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Demonstration Testing of a Long-Life 5-Lbf (22-N) MR-106L Monopropellant Hydrazine Rocket Engine Assembly

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Aerojet has conducted demonstration testing to flight-qualify the MR-106L, the new model designation for the Long-Life MR-106E 5-Lbf (22-N) thrust class monopropellant hydrazine (N_2H_4) Rocket Engine Assembly (REA). This engine features a thruster with a dual catalyst bed decomposition chamber comprised of spontaneous and non-spontaneous catalyst, a dual seat solenoid valve, a dual element valve heater, two valve thermistors, a dual element catalyst bed heater, and a Platinum Resistance Temperature Device (RTD) for catalyst bed temperature telemetry. Severe dynamic environments and high throughput requirements were met by using an improved MR-106E thruster design with superior structural and thermal characteristics, using Shell-405 spontaneous catalyst, and by changing catbed screen material, mesh, and wire size for better catalyst retention capability. A single MR-106L Demo REA followed a qualification approach compliant with MIL-STD-1540D and MIL-HDBK-340A including acceptance testing and qualification level functional, vibration, shock, and hot-fire tests. The Demo REA was subjected to a random vibration of 29.1 grms and shock with peaks near 10,000 g's. Limit Duty Cycle (LDC) pulse performance was characterized as a function of on-time, feed pressure, and valve voltage. The lifetest mimicked a blowdown feed system and spanned 400-85 psia (27.6-5.9 bars). The Demo REA accumulated 120,511 pulses and a total impulse of 126,205 lbf-sec (561,388 N-s) with 574 lbm (260 kg) of high-pure hydrazine throughput. The engine ran exceptionally well throughout the lifetest. Like the MR-106E, the MR-106L provides a nominal thrust of 5 lbf (22 N) at 230 psia (15.9 bars) with nominal steady state specific impulse of 234 lbf-s/lbm (2295 N-s/kg).

Nomenclature

C	=	Time to Centroid
c^*	=	Characteristic Velocity
F	=	Thrust
I_{bit}	=	Impulse Bit
I_{sp}	=	Specific Impulse
I_{tot}	=	Total Impulse
P_c	=	Chamber Pressure
P_f	=	Feed Pressure
R_{pp}	=	Maximum Peak-to-Peak Roughness
R_{2S}	=	2-Sigma Peak-to-Peak Roughness
R_{avg}	=	Average Peak-to-Peak Roughness
t_p	=	Time to Pulse (Time to Half Impulse)
T_c	=	Chamber Temperature

Acronyms

ACS	=	Attitude Control System
ATP	=	Acceptance Test Procedure
BOL	=	Beginning Of Life
CT	=	Computer Tomography (Scan)
DC	=	Duty Cycle
D&I	=	Disassembly & Inspection
EOL	=	End Of Life
FOD	=	Foreign Object Damage / Debris
LDC	=	Limit Duty Cycle
MIPS	=	Mechanical Induced Pyro-Shock
MR	=	Monopropellant Rocket
REA	=	Rocket Engine Assembly
RTD	=	Resistance Temperature Device
SOH	=	State of Health

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I. Introduction

Aerojet has successfully completed flight qualification of the Long-Life MR-106E Rocket Engine Assembly (REA), now designated as the MR-106L. The MR-106L is a 5-Lbf (22-N) thrust class monopropellant hydrazine (N_2H_4) engine. This new configuration was developed and qualified for a fast-paced program requiring a simple and robust engine with successful design heritage that would provide broad control authority with significant life capability. A single engine configuration was desired for spacecraft control to minimize schedule risk, simplify the propulsion system, and lower overall cost. As a result, this engine configuration had to provide routine ACS and Delta-V maneuvers with varying duty cycles and significant total impulse. Maneuvers will include steady state thrust, thrust with off-impulse for control, repetitive pulsing, and single pulse impulse bits of varying magnitudes.

The MR-106L is an inherently robust rocket engine with a long and successful heritage in design, qualification, and flight. The MR-106 thruster was originally developed for the HAS/Peace Courage program over 25 years ago. Development of the E-Class thruster started in 1990 for GPS and first flew in 1995. MR-106 engines are currently used on spacecraft (NEAR, Lunar Prospector, Genesis, Mercury Messenger, SBIRS, GPS IIR, GPS IIF, A2100™, various military) and on the upper stages of launch vehicles (Atlas and Titan Centaur, Delta III & IV). Approximately 1500 MR-106 engines have been delivered, of which nearly 400 have flown, accumulating over 600 thruster years on-orbit. In 2002, an internal development effort helped consolidate the MR-106 product family by improving the thruster sub-assembly for superior thermal and structural characteristics. After qualification, thirty-six were delivered. In 2003, the design changes discussed in the next section were made to create the Long-Life MR-106E. Eleven units have been delivered for one program and sixty-four more are in work on three other programs. The Long-Life MR-106E, now designated the MR-106L, has interchangeable thrust and specific impulse with the MR-106E (see Figure 1) but with superior structural, thermal, and throughput capability.

The qualification of the design was validated with analysis, similarity, and demonstration testing. The term "demonstration testing" is used herein to describe the combined MIL-STD-1540D[1] verification methods of test, inspection, and demonstration. The demonstration test program was developed and conducted in compliance with MIL-HDBK-340A[2] in a "test as you fly" approach. A flight-representative development test article (the Demo REA) was fabricated, acceptance tested, and inspected with the same processes and procedures as the delivered flight units before conducting qualification level dynamic environment tests, hot-fire characterization tests, and a full hot-fire life test with margin. This paper presents the MR-106L demonstration test program and the new capabilities, which are shown in Figure 2 and summarized in Table 1.

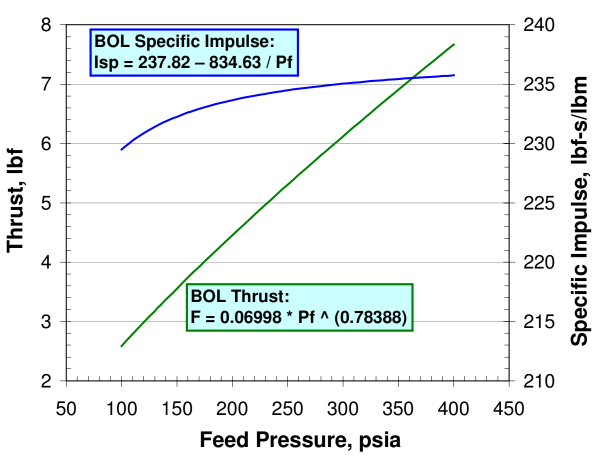


Figure 1: MR-106E & MR-106L nominal steady state performance. *Beginning Of Life (BOL) performance is typically within ±5% of nominal.*

