NASA Propulsion Concept Studies and Risk Reduction Activities for Resource Prospector Lander

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The Resource Prospector mission is to investigate the Moon's polar regions in search of volatiles. The government-version lander concept for the mission is composed of a braking stage and a liquid-propulsion lander stage. A propulsion trade study concluded with a solid rocket motor for the braking stage while using the 4th-stage Peacekeeper (PK) propulsion components for the lander stage. The mechanical design of the liquid propulsion system was conducted in concert with the lander structure design. A propulsion cold-flow test article was fabricated and integrated into a lander development structure, and a series of cold flow tests were conducted to characterize the fluid transient behavior and to collect data for validating analytical models. In parallel, RS-34 PK thrusters to be used on the lander stage were hot-fire tested in vacuum conditions as part of risk reduction activities.

Nomenclature

ACS	=	Attitude control system
AF	=	Air Force
COPV	=	Composite overwrapped pressure vessel
COTS	=	Commercial-off-the-shelf
DACS	=	Divert attitude control system
ΔV	=	Velocity change
GNC	=	Guidance, navigation, and control
Hz	=	Hertz
ISE-100	=	Abbreviation of 100-lbf descent thruster
ISE-5	=	Abbreviation of 5-lbf ACS thruster
LCH4	=	Liquid methane
LOX	=	Liquid oxygenMDA = Missile Defense Agency
MMH	=	Monomethyhydrazine
MSFC	=	NASA Marshall Space Flight Center
MON-25	=	Nitrogen tetroxide with 25% of nitrogen oxide in mass.
NASA	=	National Aeronautics and Space Administration
NTO	=	Nitrogen tetroxide
OME	=	Orbital maneuvering engine of Space Shuttle
PK	=	Peacekeeper
RP	=	Resource Prospector
RS-34	=	Abbreviation of ACS PK thruster
SRM	=	Solid rocket motor
TCM	=	Trajectory correction maneuver
WSTF	=	NASA White Sands Test Facility

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I. Introduction

NASA's exploration roadmap[1] is focused on developing technologies and performing precursor missions to advance the state of the art for eventual human missions to Mars. One of the key components of this roadmap is various robotic missions to Near-Earth Objects, the Moon, and Mars to fill in some of the strategic knowledge gaps. The Resource Prospector (RP) project[2] is one of these robotic precursor activities in the roadmap. RP is a multi-center and multi-institution project to investigate the polar regions of the Moon in search of volatiles. The mission is rated Class D and is approximately 10 days in duration, assuming a five day direct Earth to Moon transfer.

Because of the mission cost constraints, a trade study of the propulsion concepts was conducted with a focus on available low-cost hardware for reducing cost in development, while technical risk, system mass, and technology advancement requirements were also taken into consideration. The propulsion system for the lander is composed of a braking stage providing a high thrust to match the lander's velocity with the lunar surface and a lander stage performing the final lunar descent. For the braking stage, liquid oxygen (LOX) and liquid methane (LCH4) propulsion systems, derived from the Morpheus experimental lander[3], and storable bi-propellant systems, including the 4th stage Peacekeeper[4] (PK) propulsion components and Space Shuttle orbital maneuvering engine (OME), and a solid motor were considered for the study. For the lander stage, the trade study included miniaturized Divert Attitude Control System[5] (DACS) thrusters (Missile Defense Agency (MDA) heritage), their enhanced thruster versions, ISE-100 and ISE-5[6], and commercial-off-the-shelf (COTS) hardware.

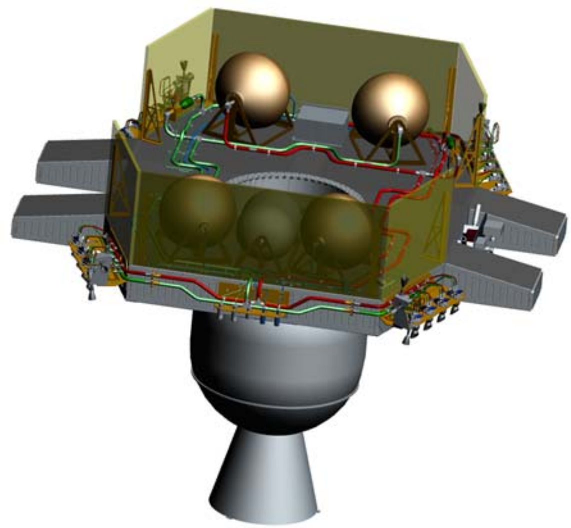

Figure 1. Government version of RP lander. SRM for braking stage and a bi-prop system for lander stage.

