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APPLICATION OF ADVANCED COATING TECHNIQUES TO ROCKET ENGINE COMPONENTS

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ABSTRACT

The materials problem in the space shuttle main engine (SSME) is reviewed. Potential coatings and the method of their application for improved life of SSME components are discussed. A number of advanced coatings for turbine blade components and disks are being developed and tested in a unique multispecimen thermal fatigue fluidized bed facility at IIT Research Institute. This facility is capable of producing severe strains of the degree present in blades and disk components of the SSME. The paper summarizes the potential coating systems and current efforts at IITRI those are being taken for life extension of SSME components.

INTRODUCTION

The space shuttle main engine (SSME), a space transportation system, utilizes two preburners which supply hydrogen-rich combustion gases to drive the turbines of the main oxygen and hydrogen turbopumps. The service life requirement for all parts of the space shuttle is 40 flights, but to date, none of the blades made of MAR-M246 (Hf mod.) has been able to fly more than five or six flights. Efforts are therefore necessary to find a way to prevent the cracks and prolong the life of blades which, in turn, will extend the life of the turbopumps involved. This paper examines the extraordinary requirements in the SSME, the current materials in use for the turbine blades and disks, and the probable coatings which could extend the life of the component.

SPACE SHUTTLE MAIN ENGINE (SSME)[1-6]

The SSME utilizes two preburners which supply hydrogen-rich combustion gases to drive the turbines of the main oxygen and hydrogen turbopumps, as pictured in Figure 1. The gases after exhausting from the turbines are ducted to the main combustion chamber injection through the hot gas distribution manifold. Additional oxygen is mixed with the gases and burned in the main combustion chamber. Oxygen and hydrogen turbopumps are mounted directly to their respective preburners, and the

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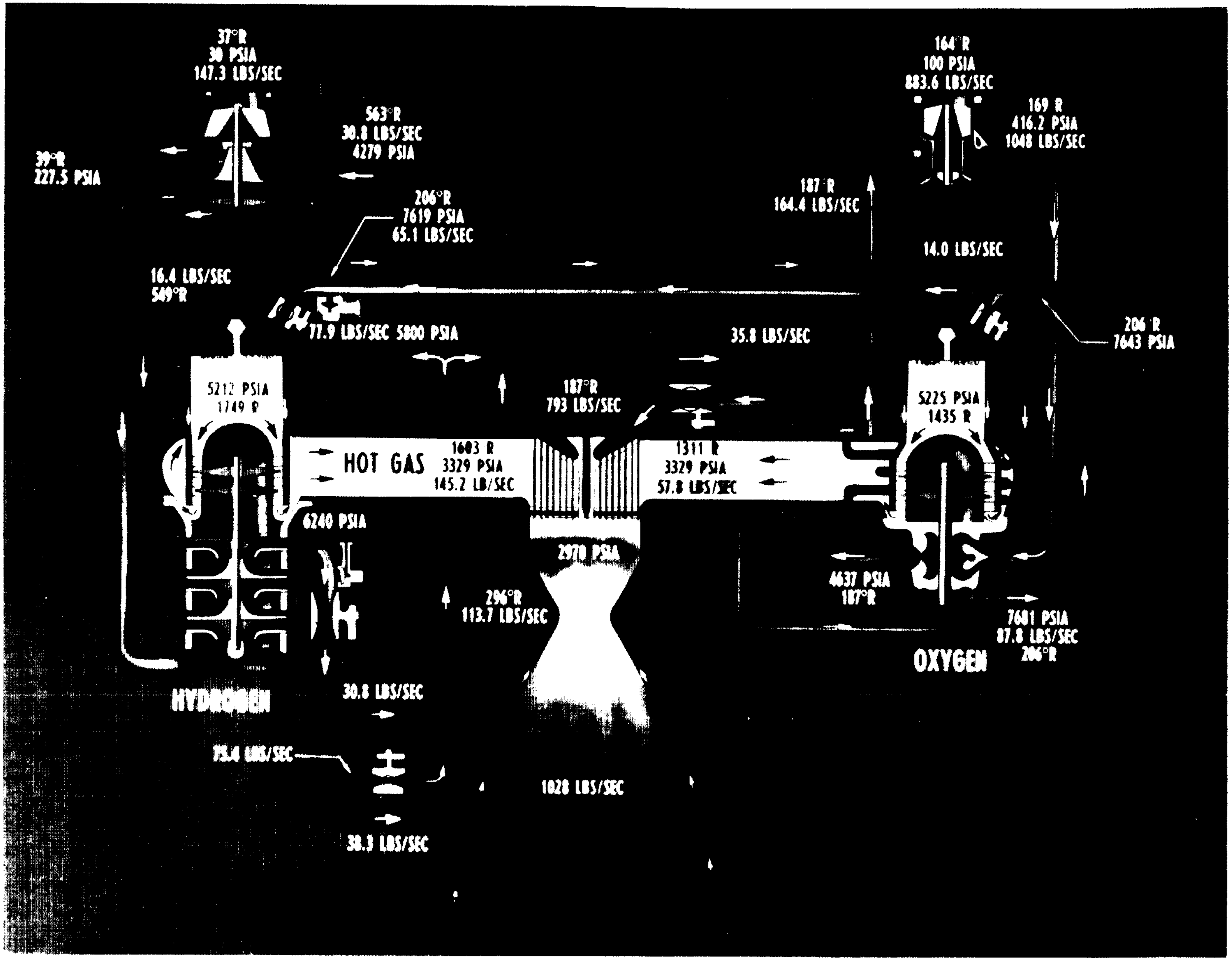

Figure 1. Space shuttle main engine. (Source: Rocketdyne.)

assemblies are supported by a hot gas manifold. The hot gas manifold and the preburners are cooled with liquid hydrogen, maintaining the structural members of the assembly at near ambient temperature. Other details on the SSME can be found elsewhere.[1]

The blades and nozzles are made from MAR-M246 (Hf mod.) material. The alloy composition is given in Table 1. The alloy possesses an optimum combination of high rupture strength and ductility, good creep resistance, and the highest static mechanical properties of nickel alloys from room temperature to 327°F (1800°C) as shown in Table 2.

First-Stage Blades[3]

The first-stage blades experience high-temperature spikes, but of short duration, during start-up engine transients. Gas temperature may reach 6000°F (3060°C), and durations of 0.425 s are common for spike existence. The actual temperature of the blade material is not known; however, incipient melting of the surface has been observed which indicates approximately 2240°F (1226°C). Strains in the blade material at these temperatures may reach 1.7% in./in., but analysis is still in progress. Although information on coating properties (e.g., shear strength or adhesive strength) is not available, simple calculations of shear stress with coating thickness indicate that it is preferable to keep coating thickness as thin as possible. A plot of shear stress as a function of coating thickness is presented in Figure 2 to illustrate the relationship of shear stress to coating thickness. The formula used here for the calculation of shear stress is as follows:

$$\text{Shear stress } (\tau) = \rho t \, R w^2 / g_c$$

where ρ = 0.252 lb/in.^{3} (density of coating)

R = 4.5338 in. (mean radius)

w = 39,000 rpm/60$\cdot 2\pi$ = 4084 rad/s (rotation speed)

g_c = 386 (gravity acceleration)

t = thickness, mils

If the bond strength of the coating is only 900 psi, then 10 mils of coating can be tolerated. However, depending on the bond strength, the thickness of the coating would vary as shown in Figure 2.[3] The temperature distribution in the shank area has been found to be critical as melting and cracking have been observed. Figure 3[3] indicates that a considerably thick coating would be necessary to reduce surface temperature to 1500°F (815°C) levels. This means that two types of coatings of different thickness, for the airfoil and for the shank area, would be required.