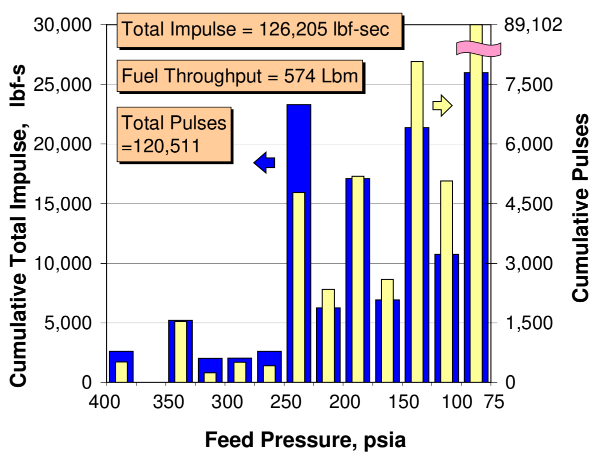


Figure 2: Operation of the MR-106L Demo REA. *Operation mimicked a blowdown feed system from 350-85 psia with 400 psia excursions, + 50% margin, prior to additional operation at lower feed pressures.*

Table 1: Demonstrated Capabilities from the MR-106L REA Demo Test.

Capability	Demonstrated
Propellant	Monopropellant Hydrazine (N_2H_4), High Purity Grade per MIL-PRF-26536E[3]
BOL Thrust (See Figure 1)	7.4 lbf @ 400 psia; 2.5 lbf @ 100 psia (33 N @ 27.6 bars; 11 N @ 6.89 bars)
BOL Specific Impulse (See Figure 1)	236 lbf-s/lbm @ 400 psia; 229 lbf-s/lbm @ 100 psia (2314 N-s/kg @ 27.6 bars; 2246 N-s/kg @ 6.89 bars)
Random Vibration (See Figure 10)	29.1 grms 3 axis, 3 minutes per axis, with catbed heater powered
Shock (See Figure 11)	10,000 g's peak @ 4000-8000 Hz 3 axis, 2 shocks per axis, with catbed heater powered
Total Pulses (see Figure 2)	120,511
Total Impulse (see Figure 2)	126,205 lbf-s (561,388 N-s)
Total Hydrazine Throughput	574 lbm (260 kg)
Total Burn Time	559 minutes
Longest Continuous Burn	4000 sec (500 sec runs x 8, ~1 minute between each)
Smallest Duty Cycle	0.020 sec on, LDC cool chamber to 200°F (93°C), ~7 min
Minimum Pulse Width	0.020 sec
Inlet Pressure Range	400 - 85 psia (27.6 - 5.9 bars)
Propellant Temperature Range	47 - 128 °F (8 - 53 °C)
Valve Voltage	22 - 34 vdc with 10 vdc suppression (suppression was changed to 1.8 vdc for production testing)
Number of Cold Starts	1 @ 70°F (21°C); 617 @ ~200°F (93°C)

II. Design Improvement Effort

The primary objective of the design improvement effort was to identify and implement changes to the MR-106E thruster that would mitigate the risks associated with attaining long life. This was characterized by the need to increase the total impulse of the engine from the previous qualified level of 21,546 lbf-s (95,841 N-s) to 55,000 lbf-s (244,652 N-s) plus 50% margin for a total of 82,500 lbf-s (366,978 N-s). The total number of required pulses was 20,000 plus 50% margin for a total of 30,000. These objectives were realized by making changes to the catbed and bedplate screens, selecting Shell-405 spontaneous catalyst for the upstream catalyst bed, and using high-purity grade hydrazine per MIL-PRF-26536E[3]. Future production builds will use either Shell-405 or S-405[†].

To improve catalyst retention capability, changes were made to the metal material, wire size, and mesh size of the catbed screen and bedplate screen. Catalyst retention can be compromised by screen damage caused by nitriding of the metal. Nitriding means that a material has become embrittled by reaction with nitrogen, which worsens with increased time at high temperatures in a nitrogen rich environment (e.g., catalytic decomposition of hydrazine). Nitriding of screens can cause them to lose ductility, which can lead to cracking, tearing, and/or dislodging from their welds. This in turn can cause the spontaneous and non-spontaneous catalyst beds to mix, or the loss of catalyst from the chamber. Either of these would result in degraded performance, possible engine washout, and shortened life. The new screen material was chosen for its known superior nitride resistance[5,6]. It also forms and welds easily and is a material with which Aerojet has significant experience. In addition, the screen mesh size was selected to provide a larger wire diameter, providing additional margin since nitriding occurs by surface penetration. Finally, the mesh size was selected to help trap fractured catalyst particles. A conservative prediction indicated that the screens would adequately resist nitriding damage and contain sufficient catalyst for smooth operation through at least 100,000 lbf-s (444,822 N-s). This prediction was validated with margin during the test program.

[†] References to "S-405" identifies catalyst produced by Aerojet under license from Shell Oil Company.[4]

The second change to achieve higher throughput was to use Shell-405 14-18 mesh catalyst for the upstream catbed. Shell-405 catalyst is recommended by the "Long Life Manual"[5] and other reports[6-8] for use in long life catalyst engines and is generally believed to have better life capability than the Aerojet LCH-227 previously used in the MR-106 engine. Although 14-18 mesh catalyst has higher chamber pressure roughness than finer mesh ranges, it was chosen because it has historically been proven stronger and more resilient in long-life applications.

A third design change was more mission-driven and considered secondary to achieving long life. This change was to use a higher power catbed heater recently developed and qualified for another Aerojet engine. Catalyst life can be improved if repeated hydrazine-catalyst reactions occur at start temperatures of 200°F (93°C) or higher[5,6]. Depending on thermal environments and the mission profile, heating with a standard catbed heater may take substantial time to reach this and/or may be inadequate in colder bias conditions. While occasional starts below this are acceptable, and recent testing has demonstrated significant cold start capability for the MR-106 thruster[9], there was still concern with the long life mission profile and the customer's desire to minimize preheat delay time. Since sufficient spacecraft bus power was available, the higher power catbed heater was selected. This heater provides 6.6 watts/element @ 28 vdc, compared to the standard MR-106E heater's 3.0 watts/element @ 28 vdc. Future programs must evaluate which heater is best depending on the mission, environments, and power constraints.

III. Test Hardware

Figure 3 shows the MR-106L Demo REA and Figure 4 identifies the components. Because the program started when S-405 catalyst was just finishing the technology transfer to Aerojet[4], the Demo REA and the first eleven production engines used Shell-405 catalyst. A dual-seat valve is used for control while providing two sealing barriers against leakage. A dual-element valve heater is used to maintain warm temperature to prevent hydrazine freezing. The flight computer makes readings from a thermistor, with a backup for redundancy, to monitor valve temperature in a feedback control loop to the valve heater. An alternative approach is to use thermostats that are directly wired to the valve heater. A dual-element catbed heater is used to pre-heat the catalyst bed prior to valve actuation. A Platinum Resistance Temperature Device (RTD) is used as a catbed temperature sensor.

