Resource Prospector Lander: Architecture and Trade Studies

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***Abstract*— NASA's Resource Prospector (RP) is a multi-center and multi-institution collaborative project to investigate the polar regions of the Moon in search of volatiles. The mission is rated Class D and is approximately 10 days. The RP vehicle comprises three elements: the Lander, the Rover, and the Payload. The Payload is housed on the Rover and the Rover is on top of the Lander. The focus of this paper is on the Lander element for the RP vehicle. The design of the Lander was requirements driven and focused on a low-cost approach. To arrive at the final configuration, several trade studies were conducted. Of those trade studies, there were six primary trade studies that were instrumental in determining the final design. This paper will discuss each of these trades in further detail and show how these trades led to the final architecture of the RP Lander.**

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1. INTRODUCTION

NASA's exploration roadmap 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 is one of these 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. The RP vehicle comprises three elements: the Lander, the Rover, and the Payload. The Payload is housed on the Rover and the Rover is on top of the Lander. The focus of this paper is on a Lander element for the RP vehicle.

The design of the Lander was requirements driven and focused on a low-cost approach. Several trade studies were completed on the following subsystems to arrive at the final architecture: the primary structure, communications, power, thermal, propulsion, and avionics. In each of the trade studies, the cost and the mass were primary figures of merit, along with other figures of merit specific to the trade.

In this paper, we will first review the current RP Lander architecture. Then, we will discuss the trade studies for each of the subsystems mentioned above. We will also discuss the figures of merit for the trades and how we arrived at the ultimate design decisions.

2. RP LANDER ARCHITECTURE

The RP lander is a joint development between MSFC, JSC, and APL. The requirements for the lander were influenced by the RP mission requirements. Derived requirements were crafted to focus the development on a lander which would be the lowest cost option. This was frequently done at the expense of mass and risk. The vehicle has two radiators that house all of the electronics, and one isolated optical bench for the Guidance, Navigation, and Control (GNC) instrumentation. A spherical communications antenna is housed on the solid rocket motor (SRM), but the majority of the communications system is located on the rover. An overview of the lander is shown in the following block diagram (Figure 1).

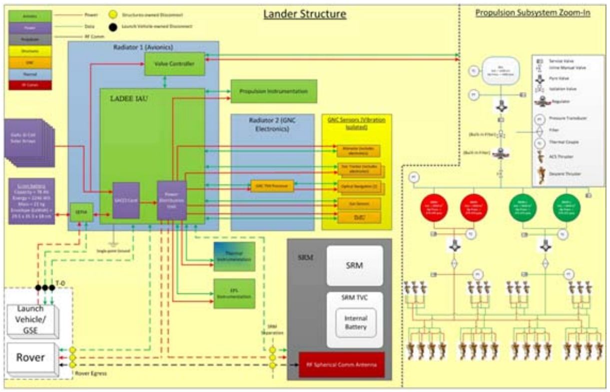

Figure 1: Lander system block diagram.

A series of philosophy decisions drove the overall architecture, and are reinforced in the lower level trades within the subsystems.

The first philosophy decision is with regard to risk. The agency can classify missions based on a variety of factors, leading to an agreed to approach for risk which feeds into the design. With the assumption of maintaining this as a low cost, low priority mission, a much riskier design posture is acceptable. The baseline approach to in-flight failures is to design for a single string approach. Unless there is a possible impact to personnel safety on the ground, a zero fault tolerance rules is being used. As the design matures, specific failure modes will be investigated on a case-by-case basis to address low cost areas where redundancy will significantly increase reliability.

The next approach is driven by the expected payload, a rover. One of the most challenging aspects of a rover deployment on the surface is the mechanism. Reach back to our low cost approach drives us to minimize the number of mechanisms, thereby reducing complexity and mass. An approach to eliminating some rover mechanisms is to allow direct egress from the lander to the surface. This approach is literally built right in to the primary structure in the form of fixed egress ramps.

To facilitate the rover egress on the fixed ramps, an effort was made to have the final landing height as close to the surface as possible. This drove a number of design decisions. First, the traditional Apollo style landing legs were replaced with crushable pads, directly transferring the landing energy into the primary structure. Second, the propulsion system required a design where the engines would not impact the surface. The perimeter placement of the engines allowed efficient heat radiation with mounting points allowing the large nozzles to have clearance above the surface.

To reduce requirements on landing control, an approach was taken to maintain a low center of mass. This drove the design to a pancake shape, only limited by standard launch vehicle fairing sizes. Tank placement was distributed across the pallet-type structure's deck. Taking into account the density differences of the fuel and oxidizer, and off center placement of tanks, a diagonal balancing was performed. The tanks in quadrants one and three are fuel, and quadrants two and four are oxidizer.

In the effort of lowest cost lander, a philosophy of selecting high "technology readiness level" components was required. Given the short development cycle (needed to keep manpower costs low), there was no time to spend in developing new technologies. The majority of the components have flown on previous NASA missions, and therefore required very limited new qualification data.

3. TRADE STUDIES

Primary Structure

For the RP mission, the primary structure of the lander has to carry all the components on the vehicle and survive all phases of the mission to safely deliver the rover and payload to the surface of the moon. With this in mind, and in keeping with the low-cost approach, there were five potential options for the primary structure. These were an isogrid/grid stiffened design, flanged grid stiffened, open grid stiffened, composite (metal-core-metal), and a baseline of sheet metal. The figures of merit for evaluating these five potential designs were the first frequency of the structure, the mass, and the cost.

