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New Generation of High-Performance Engines for Spacecraft Propulsion

Sanders D. Rosenberg* and Leonard Schoenman†
Aerojet Propulsion Division, Sacramento, California 95813

The technology readiness for a new class of high-performance storable bipropellant engines has been demonstrated at three different thrust levels, i.e., 5-, 15-, and 100-lbf thrust. Improved high-temperature materials and combustion chamber design permit the removal of fuel film cooling and its associated contamination and performance losses without compromising pulsing duty cycle flexibility. The iridium/rhenium composite chamber material greatly increase the thermal margin over existing engine designs which use disilicide-coated columbium thrust chambers by increasing the allowable operating temperature to 2204°C (4000°F). The companion technologies for providing metallurgical joints between the different materials used for the injector, chamber, and high area ratio skirt have also been demonstrated and incorporated into flight type engine designs.

Introduction

PROPELLANT used for orbit insertion and attitude control is the largest single item contributing to the mass of most satellites at launch. Not only does this increase the cost of placing the satellites in orbit, but the depletion of propellant generally limits satellite life. Any design improvements that can decrease satellite propellant requirements or make more effective use of propellants will have significant economic benefit.

Most of the rocket engines in use in today's satellites are either relatively low-performing hydrazine monopropellant thrusters, or liquid bipropellant engines employing nitrogen tetroxide (NTO) and monomethyl hydrazine (MMH) as propellants. The bipropellant engines delivered performance is considerably lower than theoretically attainable because of the way the combustion chamber must operate due to material limitations. Today's low thrust engines (under 1000-lbf) employ disilicide-coated columbium chambers which have a nominal upper use temperature of 1316°C (2400°F) with approximately 10 h of life. To maintain the chamber walls at or below this level, a significant amount of the fuel is injected onto the chamber wall as fuel film coolant which does not burn completely before exiting the chamber. In addition to degrading efficiency, the unburned fuel also represents a potential source of plume contamination which has an adverse effect on sensors, solar cells, and other onboard instruments. The amount of fuel film cooling required to maintain acceptable wall temperatures is typically 30–40% for 5-lbf thrusters and 15–30% for 100-lbf thrusters. This results in a loss of 5–10% of the available specific impulse, i.e., 15–30 s. In order to obtain the highest possible performance from these bipropellant engines, two new technologies are required. The first is an oxidation resistant material system that can operate at temperatures in excess of 1927°C (3500°F) for tens of hours. The second is a method of attaching the chamber to the injector without overheating the injector and valve or transferring an excessive amount of heat to the engine/spacecraft interface. The injector must be maintained at a temperature low enough to prevent oxidizer vapor lock. The valve must be maintained below temperatures which may damage its soft goods. This head end temperature limitation is in the 121–149°C (250–300°F) range.

This new class of engines, developed with extensive NASA support, is capable of operating at close to 100% combustion efficiency without compromising thermal design margin, i.e., front end thermal control and upper temperature limits of the combustion chamber wall.

This article provides experimental data validating advanced engine designs at three thrust levels, 5-, 15-, and 100-lbf. The hot-fire test results demonstrate specific impulse improvements of 15–25 s over conventional fuel film cooled columbium chamber designs, while operating at maximum chamber temperatures significantly below the demonstrated material system capabilities which is in excess of 2204°C (4000°F).

Design

Valve

The 5-, 15-, and 100-lbf engine assemblies fabricated and tested to date are shown in Figs. 1, 2, and 3, respectively. All of the engines employ a bipropellant torque motor valve which is free of sliding or rubbing surfaces that can bind, induce wear, or generate contamination. Valves of this general type have demonstrated the ability to deliver over 1,000,000 cycles without sticking, leaking, or shifting the oxidizer-fuel lead-lag relation on startup and shutdown. The valve incorporated into the 15-lbf engine design has series redundant shutoff seals and is flight qualified.

