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Monopropellant Hydrazine 700 lbf Throttling Terminal Descent Engine for Mars Science Laboratory

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A technology program conducted by Aerojet and the NASA Jet Propulsion Laboratory has been underway since 2001 to develop a rocket engine for terminal descent of an exploration vehicle to the Martian surface in 2009 for NASA's Mars Science Laboratory mission. The MR-80B monopropellant engine is fueled by hydrazine with a throttle range of 7 to 810 lbf thrust. The design is based on the terminal descent engine used on the highly successful 1976 Viking Landers. The primary objective of the technology program is to resurrect the 1976 design, make improvements, and demonstrate through hot fire testing that current mission performance requirements can be met. The original Viking terminal descent engine included multiple small exhaust nozzles to diffuse the exhaust plume to eliminate a recirculation zone that could contaminate spacecraft instrumentation and avoid potential extensive erosion of the landing site. The new upgraded engine utilizes a single nozzle to improve performance, since landing site erosion concerns have been eliminated. Development activity on seven engines is complete and fabrication of flight engines is underway. The Critical Design Review and the formal flight qualification test program are complete pending verification of a late modification extending the engine nozzle. This paper outlines the development testing program with insight into the engine design and performance capabilities and discusses the most recent work, including demonstration of pulse mode operation and the addition of nozzle extensions.

Nomenclature

CT	=	Computed Tomography
EMA	=	Electromagnetic Actuator
F	=	Thrust, lbf or Newtons
GNC	=	Guidance, Navigation and Control
Isp	=	Specific Impulse (lbf-sec/lbm or sec)
lbf	=	Pounds (Force)
MLE	=	Mars Lander Engine
MSL	=	Mars Science Laboratory

I. Introduction

The Mars Science Laboratory (MSL) mission will deliver a robotic rover to the surface of Mars using eight throttleable Mars Lander Engines (MLEs) for the final descent and deployment of the rover. The MLEs will be used for attitude control, deceleration, and hover of the spacecraft. During the hover phase, a rover will be lowered onto the Martian surface. After deployment of the rover, the MLEs will move the spacecraft away from the landing site in a flyaway maneuver. A technology development program started in 2001 leverages the MR-80 Viking engine design and adapts it to the MSL mission.

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The objective of the current effort is to complete delta-qualification of a 700 lbf (3114 N) thrust class throttleable monopropellant hydrazine thruster and produce nine flight engines (8 flight and 1 spare) to perform the powered descent and deployment phases of 2009 MSL mission after exposure to acceptance testing and worst case flight environments. Several thruster parameters and critical MSL requirements successfully achieved during the MLE qualification testing are outlined below as Table 1.

Table 1: MSL Requirements vs. Demonstrated

Parameter	Requirement	Demonstrated
Thrust	90-674 lbf (400-2998 N)	7-810 (31-3603 N)
Specific Impulse	191-217 sec	204-223 sec
Starts	5	7
Burn Time	>220 sec	350 sec
Throughput	342 lbm (155 kg)	612 lbm (278 kg)
Valve Response - 90%	< 80 msec	50 msec

Thruster Description

The MLE is a derivative of the highly successful MR-80 engine used for the two Viking missions to Mars in 1976[1]. The MLE differs from the Viking engine primarily in its single nozzle, modified throttle valve, catalyst bed retention cylinders, and materials. The multiple nozzle configuration of the original Viking terminal descent engine was intended to eliminate spacecraft instrumentation contamination caused by a recirculation zone and to avoid potential landing site erosion[2,3,4]. The change in catalyst from Shell-405 in the Viking engine to S-405 represents a change in manufacturer, but no change in performance[5,6]. A comparison of the Viking and MLE design is shown in Figure 1. Figure 1 also includes photos of the Viking and MLE engines.