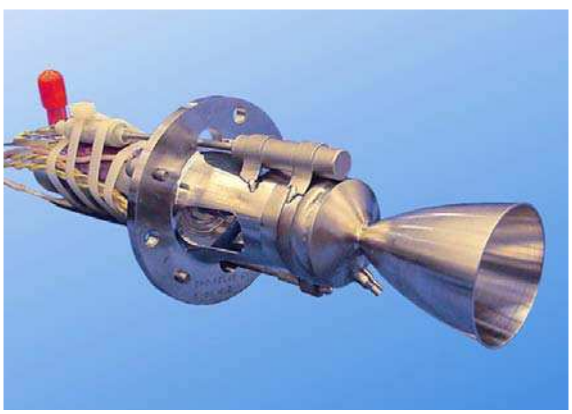


Figure 3: The MR-106L Demo REA.

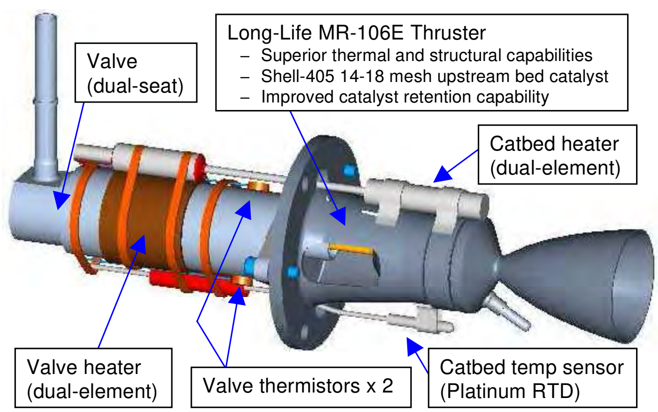


Figure 4: The MR-106L Demo REA and its components.

Although a test unit is ideally identical to that used for production, schedule and cost precluded this option on the demonstration program. Thus, a "flight-like" Demo REA was assembled from available parts and components and all deviations were fully documented and reviewed. The differences were inconsequential to the new capabilities being validated and did not compromise the quality comparison of the Demo REA to the production REAs.

The three critical differences were all part of the thruster subassembly, which is the thruster prior to having catalyst packed and a nozzle attached. The subassembly was rescued from the Material Review Board (MRB), where it was discrepant because of a small tooling blemish on the standoff and because the injector stem height was 6% short. The concern that the short height might alter impulse bits because of a change in the hold-up volume was compensated for by specially sizing the spacers used between the valve and the thruster. The standoff blemish and the slightly thinner spacers were inconsequential to dynamic environment testing, and the resulting hold-up volume was near nominal. The last difference was that the rescued thruster subassembly had thru-holes for mounting the engine, and the final product for the program was to use threaded-holes. To ensure comparable results, the Demo REA was mounted with high torque values during its shock testing. As a result of this review and these actions, the Demo REA was sufficiently representative of the delivered products for the purposes of the test program.

IV. Test Program

Figure 7 presents the test flowplan for the MR-106L Demo REA. In accordance with aerospace standards[1,2], testing followed the same build and test logic expected for the delivered flight products. That is: the Demo REA was built and acceptance tested in the same manner as the production engines prior to beginning qualification level dynamic environment and hot-fire lifetests. These tests were performed in the worst order expected in the product's lifetime. Additional tests were conducted that did not invalidate the qualification tests. These included periodic functional tests that would have helped locate the source or cause of a discrepancy should one have occurred, CT-Scans, and various hot-fire testing to characterize engine performance.

All testing except shock was performed at Aerojet. Hot-fire was conducted in a vacuum chamber simulating an altitude of 300,000 ft (91 km) with a vacuum pressure of ~10^{-3} Torr. Vibration was conducted on an Unholtz-Dickie Vibration Table. Shock was performed off-sight using a Mechanical Induced Pyro-Shock (MIPS) simulator, which uses a pendulum and pneumatic hammer to induce shock.

Figures 8 and 9 show the Demo REA feed pressure blowdown for cumulative total impulse and cumulative pulses, respectively. Hot-fire testing was classified in four groups: Pre-Life (boxes 4 and 10 in Figure 7), Mission (box 11), Margin (box 12), and Extension (box 14). These tests are explained in subsequent sections.

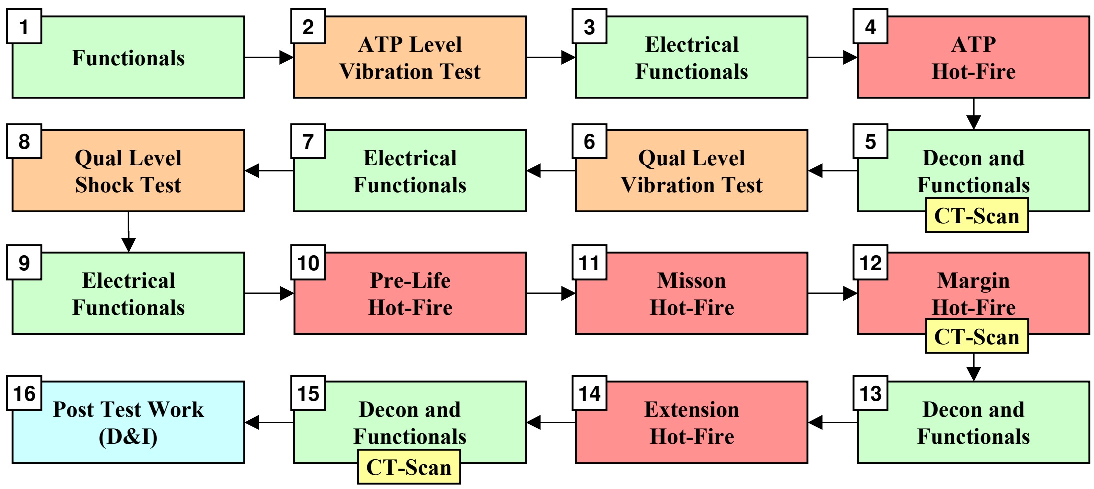
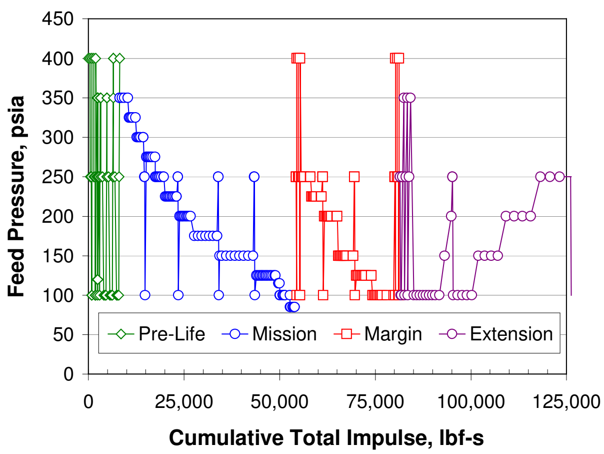


Figure 7: Test flowplan for the Demo REA.



