosion by sand and other smaller particles passing through the inlet. Here, the use of composites may be appropriate when used with thin titanium sheaths bonded around the individual blades. The low density of graphite/epoxy composite systems for example (approximately 1.66 g/cm^3 (0.06 lb/in^3)) compared with monolithic titanium alloys (4.71 g/cm^3 (0.17 lb/in^3)) can add a further significant benefit. Despite the addition of the denser titanium sheath for erosion protection, a weight saving would result. The saving in blade weight can, of course, be escalated to still greater benefits due to potential reductions in the supporting disk weights.

Disks

Prealloyed powder processing (refs. 31 to 33) holds promise for providing superalloys with increased strength for turbine disk applications. Although current powder metallurgy (PM) disk alloy development is concerned principally with existing alloys, such as René 95 (ref. 34), IN-100 (ref. 35) and Astroloy, work is underway with more advanced alloys which show significant improvements in strength (fig. 12). Even further strength increases are anticipated over the next decade. For example, for PM processing, alloys can be designed specifically to accommodate larger quantities of strengtheners without encountering the segregation which occurs when they are cast. In this way, the feature of greater structural homogeneity resulting from the prealloyed powder process can be used to considerable advantage.

Figure 12 shows the 650° C (1200° F) yield strengths of promising candidate PM disk alloys such as AF 115 (ref. 36) and IIB-7 (ref. 37). These alloys can be processed to have 650° C (1200° F) yield strengths in excess of 1380 MPa (200 ksi).

It is appropriate to consider some very recent results obtained by one of the authors that further illustrate the potential strength capability of PM disk alloys. By careful processing procedures involving PM techniques (ref. 38) extremely high tensile strengths (up to 2865 MPa (415 ksi)) were obtained at 480° C (900° F), the normal disk hub temperature. Figure 13 shows a comparison of strength and ductility of an alloy so processed, the nickel base alloy NASA-TRW VI-A, with conventionally processed disk alloys. No significant reduction in ductility occurred despite the high tensile strengths obtained.

Decreased life cycle cost represents one of the highest priority objectives for improving gas-turbine disks. The two most important factors in achieving this objective are reductions in initial cost by using prealloyed powder processing and improvements in cyclic life. Figure 14 compares the prealloyed powder process schematically with the conventional casting and forging process. It is apparent that less starting material (approximately 45 percent), fewer operational steps, and less machining (approximately 50 percent cost reduction) are required to reach the final disk shape. Of the process variations, the opportunity for greatest cost reduction exists for the hot isostatically pressed (HIP) to near net shape method, with a smaller cost reduction possible for either the extruded or HIP plus warm work approaches.

Although much stronger disk alloys are possible through PM processing, further research is required to assure a commensurate improvement in cyclic life. There are two aspects of this problem.

One is illustrated in figure 15, which shows the 650° C (1200° F) fatigue crack propagation behavior of several disk alloys (ref. 39). Fatigue crack propagation rates increase among the alloys from conventionally processed Waspaloy to PM NASA IIB-7. This increase is in the same order of increasing cyclic strength and of increasing life-to-crack initiation as observed in 650° C (1200° F) strain-controlled, fully reversed-strain, fatigue tests conducted at the low strain ranges appropriate to disks. Thus, while the pursuit of higher strength has produced a greater resistance to crack initiation, it also appears to lead to poorer resistance to crack propagation. Obviously, this trend cannot be ignored and in the future, crack growth behavior must be considered early in disk alloy development programs.

The second aspect of the problem with PM disk alloys currently receiving much attention is concern over defects. Inclusions such as oxides are not removed as in a cast ingot where they tend to float to the top. Also, there is the possibility of porosity in PM materials produced by the entrapment of the argon used during atomization in powder manufacture. Such defects may greatly affect fatigue behavior. While average fatigue properties of PM alloys are improved, it appears that their variance is also greater. Indeed, the minimum fatigue properties of the advanced PM alloys may show relatively little improvement over those of conventional alloys. Considerable effort is expected to be devoted to reducing the defects present in ingots used for conversion to powder as well as those introduced in the powder making process.

Hot-Section, High-Stress Application - Turbine Blades

For helicopter-engine high-stress applications, such as turbine blades, substantial increases in use temperature can be expected by using directional structures of several types (fig. 16). Initial use-temperature increases were made with conventional nickel-base alloys such as directionally solidified (DS) Mar M-200 (ref. 40), in which grains were aligned parallel to the direction of the major stress axis. Further increases can be expected from monocrystals (ref. 41). A summary of these two approaches is provided in reference 42. The directionality of structure is also the key to the ODS + γ' systems and tungsten-fiber superalloy composites.

It should be noted that only a general, qualitative use-temperature capability is identified for the various types of directionally solidified systems shown in figure 16. Recent work with DS superalloys and monocrystals indicates that some of these materials can attain use temperatures on the order of 85° C (150° F) over conventionally cast superalloys.

Directionally solidified (DS) eutectics. - Major research and development effort has been expended in industry and government to exploit eutectic systems. Figure 17 provides a comparison of the 1000-hour stress-rupture properties (refs. 43 and 44) of the major eutectic systems currently under study and hafnium-modified DS Mar M-200. At turbine-blade operating conditions, the DS eutectics now afford about a 30° to 80° C (50° to 150° F) use-temperature advantage.

Figure 18 shows the microstructure of the two major types of eutectic systems, a typical rod- or fiber-reinforced system (HAFCO) and a typical lamellar reinforced system ($\gamma/\gamma'-\delta$). The matrix has been etched away to bring into relief the two types of reinforcing phases. In both systems a relatively ductile matrix is reinforced by a brittle phase. Most rod or fiber (not all are perfect rods as may be seen from the figure) reinforced systems use carbide fibers (HfC in HAFCO and TaC in both CoTaC and NiTaC) ranging from 5 to 15 volume percent (ref. 45). The matrices are generally complex. For example, the matrix of the NiTaC systems consists of a γ' precipitate within a γ-nickel solid solution containing Cr and Al to provide oxidation resistance and precipitation strengthening. Additions of W or Mo provide additional solid-solution strengthening. The $\gamma/\gamma'-\delta$ alloy Ni-20Cb-6Cr-2.5Al is typical of the lamellar-reinforced systems. The δ (Ni_3Cb) lamellae make up approximately 40 volume percent, thus contributing to very low transverse ductility at low and intermediate temperatures, as will be discussed subsequently. The matrix consists of a γ' precipitate within a complex nickel solid solution.

A number of problem areas must be resolved before eutectic systems can be utilized effectively. Foremost and common to all systems, is the slow growth rate DS eutectics require in their manufacture, typically less than 3 cm/hr (refs. 44 and 46). Work is being directed toward achieving more acceptable rates from a blade fabrication standpoint.

Another problem with some eutectics is that of thermal instability during thermal cycling. A visual example is given for the CoTaC system in figure 19 (ref. 47). It may be seen that by suitable compositional changes such instabilities can be overcome. The thermal instability demonstrated (fig. 19(a)) during cycling 2000 times between 425° and 1100° C (797° and 2012° F) could have been caused by a number of factors, whose individual roles are not, as yet, well defined. Possible factors are thermal coefficient of expansion mismatch between fibers and matrix, fiber solubility in matrix, fiber surface energy of formation, and imperfections of reinforcing fiber phase (refs. 45, 47, and 48). Figure 19(b) shows that by substituting HfC for TaC fibers instability during thermal cycling was totally eliminated (ref. 49). A better understanding of the importance of the individual factors that can contribute to such thermal instability must be obtained, but it is apparent that the problem is not insurmountable.

A further area of concern with some eutectic systems is that of low transverse ductility at intermediate temperatures, thus posing a problem with blade fatigue. Another problem is how to adequately design the blade roots because of the need to accommodate the normally imposed shear and bending stresses. The solution might require modified (lower stress) root designs or conventional superalloy bonded roots for advanced "high work" blades (ref. 50). Figure 20 shows the transverse tensile fracture strain for representative eutectic systems, compared with a currently used superalloy, hafnium-modified DS Mar M-200. The rod- or fiber-reinforced systems (NiTaC and CoTaC (refs. 44 and 51)) have reasonably good room- and intermediate-temperature (760° C (1400° F)) transverse ductility, which compared well with hafnium modified DS Mar M-200 (ref. 46). However, the lamellar $\gamma/\gamma'-\delta$ alloy system has relatively low transverse ductility at these temperatures. So far, attempts to improve the transverse ductility have achieved only marginal results; therefore, this alloy system is likely to see only limited application as an aircraft turbine engine blade material.

A recent development is the $\gamma/\gamma'-\alpha$ eutectic alloy. The α phase consists of molybdenum fibers that reinforce the γ/γ' matrix. Recent results indicate that this alloy system has about a 35° C (60° F) higher use temperature for 1000-hr stress-to-rupture life than the $\gamma/\gamma'-\delta$ alloy, together with significantly improved transverse ductility and tensile shear strength. In this case, however, creep shear strength appears to be marginal (refs. 52 to 55).

In summary, the DS eutectic systems have considerable potential, but are characterized by problems that are difficult to solve. Although these alloys possess some attractive properties, they do not appear to be as promising as was previously believed (ref. 1) for near-term applications in high-stress turbine blading. It is probable that in the next 10 years DS eutectics will see only limited service as turbine blades in turbine engines for helicopters or other aircraft.

Composites. - Of all the directional metallic systems (fig. 16), tungsten fiber reinforced superalloys afford the highest use-temperature potential for turbine blades. Considerable research is underway (refs. 56 to 62) to study materials that combine the high-temperature strength of tungsten wires with the ductility, toughness, and oxidation resistance of a superalloy matrix. Potential stress-rupture strength values of over 345 MPa (50 000 psi) for 1000-hr rupture at 1090° C (2000° F) have been reported (ref. 63). The significant strength advantage of tungsten-fiber-reinforced superalloys over DS eutectics (up to $2\frac{1}{2}$ times) and current superalloys (up to five times) is apparent from figure 21, which shows a comparison of their 1090° C (2000° F) 1000-hr density normalized data. The composite data (solid lines) were obtained with a uniform reinforcement of 70-volume-percent fibers along the length of the specimens. For an average 30-volume-percent reinforcement with varying fiber content along the blade span, calculated results (dashed lines) are shown which take into account potential composite degradation due to fiber-matrix interaction over the 1000-hr exposure. To make the 30 volume percent of reinforcing fibers a viable concept, the fibers must be suitably distributed along the blade span to accommodate spanwise variation in blade centrifugal force. A 30-volume-percent reinforcement would result in composite blades of approximately the same weight as current superalloy blades. Such weight equivalence would be accomplished by taking advantage of the greater stiffness and strength of the composite to reduce blade thickness and taper. Excellent high-temperature impact strengths also have been obtained with these composites (ref. 64). Notched Charpy impact strengths of 39 J (29 ft·lb) have been obtained at 1090° C (2000° F). Tungsten-fiber-reinforced superalloy composites also have excellent thermal conductivity as contrasted to superalloys without reinforcement (ref. 65). It has been calculated that it is possible to increase the permissible inlet gas temperature for cooled tungsten-fiber-reinforced FeCrAlY composite blades over that for conventional superalloy cooled blades and thus improve engine performance (ref. 66). Of course, the fabrication difficulties associated with suitably effective cooling configurations within the small composite helicopter blades would have to be overcome to fully realize this advantage.

To date, the combination of properties demonstrated for the tungsten - 1 percent ThO_2/FeCrAlY composite make it attractive for first-generation composite turbine blades. This material has a high melting point, excellent oxidation and corrosion resistance, and only limited fiber-matrix reaction at proposed operating temperatures. Fortunately, the W-1 percent ThO_2 wire used as a reinforcement is commercially available at relatively low cost. It is currently used as lamp filament wire. A significantly stronger but higher cost material is tungsten-hafnium-carbon (W-Hf-C) wire (ref. 67). Additional work is needed, however, to establish more efficient manufacturing methods for the W-Hf-C wire before it can become cost effective. The thermal fatigue resistance of the W-1 percent ThO_2 reinforced FeCrAlY (ref. 68) was shown to be excellent, the material having withstood 1000 cycles between room temperature and 1090° C (2000° F) with no detectable degradation.

Successful solid and hollow turbine-blade fabrication using W-1 percent ThO_2 reinforced FeCrAlY has been demonstrated (ref. 69). Diffusion bonding of monolayer composite plies was selected as the most promising, cost effective method of fabrication for the blades. The composite plies consisted of aligned fibers sandwiched between layers of FeCrAlY. This approach has the capability for achieving the desired accurate fiber distribution and alignment and also limits fiber-matrix reaction. Blade shapes can be produced that are close enough to final dimensions so that only root machining and touchup grinding of the airfoil are required to attain final shape. The concept used for fabricating a composite turbine blade is shown in figure 22. Briefly, the number and shape of monolayer plies needed to produce an airfoil are cut from monolayer sheets. The plies are then pressed around a leachable steel core. The core is leached out with acid after pressing. A superalloy root attachment is then bonded onto the composite airfoil. An end cap can then be bonded onto the airfoil. To enhance cooling effectiveness, an impingement cooling insert can be placed in the cavity and bonded to the blade root. Figure 23 shows a hollow composite JT9D airfoil without the root attachment fabricated by this technique.

The blade fabrication sequence described was used as the basis in a cost analysis evaluation (ref. 70). The manufacturing costs of a JT9D turbine blade made of a DS eutectic, an oxide-dispersion-strengthened superalloy, and the tungsten-fiber/FeCrAlY composite were compared with the cost of a production JT9D blade made of a DS superalloy. The analysis indicated that manufacturing costs for the fiber-reinforced composite blade would be competitive with current costs of the production blade.

Oxide-dispersion-strengthened + γ' alloys. - Recent advances in the production of oxide-dispersion-strengthened (ODS) alloys have introduced them as contenders for advanced turbine-blade application. For this use the high strength at intermediate temperatures of a γ'-strengthened alloy (needed at the blade root) is combined with the high-temperature strength derived from the oxide dispersion (needed in the airfoil). At 1090° C (2000° F) the creep-rupture life of experimental ODS superalloys, such as ODS WAZ-D (ref. 71) and the very recently developed MA-6000E (ref. 72), compare favorably with conventionally cast, random-polycrystalline-strengthened alloys and DS eutectic alloys. As shown in figure 24 these alloys have use temperatures in the range 1150° C (2100° F) for a 1000-hr life at a stress of 103 MPa (15 ksi). MA-6000E, developed by the International Nickel Co., has demonstrated improved rupture ductility (3 percent) compared with previous ODS superalloys (1 percent for ODS WAZ-D); it has also demonstrated good oxidation and excellent corrosion resistance in comparative tests with conventional superalloys. An added advantage of the ODS alloys is their higher incipient melting temperature, which is derived from the compositional uniformity inherent in processing by powder metallurgical techniques.

Although ODS superalloys have attracted considerable interest from manufacturers of gas-turbine engines, a great deal of work in several areas, including coatings, forging, and texture control, must be performed before they can be applied to helicopter turbine engine blades.

