SMALL SPACECRAFT IN SUPPORT OF THE LUNAR EXPLORATION PROGRAM

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This paper analyses the ability of small, low cost spacecraft to deliver scientifically and technically useful payloads to lunar orbit and the lunar surface, in particular precursor mapping, infrastructure and in-situ resource utilization functions, that are necessary prior to human return as part of the US Vision for Space Exploration. It is based upon a technical study of the NASA-Ames Research Center's Small Spacecraft. Following an overview of the generalized capabilities of small spacecraft in comparison to the objectives of the robotic lunar exploration program, the paper documents the mission planning (including trajectory, launch stack and timeline), and overall spacecraft design (including mass budget, structure, propulsion, thermal, electrical power, descent guidance, navigation and control, and telecommunications) for a lunar lander mission. The study shows that spacecraft subject to the constraints laid out, in particular within a budget of < $100 Million and which can be launched on one of the next generation affordable launch vehicles such as Falcon-1 or Minotaur-V, can deliver payloads of 5-50 kg to the lunar surface or 10-200kg payload to lunar orbit. The payloads carried would be capable of covering most of the functions of lunar missions that are needed prior to human arrival, as identified in NASA's Lunar Robotic Architecture Study, with the exception of the bulk ISRU tasks of the 'Lander Rover' (In-situ Resource Utilization (ISRU)) mission. The key advantages of smaller spacecraft are reduced costs and schedules.

I. INTRODUCTION

This document summarizes the ability and limitations of small spacecraft to do space exploration missions for NASA. It focuses on a 'Micro Lunar Lander' Case Study developed by NASA's Ames Research Center (ARC. It is intended to document the key assumptions, analysis and trades performed. The paper is based upon an analysis performed in a broader study effort of the Small Spacecraft Office (SSO) at NASA-ARC that is focused on the development of a common satellite bus design that would adaptable to a variety of missions, both exploration and scientific, at a vastly reduced cost.

A. Small Spacecraft

Over the last decade, small spacecraft have gone from novelty to having significant functional capabilities. An increasing number of capabilities (although by no means all) can be achieved with small spacecraft due to the general miniaturization of technology. Such missions can be done at a fraction of the cost and schedule of existing missions. This is causing a mini revolution in the space field by allowing many more actors to access and utilize space. Several countries, companies and universities are beginning to use small spacecraft in many areas of civil and military uses in order to get more from space within

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given budget constraints. Not all actors have embraced this trend and the US government included has yet to embrace the ability of small satellites to do many of their missions.

To illustrate the problem with the status quo, the SBIRS High spacecraft were conceived and designed in the early 1990s. The sensors that were chosen were the best at the time and reportedly billions of dollars were spent on their development. Today those sensors are less capable than many that are readily available at miniscule costs on the commercial market. Instead of developing high cost technology such as these, the government could do better just by decreasing the development cycle from 10-20 years to 1-2 years.

Key Characteristics of Small Spacecraft Missions[1]

1. Low cost (~ $50-100M)
2. Fast turn around (12-36 months from authority to proceed (ATP) to launch)
3. Use of latest technology
4. Use of next generation of affordable launch vehicles (Minotaur V and Falcon I)
5. Use off-the-shelf technologies wherever possible (both commercial and other)
6. Leveraging technologies from the US Department of Defense (DoD)
7. Higher risk missions (Class D as per NPR 7120.5D)

Key Advantages of Small Spacecraft

1. Low Cost
2. Decreased schedule
3. Increased number of missions (as a result of (1) and (2)), allowing:
   a. A fast learning cycle for spacecraft development
   b. High public participation and attention
   c. Many opportunities for international collaboration
   d. Exciting focal points for children with the potential to help increase interest in science and engineering, and in particular to attract those into the space sector
   e. A small overall program risk
   f. Ability to do more high-risk missions (e.g. testing new systems of a lower technology readiness level)

Key Limitations of Small Spacecraft

1. Size and mass constraint make small spacecraft not directly useful for some set missions. In the case of robotic lunar exploration this means in particular that heavy ISRU equipment and large rovers are not feasible.
2. Higher individual risk on missions with potential negative political ramifications
3. Reliance on yet to be proven launch vehicles, or on being a secondary payload on a larger mission
4. Sometimes more expensive per unit mass of spacecraft

B. Summary of Current Architecture for Lunar Exploration

Given the Presidential goal of performing "extended human missions to the moon as early as 2015, with the goal of living and working there for increasingly extended periods," there is a need to answer certain questions, in particular about resources and the potential location for the outpost, prior to human missions. The tasks of the robotic exploration program that need to be completed, in order to expedite human missions efficiently and safely, are well spelled out in the *LRAS*[2]. A summary of the mission architecture therein is given below:

Mission 1: Lunar Reconnaissance ('LRO-like') [2010]. Tasks: visual & topographical maps, hydrogen map, radiation environment.

Mission 2: Fixed Lander [2011] Tasks: precision landing, dust characterization, regolith composition and thickness, lighting and thermal ground truth.

Mission 3: Communications Orbiter (co-manifested with Fixed Lander) [2011] Tasks: partial coverage of south polar region.

Mission 4: Mobile Lander (North Pole) [2013] Tasks: water presence in 20 sites of shadowed crater, radiation shielding of regolith, effects of lunar environment on life and mechanical structures.

Mission 5: Lander Rover (South Pole) [2015] Tasks: ISRU of O_2 and H_2O (produce up to 1000kg), fluid experiment, 30km roving.

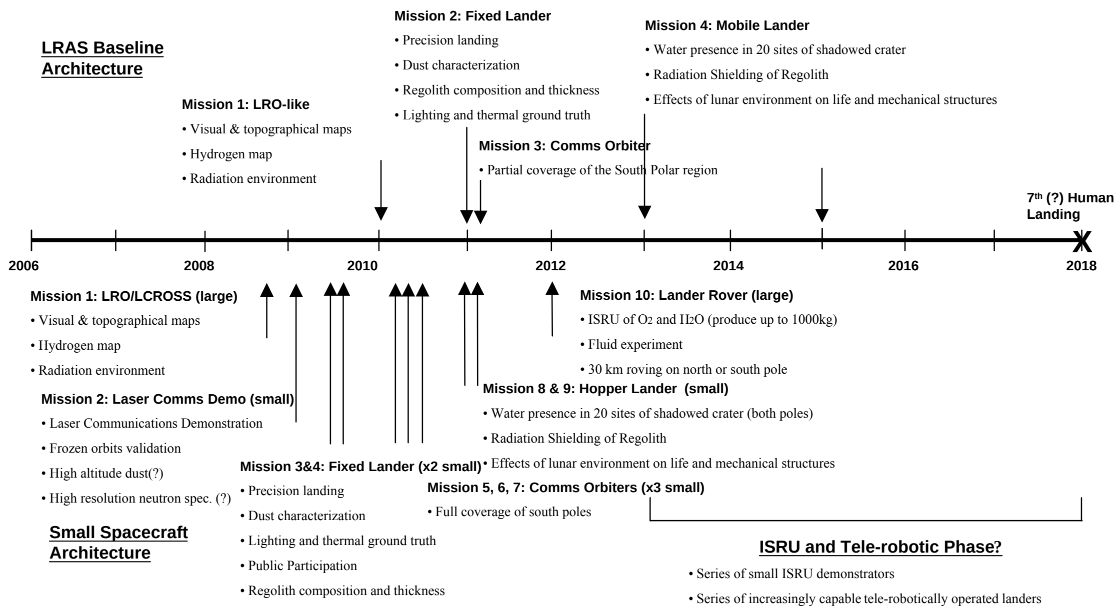

Figure 1 Small Spacecraft Architecture Concept as compared to the *LRAS* Architecture

C. What can small spacecraft missions do for the Lunar Exploration Architecture?

The approximate throw mass of Falcon-1 and Minotaur-V launch vehicles to lunar orbit and lunar landing are shown in Table 1.

