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4000°F Materials for Low-Thrust Rocket Engines

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The use of layered wall construction composed of various combinations of rhenium and platinum group metals, ceramic materials, carbon-based composites, ceramic matrix composites, and newer classes of intermetallic compounds suitable for use to 4000°F is discussed. The maturity of each of the respective materials is based on the reported liquid rocket engine firing life using NTO/MMH, NTO/N_2H_4, and O_2/H_2 propellants at combustion pressures of 100–200 psia.

Introduction

COOLING of today's low-thrust, liquid-bipropellant rocket engine combustion chamber requires the use of significant amounts of fuel film cooling because the most common material of construction, a disilicide-coated niobium alloy, is limited in operating temperature to about 2500°F, which is only 50% of the propellant combustion temperature. This method of cooling requires degrading the engine performance in order to obtain the required life. Elimination of the fuel film cooling will result in a cleaner burning engine and significantly higher specific impulse and, therefore, reduced launch costs and/or extended spacecraft life. Elimination of the fuel film cooling requires a combustion chamber material that can operate in a passive, radiation-cooled mode, at temperatures ranging from 3500 to 4000°F.

Initial reports[1,2] documenting the successful rocket engine tests at these temperatures using rhenium chambers fabricated by chemical vapor deposition were made in 1983 and 1984. These early demonstrations comprised of hundreds of seconds of continuous rocket engine firing were aided by fuel film cooling to preclude oxidation rather than to reduce chamber temperature.

A considerable investment was made, between 1980–1993, to develop lightweight, 3000°F and higher, oxidation-resistant carbon and silicon carbide-based composite materials for air-breathing propulsion systems. The low-thrust liquid rocket propulsion community[3,4] incorporated these material advances to extend operating temperature capabilities beyond the 2500°F limits of disilicide-coated niobium alloys.

High-temperature materials and manufacturing process developments for rocket propulsion can be divided into two application categories (Table 1). The first is the materials technology required to support space defense (SD) requirements where emphasis is placed on high-strength-to-weight ratio materials and a capability for operating at high temperatures for short periods. The approach taken for these applications was to miniaturize the designs by increasing operating pressure. The high-strength-to-weight ratio carbon–carbon and carbon–silicon carbide composite materials were pushed to their upper operating temperature limits. The results were an order of magnitude increase in thrust-to-weight ratios and reduction in size over the conventional designs that existed in the early 1980s. These types of engines operated at chamber pressures of 500–2000 psi for durations up to 30 s.

The second application for high-temperature materials research for liquid rocket propulsion addressed the need for improved performance of the low-thrust onboard liquid axial engine (LAE) propulsion systems required for placing communication, weather, and surveillance satellites in a wide range of orbits (including geosynchronous), and for planetary exploration. In contrast to the SD-related missions, propellant utilization efficiency is much more important. These engines must operate for many hours, consuming thousands of kilograms of propellant, which may comprise up to 80% of the payload total mass at separation from the booster. An ambitious mission such as comet rendezvous asteroid fly-by (CRAF) required a firing life of 8 h, more than 200 starts, and single burns lasting several hours. For flight qualification a single engine would be expected to demonstrate a capability of two times a nominal mission. Flight qualification needs of 6 h of accumulated firing time in several burns are more common.

The major emphasis of this article is the latter class of missions requiring both maximum performance and long life. The discussions include the following topics: 1) operating environments and life-limiting mechanisms, 2) classes of high-temperature materials, and 3) rocket engine test results.

Important practical design issues such as providing leak-free joints of the various material combinations while attaching a 4000°F combustion chamber to an injector, valve, and thrust mount, which cannot be permitted to exceed 300°F, are not specifically covered. However, it is appropriate to note that metals having high ductility are easier to use compared to nonductile ceramic composites. Ceramic composites may also have the disadvantage of being permeable if not properly densified or otherwise sealed.

Operating Environments and Life-Limiting Mechanisms

Table 2 identifies nine significant material-related life-limiting mechanisms encountered in the operation of a rocket engine combustion chamber. The operating environments these materials are exposed to is severe. Failure can be the result of thermal fatigue caused by rapid heating and cooling rates and the resulting inelastic strain induced by these thermal gradients. Chemical reactions between the high-temperature

Table 1 Typical missions for low-thrust engines

Missiles and space defense
Burn time, 5–30 s
Restarts, many, more than 100
Short pulses, many
Orbit transfer, LEO to GEO
Burn time, 7,200–10,000 s
Restarts, 10–100
Short pulses, none to few

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Table 2 Life-limiting mechanisms