Pallet structures for use on flight vehicles, or to be transported by them, have minimal natural frequency requirements levied on them; these are typically driven by the fundamental natural frequency of the launch vehicle itself. Finding the lightest weight flat or pallet structure given this constraint can be challenging as the design space is often large and the pallet geometry does not typically fit within classical closed form frequency solutions [1]. Presented here is a trade to select a pallet structure based on the natural frequency requirement. Four problems are considered in the trade space: isogrid/grid stiffened, flanged isogrid, open isogrid and metal-core-metal composite.

For each of the construction methods the optimal synthesis method of genetic algorithms coupled with finite element analysis/modal analysis to optimize structures for a required natural mode frequency was used to search through the design space for a set of good designs. The analytical setup including boundary conditions, geometry and lumped mass is shown in Figure 2.

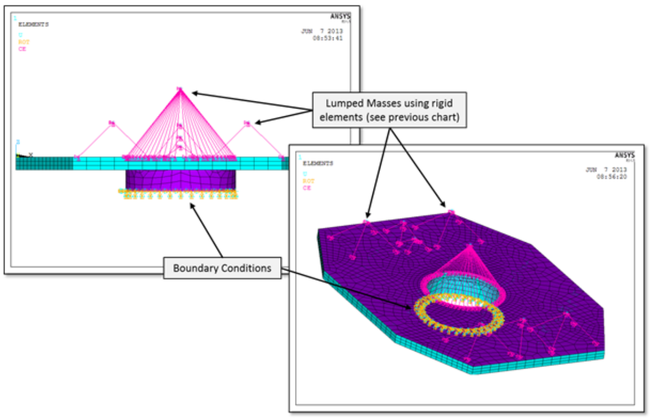

Figure 2: Boundary Conditions and Mass – Thickness of deck is notional only [2].

The objective function formulation is given as follows, where the mass and stress serve as governing constraints or penalties:

$$Objective = \frac{1}{f} \tag{1}$$

While the stress levels were verified to be benign, they were still checked for every function evaluation.

The genetic input parameters for each design option were held constant (generations: 200, population size: 12, mutation rate: 12%, multipoint crossover and a generational gap: 0.7).

The dimensions, mass, and frequency for each design option is given in the following table (
Table 1). The major difference between the isogrid/grid stiffened options is the stiffening of each design, i.e. the flanged isogrid design has a much higher stiffness than the open isogrid design. The composite design was optimized not for a frequency but for mass. As is the nature of composite structures, they possess high stiffness for relatively low weight; therefore achieving the minimum natural frequency was easily achievable absent of any optimization. With that in mind, it was decided to pursue the lightest weight structure and thus the individual curve for the composite design. It is also clear from
Table 1 that none of the isogrid designs met the minimum required 25Hz natural frequency. There are stiffening options that could be used to possibly achieve a higher natural frequency, for example higher frequencies and lower mass can be achieved with local geometry refinement of the grid stiffened designs. As represented in the table, these designs all have constant properties throughout, not geometry variance.

Table 1: Dimensions, mass, and frequency for each of the design options.

Isogrid	
Design Parameter	Value
b	*1.9 mm*
d	*46.2 mm*
t	5.1 mm
h	94.0 mm
Natural Frequency	*17.7 Hz*
Mass	*180 kg*
Flanged Isogrid	
Design Parameter	Value
b	*1.95 mm*
d	*43.7 mm*
t	2.54 mm
h	198.2 mm
w	8.4 mm
c	2.3 mm
Natural Frequency	20.2 Hz
Mass	174 kg
Open Isogrid	
Design Parameter	Value
b	*5.2 mm*
d	*48.3 mm*
h	52.2 mm
Natural Frequency	15.1 Hz
Mass	270 kg
Metal-Core-Metal Composite	
Design Parameter	Value
t1	*2.0 mm*
t2_matl	HRP Core
t2	127 mm
t3	*2.0 mm*
Natural Frequency	35 Hz
Mass	60 kg

From the construction methods considered in this study, the composite pallet structure is the only one to meet the frequency requirements and possesses the lightest mass. However, when considering the associated cost, the composite construction is the most expensive and thus does not meet the main criteria of lowest cost possible. Summarizing the main figure of merit and including the baseline sheet metal design in the table below (

Table 2) shows that the baseline sheet metal design best meets the requirements of the designs investigated.

Table 2: Summary of primary structure options and figures of merit.

	1st Frequency (Hz)	Mass (kg)	Cost
Isogrid/ Grid Stiffened	17.7	180	1:Lowest
Flanged Isogrid	20.2	174	2:Mid
Open Isogrid	15.1	270	N/A
Compositel	35	60	4:Highest
Baseline Sheet Metal	~35	83	3:Mid

Communications

At the start of the Project, both the rover and the lander had independent, direct to Earth, communication systems and carried the associated design in the Master Equipment List (MEL) and Power Equipment List (PEL). However, any communication system on the lander needs to be compatible and work in conjunction with the rover communication system to support the data transfer requirements of the overall mission through all its phases. Towards that end, two trade studies were conducted on the overall communications architecture for the mission as the requirements evolved. For both trades, common figures of merit that were considered were system cost, weight, power, performance, risk (development and system complexity), operations complexity and robustness. Details of the assumptions and ground rules, requirements, options considered and the baselined communications architecture that resulted from the two trades are described in the following paragraphs.