Ceramics. - The ultimate in potential use-temperature for turbine-blade applications still resides in ceramic materials. As shown in figure 16 use temperatures of 1200° to 1370° C (2200° to 2500° F) can be expected. In addition to the potentially low cost of ceramics (about 1/10 that of superalloys), their low density (about 1/3 that of superalloys), together with their high strength-to-density ratios, make ceramics particularly desirable for rotating turbine blades where the primary stresses result from centrifugal forces. However, it is not reasonable to expect the characteristics of essentially no ductility and very low impact resistance to be circumvented to the extent necessary to permit the use of ceramics as helicopter engine turbine blades before the last decade of this century. There is a much greater likelihood that ceramic turbine blades will see service in ground power installations or automobiles considerably before this.

Although it is not feasible to produce ductile ceramics, they can be made more impact resistant. Modest, although not consistently obtainable improvements in impact resistance have been obtained to date. Figure 25 shows in chronological order the increase in both room-temperature and 1315° C (2400° F) impact strength achieved with Si_3N_4 by various investigators. The approaches included increasing the purity of α-phase Si_3N_4 powders (ref. 73), applying a lithium aluminum silicate energy-absorbing surface layer (ref. 74), and carburizing to create a compressive layer (ref. 75). The last mentioned approach resulted in a 1-J (10-in·lb) impact strength, a factor of 10 increase over as-received Si_3N_4.

More impressive improvements in impact strength of Si_3N_4 have resulted from reinforcing hot-pressed Si_3N_4 with 25-volume-percent Ta wire of 0.06-cm (25-mil) diameter (refs. 76 to 78). Drawbacks to this Si_3N_4-Ta material are its more than doubled density over monolithic Si_3N_4 (6.55 g/cm^3 versus 3.2 g/cm^3) and the possibility of catastrophic oxidation of any exposed Ta when the material is placed in service. Most recently, the use of an energy-absorbing surface layer has given encouraging results (ref. 79). In preliminary work porous, reaction-sintered silicon nitride layers on dense Si_3N_4 have provided ballistic impact strengths of 14 J (10 ft·lb) at both room temperature and 1370° C (2500° F). Without this protection, the ballistic impact strengths of the Si_3N_4 at room temperature and 1370° C (2500° F) were only 2 and 2.9 J (1.4 and 2.1 ft·lb), respectively). The impacting particles were approximately 4-mm, hardened steel balls having impact velocities from 100 to 130 m/sec (350 to 425 ft/sec). Further improvements in impact resistance, particularly as regards reproducibility, are anticipated in current work, although much higher values of impact strength are not likely.

To use ceramics for turbine blades, the designer must deal with materials of essentially no ductility. Very early NACA (National Advisory Committee on Aeronautics) work recognized the need to accommodate this lack of ductility by designs employing cushioning interfaces between blades and disk and generous radii at the blade roots (refs. 80 and 81). Figure 26 illustrates some early, partially successful attempts in this area (ref. 81). The cushioning of the interface materials also acted to distribute the stress in the root attachments. Ductile, porous, or screen material interfaces were especially beneficial. Interestingly, as in the early NACA work, recent investigations (refs. 82 and 83) have shown that the use of ductile interlayer materials, such as platinum, is highly beneficial. Such materials distribute the stresses and prevent chemical reaction between the ceramic blades and the metal disks. These methods were used to construct full-scale (10-cm (4-in.) span) blades that, 20 years ago, were operated for over 240 hr at continuous, full-engine power without root failures. This was accomplished at a time when these engines were qualified for military service with a 150-hr test. The brittle airfoils could not withstand the impact of typical foreign objects, however.

Of course, separately mounting blades on a disk is not the only viable approach for helicopter applications. Another involves separately fabricated ceramic hubs and blade rings that may then be joined, for example, by diffusion bonding. Still another approach is injection molding. Here, a ceramic powder is mixed with a plasticizer and injected at a low temperature (approximately 200° C (400° F)) into a closed die of the hub and blade combination. The die is removed, the green body is heated to burn out the plasticizer, and finally the body is sintered at high temperature in an inert atmosphere.

The early attempts to use ceramics and cermets as turbine blades were handicapped by weak materials, immature processing procedures, and crude stress-analysis techniques. Fortunately, designers today have far superior materials (SiC and Si_3N_4) and design tools. Today's tools are based on fracture mechanics and sophisticated three-dimensional, finite-element stress analyses, made possible by the advent of large computers. The three-dimensional, finite-element stress analysis permits the determination of the complete stress distribution throughout the component so that the designer can pinpoint local stress conditions that may result from stress concentrations.

Fracture mechanics can be used to determine the crack-growth characteristics of the materials under design loads. Since the strength of ceramic materials is determined by initial flaw size and distribution and since there is essentially no ductility to arrest crack growth, the careful application of fracture mechanics is even more important than in the design of highly stressed metallic systems. The ceramic materials must be characterized by determining the sustained-load, subcritical crack growth and the cyclic-load crack growth. Fortunately, a good basis for this work has been established by the U.S. National Bureau of Standards and others (refs. 84 to 86) working with glass, Al_2O_3, SiC, and Si_3N_4. Note, however, that direct, quantitative comparisons of the ceramic fracture toughness data obtained to date with metal data can be misleading because of differences in the fracture toughness specimen sizes involved. Relatively thick (on the order of several centimeters) metal specimens are used, whereas ceramic specimen thicknesses are typically about 0.8 mm (30 mils). A rough indication of the relative fracture toughness of these materials is that heat-treated steel has 30 to 50 times the fracture toughness of hot-pressed Si_3N_4. Research is being conducted at the NASA Lewis Research Center to standardize testing methods for determining the fracture toughness of ceramics.

Despite the difficulties and limitations, however, the tools available to the designer today make designing around the ceramic ductility problem appreciably more likely than in the past. Nevertheless, the challenge is to learn how to use these materials effectively. Their extreme brittleness makes them far more sensitive to internal flaws (inherent in their manufacture) and to surface flaws resulting from accidental damage than any materials used heretofore in critical, high-stress hot-section helicopter gas-turbine blade applications.

ENVIRONMENTAL PROTECTION

All of the metallic systems discussed for helicopter gas-turbine stator vanes and blades require surface protection to realize their high use-temperature potential for long service. The turbine combustion gases pose problems of oxidation and hot corrosion which vary in severity with the type of fuel and the atmospheric environment. In addition, the turbine's cyclic operation causes thermal fatigue cracking. Considerable progress has been made in developing conventional coatings and, in particular, thermal barrier coatings to provide good resistance to these failure mechanisms. In addition, research on fuel additives is also underway to reduce the harmful effects of impurities; however, greater effort is needed in view of the requirements for petroleum conservation which dictate eventual use of "dirty" fuels. Tolerance limits for materials and coatings must be established, and improved coating protection for helicopter turbine blades and vanes must be provided so that these fuels can be used without lowering turbine operating temperatures. Figure 27 is a summary of fuel costs as well as the impurity contents that contribute to hot corrosion and erosion in various grades of fuel. Associated with desirable reductions in fuel cost are impurity content increases of orders of magnitude which contribute to increased hot corrosion and erosion.

Metal Coatings

Recently, coating development efforts have focused on electron-beam, physical vapor deposited (EB-PVD) MCrAlY (M = Ni, Co, or Fe) coatings, noble-metal-modified PVD MCrAlY coatings (ref. 87) and aluminides (ref. 88), and duplex aluminide coatings. The latter are formed by deposition of an MCrAlX (X = Si, or Y, etc.) modifier layer before pack cementation (ref. 89). The above types of coatings provide good protection for conventionally cast and DS superalloys and DS eutectics. Because the overlay coating method is not dependent on diffusion with the substrate, it affords the opportunity for applying a wide variety of coating compositions. Because of this versatility, the method has great appeal to the coating designer. However, the high cost of the PVD method (currently the most common method of applying overlay coatings) makes development of alternative, low-cost overlay coating processes attractive. Recent advances in plasma-spray technology appear to offer a potential low-cost alternative.

Because periodic refurbishment of the coatings discussed above is presently required, more durable metallic coatings must be found for the present generation of engine alloys. For the future, specific coatings must be developed for the advanced higher temperature alloys such as the ODS + γ' alloys and monocrystals.

The nature of the problem with ODS alloys with the overlay coatings, is illustrated in figure 28 (ref. 90). Certain overlay coatings on TD-Ni-CrAl exhibit large weight losses after about 1000 hr of Mach 1 burner-rig cyclic exposure between

1100° C (2012° F) and room temperature. As illustrated by figure 29, the overlay coating can be severely degraded after as little as 500 hr, and Kirkendall porosity is evident at the coating and substrate interface. Diffusion barriers may be one procedure for slowing down the interdiffusion reaction and reducing interface porosity (ref. 91).

As metal and/or cooling-air temperatures are raised, the need for coating the increasingly complex internal cooling passages of turbine airfoils arises. This problem is, of course, still more severe in small helicopter engine blades. Present internal coating efforts are directed toward the application of economical aluminized MCrAlX coatings.

Thermal Barrier Coatings

The concept of an insulating coating on cooled components of gas turbines has always offered an attractive means of improving performance by allowing higher operating temperatures or by reducing cooling-air consumption (ref. 92). It can also improve component durability by reducing the substrate metal temperature. The concept of thermal barriers is illustrated in figure 30 (from ref. 93). The benefits result from the large temperature drop ΔT that occurs across the insulating oxide layer. The ΔT range, from 50° C to more than 200° C (90° to 360° F) for a 0.25-mm (0.010-in.) thick ceramic layer (zirconia), depends on the specific application and pressure ratio of the engine. For example, the calculated benefits of a 0.18-mm (0.007-in.) zirconia coating applied to the first-stage blades of a JT9D engine are listed in figure 31 for sea level take-off conditions (ref. 94). Similar metal temperature reductions were calculated for a 0.25-mm (10-mil) zirconia coating on the first-stage blades of an F100 engine (ref. 95). Here it was determined that cooling air to the blades could be reduced by 1.2 percent of the engine airflow. This is equivalent to about a 0.3-percent improvement in thrust specific fuel consumption (TSFC) Alternatively, in both engines, durability could be improved by a factor of about 4. It is of interest to note that for ground-power gas turbines benefits in performance resulting from application of thermal barrier coatings are also significant (refs. 96 to 98).

The development of the $ZrO_2 \cdot 12\ Y_2O_3$ ceramic/Ni-16 Cr-6 Al-0.6 Y (in percent by weight) bond-coat duplex coating system was one of the milestones in thermal barrier coating technology (ref. 99). The microstructure of this coating is shown in figure 32. This coating is more durable than the more conventional $ZrO_2 \cdot CaO$ and $ZrO_2 \cdot MgO$ systems used with nickel aluminide, Nichrome, and even NiCrAlY bond coats. The durability of the $ZrO_2 \cdot 12\ Y_2O_3$/NiCrAlY system is exemplified by the data in figure 33 (from ref. 99).

A second milestone in thermal-barrier coating technology was the succesful completion of an engine test (fig. 34, ref. 100). These two accomplishments indicated that thermal barrier coatings are feasible in turbine airfoil applications and sparked intense interest throughout the gas-turbine engine community (ref. 101).

The $ZrO_2 \cdot 12\ Y_2O_3$/NiCrAlY system was also evaluated on first-stage turbine blades of a JT9D engine for 264 hr (1424 cycles) (ref. 94). Turbine-inlet temperature reached 1430° C (2600° F). After 39 hours most of the zirconia layer had spalled from the central region of the leading edge. After another 225 hours, additional ceramic spalled from part of the pressure surface in the 70 percent span region near the trailing edge. Analysis indicated that these two areas were the hottest regions of the blade. In cooler areas, the coating was intact.

Recently, further significant improvements in the durability of the $ZrO_2 \cdot Y_2O_3$/NiCrAlY system were made (ref. 102) by decreasing the Y_2O_3 level from 12 (which imparts a nearly fully cubic stabilized structure) to 8 weight percent (which results in only partial stabilization). Additional improvement resulted from the reduction of the NiCrAlY yttrium level from 0.6 weight percent to 0.15 to 0.35 weight percent. This change is believed to enhance the stability of the oxide/bond coat interface region. These modifications have more than doubled coating durability (i.e., cycles to first crack) as illustrated in figure 35.

With trace-metal contaminated fuels, the $ZrO_2 \cdot 12\ Y_2O_3$ system has not proven to be nearly as durable as with clean fuels. Performance in contaminated fuels (ref. 103) is a strong function of contaminant level as illustrated in figure 36. The coating is more sensitive to vanadium than to sodium. Thus, in helicopter applications, where clean fuels are presently used but where sea salt may be present in the air, coating durability appears to be less of a problem than in stationary gas turbines that burn heavy fuels. This situation can, of course, change as less clean fuels go into helicopter service. Several coating systems with significantly improved tolerance to Na and V have recently been identified (ref. 103; see fig. 37).

In the JT9D engine test, in burner-rig oxidation and corrosion tests, and in creep, tensile, and compression tests, the $ZrO_2 \cdot 12\ Y_2O_3$/NiCrAlY system has failed in the ceramic layer close to the bond coat. Cohesive strength measurements have confirmed that this region of the coating is the weak link (ref. 104). (See fig. 38.) The interface region has a strength of only 6.2 MPa (930 psi), while the bulk oxide layer and the bond NiCrAlY layer are about four times stronger. Figure 38 also illustrates the steady-state thermal expansion mismatch considerations associated with ceramic coatings. Mismatch strains are greatest in the region where the coating has the least strength.

Although significant progress has been made in thermal barrier coating technology over the past several years, the technology is still immature. In particular, the need to translate the advances into sizes that are applicable to small engines used in helicopters is apparent.

TRENDS IN TURBINE COOLING TECHNOLOGY

Interest in improving helicopter gas-turbine-engine cycle performance by increasing turbine-inlet temperature places a continuing emphasis on the development of more efficient and flexible turbine cooling systems. As inlet gas temperatures increase, a departure from convection-cooled configurations toward more complex airfoil cooling systems involving combinations of impingement and convection cooling and surface film cooling are required (ref. 105). As gas temperature and pressure increase, convection cooling requirements rise sharply. Use of a more sophisticated cooling system, such as film cooling, can substantially decrease the cooling flow required at current gas temperatures and pressures. Furthermore, it can permit operation at much higher gas temperatures and pressures without the need for exorbitant cooling flows. However, problems associated with cooling small turbines such as those used in helicopter engines are far more difficult to solve than those of the larger turbines of high-bypass turbofan engines. Primarily, the physical difficulties involve not having sufficient internal surface cooling area available to incorporate suitably effective cooling configurations into the small turbine blades. Also, the lower aspect ratio (blade span-to-chord), generally characteristic of the smaller helicopter turbine blades, results in more complex, and less understood, secondary-flow mechanisms of the gas stream. As a result, prediction of the local gas environment is less accurate. Also, surface roughness and defects, as well as tip clearances and blade dimensional tolerances have a relatively larger effect on the thermal and aerodynamic performance of small turbines than large turbines. These factors, along with other key performance obstacles associated with small size turbines are listed in figure 39.