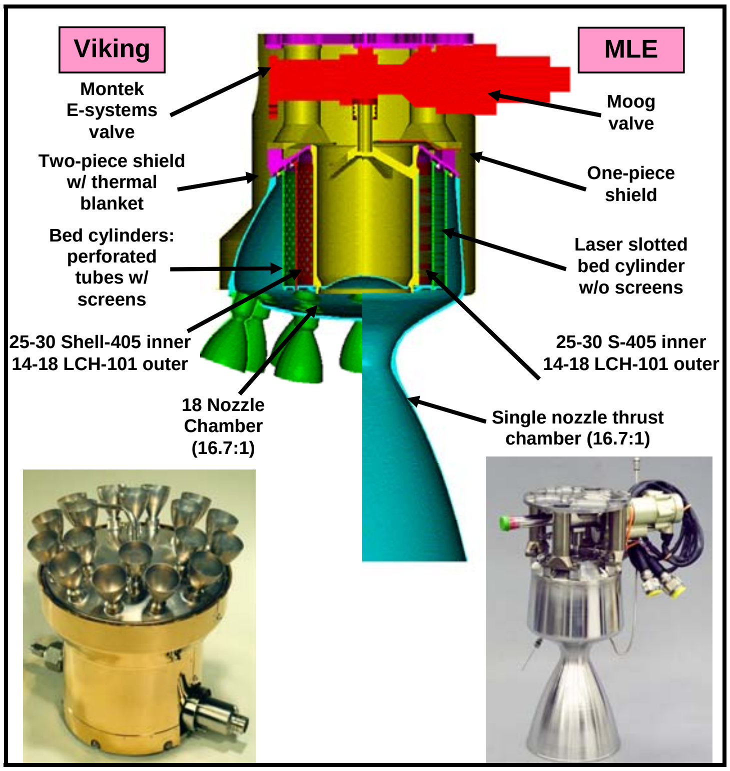

Figure 1. Comparison of Viking and MLE Designs

The MR-80B MLE has a two layer radial outflow catalyst bed design and a single thrust chamber/nozzle assembly. High purity grade hydrazine propellant is distributed to the bed via individual elements equally spaced circumferentially about the thrust axis of the engine. The allowable propellant inlet supply pressure is from 600 to 760 psia (4.14-5.24 MPa). By translating the valve pintle into the throat of a cavitating venturi, the propellant flow-rate can be varied from 0.045 to 3.600 lbm/sec (0.020-1.633 kg/s) to produce an infinitely throttleable thrust range from 7 to 810 lbf (31-3603 N), vacuum. The valve does not have a seat and is not capable of shutting off fuel flow so propellant flow is started and stopped by operating an additional valve located upstream of the throttle valve. A development throttle valve was used during acceptance and qualification testing because the flight valve was not available; however, flight units are in production at Moog. The development valve is functionally the same as the flight valve design and differs from the flight design only by the electrical interface and the electromagnetic actuator (EMA) housing. The engine has a dual element valve and catalyst bed heaters, and redundant catalyst bed and valve platinum temperature sensors.

The MLE development program consisted of seven engines which resurrected the Viking engine technology and expanded the operating range demonstrated for Viking to meet the MSL mission. A summary of these development engines is listed below as Table 2 along with the development time frame, test achievements and some engine characteristics. The first goal of the program, accomplished using the Viking 015R and Viking 020 engines, was to demonstrate the capability to increase the thrust from 635-685 lbf (2825-3047 N) by increasing the propellant flow-rate through the engine. The second goal of the program was to demonstrate successful operation and map performance of a single nozzle design with a new Moog throttle valve. This goal was achieved through completion of the MLE Development engine #1 (Dev#1) test series. The remaining development engines listed in Table 2 explored startup transients, effects of hot fuel, cold fuel, Helium saturation, and operation at the 1% throttle position. The conclusion of the development program was successful completion of all qualification tests with the Dev#4 engine. Although risk reduction efforts were conducted using all of the thrusters, this paper focuses on the acceptance and qualification of engine Dev#4

Table 2. MLE Development Prototypes

Engine Identifier	Timeframe	Total Firing, sec	Total Starts	Total Throughput, lbm	Catalyst Bed Configuration	Comments
Viking 015R	2002	234.5	5	209.0	Alternate	Rebuilt vintage Viking thruster. Thrust increased from 635 to 685 lbf (Pa=Mars)
Viking 020	2004	195.5	5	346.0	Baseline	Viking flight spare from 1976.
Dev#1	2004	334.9	8	644.0	Baseline	First throttled testing of MLE. Demonstrated range of 83-730 lbf (Pa=Mars)
Dev#2	2005	418.6	8	405.3	Baseline	All tests with cool fuel (39-51°F). Increased throttleable range down to 4.0 lbf.
Dev#2R	2006	715.9	10	755.0	Alternate	Non-flight bed design, tested to demonstrate capability as contingency plan in case Viking bed probs from Dev#2 tests persisted.
Dev#3R	2006	806.3	12	995.7	Baseline	High thrust demo, 3.76 lbm/sec (871 lbf @ Mars). Hot restart demo. Cold fuel demo (39°).
Dev#4	2006	349.2	7	612.9	Baseline	Completed qual testing (vibe, shock, environments) Demo start with ammonia poisoned bed. Hot GHe saturated fuel test.

Acceptance and Qualification Testing

Three acceptance tests and two qualification tests were conducted for a total of five starts. The acceptance test process conducted on Dev#4 was unchanged from prior MLE development engine test series and is largely based on the Viking process used in 1973. The qualification testing was based on the engine life cycle for the MSL mission.

Acceptance Testing

Figure 2 below depicts the acceptance test plan executed on Dev#4. Note that all testing is completed without the heat shield installed. The heat shield is installed subsequently –its primary purpose is to limit catalyst bed heater power required during the pre-heat of the bed.