Injectors

The three engines employ platelet injectors designed to produce approximately 1.0 lbf per element. The stainless steel

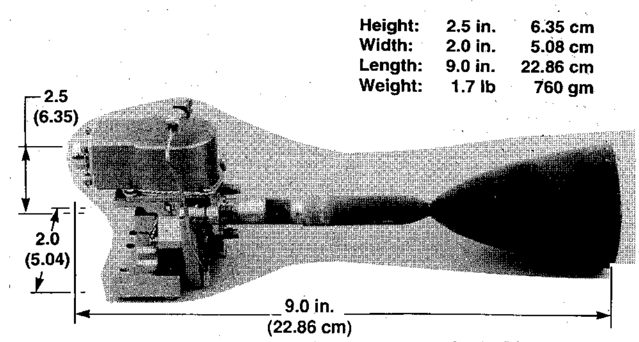

Fig. 1 Five-lbf NTO/MMH rocket engine assembly, I_{sp} > 310 s at ε = 150:1.

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*Chief Scientist, Retired, P.O. Box 13222. Associate Fellow AIAA.
†Department Manager, P.O. Box 13222. Associate Fellow AIAA.

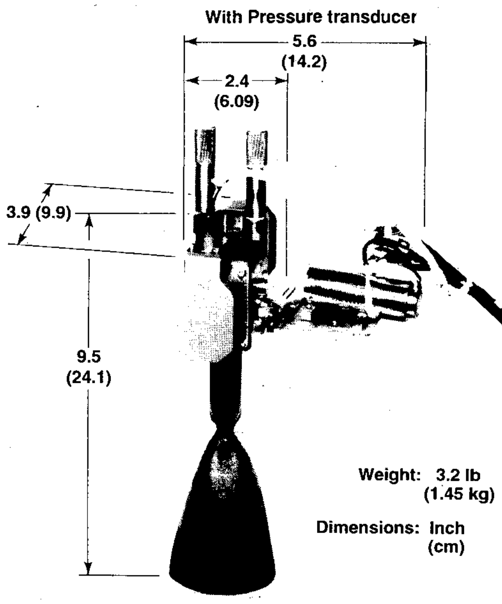


Fig. 2 Fifteen-lbf NTO/MMH rocket engine assembly, dual shutoff valves, I_{sp} = 305 s at ε = 75:1.

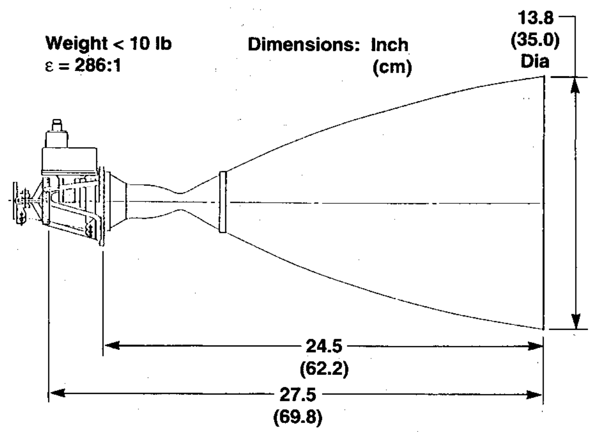


Fig. 3 One hundred-lbf NTO/MMH rocket engine assembly, I_{sp} = 320 s at ε = 286:1 with optional gimbal.

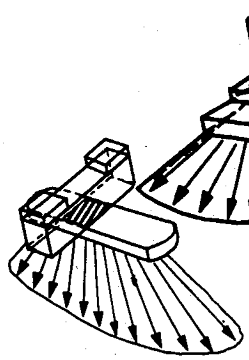


Fig. 4 Types of preatomized injection elements employed in platelet injectors (Patent no. 3,881,701).

injector has been life tested for 40 h at the 5- and 15-lbf, and 5 h at the 100-lbf thrust level. Preatomized unlike doublet elements are employed in the three designs. In the preatomized design (Fig. 4) the propellant leaves each injection orifice in an atomized state, avoiding the vexing problem of requiring perfect impingement/alignment of two small diameter streams for the purpose of atomization. Misalignment of small impinging propellant jets is a major cause of nonreproducible engine to engine performance, thermal streaking, and stability characteristics. The two higher thrust units have integral critically damped acoustic resonator cavities as part of their basic injector design. The combination of short length/diameter (L/D) orifices and the acoustic resonator cavities provides a robust design which is insensitive to flow decay, which may be caused by the presence of ferric nitrate in the oxidizer, and assures stable combustion.