It should be noted that for some helicopter applications optimization of the entire helicopter and engine system may dictate the use of radial-flow turbines or a combination of radial-flow and axial-flow turbines instead of the customary axial-flow turbines used in large aircraft engines. The potential for high efficiency in radial turbines is unquestioned and surpasses that of the axial type, but mechanical integrity of the radial type has not been proved at elevated temperatures. At present, there are no cooled radial turbines in aircraft-engine use for primary propulsion. Plans call for advanced versions of radial turbines to operate at gas temperatures that will require cooling, and there is a strong need for a cooling technology base as well as a fabrication technology base for this type of turbine. Also, extensive research in the areas of fluid flow, heat transfer, and fabrication will be needed to achieve the required durability for these efficient turbine stages.

The potential benefit of applying a thermal-barrier coating to a convection-cooled configuration has been described previously. Such benefits are not expected to be as great for the smaller turbine blades of helicopter engines because of the lower cooling effectiveness of the smaller blades. Furthermore, greater aerodynamic penalties (ref. 106) would be associated with the relatively thicker trailing edge resulting from the application of the thermal-barrier coating on the smaller turbine blades. This area of work presently lacks the attention needed to establish the exact payoff that can be expected.

HOT-SECTION LIFE PREDICTION AND DESIGN TECHNIQUES

Life prediction of a helicopter engine component requires that there be available to the designer an adequate analysis procedure, which will permit him to accurately establish its response to the expected service environment. Furthermore, this response must be compared with the capacity of the particular material/structural combination to sustain the damage imposed by the service environment. Both material/structural aspects and service conditions are different for the various key engine components, and thus each component poses a variety of multiple failure modes and complex environment, stress, and strain interactions.

To achieve success in this key area, research is required along a broad front that involves the determination of mechanics of material data in low-cycle fatigue, creep, stress-to-rupture, and the interaction of multiaxial stress fields. The development of three-dimensional time-independent and time-dependent cyclic inelastic stress analysis, and a determination of the interacting effects of the engine service environment will also be required.

Some progress is being made in these areas. This can be illustrated for a key component such as a turbine blade in the context of the interaction effects discussed above as follows: Figure 40 schematically illustrates the major factors affecting turbine-blade life. At low temperatures fatigue is the governing life mechanism; at high temperatures creep is the governing life mechanism. Environmental factors such as oxidation and hot corrosion must also be accounted for by the designer if he is to accurately predict blade life. Of course, superimposed on these effects is the problem of erosion, which can vary in intensity, depending on the aircraft service environment. By developing procedures to account for each of these aspects and coupling these procedures, a reasonable start can be made toward accurate life prediction.

Creep-Fatigue Effects

Recent efforts to account for creep-fatigue interaction (refs. 107 to 109) by means of the so-called strainrange partitioning approach (SRP), show great promise. Briefly, the technique involves combining the two directions (tension or compression) and two types (creep - time dependent or plastic - time independent) of inelastic strain to achieve four possible kinds of strainrange that may be used as the basic building blocks for any conceivable hysteresis loop. These are illustrated in figure 41 and defined as (a) tensile plasticity reversed by compressive plasticity and designated a PP strainrange, (b) tensile creep reversed by compressive plasticity and designated CP strainrange, (c) tensile plasticity reversed by compressive creep and designated PC strainrange, and (d) tensile creep reversed by compressive creep and designated CC strainrange. The four life relationships shown in figure 42 may be obtained experimentally for any material (ref. 107), or they may be calculated from a knowledge of the tensile and creep ductility (ref. 110). Once obtained, the hysteresis loop for the cycle being analyzed must be partitioned into its inelastic components, and a damage rule applied to predict the life associated with the combination of applied strainranges as described in reference 107. A typical example of the ability to predict high-temperature, low-cycle fatigue life by this method for 12 alloys is illustrated in figure 43.

This technique was recently the subject of an Advisory Group for Aerospace Research and Development (AGARD) Structures and Materials Panel Evaluation. A wide variety of materials were considered by 20 laboratories in 6 countries. The culmination of this evaluation was the Specialist's Meeting that was held in Aalborg, Denmark, in April 1978 (ref. 111). While the results varied from laboratory to laboratory, most investigators agreed that the SRP method was a significant step toward life prediction in the presence of high-temperature and cyclic stresses (ref. 112). Several concerns and recommendations emerged regarding SRP. These dealt primarily with the problems associated with the application of SRP to cases involving small inelastic strains (and therefore long lives). The difficulties associated with partitioning these narrow hysteresis loops and the past inability of SRP to handle mean stress effects were also noted. Research is underway at the NASA Lewis Research Center to address these problems.

Oxidation Effects

To fully evaluate oxidation and hot corrosion attack, we must develop reliable life-prediction techniques to account for these mechanisms. Early work in this area (ref. 113) used paralinear oxidation models to predict isothermal oxidation lives for materials forming volatile oxides. This type of analysis was extended to cover oxide spalling losses (ref. 114) and was found to predict isothermal or cyclic oxidation to within a factor of 3 of the measured material-loss values (fig. 44). Studies at the NASA Lewis Research Center of oxide spallation (refs. 115 and 116) are being used as the basis for further modification of the computerized life-prediction techniques (COREST I, II, etc.) to achieve even greater accuracy.

Coating Effects

In addition to the need for further development of coating techniques, there is a major requirement for developing reliable methods to predict coating life. Recent work at NASA (refs. 117 and 118) is directed toward developing a better understanding of the two high-temperature coating degradation mechanisms associated with turbine blade and vane applications. These mechanisms are (1) interdiffusion of key elements from the coating into the metal substrate and (2) oxidation and spallation. Equations have been proposed for a first-approximation, combined-oxidation-attack and diffusion-degradation model for NiCrAl coating systems. Experimental verification studies of actual coatings under long exposures are still required for confirmation. In addition, considerable effort will be required to incorporate the effects of corrosion and erosion on coating life into a comprehensive life prediction model.

It should be noted that once the techniques for predicting coating life have been established, they must be incorporated into the other hot-section-component-life-prediction techniques already described. Once such coupling is accomplished, the engine designer will have an immeasurably better capability for selecting the proper material and coating combination to meet his particular design life goal.

Design Approaches to Reduce Oxidation and Hot Corrosion Effects

Using the cyclic oxidation evaluation and prediction techniques described above (ref. 114), alloys in the Ni-Cr-Al system have been evaluated in cyclic oxidation and hot corrosion (refs. 119 and 120). Data from these tests have been used to make both oxidation and hot-corrosion maps (oxidation map shown in fig. 45) from which optimum compositions then can be easily identified.

Another approach to reducing hot-corrosion effects is to add inhibitors that will reduce or eliminate hot corrosion without a material system change. Efforts to find such an inhibitor are described in reference 121.

Hot corrosion testing of several nickel-, cobalt-, and iron-base alloys was performed in a Mach 0.3 burner rig to determine the effectiveness of potential fuel additives for reducing sodium sulfate attack. All tests were conducted in a cyclic mode (900° C (1650° F) for 1 hr; room temperature for 3 min). After 100 hours at maximum temperature the extent of attack was determined by measuring the maximum metal loss. All corrosion results were compared with cyclic oxidation tests made under the same conditions of time and temperature but in the absence of sodium chloride or additives. While all of the additives tested (salts of Al, Si, Fe, Cr, Zn, Mg, Ca, and Ba) reduced hot corrosion for some alloys, the most consistently effective additive was barium nitrate (fig. 46). For all alloys examined this barium additive generally reduced the corrosion attack to oxidation levels. Although much work is needed to establish cost effectiveness and the extent of a potential fouling problem, these results strongly suggest the desirability of the use of barium to substantially reduce the alkali-metal-induced hot-corrosion problem. This is particularly pertinent in many helicopter service situations involving flights in sea-salt air environments.

OBSERVATIONS CONCERNING LIFE ASSURANCE AND LIFE MANAGEMENT OF HELICOPTER ENGINES

Previous sections of this paper identified materials and structural technological advances already available to the helicopter engine designer in terms of individual engine components. The concepts expressed in this last section of the paper are intended to point out the broader aspects of what is still required to achieve intended design life, not only of individual engine components, but of the entire engine as well. They are of particular significance when considering the engine hot section, where durability is clearly one of the major concerns in high-performance helicopter (as well as large turbofan) engines. The technology for improving engine performance has outpaced the technology for achieving long life in hot-section parts. Therefore, it is of prime importance that research in the areas described be implemented, with particular emphasis placed on the engine hot section.

As described previously, it is necessary to design a component so that it will survive the complex environment imposed by the engine for some minimum cycles or minimum time. As was indicated for the turbine blade, one of the key hot-section components, the environment imposed can cause failure in various ways and combinations thereof, including creep, stress rupture, fatigue, oxidation, hot corrosion, and erosion, not to mention alloy embrittlement due to possible formation of embrittling phases (ref. 122).

Both life assurance (defined as the ability to design hardware that will meet a required life time) and life management (defined as the activities that provide for appropriate inspection for failure initiation or for part removal based on known service exposure) must be developed. These plans are analogous to fracture control plans used for launch vehicles. A key aspect involves recognition and understanding of the potential failure modes. Both life assurance and life management must be considered together because of the commonality of the disciplines involved and their dependence on an understanding of failure mechanisms.

To provide for safe-life design of a typical hot-section engine part, for example, it is necessary to know accurately (1) the local gas temperature, (2) heat transfer to the part, (3) local and average part temperatures, (4) local and average stresses and strains, (5) the surface environment, (6) appropriate material properties in the part, (7) the response of the material to various failure mechanisms induced by the temperature, stresses, strains, and surface environment. Finally, the methodology must be validated in test rigs through the component level, and in engines. To enhance the technology for safe-life design, advances are needed in a variety of disciplines. For example, items (1) to (3) involve the disciplines of aerodynamics, heat transfer, and fluid flow, whereas items (4) to (7) involve various disciplines associated with materials and structures.

In addition to designing each component to attain a desired design life on the basis of the factors already described, it is also necessary to view each component in the broader context of its function as an integral part of the total helicopter structure. Thus, the structural stability of the component when it is put in service becomes a vital part of the design process. In service, components are subject to dynamic structural loadings (i.e., maneuver loads, blade-shaft-disk interactions, unbalance effects, interface compliance, and inertial effects of engine-airframe interaction). All of these structural factors and their interactions must be accounted for, preferably by easily applied computerized design techniques, if the desired component design lives are to be met.

Finally, more effective life management requires that advances be made in nondestructive inspection methods. Advanced methods must be developed that can be used during production to provide greater assurance that the part will survive for a desired lifetime; others, to monitor component operation during engine service to guide component removal and replacement. More appropriate methods and times for engine inspection in service must be found to insure that incipient failures are located before they propagate to catastrophic part or engine failure.

Guidance must be established for part removal based on indications of incipient failure or accumulation of cycles or hours of exposure. (An example of guidance policy here would be that if one blade on a high-pressure turbine stage shows incipient failure, all of the blades from that stage should be replaced.)

It is the intent of the foregoing to emphasize that a broader research approach is needed to achieve improved helicopter engine durability (and for that matter, larger turbofan engine durability). The technology advances in the various disciplines cited, appropriately integrated and incorporated into a comprehensive structurally oriented design approach, would, in the authors' view, add significantly to obtaining solutions in this key area.

SUMMARY

Significant payoffs in terms of turbine engine performance, as well as, initial and operational cost reduction can be achieved by utilizing advances in materials, environmental protection, structural concepts, and life-prediction techniques. These payoffs would be applicable to helicopters as well as fixed-wing aircraft. The degree of improvement varies, depending on the engine component and the aircraft system under consideration. It is influenced by constraints of geometry, fabrication, and of course, the technological state of the art. Although major technological advances have already been made, the technologist still faces formidable challenges before such advances can be applied routinely to helicopter engine systems.

Reviewing the picture as it stands at present, composite technology (including both advanced materials and design methodology) is rapidly nearing the point where composite static structures should be considered for applications such as the helicopter front frame. Advanced temperature composite systems also show promise for reduced weight and cost in rotating components such as compressor blades. For disks, prealloyed powder superalloys afford opportunities for both increased strength and reduced fabrication cost. For low-stressed, hot-section components such as combustors and vanes, ODS alloys have approximately a 100° C (180° F) higher use-temperature potential than conventional superalloys. Ceramics afford the highest use-temperature potential, on the order of 1400° C (2600° F), of all materials for stator vanes, with SiC and Si_3N_4 being particularly promising. For highly stressed, hot-section components such as the turbine blades, directionally structured materials have major advantages over conventionally cast polycrystalline superalloys: Monocrystals, directionally solidified eutectics, and tungsten-fiber-reinforced superalloys are all potential candidates. Recent strides in the latter system suggest that blade use-temperatures on the order of 1090° C (2000° F) are possible. Although the highest use-temperature capability for turbine blades resides with ceramics, with potential use temperatures as high as 1370° C (2500° F), major changes in fabrication techniques and design philosophy must be adopted if these materials are to be used successfully. Thus, fracture mechanics concepts and advanced three-dimensional, finite-element stress analysis will have to be used to deal with these materials of essentially no ductility. Their initial use will probably be limited to relatively low average stress areas.

To benefit fully from the increased use temperature of advanced materials, adequate environmental protection schemes must be developed to overcome the lower oxidation resistance that goes hand in hand with higher metal temperatures. This problem is compounded by eventual use of cheaper, "dirtier" fuels that contribute to hot corrosion and erosion. Substrate temperature reduction in cooled turbine components by means of thermal barrier coatings is promising, although the improvement in the smaller blades of helicopter engines will probably not be so great as for the larger blades of turbofan engines. Similarly, fabrication difficulties with the smaller helicopter engine blades place restraints on the application of some of the more sophisticated internal cooling configurations.

Finally, major strides are being made in the development of more sophisticated analytical methods of life prediction. New methods include such techniques as strainrange partitioning for predicting high temperature, low cycle fatigue life, computerized methods such as COREST II for predicting oxidation resistance, and advanced techniques to account for coating life. Of course, additional research is needed before these methods can be appropriately combined to allow the helicopter turbine engine designer to select the optimum material and structural combination for a particular design goal. Nevertheless, with the methods already in hand, a considerable change in the "build them and break them" design philosophy has been made. Their application can add appreciably to more accurate helicopter-engine component life assessment before service and result in significant operational cost reduction.

