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ABSTRACT

A 10,000 lbf thrust chamber was developed for the Upper Stage Flight Experiment (USFE). This thrust chamber uses hydrogen peroxide/JP-8 oxidizer/fuel combination. The thrust chamber comprises an oxidizer dome and manifold, catalyst bed assembly, fuel injector, and chamber/nozzle assembly. Testing of the engine was done at NASA's Stennis Space Center (SSC) to verify its performance and life for future upper stage or Reusable Launch Vehicle applications. Various combinations of silver screen catalyst beds, fuel injectors, and combustion chambers were tested. Results of the tests showed high C* efficiencies (97% - 100%) and vacuum specific impulses of 275 – 298 seconds. With fuel film cooling, heating rates were low enough that the silica/quartz phenolic throat experienced minimal erosion. Mission derived requirements were met, along with a perfect safety record.

1. Introduction

There is a recognized need for space-storable, nontoxic propulsion systems. Orbital Sciences Corporation is under contract to the NASA Marshall Space Flight Center, in cooperation with the Air Force Research Lab, to develop and demonstrate in flight a new low-cost liquid upper stage based on high concentration hydrogen peroxide (HTP) and JP-8. The Upper Stage Flight Experiment (USFE) focuses on key technologies necessary to demonstrate the operation of an inherently simple propulsion system with an innovative state-of-the-art structure.

Orbital Sciences Corporation has completed two series of development tests (DVT-1, DVT-2) on a hydrogen peroxide/ JP-8 engine for the USFE program. These tests were conducted during two testing campaigns beginning in late 1998 and concluding in early 2000. Some key hardware improvements were made during this testing resulting in an engine that fully met the design requirements. A summary of hardware configurations and test results is presented here as well as the basic thrust chamber design.

2. USFE 10k Engine Design

The engine consists of pneumatically-actuated ball valves, propellant feed-lines, the oxidizer dome with a mount for gimbal attachment, a catalyst bed to convert the HTP into oxygen and superheated steam, a fuel injector, and an ablative chamber and nozzle. Figure 1 shows the thrust chamber assembly (TCA).[1] Low material and design costs coupled with robust margins were the guiding philosophy toward selecting a design.

The engine design and operating parameters are provided in Table 1. The engine develops 10,000 lbf of thrust at vacuum conditions with a 40:1 expansion ratio nozzle. Chamber pressure was chosen to be 500 psia, which spans the operating regimes of pressure-fed and pump-fed systems. Based on a demonstrated C* efficiency of 0.97 and an assumed nozzle efficiency of 0.98, the delivered vacuum specific impulse is 275 seconds at a mixture ratio of 4.7.

The catalyst, injector, and ablative chamber designs are based on results from two sets of subscale tests which used a 50 lbf monopropellant thruster for catalyst bed screening and a subscale bipropellant thrust chamber for injector development tests. The subscale configuration captured key design features of the fullscale catalyst bed, the injector, and the chamber.

Historical designs were used to size the thrust chamber.[2] To ensure autoignition of the fuel, a contraction ratio of about seven was chosen. The resulting chamber inner diameter was ten inches. Maintaining this inner diameter in the catalyst bed led to a bed mass flux (G) of 0.4 lb/s-in^2, which is also within the historical operating range of silver screen-based catalyst beds.

3. Engine Test Facility

Full scale tests were conducted at the E-3 Test Cell at NASA Stennis Space Center[3] (Figure 2). The E-3