Trade 1 - The first communication system trade was performed in the December 2012 timeframe. The main purpose of the trade was to determine the communications architecture for the RESOLVE Polar Lander Mission, specifically focusing on allocating functionality between the Lander and Rover elements using the mission/design parameters that were known at that time. The focus of this trade was to allocate functions based on surface operations. The going in direction was that low cost was the highest priority. The ground rules/assumptions used as part of the trade are given in Figure 3.

- **Class D Mission:** 0 Fault Tolerant
- **Launch:** Spring 2017
- **Mission Profile** is 5 day cruise & 6-7 day surface ops.
- **Cruise**
  - Spin-Stabilized/BBQ roll
  - DSN Tracking (No RF ranging needed after landing)
- **Landing:**
  - Design for both S. Pole (Cabeus) & N. Pole (Fibiger).
  - Direct descent (no Lunar orbit)
- **Surface Ops**
  - No functions required of Lander post landing.
  - Max rover distance from Lander = 2 km
  - **Comm elevation angles** from lunar surface are 5-10 degrees above local lunar horizon (assume multipath degradation 8-16 dB)
    - Lunar Terrain 20 meters resolution
    - DTE evaluation is only on Lunar Surface, not as the Rover drives into a Crater, would need specific crater to evaluate this.
    - Radio Propagation did not take into account terrain.
  - **Surface Thermal:** Rover must operate through 6 hours of shadow.
  - Rover handles relative NAV (NOT using RF)
  - No RF ranging required post landing
- **Coverage**
  - 100% Transit/Landing
  - 80% Surface Ops
- A 3 dB link margin on all links
- Data Rates:
  - Cruise Fwd: 2-4 kbps, Rtn: 1.2 kbps
  - Landing Fwd: 2-4 kbps, Rtn: 84 kbps
    - To support real-time downlink of heavily compressed descent imagery.
  - Roving Fwd 2-4 kbps, Rtn 2 – 420 kbps

Figure 3: Assumptions and performance requirements for Trade 1.

The team evaluated five main options with many of these having a few different sub-configurations. The main options considered were:

1. Lander Direct to Earth (DTE) w/ Relay to Rover
2. Rover DTE (Lander uses Rover Communications during Cruise)
3. Rover DTE and Lander DTE (totally separate systems)
4. Rover DTE and Lander DTE with Relay/Cross Link
5. Orbiting Communications Relay

After an initial evaluation, Option 4 and Option 5 were ruled out. Option 4 exceeded the functional requirements for redundancy and its complexity/cost was too high, whereas Option 5 was cost prohibitive given the budget constraints. A sub-study looked at the frequencies, infrastructure assets, etc. and recommended using the X-band frequency for both the command and telemetry links due to spectral availability, spectral efficiency, and more widely available Deep Space Network (DSN) support. Reference architectures and their associated MEL/PEL link margins, communication coverage and an estimate of the system Design, Development, Test, and Evaluation (DDT&E) costs were developed and analyzed. These options were evaluated against the figures of merit used and ranked. The different metrics were weighted against their importance where lowest cost was the most important factor, followed by mass, power, system complexity, etc. Based on the options traded, the recommended architecture was Option 2 (Rover DTE) communications with X-band and with the Lander using the rover communication system during the transit phase. Post-trade, the decision was made to maintain an independent lander DTE link and an independent rover DTE, both at X-band. However, the integrated approach to communication during the cruise phase had not been completed.

After the first trade, the overall mission requirements were refined, the landing site and mission dates were changed, and the original requirements for real-time downlink of descent video was changed to recording the video and doing a post-landing dump. Based on these changes another trade with the updated requirements was conducted with the same figures of merit as the first trade.

Trade 2 - The second communication system trade was performed in the May-July 2013 timeframe prior to RPM Mission Concept Review (MCR). This trade looked at an integrated approach to communications for all mission phases using the revised requirement set. The updated ground rules and assumptions for this traded are given in Figure 4 and Figure 5.

- **Class D Mission:** 0 Fault Tolerant
- **Launch: April 2018**
- **Mission Profile** is less than 10 day cruise (5 day ideal) & 6-7 day surface ops.
- **Cruise**
  - Spin-Stabilized/BBQ roll at 1 RPM
  - DSN Tracking (No RF ranging needed after landing)
- **Landing:**
  - **Design for S. Pole (North-West of Haworth Site)**
  - **4/30/2018 05:46 UTC**
  - Direct descent (no Lunar orbit)
- **Surface Ops**
  - No functions required of Lander post landing.
  - Max rover distance from Lander = 3 km
  - **Comm elevation angles** from lunar surface are 4°-10° above local lunar horizon (multipath degradation assumed to be 8 dB for high-gain antenna, 16 dB for low-gain ant.)
    - Lunar Terrain 20 meters resolution
    - DTE evaluation is only on Lunar Surface, not as the Rover drives into a Crater, would need specific crater to evaluate this.
    - Radio Propagation did not take into account terrain.
  - **Surface Thermal:** Rover must operate through continuous 6 hours of shadow. 100% comm Coverage assumed in shadow operation.
  - Rover handles Relative Nav (NOT using RF)
  - No RF ranging required post landing

Figure 4: Ground rules and assumptions.