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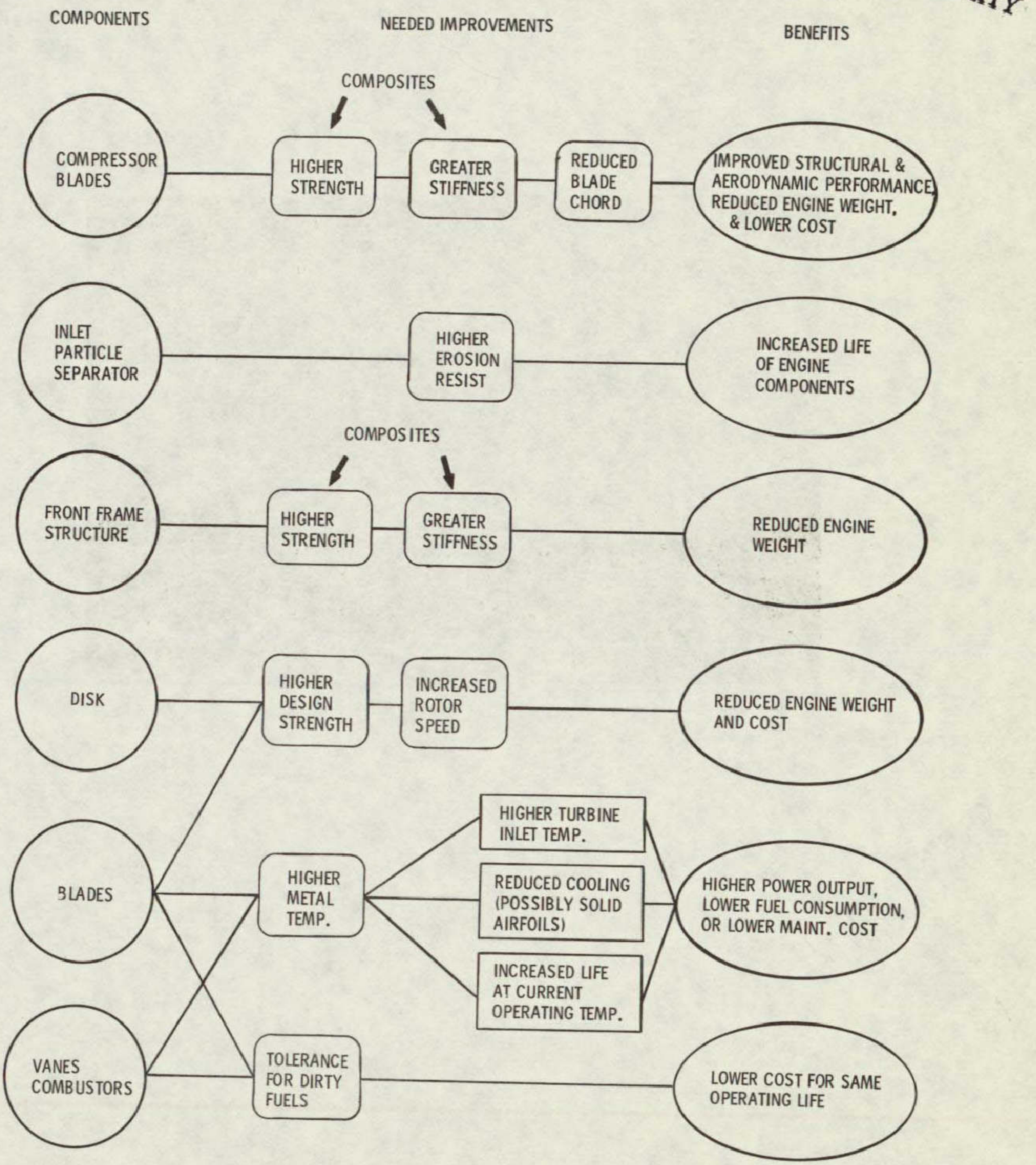


Figure 1. - Helicopter gas-turbine engine payoffs from advanced materials and structural components.

E-9922

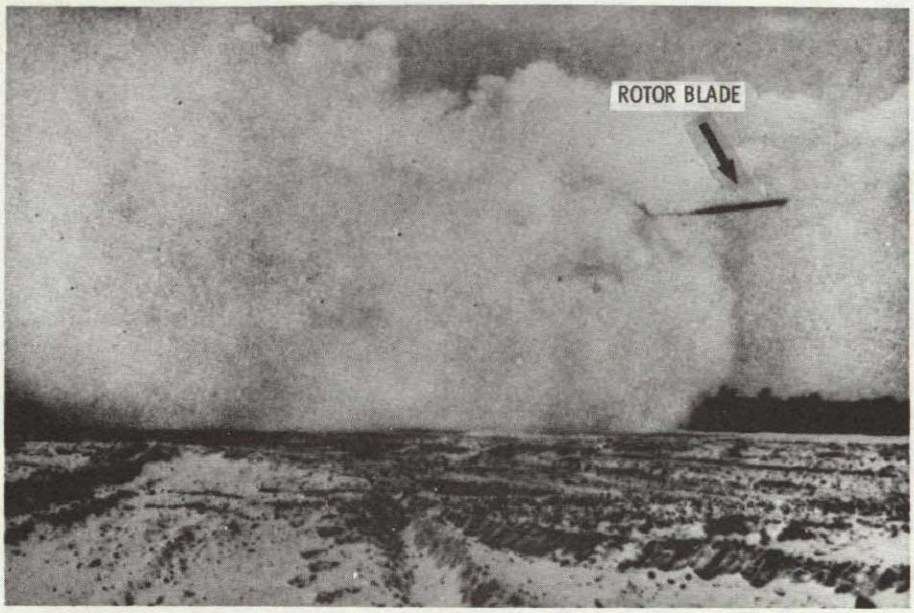
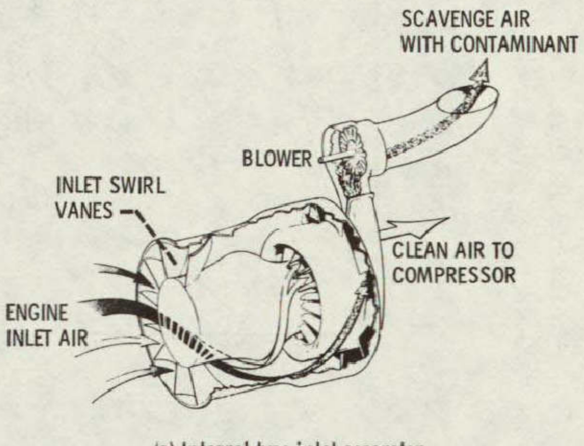
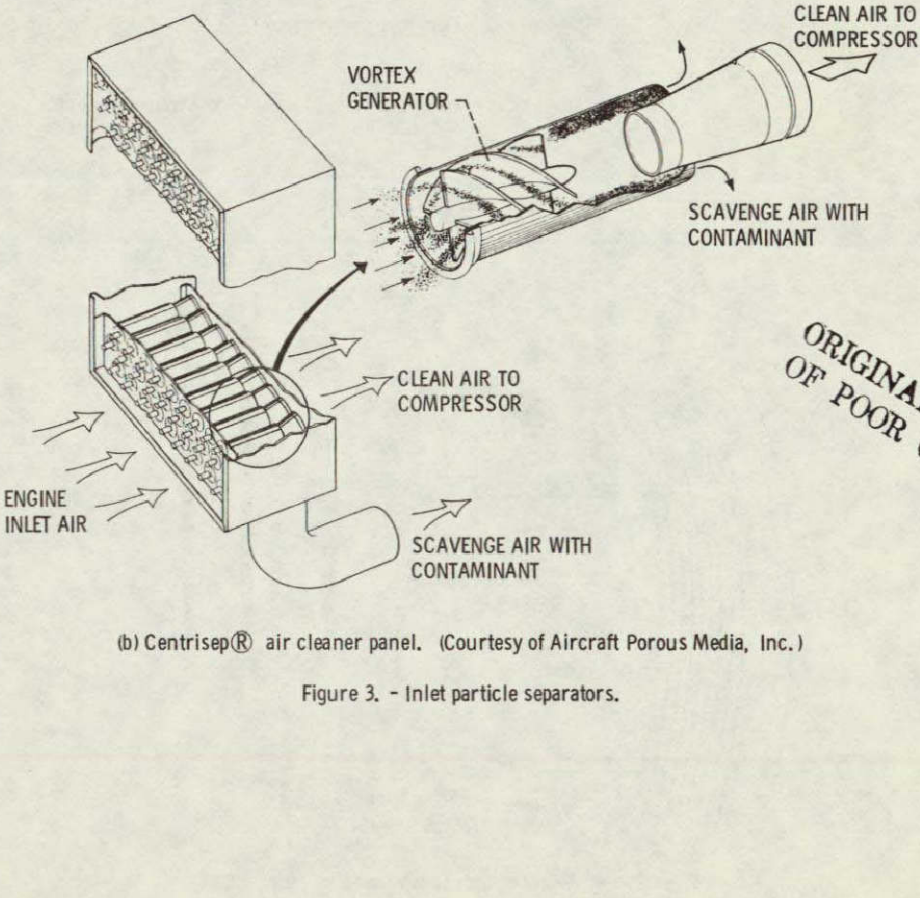


Figure 2. - Helicopter foreign-object-ingestion problem.

E-9924



(a) Integral type inlet separator.



(b) Centrisep® air cleaner panel. (Courtesy of Aircraft Porous Media, Inc.)

Figure 3. - Inlet particle separators.

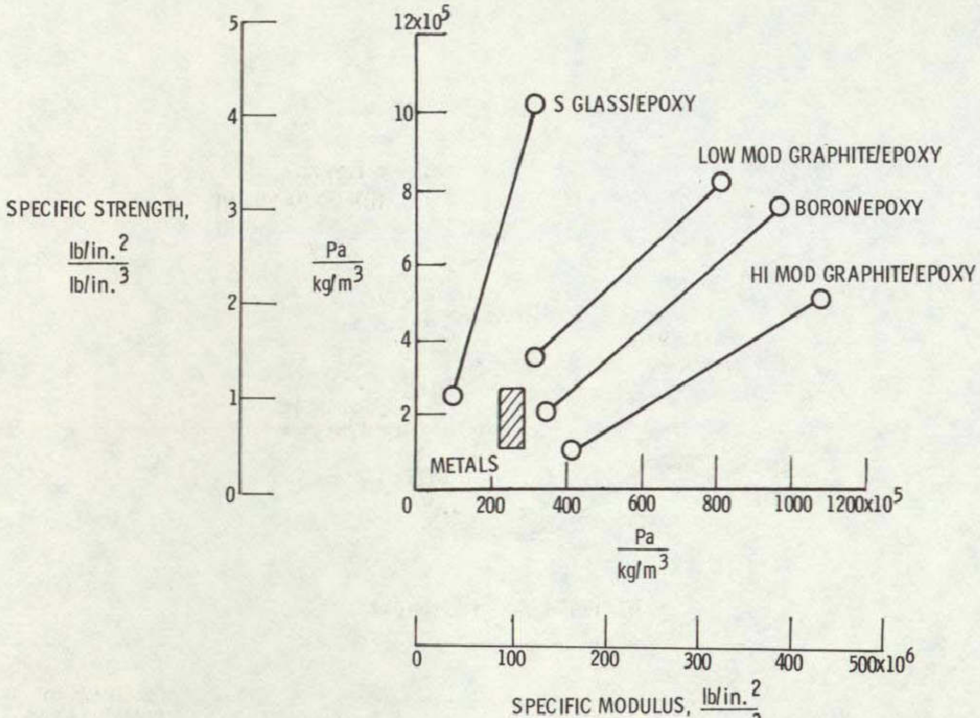


Figure 4. - Advantages of composites over metals. Room temperature data.

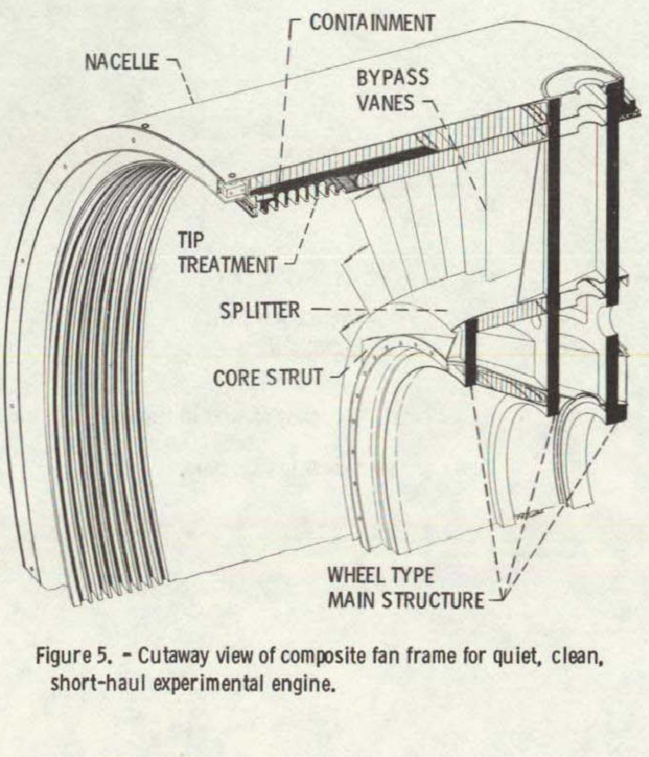


Figure 5. - Cutaway view of composite fan frame for quiet, clean, short-haul experimental engine.

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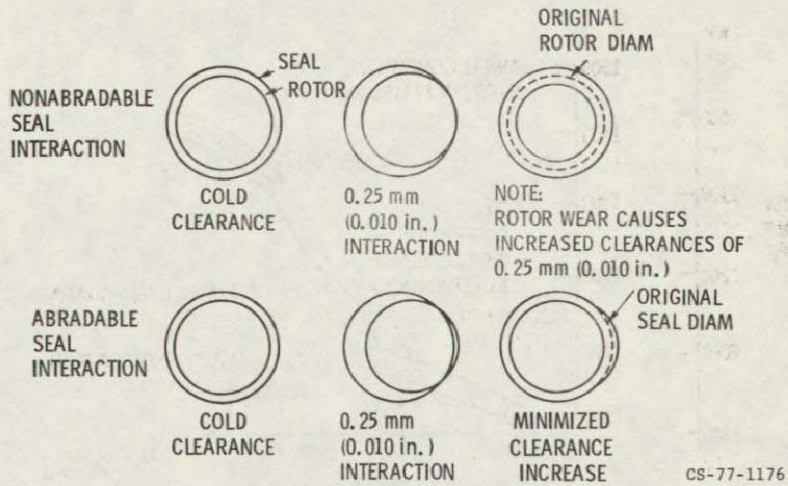

Figure 6. - Abradable seals can reduce postrub clearance by 70 percent.