Melting
Oxidizing wall environment
Nonequilibrium combustion and transient operation
Rapid heating and cooling, thermal gradients, thermal shock
Interdiffusion of layered walls or microcomposites
Loss by evaporation
Creep, fatigue, grain growth

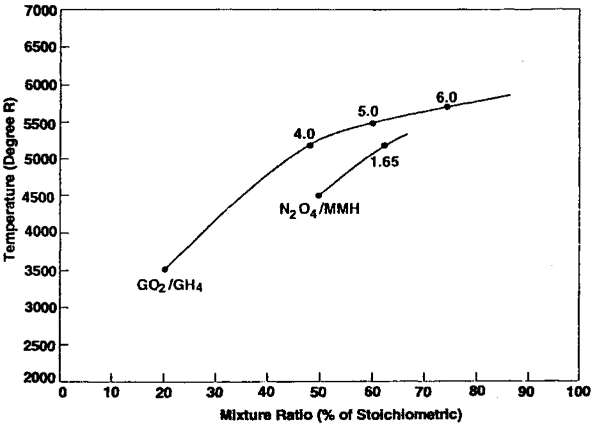

Fig. 1 Combustion temperature of liquid-rocket propellants.

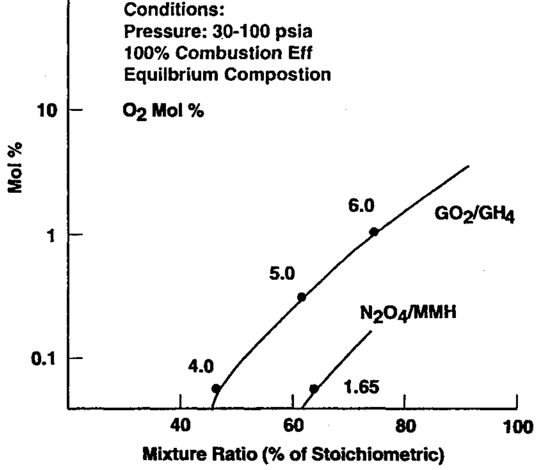

Fig. 2 Significant amount of oxygen is present in the combustion products.

gas and the chamber wall materials are accelerated by high pressure and high-gas flow velocity. The combustion chemistry is complex, being both time and space dependent. Combustion chamber wall exposure starts with the raw propellants near the injector, transitions to partially reacted combustion products, and then to more completely reacted combustion products as the flow progresses down the chamber.

Table 3 identifies the two liquid rocket propellant combinations of major interest and some of the resulting combustion products. These propellants produce combustion temperatures in the range of 5000–6000 R as shown in Fig. 1. Although liquid rockets normally operate in a slightly fuel-rich environment, because this provides the highest specific impulse, it does not preclude the possibility of local regions that are fuel lean due to a number of different phenomena, including imperfect mixing of the fuel and oxidizer and differences in the rates of oxidizer and fuel vaporization. At the very high combustion temperatures equilibrium chemistry predicts the presence of dissociated species such as O, OH, and significant amounts of O_2 as shown in Figs. 2 and 3. The CO_2 and H_2O can act as oxidants for many metals and H_2O, H_2, NH_3, O_2 and CO_2 all will react with carbon-based materials, some very rapidly. Some transient species such as $(CN)_2$ can react with candidate materials at rather low temperatures.

Candidate Materials

A large number of materials have melting temperatures above 4000°F. These can be considered for individual use or in combination with other materials to produce engineered composites that will provide the desired behavior. The selection criteria are based on high-temperature strength, strength-to-density ratios, material loss rates by surface oxidation on the chamber internal surface (Fig. 4), and by evaporation in vacuum on the chamber exterior (Table 4). To date, no single material has been found to provide all of the qualities required

Table 3 Operating environment

Propellants
N_2O_4/MMH, O_2/H_2
Combustion temperature 3300–3588 K (5000 to 6000°F)
Chemistry:
O_2/H_2
H_2O, O_2, H_2, OH, O, H
N_2O_4/MMH
CO, CO_2, H, H_2, H_2O, NO, N_2, O, OH, O_2, HCN, NH_3, $[CN]_2$, HNO_3
Aerothermal-gas velocity
3000–5000 ft/s
1000–1600 m/s

