

Materials for Advanced Rocket Engine
Turbopump Turbine Blades

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Abstract

A study program was conducted to identify those materials that will provide the greatest benefits as turbine blades for advanced liquid propellant rocket engine turbines and to prepare technology plans for the development of those materials for use in the 1990 through 1995 period. The candidate materials were selected from six classes of materials--single-crystal (SC) superalloys, oxide dispersion-strengthened (ODS) superalloys, rapid solidification processed (RSP) superalloys, directionally solidified eutectic (DSE) superalloys, fiber-reinforced superalloy (FRS) composites, and ceramics. Properties of materials from the six classes were compiled and evaluated and property improvements were projected approximately 5 years into the future for advanced versions of materials in each of the six classes. The projected properties were used in turbine blade structural analyses based on the design and operating conditions of the Space Shuttle Main Engine. It was concluded that the materials that warranted development were the SC superalloys for use at 870 C (1600 F), the FRS composite for use at 870 C (1600 F) and 1093 C (2000 F), and the ceramic for use at 1093 C (2000 F).

Introduction

Requirements for improved performance and longer life in advanced liquid propellant rocket engines have focused attention on the potential benefits of advanced, high-temperature structural materials for rocket engine turbopump turbine blades.

Compared to aircraft gas turbines, rocket engine turbines experience very severe thermal start/stop transients, high operating speeds, and hydrogen environments which result in the following unique requirements for rocket engine turbine blade materials: high thermal strain low-cycle fatigue strength; high mean stress high-cycle fatigue strength; resistance to hydrogen environment embrittlement; thermal shock resistance; and relatively short time stress-rupture/creep strength. Fatigue requirements are particularly severe as evidenced by the fact that the directionally solidified, hafnium modified MAR-M246 (MAR-M246(Hf) (DS)) turbine blades in both the High Pressure Fuel Turbopump (HPFTP) and High Pressure Oxidizer Turbopump (HPOTP) of the Space Shuttle Main Engine (SSME) are subjected to life-limiting fatigue cracking as outlined in Table 1. Thus, there is a need for improved turbine blade materials for increased life under current SSME turbine operating conditions and for increased temperature capability for improved rocket engine performance for future liquid propellant rocket engines.

Table 1. SSME Turbine Blade Life Limitations

<ul style="list-style-type: none"> • DESIGN LIFE GOAL - 55 ENGINE FIRINGS, 27,500 SECONDS • 1st STAGE HPFTP BLADES <ul style="list-style-type: none"> • TRANSVERSE LEADING EDGE AIRFOIL CRACKS - LCF + HCF • CURRENT LIFE ~ 6-8 FIRINGS • 2nd STAGE HPFTP <ul style="list-style-type: none"> • TRANSVERSE TRAILING EDGE SHANK CRACKS - HCF • CURRENT LIFE ~ 6 FIRINGS • 1st STAGE HPOTP <ul style="list-style-type: none"> • TRANSVERSE TRAILING EDGE FIRTREE CRACKS - HCF • CURRENT LIFE ~ 6 FIRINGS
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Currently, there are some quite sophisticated efforts underway to further extend the capabilities of turbine blade materials. These include not only the various classes of superalloys (single crystal (SC) superalloys, oxide dispersion strengthened (ODS)

superalloys, rapid solidification-processed (RSP) superalloys, fiber-reinforced superalloys (FRS), and directionally solidified eutectic (DSE) superalloys) but also structural ceramics. Thus, a program was conducted to identify those materials from among these six classes that would provide the greatest benefits as turbine blades for advanced liquid propellant rocket engine turbopump turbines in the 1990 through 1995 period, and to prepare development plans for advanced versions of those materials. The program included the gathering, evaluation, and projection of data for materials believed to show promise for the proposed application and performing structural analyses of blades using the selected materials.

The configuration and operating conditions of the SSME high pressure fuel turbopump first-stage turbine blade were used for the blade analyses. The turbine blade steady-state operating temperatures of interest range from 870 C (1600 F), the approximate steady-state operating temperature of turbine blades in the SSME, to above 1093 C (2000 F). Thus, the analyses and comparison of the potential of the candidate materials were conducted for those two temperature regimes. Based on the blade analyses, development plans were prepared and benefit analyses performed to permit the identification of those materials offering the best balance of benefits to the system, risk, and development costs.

Results and Technical Discussion

Evaluation and Projection of Data

The mechanical, physical, and environmental effects properties required to analyze and compare the potential of the various candidate materials for use in rocket engine turbine blades in the 1990 to 1995 period were acquired, evaluated, and projected.

As noted earlier, properties were required for structural analyses at both 870 C (1600 F) and 1093 C (2000 F).

A developmental time of approximately 5 years was assumed, at which time the specific optimum turbine blade material must be identified. Therefore, for each of the six classes of materials, mechanical properties were projected 5 years into the future, based on the properties of the best of the current materials and anticipated improvements. The projected mechanical properties were converted into design (minimum) values for use in blade structural analyses. For other properties, for example, thermal properties, current typical values were used in the analyses.