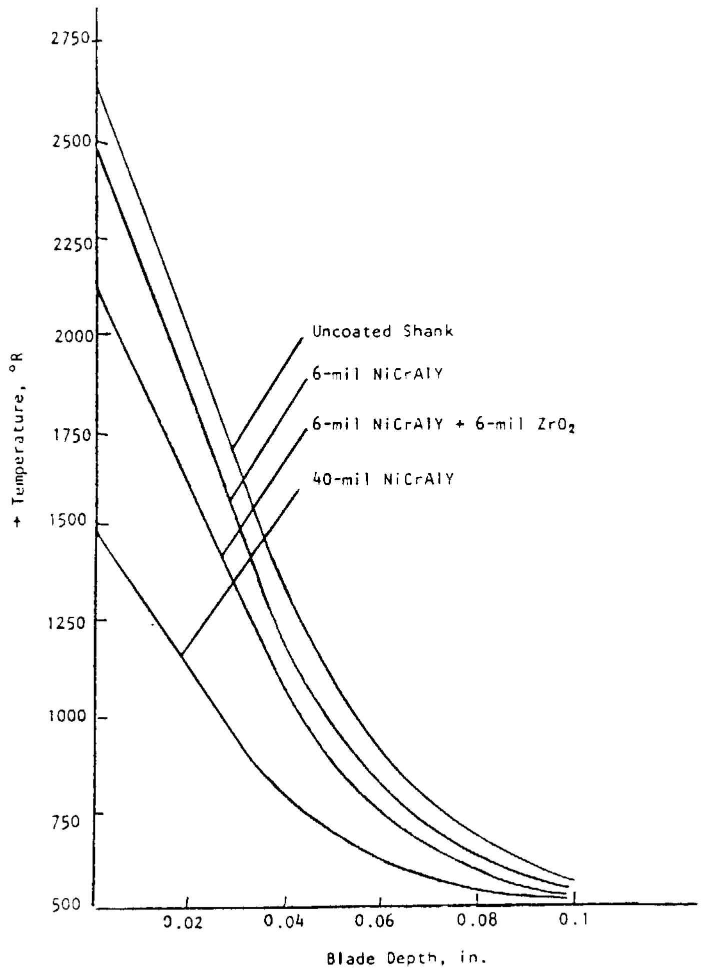

Figure 3. Temperature distribution in shank trailing edge. (Source: NASA-Marshall.[3])

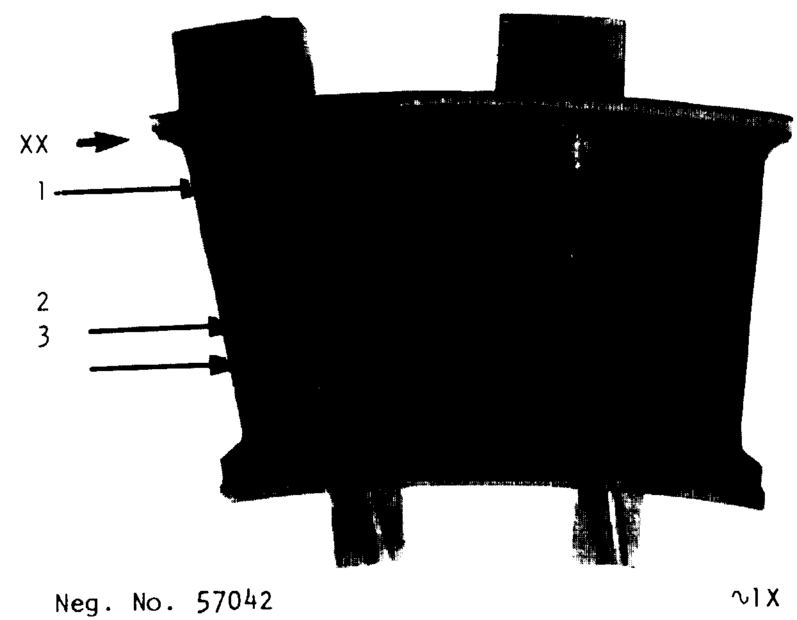

Figure 4. Photograph of the nozzle section received by IITRI for failure analysis.[1] (XX, 1, 2, 3 show approximate locations of cross-sectioning.)

Second-Stage Blades[3]

The second-stage blades are exposed to different conditions, being further downstream in the turbine exhaust. Most of the work in this case has been devoted to steady-state operating conditions due to the high-cycle fatigue cracking that has been observed in these blades.

Gas temperatures in the second stage reach 1300°F (704°C), and due to the discoloration of blades that have been hot-fired, metal temperatures are about the same. The thermal and mechanical stresses in these blades combine to produce mean stresses that approach yield in local areas. Strains are on the order of 0.5% in./in.

Current Blade Coatings and Their Problems[1]

The first-stage turbine blades are coated with NiCrAlY and a thermal barrier, yttria-stabilized zirconia. The second-stage blades do not have zirconia and have only thermally sprayed NiCrAlY. The coating, however, often spalls and reduces the service life of the blades. The details of coatings, which were derived mainly from aircraft blade coatings, can be found elsewhere.[1]

Disks and Their Coatings[1]

The disks are forged from Waspaloy X, which is thermomechanically processed (TMP) and heat treated (see composition and treatment in Table 3). The alloy has high strength and ductility at cryogenic and elevated temperatures, and has high strength-to-weight ratio. The alloy undergoes forging operations (the final 40 to 50% reduction) at temperatures below the gamma prime solvus. Upon subsequent heat treatment, only part of the residual strain (from low-temperature deformation) is relieved and the strengthening due to retained strain and precipitation hardening is synergistic. Notch rupture ductility is greatly improved because the resultant microstructure consists substantially of uniform deformed grains free from equiaxed recrystallized structure. The strengthening is accomplished only through heat treatment. The turbine disk is gold plated to protect the substrate from hydrogen embrittlement, but spalling and tearing of the gold plating have been observed.

FAILURE ANALYSIS OF SSME COMPONENTS[1,3]

Metallurgical Examination of a Nozzle

A macro-optical picture of a turbine nozzle section from the SSME received by IITRI is shown in Figure 4.[1] The leading edge of the nozzle was severely damaged, as illustrated. A closer examination of the leading edge by scanning electron microscope showed an uneven surface with large protrusions (Figure 5a). The surface appeared molten and severely deformed. Cracking and separation of grain boundaries were also apparent on the surface (see Figure 5b). The extensive flow of material near the grain boundaries suggested a molten state for the surface, as shown

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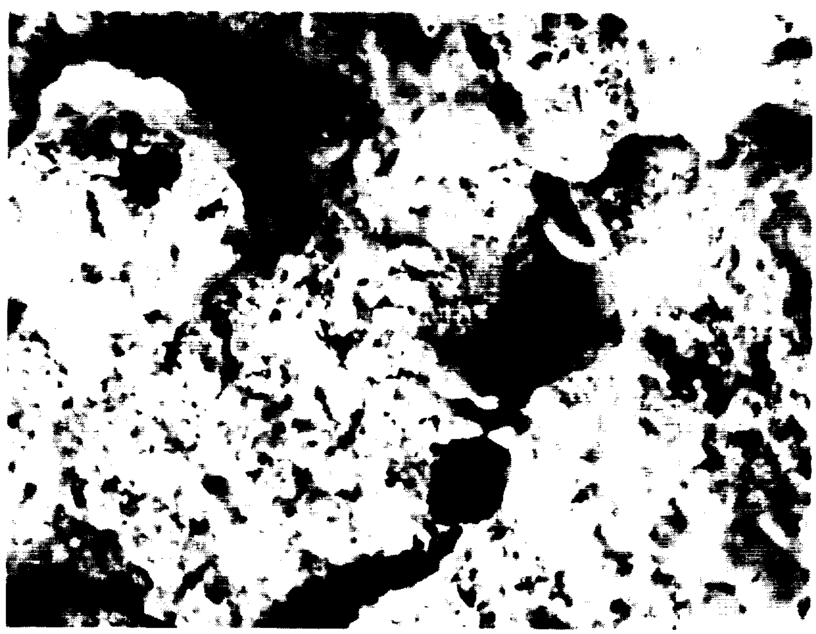
(a)



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(b)

Figure 5. Scanning electron micrographs of the leading edge of the nozzle shown in Fig. 4. (a) Surface melting and "protruded" surface of nozzle, (b) cracks in outer surface of the nozzle.