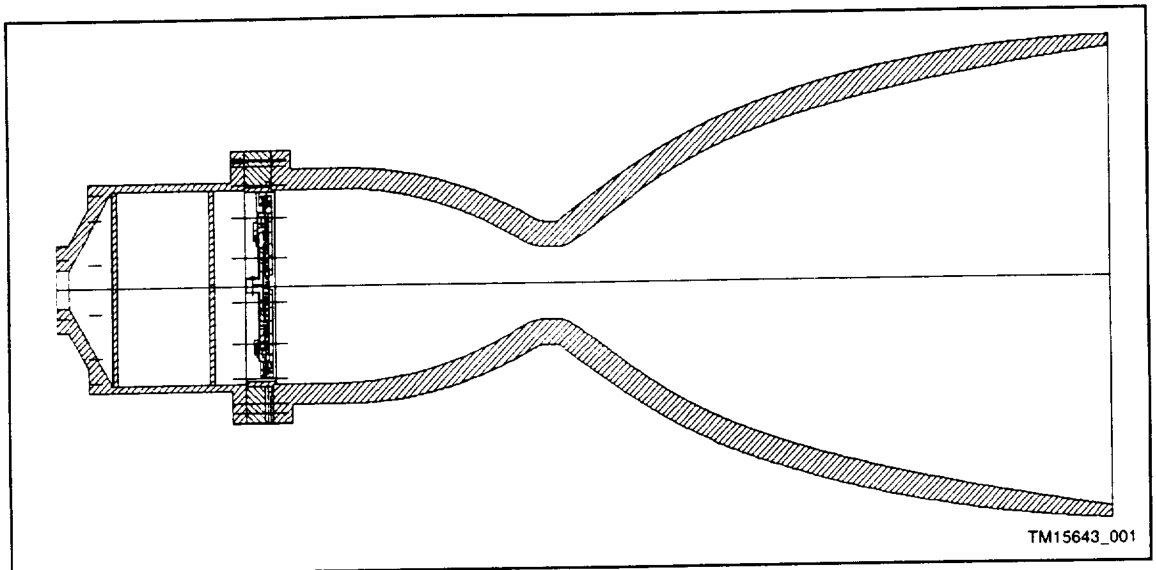

Figure 1. USFE Thrust Chamber Assembly. TCA consists of oxidizer dome, catalyst bed assembly, fuel injector, and ablative chamber/nozzle. The chamber produces 10,000 lbf of thrust in space at a chamber pressure of 500 psia with an expansion ratio of 40.

TABLE 1. DESIGN AND OPERATING PARAMETERS OF USFE ENGINE

Parameter	Value
Propellants	90% HTP/JP-8
Vacuum Thrust, lbf	10,000
Chamber Pressure, psia	500
Mixture Ratio	4.7
Nozzle Expansion Ratio	40 (five for ground tests)
Chamber Contraction Ratio	7.1
Delivered Specific Impulse, sec	275 (in vacuum)
Flowrate, lb/s	36.0
Burn Time, sec	200
Engine Envelope	60 in. long, 40 in. diameter

Test Facility is a versatile two-cell test complex for component development testing of combustion devices, rocket engine components and small engines and boosters. Cell 2 features a skid-based design concept in a vertical-fire configuration. In this concept, all test specific hardware (run tanks, run lines, and test article) are mounted to a platform that is bolted above the existing 8 ft. wide by 17 ft. deep concrete flame bucket. The existing platform contains a 500 gallon oxidizer run tank and a 250 gallon fuel run tank. At the nominal flow, a maximum run duration of 150 seconds was possible. Figure 3 shows a schematic of the run line instrumentation and flow control.

4. Engine Test Configurations

Table 2 lists the configurations used throughout the development testing. Two engine dome/catalyst bed designs were tested. The first dome/catalyst bed shown in Figure 4 is a catalyst bed insert design to facilitate rapid catalyst bed change outs during testing. The second design is an integrated dome/catalyst bed (Figure 5). The internal flow geometry of the integrated configuration (also referred to as "flight-like") will be retained for the flight engine. Four fuel injector designs, three of which are shown in Figures 6-8, were tested. The first was a steam-port design. The second injector tested was a ring/steamport design that utilized fuel film cooling to control throat heating in the nozzle. The inner region utilized a steam port design similar