Single Crystal (SC) Superalloys. Data acquisition was initiated for over a dozen SC alloys. Based on an early comparison of properties, NASAIR-100, PWA 1480, and the CM SX alloys were selected as having the greatest near-term potential for rocket engine turbine blades, however NASAIR-100 was found to be very susceptible to hydrogen environment embrittlement (Ref. 1). Thus, the projected values for the mechanical properties for single-crystal superalloys are based on consideration of the properties of PWA 1480 and CM SX series alloys, with the majority of the projected values based on CM SX-2 data. Assumptions made in generating these projected properties were that single-crystal superalloys would be used in rocket engines at operating temperatures of approximately 870 C (1600 F), and that the projected properties appear to be attainable in 5 years through processing developments such as high gradient casting and hot isotatic pressing which reduce porosity that serve as fatigue crack initiation sites and crystal orientation control. In particular, high gradient casting has been shown in the laboratory to provide dramatic increases in high cycle fatigue properties (Ref. 2) and it was assumed in the projections that this process could be transformed into an industrial process in 5 years.

Oxide Dispersion Strengthened (ODS) Superalloys. After evaluation of various ODS alloys, it was concluded that Inconel MA 6000 which combines coherent γ' and incoherent Y_2O_3 particles in a nickel-based

solid solution strengthened matrix, is the only commercial ODS superalloy that may be considered a candidate turbine blade material for use in the 870 C (1600 F) to 1093 C (2000 F) temperature range. International Nickel Company has also developed higher δ' volume fraction ODS superalloys, and the most promising of these alloys, Alloy 51, has been well characterized (Ref. 3,4).

To project the potential physical and mechanical properties of ODS alloys 5 years into the future, the properties of Inconel MA 6000 and of Alloy 51, were examined and it was assumed that with appropriate development effort, the higher of the strength properties would be available at that time.

Rapid Solidification Processed (RSP) Superalloys. After evaluation of various RSP alloys, it was concluded that those showing the greatest promise for rocket engine turbine blades were the Pratt & Whitney Rapid Solidification Rate (RSR) processed Alloy 185 (a Ni-Al-Mo ternary alloy) and an Alloy X variant optimized for turbine blade applications that were developed under a DARPA/AFWAL contract (Ref. 5). Thus, projected properties were based on the properties of these alloys, assuming that processing of these alloys could be developed in 5 years.

Directionally Solidified Eutectic (DSE) Superalloys. Some 22 DSE alloys were surveyed and from these, four alloys, NiTaC-13, NiTaC-14B, CoTaC-744, and AG-170, were selected for evaluation. It was concluded that of these, NiTaC-14B has the greatest promise for rocket engine turbine blades because of greater high cycle fatigue strength and transverse ductility than the other alloys.

Thus, projections for the DSE alloy mechanical properties were based on the properties of NiTaC-14B (Ref. 7) assuming that this alloy could be developed for use in rocket engine turbine blades in 5 years with additional modest improvements in longitudinal strength (10 percent) and transverse ductility (15 percent).

Fiber-Reinforced Superalloy (FRS) Composite. A study program directed at applications of FRS to rocket engine turbines recently has been completed at Rocket-dyne Division of Rockwell International (Ref. 7). The findings of that program serve as the source of the projected properties for FRS composites. The selected composite on which projected properties were based was a W-4Re-0.38 Hf-0.02C fiber-reinforced Waspaloy assuming that this material can be developed for use in rocket engine turbine blades within 5 years.

Ceramics. A list of candidate materials for ceramic turbine blades was compiled at the onset of this program. Those materials on the list had to pass rigid requirements qualifying them as potential advanced turbine blade materials. The pertinent properties essential for designing an advanced ceramic turbine blade were then compiled. A first cut evaluation reduced the candidates to silicon nitride and silicon carbide processed by various methods. Upon further evaluation sintered silicon nitride (SSN) was selected as the ceramic with the greatest promise for rocket engine turbine blades mainly because of higher strength, Weibull modulus, and resistance to thermal shock.

Comparison of Projected Mechanical Properties. Figures 1 through 6 contain comparisons of the projected properties among the 6 classes of materials. Based on these data, the SC superalloy and the FRS composite appear to show the greatest promise for use at 870 C (1600 F) and the FRS composite and ceramic show the greatest promise for use at 1093 C (2000 F). Although very little strain-controlled, low-cycle fatigue data have been generated for FRS composites, low-cycle fatigue behavior was projected using the Manson-Coffin approach. The results are shown in Fig. 6, which was derived for a composite based on 50 volume percent W-Re-Hf-C fiber in a Waspaloy matrix at 870 C (1600 F) using the overall strength of the composite but other properties of the individual components to calculate the upper and lower limits shown. Comparison of Fig. 5 and 6 show that