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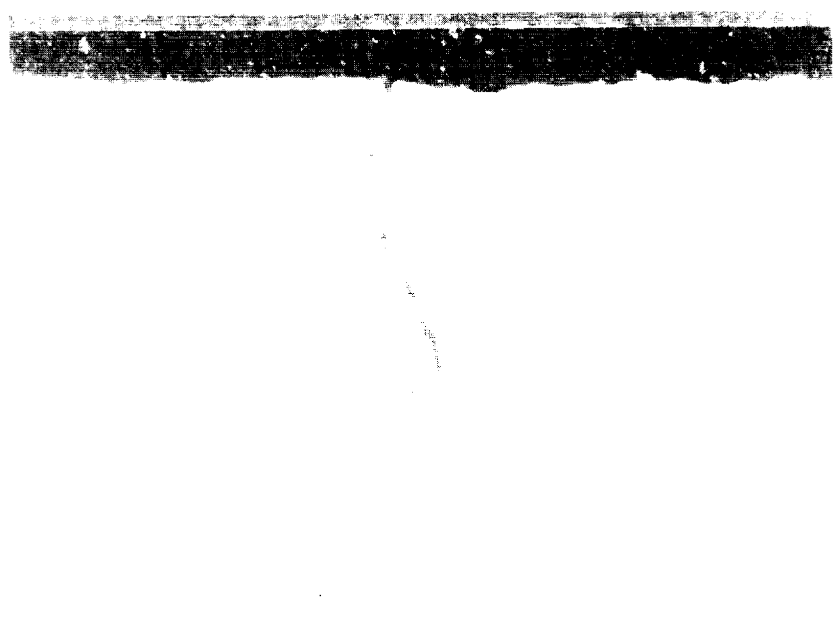
(a)



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(b)

Figure 6. Photographs showing cracks on the side walls of the nozzle (see xx line in Fig. 4.) (a) SEM photograph of the surface showing a number of cracks, and (b) optical photograph of the cross-section of the side walls showing the propagation of cracks along the carbide network of the base alloy.

in Figure 5b. The nozzle sample was cut using conventional blade material, but part of the cutting of the leading edge was done in a slow diamond cutter in order to prevent artifacts in the microstructure of the damaged layers.

The nozzle side walls showed a series of cracks which propagated along the carbide network of the alloy (Figures 6a and b). The optical cross-sections taken in the interior of the nozzle showed sound metal, but a few pores decorated by "oxide rims" were also observed.

Further optical examination of the cross-sections of the nozzle leading edge revealed the severity of the damage in this area. In the middle of the nozzle leading edge, a large amount of material was molten, and porosity near the primary MC carbides was also observed (Figure 7a-d). The presence of "casting holes" near the outer surface supported surface examination revealing the near-molten state of the nozzle surface. Also, cracks at the grain boundaries and near the MC carbides and eutectic gamma prime precipitates were observed by the optical microscope as shown in Figure 7.

Earlier work on uncoated turbine blades was aso undertaken by Rocketdyne.[3] The main cracking problems associated with high pressure fuel turbopump first-stage turbine blades were the following:

<u>Airfoil</u>

Radial

Transverse leading and trailing edge

<u>Platform</u>

Intergranular and interdendritic

<u>Shank</u>

Radial

Transverse leading edge

<u>Fir tree</u>

Transverse bottom hole.

Previous failure examinations of turbine blades, summarized in Table 4, resulted in many corrective measures in coating selection for SSME components.[1-6]

A schematic representation of radial airfoil cracks is shown in Figure 8. The intergranular cracks were caused by thermal fatigue and were eliminated by application of blade coatings. The platform cracks were both intergranular and interdendritic and were also caused by thermal fatigue. A typical schematic representation of such cracks is shown in Figure 9. Additionally, there were signs of overburnt areas in the

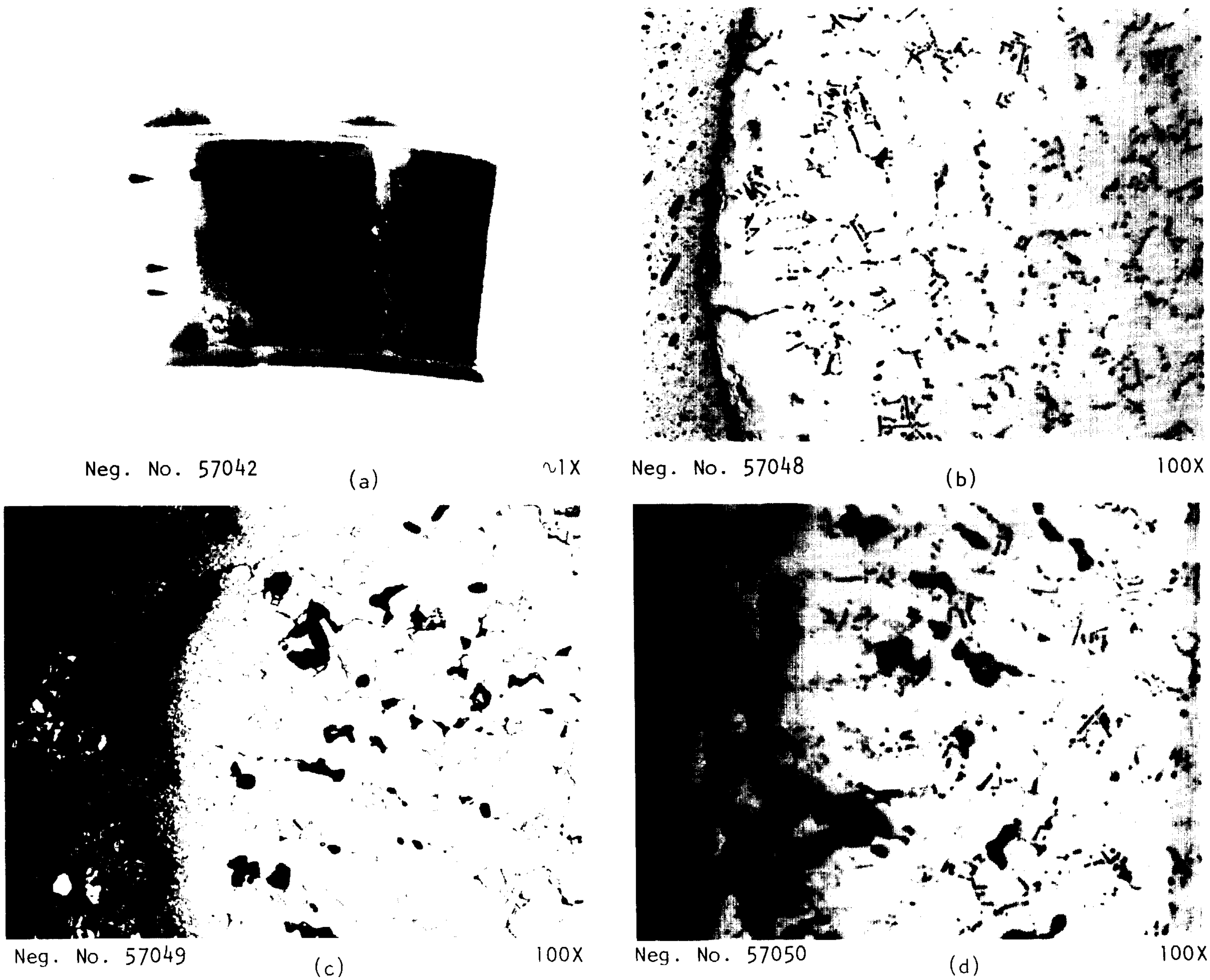


Figure 7. Optical photographs of the nozzle blade section and various cross-sections at three locations. (a) Cross section of disk, (b) 1, (c) 2, and (d) 3.