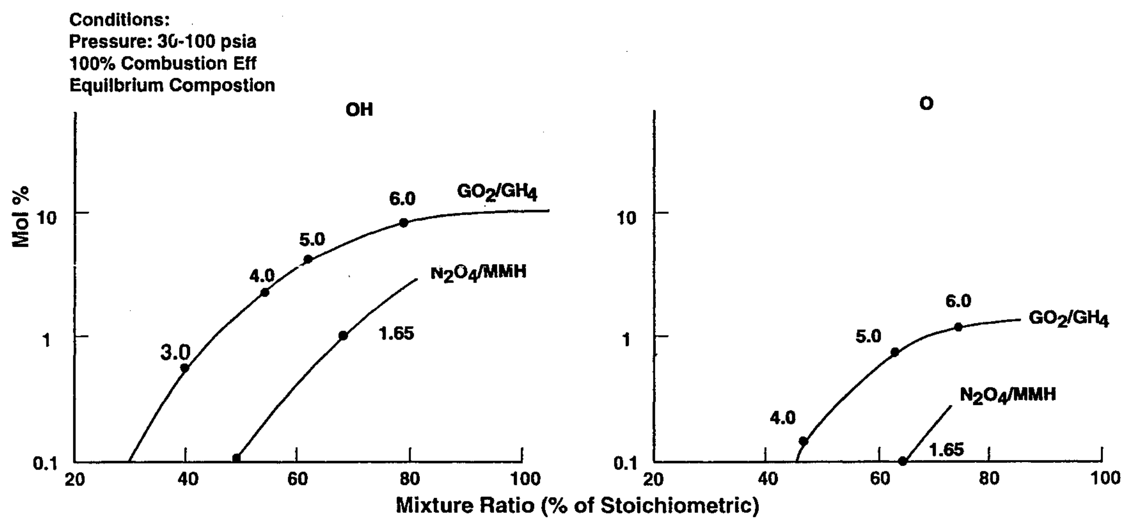

Fig. 3 High concentrations of O and OH exist. These will degrade most materials.

Table 4 Material evaporation rates at 4000°F in a vacuum is a design consideration

Material	Loss Mil/h
HfC	Nil
Tungsten	0.00025
Rhenium	0.0040
Carbon	TBD
Iridium	1.0
Molybdenum	2.0
ZrO_2	3.0
Al_2O_3	40
Rhodium	50
Platinum	60
BeO	100
MgO	8,000
SiO_2	>10,000
BN	>10,000

Table 5 Classes of candidate high-temperature materials

Platinum group and refractory metals
  Iridium/rhenium, platinum-rhodium alloys
Carbon matrix composites
  C/C, oxidation inhibited, C/C
Ceramic matrix composites
  Si_3N_4/SiC, SiC/SiC, C/SiC, C/HfC
Cermets and intermetallic compounds
  Y_2O_3, HfO_2, ZrO_2 + M^a
  HfC, NbC, TaC, ZrC + M^a
  $ZrRe_2$, $HfIr_3$, $ZrIr_3$

[a] Where M = (Re, W, Mo, Ir)

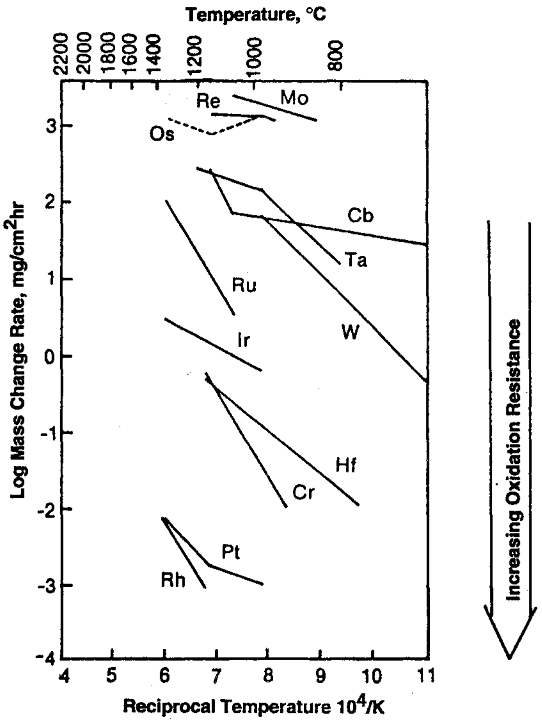

Fig. 4 Oxidation resistance is the key criterion.

to withstand the high temperature, structural, and chemical environments at 4000°F. The engineered material approach that has been most successful is to select combinations where one material is a load-carrying member, a second is for preventing the primary structural material from being chemically attacked, and sometimes a third to act as a thermal barrier providing additional life enhancement. Where more than one material is employed the relative thermal expansion rates and

