NASA/TM—2010-216218 CD–2006–03

COMPASS Final Report: Low Cost Robotic Lunar Lander

Melissa L. McGuire and Steven R. Oleson
Glenn Research Center, Cleveland, Ohio

December 2010

NASA STI Program . . . in Profile

Since its founding, NASA has been dedicated to the advancement of aeronautics and space science. The NASA Scientific and Technical Information (STI) program plays a key part in helping NASA maintain this important role.

The NASA STI Program operates under the auspices of the Agency Chief Information Officer. It collects, organizes, provides for archiving, and disseminates NASA's STI. The NASA STI program provides access to the NASA Aeronautics and Space Database and its public interface, the NASA Technical Reports Server, thus providing one of the largest collections of aeronautical and space science STI in the world. Results are published in both non-NASA channels and by NASA in the NASA STI Report Series, which includes the following report types:

- TECHNICAL PUBLICATION. Reports of completed research or a major significant phase of research that present the results of NASA programs and include extensive data or theoretical analysis. Includes compilations of significant scientific and technical data and information deemed to be of continuing reference value. NASA counterpart of peer-reviewed formal professional papers but has less stringent limitations on manuscript length and extent of graphic presentations.

- TECHNICAL MEMORANDUM. Scientific and technical findings that are preliminary or of specialized interest, e.g., quick release reports, working papers, and bibliographies that contain minimal annotation. Does not contain extensive analysis.

- CONTRACTOR REPORT. Scientific and technical findings by NASA-sponsored contractors and grantees.

- CONFERENCE PUBLICATION. Collected papers from scientific and technical conferences, symposia, seminars, or other meetings sponsored or cosponsored by NASA.

- SPECIAL PUBLICATION. Scientific, technical, or historical information from NASA programs, projects, and missions, often concerned with subjects having substantial public interest.

- TECHNICAL TRANSLATION. English-language translations of foreign scientific and technical material pertinent to NASA's mission.

Specialized services also include creating custom thesauri, building customized databases, organizing and publishing research results.

For more information about the NASA STI program, see the following:

- Access the NASA STI program home page at *http://www.sti.nasa.gov*

- E-mail your question via the Internet to *help@sti.nasa.gov*

- Fax your question to the NASA STI Help Desk at 443–757–5803

- Telephone the NASA STI Help Desk at 443–757–5802

- Write to:
  NASA Center for AeroSpace Information (CASI)
  7115 Standard Drive
  Hanover, MD 21076–1320

NASA/TM—2010-216218 CD–2006–03

COMPASS Final Report: Low Cost Robotic Lunar Lander

Melissa L. McGuire and Steven R. Oleson
Glenn Research Center, Cleveland, Ohio

National Aeronautics and
Space Administration

Glenn Research Center
Cleveland, Ohio 44135

December 2010

Acknowledgments

The authors wish to thank the dedicated efforts of the entire COMPASS team for always producing high quality designs.

Level of Review: This material has been technically reviewed by technical management.

Available from

NASA Center for Aerospace Information
7115 Standard Drive
Hanover, MD 21076–1320

National Technical Information Service
5301 Shawnee Road
Alexandria, VA 22312

Available electronically at http://gltrs.grc.nasa.gov

Contents

COMPASS Final Report: Low Cost Robotic Lunar Lander

Melissa L. McGuire and Steven R. Oleson
National Aeronautics and Space Administration
Glenn Research Center
Cleveland, Ohio 44135

1.0 Executive Summary

In the Fall of 2006 the NASA Headquarters (HQ) Lunar Architecture Team/Lunar Precursor and Robotic Program (LAT/LPRP) (Program) leader Tony Lavoie was interested in how much payload can an inexpensive chemical or Electric Propulsion (EP) system deliver to the Moon's surface. Subsequently management at the NASA Glenn Research Center (GRC) (Rob Jankovsky et al.), were contacted by the NASA Ames Research Center (ARC) and asked if the COMPASS team could model a small lunar lander mission given a low cost launch vehicle and its performance characteristics. The goal of this COllaborative Modeling for the Parametric Assessment of Space Systems (COMPASS) session was to use Total Low Cost as the objective function, and design a Robotic Lunar Lander to deliver an unspecified payload (greater than zero) to the lunar surface for the lowest cost.

The spacecraft designed as the baseline out of this study was a solar powered robotic lander, launched on a Minotaur V launch vehicle on a direct injection trajectory to the lunar surface. A Star 27 solid rocket motor does lunar capture and performs 88 percent of the descent burn. The Robotic Lunar Lander soft-lands using a hydrazine propulsion system to perform the last 10 percent of the landing maneuver, leaving the descent at a near zero, but not exactly zero, terminal velocity. This low-cost robotic lander delivers 10 kg of science payload instruments to the lunar surface.

Collected in Table 1.1 is the top-level summary of each of the subsystems in the Robotic Lunar Lander design.

TABLE 1.1—MISSION AND SPACECRAFT SUMMARY

Subsystem area	Details	Total mass with growth
Top Level Robotic Lunar Lander	One week, Sunlit, global access, 25 W for payload on lunar surface	462 kg
Mission and Operations	Chemical direct injection mission (a.k.a. Surveyor), Minotaur V to TLI and direct lunar injection, solid rocket motor for descent, hydrazine for landing	
Guidance, Navigation and Control (GN&C)	Avionics hardware located on top of lander for improved cooling: Star trackers, Sun sensors, flight control electronics and radar altimeter for landing	24 kg
Launch	Minotaur V Launch, 92 in. payload fairing	
Science	Science payload as yet undetermined. Used as a metric in the design.	15 kg
Power	125 W GaAs solar array (two-axis pointing) on landed vehicle, 100 W housekeeping, 25 W to science instruments	31 kg
Propulsion	Descent motor: Star 27 Solid Rocket (disposed before landing), hydrazine propulsion system for final landing	44 kg
Structures and Mechanisms	Mechanisms to separate Star 27, and point solar arrays. Structural designed for 12 g axial launch, 5g axial landing loads, AL 2090 material, hexagonal bus with three Fixed leg landing system using Doppler radar (antennas on feet)	48 kg
Communications	High gain antenna (HGA) on lander	8 kg
Command and Date Handling (C&DH)	General processor (high speed, 1394 B), harness, wiring	10 kg
Thermal	Fixed radiator panels, heat sinks, heaters, multilayer insulation (MLI) blankets	22 kg

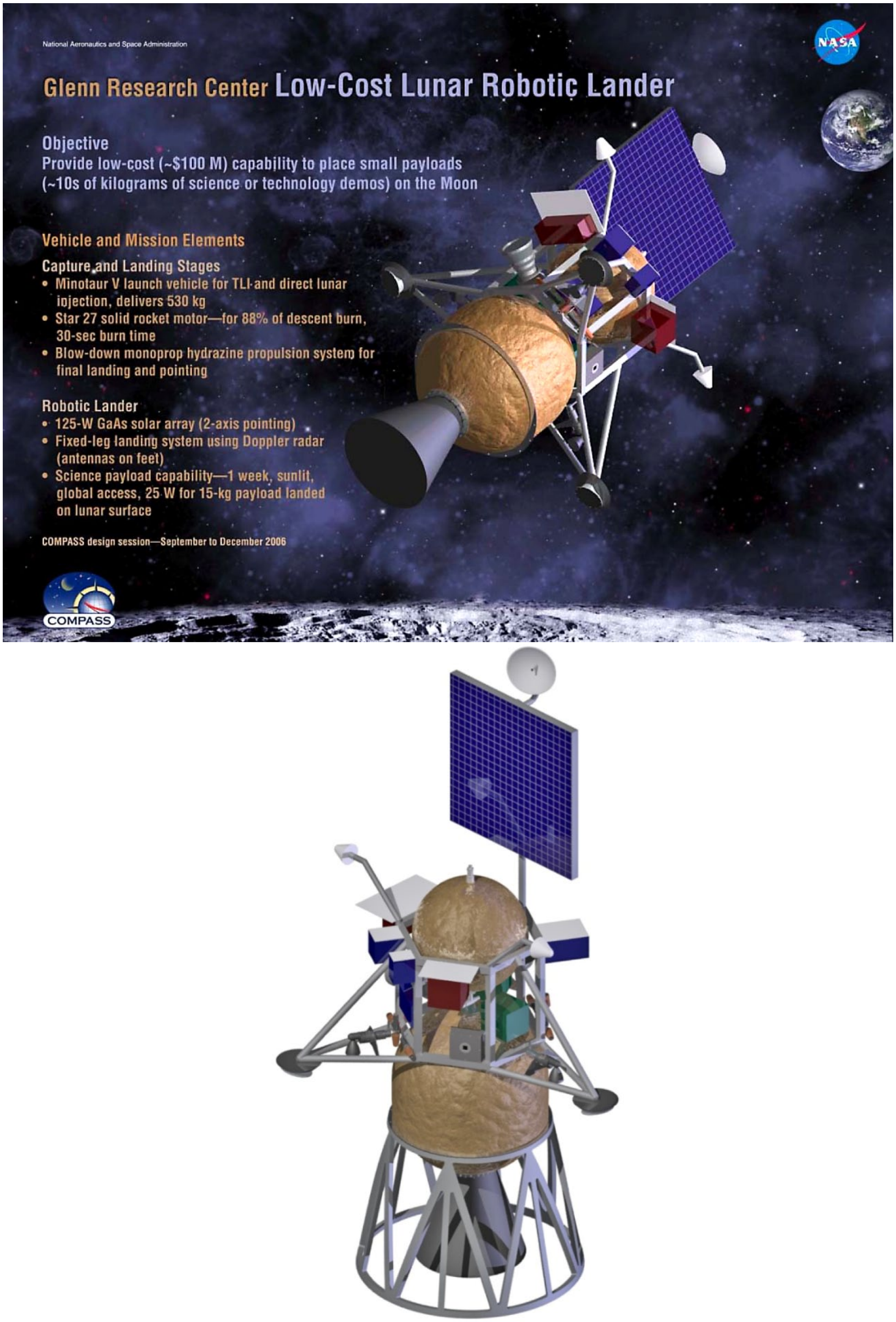


Figure 1.1—Robotic Lunar Lander shown on top of Star 27 solid rocket motor.