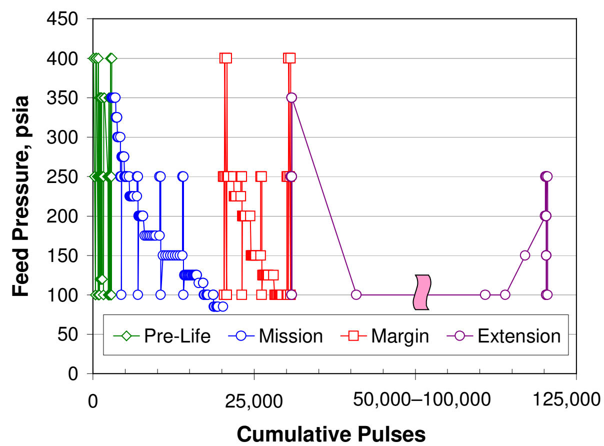


Figure 8: Blowdown impulse for the Demo REA. **Figure 9: Blowdown pulses for the Demo REA.**

"Pre-Life" sequences established Beginning Of Life (BOL) acceptance criteria by firing the same duty cycle as Acceptance Test Procedures (ATP) and included fuel temperature characterization. "Mission" sequences included Limit Duty Cycle performance characterization. "Mission" & "Margin" sequences mimicked the first mission with feed system blowdown from 350-85 psia. "Extension" sequences repeated fuel temperature characterization and was performed in reverse-blowdown to mitigate risk. State-Of-Health (SOH) and ATP hot-fires were periodically conducted to ease evaluation of engine performance throughout life.

A. Functional Tests

The purpose of the functional tests shown in Figure 7 was to ensure the hardware's mechanical and electrical operational requirements were met. Most were periodically repeated throughout the test program to ensure performance had not changed unacceptably due to other tests. Functional tests included internal valve seat leakage, external leakage up-to and including the nozzle with valve seats open, proof pressure testing with valve seats opened and closed, gas flowrate to screen for Foreign Objects or Debris (FOD), insulation resistance and circuit resistance for all electrical components, valve voltages for pull-in and drop-out, dimensional measurements for interface requirements, and workmanship inspections. Functional test requirements were developed with Aerojet's assistance using component specifications and historically acceptable criteria. The test program completed successfully without compromising or failing any of the requirements from these tests.

B. Acceptance Test Procedures (ATPs)

Acceptance testing of the Demo REA covers boxes 1-5 in Figure 7. The ATP was identical to that planned for production units and similar to those used for most Aerojet monopropellant rocket engines. The purpose of ATP is to demonstrate specification conformance and acceptability of each deliverable item. The ATP was designed to precipitate any incipient failures due to any latent defects in parts, materials, or workmanship. As with most liquid rocket engines, acceptance testing included random vibration, hot-fire, and functional tests. All ATP tests completed successfully without issue.

1. ATP Random Vibration

Random vibration is performed to screen for any structural or electrical defects that would occur during launch. Vibration was performed once for each orthogonal axis for 60 seconds at the maximum levels expected during launch. The acceptance level spectrum is shown in Figure 10 and resulted in a gravity load of 14.6 grms.

2. ATP Hot-Fire

ATP hot-fire is performed to validate an engine operates in a smooth, consistent, controlled, and expected manner. It also serves as a wear-in test that detects material or workmanship defects that could occur early in life. For the qualification test, it also serves as a baseline of thruster operating characteristics for all future engines. By repeating ATP hot-fire after qualification level dynamic tests, it may also highlight the possible effect of those tests.

Table 2 presents the ATP hot-fire duty cycle used on the Demo REA and subsequent production engines. The duty cycle was designed to verify workmanship for all conditions of the planned mission while providing additional performance data for the customer. Sequence "0" is used as a conditioning sequence for the catalyst, since this is when it is exposed to hydrazine for the first time. The remaining sequences were organized and designed for minimum test time while maximizing repeatability by ensuring all engines will be operating with the same equilibrium thermal conditions at the end of each sequence, when the most critical performance data is collected.

The calculations and evaluations of hot-fire performance are conducted using methods and terminology compliant with aerospace standards[10-12]. For steady state sequences, the standard requirements of thrust, specific impulse, and chamber roughness were imposed. Both steady state and pulse mode data and performance are reviewed by the program engineer to ensure the engine is acceptable and within family.

The first ATP hot-fire of the MR-106L Demo REA was successful. All performance variables and parameters were within specification and in-family with expected performance, and all post-fire functionals were passed. This concluded the ATP testing portion of the demonstration test program.

Table 2: ATP Hot-Fire Duty Cycle for the MR-106L Demo REA

Seq. No.	Inlet Pressure (psia)	Catbed Start Temp T_c (°F)	On Time (sec)	Off Time (sec)	No. of Pulses	Sightglass Δm Data
0	400	$\geq$200	0.020	0.5	100	N/A
1	400	400±20	60	–	1	N/A
2	250	380-1000	0.020	0.980	150	Last 10
3	250	400±20	60	–	1	N/A
4	100	380-1000	0.020	0.980	150	Last 10
5	100	400±20	60	–	1	N/A
6	400	200-205 ea.	0.020	LDC	10	–

C. Qualification Dynamic Tests

The purpose of qualification dynamic environment tests is to expose a qualification test unit to the extreme levels of vibration and shock that could occur during launch, separation, and solar panel deployment; and to do so for conservatively longer times than expected with repetitions. As with the expected life sequence of flight units, this is done prior to the lifetest. Thus, any dynamic-induced phenomena that might jeopardize life have been included.

A unique requirement applied to both tests was to turn one catbed heater element on at 30 vdc for 12 minutes prior to and during the test while blowing a fan on the engine. The fan simulated the pre-launch fairing cooling system. This test was requested by the customer based on a plan to use the heater in this fashion up-to and during launch, thus allowing immediate use of the engine after spacecraft launch separation. The heater testing occurred without issue with a peak chamber temperature of ~85°F (29°C). After completing each test, the engine was given a single vertical shake with the nozzle down to confirm there were insignificant catalyst fines created.

Subsequent examination, dimensional inspection, and success of the lifetest validated that the high dynamics did not cause any life limiting damage to the Demo REA or its components.

1. Qualification Vibration

The qualification random vibration spectrum is shown in Figure 10. The test was performed once for each orthogonal axis for 180 seconds at the extreme flight launch levels. This was effectively 6 dB above the acceptance levels and resulted in a gravity load of 29.1 grms.