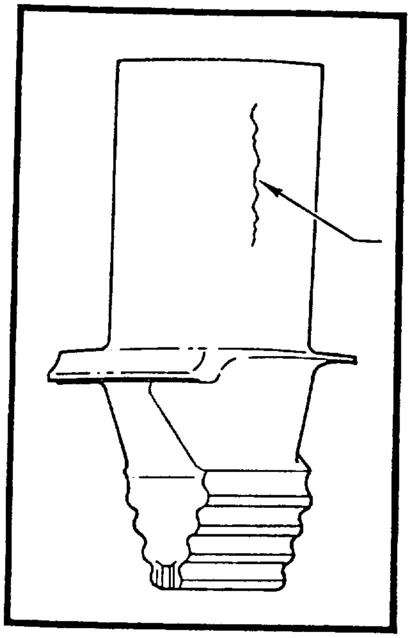


Figure 8. Schematic representation of radial airfoil cracks in first-stage blades. (Source: Rocketdyne.[3])

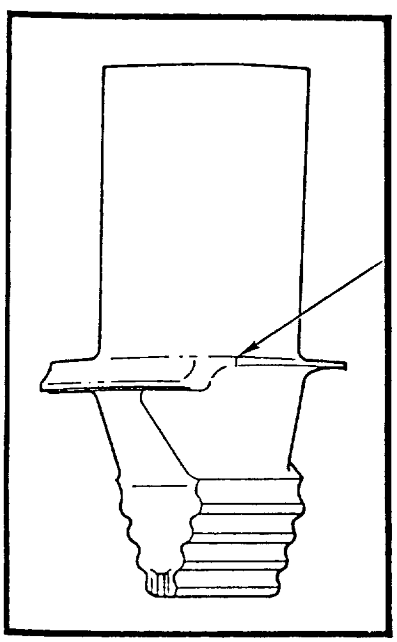


Figure 9. Schematic representation of platform cracks in first-stage blades. (Source: Rocketdyne.[3])

shank or fir-tree sections suggesting a localized burning of trapped, localized oxygen-enriched mixture.

Turbine Disk[3,4]

The turbine disk is gold plated to provide resistance to hydrogen diffusion. In initial trials, however, gold was missing locally after hot firing or even damaged during fabrication or assembly. Possible causes postulated for such cases were as follows:

High current density strikes
- fir-tree edges

Lifting of maskings
- rinsing problems

Gold alloys with Ni strikes during blister test and hot firing
- gold stripper does not remove Au-Ni
- weak bond to Au-Ni

Broaching smears substrate surface

Thick gold coating with weak bond peels at operating conditions

Mechanical damage.

In order to limit gold exfoliation, specifications for gold plating were rewritten in which trace impurities were limited, solutions were required to be filtered, and strike initiation was controlled according to specification RA1109-009.[4] Measurement of gold film thickness, however, varied from place to place on the same component, and it was agreed that the maximum gold thickness for oxidizer disks must be increased (now 0.0025 in.) to achieve 0.002 in. minimum.

The problem was quite severe since the properties of Waspaloy TMP in hydrogen are adversely affected as shown in Figure 10. The elongation and low cycle fatigue data clearly show the importance of gold coating. This affects the allowable speed of the disk as shown in Figure 11.

Analysis work at NASA-Marshall[3] has shown that a gold plating thickness of 0.002 in. is indeed required as it acts as a barrier to hydrogen diffusion (D): D(H_2) in Au or D(H_2) in Pt = 3.4×10^{-9} cm^2/s. The effectiveness of gold plating, however, is reduced by porosity. Bubbles form on the surface during the plating process, but such conditions can be avoided by agitation during plating. During heat-up, however, the dissolved hydrogen migrates to the coating/metal interface and debonds the gold plating from the substrate.

Also, it is suggested that centrifugal force on the gold plating and thermal stresses due to difference in thermal expansion between

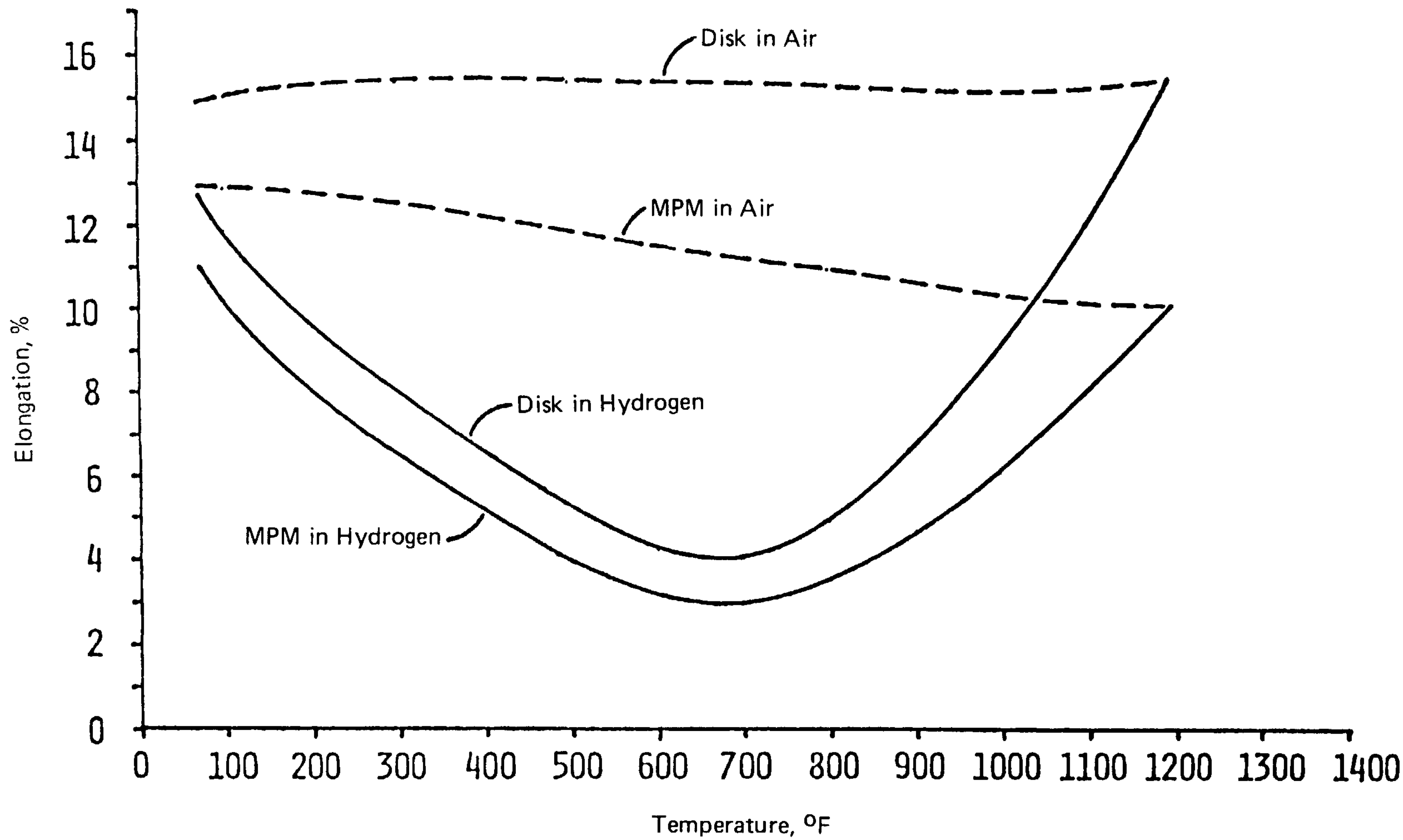


Figure 10. Ductility (measured as elongation of Waspaloy as a function of temperature in air and in hydrogen. (Source: Rocketdyne.[3])