Combustion Chambers

An iridium-lined rhenium combustion chamber operates in a simple radiation cooled mode, without film cooling. Rhenium was selected because it provides good low-temperature ductility, and has a melting point of 3180°C (5756°F) that exceeds the maximum of combustion temperature of the NTO/MMH bipropellant combination by more than 260°C (500°F). The liner—iridium—was selected because of its chemical inertness, i.e., oxidation and oxygen diffusion resistance, its high melting point of 2454°C (4449°F), and a thermal expansion coefficient that is nearly the same as rhenium. The close

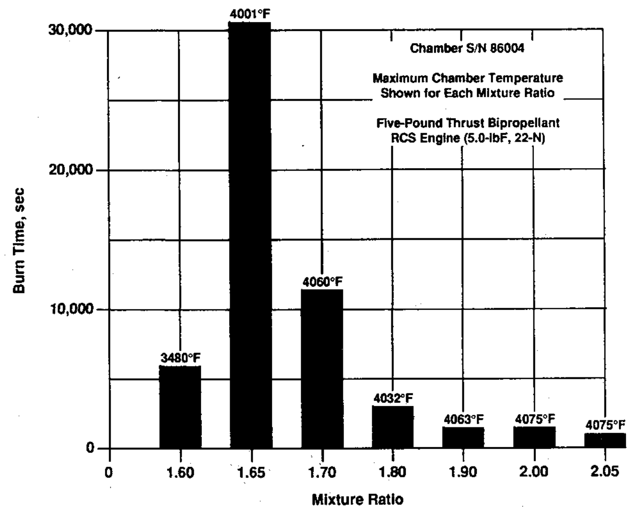


Fig. 5 More than 54,000 s of hot-fire testing verifies life.

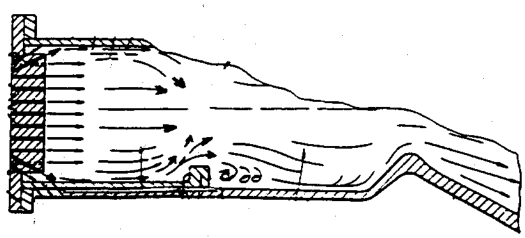


Fig. 6 Two-stage rocket combustor for controlling head end cooling and obtaining maximum performance (Patent no. 4,882,904).

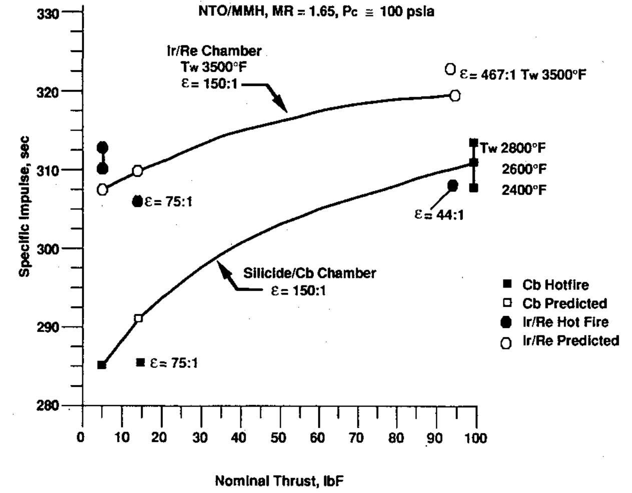


Fig. 7 Iridium/rhenium chambers enable increased performance.

match of the coating/substrate thermal expansion overcomes the thermal cycle life limitations of the disilicide-coated columbium system in which the expansion rates are significantly different, and the coating cracks and refuses following each thermal cycle.

The equilibrium operating temperature for these engines ranges from 1760 to 1927°C (3200 to 3500°F) when all of the propellant is fully burned. Small, local reductions of either oxidizer or fuel has little or no influence on chamber wall temperature. Greater reductions cause the wall temperature to drop, improving the thermal design margin in an off-design operating mode. As shown in Fig. 5 and discussed in Refs. 1–3, a life of 15 h has been demonstrated on a single chamber at 2204°C (4000°F), without failure, over a wide range of mixture ratios. This operating temperature could only be attained by reducing the emissivity of the exterior surface of the combustion chamber in order to reduce radiation cooling. The chamber design allows the use of radiation shields, if required, without exceeding the demonstrated engine operating temperature capabilities.