The configuration of using the solid motor and the PK components while meeting the requirements was selected due to its lowest cost. The reference concept of the lander is shown in Figure 1. In the current reference configuration, the solid stage is the primary provider of velocity change (ΔV). It will generate 15,000-lbf of thrust with a single burn of ~ 80 seconds. The lander stage is a bi-propellant, pressure-regulated, pulsing liquid propulsion system which will perform all other propulsive maneuvers.

Moving forward in maturing the liquid propulsion concept, a series of risk reduction activities including the system design, the propulsion system cold flow test, and thruster hot-fire tests[7] were also conducted. For the cold flow test, a simulated propulsion system based on the drawings of the early flight design was built to evaluate the feasibility of using available PK and COTS propulsion components in the lander application. This buildup also served as a mockup for demonstrating the integration of a propulsion system to a flight-like lander structure. The propulsion cold flow test provided data to characterize the steady state flow and pressure conditions as well as transient behavior. The testing included a focused parametric study (variations in operating conditions, simulated variations in metal diaphragm pressure drops, etc.) on steady state operation, slump and water-hammer effects due to the combination of opening and closing of the thruster valves, and system priming (initial system activation).

For the descent thrusters, PK RS-34 70-lbf thrusters[4] used on the 4th stage PK propulsion were selected for the lander stage. However, the usage of such thrusters for the RP mission is slightly outside of the qualified regime of the RS-34. Moreover, the aging of the hardware was also a concern for the project. To reduce the risk of using the thrusters for the mission, a hot-fire test program on the thruster was conducted in vacuum conditions at NASA White Sand Test Facility (WSTF) in New Mexico.

This paper will provide an overview of the propulsion concept studies and following-on risk reduction activities. The selected propulsion concept will also be discussed and results of the risk reduction efforts will be highlighted on the paper.

II. Propulsion Trade study

The primary focus of the trade study was to find ways of reducing cost in propulsion system development, technical risk, system mass, and technology advancement requirements were also aspects of possible designs which were taken into account. The reference propulsion configuration has two stages: a separable solid rocket motor (SRM) for the braking function and a bi-propellant, pressure-regulated, pulsing liquid stage to perform all other

GNC functions. The SRM stage, baselined on ATK STAR-48BV[8], delivers high thrust with a short-duration burn before its separation for the terminal descent. The liquid stage is responsible for performing necessary trajectory correction maneuvers (TCM), controlling the attitude of the spacecraft, and performing terminal descent to the lunar surface. The liquid stage is comprised of the tanks, lines and components, twelve five-pound ACS engines, and twelve one hundred-pound descent engines. Although the liquid stage is periodically operated throughout the flight mission, a significant amount of propellants is consumed during the terminal descent. To ensure the study figures of merit were fully captured, several parts of the trade studies investigated departures from the baseline configuration.

For the braking stage, LOX and LCH4 propulsion system, derived from Morpheus experimental lander, and storable bi-propellant systems, including the 4th stage Peacekeeper (PK) propulsion components and Space Shuttle OME are considered.. It should be noted that the PK propulsion components and Space Shuttle OME are government-owned hardware which can be used without procurement cost, but would have costs associated with modifications to the government owned hardware if required.

For the lander stage, the study includes miniaturized DACS thrusters (MDA heritage), their enhanced thruster versions, ISE-100 and ISE-5, and COTS hardware. The rationale for selecting the propulsion configurations and components for the study is summarized in Table 1 below.

Table 1. Rationale for selecting propulsion configurations for trade study

Concepts	Rationale
Braking Stage	
SRM	Already qualified and operational in space, system simplicity, high propellant mass ratio.
LOX/LCH4	Non-toxic, high performance, and provide an opportunity to demonstrate technology for future exploration.
Hypergolic bi-prop	Flight qualified, low cost because of using operational government-own propulsion components, such as 4th stage Peacekeeper and Space Shuttle OME
Lander Stage	
DACS (ISEs and existing engines)	Low weight, operated with cold propellants for savings in heater power, and provides an opportunity to demonstrate technology for future science missions.
COTS	Already qualified and operational in space
Available Peacekeeper hardware	Flight qualified, low cost because of using operational, government-own propulsion components of 4th stage Peacekeeper missile

Several criteria were established for the down selection. The first criterion was that the lander mass should be within the capability of the launch vehicle selected, which at the time of this study was the SpaceX Falcon 9 v1.1. The second criterion was the cost of developing the propulsion system should be within the projected allocation. The direct costs were considered, but also in the distribution of manpower across agencies and contractor organizations. The final criterion was the schedule and technical risk level for accomplishing the mission. Based on the these metrics, three potential propulsion configurations were down-selected from a total of eleven configurations. The summary of pros and cons for the original reference and options are listed in Table 2 below. Risks for the options are also qualitatively assessed and listed in Table 3 below.