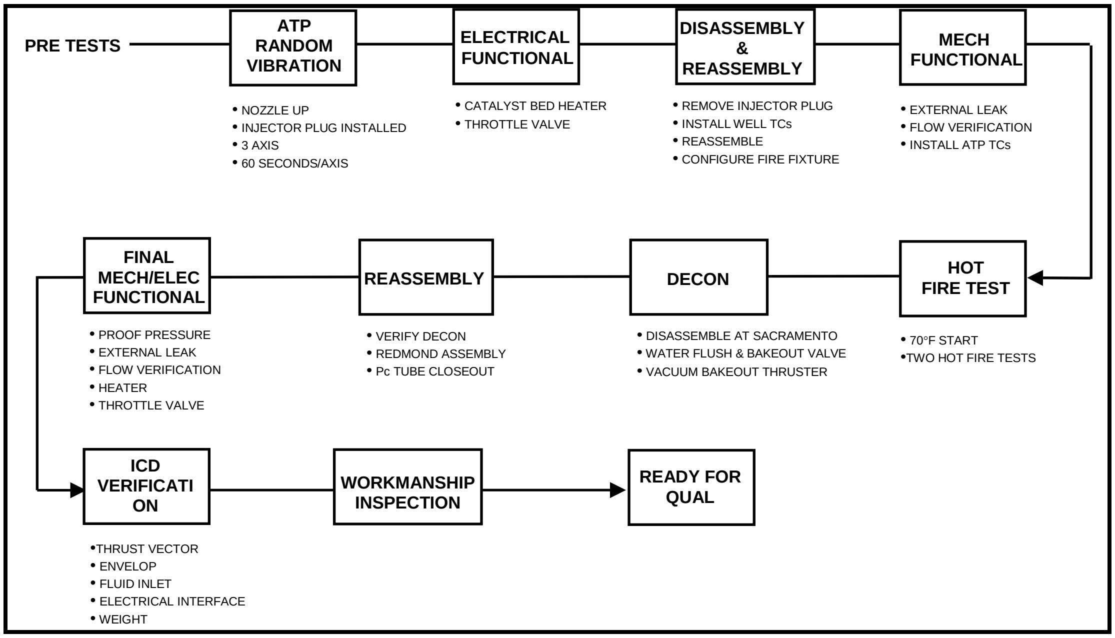

Figure 2. Acceptance Test Plan

As part of the pre testing, a series of x-ray computed tomography (CT) scans of the catalyst bed were conducted to assure the injector was fully packed with catalyst during assembly and that no voids were present. Catalyst bed voids can cause unpredictable engine performance. No voids were found on the engine CT scan.

Acceptance level random vibration testing was conducted next. The vibration fixture orients the engine with the MLE thrust axis 25 degrees off vertical in a nozzle up orientation as dictated by JPL to reflect the canted angle of the engine as installed on the spacecraft for launch. The x-axis is perpendicular to the thrust axis inline with the throttle valve, the y-axis is parallel to the thrust axis, and the z-axis is perpendicular to the thrust axis and perpendicular to the valve EMA axis. The test successfully completed 60 seconds duration per axis. The first resonant frequency of the engine for the X, Y, and Z test axes was 220, 265, and 215 Hz, respectively. After the vibe test the MLE was disassembled and the injector plug removed with less than 2.2×10^{-5} lbm (0.01 grams) of catalyst collected, a fraction of a percent of the initial catalyst mass. The engine was visually inspected for damage and none was found. Electrical functional tests were conducted on the catalyst bed heater and the catalyst bed heater passed circuit resistance and insulation resistance testing. The engine was reassembled and externally leak tested. The final functional test performed was a gas flow test to verify no foreign objects or debris were located within the assembly which would obstruct propellant flow. The unit passed all tests before proceeding to hot fire testing.

Hot fire testing was in a high altitude test facility capable of firing an engine producing 15,000 lbf (66.7 kN). The test objective was to determine the health of the engine and to quantify steady state and dynamic engine performance. Engine health is established via thrust step response and quasi-steady state operation at the end of each step change. Dynamic performance is evaluated during four sinusoid cyclic sequences. The ATP firing objective is to demonstrate compliance with JPL requirements for throttle range, minimum specific impulse, thrust response, stability, linearity, and engine backpressure.

The engine was instrumented to acquire surface and gas temperature data along with system and chamber pressure data. Thrust data was not taken for the ATP firings; however, the chamber pressure and nozzle throat area along with an appropriate thrust coefficient was used to calculate thrust. Martian ambient pressure of 0.24 psia (1655 Pa) was used when calculating thrust. The engine and propellant were temperature conditioned to a requirement of 77-122°F (25-50°C) prior to the start of a hot fire test. For testing, the engine was oriented with the nozzle firing down.

The ATP duty cycle consists of fourteen (14) steps of 2-3 seconds duration and four (4) sinusoids to demonstrate compliance with JPL specification requirements. Each ATP firing is 39.2 seconds duration. The minimum duration of each thrust step is two seconds to allow equilibration of the turbine flowmeters. The test duty cycle and thrust response is shown in Figure 3.