2.0 Study Background and Assumptions

2.1 Introduction

The goal for this study was to design the architecture necessary to deliver a low cost Robotic Lander to the Moon's surface using a small class launch vehicle. The overall study objective was to provide a low cost (~$100M) capability to place small payloads (10's of kilograms of science or tech demos) on the Moon. The science and power goals were as follows:

- One week mission duration on sunlit portion of the Moon (i.e., approximately on half of an average lunar day from "Sun up" to "Sun down" at a location on the surface of the Moon. This is dependent on choice of latitude/longitude of the landing site)
- Provide global access to the science payload, i.e., deliver science anywhere on the lunar surface

2.2 Assumptions and Approach

Starting with similar missions, the following missions were looked at as historic references for the COMPASS design session:

- Start with previous missions
  - Lunar
    - Surveyor (http://www.lpi.usra.edu/expmoon/surveyor/surveyor.html)
      - Landed ~33 kg of payload with 1960s technology off of the first Atlas Centaur vehicle
      - Lunar Prospector (http://lunar.arc.nasa.gov/)
      - Clementine (http://www.cmf.nrl.navy.mil/clementine/)
    - Mars
      - Viking (http://www.nasa.gov/mission_pages/viking/)
      - Pathfinder (http://marsprogram.jpl.nasa.gov/MPF/)
      - Mars Polar Lander (http://mars.jpl.nasa.gov/msp98/)
      - Mars Exploration Rover (MER) (http://marsrover.nasa.gov/home/index.html)

Starting points for the specific areas of the design are

- Power available for payload: 25 W
- Launch vehicle options to consider
  - Minotaur I, IV, V (http://www.orbital.com/SpaceLaunch/Minotaur/index.html)
  - Falcon 1
- Stages to design: TLI, Lunar capture and descent, Lander
- Propulsion Options
  - Chemical: Solids, etc.
  - Electric Propulsion
  - Combination of both

2.2.1 Survey Starting Points

The Surveyor spacecraft were used as starting points in this design. From the historical JPL website, the description of the Surveyor program reads as follows:

The Surveyor program consisted of seven unmanned lunar missions that were launched between May 1966 and January 1968. Five of these spacecrafts, Surveyor 1, 3, 5, 6, and 7 successfully soft-landed on the lunar surface. In addition to demonstrating the feasibility of lunar surface landings, the

Surveyor missions obtained lunar and cislunar photographs and both scientific and technological information needed for the Apollo manned landing program. Four spacecraft, Surveyor 1, 3, 5, and 6, returned data from selected mare sites from Apollo program support, and Surveyor 7 provided data from a contrasting rugged highland region.

For a sanity check in the bottoms up subsystem mass numbers, Table 2.1 calls out the main subsystems in the surveyor spacecraft and their relative magnitude of total dry mass.

TABLE 2.1—SURVEYOR SUBSYSTEM RELATIVE MASS DISTRIBUTION

Surveyor reference	Mass (kg)	Percent of dry
Flight control	32.8	8
Communications	38.7	9
Radar	15.4	4
Propulsion	99.7	24
Power	48.6	12
Mechanical/structural	142.8	35
Science	33.5	8
Dry total	411.5	

From the Surveyor Web site, the surveyor spacecraft are described as:

Each spacecraft weighed 1000 kg at launch, was 3.3 m high, and had a 4.5-m diameter. The tripod structure of aluminum tubing provided mounting surfaces for scientific and engineering equipment. Onboard equipment consisted of a 3-m-square solar panel that provided approximately 85-W output, a main battery and 24-V nonrechargeable battery that together yielded a 4090-W total output, a planar array antenna, two omnidirectional antennas, and a radar altimeter. The soft landing was achieved by the spacecraft free falling to the lunar surface after the engines were turned off at a 3.5-m altitude. Operations began shortly after landing.

Figure 2.1 is the surveyor spacecraft with major subsystem components identified.

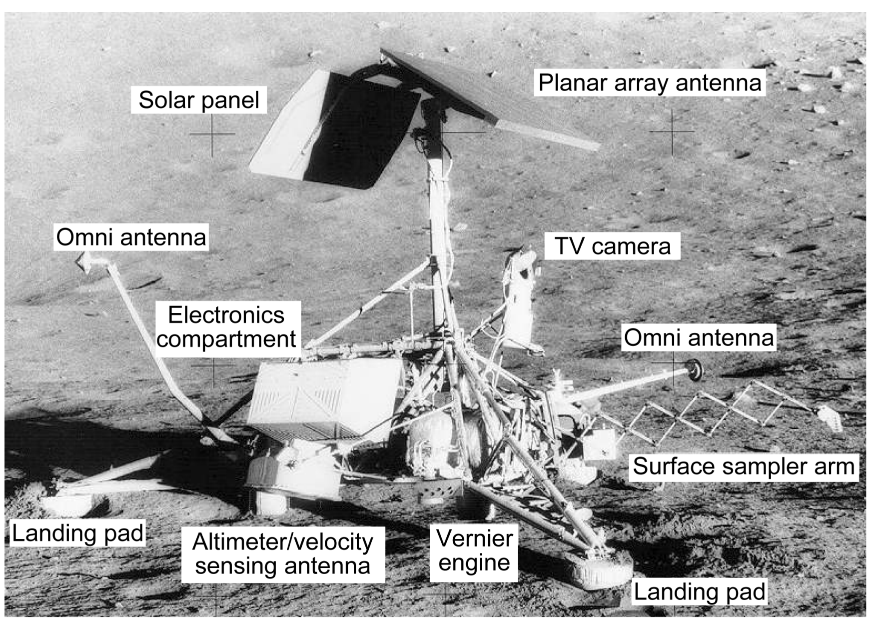


Figure 2.1—Surveyor Lander.

2.2.2 Fault Tolerance

In the interests of minimizing costs this design is assumed to use a zero-fault tolerant approach.

2.3 Growth, Contingency and Margin Policy

Mass Growth: The COMPASS team uses the ANSI/AIAA R-020A-1999, *Recommended Practice for Mass Properties Control for Satellites, Missiles, and Launch Vehicles*. Table 2.2 shows the Percent Mass Growth separated into a matrix specified by level of design maturity and specific subsystem.

TABLE 2.2—PERCENT MASS GROWTH ALLOWANCE (MGA)

Code	Design Maturity (Basis for Mass Determination)	Percent Mass Growth Allowance									
		Electrical/Electronic Components			Structure	Thermal Control	Propulsion	Batteries	Wire Harnesses	Mechanisms	Instrumentation
		0 to 5 kg	5 to 15 kg	>15 kg							
E	Estimated (preliminary sketches)	30	20	15	18	18	18	20	50	18	50
L	Layout (or major modification of existing hardware)	25	20	15	12	12	12	15	30	12	30
P	Pre-Release Drawings (or minor modification of existing hardware)	20	15	10	8	8	8	10	25	8	25
C	Released Drawings (calculated values)	10	5	5	4	4	4	5	5	4	5
X	Existing Hardware (actual mass from another program)	3	3	3	2	2	2	3	3	2	3
A	Actual Mass (measured flight hardware)	0	0	0	0	0	0	0	0	0	0
CFE	Customer Furnished Equipment	0	0	0	0	0	0	0	0	0	0

The percent growth factors are applied to each subsystem, after which the total system growth of the design is calculated. The COMPASS system designed to total growth of 30% or less. An additional growth is carried at the system level in order to add up to a total system growth of a maximal 30% limit on the dry mass of the system. Note that for designs requiring propellant, growth in propellant is either carried in the propellant calculation itself or in the ΔV used to calculate the propellant required to fly a mission.

From the Discovery AO (Announcement of Opportunity): Definitions of Contingency and Mass Contingency (or Reserve), when added to a resource, results in the maximum expected value for that resource. Percent contingency is the value of the contingency divided by the value of the resource less the contingency.