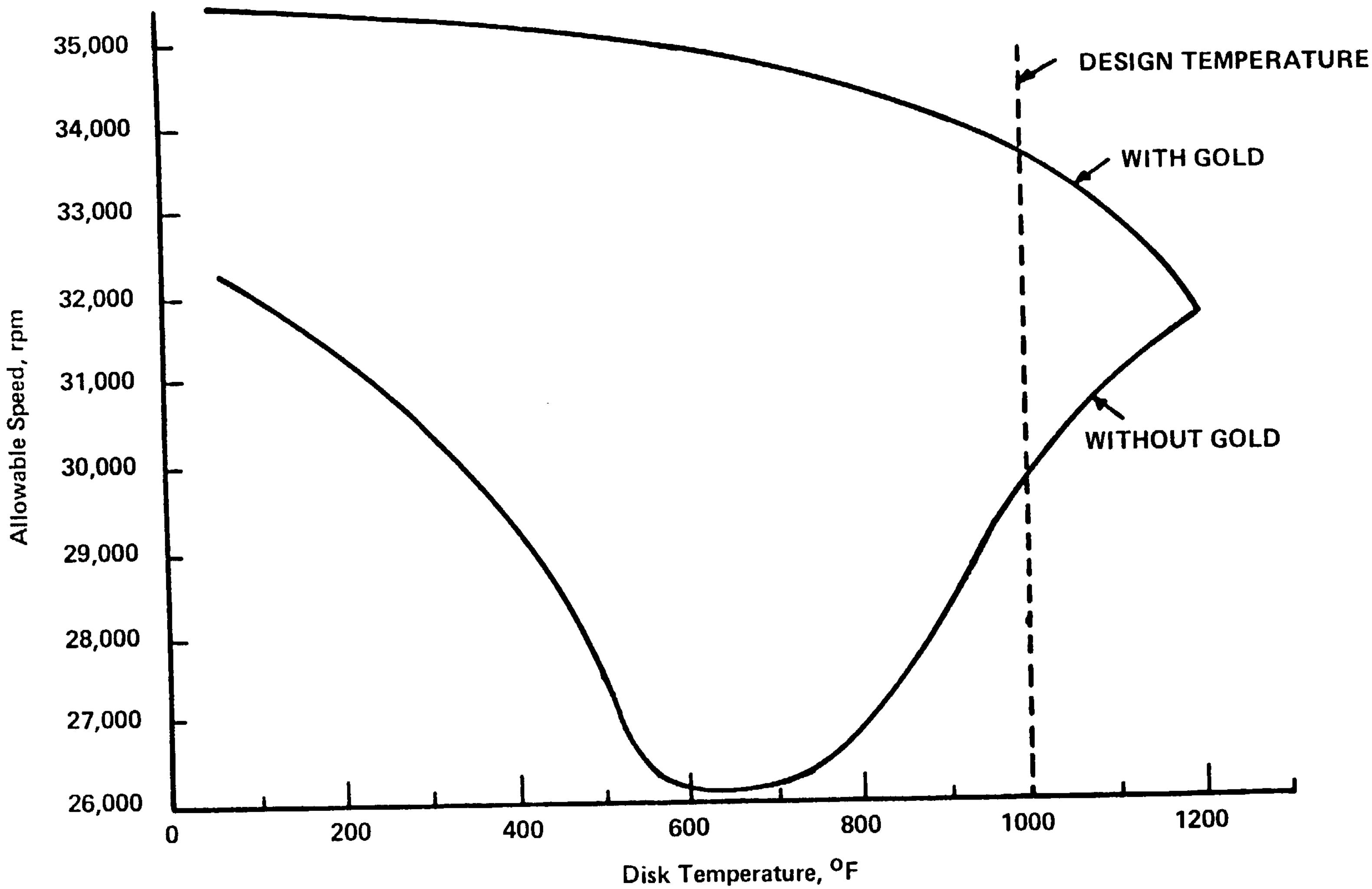


Figure 11. Allowable speed with or without gold for the first-stage disk at various disk temperatures. (Source: NASA-Marshall.[3])

gold and Waspaloy are the main source of stresses, which can exceed the shear strength of gold or the bond strength between gold plating and Waspaloy. Thermal stresses of the order of 13,000 psi at 1200°F (649°C) will be developed which exceed the shear strength of gold. This number does not include thermal stress due to temperature gradient.

IITRI's analysis is in line with NASA-Marshall results[3] in that baking of parts after electroplating will not remove all hydrogen trapped in the coating, and further voids in the coating left by exiting hydrogen will act as short-circuit paths for hydrogen entry. It is important to increase bonding between the coating and the disk for durability. Any metallic impurities of the plating have been known to cause exfoliation of gold from the surface and must be minimized.

Not only is the surface finish of the base metal important, but also maintenance of all of the pretreatment systems prior to the use of a gold electroplating system. Water that is used in rinsing should also be clean.

Developmental Work at NASA-Marshall for Turbine Blade Coatings

Some developmental work was carried out at NASA-Marshall[5] to test a number of coatings, but cracks appeared on almost all coatings. The coatings ranged from pure alloy to advanced gas turbine coatings. Many variables were included in the test program, but none of the coatings survived the testing. It should be stated, however, that test conditions in the laboratory were very severe compared to actual operation.

In these tests, the thermal cycle was characteristically a nominal maximum chamber temperature of approximately 1700°F (927°C) and a nominal maximum chamber pressure of approximately 2500 psi, down to a nominal minimum chamber temperature of -350°F (-212°C) and a nominal minimum chamber pressure of 160 psi. Figure 12 is a plot of a high, a nominal, and a low chamber temperature seen during the course of the testing. For simplicity, two of the schematics prepared by NASA-Marshall are shown in Figure 13 illustrating the type of cracking. In fact, all the coatings of various compositions applied by conventional methods had cracks of the type shown in Figure 13.[5]

Analysis Summary

First-Stage Blades

Thermal transient temperature	2240°F (1226°C)
Strain	1.7% in./in.