2. Qualification Shock

The qualification shock spectrum is shown in Figure 11. Shock was performed twice for each orthogonal axis at the maximum expected flight levels. The low-end tolerance requirement was imposed and a best effort was made to stay at or above nominal values for at least 50% of the response spectrum. Upper-end tolerances were considered guidelines only, with consideration to avoid extremely high levels of frequencies of specific concern to the valve and catalyst bed. Testing was first done on a mock-up engine to tune the MIPS system for the appropriate response.

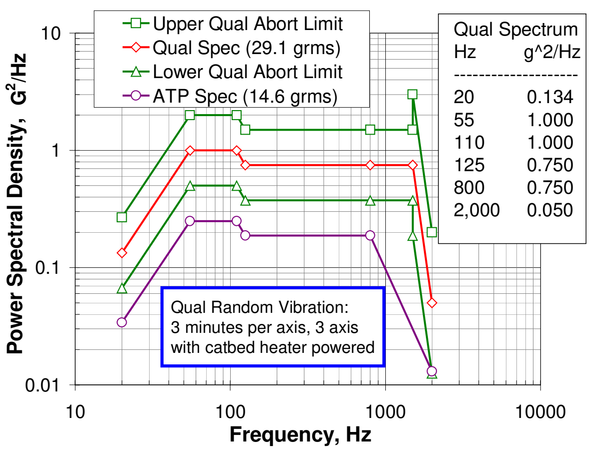


Figure 10: Demonstrated vibration for the Demo REA.

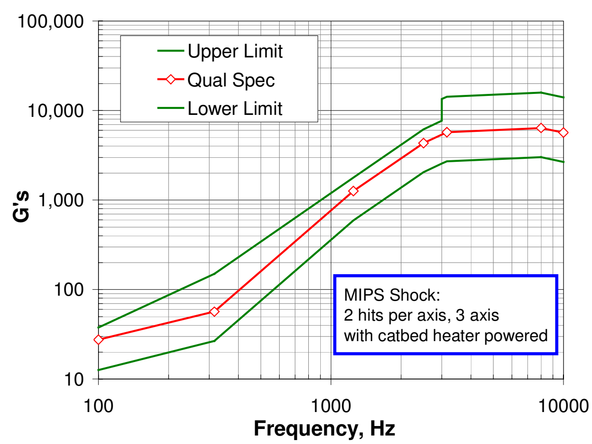


Figure 11: Demonstrated shock for the Demo REA.

D. Pre-Life Hot-Fire Tests

After re-installing the Demo REA into the vacuum test facility, several non-intrusive tests were conducted before the lifetest. The resulting blowdown and pulse profiles are shown in Figures 8 and 9, respectively. The following subsections describe three of these special tests.

1. ATP Hot-Fire #2

This repeat of the ATP hot-fire in Table 2 was conducted to allow a direct comparison of the engine operating characteristics before and after the qual-level dynamic environment tests. The comparison showed the chamber pressure had increased by 40-80%. This increase was not surprising and is believed to be caused by a looser catalyst bed, which would result from shock testing breaking up catalyst particles and creating fines that are blown out during hot-fire. Steady state performance was unchanged. Pulse mode specific impulse and impulse bit may have decreased slightly (2-7%), but were within the trend that continued throughout the lifetest.

2. LDC Pulse Performance Mapping

The primary purpose of this test phase was to characterize the magnitude and trade during Limit Duty Cycle (LDC) operation between impulse bit and impulse bit repeatability as a function of on-time, valve voltage, and feed pressure. This was desired by the customer's ACS team to help plan for flight control.

A secondary interest was that of time-to-centroid (*C*) and the time-to-½-impulse, called the time-to-pulse (t_p). Centroid is the time-leveraged center of impulse[10]. It is reviewed during testing because long centroids can highlight problems such as a leaky valve, voids that are taking a long time to drain, pressure transducer zero shift problems, or even external leaks. It is a parameter of interest during positional change of spinning or rotating spacecraft where the application of force long after the command might affect the maneuver. In contrast, the time-to-pulse is the median impulse time, defined as the time when half the impulse has occurred. This parameter is of interest to three-axis stabilized spacecraft and for the ACS maneuvers of rotating spacecraft. Note that these are incorrectly defined as the same in older aerospace standards and handbooks[11,12]. This is because prior to computers, planimeters were used to calculate area and time-to-centroid was used as an estimate for time-to-pulse. Time-to-centroid is longer than time-to-pulse and is comparatively less repeatable from pulse-to-pulse, test-to-test, and engine-to-engine.

For this test series, a standard set of fifteen sequences was repeated at three valve voltages. Each voltage setting had tests done at three feed pressures. Each feed pressure had five sequences of on-times. Ten pulses were done for each sequence in a LDC mode where the pulse initiated when the measured chamber temperature cooled to ~200°F.

Figure 12 shows the time-to-centroid and time-to-pulse versus commanded on-time as a function of feed pressure. As expected, centroid is more sensitive to feed pressure and less repeatable than the time-to-pulse. The time-to-pulse is nearly linear with on-time while centroid is not. Note that the two parameters slowly approach one another as steady state operation is approached.

Figure 13 shows the impulse bit versus commanded on-time as a function of feed pressure and valve voltage. As expected, LDC impulse bit is more sensitive to feed pressure than valve voltage. Impulse bit non-repeatability averaged around ±0.6%, indicating the characterization was well performed. Non-repeatability would likely increase in-flight where thermal environments and catbed temperature conditions would vary.

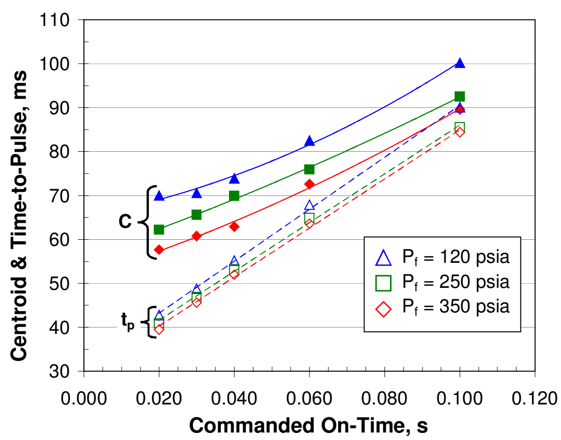


Figure 12: LDC Time-to-Centroid and Time-to-Pulse vs commanded on-time at BOL (28 vdc).