Table 1
BASIC MISSION OPTIONS

Concept	Rocket Motors (Launch/TLI/breaking/descent)	Wet Mass (kg)	Payload (kg) [3]	Cost ($M) [4]
Lander 1	Falcon-1/Star-30BP/Star-15G/KKV	45 [5]	**7.2**	65
Lander 2	Minotaur-V/-/Star-15G/KKV	86 [6]	**14.8**	88
Lander 3	Minotaur-V/-/Star-27/KKV	143	**70.3**	97
Orbiter 1	Falcon-1/Star-30BP/KKV	68 [7]	**16**	55
Orbiter 2	Minotaur-V/-/Star-15G/-	357	**198**	125

Given the priorities identified in the *LRAS* and the limitations of small spacecraft identified above, in particular a payload mass limit of < 50 kg), an initial study of the instruments to achieve the 15 objectives given, shows that:

1. The only task (of the 15) definitely not possible with current technology on small spacecraft missions is that of large scale (e.g. > 1000 kg mass) ISRU of O_2 and H_2O.
2. Tasks that are in a grey area – that are possible with small satellites but which may or may not be preferential to do with small missions and require further analysis – include:
   a) 30km roving[8]
   b) Water presence in 20 sites of shadowed crater[9]
   c) 1-year operation with periods of shadow

On first analysis, Mission 5 of *LRAS* needs to remain a large lander and there is a need for further study to decide whether Mission 4 could be done more cost effectively with small spacecraft or not. Given that LRO is proceeding, a small spacecraft architecture might replace Missions 2, 3 and 4 with several (e.g. 4-10) small missions and an accelerated overall schedule and with reduced overall cost. Mission-II could be replaced by two small fixed landers compared with *LRAS*, Mission 3 accomplished by four small communications orbiters (which would have the added advantage of providing permanent coverage of south polar region), and Mission 4 by two small hopper landers (one on each pole) as per the timeline below. The total cost would be considerably less than the LRAS baseline and over a shorter period of 2006-2013 leaving room for a telerobotics and ISRU phase.

II. THE MICRO LUNAR LANDER

A. Mission Introduction

The design to follow is a small unmanned lunar lander which could be developed in under 36 months from authority to proceed (ATP) and for a total mission cost under $100 million. The goal of the mission is to work down the decision tree of exploration relevant questions regarding the surface of the moon, as well as descent technologies in order to enable, increase the effectiveness and safety of human missions. The former would include exploring the dust environment, obtaining detailed terrain mapping and localized composition of the regolith, the form and extractability of hydrogen at the poles and the nature of the peaks of eternal light on the poles. This would, in approximate terms, cover the goals of the 'Fixed Lander' mission in *LRAS*.

The technical concept utilizes a core set of hardware that was leveraged from existing US DoD investments. The propulsion system concepts under consideration are from the DoD's Missile Defense Kinetic Kill Vehicle programs such as EKV, THAAD, ASAT and LEAP. The avionics and software are based on the XSS and NFIRE programs. The DSMAC image based navigation system is from the Tomohawk cruise missile. Most other technologies are commercial off-the-shelf in order to avoid development costs and schedule implications. In addition, much of the guidance and navigation control hardware is has been used extensively by NASA, the US Air Force Research Laboratory (AFRL), the Defense Advanced Research Projects Agency (DARPA) and others.

B. Objectives

Political:

1. To demonstrate progress towards the US President's vision that "Beginning no later than 2008, we will send a series of robotic missions to the lunar surface to research and prepare for future human exploration."
2. Make steps towards the "goal of living and working there for increasingly extended periods": for which there is a clear need to answer certain questions, in particular about resources and the potential location for the outpost, prior to human missions.
3. Demonstrate a publicly visible step towards the lunar exploration program before the end of the political cycle at the end of 2008.

Managerial:

4. To successfully develop and deploy a soft-landing spacecraft onto the Lunar surface with the following boundary conditions
   a) Timescale: < 36 month from ATP to launch

b) Cost: < $100M (including launch)

Technical:

5. Retire operational and technical risks for human lander missions, including to safety test the descent algorithm to be used for the human landers
6. Demonstrate landing precision of <1km.

Scientific:

7. Develop a design that could support future scientific payloads
8. To investigate, if possible:
   a) The Lunar Dust characteristics
   b) The Hydrogen quantity and form in the regolith at the Lunar equator.

Public Exploration:

9. To provide an opportunity for real public participation in the mission, through, for example, real time data streaming on descent.

C. Mission Requirements

The following represent a set of Level 1 requirements for the Micro Lunar Lander. The intent is that these requirements represent a cost effective approach to fly as a Class D mission. The lander shall:

1. Be compatible with either the Falcon 1 or Minotaur V launch vehicle.
2. Have a minimum of 5 kg payload to the lunar surface, and 130 Watts minimum over 80% of the lunar day.
3. Be based on a common/modular spacecraft bus platform. This would be suitable for a variety of missions, such as:
   a. 130 kg lander with a minimum of 50 kg payload to the lunar surface.
   b. Lunar communications orbiter
   c. "X-Nav" navigation payload
   d. A near earth object mission
4. Be designed for equatorial landing but be adaptable for polar landing.
5. Be operational for at least one lunar day (14 earth days) and one hour after sunset to measure the dust phenomena of the terminator.
6. Perform a descent that[10]
   a. Impacts the lunar surface with vertical and horizontal velocity components of up to 4 m/s and 1 m/s respectively. [11]
   b. Survives impact (not tip/roll over) from landing on slopes up to 15 degrees. [12]
   c. Survives impact with obstacles, such as rocks, a maximum of 10 cm in size.[13]
   d. Has a landing accuracy < 1 Km, 1σ.
7. Support a camera system that is capable of taking 360 degree stereoscopic images. The camera height shall be approximately 1.8 m from the lunar surface.
8. To the fullest extent possible test the landing hardware and software proposed for the human missions to come, in order to help expedite those missions more quickly and safely.
9. Have the capability to support future payloads as mass/power availability allows as practical:
   a. Dust characterization instruments
   b. Neutron spectrometer
   c. X-ray spectrometer

D. Mission Planning

Launch Vehicle: Minotaur V[14] [= 464 kg to TLI]

Trajectory: Hohmann[15] (~5 day)

Mission Duration: 12 days (1+ years at polar sites)

Design Constraint: Designed such that it could potentially be launched on a Falcon 1

Descent: In close accordance with NASA-JSC descent algorithm being developed for human landers[16] [e.g. Drop-off altitude 2.4km, time of flight 84s].

The spacecraft under consideration is "Lander 2" option in Table 1. Table 2 provides the stack mass breakdown.

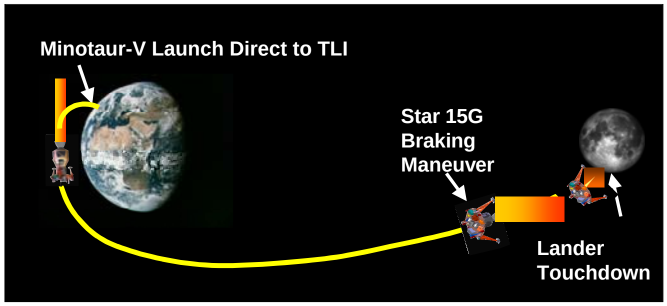

Figure 2 Trajectory Overview

Table 2
STACK MASS BUDGET

Description	Mass	ΔV
Lander at Launch	**85.6**	**1023**
Total lander propellant at launch	25.7	
Total landed mass minus prop inert mass	49.0	
Payload Landed	**59.8**	838
Landing Fuel	20.4	
Payload w/o Star 15G	**80.2**	
Interstage masses	2.57	
Mass Star 15G Inert	13.4	
Payload to Moon	**96.1**	**2113**[17]
Star 15G Fuel	110.0	
Payload after separation from the Minotaur-V with correction burn	**207**	64[18]
Correction Burn	5.35	
Payload in TLI	**212**	
Payload Adaptor and separation structure[19]	21.6	
Payload needed into TLI	234	
Excess launch capacity	230[20]	
Minotaur-V to TLI	**464**	

Minotaur V Launch Vehicle

Due to its availability, reasonable cost compared to most launch vehicles of this class and TLI mass capability, the Minotaur V has been selected as the baseline launch vehicle platform for the MLL. The Minotaur V combines elements of government-furnished decommissioned Peacekeeper boosters with technologies from proven Pegasus®, Taurus® and Minotaur launch vehicles. The vehicle consists of three Peacekeeper solid rocket stages, a commercial Star 48 fourth stage motor and subsystems derived from established space launch boosters. The fifth TLI stage is a Star 37 housed in a structure that includes an integrated SAAB 98.6 cm diameter Marmon clamp band style separation system.