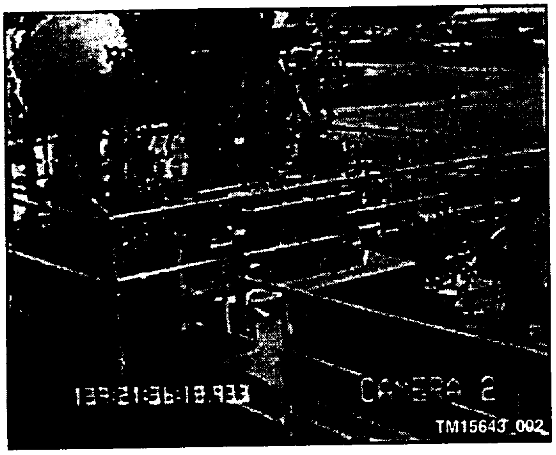

Figure 2. Thrust Chamber Testing of USFE at E-3 Test Cell at Stennis Space Center. Chamber produces approximately 7300 lb$_f$ of thrust at sea level with a 5:1 nozzle expansion.

to the first injector. A third injector whose design was a derivative of the Gamma engine was tested. After being damaged in the first series of development testing, the ring/steamport injector was redesigned (Figure 8). The new all ring design met all performance requirements and was successfully used on the 150 second test at the end of development testing.

Three combustion chamber/nozzle assemblies were used during development testing. The first is the copper heat sink chamber. This chamber is an instrumented (pressure and temperature) chamber used for short duration testing. The second design tested was constructed of silica phenolic with a glass phenolic overwrap as shown in Figure 9, and the third had a quartz phenolic insert in the throat.

5. Test Results

5.1. First Development Test Series

Key test results are shown below. All of the tests were conducted with 85% concentration peroxide.

- 300 seconds of bipropellant operation using ablative chambers
- Over 700 seconds of run time on a single catalyst bed
- Demonstrated throat recession rates of less than 0.001 inch/sec at O/F 5.85
- Demonstrated multiple restarts
- Demonstrated throttling to 10% in monoprop mode and 20% in biprop mode

5.1.1. Injector Performance

Figure 10 shows C* efficiency versus oxidizer to fuel ratio for the steamport and ring/steamport injectors. The data show a trend of decreasing efficiency with increasing O/F ratio. Table 3 shows the throat heating rate comparison for the steamport and ring/steamport injector as a function of O/F. As shown in the table, the ring/steamport injector provides a considerable improvement in throat heating rate as the O/F increases.

5.1.2. 140 Second Test

This long duration test was completed 19 May 1999. The engine configuration in this test was configuration 2 (Table 2). The O/F ratio was 4.9 and the chamber pressure was 495 psi. The chamber pressure varied over the course of the test. At the end of the test the pressure was 35 psi less than the starting pressure. This pressure drop was due to the erosion in the throat of the nozzle. Figure 11 shows a post test photo of the nozzle. The view is from the nozzle exit plane looking into the engine. As shown in the figure, there was substantial silica