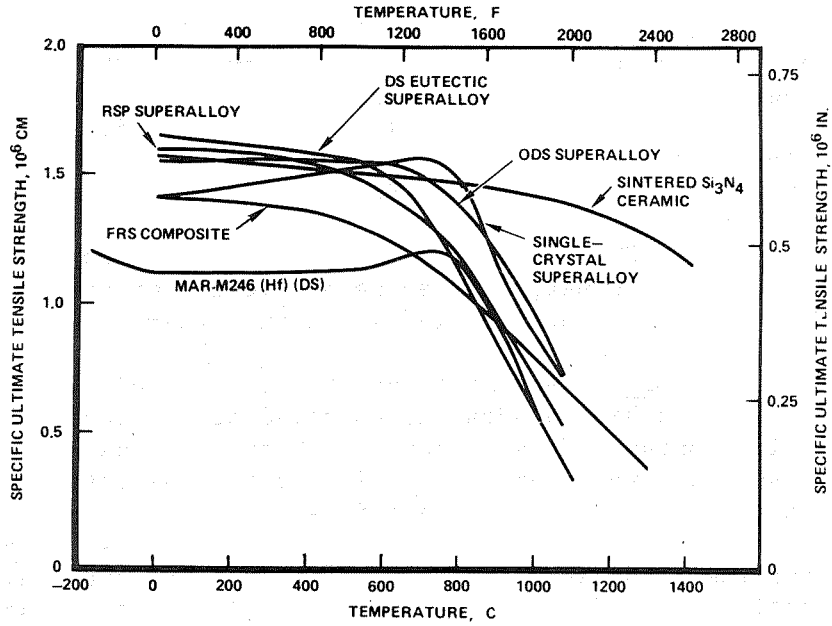


Fig. 1. Projected Design Specific Ultimate Tensile Strength

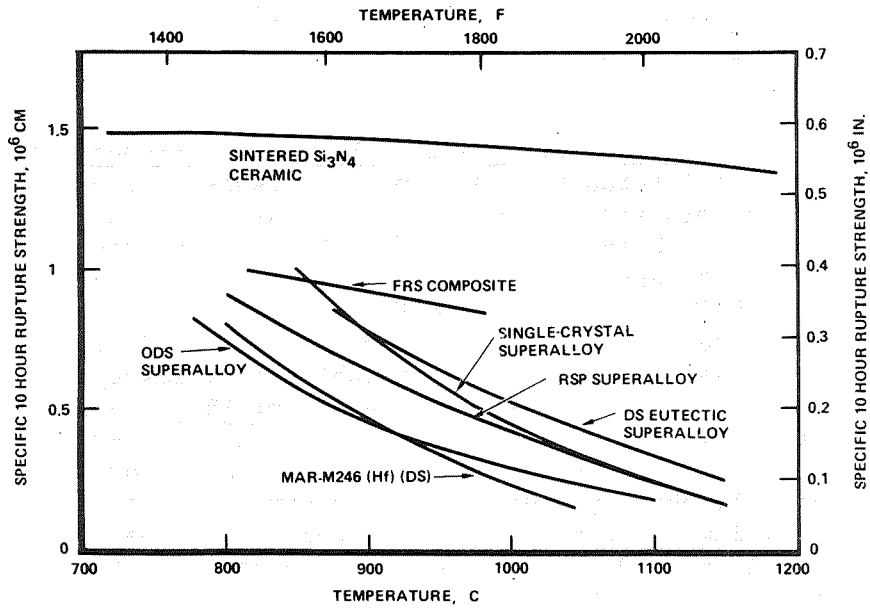


Fig. 2. Projected Design Specific 10-Hour Stress Rupture Strength

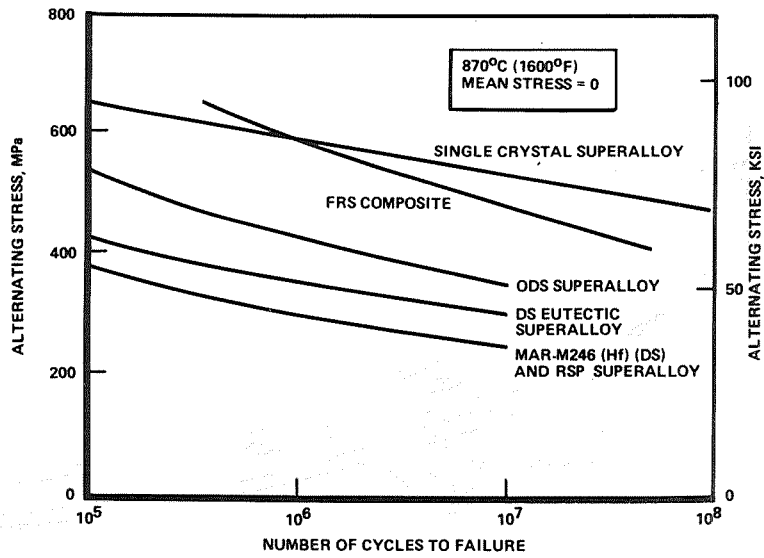


Fig. 3. Projected Design High-Cycle Fatigue Strength

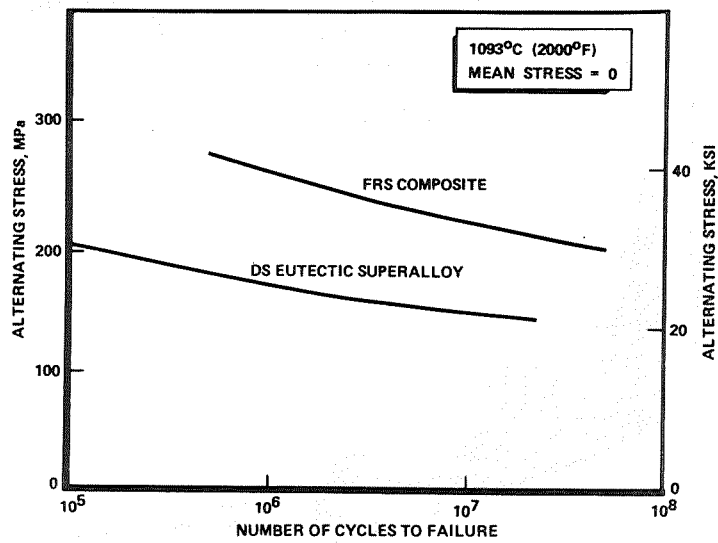


Fig. 4. Projected Design High-Cycle Fatigue Strength

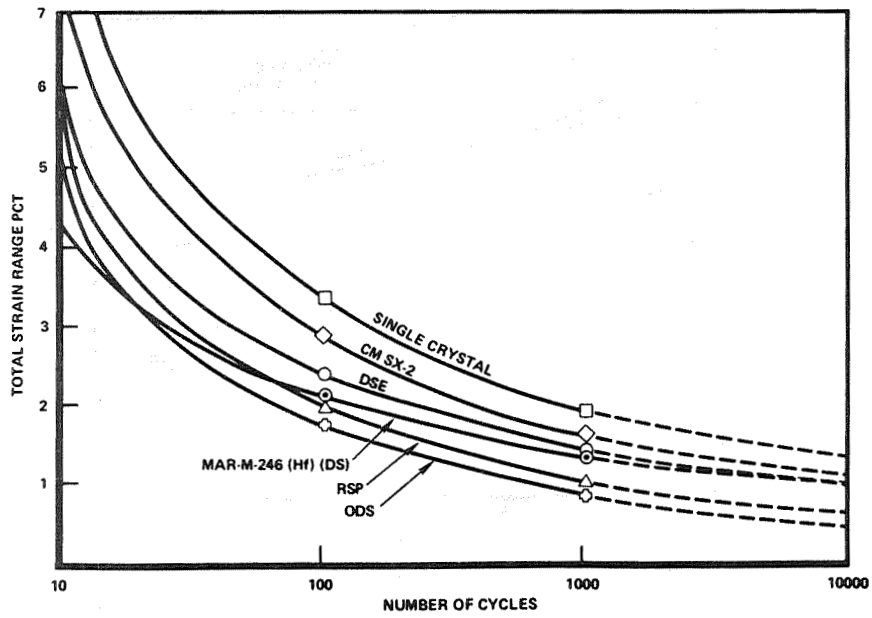


Fig. 5. Projected Design Low-Cycle Fatigue Properties (870 C, 1600 F)

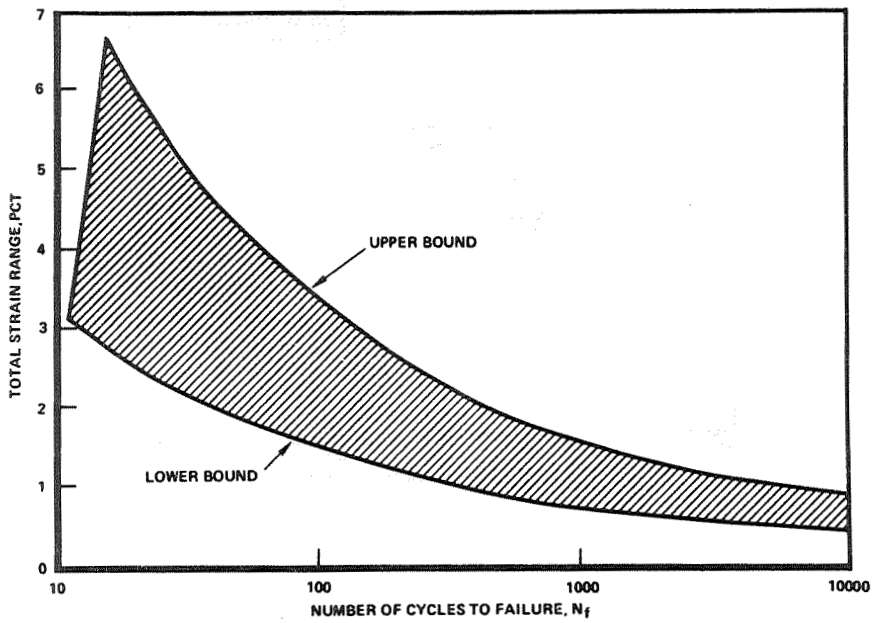


Fig. 6. Projected Design Low-Cycle Fatigue Properties for FRS Composite (870 C, 1600 F)

the projected extremes for the FRS composite embrace the entire behavior for all the other alloys considered during this program.

Turbine Blade Structural Analysis

Turbine blade structural analyses were conducted using the projected properties for the advanced material in each of the six classes so that the impact on performance of using each of these materials could be compared with each other and with MAR-M246(Hf)(DS) and two current advanced materials, the CM SX-2 SC superalloy and the NiTaC-14B DSE superalloy. The analyses were conducted for a steady-state operating temperature of 870 C (1600 F) for eight materials: the projected material for each of the six classes and for MAR-M246(Hf)(DS) and CM SX-2 SC superalloy. Analyses also were conducted for a steady-state operating temperature of 1093 C (2000 F) for four materials: the projected DSE, FRS, and ceramic materials and the NiTaC-14B alloy. The turbine blade on which the structural analyses were based was the first-stage blade of the SSME high pressure fuel turbopump.

Stress Rupture Evaluation for 870 C (1600 F).

The stress rupture evaluation for 870 C (1600 F) was performed using the calculated mean stress due to centrifugal loading for the reference MAR-M246(Hf)(DS), calculating the mean stress for each of the other materials based on the ratio of its density to that of MAR-M246(Hf)(DS), and comparing these calculated stresses to the stress-rupture strengths from the previous section. The ranking order of the materials from the stress rupture evaluation for 870 C (1600 F) is shown in Table 2.

High-Cycle Fatigue Evaluation for 870 C (1600 F).