Table 6 Potential cermets of rhenium metal for 4000°F applications[a] based on compatibility testing

Metal	Oxide	Test temperature °C	Test temperature °F	Reaction vacuum or inert
Re +	ThO_2	2350	4241	No
	HfO_2 + Y_2O_3	2350	4241	Minimal
	ZrO_2 + Y_2O_3	2350	4241	Minimal
	HfN_2	2500	4531	No
	TaB_2	2350	4241	Yes
	NbB_2	2250	4081	Yes
	HfC	2500	4531	No
	NbC	2500	4531	No
	TaC	2500	4531	No
	ZrC	2500	4531	No

[a] Manning, C. R., and Stopps, R. F., "High Temperature Cermet's Compatibility," *Journal of the Ceramic Society*, Vol. 51, No. 8, 1968.

Table 7 Examples of high-temperature stable intermetallic compounds (Engel–Brewer)

Compound	Melt temperature °C	Melt temperature °F
$ZrRe_2$	3020	4976
$HfIr_3$	2737	4467
$ZrIr_3$	2535	4103
$ZrPt_3$	2470	3986
$HfPt_3$	2400	3860
ZrRe	2170	3446

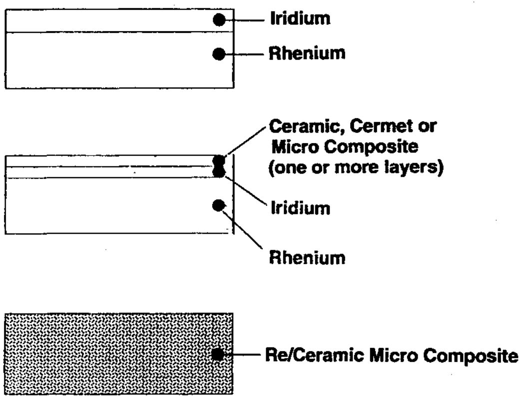

Fig. 5 Methods of providing oxidation protection for rhenium.

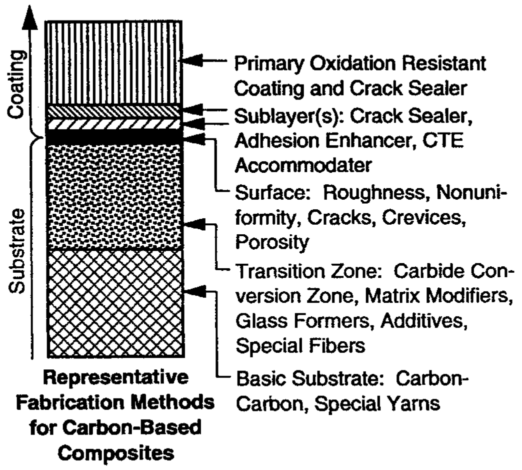

Fig. 6 Fiber-reinforced composites can be fabricated by numerous methods; carbon fibers need a protective coating.

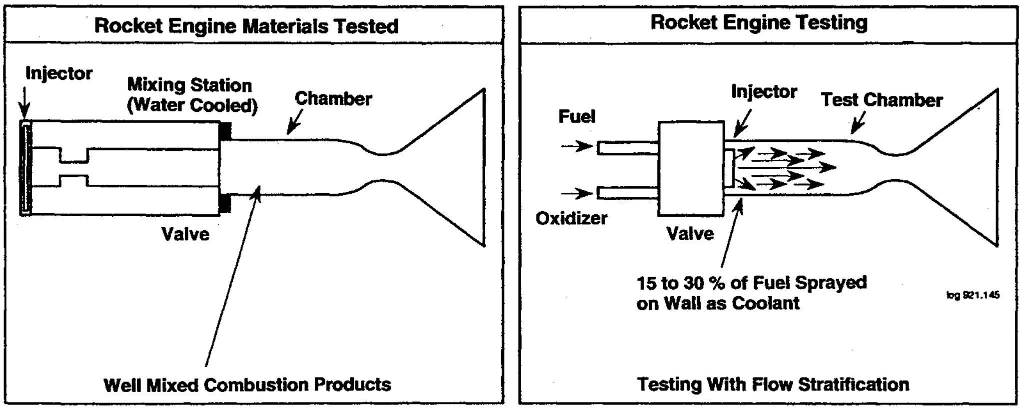

Fig. 7 Test methods for high-temperature rocket engine materials.

interdiffusion rates at high temperatures must also be considered. Several materials systems that have melting temperatures below 4000°F have been included in this survey because of their uniquely superior oxidation characteristics.