Table 1 Five-lbf thrust RCS engine (5.0-lbf, 22-N) steady-state performance and chamber temperatures, 150:1 area ratio

I_{sp}: 313 s, MR: 1.65				
Mixture ratio, w_{ox}/w_{fu}	Chamber pressure, psia	Duration, s	Specific impulse, s	Chamber wall temperature,[a] °F
1.59	115.7	300	307.2	3517
1.63	111.3	350	309.8	3552
1.64	114.9	90	313.7	3526
1.65	116.3	90	310.7	3566
1.66	116.3	320	313.0	3582
1.68	115.6	90	317.7	3607
1.68	115.6	90	317.7	3607

[a]Optical pyrometer measure. $\dot{w}$ = propellant flow rate.

Table 2 Fifteen-lbf thrust RCS engine (15-lbf, 67-N) steady-state performance and chamber temperatures—*S/N* 2, 75:1 area ratio

I_{sp}: 305 s, MR: 1.65					
Test no.	Chamber pressure, psia	Mixture ratio w_{ox}/w_{fu}	Duration, s	Specific impulse, s	Chamber wall temperature,[a] °F
121	105.1	1.65	100	305.0	3438
124	80.4	1.62	100	302.1	3297
126	140.2	1.64	100	305.0	3385
128	83.3	1.86	20	299.9	3486
130	104.6	1.88	20	295.4	3553
131	103.3	1.41	20	301.4	3310
132	122.0	1.50	20	301.6	3359
133	130.5	1.40	20	300.6	3310

[a]Optical pyrometer measure.

Two each of the 5- and 15-lbf iridium-lined rhenium chambers were built with full area ratio nozzles of 150:1 and 75:1, respectively, following a development period during which three 5-lbf, 8.4:1 area ratio uncoated rhenium, and five identical iridium-lined rhenium chambers were fabricated and tested for tens of hours and hundreds of thousands of firing cycles.[1] Four 100-lbf chambers have been built. The first two units employed mechanically held nozzle extensions to provide performance and life data at an expansion ratio of 44:1.[4,5] The second two units are being fabricated[6] with 286:1 expansion ratio metallurgically joined, disilicide-coated columbium nozzle extensions. The joints between the rhenium and columbium which also have nearly identical expansion coefficients will operate conservatively at temperatures below 1093°C (2000°F).

All of the advanced engines successfully tested to date have included a device [the two-stage combustor design (U.S. Patents 4,882,904, 4,936,091) (Fig. 6)] that forces the fuel film coolant to leave the chamber wall after the head end cooling is completed. The heated fuel film coolant ejected from the wall is forced to mix and burn with the balance of the core flow propellants. Attempts to operate without this device have resulted in both lower performance and reduced chamber life. In order to guarantee a high thermal design margin, the device, which induces the secondary mixing and combustion, is fabricated from a platinum alloy. The joining technology has advanced to a stage where an all metallurgically joined stainless steel injector, platinum alloy head end, and iridium-lined rhenium chamber can be fabricated, i.e., no bolted joints or hot gas seals are required. Because of the high cost of platinum and other performance advantages, the head end of the higher thrust 100-lbf engine replaces the platinum with a fuel regeneratively cooled stainless steel chamber segment which permits the rhenium chamber to be metallurgically attached to the injector/valve assembly. The actively cooled head end enables continuous burns of any duration, (tens of hours) without producing a thermal load across the engine/spacecraft interface.

Test Data

A comparison of the specific impulse measured for the 5-, 15-, and 100-lbf engines is displayed in Fig. 7 and Tables

Table 3 One hundred-lbf thrust bipropellant axial engine (100-lbf, 445-N) steady-state performance and thermal mapping tests with 44:1 area ratio