Margin is the difference between the maximum possible value of a resource (the physical limit or the agree-to limit) and the maximum expected value for a resource. Percent margin for a resource is the available margin divided by its maximum expected value.

Power Growth: The COMPASS team uses a 30% margin on the bottoms up power requirements in modeling the power system. See Section 3.1.2 for the power system assumptions.

2.4 Mission Description

2.4.1 Mission Analysis Assumptions

2.4.1.1 Landing Options

The following set of bullets describes the options for landing a Robotic Lunar Lander that were discussed during the brain storming portion of the COMPASS design sessions. The powered landing was chosen in the baseline chemical configuration.

- Powered hover or 'drop'
- Terminal Systems
  - Legs (crushable)
  - Vented Airbag (no bounce)
    - The 100 kg rover/instrument example shows feasibility of landing from drop height of 100 m. At this height, the landing velocity is 18 m/s.
    - It is assumed that a payload will be oriented and released so that the bag side remains down (without tumble) and the package will need to land with a single bag stroke. Vent patches can be designed so as to attenuate most of the kinetic energy in the impact and keep the package from contacting the surface and rebounding.
    - With a 1/6 G environment, a rebounding bag similar to the Mars rover landing could bounce for a significant amount of time and could settle in an undesirable location. Deceleration rates are highly variable depending on the design chosen. However, the need for the bag appears feasible to keep the landed package below 30 G's to avoid special shock hardened technology for the instruments. (http://marsrovers.nasa.gov/mission/spacecraft_edl_airbags.html)
    - BEAGLE—The British-led BEAGLE2 project is the probe for the European Space Agency's (ESA) Mars Express mission. This airbag design makes use of a multisectioned bag assembled to form a single sphere. The system went through extensive and successful testing at NASA's Plum Brook and JSC facilities. The airbags were integrated to the Lander in January 2003, and launched the following June. Unfortunately the airbags did not have an opportunity to function, due to an undetermined overall mission failure
  - Survivable penetrator
  - Crushable engines/structure (dual use structure)

2.4.1.2 Main Propulsion Options

Table 2.3 shows the comparison between case 1 (baseline—chemical) and Cases 2 and 3 using EP for lunar transfer.

2.4.2 Main Mission Trajectory Options

2.4.2.1 Ballistic Earth-Moon Transfer (Direct Hohmann)

The baseline case chosen used a direct chemical injection Hohmann transfer to the Moon. The launch vehicle injects the Robotic Lunar Lander/Star motor stack directly to Trans-Lunar Injection (TLI). The Star Motor performs approximately 88% of the descent burn, and a hydrazine propulsion system on the lander performs the rest for a soft landing.

Table 2.4 breaks out the three stages in this transfer. Because of the generic design of the system tools, there is a line in the table in the Lunar Orbit Insertion (LOI) stage for useable propellant. In the baseline Case 1 detailed in this report, the Minotaur V rocket is used to perform a TLI burn direct injection to the Moon, therefore there is no useable propellant carried on the Robotic Lander portion of the vehicle. Cases 2 and 3 would have a useable propellant number in this cell where the EP system's Xenon (Xe) propellant would be book-kept.

TABLE 2.3—MISSION ANALYSIS DELTA V COMPARISON

	Case 1—Chemical Propulsion				Case 2—Electric Propulsion			
Phase name	Main DV main propulsion item		RCS DV RCS propulsion item		Main DV main propulsion item		RCS DV RCS propulsion item	
Launch from Earth								
Checkout								
Loiter to TLI window opening								
TLI opening to ignition								
TLI burn								
TLI stage disposal			10 m/s	Lander			10 m/s	Lander
Trans-Earth mid-course corrections	20 m/s	Lander	10 m/s	Lander	20 m/s	Lander	10 m/s	Lander
Trans-lunar coast								
Lunar orbit capture burn coast					4545 m/s	EP Stage		
LOI plane change burn								
Coast								
Lunar orbit circularization burn								
Lunar descent burn	2191 m/s	Descent stage	11 m/s	Lander	1499 m/s	Descent stage	11 m/s	Lander
Lunar landing burn	299 m/s	Lander			224 m/s	Lander		
	Launch: Minotaur V launch to TLI Delivers 550 kg TLI: Performed by launch vehicle LOI: None, direct landing Landing: Direct from TLI Star 27 performs 88% of the descent Vernier system used for remainder				Launch: Minotaur V launch to GTO Delivers 710 kg TLI: Performed by EP stage LOI: Performed by EP stage Landing: Descend from LLO Star 17A performs 88% of the descent Vernier system used for remainder			

TABLE 2.4—CHEMICAL DIRECT TRANSFER STAGING

LOI stage	
Useable propellant, kg	0.0
Total wet mass, kg	29.5
Inert mass, kg	0.0
EDS adaptor mass + 30%, kg	29.5
Descent stage	
Useable propellant, kg	238.3
Total wet mass, kg	266.8
Inert mass, kg	28.5
Lander stage	
Useable propellant, kg	30.3
Total wet mass, kg	165.7
Inert mass, kg	135.5

2.4.2.2 Earth-Moon Transfer Weak Stability Boundary (WSB) Analysis

Belbruno (AIAA–2000–4142) and Parker (AAS 06–132) and others (Lo, Koon, etc.) have demonstrated ballistic lunar transfers via the “weak stability boundary” (WSB). WSB transfers require more ΔV for TLI than a direct Hohmann-like (Apollo) lunar transfer, but the spacecraft is captured by solar/lunar gravity into a lunar libration orbit without any propulsive ΔV. Apogee is ~1.5×10^6 km (235 Earth radii), where the Sun’s gravity perturbs the trajectory and raises perigee to lunar-centered orbit of radius (384,000 km). Typically, the spacecraft is captured into a “halo” orbit about the lunar libration point L2 (far side of the Moon on Earth-Moon line). The spacecraft can leave the unstable halo orbit (for zero ΔV) and follow the invariant manifold (surface of trajectories) to the vicinity of the Moon. WBS transfers also require much longer trip time (>90 days to the lunar libration orbit).

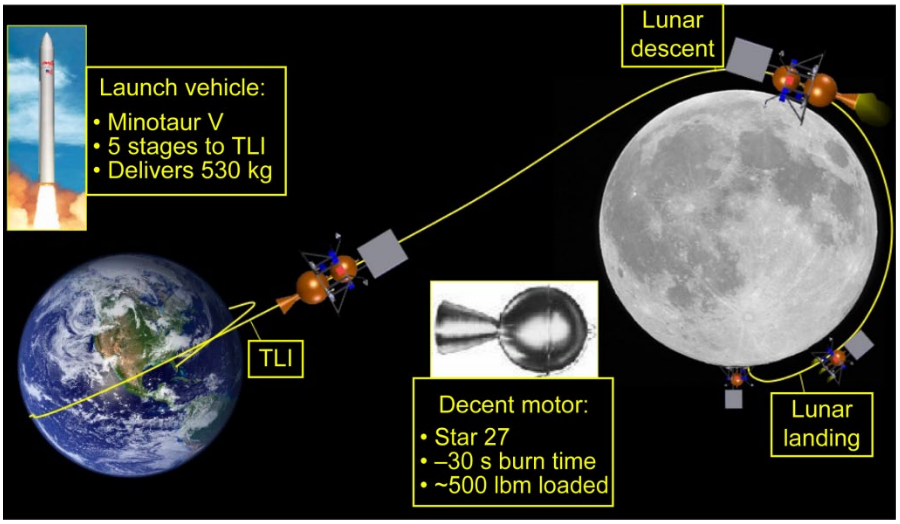


Figure 2.2—Baseline mission event timeline illustration.

2.4.3 Mission Analysis Event Timeline

Figure 2.1 shows the major events in the baseline all-chemical propulsion option (case 1). The Minotaur V Launch Vehicle lifts the Robotic Lunar Lander to TLI using five stages. The Star 27 motor performs the direct lunar descent maneuver and then is jettisoned. A small hydrazine propulsion system on the Robotic Lander performs the last landing maneuver to provide controlled touchdown and safe landing of the science instruments.

2.4.4 Mission Trajectory Details

For comparison, Table 2.5 presents the Delta V at different burns along the trans-lunar mission. The direct Hohmann type transfer was used in the baseline mission design. The WSB trajectory analysis was left for another mission if an Electric Propulsion option were found that delivered positive payload. Note that (TOF) is time of flight, or the total trip time to the Moon. A standard direct, Hohmann-type transfer takes approximately 4 days.

For the comparison from low Earth orbit (LEO) to low lunar orbit (LLO) the initial LEO was assumed to be 300-km altitude circular orbit and the target LLO was a 100-km altitude circular polar orbit.

TABLE 2.5—TRAJECTORY DELTA V TOTALS LEO TO LLO

Trajectory Method	DV_{TLI} (km/s)	DV_{LOI} (km/s)	DV_{total} (km/s)	TOF (days)
WSB	3.176	0.664	3.840	100.5
Direct (Hohmann)	3.107	0.819	3.926	4.1

Table 2.6 shows the Delta V's for comparing the mission phases of a WSB trajectory to a direct lunar transfer assume impulsive Delta V for landing (no gravity losses and no LLO parking orbit). For the direct transfer trajectory, assume perilune of incoming hyperbola grazes lunar surface. The trajectories were run starting from Initial LEO altitude of 300 km (circular) and target was lunar surface.