Capitalizing on some of the results from Trade 1, the two main options that were considered as part of this trade was 1. Have all the communications on the rover and none on the lander, and 2. Have a lander system to support transit communication requirements and a rover system to support surface communications. For each of these options, three sub-options were considered giving a total of six options that were evaluated. The six options considered were:

A. All communications on the rover
- A1. An all X-band system using DSN
- A2. S-band transponder and X-band transmitter using DSN
- A2S. An all S-band system using DSN

B. Lander Communications for transit and Rover Communications for surface ops
- B1 All X-band using DSN
- B2 S-band for transit phase and X-band for surface ops using DSN
- B2S All S-band using DSN

- **X-Band or S-Band** Fwd/Rtn for DTE
- **Link Margin** 3 dB (Bit Error Rate: 1 error per 10^5 bits)
- **Data Rate Requirements**

Phase			Forward	Return		
				User Rate	Total W/Overhead	
Cruise (95% Omni Coverage required for 4 critical TCMs)			2 kbps	4 kbps (User Rate with minimal overhead)		
Surface				Max Data Volume = 28.2 GB (5.8 days x 450 kbps)		
				Nominal High-Rate		Contingency Low-Rate
	Stopped	Payload	2-4 kbps (May need higher rates for SW updates but assumed not an issue)	450 kbps Sample Processing	600 kbps (450 kbps + 30% overhead)	2 kbps
		Rover		0		
	Roving	Payload	2-4 kbps	2-40 kbps	400 kbps (296 kbps + 30% overhead)	2 kbps
		Rover		256 kbps (for teleops while roving)		

- **Coverage**
  - 100% Emerg. availability
  - 10% Cruise
  - 100% Crit. Event/Landing
  - 100% Surface Ops

100% for all critical events.

Figure 5: Performance requirements.

During the early stages of the trade, use of the Near-Earth Network (NEN) was considered (as possible cost savings over DSN). However, once initial link calculations were performed, the use of the NEN sites was eliminated due to high power requirements to close the link. Options that did not have ANY data connections between the Rover and Lander (each with independent DTE links) were eliminated due to the requirement to dump descent imagery post-landing using the Rover communication system. The option to use Rover communications without a Lander mounted antenna was also eliminated after initial coverage analysis showed that Rover antennas alone could not achieve required coverage during cruise phase.

Reference architectures and their associated MEL/PEL link margins, communication coverage and an estimate of the system DDT&E costs were developed and analyzed for each of the six options considered. The results were mapped against the figures of merit and evaluated. The study recommendation was option A2S with all the communications using S-band and situated on the rover. An antenna on the lander was needed to meet the transit communications coverage requirements during critical events. A notional reference architecture for Option A2S is given in Figure 6. This architecture was the baseline used for the Lander Communication system during RPM MCR.

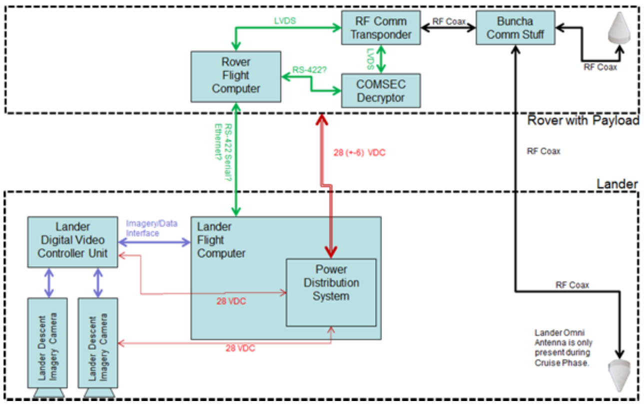

Figure 6: Notional reference communication architecture for Option A2S.

Power

The electrical power system for a spacecraft must produce, store, manage and distribute electrical power to spacecraft loads. Custom designs and hardware are the optimum solution to produce an electrical power system with the highest efficiency and best function; however, off the shelf hardware (OTS) or reutilization of existing designs is an acceptable way for projects to lower cost. There are many OTS options available for use that provide electrical power production, storage, management and distribution. Typical OTS avionics packages today can be obtained with flight heritage and offer multiple channels easily configured to provide 0 – 5 amps of protected output current. Versions are available that provide radiation shielding if required and additional shielding may be added with exterior enclosures. The OTS avionics packages with integrated power system management are commonly for use of an electrochemical battery to store energy and a photovoltaic array to produce power from sunlight. The power system management is built into the avionics package and marries the solar array to the battery while managing the power bus within specified limits. OTS hardware for electrical power systems other than solar array/battery is not readily available. The current baseline design for the Lander utilizes an off the shelf avionics package manufactured by Moog Broad Reach Engineering. This avionics hardware is the same equipment that was successfully flown on the LADEE (Lunar Atmosphere and Dust Environment Explorer) mission. This avionics package utilizes an expandable card chassis and can easily be configured to meet the requirements of the baseline Lander. A SACI (Solar Array Charge Interface) card manages the interface between the solar array panels and the battery.