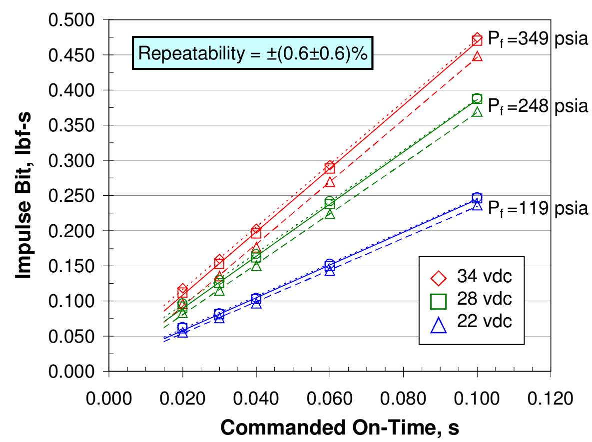


Figure 13: LDC Impulse bit vs commanded on-time at BOL.

Operation was LDC pulses with chamber starts at 200°F. Data is from a production engine using 1.8 vdc valve suppression. The Demo engine had nearly equivalent results at 10 vdc. Both were done at the customer's request.

3. Fuel Temperature Characterization

The purpose of this test was to demonstrate the engine functioned properly with both hot and cool hydrazine. Cold-start testing was not required because of past demonstrations[9]. Tests at various feed pressures and duty cycles were conducted at three nominal fuel temperatures. Steady state runs were repeated during the extension testing when a turbine flowmeter was added. This was necessary because the Coriolis flowmeter measured poorly in the first test because of feed system oscillations created by the use of the non-standard fuel conditioning system. None of the steady state performance parameters had trends with fuel temperature that was worth characterizing. In pulsed operation, hotter fuel caused a slight increase in specific impulse and centroid (by way of longer ignition time and longer decay time). However, these trends were minimal and not worth characterizing. The results validated that the engine was capable of operating with fuel temperatures of 40-120°F (4-49°C) without changing performance.

E. Hot-Fire Lifetest

The purpose of a lifetest is to demonstrate the capability of a design to perform within specification limits for the maximum duration and/or cycles of operation expected in flight. Aerospace standards[1,2] recommend a lifetest for any design where the unit may have performance degradation or a failure mode from wear-out, drift, or fatigue. Test conditions need to simulate those of importance to life and the test should be designed to demonstrate the ability of the unit to withstand maximum operating time and the maximum number of operational cycles predicted during its entire life with suitable margin.

The hot-fire lifetest of the MR-106L Demo REA covers boxes 11-14 in Figure 7. The lifetest was divided into three test phases: Mission, Margin, and Extension. The resulting blowdown and pulse profile for these tests are shown in Figures 8 and 9, respectively. Table 3 presents an operational summary for each test phase.

Table 3: Operation Summary for the MR-106L Demo REA

Test Phase	Feed Pressure Test Range		Total Impulse		Pulses
	psia	bars	lbf-s	N-s	
Pre-Life	400 – 100	27.6 – 6.89	940	4,181	413
Mission	350 – 85	24.1 – 5.9	54,375	241,872	20,349
Margin	250 – 100	13.8 – 6.89	26,339	117,162	10,055
Extension	100 – 250	6.89 – 13.8	44,551	198,173	89,694
		Total:	126,205	561,388	120,511

The mission and margin tests were developed by the customer with Aerojet's assistance. The mission test simulated the nominal blowdown operation expected for the first mission and included mission phases for propulsion initiation, orbit insertion, orbit maintenance, orbit transfer, and spacecraft disposal. The margin test re-started this simulation at mid-mission feed pressure since it is more likely that additional use of engines due to on-orbit anomalies would occur later in life. A margin of 50% was chosen because the mission phase is already conservative with regards to a single engine.

The mission and margin tests were broken up into multiple blocks simulating various flight phases. Each block had multiple sequences that were developed to simulate the expected operation during that block under worst case operation. As a result, the lifetest included operation at steady state, pulsed LDC, and pulsed repetition with one second control cycle modes of various duty cycles, both on-impulse and off-impulse. As planned for the mission, the first pulse of the first sequence started with a catbed temperature of ~70°F (21°C).

The extension test was added by Aerojet to demonstrate the additional life capabilities of the design beyond what was required for the funding program. The extension test was run in reverse pressure blowdown up to the mid-mission pressure. This way, additional cycles could be maximized prior to any event that might terminate the test, which would more likely occur at higher feed pressures.

Table 4 shows the State-Of-Health (SOH) sequences that were periodically conducted to provide a snapshot of engine health throughout life. This SOH test was performed every ~5000 lbf-s (22,241 N-s) throughout the life and margin tests, including one at the beginning and end of each test. One was also done at the beginning, middle, and end of the extension test. To provide additional information, the ATP duty cycle from Table 2 was also repeated at the end of the mission and margin tests. Their placement can be detected in Figures 8 and 9 by noting the data points at 250 psia for SOH and 400 psia for ATP.

Table 4: State-Of-Health (SOH) Hot-Fire Duty Cycle for the MR-106L Demo REA

Seq. No.	Inlet Pressure (psia)	Catbed Start Temp T_c (°F)	On Time (sec)	Off Time (sec)	No. of Pulses	Sightglass Δm Data
SOH-1	250	$\geq$380	60	–	1	N/A
SOH-2	250	380-1000	0.020	0.980	150	Last 10
SOH-3	100	$\geq$380	60	–	1	N/A

The SOH and ATP firings validated that the engine operated acceptably throughout life with relatively stable and smooth performance. End Of Life (EOL) performance met BOL performance requirements, exceeding expectations. Roughness leveled off or began decreasing near 45,000 lbf-s (200,170 N-s); well below expected or disconcerting limits. The following two subsections describe the steady state and pulse mode performance results in more detail.

1. Steady State Operation Over Life

Figure 14 presents the thrust over life for SOH Sequences 1 & 3. Thrust remained relatively constant throughout life, varying only as much as expected due to test-to-test variation. The EOL thrust was within 1.7% of the BOL thrust, well within the 5% that was required at the end of the life test and desired at the end of the margin test.

Figure 15 presents the steady state specific impulse over life for SOH Sequences 1 & 3. Specific impulse was fairly constant through 95,000 lbf-s (422,581 N-s) and then dropped ~5 s (2 %) by the end of testing. Chamber temperature and specific impulse during the extension sequences confirm this was a slow and steady decline. Note that the specific impulse climbed slowly between 50,000-95,000 lbf-s (22,241-422,581 N-s). Characteristic velocity had the same trend, which indicates a gradual decrease in catalyst activity that resulted in less ammonia dissociation and therefore less endothermic losses. The slow decrease after this suggests catalyst activity dropped below a critical threshold (less exothermic gains) and/or a looser catalyst bed and/or void(s) resulting in deeper hydrazine penetration. The EOL specific impulse met the BOL requirements, validating acceptable fuel efficiency over life.