Test no.	Duration, s	*Pc*, psia	Thrust, lbf vac	MR, $\dot{w}_{ox}/\dot{w}_{fu}$	I_{sp} vac, s	Throat, °F
116	1.00	103.9	100.0	1.61	——	——
117	4.23	103.0	103.2	1.61	306.6	2869
118	5.0	103.0	104.0	1.62	308.7	2623
119	19.60	102.6	103.9	1.65	309.4	3154
		102.5	103.8	1.65	309.4	3264
120	19.3	101.2	102.3	1.50	308.5	3096
		101.0	102.0	1.49	308.2	3132
121	5.00	78.9	79.7	1.65	304.3	2834
123	60.0	77.6	78.2	1.63	304.0	3503
		77.6	78.3	1.63	304.2	3504
124	13.59	102.4	103.1	1.61	309.6	2980
125	15.96	101.9	103.1	1.82	309.3	3350
126	6.10	125.3	126.5	1.65	308.1	2600
127	6.16	122.5	123.7	1.80	309.0	2615
128	11.80	102.0	103.7	1.63	309.4	3011
		102.7	103.4	1.63	309.3	3192
129	34.60	78.1	78.1	1.64	303.3	3534

1, 2, and 3, respectively. The figure also shows published data for similar classes of engines using fuel film cooled, disilicide-coated columbium chamber materials and a conventional smooth wall combustion chamber design. The test data at the 5-lbf thrust level and 150:1 area ratio demonstrate a specific impulse of 310–313 s. This is 25 s higher than flight qualified designs in use today. The measured specific impulse for the 15-lbf thrust engine (Table 2) at an area ratio of 75:1 is 305 s compared to 286 s for the flight qualified design of the same area ratio. The minimum expected specific impulse for the same engine with a 150:1 expansion ratio is 310 s. This level of performance is attained at an operating temperature that is 260°C (500°F) below the full life operating capabilities of the chamber material system and the engine design.

The measured specific impulse of the 100-lbf thrust engine at an area ratio of 44:1 is 309 s (Table 3). The delivered specific impulse for this same design with a 150:1 and 286:1 expansion ratio is 318 and 321, respectively.

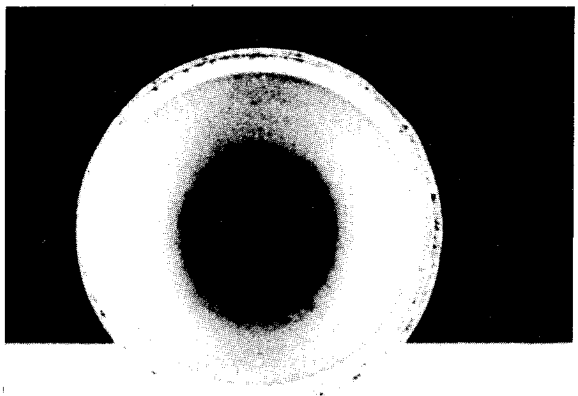


Fig. 8 Throat and nozzle of 5-lbf iridium/rhenium chamber after 15 h of hot-fire testing.

Robustness

A highly robust engine design results from the high thermal design margins of the selected materials, the method in which they are joined, and the basic valve design. A summary of the experimental hot-fire test data base for the seven 5-lbf thrust engines, two 15-lbf engines, and four 100-lbf engines established over the last 5 yr is presented in Table 4. Two of the 5-lbf units were retired after 8 h with no evidence degradation. One was retired in "as new" condition after 15 h of operation, as shown in Fig. 8. Because emphasis was placed on pushing the materials to their respective limits, all of the testing was configured to force the chamber to operate at temperatures in excess of 2004°C (4000°F), which is approximately 260°C (500°F) greater than would be expected in a normal radiation cooled mode. Figure 5 shows the time-temperature mixture ratio life history for the chamber shown in Fig. 8. Much of the total burn time was accumulated at mixture ratios greater than 1.6, a potentially much more degrading condition.

Durability/Flexibility

The operational performance capability of each of the three engines, based on actual hot-fire test data, is summarized in Tables 5–7.

Specific pulse duty cycle test results from a 100,000 restart demonstration for the 150:1 area ratio 5-lbf unit, S/N 88001, are given in Table 8. The testing covered a percent on time ranging from 10 to 90%. Numerous single burns in excess of 1.0 h have been conducted routinely without encountering thermal limitations.