Excursion 1: Falcon-I (Table 1: Lander 1)

For flexibility of launch the mission requirements were set to include that the spacecraft be designed such that it is possible, at least with only relatively small adjustments, to be launched on a Falcon-I. Since the Falcon-I has a lower payload delivery capability than the Minotaur-V, the spacecraft mass is constrained by the Falcon-I: analysis was initially performed to arrange the stack in the Falcon-I to maximize payload on to the lunar surface. This was used to calibrate the Minotaur-V stack to ensure that only relatively minor changes could be made to the design in order to launch on the Falcon-1.

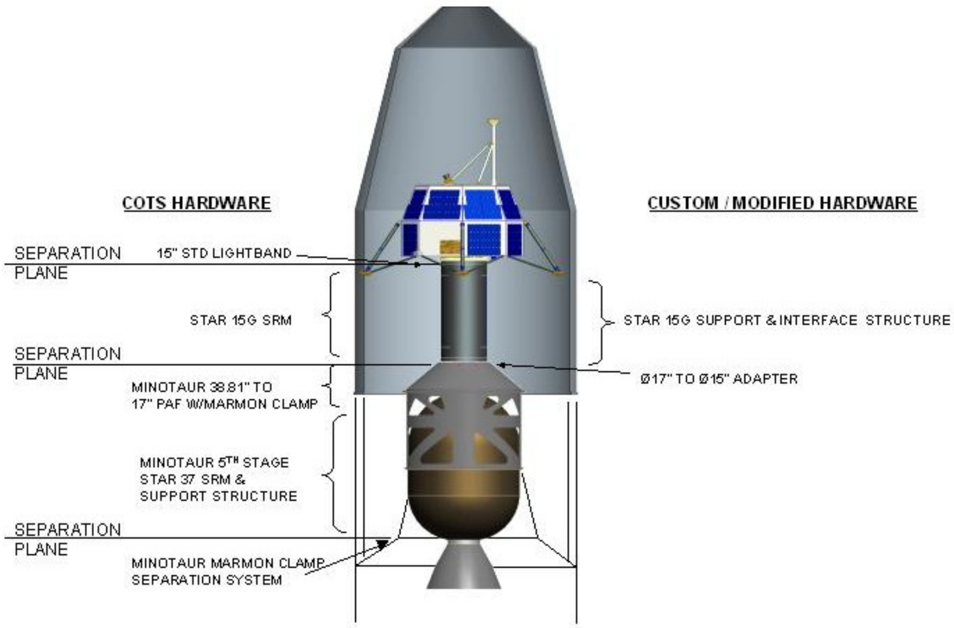

Figure 2 Micro Lunar Lander Stack in the Minotaur-V fairing

This design is different from the Minotaur-V stack in several ways:

1. The Minotaur-V delivers to TLI, whereas the Falcon-1 delivers to LEO so the post booster separation stack needs to have a TLI burn (in this case using a Star 30BP)
2. A weak stability boundary trajectory is used (rather than the Hohmann for the Minotaur-V) since there is a much tighter launch mass constraint
3. The Falcon-1 payload fairing is less wide and as such the legs of the lander design have to be able to retract for launch on this vehicle.
4. Removal of 2 propellant tanks.

Result: if this would be the same spacecraft design as discussed in the rest of this document then this stack would result in payload of just 7.2 kg. This is with no system reserve so in practice is approximately half the payload. Further analysis is needed in two areas which has yet to be completed which could enable a greater payload to be launched on the Falcon-1: (1) interstage masses for this stack; and (2) the minimum frequency of the payload.

In addition, changes to the baseline design that may allow greater payload on a Falcon-1 with but slightly reduced capability would include: (1) reducing the descent fuel mass (thus reducing the capability of testing the human landing algorithms); (2) reduction in the battery mass (causing a reduced operation time, perhaps a few days); (3) reduced solar arrays (causing more minimalist power consumption by the payload, radio transponder etc); and (4) reduction in thermal control hardware. These options would reduce the spacecraft dry mass to 35.6 kg thus improving the payload mass to 5.4 kg (without system reserve).

Excursion 2: Full Minotaur-V (Table 1: Lander 3)

Just for scale, if one were to use the total payload capacity of the Minotaur-V it would enable a lander approximately 170 kg in mass at launch (wet), 135 kg landed (dry) and a payload in the range of 70 kg to the lunar surface. This spacecraft would require some significant re-design in structure and elsewhere and so is not easily re-adaptable to the Falcon-1.

E. Spacecraft Subsystem Overview

The Micro Lunar Lander design is shown in Figure 4. The spacecraft is modular in design. It consists of a Common Bus Subsystem and a Propulsion Subsystem. The Propulsion Subsystem is reconfigurable to hold either 2 or 4 Tanks for additional payload capacity.

Table 3
SPACECRAFT MASS BREACKDOWN

Key Components	Mass
Structure – ARC	**13.6**
- Four lander legs with carbon rod strut design	3.0
- Hybrid Spaceframe & sandwich panel construction	5.5
Propulsion	**9.0**
- Descent Main Engine	4.0
- Propellant and Pressurant Tanks	5.0
Electric Power	**6.0**
- Batteries – Secondary Li-Ion 130 Whr	3.5
- Fixed Solar Arrays with 65° orientation from horizontal- 1.6 m^2	2.5
Flight Control	**4.5**
- Flight Software - ARC	0
- IMU – Honeywell LN-200S	0.8
- Startracker – Aero-Astro	0.6
- Radar Altimeter – Honeywell HG8500	1.4
- DSMAC – Raytheon	1.8
- Avionics – Broadreach Integrated Avionics Unit	
Command and Data Handling	**5.0**
- RAD750 PCI – BAE Systems	0.3
Telecommunications	**1.6**
- Patch Antenna (omni S-band)	0.6
- Transponder – Aero-Astro	0.6
Other	**9.9**
- Harness	2.4
- Thermal	3.0
- Reserve (10% spacecraft mass)	4.8
Payload	**10.0**
- Dust characterization (distribution and dynamic behavior) instrument	
- Stereographic Camera	
- Neutron Spectrometer	
- LIDAR	
- High gain antenna (to stream descent for public outreach)	

F. Structure

The space frame structure is composed of tubes, fittings, and honeycomb panels. Mass and capability of in house fabrication were considered critical, although integration, cost, ease of assembly, etc. are also considered. The structure must handle launch loads and provide attenuation of impact loads. Critical to the mission is its ability to be a stable platform on the moon that does not topple during landing. A series of trade studies were performed, which are summarized in Table 4, but which are not expanded upon in this paper.

Following several design iterations the design finalized was a configurable modular spacecraft bus. In this way it would be suitable for a variety of missions, such as: the Lander 3 concept in Table 1, a lunar communications orbiter or an "X-Nav" navigation payload. For the design in this paper the following modules were used: propulsion module, legs module, main spacecraft module and extension module.

Table 4
SUMMARY OF STRUCTURE TRADES

Trade Description	**Results**
Spacecraft launch orientation	Legs down ("Live bug")
Two Tanks vs. Four Tanks	Four tanks
Number of Lander legs (3 vs. 4)	Four legs
Leg Construction (Struts vs. Beams)	Struts
Structure Type (Space frame Truss structure vs. Sandwich Panels)	Hybrid
Materials for truss structure (Composite vs. Metal)	Carbon tube/ Al fittings
Joint Design Trade (Tabs vs. Ball vs. One Bolt vs. Weldments vs. Slip Joint Node Fittings vs. Monolithic Machined Joint	Down-selected to Tabs & Ball
Equipment Layout (two independent modular primary subsystems of Propulsion and common bus vs. integrated)	Independent modular
Fixed vs. retractable legs	Fixed

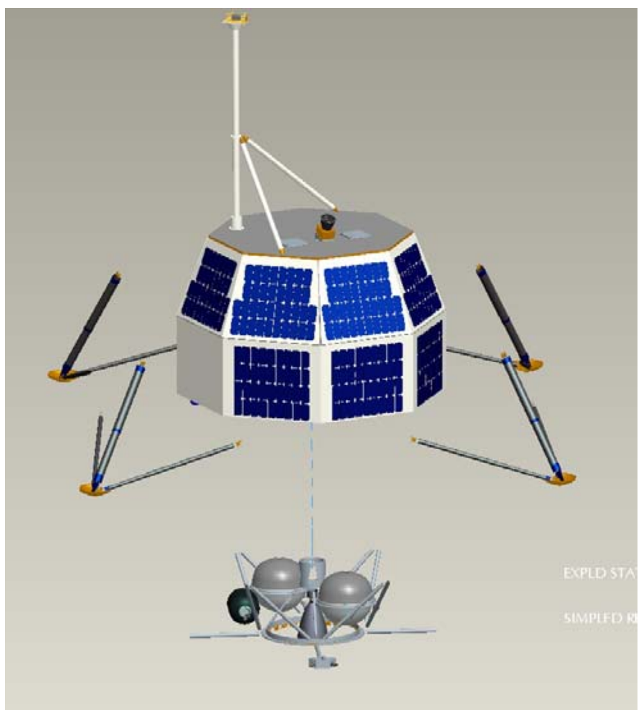

Figure 4 Exploded Diagram of the Spacecraft Design

Load Analysis

ProMechanica models were made using beam and shell elements. Non-structural masses representing all components were applied as point masses at appropriate nodes. The models were constrained at the light band for the launch and braking load cases, at the thruster for the thruster case, and at the appropriate foot for the landing cases. Three landing cases were analyzed with the FEM: impact on one foot, two feet, and all feet.