The option 1 had the lowest cost while the lander mass was within the launch capability; hence, this option was selected as a new reference for the propulsion system. It should be noted that the PK thruster is capable of generating approximately 70-lbf; therefore the lander will require sixteen PK thrusters for terminal descent instead of twelve 100-lbf thrusters as stated on the original reference. Because of this selection, the risks identified in Table 3 have been mitigated. The PK thrusters were tested in vacuum conditions. The test results indicate that the thrusters are able to meet the mission requirements.

Table 2. Summary of pros and cons among propulsion configuration options

Option	Config.	Cost	Lander Mass	Pros	Cons
Original Reference	ISE/ SRM	High	Low	• Lightest weight • New technology demo. for future mission • Reduced heater requirements	• Highest cost • High risks (technical and schedule)
Option 1	PK/ SRM	Low	Medium	• Lowest cost, hardware available without cost.	• Moderate weight increase • Lowest performance • No technology demo.
Option 2	Existing DACS/ SRM	Medium	Medium	• New technology demo. future • Potential reduced heater requirements • New technology demo. future	• Moderate cost • Moderate weight increase
Option 3	Mono Prop hydrazine/ SRM	Medium	High	• Low/moderate cost • Simple , reliable system w/ extensive flight data	• Heaviest • No technology demo.

Table 3. Qualitative assessment of risks on propulsion options

Option	Configuration	Risk
Original Reference	ISE/ SRM	• Thrusters are still in development phase. • 1st use of MON-25/MMH in space and at wide temperature range
Option 1	PK/ SRM	• Aging hardware, especially valve soft-good. • Thruster nozzle made of Beryllium (toxic) • Minimum impulse bit repeatability
Option 2	Existing DACS/ SRM	• Hardware modification (new Teflon valve seal) • 1st use of MON-25/MMH in space. • Relatively high pressure system
Option 3	Mono.Prop hydrazine/ SRM	• Potential interference w/ optical landing devices during the descent due to continuous thruster operation (throttling instead of pulsing) • Plume effects to SRM • Thrusters are not in production. • High pressure operation & large size of feed lines & large tanks

III. Propulsion System Design

The flow schematic and mechanical design of the liquid propulsion for this lander are presented in Figures 2 and 3, respectively. Hypergolic propellants monomethylhydrazine (MMH) and nitrogen tetroxide (NTO) will be supplied to sixteen 70-lbf class descent thrusters and twelve 5-lbf attitude control thrusters. These thrusters are grouped into four thruster modules that are located on corners of the lander. All thrusters will operate with pulsing modes for precision and soft landing. The descent thrusters will provide main thrust for trajectory correction maneuvers and terminal descent, while the ACS thrusters will perform pitch, roll, yaw, and nutation damping.