The high-cycle fatigue evaluation for 870 C (1600 F) was performed using mean stresses calculated as described for the stress rupture evaluation and basing the alternating stress on test data obtained by vibrating an extensively strain-gaged MAR-M246(Hf)(DS) blade at its fundamental frequency. The

Table 2. Stress Rupture Evaluation at
870 C (1600 F)

MATERIAL	10 HOUR RUPTURE STRENGTH CALCULATED MEAN STRESS
MAR-M246 (Hf) (DS)	1.51
Si ₃ N ₄ CERAMIC	3.80
FRS COMPOSITE	2.68
SINGLE CRYSTAL SUPERALLOY	2.31
DS EUTECTIC SUPERALLOY	2.31
RSP SUPERALLOY	1.94
CM SX-2 SC SUPERALLOY	1.92
ODS SUPERALLOY	1.36

alternating stress was assumed to be due only to pressure pulses and thus was assumed to be the same for the different materials.

High-cycle fatigue capabilities were analyzed using a modified Goodman diagram that shows the failure limits for combinations of mean and alternating stresses. The failure limits are established by the materials endurance (10^7 cycles) limit, ultimate tensile strength, and 10-hour stress-rupture strength. Figure 7 shows a modified Goodman diagram for MAR-M246(Hf)(DS) for four blade locations. The failure limit for a given ratio of mean and alternating stresses is represented by the length of a ray from the origin to the limit line (the limit length). The portion "used up" by the stresses at a given position on the blade is the length of the ray from the origin to that point (the point length). The high-cycle fatigue factor of safety is the limit length divided by the point length. The high-cycle fatigue minimum factors of safety for blades of all the materials for 870 C (1600) are listed in Table 3.

Low-Cycle Fatigue Evaluation for 870 C (1600 F).
Different thermal strains will occur in blades of different materials when subjected to the same hot gas transients. The thermal strains are proportional to

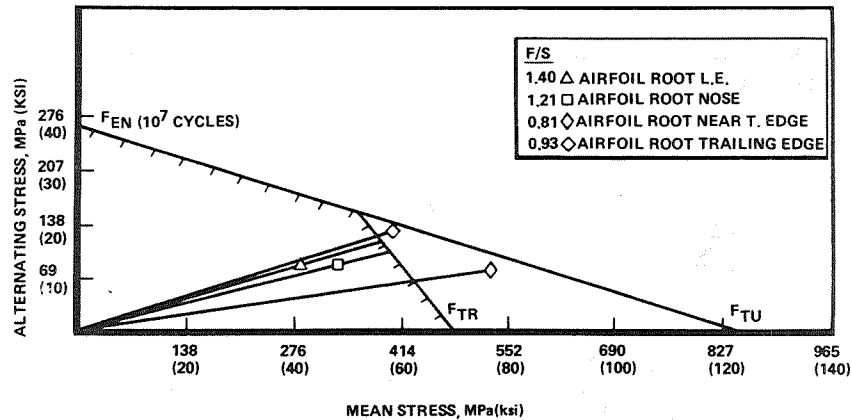


Fig. 7. MAR-M246 (Hf) (DS) Goodman Diagram at 870 C (1600 F)

Table 3. High-Cycle Fatigue Evaluation at 870 C (1600 F)

MATERIAL	HCF MINIMUM FACTOR OF SAFETY (FROM GOODMAN DIAGRAM)
MAR-M246 (Hf) (DS)	0.81
Si ₃ N ₄ CERAMIC	1.65
FRS COMPOSITE	1.24
SINGLE CRYSTAL SUPERALLOY	1.23
DS EUTECTIC SUPERALLOY	1.23
CM SX-2 SC SUPERALLOY	1.03
RSP SUPERALLOY	1.03
ODS SUPERALLOY	0.71

the coefficient of thermal expansion (α) and the difference between the bulk temperature and the temperature at a point. For a given geometry, the thermal transient temperature gradients were assumed proportional to the reciprocal of the thermal diffusivity of the material, which is a function of its thermal conductivity (K), its density (ρ), and its specific heat (C).

Using MAR-M246(Hf)(DS) for a reference, the relative thermal strains for the other materials were calculated as follows:

$$\frac{\Delta\epsilon_x}{\Delta\epsilon_{\text{MAR-M246}}} = \frac{\alpha_x \left(\frac{K}{\rho C} \right)_{\text{MAR-M246}}}{\alpha_{\text{MAR-M246}} \left(\frac{K}{\rho C} \right)_x} \quad (1)$$

A strain range of 3 percent was assumed for MAR-M246(Hf)(DS) and the corresponding strain ranges for the other materials were calculated using Eq. (1). Using these strain ranges and the low-cycle fatigue curves of Fig. 5 and the lower-bound curve of Fig. 6 for the FRS composite, a cyclic life for a temperature transient that causes a 3-percent strain in MAR-M246(Hf)(DS) was calculated. The calculated comparative cycle lives are given in Table 4.

Table 4. Thermal Fatigue Evaluation
870 C (1600 F)

MATERIAL	CYCLIC LIFE FOR A TEMPERATURE TRANSIENT WHICH CAUSES A 3% STRAIN IN MAR-M246 (Hf) (DS)
MAR-M246 (Hf) (DS)	24
FRS COMPOSITE	>1000
SINGLE CRYSTAL SUPERALLOY	240
CM SX-2 SC SUPERALLOY	130
DS EUTECTIC SUPERALLOY	120
RSP SUPERALLOY	120
ODS SUPERALLOY	65

Conclusions From Structural Analyses for 870 C (1600 F). Based on the results shown in Tables 2, 3, and 4, the single-crystal material appears very attractive for the 870 C (1600 F) turbine blade application. The ODS superalloy offers little or no improvement over MAR-M246(Hf)(DS) at this temperature. The RSP and DSE superalloys do provide improvements in capabilities over MAR-M246(Hf)(DS) but they

overall, do not measure up to the SC superalloy, and they are in earlier stages of development than the SC superalloys. The FRS composite also appears attractive for use at 870 C (1600 F) although there are open questions with regard to thermal fatigue capabilities. The FRS composite would require blade design modifications in the attachment area and the FRS composites are in an earlier stage of development than the SC superalloys. From stress-rupture and high-cycle fatigue considerations, the ceramic looks promising. However, its ability to withstand the thermal transients is of concern. The 870 C (1600 F) temperature is too low to require the unique high temperature capabilities of the ceramic and its development for this temperature application does not appear justified.