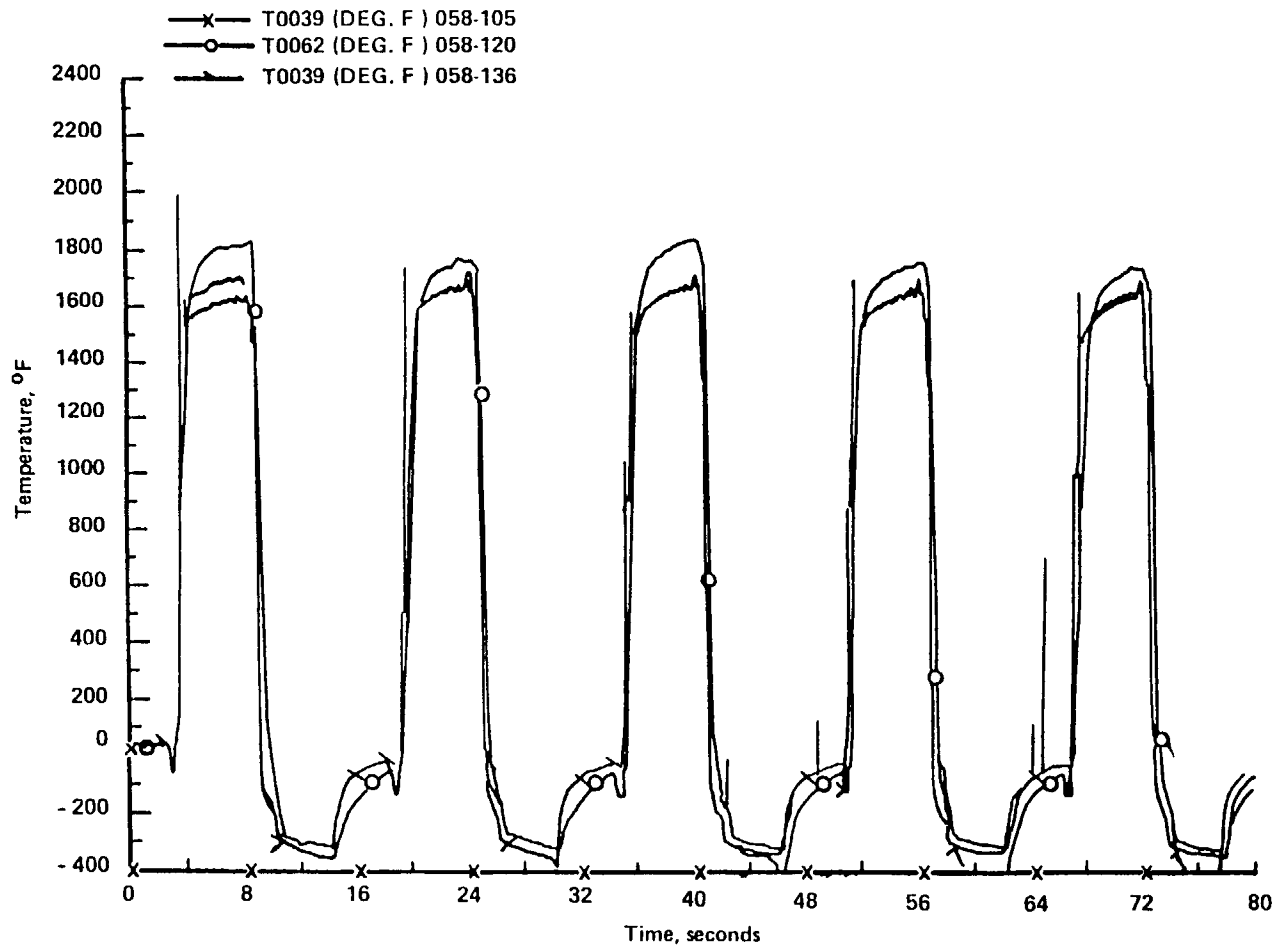
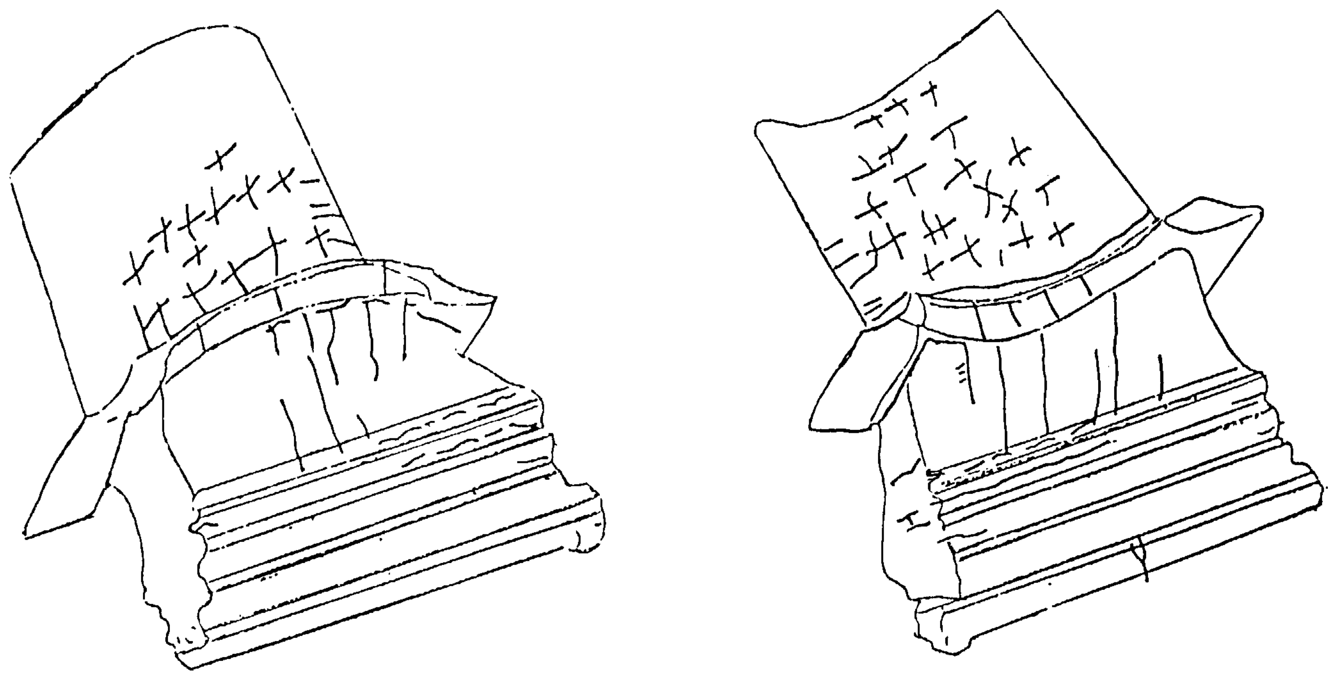


Figure 12. Variations in chamber temperature in NASA-Marshall's thermal fatigue rig.[5]



0.003'' NiCrAlY Undercoat (-400 mesh)
0.004'' Cr_2O_3-50NiCrAlY (-400 mesh)

(a) (b)

Figure 13. Schematic representation of cracks formed on (a) concave and (b) convex sides of turbine blades tested in NASA-Marshall's thermal fatigue rig simulating the operation of space shuttle main engine.[5]

Current coatings	NiCrAlY Cracks and spalling
Melting/cracking	Observed on first-stage blades without coating and in shank area. A considerably thicker coating can be adopted for shank area.

Second-Stage Blades

Blade temperature	1300°F (704°C)
Strain	0.5% in./in.
Current coating	MCrAlY - cracks
Some discoloration	In shank areas discoloration is observed. Cracking was observed even in newly recontoured blades.

PREVENTIVE METHODS AND TESTING OF COATINGS FOR SSME COMPONENTS

Preventive Measures[6]

Analytical work has led to the discovery of two causes for the cracks in turbines. One involves a temperature and start shock on the blades in the hydrogen pump. Just prior to start of the engine, liquid hydrogen at -423°F (-252°C) is allowed to flow through the pump. Though no hydrogen flows into the turbine disks and onto the blades, it is extimated that the blades are at about -300°F (-184°C).

At the start of the engine, and within 4 s, the temperature of the blades changes from -300°F (-184°C) to close to +1500°F (815°C). This produces a temperature shock and apparently induces a surface crack just under the thin platform that separates the airfoil and the shank of the blade.

A second shock, which can affect other blades, is apparently caused by a thermal environment,[6] and involves hot fuel-rich steam colliding with cold hydrogen--again, just under the platform. The steam crosses horizontally along the platform, but tends to leak under the platform to flow around the edge of the shank. As it reaches the machined and as-cast radii, the steam collides with cold hydrogen.

The surface crack here has been analyzed as a "temperature discontinuity," where the steam is about 1500°F (815°C) and the hydrogen is -100°F (-73°C).

A three-step procedure has been proposed which could eliminate the possibility of the blades developing the cracks, regardless of the type of shock or temperature difference.

The first step in the procedure is to hand-finish or smooth out the junction point where the machined radius meets the as-cast radius. This will eliminate any possible point where the temperature difference will find a leverage to induce stress in the surface. The hand finish is to "spread" the stress load on the blade in the thermal environment. Once the hand-finishing is completed, the shank of the blade, just beneath the platform but above the fir tree, is shot-peened to provide a "residual compressant" to increase the strength of the shank.