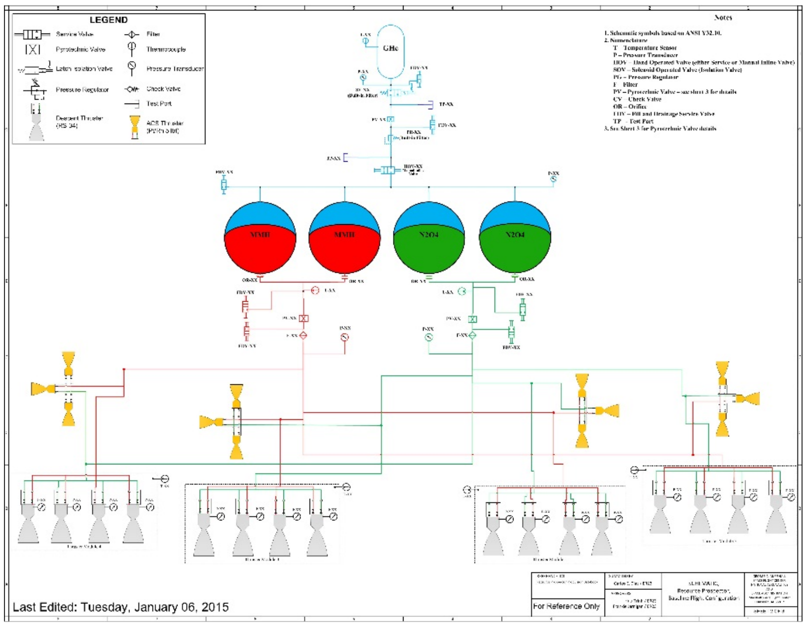

Figure 2. Flow schematic of liquid propulsion for lander stage

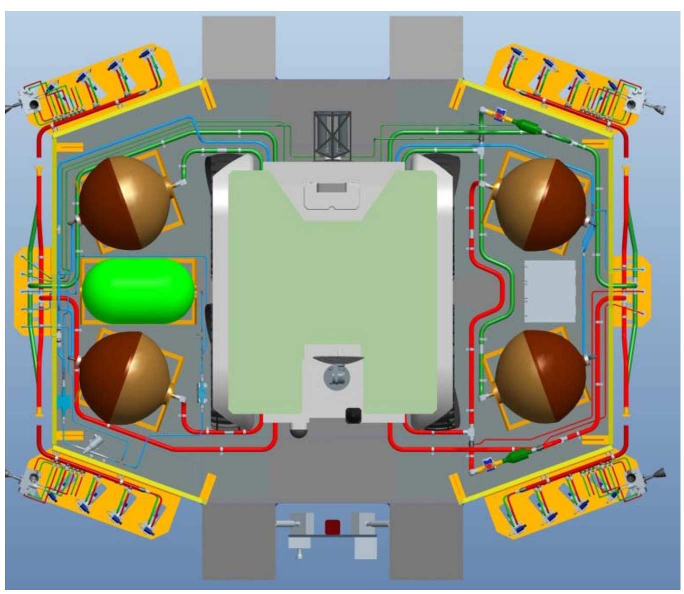

Figure 3. Mechanical design of liquid propulsion system

With respect to the propellant storage and distribution system, four metal diaphragm tanks, two connected-in-parallel tanks per propellant, will be used along with a high-pressure composite overwrapped pressure vessel (COPV) for the helium pressurant gas. The metal diaphragm tanks offer the advantage of active propellant control to eliminate technical issues, such as propellant slosh and gaseous pressurant entrainment to the engines, while gaining high propellant expulsion efficiency (~98%). Using government available PK propulsion components along with

some COTS hardware will significantly reduce the cost of the spacecraft while maintaining the technical and schedule risk at a minimal level.

IV. Propulsion Cold Flow Test

In parallel with the flight system design activities, a simulated propulsion system, shown in Figure 4, based on flight drawings was built for conducting a series of water flow tests to characterize the transient fluid flow of the propulsion system feed lines and to verify the critical operation modes, such as system priming, waterhammer, and crucial mission duty cycles. The primary objective of the cold flow testing was to simulate the RP propulsion system fluid flow operation through water flow testing and to obtain data for anchoring analytical models. Other test objectives included the following:

- Demonstrate the propulsion hardware assembly/integration (thruster modules, feed lines, prototype flight structure, etc.)
- Demonstrate key propulsion operation characteristics (conceptual usage profiles, propulsion priming process, etc.).
- Demonstrate available Peacekeeper components to be used for the RP.
- Develop a propulsion system mockup to gain knowledge on hardware as well as operational interfaces with other sub-systems.

Though the cold flow test article was based upon the flight system design, some deviations from the flight design were made based on cost and schedule constraints, as well as component availability, producibility, and test considerations. The analytical models developed for this system were used to predict the transient and steady state flow behaviors in the actual flight operations, and to date all analytical modeling efforts have been completed for this test activity.

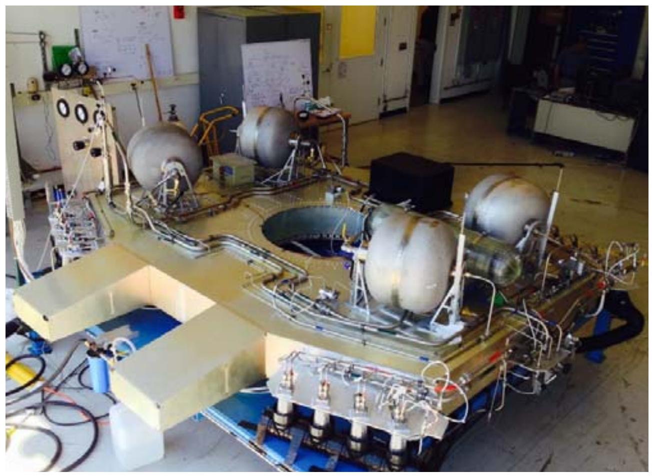

Figure 4. RP Cold Flow Test Article

Priming

The priming testing was divided into multiple phases due to hardware and design changes during the test activities. Initially, the team planned to simulate the pyro-valve actuation with a burst disk. After conducting multiple tests and getting inconsistent results from the burst disks, the team decided to implement a remotely-operated ball valve in place of the burst disks. Implementation of the ball valve provided marginally more consistent results and with much higher surge pressures, as can be seen from Figure 5. Those surges were in excess of 2500 psi, which exceeded many of the flight component’s rated pressures.