The materials worthy of discussion, based on published data, are divided into the four categories given in Table 5. Figures 5 and 6 and Tables 6 and 7 identify the combinations of materials being considered in each of these categories.

Material Systems Under Evaluation

Each of the material systems identified in the previous figures is being evaluated either under laboratory conditions or in actual hot-fire rocket testing which, in nearly all cases, is much more stressful. The balance of this presentation will only address the hot-fire rocket-test results.

Experimental Rocket Engine Material Testing Methods

Two methods of testing advanced rocket engine materials are in use. These are illustrated schematically in Fig. 7. One test method involves replacing the conventional chamber material by the high-temperature test material using existing injector designs and systematically reducing the fuel film cooling to obtain higher performance and higher temperatures. This test method makes the material performance highly dependent on the specific propellant injector employed and its ability to produce uniform, well-mixed flow. The other test method is to insert a cooled mixing section between the injector and the high-temperature chamber material being tested. Either propellant or water can be used as the coolant for material evaluation, however, only propellant cooling can be considered for extending this approach to flight designs. The mixing section is designed to remove all fuel film cooling as well as injector-induced stratification, thus allowing the material to be evaluated under well-defined environmental conditions.

Test Results Using a Cooled Mixing Section and Well-Mixed Combustion Products

One of the high-temperature rocket engine materials test devices was based on a 5-lbf thrust reaction control thruster that used NTO/MMH propellants.[5-8] A water-cooled hot gas mixing section was added between an existing multielement propellant injector and the test chamber to eliminate the fuel film cooling and insure the delivery of fully mixed and completely combusted propellants to the material test section. Initial testing using uncoated rhenium chambers, whose melting temperature exceeds the combustion temperature of these propellants by 500°F, verified[5] that the combustion efficiency, based on C^*, exceeded 99% and the adiabatic chamber wall temperature was within a few hundred degrees of the 5000°F theoretical combustion temperature. Figure 8 shows a photograph of this test device and a sample test chamber. Rhenium alone exhibited a slow loss of material in long duration tests at the O/F ratio of highest performance, greater than 1.6, and also during hot pulsing operation where thousands of restarts are required. The latter is attributed to the loss of mixture ratio control during the shutdown transient.[9] Replacement of the rhenium chamber with an iridium-lined rhenium chamber produced a significant improvement in durability. A single chamber accumulated of firing duration 15 h at 4000°F and optimum performance conditions[8,10] with no measurable material loss. These results were reproduced with duplicate injectors and chambers, where 8 h of firing life without failure were attained.

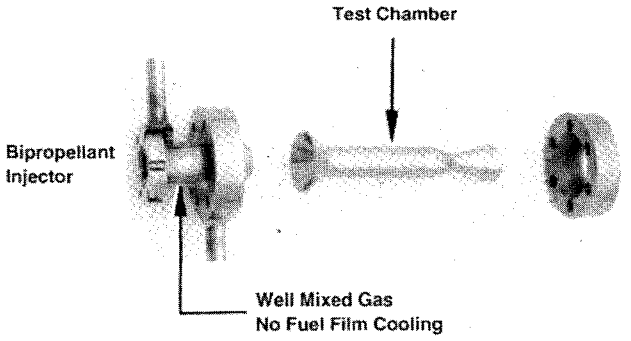

Fig. 8 Standard high-temperature rocket materials test chamber.

A second, similar rocket engine materials test device that permitted the same chamber design to be tested using hotter oxygen and hydrogen propellants is being used as an advanced materials test bed.[10-14] Oxygen/hydrogen propellants can produce an order of magnitude higher concentration of oxidizing species than NTO/MMH at optimum performance as shown in Fig. 2.

These two rocket materials test devices have been used to evaluate a total of 29 different chambers of eight different material systems as defined in Tables 8 and 9. As of June 1993 the total firing time added up to over 200,000 s. Most of the experience to date has been accumulated on iridium/rhenium chambers. One of the standard test chambers has accumulated 54,000 s of operation, using NTO/MMH propellants.[8] Nearly all of the firing time was at temperatures in excess of 4000°F. Testing on this chamber was terminated because of funding limitations, not material durability. Figure 9 shows the burn time, temperature, and mixture ratio history of this unit. During this period there was no measurable change in throat diameter and no visible penetration of the iridium liner. An identical chamber design has been fired 122 times[12] using oxygen and hydrogen propellants for an accumulated 51,166 s at mixture ratios from 3 to 8, but primarily 3 to 5.5 with no observable signs of failure. Chamber pressures ranged from 95 to 106 psia and the temperature was primarily 3100°F,