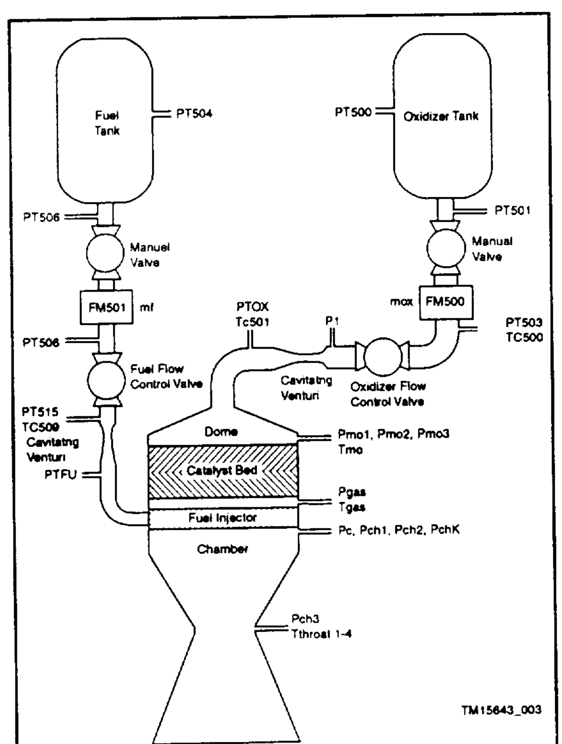

Figure 3. Schematic Diagram of the Engine and Test Stand

TABLE 2. ENGINE CONFIGURATIONS TESTED

Configuration No.	Dome/Catalyst Bed	Injector	Chamber
1	Insert	Steam Port	Copper
2	Insert	Steam Port	Silica Phenolic
3	Insert	Ring/Steam Port	Copper
4	Insert	Ring/Steam Port	Silica Phenolic
5	Insert	Ring	Copper
6	Insert	Ring	Silica Phenolic
7	Flight	Ring	Copper
8	Flight	Gamma Derivative	Copper
9	Flight	Ring	Silica Phenolic
10	Flight	Ring	Silica/Quartz Phenolic