Structural Analyses for 1093 C (2000 F). A representative advanced high temperature staged combustion rocket engine design incorporating turbopumps for operation at approximately 1093 C (2000 F) was developed under a High Temperature Turbine Program (Ref. 8) conducted at Rocketdyne for AFRPL. Blade design and operating conditions for the structural analyses for the 1093 C (2000 F) temperature were taken from that program. The evaluation of the four candidate alloys for use in advanced rocket engine high temperature high pressure fuel turbopump turbine blades operating at 1093 C (2000 F) was directed at two modes of material failure--stress-rupture and high-cycle fatigue. The lack of comparative data or established analytical techniques precluded comparisons of thermal fatigue capabilities. This points up the need for appropriate thermal fatigue data for these materials. The procedures for comparing stress-rupture and high-cycle fatigue capabilities were as discussed for the 870 C (1600 F) temperature.

The comparisons of the candidate materials for stress-rupture and high-cycle fatigue for the 1093 C (2000 F) application are contained in Tables 5 and 6. It is obvious that the DSE materials do not have adequate properties for this application. Both the FRS composite and ceramic have excellent properties

Table 5. Stress-Rupture Evaluation
at 1093 C (2000 F)

MATERIAL	10 HOUR RUPTURE STRENGTH CALCULATED MEAN STRESS
Si ₃ N ₄ CERAMIC	3.48
FRS COMPOSITE	2.15
DS EUTECTIC SUPERALLOY	1.03
NiTaC-14B DSE SUPERALLOY	0.93

Table 6. High-Cycle Fatigue Evaluation
at 1093 C (2000 F)

MATERIAL	HCF MINIMUM FACTOR OF SAFETY (FROM GOODMAN DIAGRAM)
Si ₃ N ₄ CERAMIC	2.21
FRS COMPOSITE	1.34
DS EUTECTIC SUPERALLOY	0.82
NiTaC-14B DSE SUPERALLOY	0.73

and development of these materials for the 1093 C (2000 F) rocket engine turbine blade application appears justified.

Categorization of Materials. On the basis of the results of the turbine blade structural analyses, the candidate materials were categorized as follows:

1. Material warrants further development as a beneficial alternate material for the current SSME turbine blade design
 - a. Single-crystal (SC) superalloy--for use at 870 C (1600 F)
2. Material should be dropped from consideration
 - a. Oxide dispersion strengthened (ODS) superalloy, rapid solidification processed (RSP) superalloy, and directionally solidified eutectic (DSE) superalloy

3. Material warrants further development but design modifications would be required
 - a. Fiber-reinforced superalloy (FRS) composite--for use at 870 C (1600 F) and 1093 C (2000 F)
 - b. Ceramic--for use at 1093 C (2000 F)

Technical Development Plans

Based on the technology needs for each material, technical development plans, schedules, and costs were outlined for the material categorized as warranting development for use in advanced rocket engine turbo-pump turbine blades. The major technology needs identified for the materials warranting development are given in Table 7.

Table 7. Technology Needs

<ul style="list-style-type: none">• ADVANCED SC SUPERALLOY<ul style="list-style-type: none">• DEVELOP HIGH GRADIENT CASTING PROCESS• CONFIRM PROPERTY IMPROVEMENTS• ESTABLISH AS VIABLE COMMERCIAL PRODUCTION PROCESS• FRS COMPOSITE<ul style="list-style-type: none">• DEVELOP DESIGN DATA BASE• ESTABLISH PRODUCTION OF REQUIRED QUANTITIES OF HIGH-STRENGTH TUNGSTEN ALLOY WIRE• DEVELOP BLADE DESIGN TECHNOLOGY• ESTABLISH COMMERCIAL BLADE PRODUCTION PROCESS• CERAMIC<ul style="list-style-type: none">• IMPROVE RELIABILITY AND TOUGHNESS• DEVELOP DESIGN DATA BASE• DEVELOP NDE METHODS• IMPROVE DESIGN METHODOLOGY
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It was estimated that if technical development programs are implemented, these materials could be qualified for use in rocket engine turbine blades by 1990 for the advanced single-crystal superalloy, by 1991 for the FRS composite, and by 2000 for the ceramic. It should be noted that the technical development program outlined for the ceramic depended heavily on, and was built upon, the DOE/NASA plan

for developing an advanced terrestrial heat engine (Ref. 9), the so-called Ceramic Initiative Program.

Cost Benefit Analysis

A study was made to assess the cost benefits that can be realized by the use of advanced turbine blade materials for an advanced SSME-type engine. Cost benefit analyses were conducted for two areas: analysis of the cost savings that can accrue from extended blade life, operating at present temperature levels, i.e., 870 C (1600 F), and analysis of the increased income that will result from the increased payload that can be realized by operating the turbines at 1093 C (2000 F).