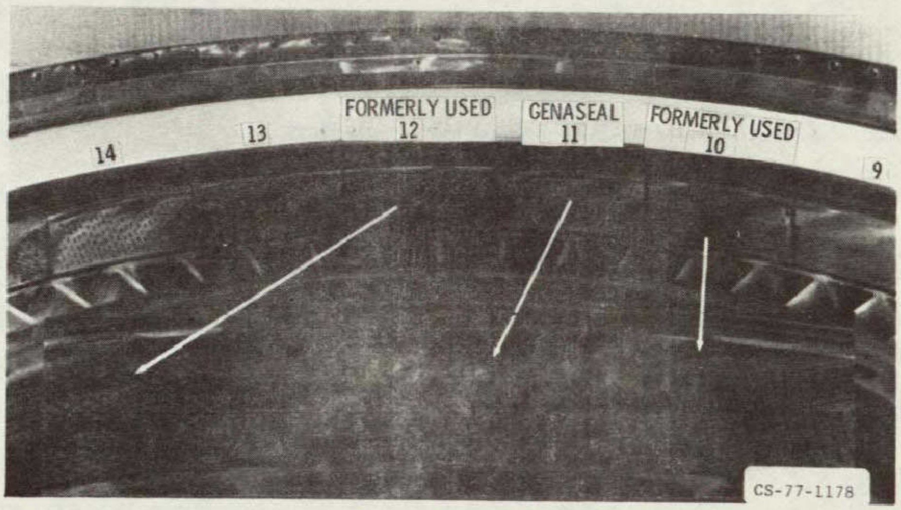

Figure 7. - Abradable porous NiCrAlY (Genaseal) shows improvement over formerly used material after 1000 test cycles in CF6-50C engine ground tests. Genaseal numbers 11 and 14; formerly used seal material numbers 9, 10, 12, 13.

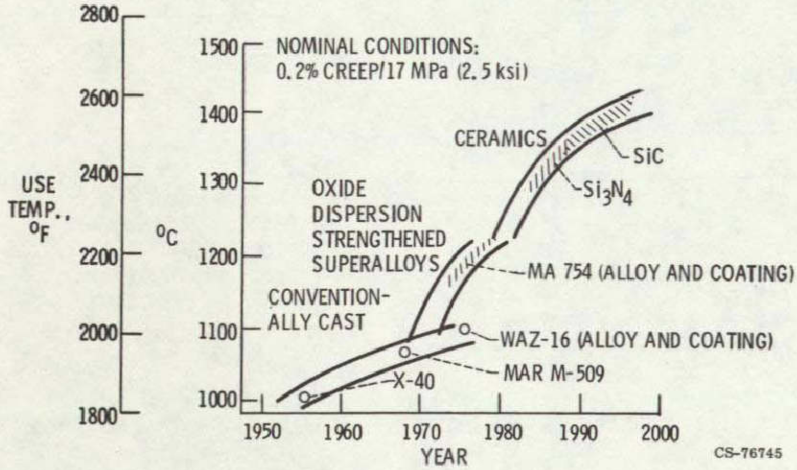

Figure 8. - Increased use temperature projected for ODS superalloys and ceramics for high-temperature, low-stress applications.

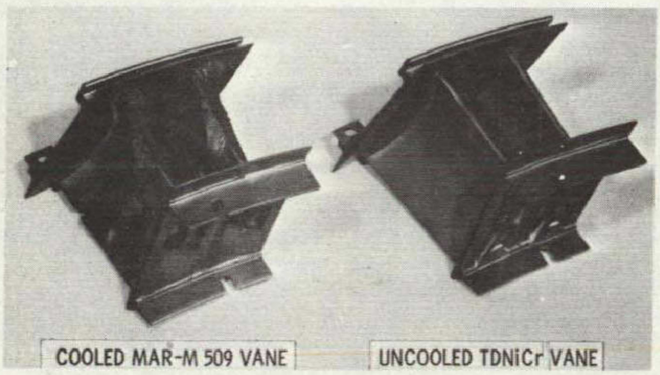

Figure 9. - Superiority of ODS vanes at overtemperature conditions.

CS-76752

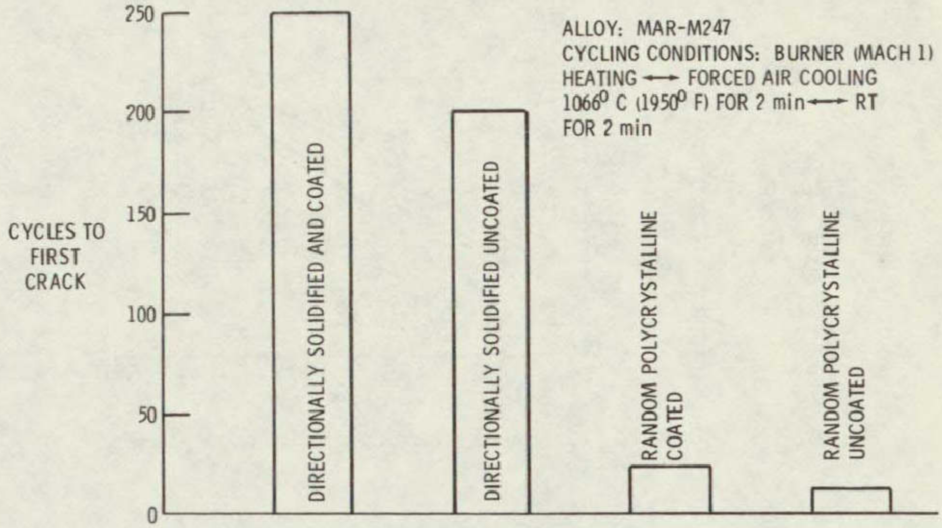

Figure 10. - Directional solidification increases the thermal fatigue resistance of superalloys.

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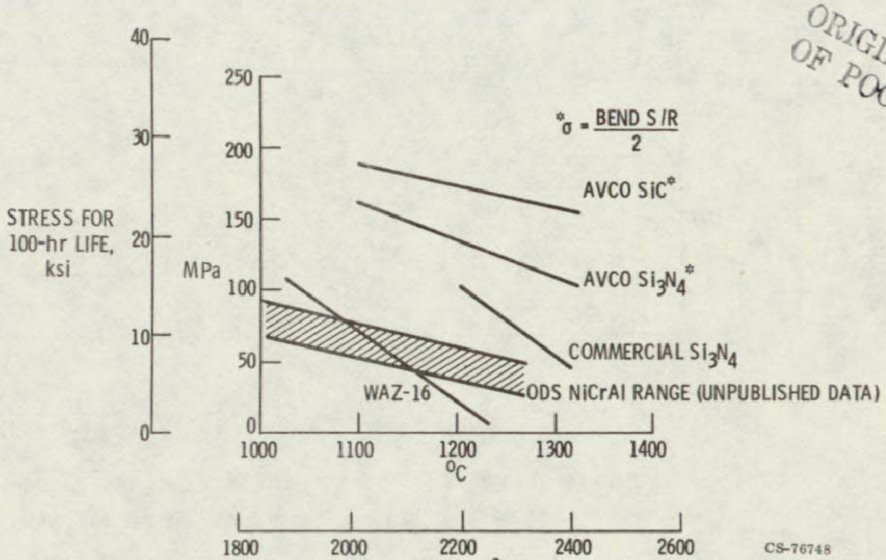

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Figure 11. - Ceramics show significant stress-rupture-strength increase over metallic vane materials.

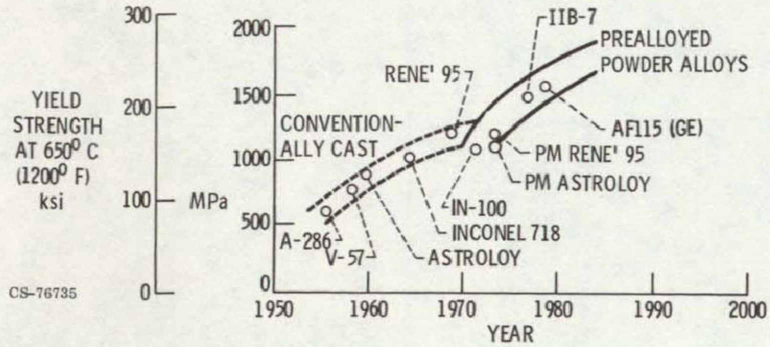
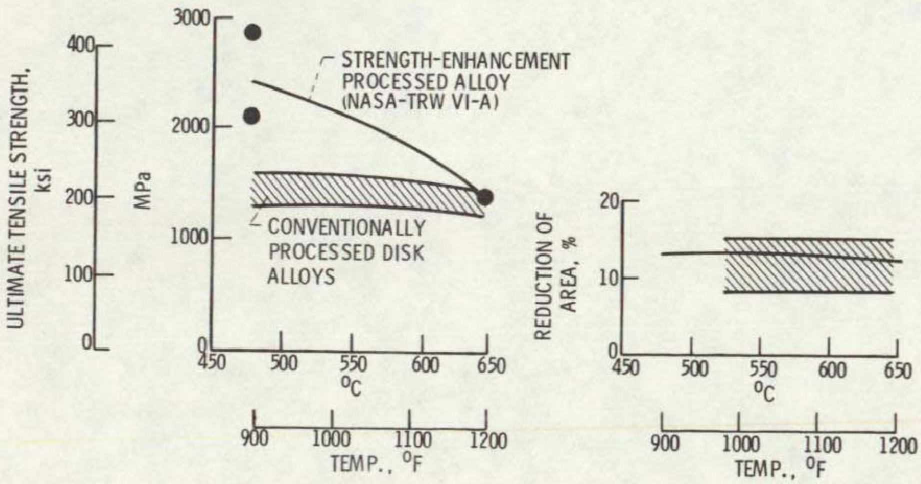


Figure 12. - Increased yield strength projected for prealloyed powder alloys.



(a) Greater strength.

(b) Comparable ductility.

Figure 13. - Advantages of prealloyed-powder strength enhancement process.

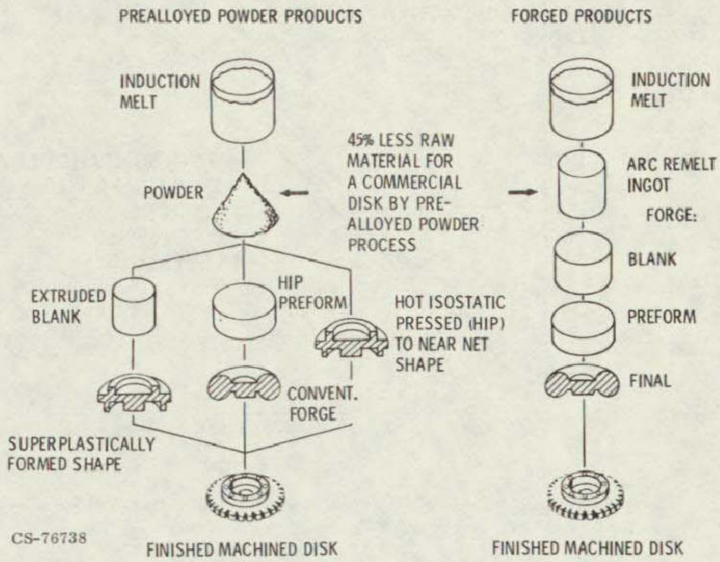


Figure 14. - Prealloyed powder process permits lower disk fabrication cost than conventional forging process.

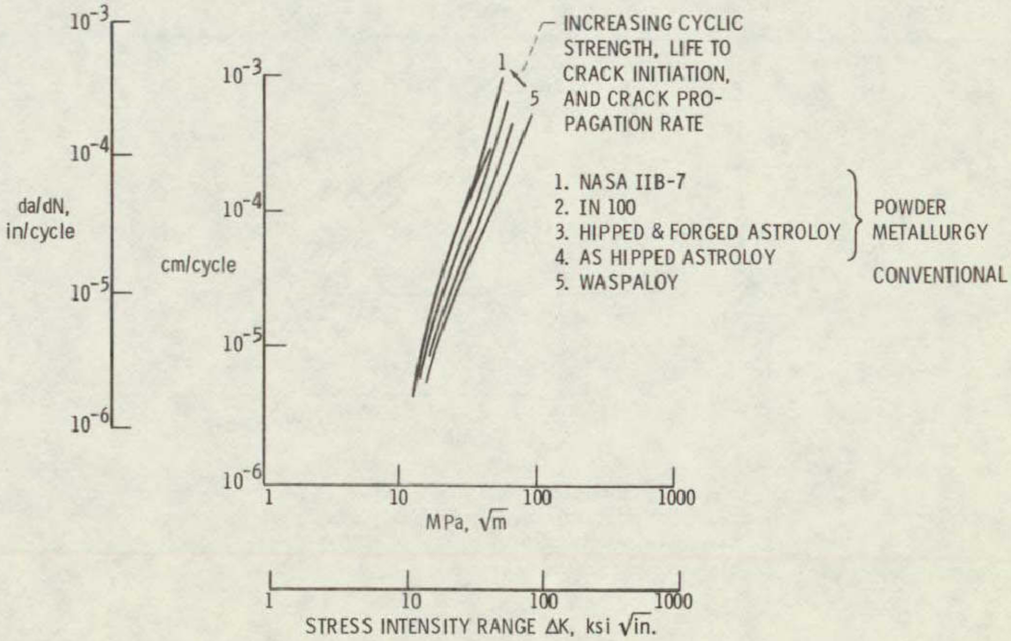


Figure 15. - Cyclic crack growth behavior of disk alloys at 650° C (1200° F).

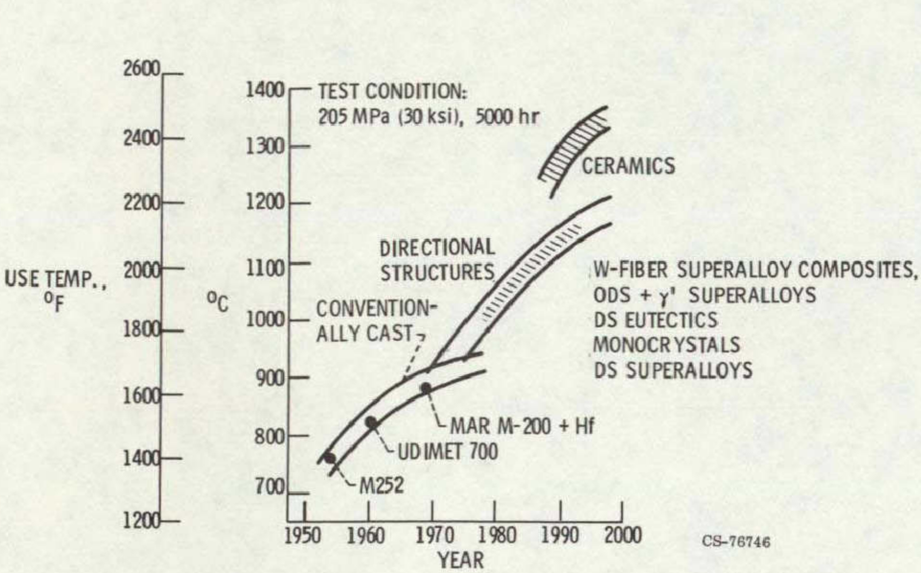


Figure 16. - Increased use temperature projected for directional structures for high-stress, high-temperature blade applications.