Table 8 Materials tested in 5-lbf rocket materials test chamber

Structural material	Oxidation liner material	Number tested, NTO + MMH/O^2 + H^2	Total life, NTO + MMH/O^2 + H^2	Temperature range, °F
Rhenium	——	3/0	11,000/-	2,000–3,700
Rhenium	Iridium	5/2	128,168/60,000	3,500–4,300
Rhenium	Iridium/CM	1/7	1,200/10,750	3,500–4,600[a]
G. S. platinum	——	4/0	6,000/-	2,000–3,000
Carbon/carbon	Acc 4 Pack diffusion	2/0	327/-	2,300
Carbon/CVI HfC	——	1/1	105/161	2,960[b]
SiC/K-Karb	——	1/0	220/-	1,800[c]
C-103	R512E	1/0	4,170/-	2,610[c]
Total		19/10		

[a] Various types of ceramic and cermet liners. [b] Mechanical failure. [c] With FFC.

Table 9 Test history of iridium/rhenium chambers using the rocket engine materials tester

Thrust, lbf	Propellant	Maximum ID temperature, °F	Maximum or range, O/F	Starts	Full thermal cycles	Test duration, s
5	NTO/MMH	4,200	2.1	3,638	37	31,369
5	NTO/MMH	4,100	1.7	14	14	13,016
5	NTO/MMH	4,300	1.7	157	74	28,426
5	NTO/MMH	4,075	1.5 to 2.1	2,701	70[c]	>54,431
5	NTO/MMH	3,920	1.4	10	9[c]	>926
5	O_2/H_2	4,000	3–8	122	122	51,166
5	O_2/H_2	3,870	4–6	36	36	9,455
5[a]	O_2/H_2	4,232	4–11[b]	25	25	1,665
5[a]	O_2/H_2	4,576	4–11[b]	24	24	1,941
5[a]	O_2/H_2	4,200	4–11[b]	23	23	1,812
5[a]	O_2/H_2	4,509	4–11[b]	18	18	1,470
5[a]	O_2/H_2	4,528	4–11[b]	27	27	1,933
5[a]	O_2/H_2	4,533	4–11[b]	24	24	1,931
5[a]	O_2/H_2	4,401	4–11[b]	48	48	3,141
All tests at 90–130 psia						Total = 202,682

[a] These chambers include a ceramic liner on the ID.
[b] Test sequence: 100 s total at $MR = 4$, 7 tests, 320 additional s at $MR = 6$, 3 tests, 920 additional s at $MR = 8$ in 5 tests, balance $MR = 11$, 60 s/test until failure. Recalibration indicated actual MR 5 to 14, not 4 to 11 (Ref. 14).
[c] >Chambers show no evidence of coating loss or cracking due to fatigue. Ultimate life capability not yet evaluated.

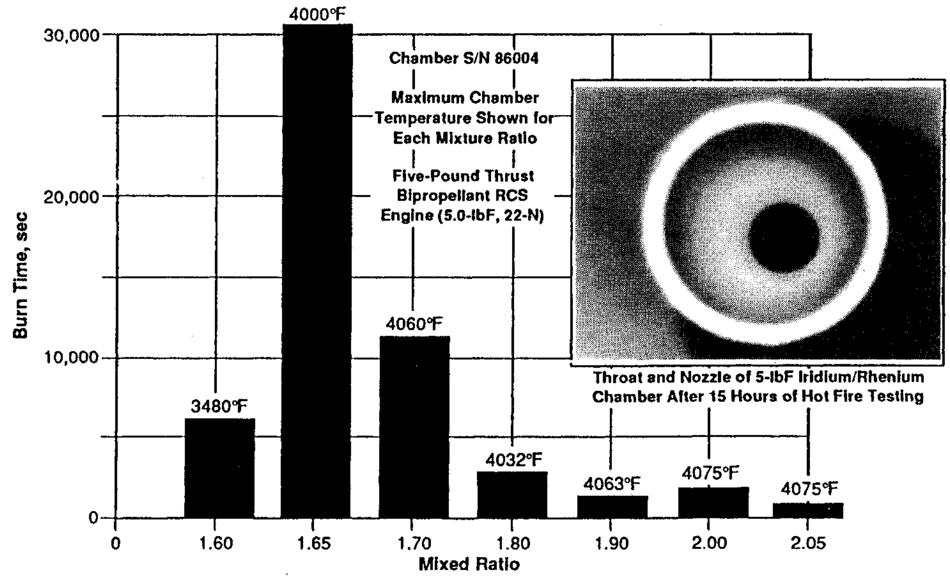

Fig. 9 More than 54,000 s of hot-fire testing on single iridium-lined rhenium chamber verifies life, NTO/MMH.

however, 4000°F was obtained for short times at the higher mixture ratios. A facility problem, rather than material durability, limited the firing duration.