Increased Turbine Blade Life. The life of SSME high pressure fuel and oxidizer turbopump turbine blades at the present is approximately 3000 seconds. Development programs are underway that are expected to extend blade life to 5000 seconds by 1989 so this life was used as the basis for calculating cost benefits from increased life. The cost benefits result from decreasing the number of pump rebladings required. The cost of replacing the blades in an SSME turbopump including pump removal and reinstallation, reblading the turbopump, testing and propellants, and shipping is approximately \$1 million.

The average reblading cost per Orbiter mission or flight is shown as a function of blade life in Fig. 8. At the expected blade life of 5000 seconds, the average cost is \$600,000 per Orbiter flight. The steep slope of the curve in Fig. 8 for a blade life under approximately 10,000 seconds illustrates the significant gains to be made by relatively small increases in blade life.

Figure 8 also shows the total program cost incurred as a function of blade life. For the case of a five-Shuttle fleet with an Orbiter life of 100 missions each, the total cost for a blade life of 5000 seconds is \$300 million.

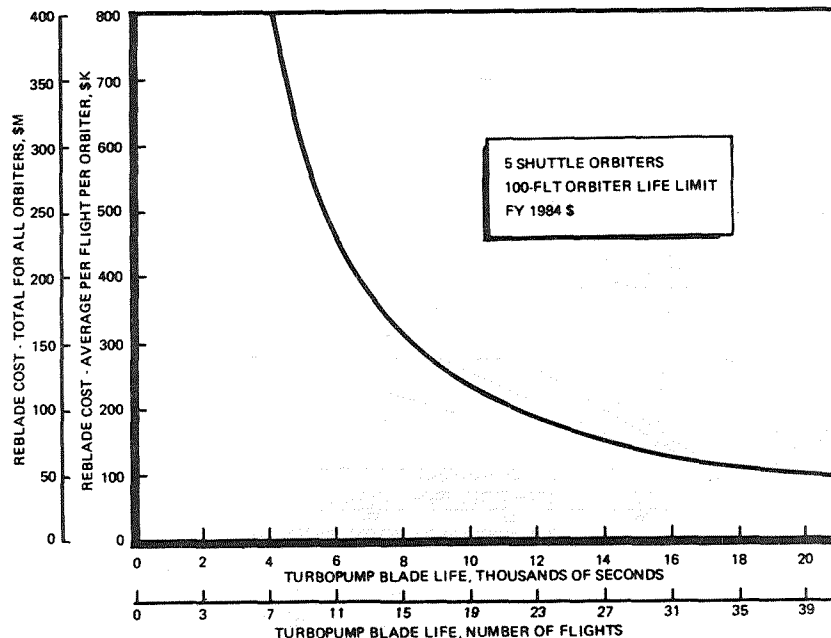


Fig. 8. Cost to Reblade Turbines as a Function of Blade Life

Figure 9 shows the maintenance cost savings realized by extending blade life from a reference value of 5000 seconds to various longer durations. Using the same case as above, i.e., at 500 missions, the gross cost savings in going from 5000 to 15,000 seconds blade life is \$240 million by reducing the total cost of reblading all engines of the Orbiter fleet from \$300 million to \$60 million.

Superimposed on the graph are cost levels that represent upper and lower estimates of the development effort. The material characterization and blade development costs in this example are set at \$10 million. This represents the lowest limit of development cost (i.e., just the technology acquisition) for turbine blade life extension using new blade materials. It assumes that turbopump and engine recertification would be charged entirely against other development or turbopump improvement program costs.

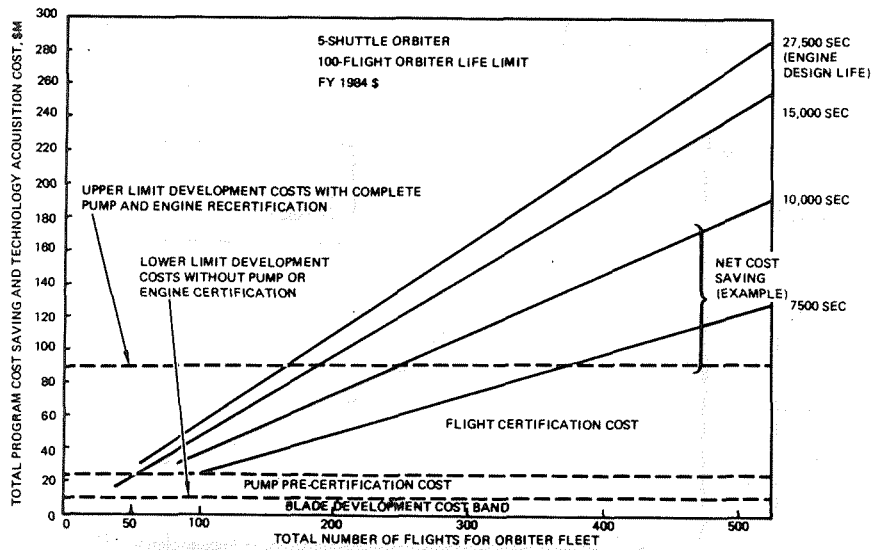


Fig. 9. Cost Saving Realized by Extending Blade Life From 5000 Seconds to Duration Shown

The upper limit of development costs for turbine blade life extension by new material substitution consists of the sum of blade development cost (\$10 million), turbopump precertification testing (about \$15 million), and engine flight certification (about \$65 million), i.e., a total of \$90 million. The development cost of \$90 million would be applicable only if turbine blade material substitution were the sole change in an advanced SSME-type engine requiring complete pump and engine recertification. However, it can be assumed that in a future advanced SSME-type engine, advanced turbine blade materials will be incorporated at the design stage, suggesting that the lower development cost level applies.