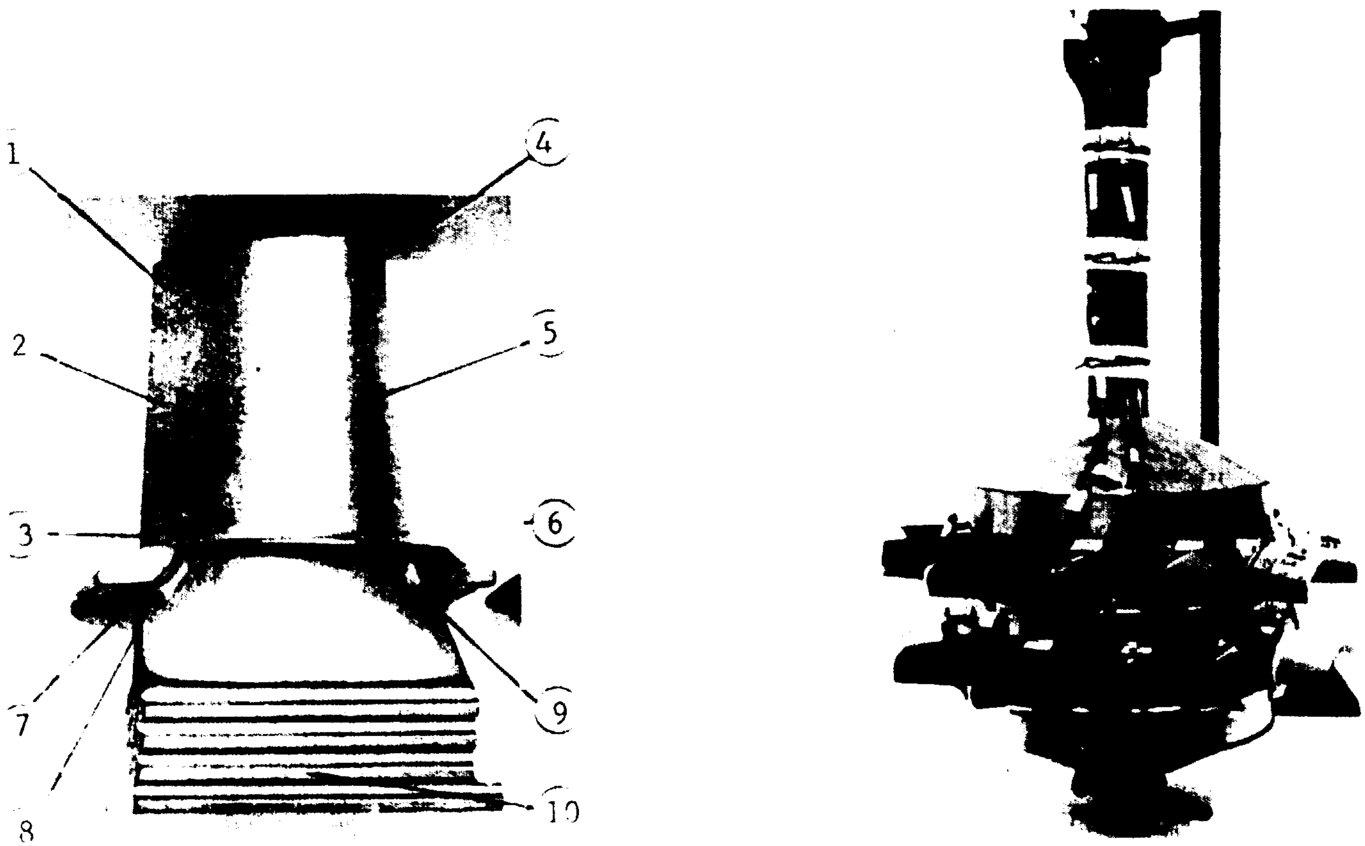
In addition to the hand-finish and peening, a two-step coating of the shank is being sought. The first step is to apply a very thin coating of "NiCrAlY" plasma coating, which is a combination of nickel, chromium, aluminum, and yttrium. A second coating, again with NiCrAlY combined with a like amount of zirconium dioxide, is applied to an area just under the platform where the machine and as-cast radii come together. These coatings could provide relief from the temperature discontinuity at that point.

Coating Testing for SSME Application[1]

IITRI has compiled comprehensive reports on coating and potential coatings which can be used for durability of SSME components.[1] IITRI has conducted thermal fatigue testing for over 20 years which includes testing for turbine applications in the space shuttle.[7] A total of 11 plasma spray coated and 13 uncoated directionally solidified MAR-M246 alloys, including single-crystal MAR-M246 blades, were tested for possible application in the SSME. Blade coatings on the airfoil tested at IITRI included several metal/oxide thermal barrier layers based on Al_2O_3, Cr_2O_3, and ZrO_2 as shown in Table 5. The 24 turbine blades were tested simultaneously for 3000 cycles in fluidized beds maintained at 1745° and 78°F (952° and 25°C) using an asymmetrical 360-second thermal cycle. Figure 14 shows a typical blade tested and the thermocouple installation to monitor temperature changes.

Almost all blades showed microscopically visible spalling, particularly at the radius of the trailing edge. Coatings for No. 4, multilayer Ni-20Cr + (ZrO_2-5CaO), and No. 6, multilayer Ni-20Cr + 30(Ni-20Cr)-70(ZrO_2-5CaO) + (ZrO_2-5CaO), showed extensive spallation after 3000 cycles.

The thermal cracking at the base of blades can be explained as follows. A significant contribution to thermal fatigue results from geometry changes of a component. Generally, thermal fatigue failure results from the imposition of a tensile stress on a thin section by adjoining thick sections of a component or specimen. Aircraft turbine blades designed for minimum airfoil mechanical stresses have relatively little difference in cross-section in the gas path region, i.e., on the airfoil. For this reason, small aircraft turbine blades are resistant to thermal fatigue failures. When thermal fatigue cracking does occur, it tends to localize at the thinnest blade section with



Neg. No. 48585 1.5X

(a) Thermocouple locations

Neg. No. 48651 0.5X

(b) Thermocouple installation

Figure 14. Experimental design for calibration tests at IITRI showing attachment of ten thermocouples in a NASA-funded program.[7]

the greatest thermal gradient (i.e., at the trailing edge) near the blade root. Thermal cracking in the upper blade sections is difficult to obtain in testing because of the inability to develop significant thermal gradients in the thin blade cross-section.

One of the most significant findings of IITRI's study was that etched directionally solidified (D.S.) blades indicated measurably lower oxidation in 3000 cycles than unetched D.S. or single-crystal blades.[7] Whether this observed effect would have continued on further thermal cycling is not certain. Since thermal fatigue is known to be a structure-sensitive property, a fine homogeneous structure would be helpful. Etching would ensure uniform properties at the surface which could help delay the surface cracks and thus improve the thermal fatigue properties.

The fact that the high-pressure hydrogen turbopump in the SSME develops more cracks than the high-pressure oxygen turbopump suggests not only that thermal fatigue properties are affected by thermal shock by the mechanism explained here, but that hydrogen entry into the substrate could also be important.

NASA-Marshall has also conducted coating development in a specialized thermal fatigue rig simulating the SSME conditions in its laboratory which also suggested that coatings normally applied on gas turbine blades are not good for SSME application and further development of coatings is necessary for strain tolerance.

CURRNET PROGRAM ON COATING SYSTEMS

The analysis of many studies cited here has revealed that coatings currently used for aircraft engines would not have adequate lifetime and further development work is necessary.

In thermal barrier systems, both two- and three-layer concepts are being be explored. An additional outer layer of aluminides on top of zirconia could change the stress distribution and would fill in the cracks due to oxidation of the reactive element aluminum.

For thermal barrier systems, attractive coatings being studied are:

(1) Basic two-layer coatings (electron beam, ion plating and plasma-sprayed coatings), for example:

- MCrAlY or MCoCrAlY-$ZrO_2 \cdot 8\%\ Y_2O_3$
- MCrAlY-$ZrO_2 \cdot 6\%$ MgO
- MCrAlY$\cdot 25\%$ $MgOZrO_3$
- MCrAlY-$ZrO_2 \cdot 6\%\ Y_2O_3$(0.5 Sr) (very low diffusivity of Sr in ZrO_2)
- MAlYTa-$ZrO_2 \cdot 8\%\ Y_2O_3$

(2) Three-layer coating, for example:

- MCrAlY or $ZrO_2 \cdot 8\% \ Y_2O_3$ with an outer Al layer and heat treated to oxidize aluminum to form Al_2O_3

(3) Surface modified coating, for example:

- basic two-layer coating with Y or Si ion implanted to change fracture mode
- MCrAlY with a thin layer of $ZrO_2 \cdot 8\% \ Y_2O_3$ (500 Å), and ion mix the top layer
- MCrAlY with a thin layer of $ZrO_2 \cdot 8\% \ Y_2O_3$ (500 Å), ion mix the top layer and deposit further layer of $ZrO_2 \cdot 8\% \ Y_2O_3$ with plasma gun

(4) Graded coatings, for example:

- coat MCrAlY and $ZrO_2 \cdot 8\% \ Y_2O_3$ initially under ambient pressure and reduce the pressure in the coating chamber from ambient to low pressure during coating application of $ZrO_2 \cdot 8\% \ Y_2O_3$. (This would give density graded coatings.)
- vary chromium and aluminum content in the MCrAlY coatings with chromium being maximum and aluminum being minimum in the outer edge.
- laser glaze the atmospheric sprayed coatings for graded density.