Specific load cases were evaluated for: Launch on a Minotaur of 7 g vertical, 3 g lateral, a braking burn of 15 g and landing loads 10 g (impact on 1, 2, or 4 legs)

The landing loads were considered the worst-case load condition as one must take into consideration that one leg may make contact before the other legs taking the full landing load. The results of these loads were used for the space frame tube analysis below.

Tip-over Analysis

The tip over analysis is critical to the design of the lander and is considered one of the primary design drivers. The following chart shows the maximum allowable center of gravity (CG) for a 55 kg lander when taking into account conservation of momentum at landing and conservation of energy at post impact. If a three leg lander were used, the maximum allowable CG is 25 cm for surviving a landing with a 15 degree slope, which is not achievable when coupled with a requirement to clear obstacles a minimum of 10 cm in height. The current baseline target is to design the vehicle to have a CG not greater than 40 cm.

If the final design is unable to attain a CG of less than 40 cm, analysis does show that if the Guidance navigation and control can limit the horizontal velocity to 0.5 m/s the CG could be raised to 48 cm with 4 legs. Figure 5 shows how the maximum CG varies with horizontal velocity and leg circle diameter.

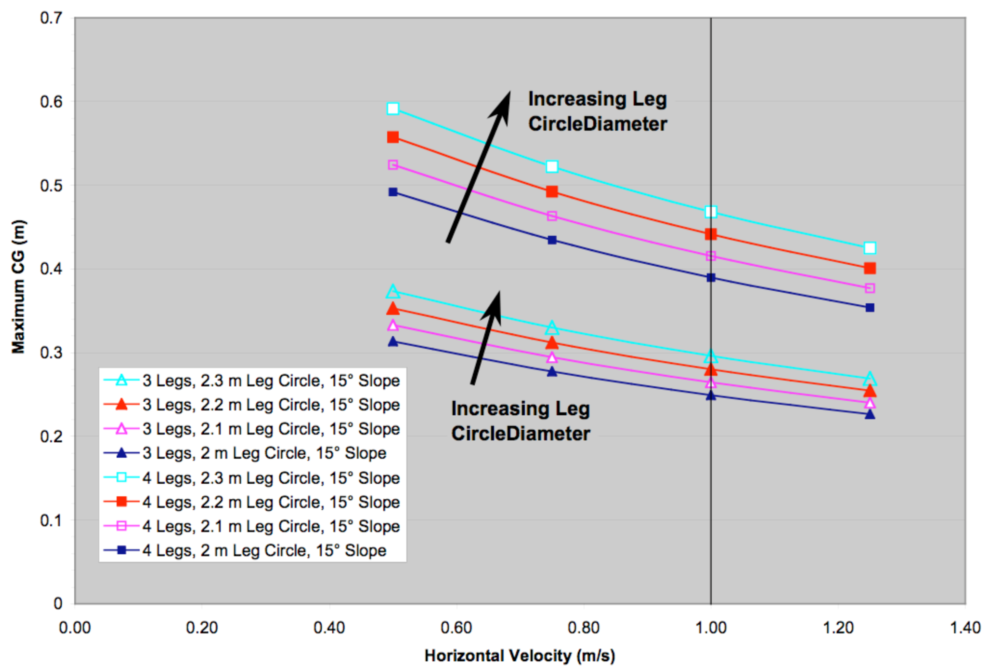

Figure 5 Tip Over Analysis: Maximum CG as a function of Horizontal Velocity and Leg Circle Diameter

G. Propulsion

The propulsion system design consists of a descent thruster, six attitude control thrusters, one fuel tank, one oxidizer tank, a pressurant tank, an ordnance valve driver card, and associated tubing and cabling. Four ACS thrusters are arranged in a bow-tie configuration for attitude control. Two ACS thrusters are oriented vertically so as to provide translational ΔV for the Trajectory Correction Maneuvers (TCMs) (the main descent thruster is shielded by the braking motor).

The pressure fed propulsion system is a modular system and consists of blow-down tanks with interchangeable engines. The engines incorporated in the lander design are kinetic kill vehicle engines: light weight pulsed modular thrust systems developed for missile defense. The four tank propulsion system can be reconfigured with two tanks.

Propulsion System Mass Wet: 21 kg

Usable Propellant Mass: 13 kg

This propulsion system is designed to provide the ΔV = 728 m/s required for the combination of ACS, TCMs and descent onto the lunar surface.

Descent Thruster Specifications[21]

Maximum Thrust (T_{max}): 3200 ± 500 N

Specific Impulse: 292 ± 10 s[22]

ACS Thruster Specifications

Maximum Thrust (T_{max}): 30 ± 10 N

Specific Impulse: 266 ± 10 s

The propulsion system above is well able to provide the thrust magnitude and modularity required for the vehicle mass with a low dry mass, is physically small enough and provides high enough specific impulse to have fuel use consistent with the mass constraints.

H. Thermal

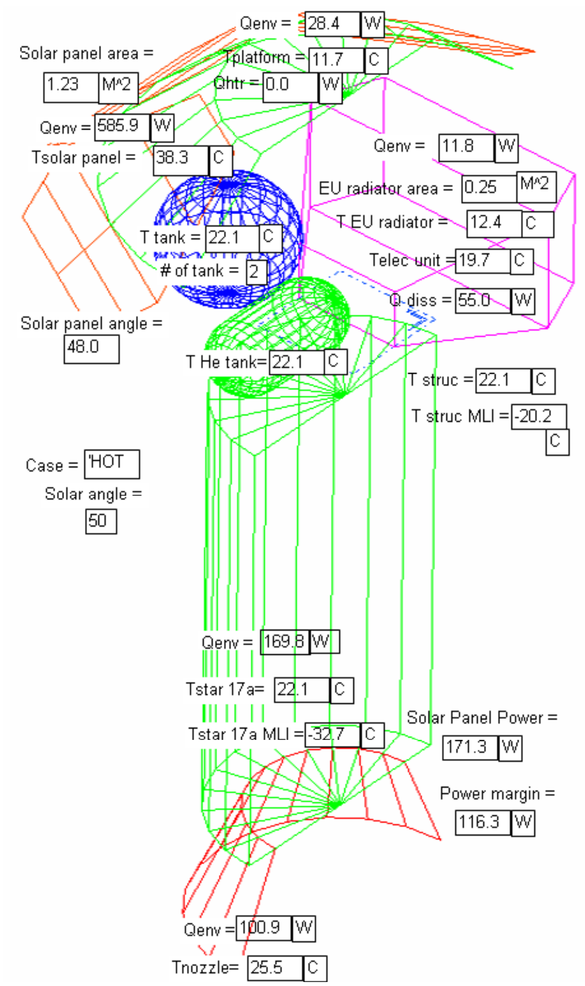

Figure 6 Thermal Analysis for case 1 'worst case hot cruise'

Simple system level thermal models were constructed to simulate the cruise and lunar surface operations. The thermal design has to accommodate both cruise and lunar surface operation, since optimizing for one condition may have adverse affect on the other operation. The assumptions were that the:

1. Lander lands on day side of equator
2. Battery is mounted on the with heat exchanger
3. Lander rotational speed is 1 rpm (minimum) during cruise
4. Total electrical load is 132 watts on lunar surface & 42 W during cruise
5. Top platform is painted white & Multi-Layer Insulation (MLI) on the reverse side
6. Electronic Unit box exterior is painted black to warm up the space frame, tanks & other components
7. MLI E* varies between 0.01 to 0.04 depending on the worst case loading
8. Lander is oriented such that the radiator panel points toward north.
9. Space frames and back of panels are covered with MLI.
10. Rocket external surface is covered with MLI.
11. Oxidizer (NTO) freeze temperature is 230 K.
12. Fuel (MMH) freeze temperature is 260 K.
13. C&DH maximum allowable temperature is 358 K.
14. Battery max. storage temperature is 358 K.
15. Battery max. operating temperature is 303 K.
16. IMU max. operating temperature is 357 K.
17. Assumed additional TPS for rocket exhaust plume.
18. Dust on solar panels & radiator are negligible after lunar landing.