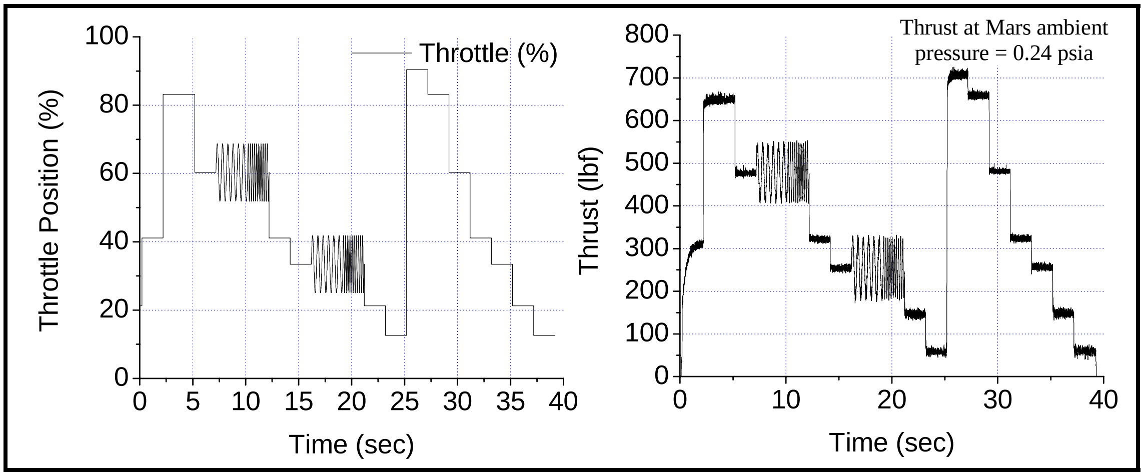

Figure 3. Acceptance Test Duty Cycle and Thrust Response (Test ATP-1)

The MLE test plan was outlined to demonstrate a throttled range of 90 – 674 lbf (400 – 3000 N) at an ambient pressure of 0.24 psia (1655Pa), in Martian atmospheric. The ATP-1 firing of demonstrated a throttled range of 79 to 731 lbf (350-3250 N).

The thrust response requirements for the four sinusoid cyclic operation sequences fired during ATP-1 and ATP-2 were acceptable per JPL requirement. The stability requirement is the chamber pressure oscillation amplitudes do not exceed ± 15 psi (10.3 kPa) over any 500 ms interval for a fixed throttle valve setting. The JPL thrust linearity requirement is that the chamber pressure versus flow-rate relationship is 77.2 psi ± 2.3 psi per 1 lbm/sec (1173 kPa ± 35 kPa per 1 kg/sec). The maximum allowable MLE backpressure requirement is 460 psia (3.17 MPa) flowing 3.1 lbm/sec (1.4 kg/s) hydrazine. This requirement assures throttle valve cavitation margin while operating at the current mission maximum allowable flowrate/thrust level. Compliance with the chamber pressure stability, thrust linearity, and backpressure operability requirements were achieved.

Post test electrical resistance checks were conducted for the engine components and throttle valve. Electrical bond of the unit was checked and successfully passed. Pre and post test measurements of the thrust vector were made to quantify any change in thrust vector from hot fire testing. The post ATP hot fire thrust vector was 0.0442 degrees out of perpendicularity to the REA mounting plane and the radial offset of the thrust vector to the center of the customer mounting bolt circle is 0.0013 inches.

<u>Qualification Testing</u>

The series of qualification tests were dictated by JPL per the engine design specification and generally reflect processing the flight units will receive after delivery from Aerojet. The MLE qualification test plan is detailed below as Figure 4.