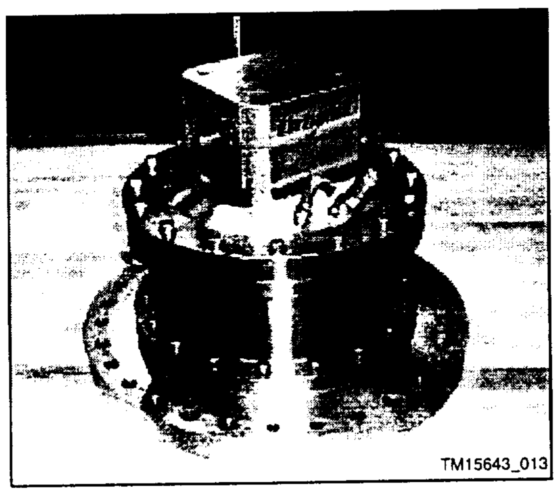

Figure 4. Catalyst Bed Insert and Dome Assembly

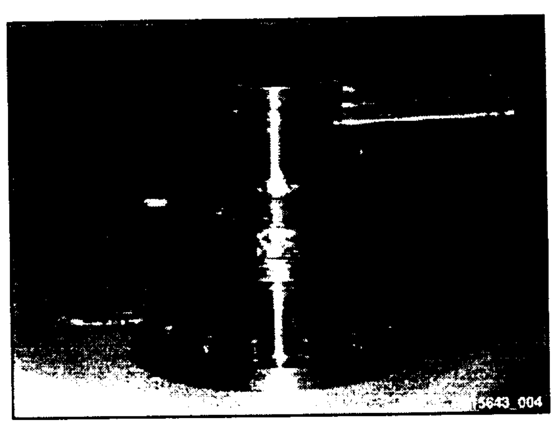

Figure 5. Integrated Dome and Catalyst Bed

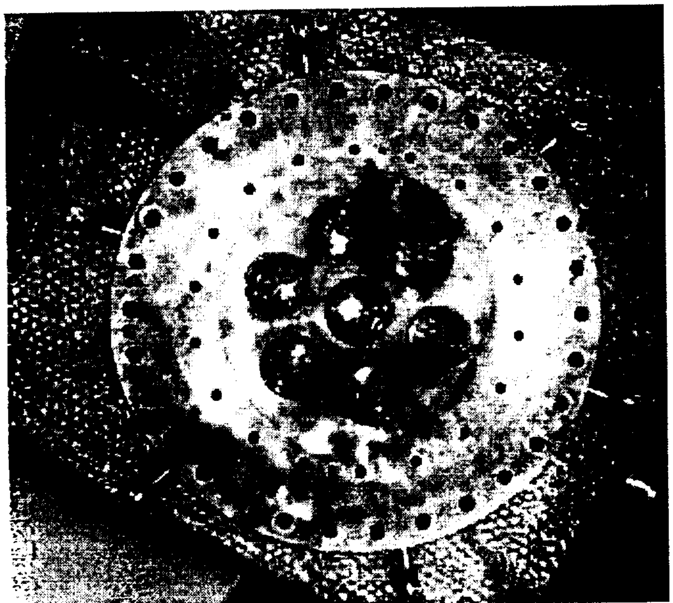

Figure 6. Steam Port Fuel Injector

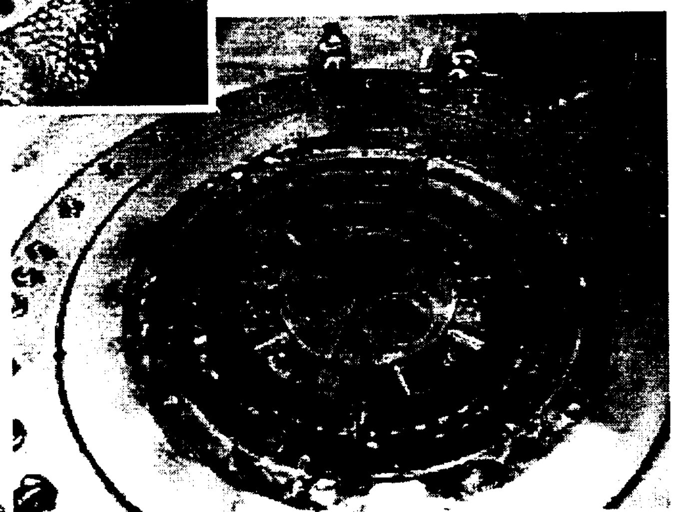

Figure 7. Ring/Steam Port Fuel Injector

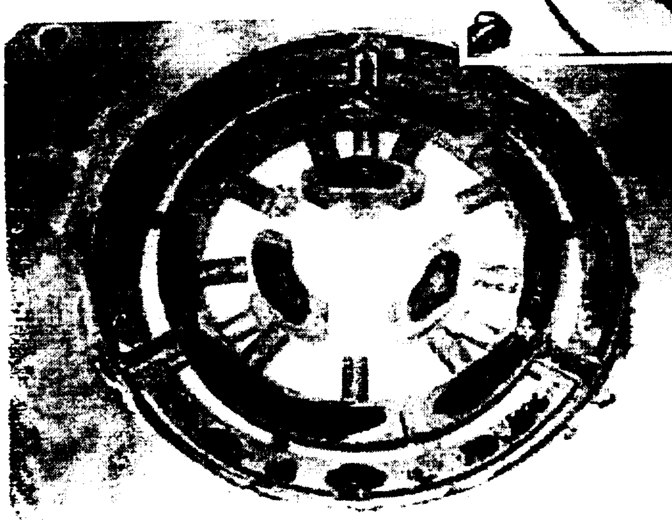

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Figure 8. Ring Fuel Injector

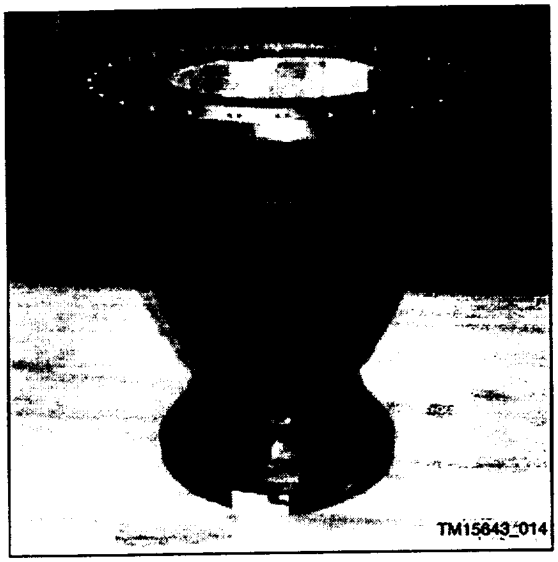

Figure 9. Composite Nozzle

TABLE 3. HEATING RATE COMPARISON FOR KM AND RING INJECTORS

INJECTOR	O/F ratio	Temp Rise [1] (°F)
Steam Port	4.1	200
	5.3	552
	5.5	595
	7.3	844
Ring/ Steamport	4.9	240
	5.8	265
	6.1	308

Notes:

1. Temperature increase at 4 seconds of bi-prop operations

flow over the entire throat. One area of the throat shows particularly significant material ablation.