TABLE 2.6—TRAJECTORY DELTA V TOTALS LEO TO LUNAR SURFACE

Trajectory Method	DV_{TLI} (km/s)	DV_{LAND} (km/s)	DV_{total} (km/s)	TOF (days)
WSB	3.176	2.365*	5.541	100.5
Direct (Hohmann)	3.107	2.490	5.597	4.1

2.5 Small Launch Vehicle Details

A number of small class launch vehicles were examined for use in this mission. The Minotaur and Falcon class vehicles were the two that were most appropriate to the launch mass required for the Robotic Lunar Lander.

2.5.1 Minotaur

Minotaur V is a five-stage evolutionary version of the Minotaur IV space launch vehicle (SLV) to provide an extremely cost-effective capability to launch small spacecraft into high-energy trajectories, including geosynchronous transfer orbits (GTO) and trans-lunar missions.

The first three stages of the Minotaur V are the unmodified Peacekeeper solid rocket motors. The stage four motor is nominally a Star-48GV. The fifth stage can be either attitude controlled or spinning. The attitude-controlled version nominally uses the same Orion-38 motor that has been extensively flight demonstrated on multiple Orbital launch vehicles, including Pegasus, Taurus, and Minotaur I. For a spinning configuration, a Star-37FM is used to provide maximum performance.

The Minotaur V avionics, structures, and fairing are common with the Minotaur IV SLV, with relatively minor changes to create the five-stage configuration. Moreover, the avionics and flight software are highly common across all Minotaur family vehicles (Orbital Sciences Minotaur V Fact Sheet pdf).

Figure 2.3 shows the "projected performance to C_3" values of the Minotaur V launch vehicle taken from the Orbital Sciences Minotaur V Fact Sheet.

2.5.2 SpaceX—Alternate Launch Vehicle Option

For comparison with the Minotaur launch vehicle family, the SpaceX Falcon launch vehicles were looked at as a possible launcher for this payload. Figure 2.4 is a graphic taken from the SpaceX site listing the relative characteristics of the family of Falcon launch vehicles. (http://www.spacex.com/falcon1.php).

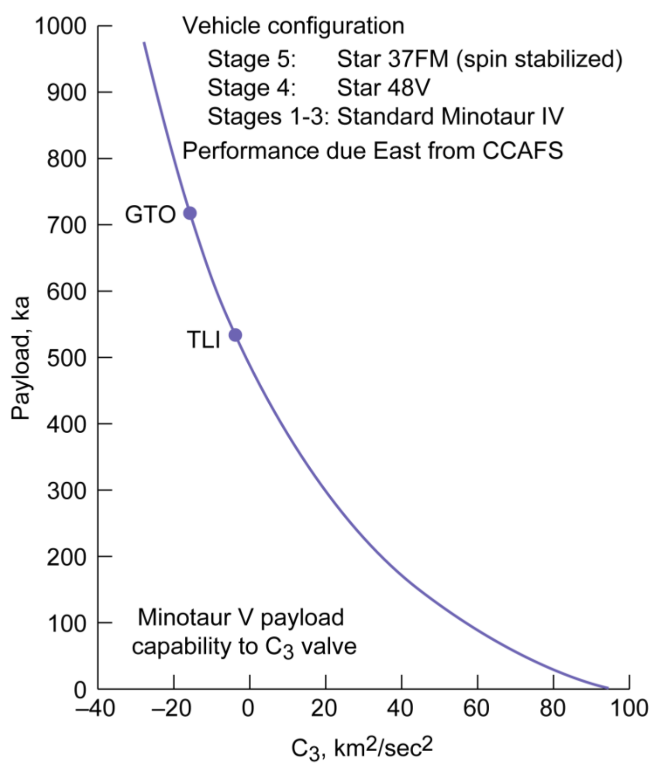
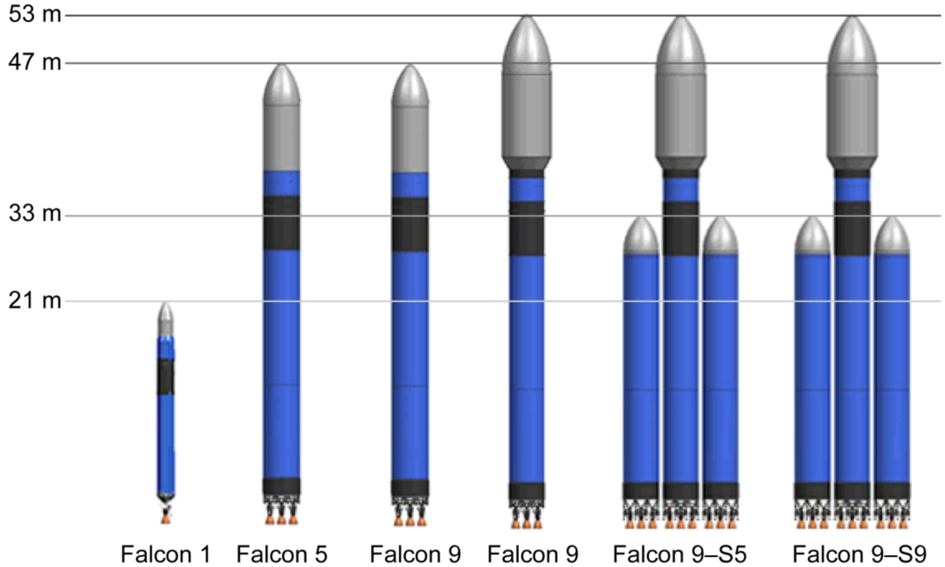


Figure 2.3—Minotaur V launch vehicle performance curve.



	Falcon 1	Falcon 5	Falcon 9	Falcon 9	Falcon 9–S5	Falcon 9–S9
LEO	570 kg	4,100 kg	9,300 kg	8,700 kg	16,500 kg	24,750 kg
GTO	—	1,050 kg	3,400 kg	3,100 kg	6,400 kg	9,650 kg
Fairing diameter	1.5 m	3.6 m	3.6 m	5.2 m	5.2 m	5.2 m
Price in millions*	$6.7	$18	$27	$35	$51	$78

*Prices are all inclusive of launch range, third party insurance and standard payload integration costs.

Figure 2.4—Space X Falcon launch vehicle family.

2.6 System Design Trade Space: Preliminary Analysis

The preliminary analysis on landed mass using the small launch vehicle performance shows that a 150 to 200 kg Landed Mass is feasible. For EP the trade is: Minotaur I with EP (150 kg LM) versus Minotaur IV with Chemical (170 kg LM).

TABLE 2.7—SYSTEM DESIGN SPACE PRELIMINARY MISSION SIZING

Chemical missions	Type	Estimate landed mass	Assumptions
Minotaur I	LEO-Direct	50 kg	12% dry stage fraction assumed
Minotaur IV	LEO-Direct	170 kg	12% dry stage fraction assumed
Minotaur V	TLI-Direct	215 kg	Solid (Isp = 285 sec)
Minotaur V	TLI-Direct	150 kg	Monoprop (Isp = 225 sec)
SEP missions	**Type**	**Estimate landed mass**	**Assumptions**
Minotaur I	LEO-LLO	148 kg	470 day total trip time
Minotaur IV	LEO-LLO	500 kg	471 day total trip time
Secondary P/L	GTO-LLO	170 kg	500 kg assumed initial mass

2.7 Baseline System Design

The basic design consists of launch on a Minotaur V to TLI on a direct trajectory (similar to Surveyor) to the Moon. The propulsion system is a combination of solid stage for descent and a Monoprop Hydrazine propulsion system on the Lander for landing. The TLI is a 4-day transfer, landing approximately 15 kg useable payload on lunar surface.

Figure 2.5 shows the Robotic Lunar Lander from two angles, with all major components labeled. The Star 27 motor is shown attached along with the spacecraft adaptor. The landing legs are extended and both the solar array and the antenna are deployed. Figure shows a simple drawing of the Robotic Lander on the Star 27 motor with the Minotaur V adaptor at the bottom. The landing legs, antenna, and solar array panel are deployed. Figure 2.7 shows the rendered lander portion of the baseline design after the star motor and associated adaptor have been jettisoned.