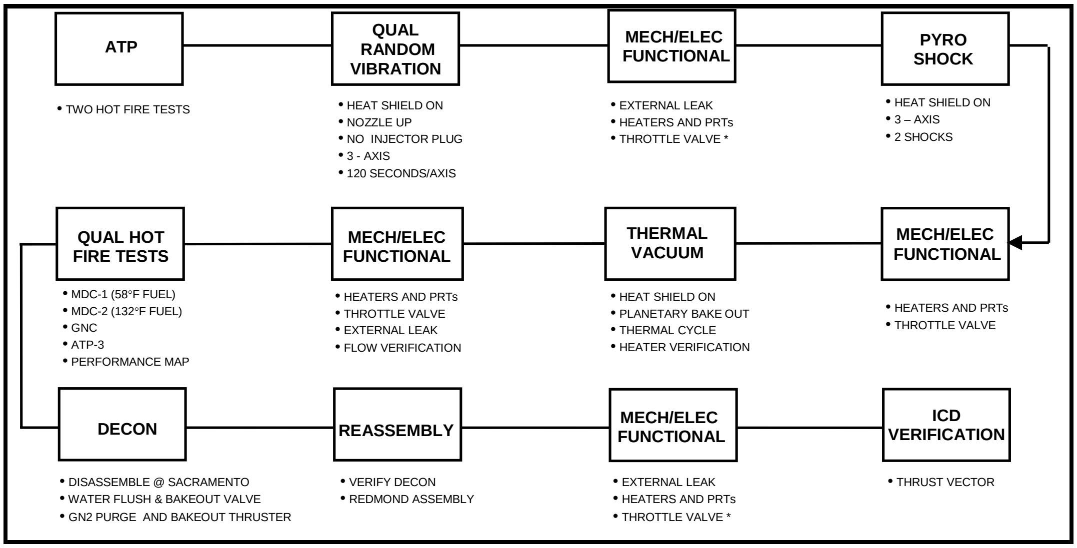

Figure 4. Qualification Test Plan

Qualification level random vibration testing was conducted using the same facility, fixture, and instrumentation used during acceptance testing. The injector plug used to collect catalyst fines for acceptance level testing was not installed for qualification testing. The throttle valve was positioned to the 100% (full open) position during testing to prohibit contact between the valve pintle and venturi throat. The test was successfully completed with 120 seconds duration per axis. The first resonant frequency of the chamber for the X, Y, and Z test axes was 215, 255, and 215 Hz, respectively. The engine passed the post test external leak inspection. Electrical bonding, resistance, and insulation resistance tests were performed on the throttle valve, valve heater, and catalyst bed heater. All components passed test requirements.

Pyrotechnic shock testing was conducting by National Technical System (NTS) using ordinance. The purpose of this test was to demonstrate survivability of the engine and all components through this environment. Following pyrotechnic shock testing, the engine electrical components were once again checked for electrical bonding, component circuit resistance, and insulation resistance. All components passed test requirements.

The purpose of the planetary protection bakeout is to sterilize all hardware before it is delivered to Mars to perform its scientific experiments. The MLE design must be capable of surviving the bakeout. The bakeout requirement is 50 hours minimum in vacuum at a temperature of 248°F (120°C). The bakeout test lasted 73 hours with a peak temperature of 285°F (140°C). The engine temperature exceeded 248°F (120°C) for 54 hours. The engine was then prepared for thermal vacuum testing.

The objectives of the thermal vacuum testing was to expose the engine to extreme hot and cold temperatures expected during the spacecraft integration, assembly test and launch operations, and the mission, -131 and 158°F (-55 and +70°C), and to verify the heater's ability to preheat the catalyst bed and valve wetted components to a required temperature. The requirement is the ability to preheat the catalyst bed and wetted components to 77°F (25°C) during a 24 hour period starting from -40°F (-40°C). During the heater performance portion of the test, transient warmup data is recorded for analytical model validation. The analytically predicted power required to achieve these start temperatures was 4.0 and 5.5 Watts for the catalyst bed and valve wetted surfaces, respectively. After completing the thermal vacuum testing it was concluded that the minimum heater power to achieve the 77°F (25°C) condition in 24 hours from a -40°F (-40°C) initial condition was 5.1 and 8.5 Watts for the catalyst bed and valve heaters, respectively. The increased power requirements were considered acceptable by JPL. After the thermal vacuum test, the engine was inspected for evidence of out-gassing contamination. All surfaces were found to be acceptable and appeared to be unchanged. The valve heater and bonded temperature sensors were also inspected to reveal no de-bonding had occurred. Electrical functional tests conducted verified acceptable electrical bonding, insulation resistance, and circuit resistance. During the final mechanical functional test, the engine was installed onto the firing fixture and cold flow tested to ensure no obstructions existed within the assembly. The engine passed the flow test and preparations were made for qualification hot fire tests.

The test cell was modified for qualification hot fire tests with the installation of a heritage thrust stand and reorientation of the engine to fire horizontally. Additionally, the propellant feed system was modified to enable Helium saturation, GHe saturation level check, and propellant temperature conditioning. There were five hot fire tests planned to be conducted, each with a specific purpose. The first test was Mission Duty Cycle #1 (MDC-1) and was derived from a JPL model prediction of MLE usage in a case where engine firing occurred at the earliest point possible after spacecraft backshell deployment. MDC-1 includes all aspects of the mission including attitude correction, deceleration, hover, and flyaway. MDC-1 was to be conducted using the minimum allowable propellant temperature 15°C (59°F). The valve throttle position and thrust are plotted against time in Figure 5. The data shows thrust response tracking very well with valve position and demonstrates the MLE’s ability to infinitely throttle to precise thrust levels. The second hot fire test was Mission Duty Cycle #2 (MDC-2) and was also derived from JPL mission model prediction of MLE usage if the engine startup occurred but the spacecraft deployment from the parachute was delayed until the correct altitude/velocity was achieved. During this period the engines are firing at the near-shutdown condition (1% throttle) for up to 80 seconds. The attitude correction and deceleration burns are followed by another near-shutdown operation for 90 seconds. The valve throttle position and thrust versus time potted as Figure 6. The MDC-2 plan called for a 159°F (70°C) maximum allowable propellant temperature with a Helium saturation point of at least 650 psia (4.48 MPa). This firing was the only one with Helium saturated propellant.