Due to these higher priming pressures seen during the testing, the system was redesigned to include a small bypass line around the pyro-valve such that the flow into the system could be metered, thereby reducing the overall priming surge. The results of that third phase of testing proved the ability of the bypass line to reduce the priming surge, as shown in Figure 6, well below 1000 psi, but results were still inconsistent. Due to schedule constraints, the priming test activities were concluded after the third phase of testing. It was concluded that even though results were still inconsistent, the implementation of the bypass line did provide the desired reduction in surge pressure to within the pressure ratings of the components.

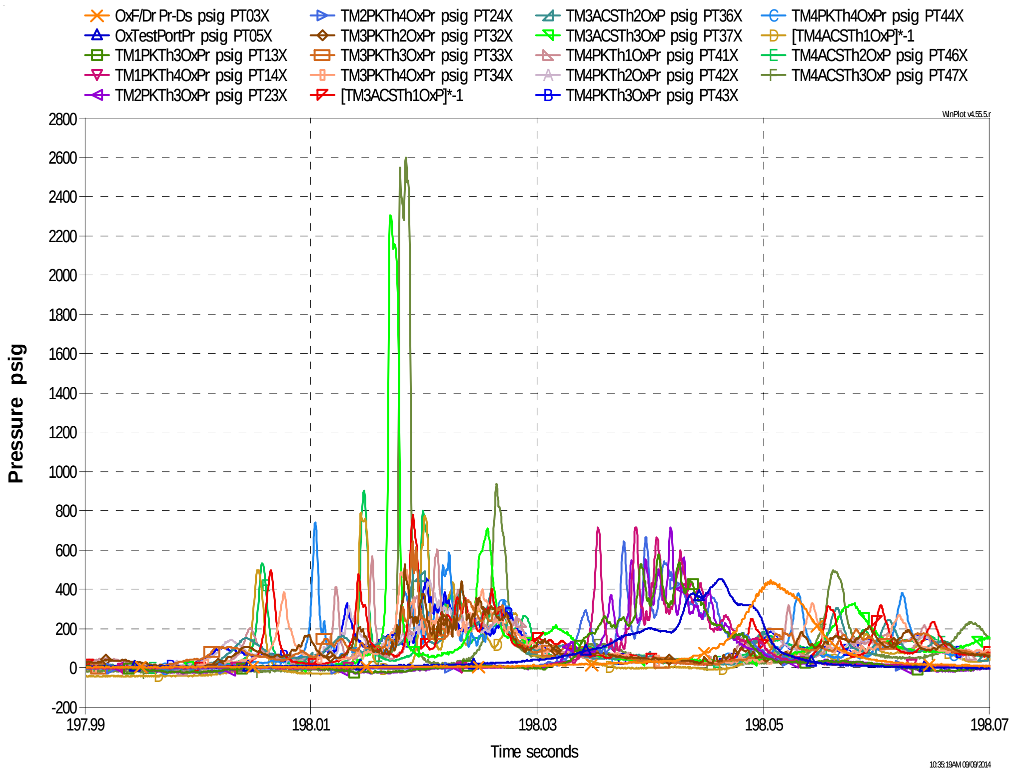

Figure 5. Priming surge with remotely-operated ball valve

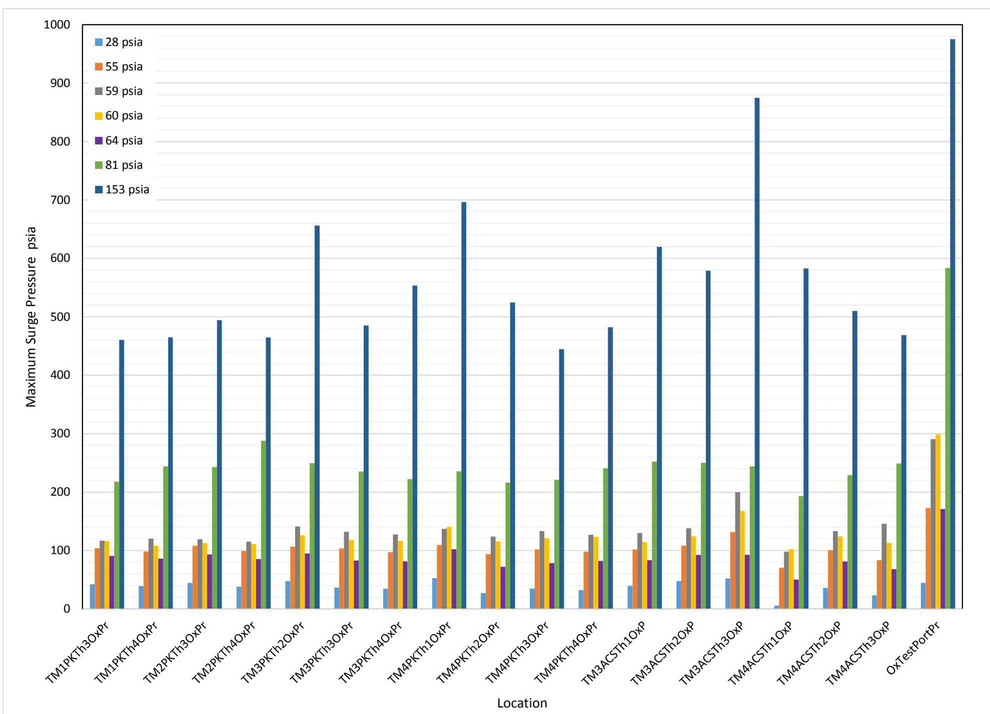

Figure 6. Priming surges with bypass line at different ullage pressures

Waterhammer

Testing was completed to evaluate maximum system surge pressures during various valve opening and closing scenarios. Scenarios tested include individual valve closure, multiple valves closing simultaneously, and valves

closing while others open at the same time. It was assumed that the thrusters would fire in sets of four (one per thruster module) to allow for stability during landing. Most of the waterhammer testing evaluated individual or multiple sets of thrusters opening and/or closing at the same time due to concerns that high surge pressures would occur with multiple sets of thrusters closing at the same time. Additionally, the system was anticipated to operate at a 25 or 50 Hz frequency, so both frequencies were tested with multiple thruster sets.