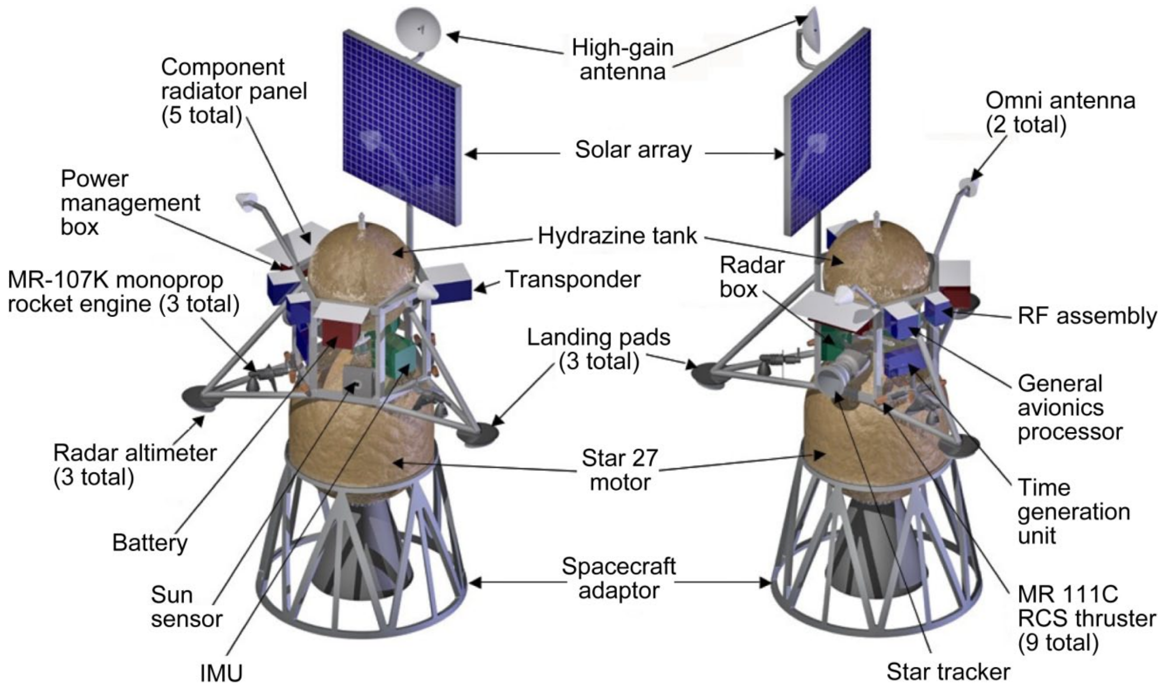


Figure 2.5—Baseline Robotic Lunar Lander/Star 27 stack with component labels.

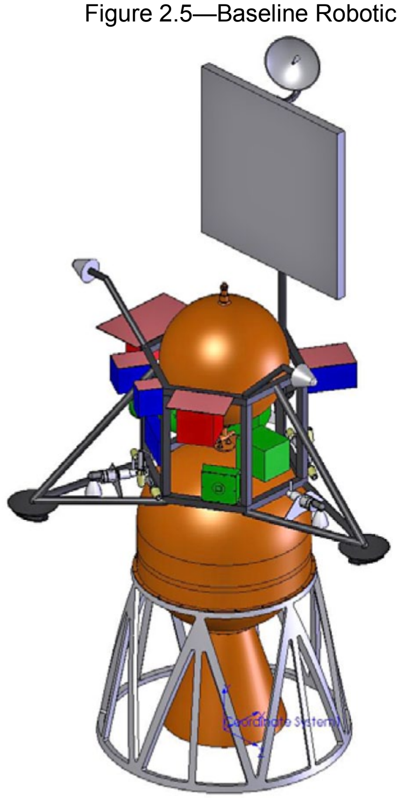


Figure 2.6—Baseline Robotic Lunar Lander/Star 27 stack.

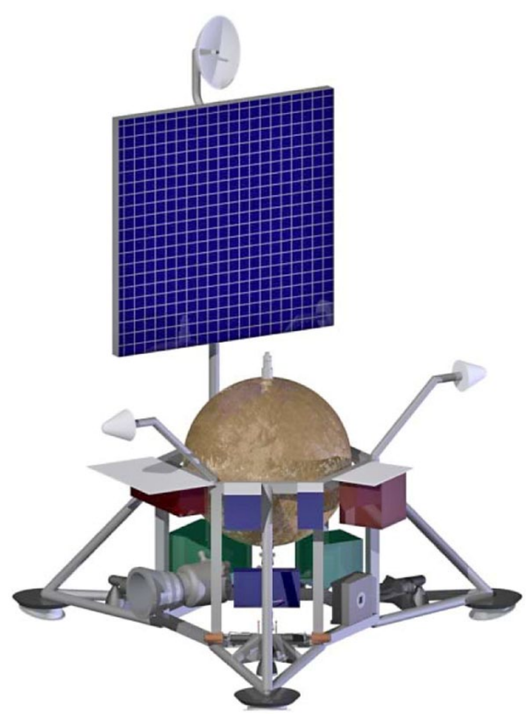


Figure 2.7—Baseline Robotic Lunar Lander: Lander portion.

3.0 Baseline Design

3.1 Top Level Design (MEL and PEL)

For the purpose of modeling in the Master Equipment List (MEL), this baseline design is broken into three stages: the LOI stage, the descent stage, and the lander stage. The LOI stage is the structure necessary to attach the Star 27 rocket to the Minotaur V rocket. The decent stage is the Star 27 Rocket, and the Lander stage is the Robotic Lunar Lander (structure, power, thermal, etc.) and science payload package along with the hydrazine system used for landing.

3.1.1 Master Equipment List (MEL)

Table 3.1 MEL is the top level MEL tool developed in the COMPASS design session. Note that the Growth (%) column displays the total top level growth percentage for the wet portions of the vehicle. This is not the way growth is used in the design session, but is still calculated in the tool. The only growth percentages to refer to are on the dry masses, not the total wet mass (i.e., with propellant). The Growth column in kg is the summation of the system level growths of the dry masses. These calculations are a function of the tool. For more clear growth calculations, see the system summary Table 3.3. The total growth percentage of the wet mass calculation need to be refined in future versions of the MEL tool. The total growth percentage using the wet mass number in the percentage calculation gives a deceptive percentage number. Growth percentage is calculated growth divided by total current base estimate (CBE) mass. When the wet mass is used instead of the dry in the CBE mass, the growth percentage is skewed.

TABLE 3.1—MEL

WBS	Description	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
01	Robotic Lunar Lander	429	7	33.3	462
01.01	Lunar Orbit Insertion Stage	25	15	4.5	29.5
01.02	Lunar Descent Stage	266	0	1.1	266.8
01.02.02	Propulsion (Chemical)	27	4	1.1	28.5
01.02.03	Propellant Management (Chemical)	0	0	0.0	0.0
01.02.04	Propellant Hardware (Chemical)	238	0	0.0	238.3
01.03	Lunar Lander Stage	138	16.7	27.7	165.7
01.03.01	Avionics, Instrumentation, GN&C	35	16	6.8	42.0
01.03.02	Propulsion Hardware (Chemical)	5	5	0.3	5.6
01.03.03	Propellant Management (Chemical)	9	6	0.5	9.6
01.03.04	Propellant (Chemical)	30	20	7.6	37.8
01.03.05	Propulsion Hardware (EP)	0	0	0.0	0.0
01.03.06	Propellant (EP)	0	0	0.0	0.0
01.03.07	Propellant Management (EP) Power	0	0	0.0	0.0
01.03.08	Processing (EP)	0	0	0.0	0.0
01.03.09	Power	24	21	6.5	30.9
01.03.10	Thermal Control (Non-Propellant)	18	15	3.3	21.7
01.03.11	Structures & Mechanical Systems	15	15	2.8	18.2

Where the MEL (Table 3.1) captures the bottoms up estimation of CBE and growth percentage line item by item from the subsystem designer, Table 3.3, wraps up those total masses, CBE and total mass after applied growth percentage. In order to meet the total of 30% at the system level, an allocation is necessary for system level growth.

3.1.2 Power Equipment List (PEL)

The PEL for nominal loads is show in Table 3.2.

TABLE 3.2—PEL

Subsystem	Peak power (W)	Keep alive power (W)
Science payload	25	10
Thermal control	20	--
Communications	34	--
Avionics	15	--
Total to size solar arrays	94	--

3.2 System Level Summary

The system level summary will track the total MEL and add on the margins and growths per the ANSI/AIAA R-020A-1999 growth schedule as described in Section 2.4. The COMPASS team used 10% of Launch Mass as the launch Margin set aside. Margin is not flown, nor can it be used as performance by the ELV. The hydrazine propellant growth was assumed at 25% (per the definitions in *Space Mission Analysis and Design* (SMAD)).

The total mass of the robotic lander is calculated by summing the total masses of the LOI stage (30 kg), the descent stage (267 kg) and the Lander stage (166 kg). This sum was 462 kg. Subtracting the propellant (238 kg on the descent stage 38 kg on the lander stage) from this number leaves a total dry mass of 186 kg. The total inert mass is calculated by subtracting the total useable propellant (268 kg summed from the stages in Table 2.3) from the ideal rocket equation from the total lander wet mass. This leaves 8 kg (total propellant—useable propellant) of total residuals trapped in the tanks and carried as inert in terms of the system mass that the propulsion system must push through the application of the Delta V. These calculations are shown in Table 3.3 in the system summary chart.

The system summary chart shows the top level total masses with subsystem growth margin applied for each of the three stages in the Robotic Lunar Lander: LOI stage, descent Stage, and lander stage.