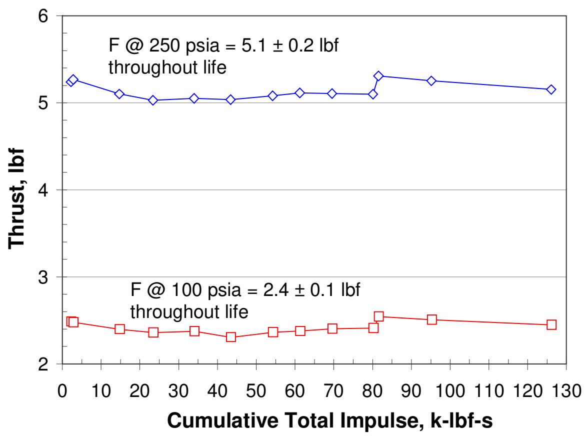


Figure 14: SOH thrust over life.

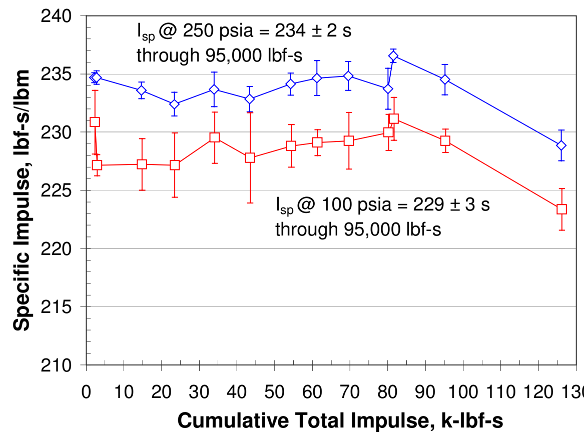


Figure 15: SOH specific impulse over life.

Error bars indicate the 95% variance in measurement.

Figures 16 and 17 present chamber pressure roughness over life for SOH Sequence 1 at P_f = 250 psia (17.2 bars) and SOH Sequence 3 at P_f = 100 psia (6.89 bars), respectively. Roughness is shown in three different ways as both a percent of average chamber pressure and in absolute magnitude. The maximum peak-to-peak roughness is the standard specification method, capturing the worst-on-worst roughness including spikes, but has no statistical value for evaluating average roughness over life. The average peak-to-peak roughness is the average of the maximum values in each one second interval and reduces the influence of spikes to show how maximum roughness changes with life. The error bars show where 95% of the one-second increment peak-to-peak values occurred. The average 2*Sigma roughness is the average of the chamber roughness within two standard deviations of the average and virtually eliminates the effect of minor spiking to better characterize true random roughness over life. The results show that at ~45,000 lbf-s (200,170 N-s), the maximum peak-to-peak roughness began to decline and the random roughness leveled out. Review of the maximum chamber pressure overshoot in the first second of operation showed it too was steady with (–11±12)% of average P_c = 135±4 psia at P_f = 250 psia (P_c = 9.31±0.3 bars at P_f = 17.2 bars), and (+22±26)% of average P_c = 65±1 psia at P_f = 100 psia (P_c = 4.5±0.1 bars at P_f = 6.89 bars).

These results show that steady state operation and performance of the MR-106L Demo REA was relatively smooth and stable throughout the lifetest.

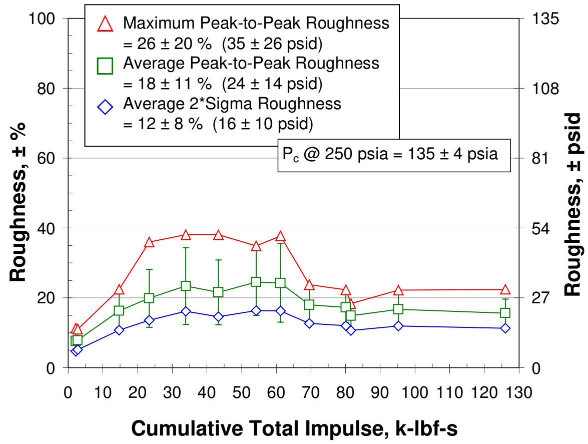


Figure 16: SOH chamber pressure roughness over life at P_f = 250 psia.

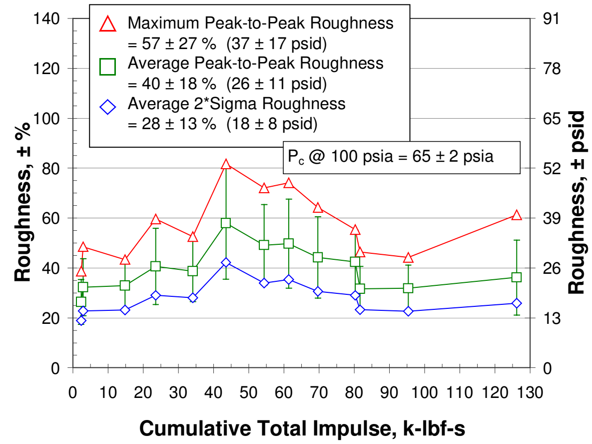


Figure 17: SOH chamber pressure roughness over life at P_f = 100 psia.

Chamber pressure roughness is shown in three different ways over life as a percent of average chamber pressure (left ordinate) and in absolute magnitude (right ordinate). The results show that roughness leveled out or decreased after ~45,000 lbf-s (200,170 N-s) and the engine ran acceptably smooth throughout life.

2. Pulsed Operation Over Life

Figure 18 presents the impulse bit and specific impulse over life for SOH Sequence 2. Both parameters dropped slowly and steadily throughout life. This is most likely caused by a slow decline in catalyst bed activity and the development of a looser bed and/or increasing void size in the upstream catalyst bed. A less active catalyst bed will be more obvious during short pulses and startups when the chamber is cool, and would result in a slower ignition time. "Void" space can result in deeper penetration and/or can fill with hydrazine that "drains" slowly / inefficiently. Both effects become less noticeable for longer pulses and for steady state runs where the catbed temperature is higher and the overall affect of a small loss in impulse bit and fuel efficiency is smaller.

Figure 19 presents the corresponding response times over life and supports these hypotheses. Ignition time drops very slowly over life, while decay time increased until ~45,000 lbf-s (200,170 N-s). These are consistent with a decrease in performance including an increase in time-to-centroid, which is also shown.

The gradual decrease in pulsed performance validated that the engine slowly and gradually lost performance without sudden trend changes that would be indicative of imminent failure. EOL performance was acceptable.