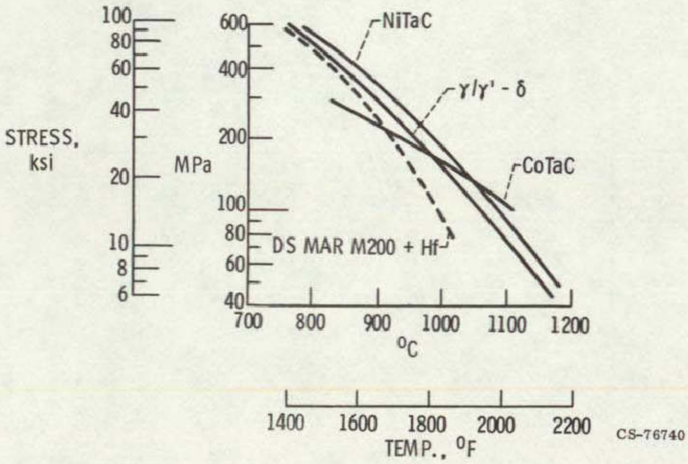
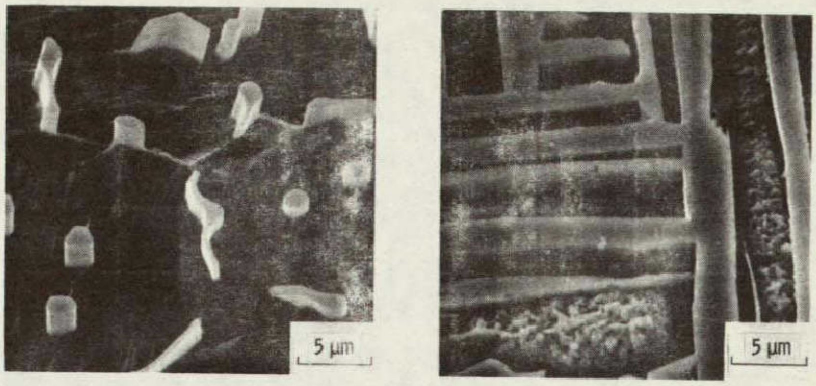


Figure 17. - Directionally solidified eutectic alloys for blade application show significant advantage over directionally solidified superalloy in 1000 hr stress-to-rupture test.



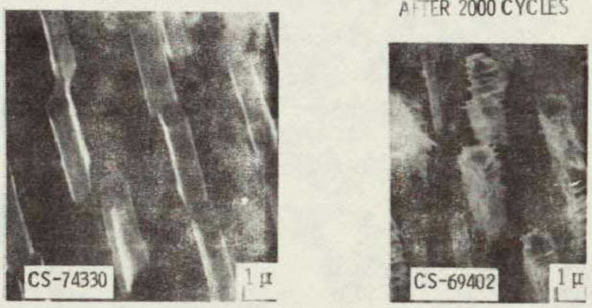
ROD HfC; HAFCO

LAMELLAR Ni_3Cb; γ/γ'-δ CS-76751

Figure 18. - Two types of D S Eutectics.

AS CAST

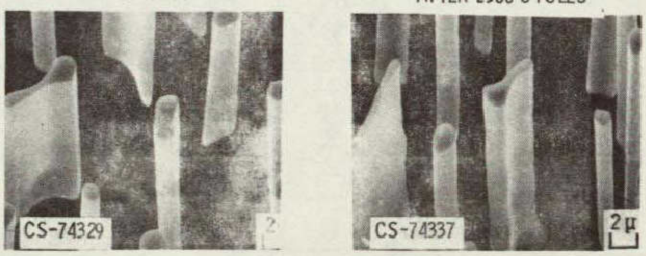
AFTER 2000 CYCLES



(a) CoTaC (Co-20 to 25 Ni-15 Cr-12 Ta-0.8 C).

AS CAST

AFTER 2500 CYCLES



(b) HAFCO (Co-20 to 25 Ni-15 Cr-10.5 Hf-0.7 C). CS-76753

Figure 19. - Compositional change may be needed to insure thermal stability of directionally solidified eutectics. Cycling conditions: 425^0 C (797^0 F)$\leftrightarrow$$1100^0$ C (2012^0 F).

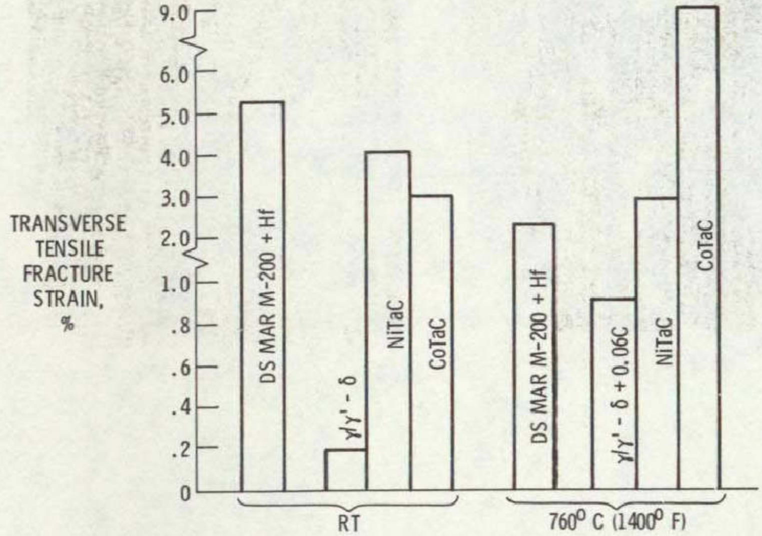
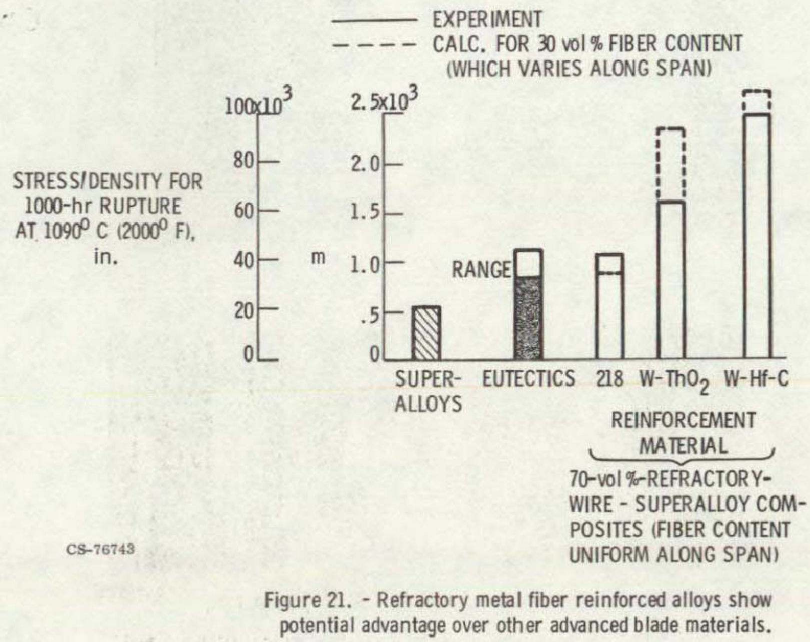


Figure 20. - Transverse ductility problem with some DS eutectics.



CS-76743

Figure 21. - Refractory metal fiber reinforced alloys show potential advantage over other advanced blade materials.

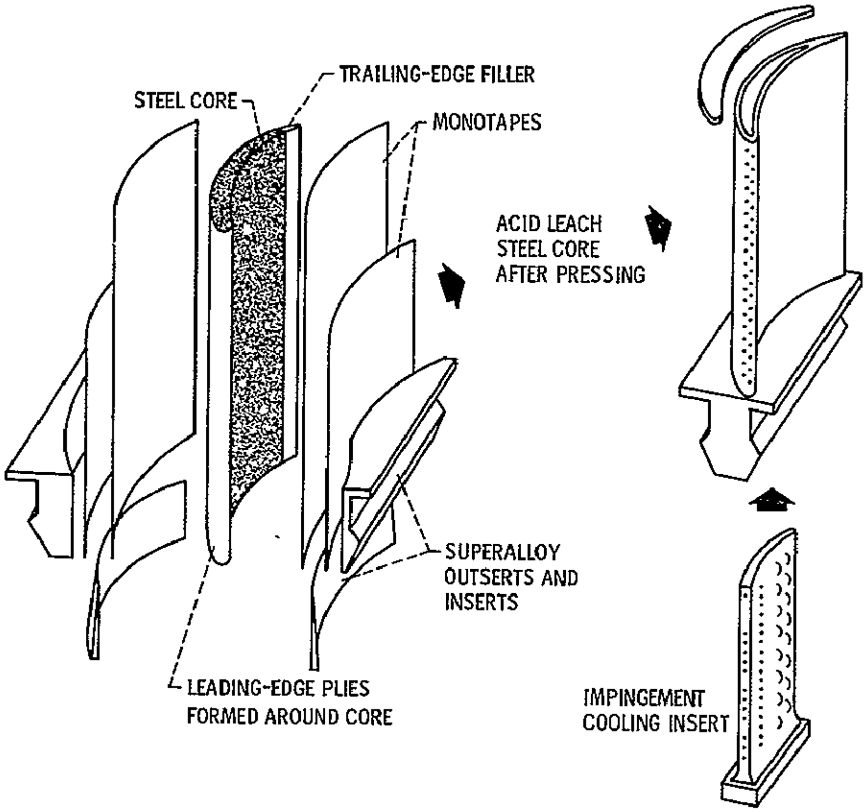


Figure 22. - Composite blade assembly schematic.

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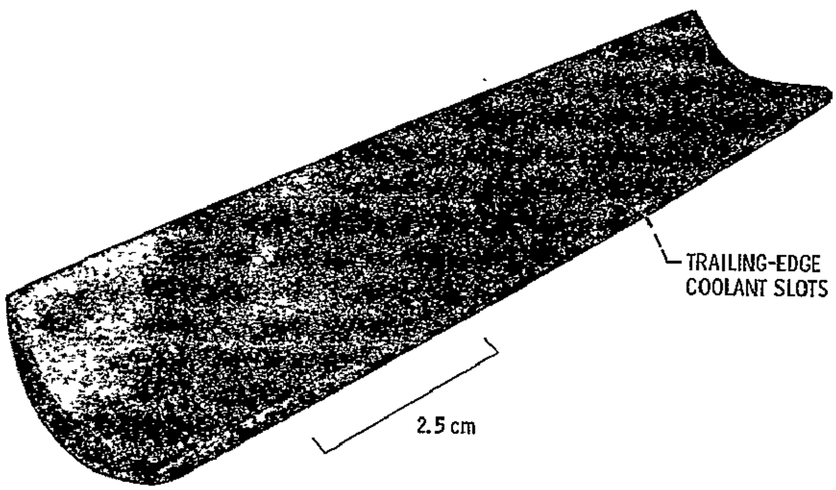

Figure 23. - Composite hollow airfoil.

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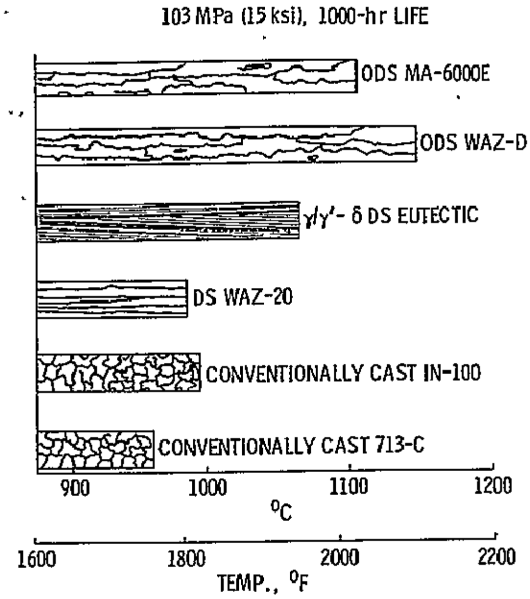


CS-76750

Figure 24. - Oxide dispersion plus γ' strengthening provides use-temperature increases over conventionally cast and DS alloys.

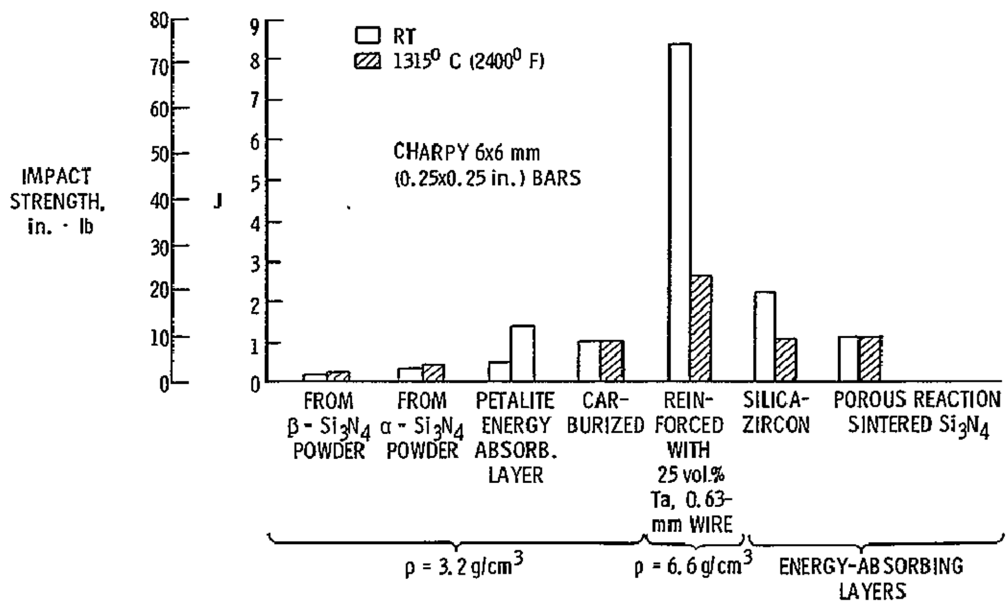


Figure 25. - Impact strength improvements in silicon nitride ceramic.