Results from this testing showed that for the majority of the valve opening/closing scenarios, the waterhammer surge pressures did not exceed component pressure ratings. However, for a couple of cases for both the 25 Hz and 50 Hz testing, it was shown that the interaction between as few as two thrusters closing was enough to drive pressures in the system above some components' pressure ratings. The results of these tests were provided as inputs to our components and software requirements for the flight system.

Regulator Slam Start & Ullage Sensitivity

The PK pressurization regulator requires a minimum gas volume downstream of the regulator for proper regulator operation. The testing evaluated the ullage overshoot from the PK regulator as the total ullage volume was reduced including beyond the initial qualification box. Minimizing the ullage volume allows for either smaller tanks or more propellant loaded in each propellant tank. The volume for each of the propellant tanks was measured by filling each tank to overflow then slowly draining the tank to determine the mass removed (correlated to volume) via the sight glass level. The values for each tank were combined to estimate a height versus volume. The propellant tanks were all assumed to be identical. The sight glass level was used to load each propellant tank during test operations, with an assumed error in the total volume measurements estimated to be less than 5%.

GHe was used for the pressurant of the system during regulator test operations. The pressurant COPV was loaded to 3300 to 3400 psig for all but one test. A 350 psig relief valve was placed in the low pressure system. The isolation valve was opened and the system allowed to pressurize. The testing started with the required total ullage volume and then decreased the volume with each subsequent test. Previous testing on the regulator showed a nominal regulated pressure of 293 psig. The ullage was decreased until the relief valve opened, shown as the top curve in Figure 7. The test results indicated that the required volume downstream of the regulator could be significantly reduced from what was initially assumed from the component specification.