Results from the first steady-state tests of the 15-lbf 75:1 area ratio unit are given in Table 2. Vibration acceptance testing was successfully completed in a two-engine module. Figure 9 shows the condition of the nozzle and throat after the hot-fire and vibration test series in comparison with a

Table 4 Design fabrication and life test history of 5-, 15-, and 100-lbf thrust *Ir/Re* chambers

Thrust, lbf	Area ratio	Max temp, °F	Max *O/F*	Starts	Full thermal cycles	Test duration, s
5	8.4:1	4,200	2.1	3,638	37	31,369
5	8.4:1	4,100	1.7	14	14	13,016
5	8.4:1	4,300	1.7	157	74	28,426
5	8.4:1	4,070	2.0	2,701	70[a]	>54,431
5	8.4:1	3,920	1.4	10	9[a]	>926
5	150:1	4,000	1.9	>94,588	32[a]	>4,788
5	150:1	3,607	1.7	>100,000	28[a]	7,735
15	75:1	3,553	1.9	306	19[a]	>314
15	75:1	Not tested	Not tested	Not tested	Not tested	Not tested
100	22:1/44:1	3,500	1.7	30	30[a]	>3,381
100	22:1/44:1	3,500	1.7	33	33[a]	>15,000
100	286:1	Under construction	Under construction	Under construction	Under construction	Under construction

[a]Chambers show no evidence of coating loss or cracking due to fatigue. Ultimate life capability not yet evaluated. Total firing experience as of April 1991 159,386 s.

Table 5 Five-lbf thrust bipropellant RCS engine (5.0-lbf, 22-N)

Torque motor bipropellant valve, CRES-347 injector, PtRh Ir-lined Re chamber		
	Performance capability	
Parameter	Nominal	Demonstrated operational range
Propellants	NTO/MMH	NTO/MMH
Thrust, lbf	5.0	3.0 to 6.0
Steady-state specific impulse, s	313	303 to 317
Mixture ratio	1.65	1.6 to 1.9
Single burn duration, s	Unlimited	54,000 (Area ratio 8.4:1) 350 (Area ratio 150:1)
Valve inlet pressure, psia	220	400–106 (Blowdown 3.75:1)
Propellant inlet temperature, °F	70	50–120
Expansion ratio	150:1	8.4:1, 150:1

Table 6 Fifteen-lbf thrust bipropellant RCS engine (15-lbf, 67-N)

Torque motor bipropellant valve, CRES-347 injector, PtRh Ir-lined Re chamber

Parameter	Nominal	Demonstrated operational range
	Performance capability	
Propellants	NTO/MMH	NTO/MMH
Thrust, lbf	15.0	11.3–21.7
Steady-state specific impulse, s	305	300–306
Mixture ratio	1.65	1.4–1.8
Single burn duration, s	100	0.10–100
Valve inlet pressure, psia	220	400–150 (Blowdown 2.67:1)
Propellant inlet temperature, °F	70	50–80
Expansion ratio	75:1	75:1

Table 7 AJ10-221 One hundred-lbf thrust bipropellant ΔV engine (100-lbf, 440-N)

Torque motor bipropellant valve, CRES-347 injector, Ir-lined Re chamber

Parameter	Nominal	Demonstrated operational range
	Performance capability	
Propellants	NTO/MMH	NTO/MMH
Thrust, lbf	95	78–127
Steady-state specific impulse, s	323 ± 3	309
Mixture ratio	1.65	1.50–1.80
Single burn duration, s	Unlimited	823, Facility limited
Valve inlet pressure, psia	220	160–380
Propellant inlet temperature, °F	70	55–120
Expansion ratio	467:1	44:1
Cumulative firing life, h	40	4.1

Table 8 Five-lbf thrust bipropellant RCS engine (5.0-lbf, 22-N)

Pulse hot-fire tests, chamber S/N 88001, 150:1 area ratio total starts: 100,313, total burn time: 6,375 s