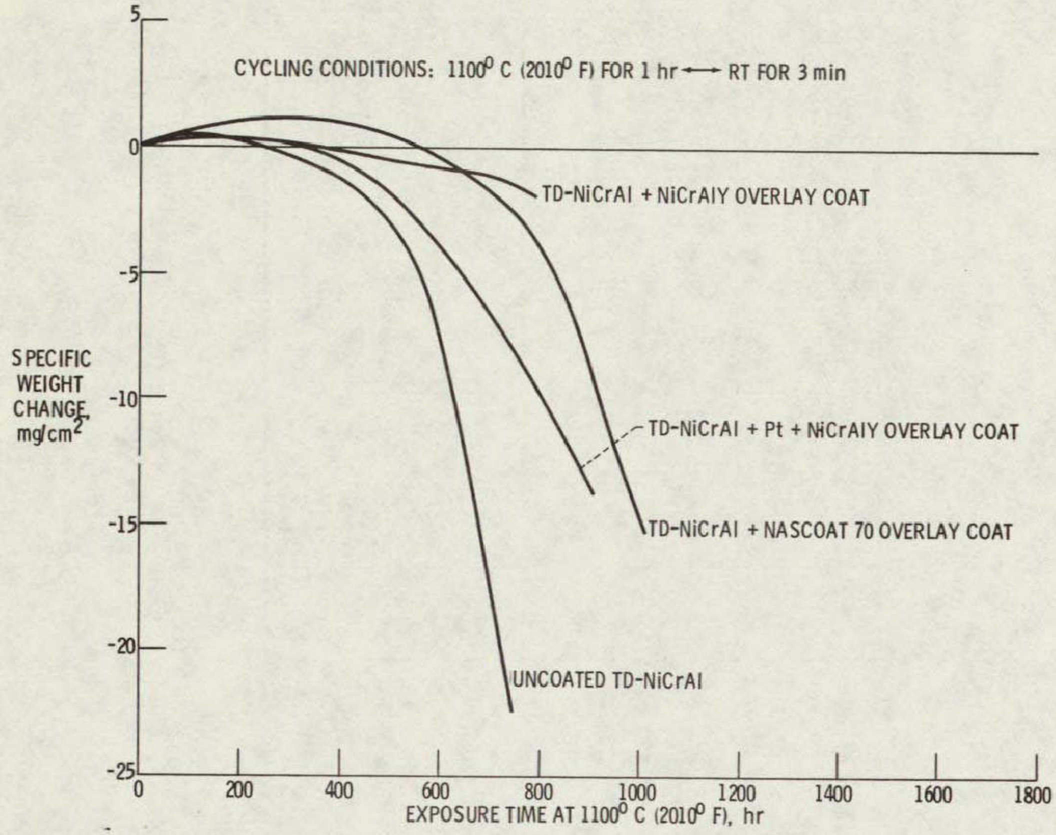
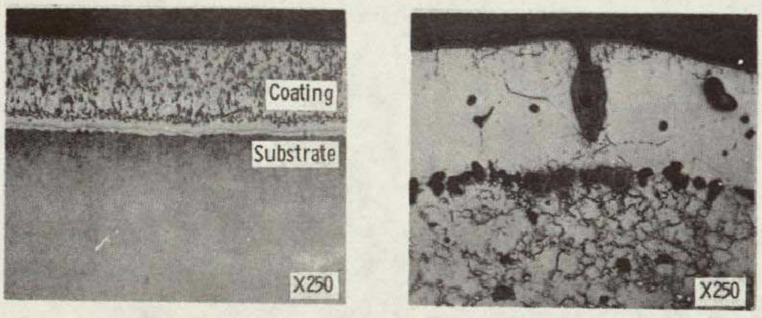


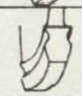
Figure 28. - Mach 1 burner rig oxidation of TD-NiCrAl at 1100° C (2010° F).



AS COATED

AFTER 500 HOURS

Figure 29.- Effect of ultrahigh-velocity, Mach 1 burner-rig oxidation of TD-NiCrAl + NiCrAlY overlay coating. Cycling conditions: 1100° C (2010° F) for 1 hr ⟷ RT for 3 min.

TYPE OF ROOT	MAXIMUM LIFE, hr	TEST RESULTS
	3	BENDING IN ROOT
	21	BENDING IN ROOT
	68	BENDING IN ROOT
	59	BENDING IN ROOT
	[a]242 (ALSO 108 CYCLES)	COMPRESSION IN ROOT CAUSED PIN TO LOOSEN; NO ROOT FAILURES
	[a]150	AIRFOIL (NO ROOT FAILURES)

[a]RUN DISCONTINUED - BLADES DID NOT FAIL.

CS-76729

Figure 26. - Early efforts to handle problem of low ductility of ceramic blades.

FUEL TYPE	IMPURITIES CONTRIBUTING TO HOT CORROSION					IMPURITIES CONTRIBUTING TO EROSION			COST, $/$10^6$ Btu	
	V, ppm	Na + K, ppm	Pb, ppm	S, %		ASH, ppm	FILTERABLE DIRT, %			
KEROSENE	0.1	0.1	----	0.1	INCREASING HOT CORROSION ↓	----	0.002	INCREASING EROSION ↓	2.80	INCREASING COST ↑
NO. 1GT[a]	0.5	0.5	0.5	LEGAL EPA LIMIT		100	~0.002		2.60	
BROAD-CUT[b] COAL-DERIVED LIQUID	<5.0	<10.0	----	>0.5		300	-------		2.25 (CALC)	
RESIDUAL OIL NO. 4GT TYPICAL	~500	~10.0	~5.0	LEGAL EPA LIMIT		~300	~0.2		2.00	

[a]ASTM 2880-76.
[b]FROM DOE REP. NO. HCP/W 3364-01/1.

Figure 27. - Dirty gas turbine liquid fuels cost less than kerosene but increase hot corrosion and erosion.

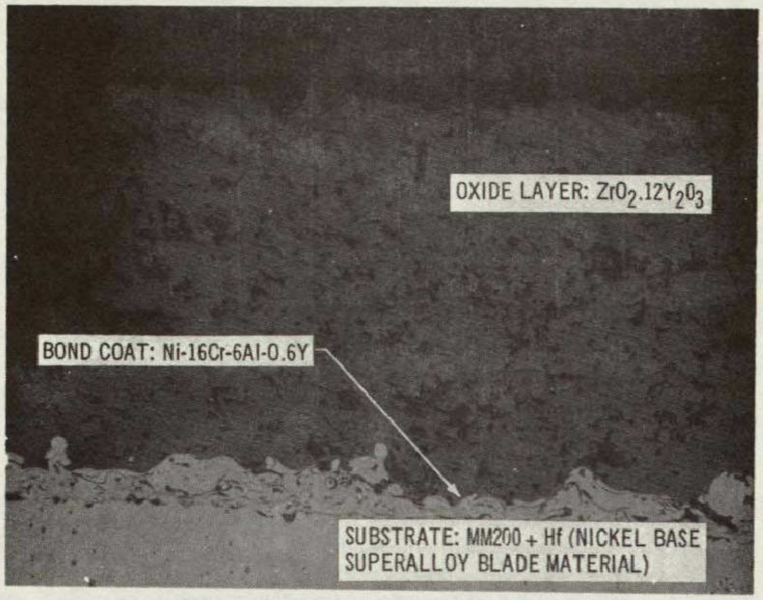


Figure 32. - Photomicrograph of the cross section of a layered thermal barrier coating.

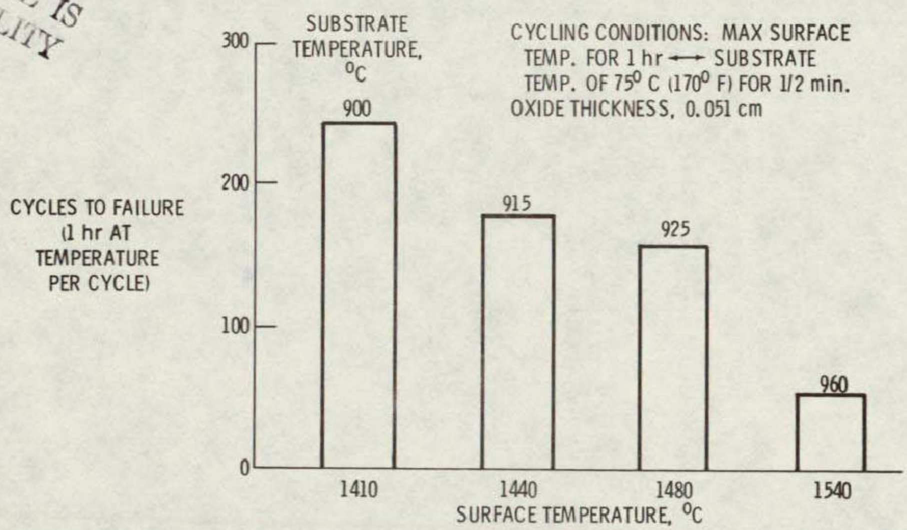


Figure 33. - Yttria-stabilized zirconia thermal barrier coating lives at high surface temperatures. Failures determined on basis of visual observation of coat thickness loss (40 to 50 percent) by erosion.

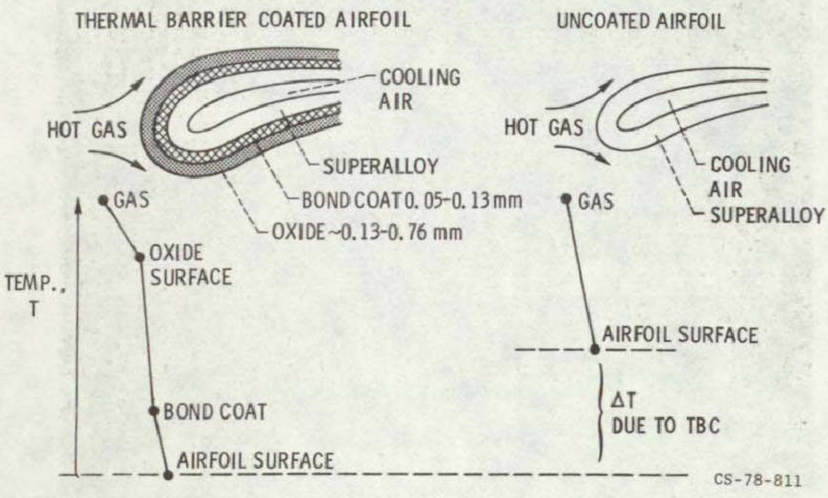


Figure 30. - Thermal barrier coatings (TBC): concept.

AVERAGE METAL-TEMPERATURE REDUCTION . . . . . . 55^{0} C (98^{0} F)

HOT-SPOT METAL-TEMPERATURE REDUCTION . . . . 67^{0} C (120^{0} F)

LEADING-EDGE STRAINRANGE REDUCTION 19%

Figure 31. - Calculated benefits of a 0.18-mm (0.007-in.) thermal-barrier coating on JT-9D first-stage blades at 70% span for sea level takeoff conditions.

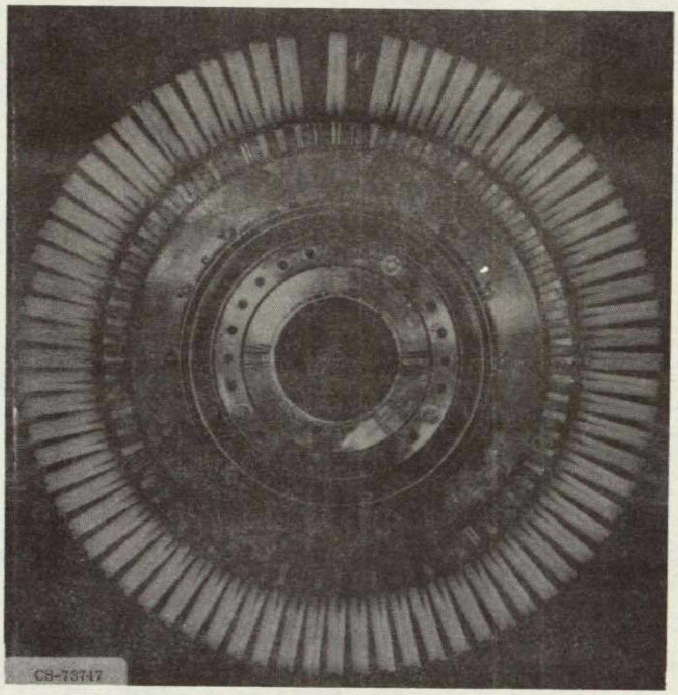


Figure 34. - Ceramic coated turbine blades. J75 first-stage rotor after 500 cycles. Turbine inlet to flame out, 1-min at maximum temp. (1370° C (2500° F)).

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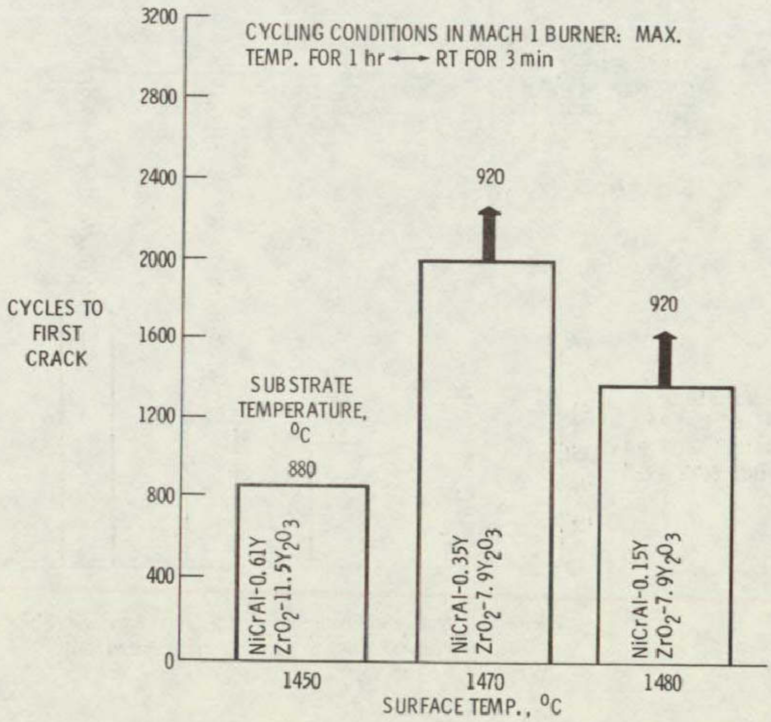

Figure 35. - Compositional modifications increase thermal barrier coating durability.

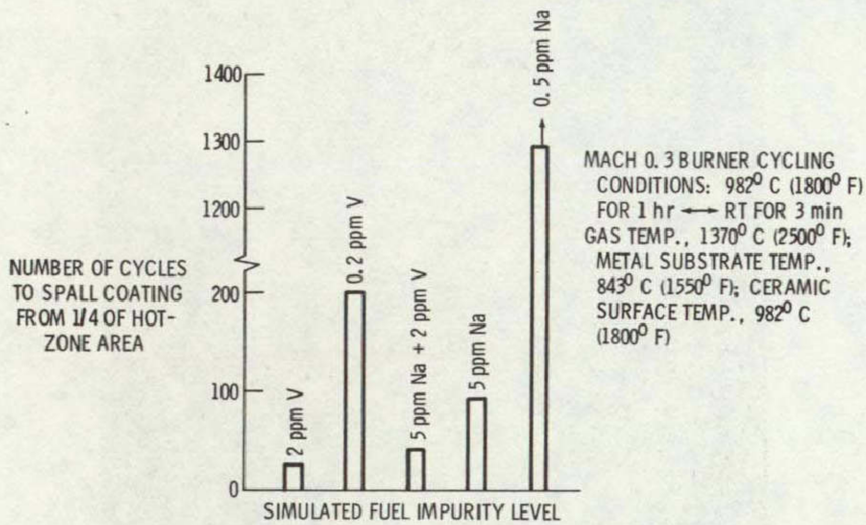

Figure 36. - Thermal barrier coatings are more sensitive to vanadium than sodium contaminant in fuels.