Results & Findings

A thermal analysis was completed for four cases of: (1) worst case hot for cruise; (2) worst case cold for cruise; (3) worst case hot for surface operations; and (4) worst case cold for surface operations. The structural and thermal design were adapted to ensure that all components and subsystems stayed within their necessary operational range. Some design modifications were made to achieve this, in particular, a solar panel was replaced with a radiator, a heat pipe and miler-gold blankets were added. The results were:

1. The worst case hot cruise (50 deg. Solar angle) component temperatures varies from 11 to 38 C
2. The worst case cold cruise (50 deg. Solar angle) component temperatures varies from -1 to 15 C
3. The worst case hot lunar surface noon operation component temperatures varies from 67 to 89 C
4. The worst case cold lunar surface operation (70 degree) component temperatures varies from 24 to 45 C
5. The cruise component temperature bandwidth is -1 to 38 C
6. Lunar surface operation component temperature bandwidth is 24 to 89 C

The design meets all the temperature requirements during cruise and lunar surface operation. It was found that limiting the cruise solar angle and lunar surface solar angle are crucial in meeting the temperature requirements and that the view to the lunar surface should be minimized.

Table 5

SPACECRAFT POWER BUDGET

		Duty Cycle %					Average Power (W)				
		Launch	Cruise	EDL	Lander Checkout	Surface Ops	Launch	Cruise	EDL	Lander Checkout	Surface Ops
C&DH											
RAD750	10.200	55	55	100	100	100	5.610	5.610	10.200	10.200	10.200
MOAB	5.500	100	100	100	100	100	5.500	5.500	5.500	5.500	5.500
DSMAC I/F	9.000	1	1	100	0	0	0.090	0.090	9.000	0.000	0.000
SACI	7.940	100	100	100	100	100	5.240	5.240	7.940	3.140	3.140
PAPI	8.850	100	100	100	100	100	0.436	0.519	3.687	2.460	3.900
IMU	15.000	100	100	100	0	0	15.000	15.000	15.000	0.000	0.000
S-Band (Rx)	1.000	100	100	100	100	100	1.000	1.000	1.000	1.000	1.000
S-Band (Tx)	32.000	1	2	2	10	100	0.320	0.640	0.640	3.200	32.000
Descent Cam	48.000	1	1	100	0	0	0.480	0.480	48.000	0.000	0.000
Propulsion	0.000	0	0	0	0	0	0.000	0.000	0.000	0.000	0.000
RF Altimeter	16.000	1	1	100	0	0	0.160	0.160	16.000	0.000	0.000
Thermal Control (H/C)	15.000	25	50	50	100	100	3.750	7.500	7.500	15.000	15.000
Solar Array Articulation	5.000	0	0	0	0	0	0.000	0.000	0.000	0.000	0.000
Payload Instruments	60.000	5	1	1	50	50	3.000	0.600	0.600	30.000	30.000
Total							**40.586**	**42.339**	**125.067**	**70.500**	**100.740**

I. Power

The electrical power subsystem is designed to generate, store, convert, and distribute electrical energy to other spacecraft subsystems. Table 5 lists the power requirements for each subsystem/component. Mission cycles highlighted in yellow are powered by batteries, green by solar cells with peak loads supplemented by batteries. Surface operations shown are for lunar days only as the power system is not designed to provide heating to survive the lunar night. Power numbers shown allocate 60 W for experiment payloads operating at a 50% duty cycle. The power requirements are 56 watts during cruise and 132 watts on the lunar surface, with 30 W included as margin. The surface operation includes 60 W for payload instruments.

J. Descent (GN&C)

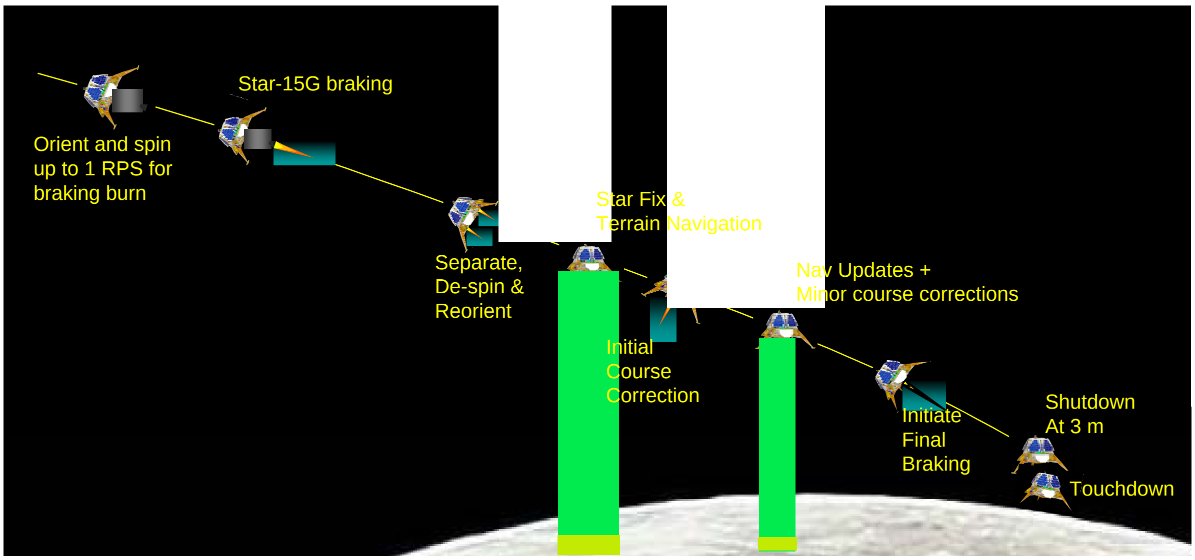

Figure 7 Basic Descent Sequence

The Guidance, Navigation and Control (GN&C) of the spacecraft during its final descent phase represents a challenge due to precision landing requirements. Guidance laws ensure minimal fuel usage and feedback control implements the guidance commands while maintaining system constraints. The challenges stems from the limitations of the propulsive units. The main thrusters and reaction attitude control systems are bang-bang in nature.

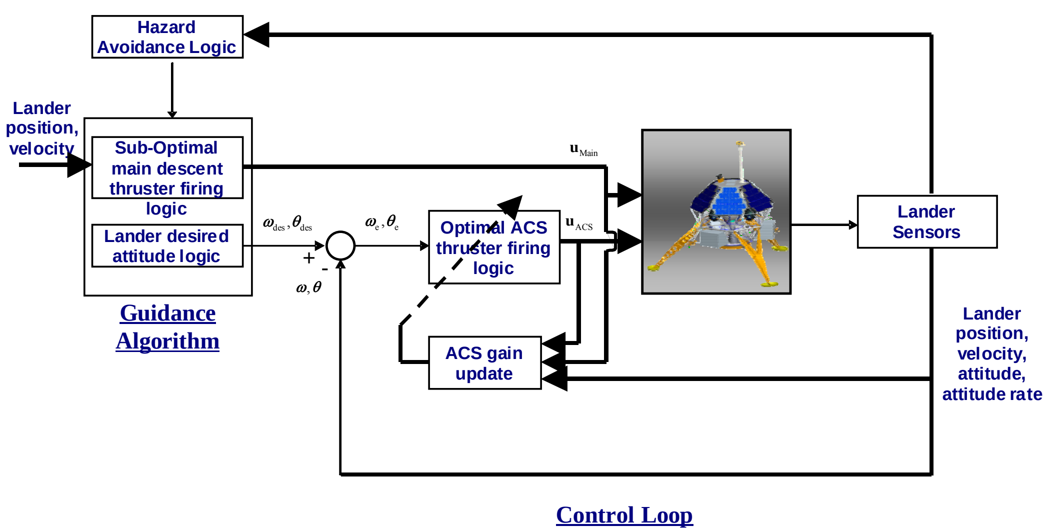

Figure 8 GN&C Control Loop

The following technologies were developed:

- A 6-dof (degree of freedom) simulation model with assumed mass distribution, inertia characteristics, main and ACS thruster characteristics.
- A design of a conservative descent velocity guidance that ensures an altitude-based velocity profile.
- A phase-plane logic based ACS attitude control for precise pointing.