The third hot fire test was called the Guidance Navigational Control duty cycle (GNC). The purpose of this firing is to provide hot fire performance that quantifies the dynamic response of the engine across the allowable spectrum of cyclic operational frequency, amplitude, and nominal throttle setting. The fourth hot fire test was a repeat of the acceptance duty cycle fired and is referred to as ATP-3. The purpose of this firing was to obtain hot fire data for determination of life effects throughout all tests. The fifth hot fire test was a performance mapping test to determine delivered thrust, specific impulse, and thrust coefficient of the MLE (CF Map).

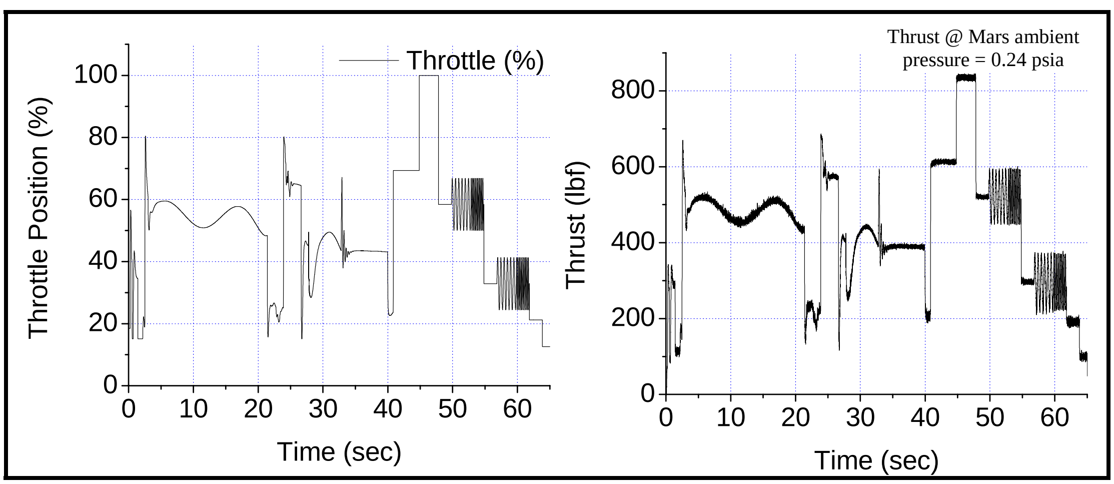

Figure 5. Qualification Test Duty Cycle and Thrust Response (MDC 1)

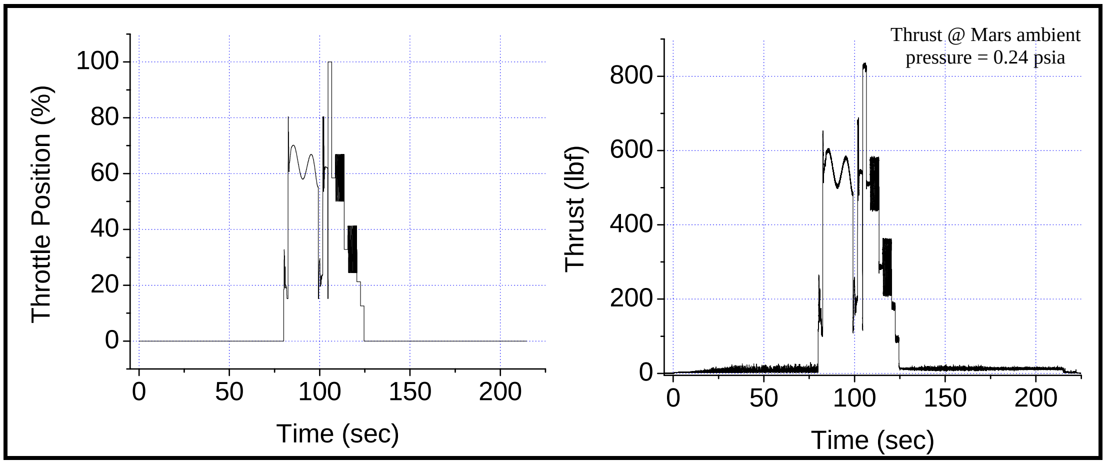

Figure 6. Qualification Test Duty Cycle and Thrust Response (MDC 2)

After testing was completed, the engine was removed, processed for decontamination, and then reassembled for post qualification hot fire functional tests. The engine was proof pressure and leak checked. Electrical functional tests conducted verified acceptable electrical bonding, insulation resistance, and circuit resistance of the electrical components. The thrust vector and thrust chamber outside diameter at a specified mid-wall elevation were measured. No significant change was found from post acceptance measurements. Additionally, no significant change was observed in the thrust chamber structure due to all acceptance and qualification test environments. A final CT inspection was conducted to reveal minor voids in both the inner and outer catalyst beds.

The data from all five qualification hot fire test sequences has been used to calculate the specific impulse against thrust in Figure 7 below. Also plotted with the test data is the MSL specific impulse requirement. All specific impulse data exceeds the minimum specific impulse requirement.