As shown in Fig. 9, extending blade life from 5000 to 15,000 seconds results in a gross cost saving of \$240 million for 500 flights (five Orbiters, 100 missions each). Subtracting the development cost band of \$10 million to \$90 million results in net cost savings of \$150 million to \$230 million. The

figure also shows that the development cost would be recovered after 25 to 190 Orbiter fleet flights, depending upon the development cost assumptions.

It is apparent that the estimated cost benefits resulting from extending life for the 870 C (1600 F) temperature are much larger than development costs for either the advanced single-crystal superalloy or the FRS composite.

Increased Turbine Operating Temperature. Raising the turbine operating temperature increases the Orbiter payload by increasing specific impulse. Specific impulse gains can be realized when the turbine operating temperature increase is accompanied by the incorporation into the engine design of advanced turbomachinery technologies (associated with higher turbopump speeds) and advanced thrust chamber cooling techniques.

For an advanced 500,000 lbf thrust, staged-combustion engine with oxygen/hydrogen propellants, with current engine technology, it is estimated that a specific impulse gain of 0.25 second is achievable if the preburner temperature is raised from 838 C (2000 R) to 1060 C (2400 R), Ref. 8. Larger gains can be realized if advanced technology in turbomachinery and thrust chamber cooling techniques are incorporated. For such engines, a 220 C (400 F) increase in preburner (or blade material) temperature is estimated to result in a specific impulse gain of approximately 0.75 second.

With a launch cost of \$1.2 million/thousand pounds of payload, based upon a Kennedy Space Center launch price for subsidized non-DOD, U.S. Government and commercial users, or \$1.9 million/thousand pounds for nonsubsidized launch costs, and a payload gain factor of 1100 pounds/sec of specific impulse, a cost benefit can be calculated. The total gross cost benefit equals the delta payload in thousand pounds, times the launch cost/thousand pounds of payload, times the total number of launches (500). The total number of launches has been established in

the groundrules as number of Orbiters (five) times number of flights per Orbiter life (100). The gross cost benefits are shown in Fig. 10 for the technology levels noted. The gross cost benefit attributable to a 200 C

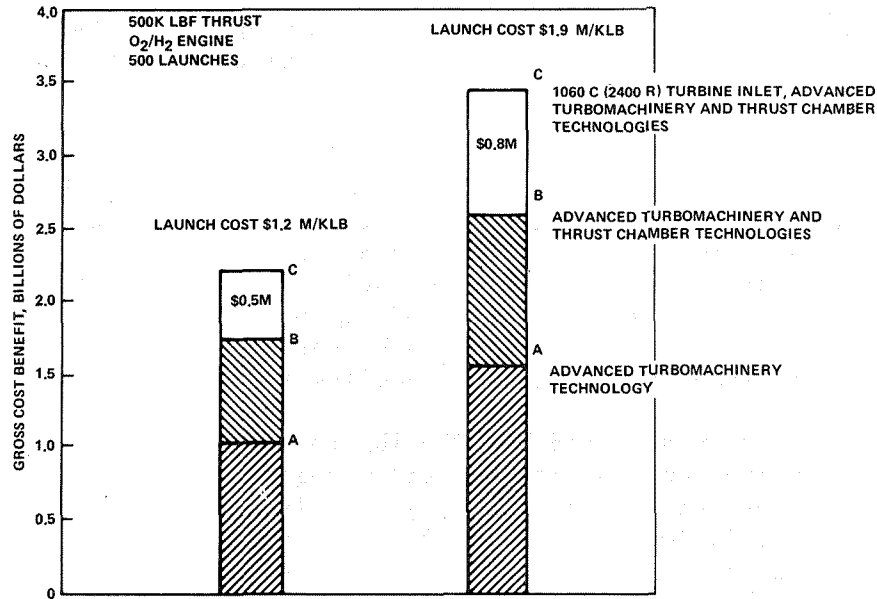


Fig. 10. Total Program Cost Benefit With Increased Turbine Inlet Temperature and Advanced TD Technologies

(400 R) increase in preburner temperature is from approximately \$500 million to \$800 million, depending upon the launch cost assumed.

From Fig. 10, it is apparent that large gross cost benefits can be achieved over the lifetime of a future Orbiter fleet if advances in turbomachinery and thrust chamber technologies are exploited fully in the next generation Orbiter engines. These gross cost benefits appear to justify the development of those technologies including the development of the FRS composite and ceramic turbine blades required to achieve the 1093 C (2000 F) turbine temperature.

Conclusions

A program was conducted to identify those materials that would provide the greatest benefits as turbine blades for advanced liquid propellant rocket engine turbopump turbines in the 1990 through 1995 period, and to prepare development plans for advanced versions of those materials. The conclusions are as follows:

1. The materials warranting development for advanced rocket engine turbine blades are the SC superalloy for use at 870 C (1600 F), the steady-state turbine blade temperature in the Space Shuttle Main Engine, the FRS composite for use at 870 C (1600 F) and 1093 C (2000 F), and the ceramic for use at 1093 C (2000 F).
2. Technology plans were outlined indicating that the time at which the materials could be developed through rocket engine certification and acceptance tests are 1989 for the SC superalloy, 1991 for the FRS composite, and 2000 for the ceramic.
3. Estimated cost benefits so far outweigh any anticipated material development costs, even factoring in development risks, that it was concluded that the programs outlined for the development of the SC superalloy, the FRS composite, and the ceramic for use in advanced rocket engine turbine blades are justified for implementation.

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