TABLE 3.3—CASE 1 (BASELINE) SYSTEM SUMMARY CHART

Robotic Lander Stage Masses (kg)	Current values with growth margin (kg)			Totals	Percent of dry mass
Subsystems	LOI stage	Descent stage	Lander stage	Total Robotic Lander	Total Robotic Lander
Avionics	0	0	42	42	23
Propulsion (chemical)	0	29	6	34	18
Prop Management (chemical)	0	0	10	10	5
Propellant (chemical)	0	238	38	276	N/A
Power	0	0	31	31	17
Thermal Control	0	0	22	22	12
Structure	30	0	18.15	48	26
Total Mass (Wet)	30	267	166	462	N/A
Lander Total Wet Mass	462				
Lander Total Dry mass (kg)	186				
Total Residuals	8				
Lander Total Inert Mass (kg)	194				
ELV Performance to TLI	477				
Gross Payload Capability	14.9				

At the time of this analysis, there was no additional growth carried at the system level for this design. The growth is entered in by the subsystem lead per instructions in Table 3.1 MGA Schedule.

3.3 Design Concept Drawing and Description

Figure 3.1 shows the side view of the Robotic Lunar Lander on top of the Star 27 motor, with the payload adaptor truss structure. Figure 3.2 shows the top down view of the Robotic Lunar Lander with dimensions.

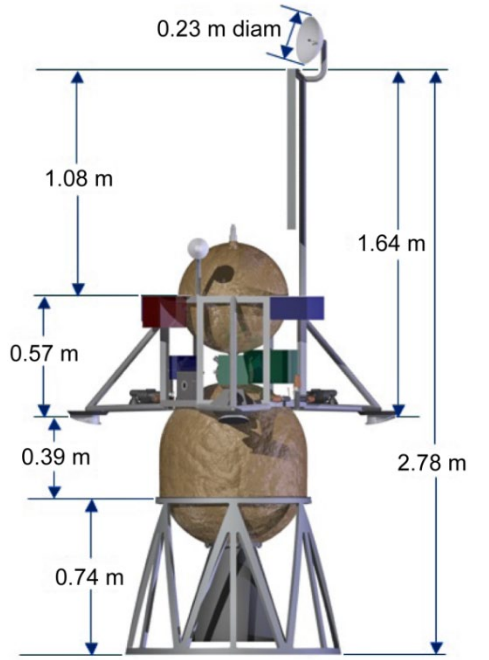


Figure 3.1—Baseline Robotic Lunar Lander: Total stack with dimensions.

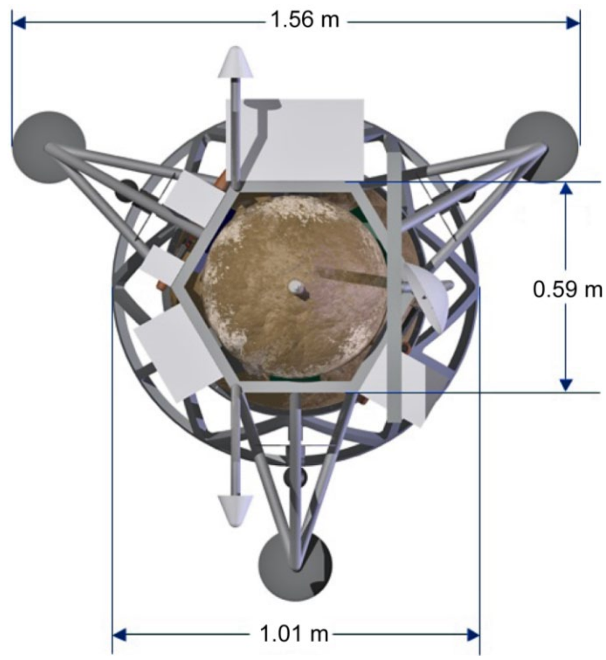


Figure 3.2—Baseline Robotic Lunar Lander: Top down view with dimensions.

4.0 Subsystem Breakdown

4.1 Communications

4.1.1 Communications Requirements

The Spacecraft RF Power level is driven by data transmission needs of the science payload. Since there is no defined science payload at this step in the design, the following two items are used as the requirements for data levied on the communications system.

- Communications Data Rate
  - Telemetry: 6 kbps
  - Command: up to 2 kbps

4.1.2 Communications Assumptions

- Deep Space Network (DSN) or Tracking and Data Relay Satellite System (TDRSS) for Earth antenna tracking capability tone ranging
- Communications system incorporates radio frequency (RF) transponder and RF antenna functions
- Transponder handles RF/digital signals between the antenna and digital interface
- RF amplification/attenuation, diplexing, mod/demod, synchronization, ranging signal turn-around (navigation (nav) transponder)
- Four conical spiral omnidirectional antennas have no active pointing mechanism
- Best antenna selected based on strongest received signal strength (antenna pointing at source)
- Signal processor performs encoding, decryption, and frame synchronization
- Space serial data system through IEEE 1394 bus (or 1553B bus)
- State vector requirements: (where am I, what direction, how fast?)
  - Doppler/tone ranging for state vector generation command data
  - Uplink—provides state vector, and timing updates, and real time commands
  - Downlink—vehicle telemetry data

4.1.3 Communications Design and MEL

The only communications hardware was on the Lunar Lander stage. This is the stage with the structure, power and science instruments ultimately landed on the lunar surface. There is no communications hardware on the LOI or Descent stages. Figure 4.1 shows the schematic of the design of the Lunar Lander communications system. Table 4.1 is the communications system bottoms up MEL. The Growth percentage is only important on the line items. Refer to the growth (kg) totals for details.

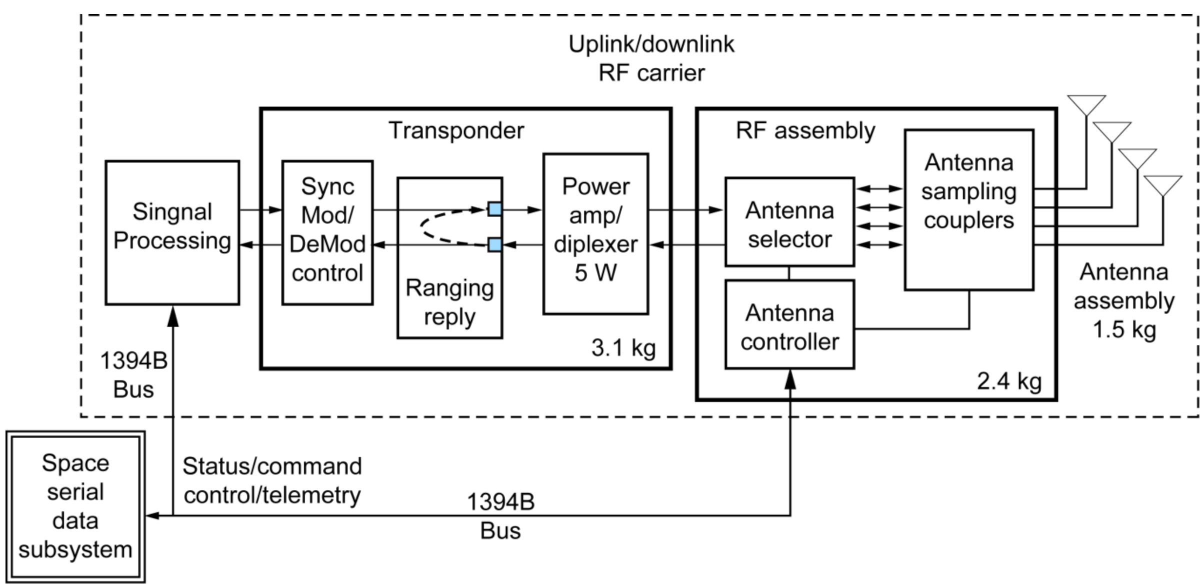


Figure 4.1—Lunar Lander communications system.

TABLE 4.1—COMMUNICATIONS SYSTEM MEL

WBS	Description	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
01	Robotic Lunar Lander	-	---	429	7	33.3	462
01.01	LOI Stage	-	---	25	15	4.5	29.5
01.02	Lunar Descent Stage	-	---	266	0	1.1	266.8
01.03	Lunar Lander Stage	-	---	138	16.7	27.7	165.7
01.03.01	Avionics, Instrumentation, GN&C	-	---	35	16	6.8	42.0
01.03.01.01	C&DH	-	---	8.3	16	1.6	9.9
01.03.01.02	Communication	-	---	6.9	11	0.9	7.7
01.03.01.02.01	S-Band Comm System	-	---	6.9	11	0.9	7.7
01.03.01.02.02	Transponder	1	3.4	3.4	3	0.1	3.5
01.03.01.02.03	RF Assembly	0	5.9	0.0	20	0.0	0.0
01.03.01.02.04	Processing Module	0	0.0	0.0	30	0.0	0.0
01.03.01.02.05	Antenna	4	0.8	3.0	20	0.6	3.6
01.03.01.02.06	Coaxial Cable	1	0.5	0.5	30	0.2	0.7
01.03.01.02.07	Installation—Mounting & Circuitry	0	1.9	0.0	30	0.0	0.0
01.03.01.03	GN&C	-	---	20.1	18	4.4	24.4

4.1.4 Communications Trades

None.