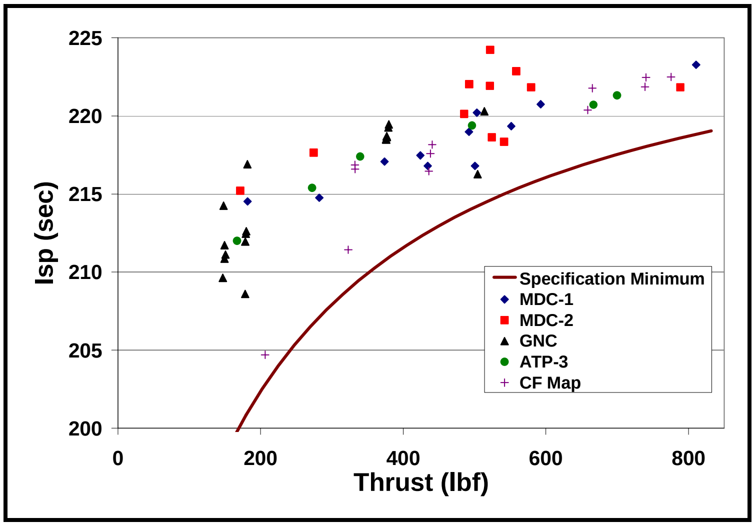

Figure 7. Qualification Test Specific Impulse vs. Thrust

Figure 8 below outlines the radial chamber surface temperature as a function of time for last test in the qualification series. Twelve thermocouples were placed circumferentially around chamber outside diameter at the axial location shown in Figure 8. Six were located at the hot zones (Tc21, Tc10, Tc14 …) in line with the injector elements and the other six (Tc19, Tc13, Tc15, …) were in the warm zones in between. A non-uniform temperature distribution may indicate issues developing within the engine. The data shows a relatively even distribution with a maximum difference in temperature of approximately 250°F (121°C).

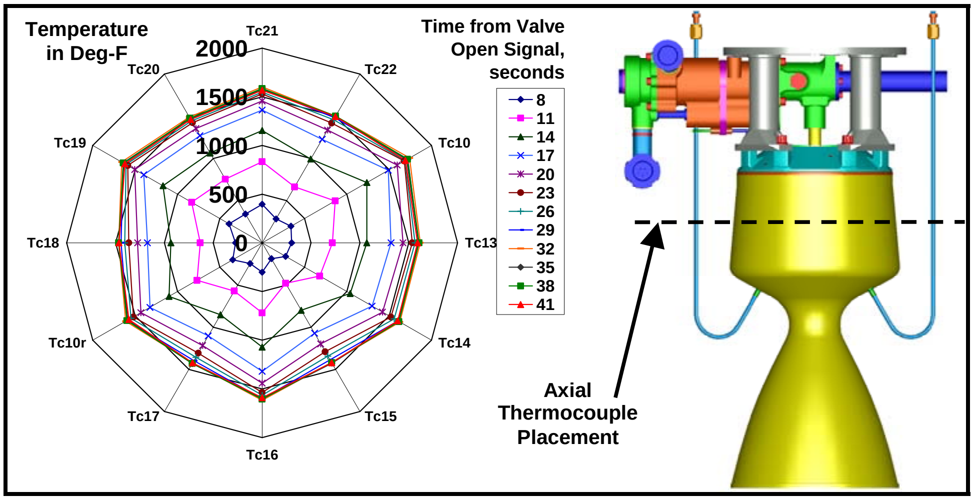

Figure 8. Catalyst Bed Temperature Distribution for Life Assessment, Dev#4

Another method to assess the health of an engine throughout life is to track catalyst bed resistance factor (Kbed) as a function of total propellant consumed. The catalyst bed flow resistance increases with accumulated firing and propellant consumption and is very predictable for the MLE even when fired under adverse conditions. The data presented in Figure 9 below shows the Dev#4 engine catalyst bed resistance factor throughout the testing performed. A large change is observed during Test #6. This change was caused by an internal pressure spike upon engine startup with an ammonia poisoned catalyst bed. Further investigation of the test setup revealed the cause of this increased resistance factor was catalyst bed contamination due to a facility propellant valve leaking overnight prior to Test #6. The Dev#4 engine survived this extremely hazardous operation and met all engine performance requirements for thrust, specific impulse, and dynamic response.