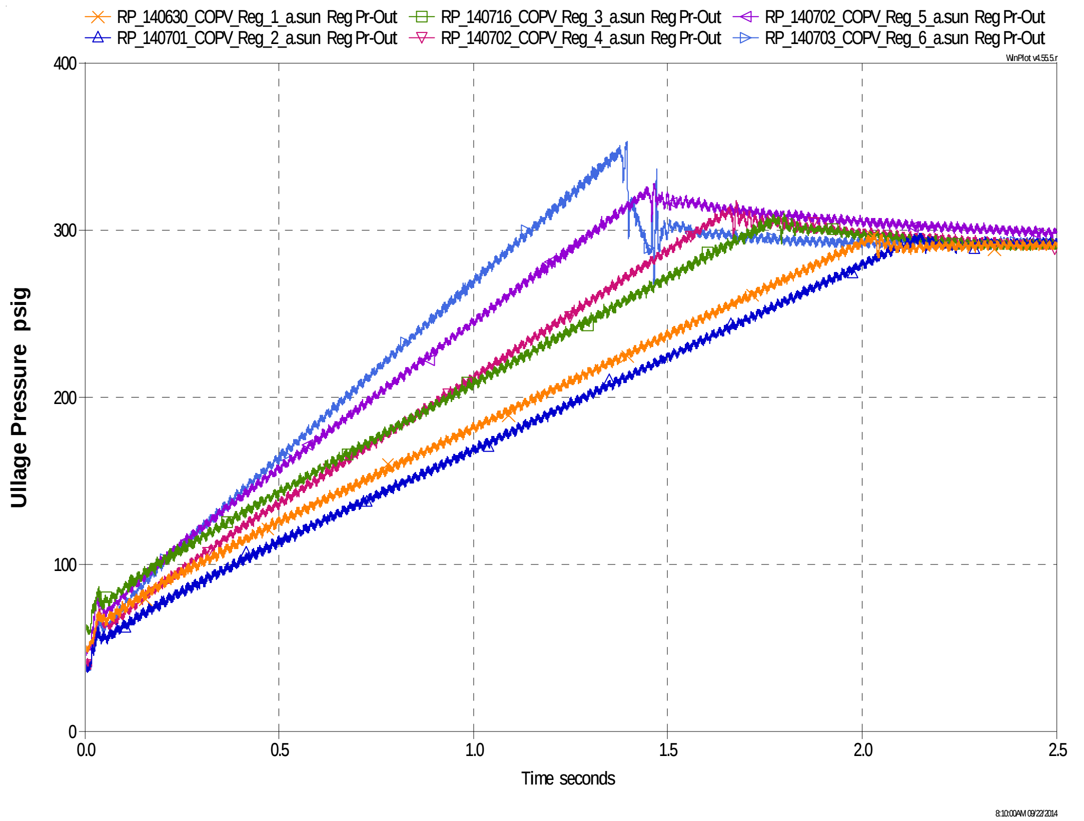

Figure 7. Ullage Overshoot Pressure with Varying Initial Ullage Volumes

V. Hot-fire test of RS-34 Thruster

The hot-fire test activity was conducted on two RS-34 thrusters which had been previously removed from the Air Force (AF) inventory of PK hardware. The thrusters were removed from PK 4th stages at WSTF as part of the demilitarization of the PK missile system and shipped to MSFC for instrumentation installation. Following installation of temperature instruments, the thrusters were shipped to WSTF for integration to the test stand prior to testing. Two thrusters were tested during the activity.

The RS-34 thruster is a flight qualified bi-propellant thruster developed by Rocketdyne that uses MMH and NTO. The chamber and nozzle are a single piece beryllium component machined from a hot-rolled beryllium ingot. The nozzle is mechanically attached to the stainless steel injector with a split-ring attachment ring which holds and seals the chamber to the injector. The injector has a port and tubing leading to an integrated strain gauge type pressure transducer. The injector assembly is mechanically attached to the inlet valve. Propellant flow is controlled by a single direct acting torque motor valve. Oxidizer and fuel control poppets are mechanically linked, and the valve is magnetically biased to maintain a normally closed position when unpowered. The thruster was mounted inside of the PK missile and conformed to the outer mold line with a scarfed nozzle. The scarf is approximately 45 degrees and causes the thrust vector to be approximately 5 degrees from the centerline of the thruster. Figure 8 shows the two thrusters that were instrumented and delivered to WSTF for hot-fire testing. Additional thermocouples are present on the bottom of the thrusters. Also, 6 uni-axial accelerometers, located at various points of interest on the thruster and test stand, are not shown in this view.

Figure 8. RS-34 Attitude Control Thruster

The thruster was designed for providing short pulses for attitude control and also for longer steady-state burns. The chamber cooling uses a fuel film cooling concept for internal regenerative (INTEREGEN) cooling that was developed for beryllium engines by Rocketdyne.

Early in the design of the RP mission, it was known the inlet conditions to the thruster may be different that the qualified regime. The pressure budget of the RP propulsion system is lower than the PK system, resulting in a lower inlet pressure at the thruster inlet. The lower inlet pressure, although close to the qualification regime, was of sufficient interest to require demonstration testing to show applicability to the RP mission. Because of this condition, the test program was designed in such a way to show that new test data could be combined with the qualification data to show the thruster could function at the lower inlet pressure with little risk to the mission. The objectives of the testing were 1) demonstrate the thruster functionality following shelf storage conditions exceeding the design life, 2) demonstrate thruster performance at RP conditions, and 3) generate thermal and performance data for system modelling.

Demonstration of functionality was an important goal of this test activity. The RS-34 thrusters have been in long term storage for much longer than their design life. The original program-required service life was 10 years from the date of manufacture, and the current age of the hardware is approaching 3x the design service life. The AF was aware that the 10 year limit might be exceeded before the PK program would be decommissioned, so they instituted a life extension program (aging and surveillance) to monitor the life of the components and subsystems. All of the PK components showed they would be acceptable for several decades before they were unusable. To show these thrusters would be acceptable for the RP mission, two were taken from storage and readied for hot-fire testing. No acceptance testing was conducted prior to hot-fire, including valve leakage assessments. Results showed all pulses and steady-state firings behaved as expected, showing the thruster could be reused with little to no check-outs on the RP mission.