A Monte-Carlo simulation for assumed 3-sigma variations in descent altitude, velocity, and position errors was conducted to document the results. The resulting trajectories are shown in Figure 9. The Micro-Lander Guidance and Control basic scenario and architecture is depicted in Figure 8.

Descent Guidance

Descent guidance is based on an altitude-velocity constraint depicted in Figure 10. This is a conservative descent velocity guidance that ensures a safe altitude-based velocity profile.

The guidance logic is as follows:

- Main Thruster is <u>turned on</u> if descent velocity magnitude is larger than the green line magnitude for the corresponding altitude.
- Main Thruster is <u>turned off</u> if descent velocity magnitude is smaller than the red line magnitude for the corresponding altitude.

The total fuel use for all runs is shown statistically in Figure 11.

Lunar Lander Control and Sensor Assumptions

The attitude control for the descent presumes two key sensor capabilities: (1) an LN200 Star tracker/IMU with three angular rates and three accelerations; and (2) a radar altimeter to provide the height above the terrain.

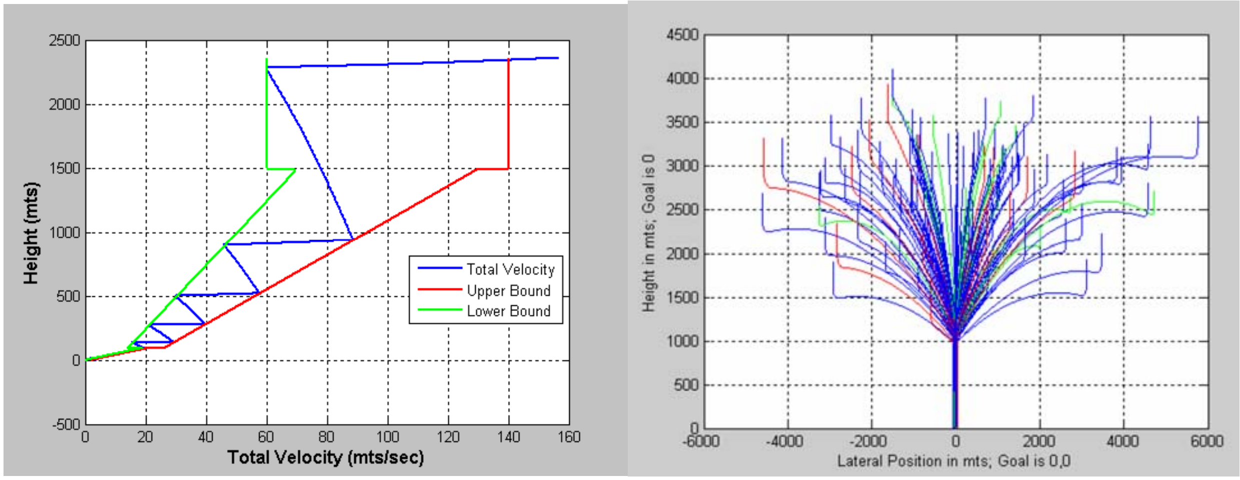

Figure 9: Monte Carlo Analysis of Ames Conservative Descent Guidance

Top Level Hardware Functional Requirements:

The Avionics Unit hardware system should shall provide the following functions:

- Execute all flight software in order to command onboard hardware systems such as Power Control and Distribution, Propulsion Systems, Payload, as well as handling receiving and transmitting of data.

- Operate properly in mission environments. This shall include limited operation during Lunar Night and a radiation environment consistent with lunar environment.

- Be able to support all interface needs stated below. Components should be COTS wherever possible and should be selected for price and mass.

The primary interfaces include:

- Flight software.
- Actuators, pyros, and propulsion system.
- Electrical Power System (EPS)..
- Telecommunications Hardware.
- GN&C sensors:
  - IMU, Star Tracker, Sun Sensor(s), DSMAC
  - Sensors for monitoring system health.
- Payload interfaces:
  - Camera(s)

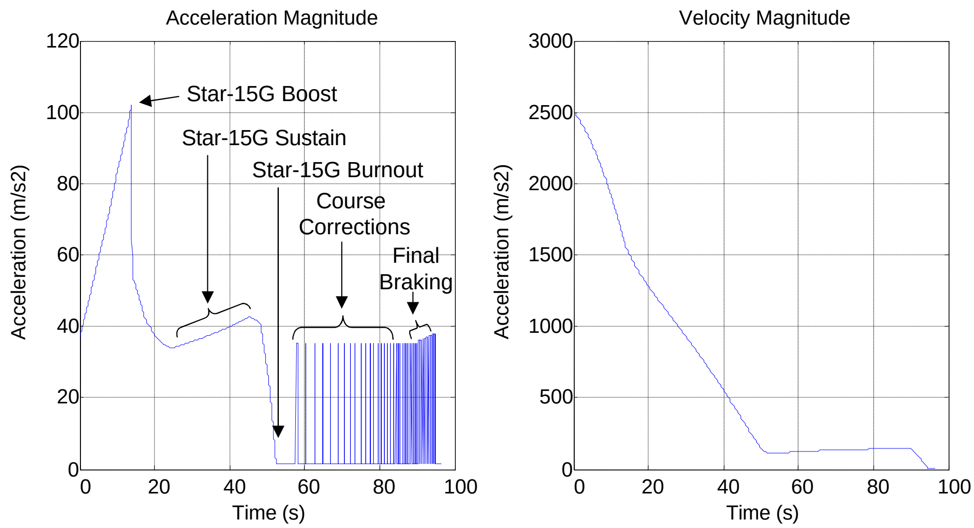

Figure 10: Acceleration and Velocity Magnitude over Time

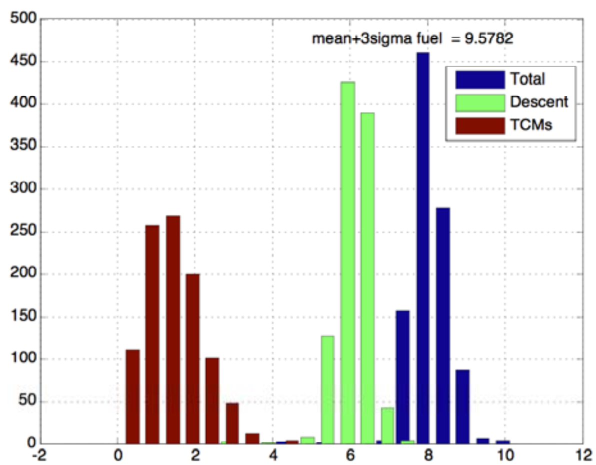

Figure 11: Fuel Usage using Ames Descent C&DH

The current Avionics Unit is planned for a 8 Slot Compact PCI Chassis Current boards include: (1) RAD750 Processor with 128 Mbytes of Synchronous Dynamic Random Access Memory (SDRAM) and 256 kbytes of Stand Up Read-Only Memory (SUROM); (2) MOAB Board; (3) Solar Array Control Integration (SACI) board; and (4) one Power-switching and Pyro Integration (PAPI) board. One slot will be reserved for DSMAC Board (as payload). The 8 Slot chassis is capable of holding 5 Command and Data Handling (C&DH) boards and 3 Power boards. The SACI board fits at the back of the chassis and does not occupy a slot.

Top Level Software Functional Requirements

In order to accomplish low cost, rapidly deployable missions, it will be necessary to utilize model based, auto-code generation techniques for developing & testing software. These techniques were successfully used by Octant Technologies for various AFRL flight projects (eg. XSS-10, XSS-11).