Solar arrays and electrochemical storage batteries are the means most often used to produce and store electrical energy; however, other options exist for producing and storing electrical energy. The means of power system management must accommodate the method of energy production and storage. Choosing the best option for energy production and storage is based on performance requirements primarily, but must also meet the requirements for cost, mass, volume, reliability, etc. Performance requirements include mission length (chronological), number of power system cycles, environment (thermal and light), rate capability, etc. If an option other than solar arrays and batteries is chosen for power generation and storage, very few options exist for OTS power management and distribution. Mass, volume, energy density and cost were the determining factors in choosing how to source and store electrical power for robotic landers.

Electrical power can be sourced with photovoltaics, fuel cells, isotopes, reactors, etc. Any mission destination with solar insolation can utilize photovoltaics. Photovoltaics are most effective within one planet of the Earth, any further from the Sun and the array size approaches being unmanageable, any closer to the Sun and the temperature is impractiably high. Permanently shadowed regions are a challenge and indicate isotopes or reactors. Deep space exploration power has come from isotopes or reactors to this point. Photovoltaics can be used for Lunar exploration. Lunar Polar regions are easier than equatorial regions. Three junction Gallium-Arsenide photovoltaic cells are baselined for the Lander. These cells have efficiencies approaching 30% at the cell level and are the current industry standard. Cost and mass primarily drove the decision.

Energy may be stored electrochemically (batteries, fuel cells) or mechanically (flywheels). Mechanical storage involves a conversion step and is not usually the most efficient. Fuel cells have application for limited durations and are especially suited on manned missions when byproduct water is needed. Electrochemical batteries are usually most efficient if the mission environment and duration is within their capabilities. Lithium-Cobalt Oxide (LiCO) cells have good cycle life, energy density and rate capability. Safety considerations for LiCO batteries are manageable for unmanned as well as manned missions. LiCO cells were chosen as the energy storage medium for the Lander. Testing is being performed on LiCO with Aluminum alloying to meet the higher temperature requirements that would be present at the Lunar Equator. This decision was also primarily driven by cost and mass.

Thermal

The thermal design of the lander is intended to be passive, simple, reasonably reliable, and low cost, consistent with the Class D philosophy. The primary function of the thermal control system (TCS) is to effectively maintain lander component temperatures within operating and survival ranges, as appropriate, from launch vehicle separation through transit, during landing, and post landing prior to rover drive-off (payload deployment) to ensure the safe and proper delivery of the surface payloads to their destination. Further, the thermal design is intended to provide an environment for the rover/payload during transit that facilitates the design and simplifies, protects, and supports operation of the rover.

A key feature of the thermal design is the use of a spinning attitude during transit with the spin axis, thru the center the vehicle and aligned with primary thrust axis of major propulsion thrusters and solid rocket motor, and perpendicular to the solar vector. This attitude evenly distributes solar heating on all external surfaces that see the sun while minimizing temperature gradients and, potentially, heater power on major components such as the solid rocket motor. This spinning attitude and the rover's placement on the lander creates a solar incidence angle for the rover/payload similar to that expected on the lunar surface.

The thermal design uses existing, flight proven components and technologies to the maximum extent possible to minimize complexity and the development costs, as well as keeping overall costs low. Most components are insulated via multi-layer insulation to minimize heat loss to and heat gain from the environment; the design and materials, of which, are tailored to the specific needs of each component. Aluminized Mylar foil and Dacron mesh are used for low and moderate temperature components, while aluminized, embossed Kapton is used for higher temperature components. Kapton outer layers are used for most blankets. AFRSI blankets are added in key places to accommodate plume induced high temperatures. An MLI blanket is also used to close out the bottom of the deck that would see the SRM (and its plume when firing) and might also provide protection against descent thruster plumes; this blanket would provide thermal protection against the cold during most other operations.

Passively controlled heaters implemented on most components are intended to provide low temperature protection of components and propellants, as necessary. An alternative approach would use software controlled heaters tied to temperature sensors which are attached to critical components throughout the lander; this method was not pursued in an attempt to save money and reduce interface and verification complexity. Current design studies are investigating the use of either mechanical thermostats or positive thermal coefficient (PTC) ceramic self-regulating heaters to eliminate any external control and minimize sensor systems. Thermal sensor instrumentation used in the lander design is designated only as health/status and will not be used for automated control to minimize cost and complexity. Health/status thermal sensors can be used to evaluate flight temperatures during a given attitude/orientation. Initial heating zones were established, based on thermal modeling, to maintain proper propellant temperatures during transit and off nominal orientations. These zones are tailored to the functions and predicted environments associated with each component. These zones will be grouped together in parallel and series to form heater circuits in subsequent design efforts. Heater redundancy, while an important consideration, has not been addressed in the design at this point. Due to the Class D status of this mission, higher risk is acceptable in order to keep the costs low. In some cases, this means foregoing the redundancy in an effort to reduce overall costs. Future work entails a failure modes and effects analysis (FMEA) to determine the potential consequences of failures due to this zero fault tolerance method. If there is a consequence that is unacceptable to the mission, specific redundant heaters may be introduced into the design.