4.1.5 Communications Analytical Methods

None.

4.1.6 Communications Risk Inputs

None.

4.1.7 Communications Recommendation

None.

4.2 Avionics

For the purpose of this design and report, Avionics consisted of the C&DH system as well as the GN&C system.

4.2.1 Avionics Requirements

The avionics system needs to provide all processing support for the Earth-Moon transit, landing and surface operations. Storage requirements were estimated to be minimal since direct communications will be available during landed science operations. The processing however will have to be quick enough to handle landing control. However, in the interest of cost, a smaller computer will be sought with an estimated MIPS performance at 1 MIPS or less and memory in the range of 8 to 16 MB.

4.2.2 Avionics Assumptions

4.2.2.1 Guidance, Navigation and Control (GN&C) Assumptions

The goal of the GN&C subsystem design was to provide for a successful lunar landing at minimum cost, and design a system that may be zero fault tolerant.

Assumptions

- Successful lunar landing requires attitude control
  - For solar panel
  - For propulsion

- For communications
- Successful lunar landing requires knowledge of position
  - For propulsion

4.2.3 Avionics Design and MEL

The GN&C system consisted of

- Hardware suite based on Surveyor
  - Surveyor was a minimalist design
  - Technological advances may make our implementation more than minimalist
- Catalog items
  - Sun sensor: Sodern Digital Sun Sensor (http://www.sodern.eu/site/FO/scripts/index.php)
    - French
  - Star tracker: Sodern SED 16
    - French
  - Inertial measurement unit: Honeywell HG9900
    - (http://www.honeywell.com/sites/aero/Military-Aircraft3_CF9E0EA52-E7BD-F572-03CA-068C338328C1_HB0A141BB-2FD0-7339-851E-22466484B4C6.htm)
- Augmented catalog item
  - Radar altimeter: Honeywell HG9550 augmented
    - http://www.honeywell.com/sites/aero/Military-Aircraft3_CC97F1FA1-ED1D-D7D4-15D8-1272F2E6C702_HFC5482F9-3DD3-2A4F-DE15-0A1BCB4CA9C6.htm

Figure 4.2 shows the schematic design of the avionics system—this is primarily the C&DH system but also includes the linkage to the GN&C system.

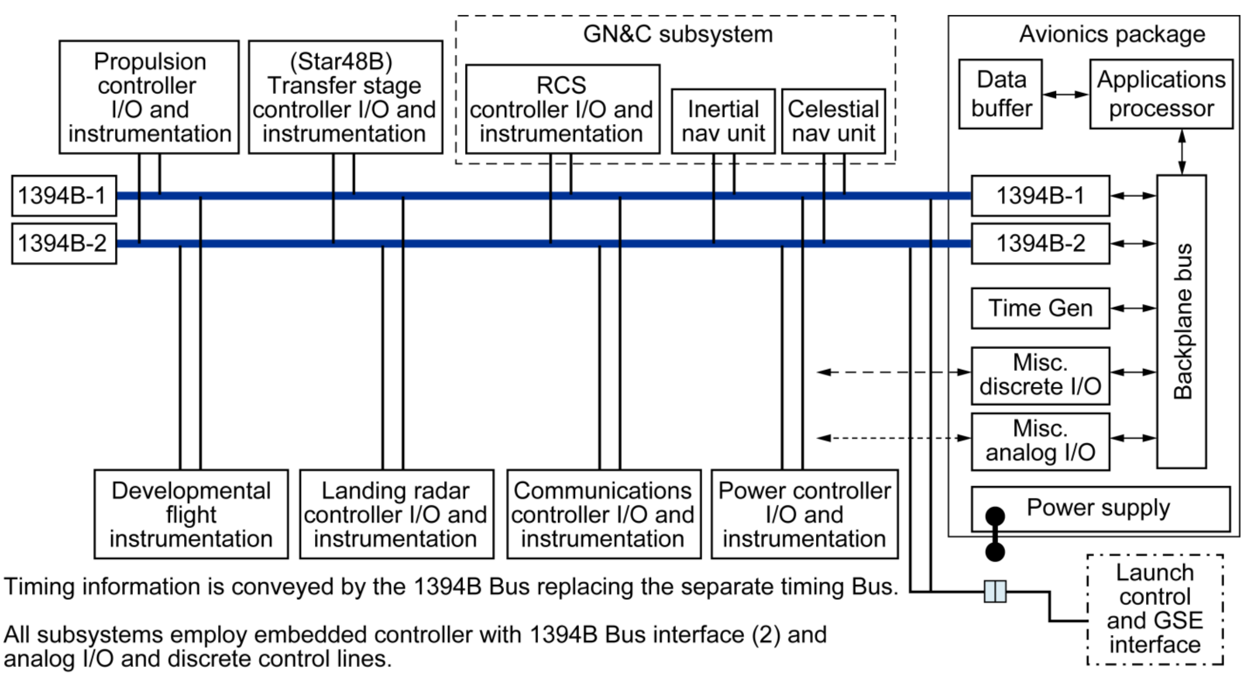


Figure 4.2—Lunar Lander avionics system.

Table 4.2 is the combined avionics MEL. The Growth percentage is only important on the line items. Refer to the growth (kg) totals for details. The total growth percentage of the wet mass calculation need to be refined in future versions of the MEL tool.

TABLE 4.2—AVIONICS SYSTEM MEL

WBS	Description	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
01	Robotic Lunar Lander	-	----	429	7	33.3	462
01.01	LOI Stage	-	----	25	15	4.5	29.5
01.02	Lunar Descent Stage	-	----	266	0	1.1	266.8
01.03	Lunar Lander Stage	-	----	138	16.7	27.7	165.7
01.03.01	Avionics, Instrumentation, GN&C	-	----	35	16	6.8	42.0
01.03.01.01	C&DH	-	----	8.3	16	1.6	9.9
01.03.01.01.01	General Avionics Processor	1	5.3	5.3	15	0.8	6.0
01.03.01.01.02	Time Generation Unit	0	0.3	0.0	30	0.0	0.0
01.03.01.01.03	Command and Control Harness (data)	1	1.2	1.2	25	0.3	1.5
01.03.01.01.04	Instrumentation & Wiring	1	1.9	1.9	25	0.5	2.3
01.03.01.02	Communication	-	----	6.9	11	0.9	7.7
01.03.01.03	GN&C	-	----	20.1	18	4.4	24.4
01.03.01.03.01	Inertial Measurement Units	1	3.0	3.0	25	0.8	3.8
01.03.01.03.02	Star Sensor	1	3.0	3.0	25	0.8	3.8
01.03.01.03.03	Sun Sensors	1	0.3	0.3	25	0.1	0.3
01.03.01.03.04	Star Tracker	0	0.0	0.0	30	0.0	0.0
01.03.01.03.05	Attitude control system	0	0.0	0.0	30	0.0	0.0
01.03.01.03.06	Flight Control Electronics	2	2.5	5.0	3	0.2	5.2
01.03.01.03.07	Radar System	1	8.8	8.8	30	2.6	11.4
01.03.01.03.08	Visualization System	0	12.0	0.0	30	0.0	0.0

4.2.4 Avionics Trades

None.

4.2.5 Avionics Analytical Methods

The avionics in this design were made on a modern serial bus design. The reasons from the designers are as follows:

- Crew Exploration Vehicle (CEV) is using it: Development (costs) is addressed by the Human Exploration Program
- Modular and light weight connections: Reduces vehicle harness weight and provides an compact interface, increased flexibility to add or remove systems.
- Widespread availability: Low cost commercial off the shelf (COTS) microcontrollers are available with built-in 1394B (or USB) interfaces.
- Single string systems do not require the expensive fly by wire 1553B bus
- Embedded microcontrollers are small, compact, power efficient and include instrumentation Interfaces and Network Connections.
- CEV is expected to employ a “hardened” version 1394B
- LXI LAN Extensions for Instrumentation—(IEEE 1558) typically applied to Ethernet also applies to virtually any high-speed serial bus that will support the protocols for time synchronization, Network Control and Hardware Triggering.
- Real Time or Deterministic operations are achieved by exploiting bus high speed.
- Execution of controllers accurate to tens of nanoseconds.

4.2.6 Avionics Risk Inputs

None.

4.2.7 Avionics Recommendation

None.

4.3 Electrical Power System

4.3.1 Electrical Power Requirements

Provide power to spacecraft and payloads. Distribute and control power to each electrical load on spacecraft.

4.3.2 Electrical Power Assumptions

- The spacecraft will only operate on the surface of the Moon during sunlit periods.
- Assumed 100 W housekeeping power (70 W with 40% growth) and 25 W payload power for nominal sizing (if housekeeping loads are lower than 100 W, then excess power may be used for payload)
- Peak power loads and periods from available hardware used to confirm energy storage size. Driving energy storage factor is the pad-fairing jettison period.
- Solar array sizing based on efficiencies/knockdown factors/solar array lay-down
- Battery sizing based on specific energy metric derived from accepted average values.
- Electronics and electrical based on specific power metric derived from historical data.
- Solar array boom/hinge sizing based on fraction of supported mass.
- Gimbals sizing based on off the shelf (OTS) the shelf mass (Moog).
- Support functions supplied by the other subsystems
  - GN&C provided pointing knowledge and orients the spacecraft in orbit/transit to maximize solar array visibility to the Sun (within 2°).
  - Structures provides housing boxes for electronics and batteries.
  - Thermal regulates the battery and electronics temperature to a safe operating range.