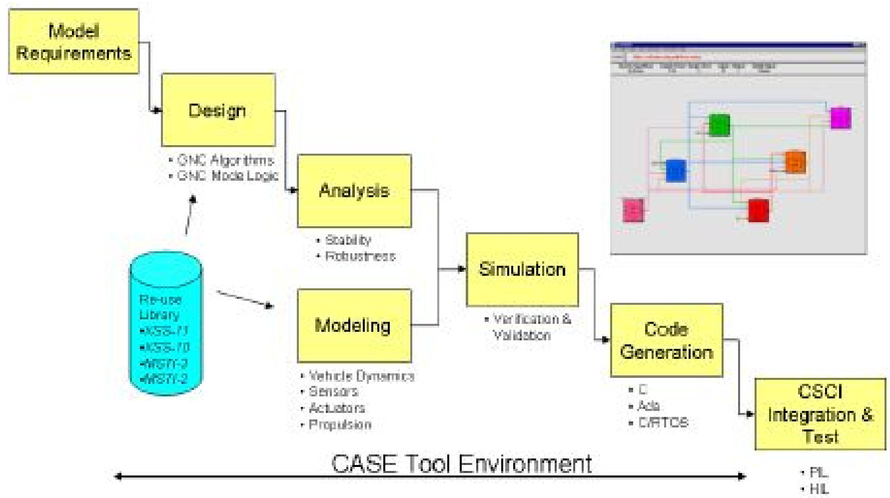

Figure 12: Model Based Flight Software Development

The onboard software system shall include: (1) a Command and Data Handling (C&DH) function, including command processing, telemetry processing, and image processing; (2) Vehicle Management System (VMS) functionality, including Electrical Power System (EPS) management, Thermal Management, Propulsion Management, Payload Monitoring, Fault Management, and Mode Control Management; (3) Guidance, Navigation, and Control (GN&C) functionality (including the following modes of operations: idle; rate capture; sun acquisition; stellar Acquisition; trans Lunar Injection (TLI); cruise; trajectory Correction Maneuver (TCM); brake; precision landing; and safe mode(s)); and (4) an Executive which will provide Task Scheduling, Interrupt Handlers, and Interface Software.

The Ground Software shall include: (1) a user interface for allowing the transmission of commands to the spacecraft. Command scripting should also be available that will be based upon state or time conditionals. Upon receipt of commands, the spacecraft should be able to execute the commands either immediately, or stored for later execution; (2) user interfaces for monitoring the spacecraft. Displays should be provided for tracking current readings, trending, and providing alarms when sensor readings are out of bounds. Displays should also be provided that show visualizations of the spacecraft and camera data; (3) means for archiving and playing back telemetry.

The primary interfaces will be to the current C&DH Avionics Hardware. This will need to control GN&C as well as other hardware systems, and to receive and transmit data. The primary interfaces include:

Avionics hardware that will run the flight software: send signals to actuators, pyros, and the propulsion system; control the power generation and utilization; transmit telemetry, and receiving commands; monitor GN&C sensor data: IMU, Star Tracker, DSMAC; and receive data from sensors for monitoring system health.

Avionics hardware for the Payload interfaces include: camera and any other scientific instruments; mission operations hardware (Telemetry decryption/ decommutation, data storage/servers, displays); and mission operations personnel/facility for commanding and monitoring the spacecraft.

Figure 12 shows the model based development approach to flight software that was used by Octant on the XSS-11 spacecraft. Note that in contrast to a traditional software development approach (Software Requirements->Software Design->Software Coding->Integration & Test), this approach starts with the development of models, and uses CASE Tools (i.e. MatrixX/SystemBuild) for designing and developing algorithms to control the models. The models & algorithms are subsequently tested in simulation. Once these are working to satisfaction, then the Code is automatically generated and ready for integration. A traditional approach for developing flight software will be used for some modules. In general GN&C will be accomplished using a model based approach, where C&DH and Vehicle Management will be accomplished using a more traditional approach.

K. Telecommunications

The communication subsystem is built around an AeroAstro modular radio, which consists of three 115 cm^3 modules. The computer interface is RS-422 serial or USB. Each module has a mass of less than 0.2 kg, which leaves approximately 1 kg for the antenna, coaxial cables, and interconnecting wiring. Using a standard product reduces cost, schedule, and technical risks while providing the needed communications capability.

Transponder Function

Measuring the distance from the earth station to the lander during flyout is done using a full-duplex coherent carrier detection system. In this mode, the received carrier is used to derive the transmitter frequency, which is 240/221 of the received carrier frequency. The AeroAstro product is compliant with the Space Ground Link System (SGLS) requirements. The transmitter output power is adjustable from 0.5 watts to 5 watts. The transmitter BPSK rate is 20 Mbps, is more than adequate.

Antenna

A single antenna meets the communication needs for both the flyout and after landing. The concept is to mount a pair of crossed dipoles on the top of the mast above the camera. This type of antenna is circular polarized directly above the lander, and horizontally polarized at the horizon. Linear polarization reduces the link margin by 3 dB, and is taken into account by referencing the antenna gain to circular polarization. The antenna is made on a light dielectric substrate with integral matching and phasing circuitry as shown below. The maximum gain is straight up, and drops off toward the horizon. The DSN antenna is circular polarized, so the gain drops by 3 dB at the horizon due to polarization alone. Horizontally polarized signals are also less susceptible to multipath than vertically polarized signals, which is beneficial for polar landings where the earth is near the horizon. A single antenna supports flyout, and polar and central landings. The estimated antenna mass is 0.1 kg.

As shown in Figure 13, the radiation pattern has a maximum gain of +3 dBic is directed normal to the antenna surface, and drops to –4 dBic as the angle approaches the horizon (assuming the antenna is level). The pattern falloff has two contributing factors. First, the pattern is more hemispherical in the elevation plane and omnidirectional in the horizontal plane, and second, the polarization goes from circular to linear horizontal. The receiving antenna is circular, which drops the effective gain by 3 dB. The horizontal plane gain is shown below. The pattern is nearly omnidirectional and is effectively horizontally polarized at this elevation angle.

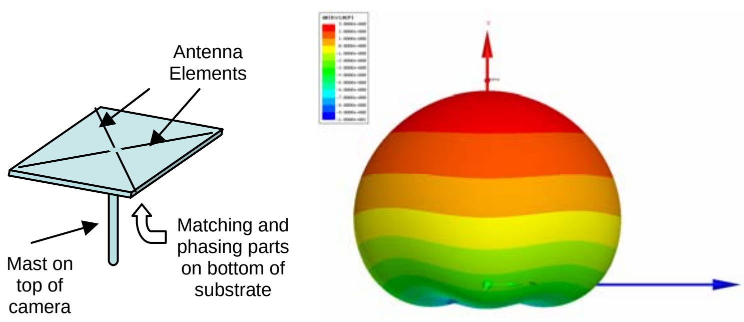

Figure 13: Antenna Design and Antenna Gain vs. Direction

When the earth is near the horizon, there is a possibility that the primary signal from the lander antenna to the DSN antenna will combine with waves that are reflected off of the lunar surface in such a way that the signals cancel each other. This effect, known as multipath, occurs when the elevation angle is low and the antenna is near a relatively flat surface. The lander's antenna is approximately two meters above the surrounding terrain. The strength of the reflected waves is a function of the surface conductivity at the operating frequency, which is around 2 GHz. An initial literature search on lunar soil conductivity, the lunar soil is not significantly conductive in the absence of water and below 200°C, and the surface attenuates RF signals. The penetration depth is approximately 100 meters at the lander's operating frequency. This suggests that multipath will not be an issue because of the poor lunar surface conductivity. Horizontally polarized signals are less susceptible to multipath effects than are vertically polarized signals, but the subject of multipath warrants additional study to assure that multipath effects will not be an issue for missions where the earth is within a few degrees of the lunar horizon.

Bit Rate

The worst case link margin is where the lander is on the lunar surface. The link margin analysis uses parameter values from the Deep Space Mission Systems (DSMS) Telecommunications Link Design Handbook. The analysis is based on a lunar surface temperature of 140°C, and a transmitter RF output power of 5 watts. The thermal noise of the lunar surface is 10 times greater than the receiver noise, which degrades the overall performance compared to a spacecraft against a cold cosmic background. Based on the margin calculation, the relationship between antenna gain and bit rate is shown in Figure 13. A 0 dBic gain antenna provides a system throughput of 50 kbps from the lunar surface. The data rate is from 20 kbps to 100 kbps as the antenna gain goes from -4 dBic to +3 dBic.

There are several trades that can be made to optimize the overall system performance. Higher gain antennas will improve throughput, but need to be pointed in the general direction of earth. Because the landing zone is known in advance, a higher gain antenna can be used with its elevation set to the approximate earth elevation angle and azimuth is known if the lander attitude is controlled on landing (if not, the rotating mast can be used to point in the correct azimuth). The command link margin is over 60 dB, which means that the lander will receive commands regardless of where the antenna is pointing. Another possibility is bore-sighting the antenna with the camera, and pointing the camera toward earth before transmitting imagery. This will result in the lowest electrical energy per transmitted bit, which eases thermal management and power issues, but following a trade analysis was not chosen for this design.