The addition of ceramic oxide overlayers on the inner surface of iridium/rhenium chambers have been reported to significantly improve oxidation resistance, thereby allowing more oxidizing operating conditions, higher temperatures, and enhance life. Test results from seven different combinations of ceramic compositions and thicknesses are reported.[12,13] Some have been successfully tested to 4500°F in an oxygen-rich environment. The conditions and durations of these tests are given in Table 9. One of these ceramic-lined iridium/rhenium chambers has accumulated 38.9 h operating with H_2/O_2 propellants at an oxygen/fuel weight mixture ratio (MR) ≈ 6 and chamber pressure of 65 psia.[14] The temperatures under these test conditions were 3450–3700°F on the inner wall, 3380–

3650 at the oxide/iridium interface, and 3300–3560 on the external surface. This chamber had 94 restarts.

Chambers fabricated from grain-stabilized platinum alloys that can operate without a liner or coating performed well at all mixture ratios, even oxidizer rich, up to temperatures of 3000°F.[5,6] Chambers fabricated from monolithic ceramic and graphite could not be tested effectively with the standard flared chamber interface design because they tended to crack at the sealing surface due to combinations of mechanical and thermal stresses. The ceramic composite designs could be attached mechanically and sealed satisfactorily using several layers of Grafoil® as a gasket material. Successful gas-tight assemblies have been demonstrated using chemically vapor deposited and/or brazed metallic transition joints. It was not always practical to conduct leak checks on composites as the chamber walls were often porous. As can be noted from the short firing durations, these materials degraded rapidly under rocket test environments.

Rocket Test Results Using Prototype Engine Configurations

Results from materials evaluation tests conducted using prototype engine configurations (no water-cooled chamber sections) are given in Tables 10 and 11. One design did include a propellant-cooled mixing device that eliminated fuel film cooling and enhanced the mixing and combustion efficiency.[15,16]

Iridium/Rhenium Rocket Test Results

Testing the iridium-coated rhenium material in actual rocket configurations has been conducted by many rocket manufacturers.[8,15,17–23] The total accumulated hot-fire time in prototype rocket configurations at three different thrust levels and three companies had exceeded 60,000 s by 1993, with many test engines still in test and showing little or no sign of wear. This is in addition to over 200,000 s in the material test chamber configuration (Table 9). Data were obtained using NTO and both MMH and N_2H_4 fuels (Table 10). Some experimenters reported[17] that the iridium was not holding up well at the head end of the chamber and throat at relatively low temperatures, i.e., less than 3000°F. Others reported[8] a failure at the throat of a 5-lbf engine after 100,000 engine pulses, a maximum temperature of 3607°F, and 7735 s of firing. Analysis of the failed region of this chemically vapor-deposited material attributed the failure to a manufacturing-related stress concentration caused by the sharp radius of curvature of the particular design. It is also reported[19] that over 3800 s of firing duration have been attained without problems on a 100-lbf thrust engine with NTO/N_2H_4 propellants. Testing is continuing. The reported maximum temperatures of less than 3100°F would seem to indicate that some degree of film cooling or flow stratification exists in this engine.

At chamber pressures of 100–150 psi NTO/MMH and NTO/N_2H_4 engines operate at temperatures of 3400–3600°F without film cooling to provide combustion efficiency of 99%.[5] Analytical methods of predicting the life limitations of iridium/rhenium combustion chambers accounting for surface oxidation and metal diffusion rates are under development.[23–25]

Composite and Other Chamber Material Test Results

The test results for ceramic matrix composite chambers is given in Table 11. The most durable class of composite material has been the silicon carbide fiber/silicon carbide matrix. This material can be employed up to about 3000°F. Above this temperature both the strength of the fiber and the ability of the protective silica surface film to resist the high shear forces at the sonic throat, degrade rapidly. The difficulty in achieving similar durability and cycle life success with carbon fiber-based materials, even those having an oxidizing inhibiting matrix such as silicon carbide or more complex boron–silicon matrix formulation is in great part due to the mismatch in thermal growth and the absence of ductility in the carbon fibers and the matrix materials. The chemical protection mechanisms of the ceramic composites lies in the self-healing ability of the matrix that softens, oxidizes flows, and fills the microcracks formed during each thermal cycle.