The results from the duration test, along with the throat heating data, indicate that the nominal O/F ratio would have to be 4.5 (with 85% peroxide) or less to meet the throat erosion constraints.

5.1.3. 31 Second Test

This test was conducted with engine configuration 4 (Table 2). The average O/F ratio during this test was 5.85 with a throat erosion rate of 0.0009 inch/ second. Figure 12 shows a photo of the nozzle from the exit plane. As shown in the figure, the nozzle throat has a small amount of silica melt. The localized heating is aligned with the fuel supply tubes for the mid and inner injector rings. The injector performance was acceptable in this test, but significant heat-induced material erosion around the steam ports led to the redesign shown in Figure 8.

5.2. Second Development Test Series

This test series was conducted with 90% peroxide, thus the injector/chamber design had to be verified at the higher operating temperature. The test

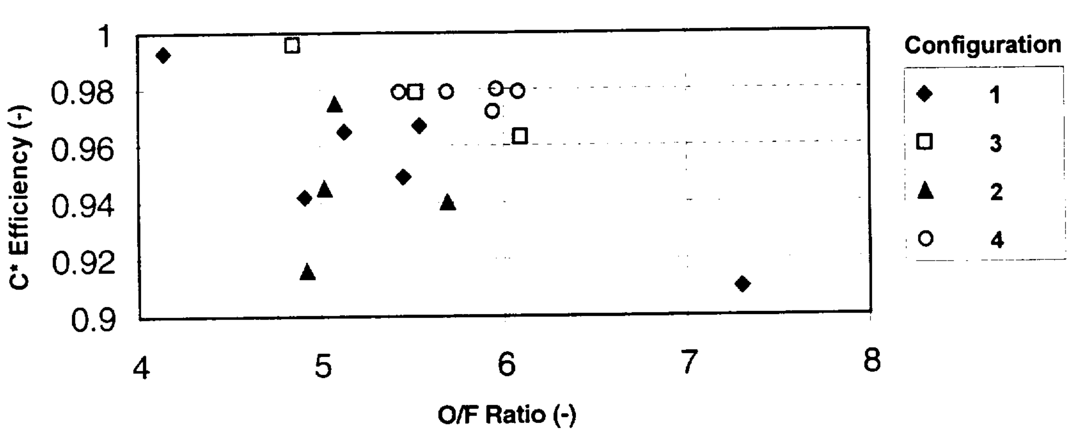

Figure 10. C* Efficiency Vs. Oxidizer Fuel Ratio

Figure 11. Post Test Photo of 150 Second Test Nozzle

results are broken down into catalyst beds, injectors, start-up characteristics, and ablative chamber performance in a long duration (150 second) test.

5.2.1. Catalyst Beds

Two catalyst bed configurations were tested in this test series. The first catalyst bed tested was the cartridge or insert configuration. A flight-like integrated dome and catalyst bed was also tested.

Figure 13 shows the catalyst bed pressure drop in bi-prop operation. The data in Figure 13 are based on operations with the ring injector only. The average pressure drop for the flight-like unit is 136 psi. The average pressure drop for the cartridge catalyst bed is 164 psi.

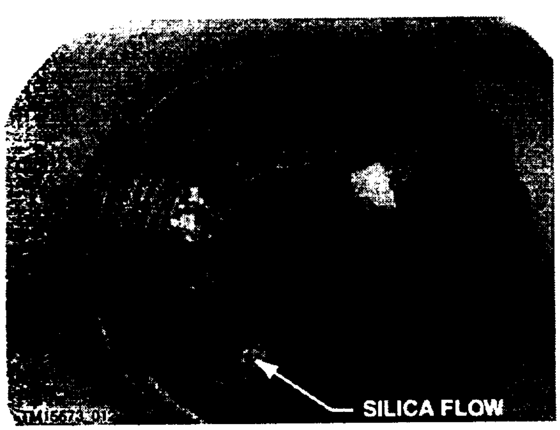

Figure 12. Post Test Photo of Nozzle

Figure 14 shows the flight-like catalyst bed pressure drop as a function of accumulated test time. The ordinate is the ratio of the pressure drop divided by the mass flux. This function is selected to normalize the pressure drop with mass flux. There is a clear trend in the figure that shows an increased pressure drop as a function of catalyst bed test time.