Test no.	Mixture ratio, w_{ox}/w_{fu}	On time, s	Off time, s	Percent duty cycle	Number of pulses
135	1.72	0.050	0.075	40	400
136	1.65	0.050	0.075	40	400
137	1.65	0.050	0.075	40	1,000
138	1.63	0.050	0.050	50	400
139	1.64	0.050	0.050	50	1,000
140	1.64	0.050	0.033	60	1,000
141	1.64	0.050	0.033	60	1,855
143	1.64	0.050	0.021	70	1,706
144	1.64	0.050	0.012	80	2,240
145	1.64	0.050	0.005	90	2,500
146	1.64	0.050	0.450	10	1,000
147	1.64	0.050	0.200	20	10,000
148	1.64	0.050	0.200	20	10,000
152	1.64	0.050	0.200	20	10,000
154	1.64	0.050	0.200	20	10,000
155	1.64	0.050	0.200	20	10,000
156	1.64	0.050	0.200	20	16,800

silicide-coated columbium chamber in production. The absence of streaks in the high-performance engine provides verification of complete and uniform combustion. The data compiled from the very first engine assembly tested indicates that all thermal and performance parameters are exactly as predicted.

The injector used in the 100-lbf thrust engine tests performed in 1989 accumulated 5 h of operation. One chamber accumulated 4 h of operation in 33 starts. The condition of this chamber after hot-fire test is shown in Fig. 10.

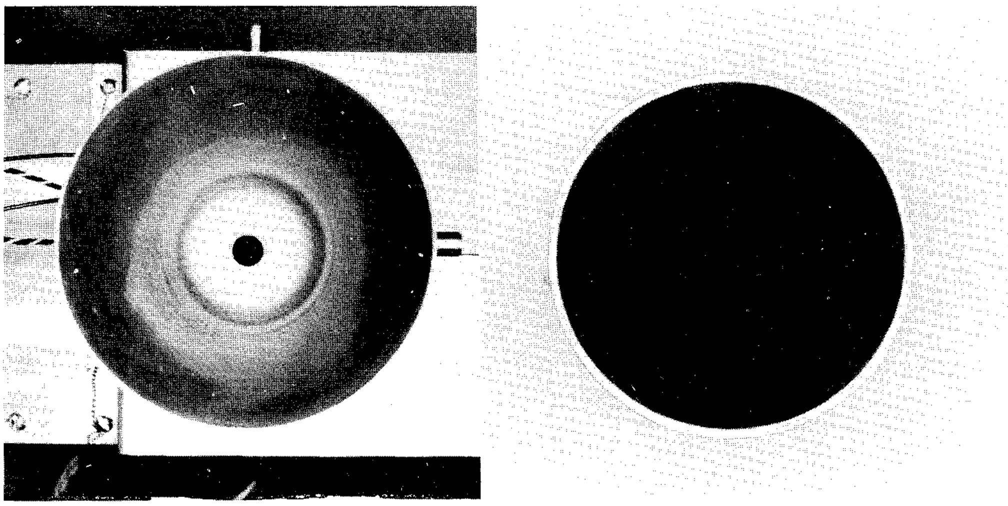
Stability

All of the engines tested have demonstrated stable combustion over the wide test range employed. There has been no evidence of unstable combustion.

Benefits

The economic benefits to be derived from this new generation of high-performance engines are significant. Table 9 displays the weight savings for a typical INTELSAT 7 class spacecraft. The 215 lbf of propellant saved in the high-thrust orbit transfer can extend the useful life of the current 12-yr system to 15 yr. Additional savings in weight or life extension are achieved when the improved performance of the lower thrust station keeping engines are included.

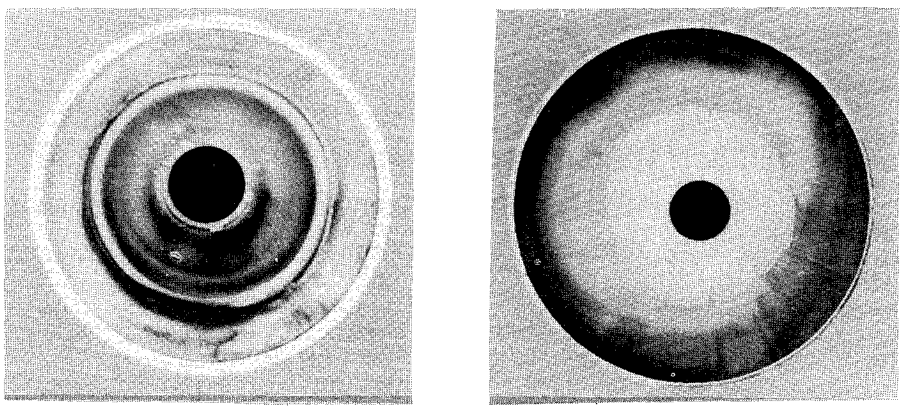
The mission benefits to the NASA planetary missions are measured by the reductions in system propellant weight given in Table 10. The benefit derived from reduction in propulsion related weight of 198 lbf for the CRAF mission are measured by the number of additional instruments and experiments to be included in the science payload package.