One of the key features of the current pallet lander is the distributed nature of the propulsion system, with 16 descent thrusters and 12 attitude control system thrusters allocated to four "thruster" modules located around the perimeter of the lander. Due to the large number and distribution of thrusters, simplification of the thruster heater system has led to a configuration that shares heat among the various heaters within a single thruster module, rather than multiple heaters per thruster. Essentially, each thruster module is surrounded with MLI (high temperature version where necessary) and the heaters are applied to the main and secondary thruster brackets upon which all the thrusters are mounted. Analyses have shown that this configuration provides the necessary thermal conditions for the thrusters during non-operation periods and still minimizes the required heater control and power resources to do so. Additional analysis is still needed to examine the efficacy during periods during which the thrusters operate and the soak back periods immediately afterwards.

The geometry of the pallet lander provides locations under the primary deck to locate avionics and ample area on the top of the deck for centralized radiators that face up with little to no obscuration. Initial trade studies attempted to allow each avionics box to have its own radiator and view to space. However, for some boxes the heat rejection was too large for available rejection area. Based on this and other considerations, such as ease of component level/subsystem level integration and testing, a "centralized" radiator concept was employed. In this concept, there is an avionics radiator on the deck near each end of the rover (conceptually, one for the integrated avionics unit (IAU) and related electronics and for the GNC related electronics). On each of these radiators a cluster of electronics is attached. The radiator area is sufficient to provide adequate heat rejection during hot conditions. The radiators have white paint as an optical coating, and some of the radiator area is covered with MLI to allow for adjustments, during the design process, to the heat rejection area as the design and heat rejection requirements targets change. This clustering of avionics on a central radiator not only provides sufficient heat rejection area, but it makes more effective use of available heat rejection area and allows the sharing and mutual heating of the clusters of electronics during colder portions of the mission. The radiators also act as part of the sheet metal deck structure, although details on thermal and vibration isolation are still to be worked out.

While a separate part of the lander, the Solid Rocket Motor (SRM) uses similar techniques to maintain thermal control. In a previous project there were significant studies surrounding the SRM, investigating various thermal management strategies, including spinning the spacecraft, black paint, heaters, and MLI. From those studies, it was found that spinning the spacecraft was the most cost-effective method for keeping the SRM within bounds. In addition, the previous studies found that plume impingement and SRM overheating were issues if no MLI were used. Thus, a combination of spinning the spacecraft and MLI were recommended as the thermal control methods for the SRM. The spin attitude is beneficial in distributing the solar load relatively evenly around the perimeter of the SRM. MLI is used to insulate the SRM. This MLI is a combination of moderate temperature MLI, which offers the best vacuum insulation performance, and high temperature MLI to protect the moderate temperature MLI and the SRM from potentially significant descent thruster plume heating. This insulation is placed on the propellant casing and can also be placed on the nozzle. In addition an insulating end cap is placed over the nozzle exit to minimize heat transfer from the inside of the SRM. This end cap is blown away during SRM firing. Although the insulation and distributed incident solar heating minimize the need for heaters, to mitigate some of the risks concerning the SRM thermal stability, additional heaters are implemented on the SRM to maintain the SRM within a relatively tight temperature range.

Propulsion

The trade study for the propulsion sub-system follows from a reference configuration which had been used as an initial concept. 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 as an ATK STAR-48V, 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 propellant is consumed during the terminal descent.

Because of the mission cost constraint, the main focus of the trade study was to find ways of reducing cost in propulsion system development, while technical risk, system mass, and technology advancement requirements were also taken into the consideration. For the braking stage, liquid oxygen (LOX) and liquid methane (LCH4) propulsion systems, derived from the Morpheus experimental lander, and storable bi-propellant systems, including the 4th stage Peacekeeper (PK) propulsion components and Space Shuttle orbital maneuvering engine (OME), were also considered for the study. It should be noted that the PK propulsion components and Space Shuttle OME are government-owned hardware which can be used without procurement cost. For the lander stage, the trade study included a miniaturized Divert Attitude Control System (DACS) thrusters (Missile Defense Agency (MDA) heritage), their enhanced thruster versions, ISE-100 and ISE-5, and commercial-off-the-shelf (COTS) hardware. The rationale for selecting the propulsion configurations and components for the study is summarized in Table 3 below.

Table 3: Rationale for selecting propulsion configurations for the 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 provides an opportunity to demonstrate technology for future exploration.
Hypergolic bi-prop	Flight qualified, low cost because of using operational government-owned propulsion components, such as 4th stage PK and Space Shuttle MOE.
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-owned propulsion components of 4th stage PK missile.

Several criteria were established for the propulsion system down selection. The cost of developing the propulsion system should be within the projected allocation. The lander mass should be within the launch capability. At the time of the study performed, the acceptable mass would be below 3600 kg since the flight mission had a baseline of using the Falcon 9 V1.1 launch vehicle. Another criterion was to assess the work schedule and technical risk level regarding whether they were suitable for the RP mission. Based on the described criteria, four options of the 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 4 and Table 5.

Table 4: Trading the major figures of merit.

Option	Config.	Cost	Lander Mass
Original Reference	ISE/ SRM	High	Low
Option 1	PK/ SRM	Low	Medium
Option 2	Existing DACS/ SRM	Medium	Medium
Option 4	Mono Prop hydrazine/ SRM	Medium	High

Table 5: Summary of pros and cons.