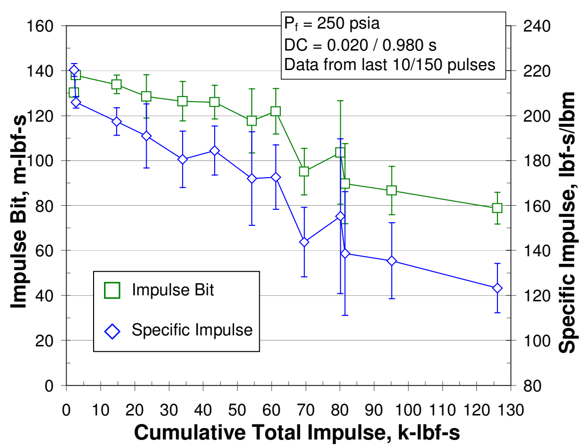


Figure 18: SOH pulse sequence impulse bit and specific impulse over life.

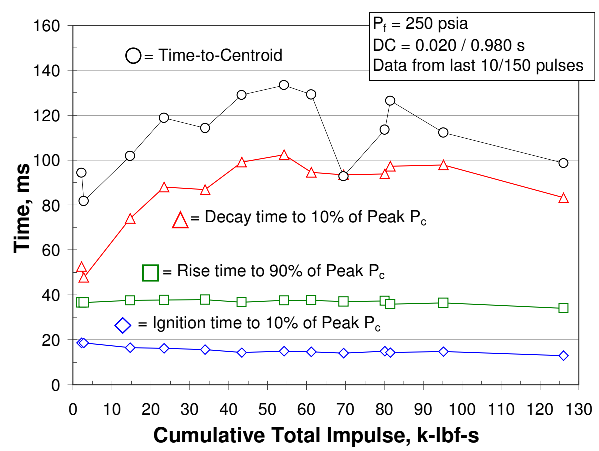


Figure 19: SOH pulse sequence centroid and response times over life.

High duty-cycle pulsed performance dropped slowly and steadily throughout the lifetest. While rise time and ignition time remained constant, the decay time (and thus centroid) increased slowly until ~45,000 lbf-s (200,170 N-s) before leveling off and then decreasing. This is indicative of the development of a looser bed and/or void that eventually stopped growing, loosened, or moved. ATP LDC firing sequences had similar general trends.

F. Post Test Work

After completing hot-fire testing, the functional tests described in Section A were repeated, and all passed. The engine underwent a final CT-Scan prior to Disassembly & Inspection (D&I). This work was done to characterize the condition of the engine at the end of the lifetest.

1. CT-Scans

The purpose of periodic CT-scans was to provide a non-destructive inspection of the catalyst bed and screens throughout life. Inspection of the catalyst bed was desired to quantify the looseness of the bed and any voids that might be present. Inspection of the screens and the surrounding catalyst was desired to validate the screens were intact and in place and that the upstream and downstream catalyst beds were remaining contained and separated.

As shown in Figure 7, three CT-Scans were made throughout the test program. The first scan was made after the initial ATP hot-fire (before the lifetest) and provided a baseline for what a normal delivered thruster would look like. The catalyst was firmly packed without signs of voids. The second scan was made after completion of margin testing. It highlighted the development of a small void in the upstream catalyst bed. The third scan was made after extension testing and showed how much the void had grown. In all scans, the screens were intact and in place.

Table 5 presents estimates of the void sizes from the last two CT-Scans. The void size is relatively small and tolerable considering the amount of throughput attained with good performance.

Table 5: Upstream-Bed (Shell-405 Catalyst) Void Size from CT-Scan Analysis of the MR-106L Demo REA

Condition	Cumulative Pulses	Cumulative Total Impulse	Cumulative Fuel Throughput	Void Size as % of All Catalyst	Void Size as % of Upstream Catalyst
Post Life & Margin	30,654	81,179 lbf-s (361,102 N-s)	367 lbm (166 kg)	3.2 %	5.5 %
End of Lifetest	120,511	126,205 lbf-s (561,388 N-s)	574 lbm (260 kg)	6.3 %	11.0 %

2. Disassembly & Inspection (D&I)

The purpose of D&I is to inspect critical materials, parts, and components for anomalous conditions and to evaluate their integrity. The critical areas of concern are generally those parts subject to fatigue failure. Because the components have been through worse life-cycle qualifications and passed, the region of concern on the Demo REA was the chamber and catalyst. Thus, D&I was performed on the thruster only.

D&I was accomplished by first removing the thruster from the valve followed by cutting the nozzle from the thruster. The internal components and catalyst beds were then removed one at a time and separately contained. General conclusions include:

- The upstream catalyst bed (Shell-405) void was small and agreed well with the CT-Scan estimates.
- No visible void was seen in the lower catalyst bed (LCH-202 catalyst).
- All O-rings and sealing surfaces were in excellent shape and still functional.
- Nitriding had embrittled the screens but they remained in position without breaking, deformity, or loss of function. The lack of cracking or distortion indicates they very likely had additional life capability.
- Nitriding had embrittled the Pc tube enough that it broke in half with normal handling. This was not a concern since delivered flight products do not have a Pc tube. However, any future test for additional throughput capacity would need to resolve this with a possible material change.
- Small cracks had developed in the chamber near the bedplate retaining feature. The largest was <0.001" (.025 mm) wide and ~0.2" (5 mm) long, extending ~1" (25 mm) around the chamber's inner circumference. The chamber was also thinning. These issues were likely caused by a combination of nitriding and creep deformation. Sufficient margin existed such that there was no concern regarding the success of the lifetest. However, additional life objectives may require an increase to wall thickness.

The catalyst was further tested and analyzed to evaluate how various characteristics changed due to the lifetest. The changes are summarized in Table 6. Surface chemistry analysis showed activity was still sufficient to permit additional efficient throughput. However, the decrease since the BOL validates conclusions from the SOH tests that catalyst activity was decreasing slowly and would likely continue to do so, resulting in a continuous and gradual decrease in performance.

Table 6: Changes in Catalyst Properties (BOL ⇒ EOL) for the MR-106L Demo REA

Catalyst Bed	Weight	Sieve Test		H_2 Chemisorption	Surface Area
		14-18	< 14-18		
Upstream Shell-405	–16%	–31%	+700%	–59%	–57%
Downstream LCH-202	–18%	–47%	+1304%	–56%	–51%

V. Conclusions

The Aerojet MR-106L Rocket Engine Assembly (REA) has been fully flight qualified. This 5-Lbf (22-N) thrust class monopropellant hydrazine rocket engine leverages the historically successful MR-106E design. The MR-106L has interchangeable thrust and specific impulse with the MR-106E but has superior structural, thermal, and throughput capability. In addition to the standard uses typical for the MR-106E, the MR-106L is well suited for long-life missions that require routine ACS and/or Delta-V maneuvers with frequently changing ranges of total impulse. The effective thrust or single-pulse impulse bit can be varied across a broad range of duty cycles to provide flexible control authority without concern of degrading engine life. This can simplify propulsion systems and lower cost by eliminating the need for multiple engine configurations with different thrust classifications. The heritage, technical superiority, quality, and flexibility of the MR-106L REA make it a value worth consideration for many missions.

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