Advanced 15 lbF Ir-Re Chamber Posttest   Standard 14-lbF RCT Columbium Chamber Posttest

Both Thrusters Use the Same Injector Design

Fig. 9 Advanced design eliminates contamination from incomplete combustion and fuel film cooling.



Throat Viewed From Injector End   Throat Viewed From Exit End

Fig. 10 One hundred-lbf chamber after 15,000 s of burn time and 33 full thermal cycles.

Table 9 High-performance engines benefit INTELSAT 7 class comsats

	kg	lb
Spacecraft weights in transfer orbit		
Spacecraft total weight	3610	7942
Spacecraft propellant weight	2160	4752
Spacecraft dry weight	1450	3190

An increase in axial engine performance from 307 to 321 s can save 215 lbf of orbit transfer propellant.

The OT propellant savings can increase the amount of attitude control-station keeping propellant by 25%, i.e., extend the useful life of a 10-yr Comsat by 3 yr.

An increase in AC-SK engine performance from 285 to 310 s can save 18 lbf of propellant, providing more Comsat life margin.

Table 10 High-performance engines benefit NASA planetary missions

	kg	lb
CRAF Planetary spacecraft		
Spacecraft total weight	6,300	13,860
Spacecraft propellant weight	4,300	9,460
Spacecraft dry weight	2,000	4,400

An increase in axial engine performance from 308 to 321 s can reduce propellant weight by 90 kg (198 lb) for the CRAF mission.

This reduction can enable addition of one experiment to a baseline 9 experiment, 12-instrument CRAF science payload.

Conclusions

The key to high engine performance with long life is the unique combination of the platelet injector, the recovery of the performance loss normally associated with fuel film cooling, and the iridium/rhenium chamber technology.

These combined technologies have undergone iterative design, fabrication, and testing research and development over the last 5 yr, and are approaching the level of maturity required to begin flight qualification programs for a wide range of candidate missions.

Acknowledgments

NASA Lewis Research Center Contracts NAS 3-25646 and 24643. Valve Design and Fabrication Subcontractor, Moog, Inc., East Aurora, New York. Chamber Fabrication Subcontractor, Ultramet, Pacoima, California.

References

[1]Lansaw, P. T., and Wooten, J. R., "High Temperature Oxidation-Resistant Thruster," Research, NASA CR 185233, Feb. 1990.

[2]Whalen, M. V., Lansaw, P. T., and Wooten, J. R., "High Temperature Oxidation-Resistant Thruster," JANNAF Propulsion Conf.,

Chemical Propulsion Information Agency 480, Vol. I, San Diego, CA, Dec. 1987, pp. 413–419.

[3]Wooten, J. R., and Lansaw, P. T., "The Enabling Technology for Long Life, High Performance on Orbit and Orbit—Transfer Propulsion Systems: High Temperature Oxidation-Resistant Thrust Chambers," JANNAF Propulsion Conf., CPIA 515, Vol. 1, Cleveland, OH, May 1989, pp. 437–443.

[4]Appel, M. A., Schoenman, L., and Franklin, J. E., "Feasibility Demonstration of a 445 N High Performance Rocket Engine," JANNAF Propulsion Meeting, CPIA 515, Vol. 1, Cleveland, OH, May 1989, pp. 445–455.

[5]Schoenman, L., and Franklin, J. E., "Feasibility Demonstration of a High Performance 100-lbF Rocket Engine," Jet Propulsion Laboratory Contract 957882, Final Rept., Jan. 1989.

[6]Jassowski, D. M., Gage, M. L., and Schoenman, L., "Advanced Small Rocket Chambers Option 1 14-lbf Ir-Re Rocket," NASA CR191014, Aug. 1992, Option 2 100-lbf Ir-Re Rocket (to be published NAS3-25646).