The conditions for coating application should be carefully chosen from those available in the existing literature. The coatings should preferably be applied by vendors having qualifications to do such work for the gas turbine industry. In IITRI work, coatings are applied either on the flat coupons or on the turbine blades themselves and tested in IITRI's fluidized bed facility shown in Figure 15. A number of blades can be tested simultaneously in the fluidized bed.

For second-stage turbine blades, both MCrAlY and aluminides are promising and should be further tested. For MCrAlY, both low- and high-chromium containing coatings can be tested at low temperature. Typical candidate coatings for second-stage blades include

- MCrAlY
- MCoCrAlY
- MCoCrAlY-Y or Si ion mixing
- MCoCrAlY + sputtered aluminum

- aluminide (inward diffusion coating)
- modified aluminide with some platinum in the pack)
- plasma sprayed nickel aluminide.

For the shank areas, where discoloration and high temperature hot spots are noticed, application of MCoCrAlY or MCrAlY with or without thermal barrier is desirable but surface finish is important in all cases. The recommended post-coating heat treatment is 1976°F (1080°C)-4 h in hydrogen/oxygen for all overlay coatings discussed here.

For turbine disks, gold or platinum coating appears to be the most suitable, though as indicated in the analysis section, bonding may remain a problem. Ion mixing of gold after gold plating or gold sputtering would indeed increase the adhesion significantly. Previous experience in another application of gold coating on glass was that the coating could be scratched by the thumb nail, but the same coating was difficult to scratch after it had been ion bombarded.[1] Some difficulty with line of sight may be present, but a suitable holder rotating in three dimensions with respect to the ion beam should be helpful. In summary, the testing possibilities are:

(a) gold plating (base)

(b) platinum plating

(c) gold sputtering + ion mixing

(d) same as (c) + electroplated gold as in (a).

SUMMARY

(1) Conventional aircraft turbine blade coatings are not suitable for SSME application, and further development work necessary to increase the strain tolerance of coatings is being carried out at IITRI and MFSC.

(2) Both delay in crack initiation and controlled crack propagation in coatings should be sought through the control of metallurgical microstructure.

(3) Advanced methods of ion plating or ion implantation are likely to produce adherent coatings necessary for the turbine disk application.

ACKNOWLEDGMENT

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TABLE 1. MAR-M246 (Hf MOD.) CHEMISTRY[1,2]

Element	Percent
Nickel	58.0
Chromium	9.0
Cobalt	10.0
Tungsten	10.0
Titanium	1.5
Tantalum	1.5
Aluminum	5.5
Molybdenum	2.5
Hafnium	1.75
Carbon	0.15

TABLE 2. MAR-M246 (Hf MOD.) PHYSICAL PROPERTIES[1,2]

Type	Incipient Melting, °F (°C)	Density, $lb/in.^3$	Modulus at 1200°F (649°C) (longitudinal direction), 10^6 psi	Poisson's Ratio	Thermal Expansion at 1500°F (815°C), in/in/°F
Directionally solidified	2230 (1221)	0.308	15.8	0.313	7.5
Single crystal	2230 (1221)	0.308	15.8	--	7.5

TABLE 3. CHEMICAL COMPOSITION[1,2] OF WASPALOY X[a]

Element	Analysis, %
Carbon	0.015-0.04
Manganese	0.10 max
Silicon	0.15 max
Phosphorus	0.015 max
Sulfur	0.015 max
Chromium	18.00-21.00
Cobalt	12.00-15.00
Molybdenum	3.50-5.00
Titanium	2.75-3.25
Aluminum	1.20-1.60
Boron	0.003-0.010
Iron	2.00 max
Copper	0.10 max
Zirconium	0.02-0.08
Nickel	Remainder

[a]Typical heat treatments are: Solution heat treatment: heat at 1800°F (982°C) for 2 1/2 h and air cool. Stabilization heat treatment: heat to 1550°F (843°C) for 4 1/4 h. Precipitation heat treatment: heat to 1400°F (760°C) for 16 1/4 h and air cool.

TABLE 4. HISTORY OF PREVIOUS FAILURE EXAMINATIONS[3]

	First-Stage Blades				Second-Stage Blades			
Component	Type of Crack	Microstructural Relationship	Cause	Solution	Type of Crack	Microstructural Relationship	Cause	Solution
Airfoil	Radial	Intergranular	Thermal fatigue	Coating	Radial as 1st stage	Intergranular cracks	Thermal fatigue	Blade coating
	Transverse (leading edge)	Transgranular	Fatigue High cycle fatigue caused by thermal spike near	Modify engine start				
	Transverse (trailing edge)	Transgranular	High Cycle Fatigue (Damper blade - bonding)	Plating of damper				
Platform Cracks	Radial	Intergranular and interdendritic	Thermal Fatigue		Radial as 1st stage (some of platform was lost)	Intergranular, interdendritic, crystallographic (one crack was transgranular)	Thermal fatigue, also HCF	Change to precision damper
Shanks	Radial	Intergranular (shallow)	Thermal Fatigue		Radial crack in shank		Thermal fatigue	
	Transgranular (leading edge)		High Cycle Fatigue		Transverse trailing edge	Geometrical risers initiation, high mean stress from thermal gradient, discoloration	Fatigue in geometrical risers and thermal gradient	Grit blast to improve contours at stress risers
Fir tree	Transverse	Bottom of fir tree (transgranular)	Overload	Blade-disk clearance increased	Trailing edge	Crystallographic fracture	Hydrogen-assisted fractures from surface, carbides (independent of geometry), crack orientation in [100] planes	

TABLE 5. SPECIMEN IDENTIFICATION FOR PLASMA SPRAY COATED MAR-M246 DIRECTIONALLY SOLIDIFIED SPECIMENS[7]

Blade No.	Serial No.	Coating Type	Composition, wt%	Thickness, mm
1	5T28	Base	Ni-20Cr	.025-.051
		Cover	50(Ni-20Cr)-50(ZrO_2-5CaO)[c]	.076-.101
2	5W21	Base	Ni-20Cr	.025-.051
		Cover	50(Ni-20Cr)-50Al_2O_3	.076-.101
3	5R6	Base	Ni-20Cr	.025-.051
		Cover	50Ni-20Cr)-50Cr_2O_3	.076-.101
4	5M10	Multilayer[a]	Ni-20Cr	.025
			(ZrO_2-5CaO)[c]	.025
5	5R7	Multilayer[b]	NicrAlY	.025
			(ZrO_2-12Y_2O_3)[d]	.025
6	4M28	Base	Ni-20Cr	.025-.051
		Cover 1	30(Ni-20Cr)-70(ZrO_2-5CaO)[c]	.089-.113
		Cover 2	(ZrO_2-5CaO)[c]	.063-.089
7	4D25	Base	NiCrAlY	.076-.101
		Cover	(ZrO_2-12Y_2O_3)[d]	.152-.203
8	5D17	Base	NiCrAlY	.025-.050
		Cover 1	30NiCrAlY-70(ZrO_2-12Y_2O_3)[d]	.089-.113
		Cover 2	(ZrO_2-12Y_2O_3)[d]	.063-.089
9	5N1	Base	NiCrAlY	.076-.101
		Cover	(ZrO_2-20Y_2O_3)[e]	.152-.203
10	5V15	Base	NiCrAlY	.076-.101
		Cover	(ZrO_2-20Y_2O_3)[e]	.152-.203
11	6X9	Base	NiCrAlY	.076-.101
		Cover	(ZrO_2-20Y_2O_3)[e]	.152-.203

[a]Six 0.025 mm thick layers (3 each) of Ni-20Cr and (ZrO_2-5CaO) beginning with Ni-20Cr.

[b]Six 0.025 mm thick layers (3 each) of NiCrAlY and (ZrO_2-12Y_2O_3) beginning with NiCrAlY.

[c]Norton 252.

[d]Zircoa.

[e]Metco.