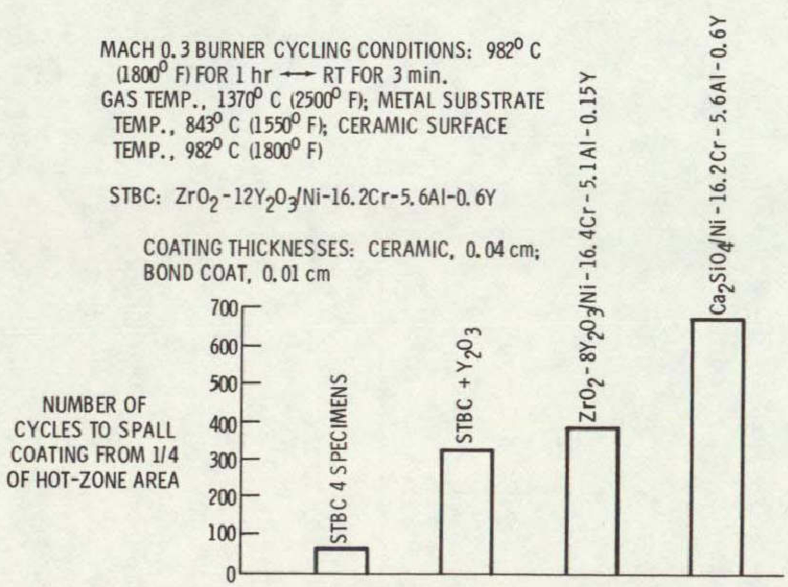

Figure 37. - Modifications to Standard Thermal Barrier Coating (STBC) improve tolerance to Na and V fuel contaminants (5 ppm Na + 2 ppm V).

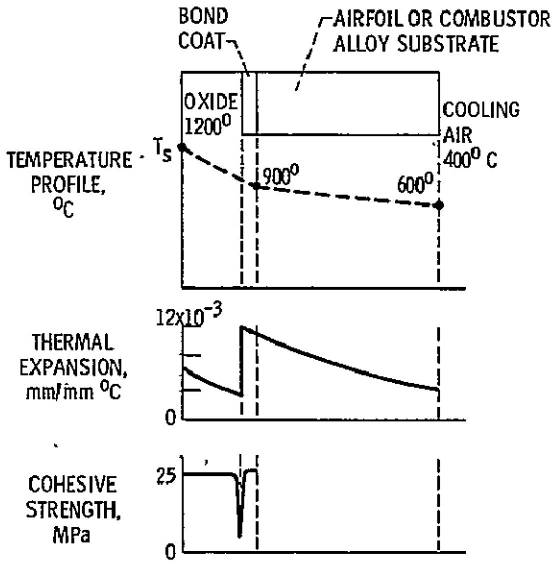

Figure 38. - Coating adherence considerations for thermal barrier coatings.

- REYNOLDS NUMBERS ARE LOWER
- BLADE-WALL CLEARANCES ARE RELATIVELY LARGER DUE TO FABRICATION DIFFICULTIES
- COORDINATE AND OTHER DIMENSIONS ARE LESS ACCURATE BECAUSE OF FABRICATION LIMITATIONS
- SURFACE ROUGHNESS IS RELATIVELY GREATER
- BLADE THICKNESSES ARE GENERALLY GREATER
    - TO AVOID DANGEROUSLY THIN TRAILING EDGES
    - TO ACCOMMODATE THE INTERNAL COOLANT FLOW PASSAGES

Figure 39. - Reasons small turbines are less efficient than large turbines.

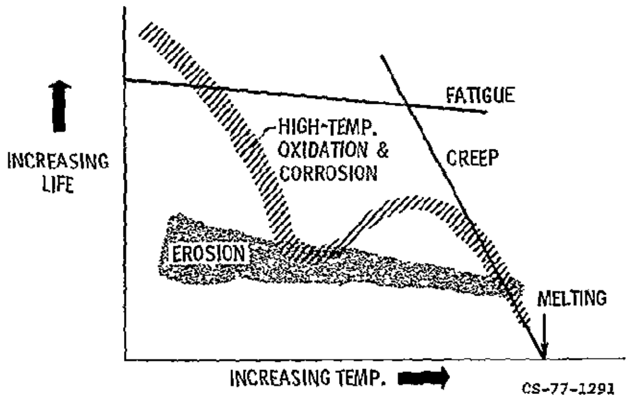

Figure 40. - Schematic of turbine blade life limits.

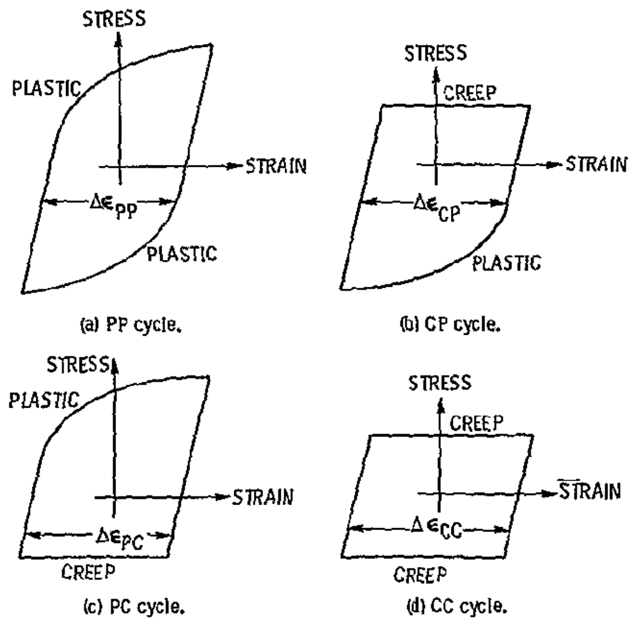

Figure 41. - Idealized hysteresis loops used in defining individually partitioned strainrange-life relationships.

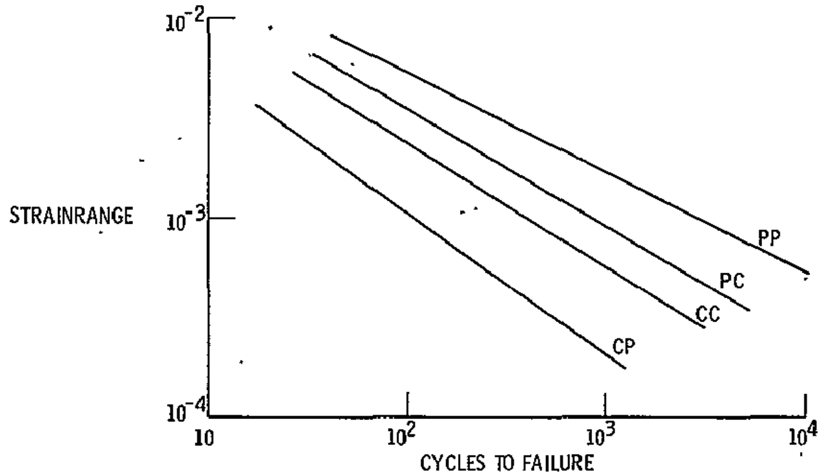

Figure 42. - Typical partitioned strainrange-life relationships used to characterize material behavior in the creep-fatigue range.

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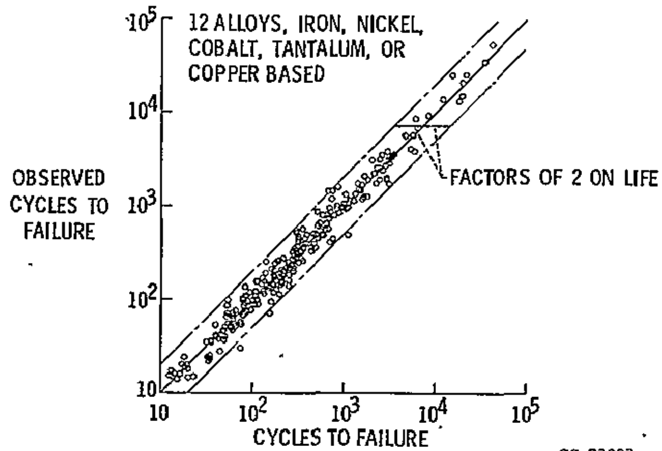

Figure 43. - Life predictability by strainrange partitioning.

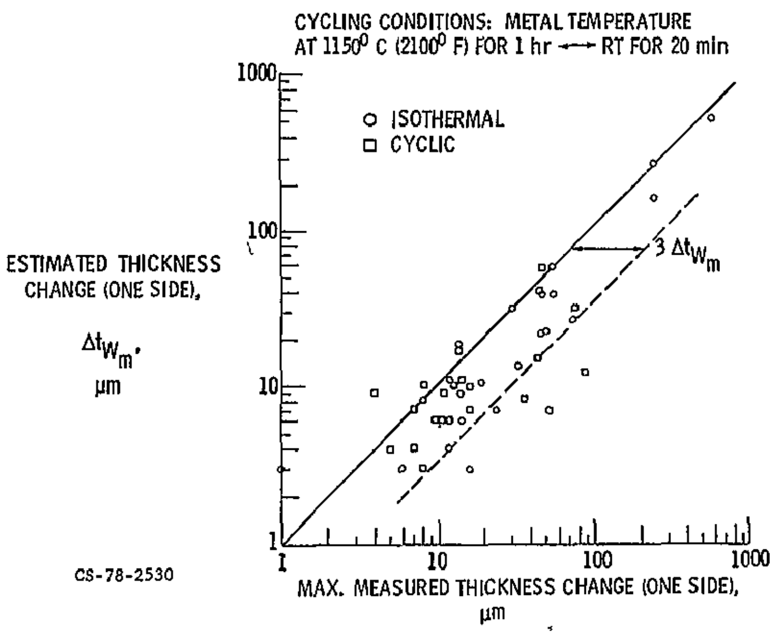

Figure 44. - COREST estimated versus measured metal thickness loss after 100 hours oxidation.

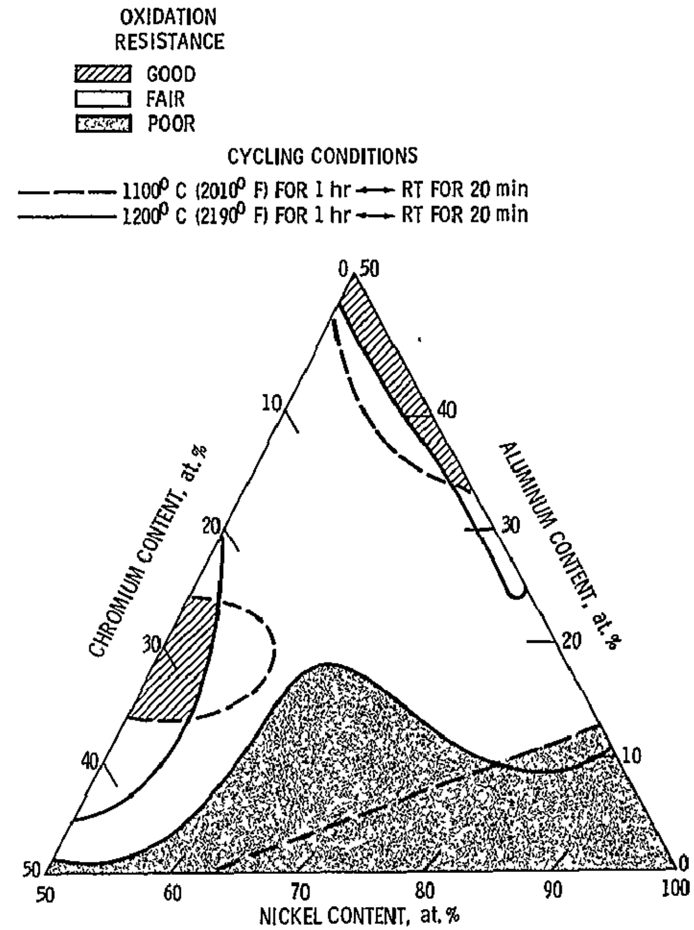

Figure 45. - Map for identifying oxidation resistant compositions for Ni-Cr-Al alloys. Total test times: 200 to 500 hours at temperature.

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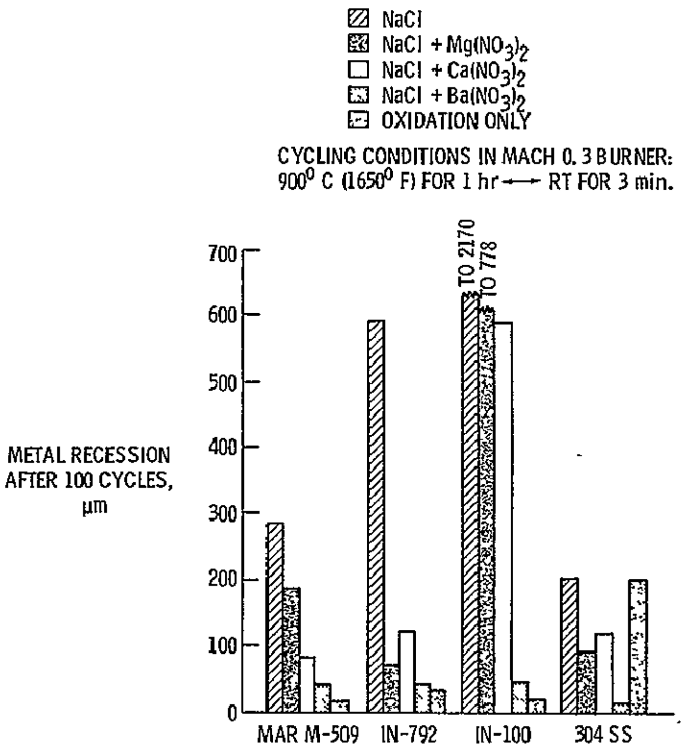

Figure 46. - Barium nitrate most effective additive for reducing hot corrosion in superalloys.

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15. Supplementary Notes		

16. Abstract

The key to improved helicopter gas turbine engine performance lies in the development of advanced materials and advanced structural and design concepts. The modification of the low-temperature components of helicopter engines (such as the inlet particle separator), the introduction of composites for use in the engine front frame, the development of advanced materials with increased use-temperature capability for the engine hot section, can result in improved performance and/or decreased engine maintenance cost. A major emphasis in helicopter engine design is the ability to design to meet a required lifetime. This, in turn, requires that the interrelated aspects of higher operating temperatures and pressures, cooling concepts, and environmental protection schemes be integrated into component design. The major material advances, coatings, and design life-prediction techniques pertinent to helicopter engines are reviewed; the current state-of-the-art identified; and an assessment, when appropriate, of progress, problems, and future directions is provided.

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