Table 10 Test history of 5-, 15-, and 100-lbf thrust Ir/Re chambers in prototype configurations

Thrust, lbf	Fuel Ox = NTO	Area ratio	Maximum temperature, °F	Maximum O/F	Starts	Full thermal cycles	Test duration, s
5	MMH	150:1	4,000	1.9	>94,588	32[a]	>4,788
5	MMH	150:1	3,607	1.7	>100,000	28	7,735
5	MMH	TBD	2,911	TBD	21	21	1,529
15	MMH	75:1	3,553	1.9	339	19[a]	>746
15	MMH	75:1	8[a]	——[b]	——[b]	——[b]	——[b]
100	MMH	22:1/44:1	3,500	1.7	——	61[a]	>3,885
100	MMH	22:1/44:1	3,500	1.7	——	87[a]	>17,194
100	MMH	286:1 and 47:1	3,600	1.9	91	81[a]	>22,700
100	MMH	286:1	——[b]	——[b]	——[b]	——[b]	——[b]
5	N_2H_2	TBD	2,551	TBD	7	7	455
100	N_2H_2	TBD	3,000	1.0	6	6[a]	>1,200
100	N_2H_4	TBD	3,060	1.1	28	28[a]	>2,640

[a] Chambers show no evidence or coating loss of cracking due to fatigue. Ultimate life capability not yet evaluated. Total firing experience as of May 1993—60,000 s. All tests at 100 to 130 psia.

[b] Not tested.

Table 11 Reported data for rocket engine testing using carbon, CMCs and cermets

Source material	Thrust		Propellants	Pressure, psia	Temperature, °F	Firing life, s	Thermal cycles
	N	lbf					
SEP-SiC/SiC	22	5	N_2O_4/MMH	283	2,900	90,000[a]	660 500,000 Pulses
Carbon/SiC	6,000	1,350	N_2O_4/MMH	283	2,700	900	7
Marquardt CERASEP	22	5	N_2O_4/MMH	100[c]	2,911	15,600	201
Ceramic matrix composite/rhenium/ZrO	22	5	N_2O_4/MMH	100[c]	4,000	2,200[b]	Not reported
Refractory composite C/HfC	110	25	O_2/H_2	16	N/A	161	Several

[a] No material loss. [b] Ref. 26. [c] Not stated, 100 psi most likely value.

An impressive 90,000 s (25 h) of accumulated firing time, including 500,000 partial thermal cycle pulses, on a single engine operating at 2900°F has been reported[3] using a SiC–SiC composite material. There was no dimensional change of the throat.

The carbon fiber, carbon matrix composites that provide outstanding high-temperature strength and high-strength-to-density ratios to over 4000°F, have not held up well in extended duration testing.[4] The problems with these composites are due to the difficulty of producing effective oxidation protective barriers between the carbon fibers and the water vapor in the products of combustion. The most effective of the carbon-based composites employ silica formers that have the ability to reseal the cracks when reheated to the softening temperature (starting about 1500°F). Silica formers (which include silicon carbide and the class of inhibited matrix materials), are only effective up to their flow temperature, which starts at approximately 3000°F. Mixing of silicon, hafnium, and other carbides is being investigated as a method for improving operating temperature, however, no significant rocket test data have been uncovered thus far.

The fourth class of materials, cermets, and intermetallic compounds with high melting temperature (Tables 6 and 7) represents a broad category that is currently under investigation. Thus far, only one engine manufacturer[26] has reported success at using a rhenium-zirconia microcomposite at 4000°F, however, the experience base is small.

Conclusions and Recommendations

1) Several materials having higher operating temperature than silicide-coated niobium have been demonstrated.

2) The iridium/rhenium-layered wall and the SiC/SiC composite are the most mature and can be considered for flight applications.

3) The iridium/rhenium material can provide 1100°F higher operating temperature than the SiC/SiC 4000 vs 2900°F.

4) The low-thrust rocket engine materials tester has been a cost-effective method for evaluating new materials. Consideration should be given to establishing an industry standardized test method to reduce testing cost and allow direct comparison of data from different agencies.

Although designs using new high-temperature materials have been successfully demonstrated, numerous issues related to design optimization, fabrication cost, quality control, and inspection remain to be resolved. The design issues include inadequate characterization of material creep and fatigue properties as a function of the fabrication processes, use temperature and time at temperature.

The investment in advanced high-temperature materials for low-thrust liquid propulsion can be paid back quickly because it enables significant performance gains that reduce launch costs and allow satellites to remain in orbit longer.

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