Option	Config.	Pros	Cons
Original Reference	ISE/ SRM	• Lightest weight • New technology demo. for future • Reduced heater requirements	• Highest cost • High risks (technical and schedule)
Option 1	PK/ SRM	• Lowest cost • Hardware avail. w/o cost.	• Moderate weight increase • Lowest performance • No technology demo.
Option 2	Existing DACS/ SRM	• New technology demo. for future • Potential reduced heater requirements	• Moderate cost • Moderate weight increase
Option 4	Mono Prop hydrazine/ SRM	• Low/moderate cost • Simple , reliable system • Extensive flight data	• Heaviest • No technology demo.

Option 1 had the lowest cost while the lander mass was within the launch capability; hence, this option was selected as the reference for the propulsion system. It should be noted that the PK thruster is capable of generating approximately 70 lbf, and the lander will require sixteen PK thrusters for terminal descent instead of the twelve 100 lbf thrusters as shown in the original reference concept. Because of this selection, risks identified in Table 6 have been mitigated. The PK thrusters were tested in vacuum conditions, and the test results indicated that the thrusters will meet the mission requirements.

Table 6: Qualitative assessment of risks.

Option	Config.	Risk
Original Reference	ISE/ SRM	• Thrusters 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 4	Mono.Prop hydrazine/ SRM	• Potential interference w/ optical landing devices during descent due to continuous thruster operation (throttling instead of pulsing) • Plume effects to SRM • Thrusters not in production. • High pressure operation • Large size of feed lines & large tanks

Avionics

Two trade studies were conducted during selection of the Avionics Architecture for the Resource Prospector mission (RPM). Given RPM is a Class D mission, high TRL Avionics were sought out. The two trade studies conducted as a part of this effort were: 1) Determine the Avionics Architecture and identify systems that were currently available and 2) Define the software processing needs for the mission.

Trade Assumptions - The driving factors in the selection of Avionics were: 1) Use avionics with flight heritage if possible to reduce the cost and schedule (build-to-print as much as possible) and 2) Avionics is to be a single string system.

The following functional assumptions were used in support of the trades.

- Provide Command & Data Handling processing for subsystems (GN&C, thermal, propulsion)
    - o Assume a separate processor for Terrain Relative Navigation (TRN)
- Provide a platform for the command and telemetry processing and interfaces
- Provide mass storage
- Provide spacecraft time maintenance
- Detect launch vehicle separation and begin the lander initialization sequence
- Lander avionics are independent from rover avionics
- Avionics fully powered during cruise and landing
- Lander functions end after successful landing and release of rover

Trade Space - It was determined that no Avionics hardware was readily available (i.e. designs existed but no flight hardware was available). Therefore, build-to-print from an existing design was determined to be the most cost effective solution. A limited market survey was conducted to identify existing Avionics designs. Build to print viability was used to down select from the identified designs (as shown in Figure 7).

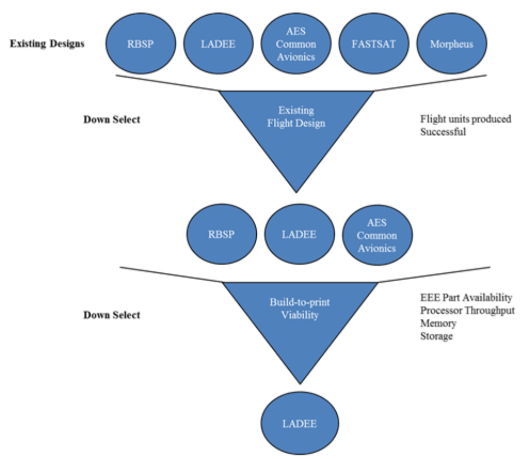

Figure 7: Trade down selection.

The down select identified the Lunar Atmosphere and Dust Environment Explorer (LADEE) Avionics design as meeting all the requirements for the Resource Prospector mission. At the time of selection, the LADEE design had just been completed and was scheduled to launch within a year. Therefore the LADEE hardware had a high TRL (8-9).

In addition to a high TRL, the LADEE hardware had the advantages of being currently available, consisting of a flexible design, and no obsolescence issues. The LADEE design consisted of a combined Avionics and Power system with the following power functions incorporated;

- Command and telemetry interface to Electrical Power Systems (EPS) functions for switched output commanding and switch status including analog telemetry provided in a single chassis.
- Primary power input interfaces for solar arrays and batteries (DC/DC converters internal to the hardware convert bus voltage to secondary voltages used by the spacecraft)
- Power switching for spacecraft subsystems, payloads, propulsion, heaters, and mechanisms

This resulted in the following Avionics architecture to be baselined for RPM.