5.2.2. Injector Performance

The ring and Gamma derivative injector plates were tested in this test series. Two important performance criteria in this engine are C* efficiency and throat heating rate. High performance (high C*) comes with a penalty in throat heating rate. Clearly, there is a balance between efficiency and throat heating when an ablative nozzle is used in a mission with 200 seconds of burn. In this case, the allowable throat erosion rate (based on radius) is 0.001 inch/second. At this erosion rate, the thrust requirement of 10,000 lbf with a 10% uncertainty can be achieved using cavitating venturis.

Figure 15 shows the C* efficiency of the ring and Gamma derivataive injectors as a function of O/F. There is a trend of decreasing efficiency at O/F ratios greater than 5. It is clear that the Gamma derivative performs very well and the efficiency is near 1.0 at an O/ F ratio of 5.

Table 4 shows the heating rates from the second series of development testing. The table shows the heating rate for the ring injector with the cartridge and flight-like catalyst beds. There appears to be a trend in the data that indicates that the heating rate is lower with the flight-like catalyst bed. It is known from the composite chamber testing, to be discussed later, that the injector had a hot zone (120 degrees or 1/3 of the circumference). The most probable reason for the heating rate discrepancy between the two catalyst beds is that the injector was turned relative to the throat thermocouples (located at two locations 90° apart) between configurations. As mentioned earlier, the Gamma derivataive injector had a very high efficiency. However, the throat heating rates were above the target heating rate of 250 °F, which was shown through data analysis to correspond to the target throat erosion rate.

The vacuum specific impulse for the ring and Gamma derivative injectors was 275 and 298 seconds, respectively. The combination of good C* and reduced throat heating make the ring injector