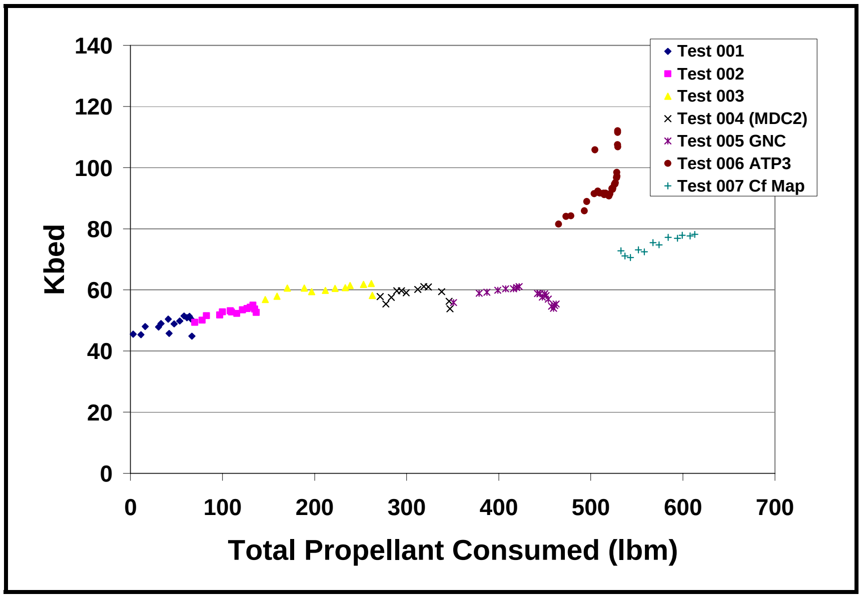

Figure 9. Catalyst Bed Pressure Drop for Life Assessment, Dev#4

Current Status[8]

A spacecraft plume impingement problem found late in the program requires a nozzle extension be added to the MLE increasing the expansion ratio and narrowing the divergence angle of the plume. The new design is complete, parts have been fabricated for the MSL flight engines, and a development engine (see Figure 10) has successfully completed vibration testing. Since the engine already meets all of the specific impulse requirements, detailed analysis of expected thrust coefficients will not be completed—thrust and Isp will be measured on the thrust stand. An additional MLE development test program funded by JPL will demonstrate 10 minutes duration steady state firing capability for potential use on future missions. The flight engines are to be delivered early in 2008.

An important off-shoot of this test has been a development program using MLE hardware for Ares. Under contract to NASA MSFC and using hardware loaned by JPL, Aerojet removed the throttle valve, installed a pulsing valve, and successfully demonstrated pulse mode operation of the MR-80-series engine. Results of this testing were compiled by MSFC[7]. The engine in the sea level test cell is shown in Figure 10.

Figure 10—MLE Engine with Nozzle Extension

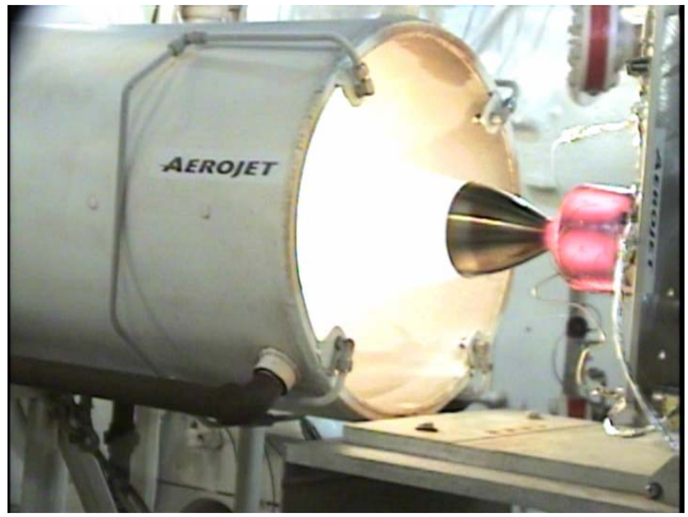

Figure 11—MLE Engine Demonstrates Pulse Mode Operation at Sea Level

Summary and Conclusions

A throttleable hydrazine monopropellant rocket engine is being developed by Aerojet as part of an ongoing technology program for the NASA Jet Propulsion Laboratory. The engine, based on the 1976 Viking Lander engine, will be used for terminal descent of an exploration vehicle to the Martian surface in 2009. The acceptance test process was unchanged from prior MLE development engine test series and was largely based on the Viking process used in 1973. The qualification testing was based on the engine life cycle for the MSL mission and included five hot fire tests. These tests demonstrated two worst case MSL mission duty cycles, engine dynamic response to a cyclical GNC duty cycle, life, thrust, specific impulse, and thrust coefficient. The acceptance and qualification testing demonstrated a throttleable range of 7-810 lbf (31-3603 N) thrust with a corresponding specific impulse range of 204-223 seconds. All thruster characteristic measured exceeded MSL's requirements.

Acknowledgments

Aerojet wishes to acknowledge NASA's Jet Propulsion Laboratory for their funding of and contribution to both the earlier Viking Program and the Mars Science Laboratory. Many individuals contributed to this program and Aerojet wishes to thank them all—Ron Carlson of JPL in particular.

As noted in Mr. Morrisey's paper[1] "The Viking Lander mission was a success because of the 10,000 people who earned the right to have their signatures on the microdots now on the surface of the planet Mars. To all of them, our gratitude."

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