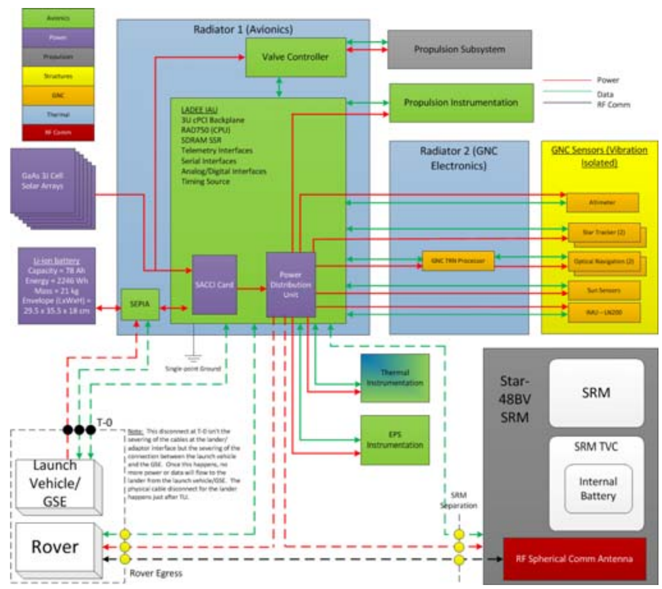

Figure 8: Avionics architecture

Goals and Objectives of Second Avionics Trade - In an attempt to further define the Avionics architecture, a second trade study was conducted. This trade focused on the software processing needs and subsequent processor selection. Benchmark testing was conducted on multiple processing platforms.

The RPM Lander software architecture and approach is based on the Morpheus project (a terrestrial Lander project at JSC). The software is based on Core Flight Software (CFS) from Goddard Spaceflight Center (GSFC). The benchmarking approach was to run the Morpheus software with only slight modifications to make it mimic the functions needed for RPM. This software was then run on multiple processors to assess performance.

The processors assessed were: AITech S950 (used on Morpheus), Space Micro Proton 400K, Maxwell SCS750 and the AITech SP0. The results are shown below in Table 7 and Table 8 (CPU usage was calculated over the ascent, hold, translate, and descent profile similar to a Morpheus Tether Test).

Figure 3 – CPU Benchmarking

Table 7: CPU Benchmarking

CPU Usage	S950 1 GHz	Proton 400K 800 MHz (1 core)	Maxwell 400 MHz	SP0 1 GHz
High	51%	54%	100%	45%
Average	15%	25%	51%	14.5%

Table 8: CPU Benchmarking continued - Issues experienced.

Processor	Issue Experienced
S950	Thermal (CPU requires additional cooling)
Proton 400K	• POSIX timers did not work correctly in VxWorks. • Current TTMR technique, used to reduce SEU induced error rate, requires unacceptable overhead for SW development (triplicating code)
Maxwell	• Size (6U vs. 3U) • Expected better performance with 800MHz, but test runs failed due to a 'random' data access (from RAM) exception error
SP0	Experienced some data drop with UDP connection, but it was because the internet port configured with simplex mode

The RAD 750 processor was determined to be inadequate to run the current Morpheus based code. No attempt was made to optimize the code which could possibly result in the SW adequately running on the RAD750.

Conclusions - The LADEE architecture and hardware was considered an acceptable solution for RPM. It was determined that there was no driving technical need to make a final processor board selection at this time.

4. SUMMARY

This paper reviews the details of the Resource Prospector Mission and specifically focuses on the lander element. The chosen architecture was discussed, as well as the philosophy in developing that architecture. Finally, we reviewed the key subsystems and the trade studies that ultimately led to their final designs.

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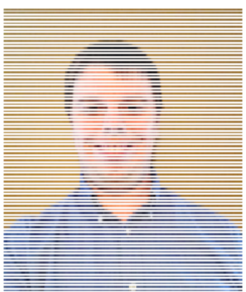
[2] Burt, Adam O., and Hull, Patrick V., "Optimization of a Lunar Pallet Lander Reinforcement Structure using a Genetic Algorithm," 15th aiaa/issmo multidisciplinary analysis and optimization conference, 2014.

BIOGRAPHY

All – I need a head shot (1.25” wide x 1.5” high) and a brief biography from everyone. The biography can include title, fields of expertise, work experience, education, and relevant personal information.

Dr. Kristina Rojdev received a Ph.D. and M.S. in Astronautical Engineering from the University of Southern California, and a B.S. in Aerospace Engineering from the University of Michigan. She has been with NASA for 10 years. She has been the lead for instrumentation on the Deep Space Habitat project, as well as the lead and principle investigator studying long-term radiation effects of composite materials for habitats. She is also involved in various radiation shielding studies for deep space missions. She is part of the systems engineering team for the RP lander and is also currently involved in SysML modeling activities for deep space habitat power systems.



Derek Calvert has a B.S. in Mechanical Engineering from the University of Alabama. He was worked with NASA as either a contractor or Civil Servant for seven years. He is currently working in the Space Systems Department as the lead systems engineer on a 12U cubesat project to demonstrate iodine propulsion. He has served on the systems engineering team at MSFC for the RP Lander project. Prior to that project, he provided systems engineering support for SLS secondary payloads integration and spent five years working operations as a flight controller for the International Space Station.

Sharada Vitalpur is the Deputy Branch Chief and Chief Architect of the Wireless and Space Communication Systems Branch at Johnson Space Center. Her expertise is in communication systems, architecture, signal processing and end-to-end analysis. She has worked on different communication systems for the Space Shuttle, the International Space Station and was the C&T System Manager for Orion. Ms. Vitalpur started her career at ITT Aerospace Corporation designing, analyzing, modeling and implementing algorithms and protocols for different DoD communication system. She has numerous publications and attended Purdue University in West Lafayette, Indiana where she received her Master of Science in Electrical Engineering and Bachelor of Science in Electrical Engineering with Highest Distinction.