Performance of the thruster at the RP conditions is also needed. Review of the qualification data, and vendor assurances that the thruster would operate at the lower inlet conditions were verified to be accurate during testing. Inlet pressures were tested to 240 psig from the nominal 298 psig. Data for thermal modeling was also generated which can be used for spacecraft integration purposes.

The thruster test setup is shown in Figure 9. Testing was conducted at the WSTF in Test Stand 401. All tests were performed at altitude simulated conditions with an approximate pressure 0.2 to 0.4 psia. The facility instrumentation includes engine inlet pressure, thrust measurement, valve current/voltage, and propellant flow rates. It should be noted that for pulsing tests less than 5 seconds, wavetube type flow meters were used, while the turbine flow meters were used for longer burn time test.

Figure 9. RS-34 Thruster Installed in TS401 at WSTF during Hot-Fire Testing

The RS-34 thrusters were tested a total of 88 times between 2 thrusters. Six tests were completed on the first thruster, the remaining were on the second thruster. The test matrix consisted of a series of pulsing and steady-state burns that were derived from the potential mission operational scenarios. Steady-state tests had a 2x margin on burn duration to show sufficient margin.

Post hot-fire data reduction showed the thruster performed exceedingly well with the lower inlet pressures of for the RP mission. Figure 10 shows a representative curve of the tested response over a wide range of inlet pressures, and mixture ratios and their impact on thrust. Included in the figure are qualification data for this engine. The offset between the qualification and test data is approximately 1.9%. Detailed data reduction was not able to uncover the cause of the variability, but build records for the engine tested and the qualification engine were not available Some variations could be attributed to build differences over the life of the PK program. There is some knowledge that performance enhancements were made during the program which may have been implemented in this set of engines that are not known to the test team. Additional causes of the offset that were ruled out were the thrust and flowrate measurement systems at WSTF. No indications were discovered in the data or the setup that indicate any issues which could have caused the offset. All testing showed specific impulse was maintained with 255-260 seconds regardless of the mixture ratios or the inlet conditions tested.

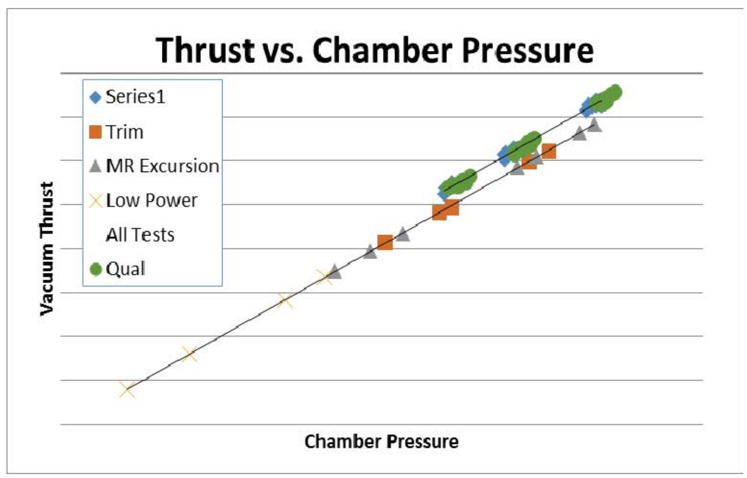

Figure 10. Thrust vs. Chamber Pressure

VI. Conclusion

The system trade study has led to the selection of a new reference design for the propulsion system that has the lowest cost and net lowest risk due to using the government-owned, flight qualified components and was deemed to be the best value by the Lander Project Manager. Although the selected propulsion is not optimized in mass and performance, it meets all mission requirements.

The mechanical design provided the layout of the propulsion sub-system and the physical interface between the propulsion components and the lander structure platform. The design was then used to build the system cold-flow test article for obtaining data to characterize the transient fluid flow operation and to anchor analytical models. The parametric test results indicated that the pressure surges from the system priming and waterhammer were within the

component operational limits. The regulator slam start tests and ullage sensitivity assessment showed that a reasonable amount of gas downstream of the regulator wass required.

The objectives of the RS-34 thruster test were met. Both thrusters performed as expected, and significant amounts of data were generated for performance and thermal modeling efforts for the RP mission. The lower inlet pressure requirement for RP has been shown to not impact the RS-34 operability. The specific impulse was unchanged from 255-260 seconds, and the thrust exhibited a linear relationship with inlet pressure to the thruster.

Overall, considerable progress has been made on the propulsion sub-system design for the government-version RP lander. The risk reduction efforts have successfully demonstrated the system design ready for the flight fabrication and development.

Acknowledgments

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