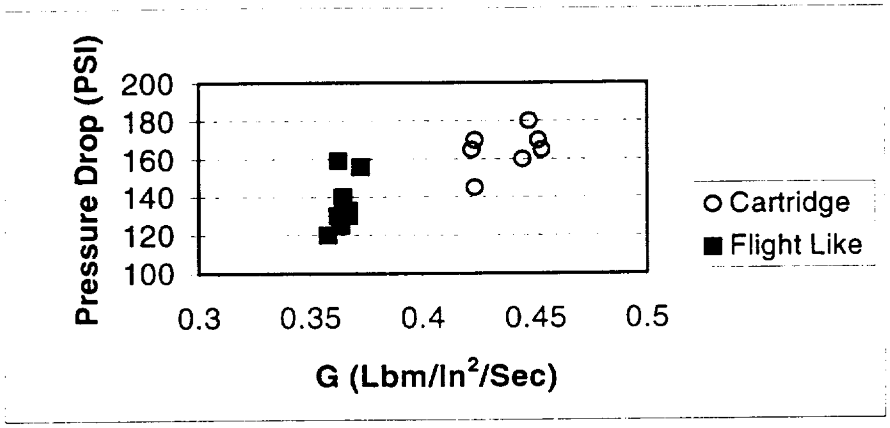

Figure 13. Catalyst Bed Pressure Drop in Bi-Prop Operation

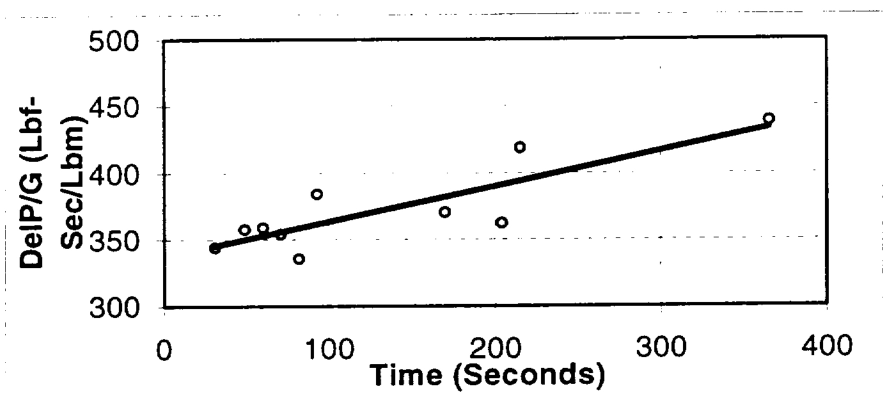

Figure 14. Flight-like Catalyst Bed Pressure Drop As A Function of Accumulated Test Time

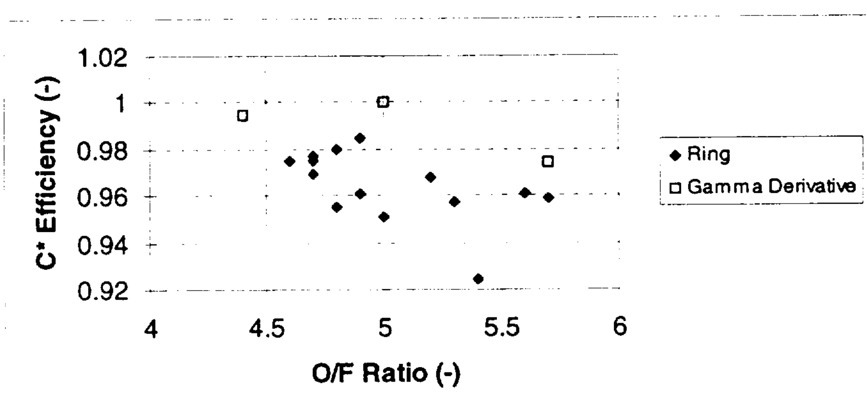

Figure 15. C* Efficiency

TABLE 4. THROAT HEATING RATES FOR DVT-2

INJECTOR	CATALYST BED	O/F RATIO[2]	TEMP RISE [1] (°F)
RING	Cartridge	4.94	76
		4.96	137
		5.31	179
		5.42	158
RING	Flight-Like	4.72	71
		5.15	100
		5.71	86
		6.16	103
GAMMA DERIVATIVE	Flight-Like	4.4	725
		5	475
		5.7	880

the best choice for this application. However, the Gamma derivative injector shows very good performance and has potential in a regeneratively cooled nozzle application, or perhaps with a composite nozzle and the addition of more fuel film cooling.

5.2.3. 150 Second Test Results

The long duration test was the last test of this test series. The purpose of this test was to verify operation of the injector and composite nozzle for a long duration burn. The flight engine profile requires a 200 second burn. Ideally, the long burn test at SSC would simulate the flight burn time. However, the peroxide tank limits the allowable run time to 150 seconds.

Figure 16 shows pretest photos of the silica/quartz phenolic chamber. Quartz phenolic is only used in the throat region with the highest heat flux due to the increased cost of the material. Figure 17 shows post test photos of the nozzle. Localized erosion is limited to an area approximately 120 degrees in circumference. The hot zone can be traced from the throat to the injector plate. An inspection of the soot formation on the chamber wall and on the injector plate supports the theory that an outer zone of the injector plate is running lean and not providing adequate cooling along the wall. Water flow tests of the injector plate will provide additional information to determine the cause of the hot zone. Results from the water flow tests will be folded into the flight injector design.

Figure 16. Silica/Quartz Phenolic Nozzle

Figure 17. Silica/Quartz Phenolic Nozzle Post Test

6. Conclusions

This test series has demonstrated the ability of this engine design to meet the mission goals. A flight-like dome was tested and performed well. The ring injector performed well with a C* efficiency of 97%. The tests show that there is a hot zone in the injector that requires some manifold improvements in the flight design. The silica/quartz phenolic chamber performed very well. With the exception of the localized hot spots, the throat showed very little erosion and qualitatively looked much better than the previous silica phenolic nozzles.

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