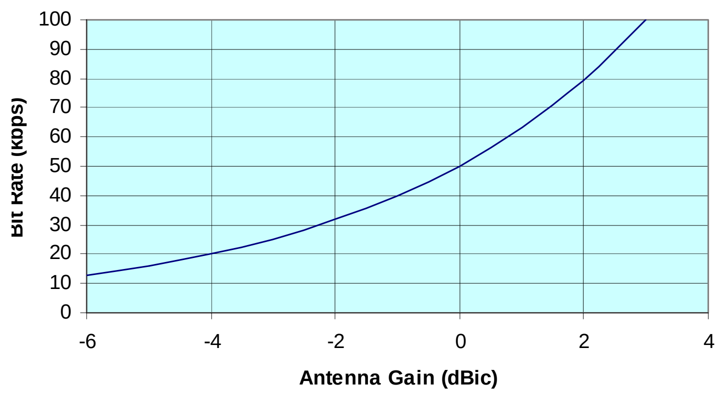

Figure 14: Bit Rate vs. Antenna Gain

L. Payload

The main purpose of this study was to confirm whether or not it is technically feasible to bring a useful payload mass to the lunar surface given the constraints applied, not to fulfill a particular scientific or technical mission. The payload mass herein suffices for the instruments needed for the large majority of the precursor lunar objectives outlined in *LRAS*. The following is an indication of the instruments that would be among those most likely to be considered:

1. Stereoscopic Camera
2. Dust Characterization Instrument
3. Neutron spectrometer (to measure the local Hydrogen content and ground truth orbital data)
4. Higher gain antenna (to stream descent imagery data for public outreach)
5. LIDAR and other potential descent hardware that might be considered for human landers (to verify performance)

M. Cost

The approximate cost breakdown is given in Table 6.

Table 6
COST BREAKDOWN

Cost Type	Assumptions	Cost
Personnel	30 FTEs for 2 yrs	18.2
Components	See Table 3	19.1
Payload	At $1M/kg	10
Launch	Minotaur-V	26
Reserve	20%	14.7
Total		**$88 M**

III. CONCLUSION

The micro lunar lander design project described herein concludes that it is technically feasible to land useful payloads (10-15kg) to the lunar surface using low mass spacecraft (86kg), very affordably (~$88M) and with a fast turn around (<36 months). These figures are for a first such mission: further ones would improve on this.

This case point gives some degree of confidence to make more general conclusions:

1. **Small spacecraft could do the same technical functions as the existing Missions 1-4 in the *LRAS* baseline at a fraction of the cost schedule of the missions planned therein.**
2. Each mission has a higher risk but the ability to do more frequent smaller missions makes the overall programmatic risk smaller.
3. Small missions will allow a faster learning cycle, which may significantly help cost and schedule risks with future larger mass missions. It could allow critical technology to be tested for the Constellation program prior to the final design being configured.
4. The shorter schedule could allow a phase of tele-robotic and ISRU missions prior to human landing which could enhance the capabilities and safety of human missions.
5. Such missions are readily able to meet some hey political objectives of the US Vision for Space Exploration and provide greater public outreach opportunities.

A small spacecraft lunar architecture could complete 14 of the 15 objectives layed out in *LRAS*. It would allow an increase number of flight opportunities which can take advantage of the latest technologies through shorter development cycles. The net result would be an evolutionary mission architecture.

ACKNOWLEDGEMENTS

This work is in essence a summary of the work of the entire lander design team centered at NASA-Ames including from NASA-Ames itself: Craig Baker, Richard Bornhorst, James Brown, James Bell, Joe Camisa, Sylvia Cox, Howard Cannon, Robert Dumais, Ian Fernandez, Ken Galal, Robert Hanel, James Head, Butler Hine, James Kennon, Kalmanje Krishnakumar, Lynette O-Leary, Lawence Lemke, Mark Mallinson, William Marshall, Owen Nishioka, Craig Pires, Steven Spremo, Mark Turner, and Alan Weston; and team members from Raytheon, Ronald Choo and Leonard Vance; and from Stellar Exploration Inc., Tomas Svitek. The project was based on a common bus design that was being developed in collaboration with Jim Watson and Joe Burt of NASA-Goddard Spaceflight Center (GSFC).

REFERENCES

[1] As conceived by NASA-Ames Small Spacecraft Office for the class of missions focused upon there.
[2] *Lunar Robotic Architecture Study*, NASA, May 2006, p. 60-63

[3] This includes 10% reserve. For scale, the RLEP2 mission was going to have a science package of 13 instruments totaling 84kg to the lunar surface and the LPRP current lander design, including a rover mass of 500kg.

[4] Cost estimates have not undergone extensive study. Only the 'Lander 1' (see Case 1 below) has had any cost analysis: the other estimates are extrapolated from this assuming, in particular, the use of relatively off-the-shelf instrumentation for payloads with a cost rate of $1M/kg for the first 20kg and $0.5M/kg thereafter. Costs include a 20% margin. Note that re-occurring costs are likely to be significantly less for duplicate missions.

[5] Assumes a 90 day Weak Stability Boundary trajectory. All other options assume a Hohmann trajectory. Note that the same spacecraft configuration is possible on a Minotaur-V, the only significant difference is a cost increase of approximately $20M.

[6] Note that this option does not use the maximum capacity of the Minotaur-V.

[7] Note that this option does not use the maximum capacity of the Falcon-I.

[8] Although carrying a rover capable of traversing 30km seems unrealistic at present for a small lander mission, the same functional results may be possible with a small satellite hopper: e.g. in Lander 2, if 15kg of propellant was taken as 'payload' then the lander could perform five 10km and five 1km hops to alternative sites.

[9] Ibid.

[10] *Lunar Surveyor Project Final Report,* NASA-JPL Technical Report 32-1265, 1969 and *Surveyor 1 Mission Report*, NASA-JPL Technical Report No. 32-1023, 31 August 1966.

[11] Ideally, the spacecraft would land with no horizontal velocity component. Surveyor missions reported values of 0.3 – 0.6 m/s. The real issue is what is realistically obtainable from the GN&C subsystem.

[12] Surveyors 1&6 landed on a slope of less than 1 degree, Surveyor 3, 12.5 degrees, Surveyor V, 19.5 degrees and Surveyor 7, 3 degrees.

[13] An obstacle size of 25 cm is desired but currently impractical for a lander of this size

[14] The Minotaur-V launch vehicle will serve a unique class of missions which current US launch vehicles cannot. They provide an ability to send 464 kg to TLI with an overall cost in the range $20-40M. No other current US launch vehicle can provide anything in the range 100-1000 kg to TLI in the cost range <$100M in a dedicated launch.

[15] Considerations for the trajectory type: (1) Fuel efficiency (alternatively put, the feasibility of putting a reasonable mass payload to the target); (2) Duration; and (3) How well the trajectory has been tested with previous missions. Hohmann trajectories allow efficient transfers in reasonable times and since the mission is not mass constrained on the Minotaur-V this allows a relaxation on the fuel efficiency.

[16] This is expected to be along similar lines to *First Lunar Outpost Powered Descent Design and Performance*, Engineering Directorate, Systems Engineering Division, August 1992, NASA JSC-25882

[17] Using STK Astrogator the velocity of impact on the lunar surface given a Hohmann transfer to the moon and if no breaking burn were performed would be 2520 m/s. This figure is this number subtracting both the target initial vertical velocity for the descent phase (60m/s).

[18]This is based on the dominant errors of the TLI injection motor, in particular, a total impulse error of 0.6% leading to an error of 92 m/s, with 20 m/s for other errors. This is more than the actual correction burns used on the Lunar Prospector mission (57m/s). It is anticipated that today a faster turn around on the position measurement of the spacecraft by DSN and on board is possible and therefore a faster implementation of the TCMs are possible, and thus lower ΔV required. The figure used in the Table comes from a worst case overall fuel use scenario when combining the TCM, ACS and descent fuel usage – which is the case where the TCM burn is less than this since that and the descent burn are anti-correlated.

[19] This is the minimal mass required to attach the spacecraft to the Minotaur-V faring. However, in reality this mass will depend upon whether a secondary is carried (in which case it could be more or less to adapt to the needs of the secondary) or could be a great deal more if no secondary is flown (in which case a dead mass of 254kg must be accounted for).

[20] Naturally, this mass could not be left unfilled else the spacecraft would receive much too create a ΔV causing a too high braking ΔV and thus reducing the mass to the moon. Thus, it could either be used through a very heavy payload adaptor or a secondary payload.