4.3.3 Electrical Power Design and MEL

Components

- Solar array: 3.6 kg (with 25% growth = 4.5 kg), 0.55 m2 (0.74 by 0.74 m)
- Solar array gimbal/electronics: 2.8 kg (with 18% growth = 3.3 kg), 8 W/axis
- Solar array hinge/boom: 4.8 kg (with 18% growth = 5.7 kg)
- Battery assembly: 2.4 kg (with 20% growth = 2.9 kg)
- Power management/control electronics: 6.8 kg (with 25% growth = 8.5 kg)
- Power distribution harness and sensor/control wiring: 4.0 kg (with 50% growth = 6.0 kg)

Table 4.3 is the power system MEL. Note that there is no power system on the LOI stage (the payload adaptor) or the Lunar Descent stage (the Star 27 motor), only on the Lander stage. This was the power to the landing propulsion system and for use on the surface. The Growth percentage is only important on the line items. Refer to the growth (kg) totals for details. The total growth percentage of the wet mass calculation need to be refined in future versions of the MEL tool.

TABLE 4.3—ELECTRICAL POWER SYSTEM MEL

WBS	Description	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
01	Robotic Lunar Lander	-	---	429	7	33.3	462
01.01	LOI Stage	-	---	25	15	4.5	29.5
01.02	Lunar Descent Stage	-	---	266	0	1.1	266.8
01.03	Lunar Lander Stage	-	---	138	16.7	27.7	165.7
01.03.09	Power	-	---	24	21	6.5	30.9
01.03.09.01	Battery System	-	---	2	17	0.5	2.9
01.03.09.01.01	Battery Assembly-Primary	0	0.0	0	20	0.0	0.0
01.03.09.01.02	Battery Assembly-Secondary	1	2.4	2	20	0.5	2.9
01.03.09.02	Solar Array	-	---	11	17	2.3	13.5
01.03.09.02.01	Solar Array Hinge/Boom	1	4.8	5	18	0.9	5.7
01.03.09.02.02	Solar Array Drive Assemblies/Motors	1	2.8	3	18	0.5	3.3
01.03.09.02.03	Solar Array Mass	1	3.6	4	25	0.9	4.5
01.03.09.03	Power Management & Distribution	-	---	10.8	26	3.7	14.5
01.03.09.03.01	Power management/control electronics	1	6.8	6.8	25	1.7	8.5
01.03.09.03.02	Power distribution/monitoring wiring harness	1	4.0	4.0	50	2.0	6.0

4.3.4 Electrical Power Trades

Options considered

- Flight proven, standard practice components.
- Triple junction solar cells on a rigid panel were assumed because this was the best trade of mass and cost.
- OTS lithium ion batteries used for low cycle geostationary communication satellites assumed for high energy density (relatively low number of cycles needed). Primary (nonrechargeable) lithium batteries were considered initially due to high “pad-fairing jettison” energy requirements, but when this time period was reduced, rechargeable batteries were optimum.
- OTS dual axis solar array drive and electronics.
- Other electronics are fairly standard but some modifications are needed.

4.3.5 Electrical Power Analytical Methods

None.

4.3.6 Electrical Power Risk Inputs

None.

4.3.7 Electrical Power Recommendation

- Lessons learned
  - Reduce the energy requirements on the pad as much as possible since that drives the battery size.
- Issues for further study
  - Need to assess solar array structures to handle various launch/thruster loads.
  - Peak loads (e.g., communications and radar) must be adjusted to have realistic (<15 min/hr) durations

4.4 Structures and Mechanisms

4.4.1 Structures and Mechanisms Requirements

The Robotic Lunar Lander and Star 27 Solid Rocket motor must fit inside the Minotaur V Launch Vehicle 92-in. payload fairing (see Figure 4.3). Mechanisms are required which separate the various portions of the spacecraft just after launch and after Star 27 rocket burnout. An additional mechanism is required to point the solar array. Payload specific mechanisms are not considered here.

4.4.2 Structures and Mechanisms Assumptions

Spacecraft must survive the following assumed loads

- Launch axial (12gs), no lateral loads used in modeling, from the Minotaur V
- Landing axial (5gs), no lateral loads used in modeling, from the Star 27 motor firing

Figure 4.3 shows the payload fairing of the Minotaur V Launch Vehicle used as the dimensions for limits in the payload packaging of the spacecraft during the design. This is the assumption used for the dimensions into which the structure and stack must fit. Note that the study used metric units in its design. However, some of the graphics from third parties use English units.

4.4.3 Structures and Mechanisms Design and MEL

Figure 4.4 shows the conceptual Robotic Lunar Lander as packaged inside the Minotaur 92-in. payload fairing. Note that the legs are deployed while in the fairing and that there is plenty of volume room for the spacecraft, antenna, legs, and solar arrays.

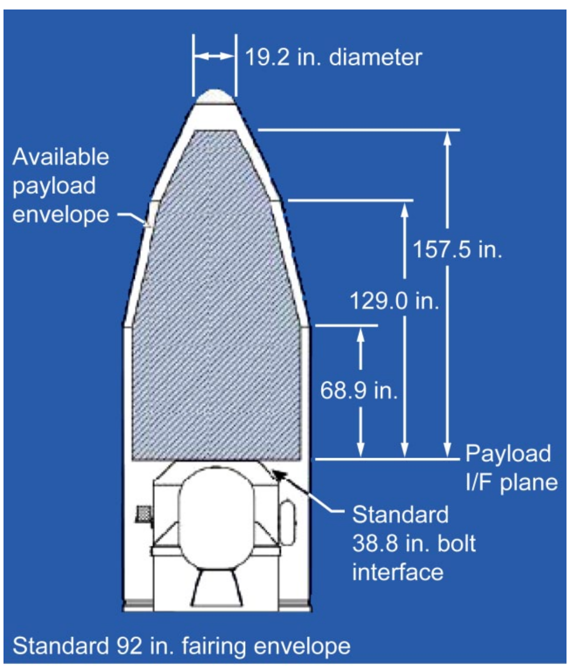


Figure 4.3—Minotaur V Launch Vehicle 92-in. payload fairing.

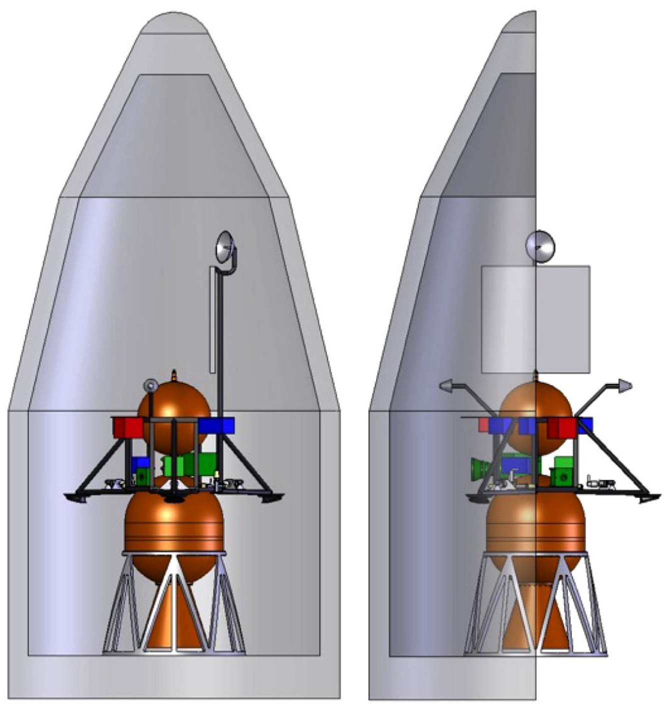


Figure 4.4—Robotic Lunar Lander with Star 27 inside Minotaur V Launch Vehicle 92-in. payload fairing.

Table 4.4 lists the Structures and Mechanical System MEL for the three stages making up the Robotic Lunar Lander. The Growth percentage is only important on the line items. Refer to the growth (kg) totals for details. The total growth percentage of the wet mass calculation need to be refined in future versions of the MEL tool. The total growth percentage using the wet mass number in the percentage calculation gives a deceptive percentage number. Typically growth percentage is calculated growth divided by total CBE mass.

TABLE 4.4—STRUCTURE AND MECHANICAL SYSTEM MEL

WBS	Description	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
01	Robotic Lunar Lander	-	----	429	7	33.3	462
01.01	LOI Stage	-	----	25	15	4.5	29.5
01.01.11	Structures & Mechanical Systems	-	----	25	15	4.5	29.5
01.01.11.01	Primary Structures	-	----	25	15	4.5	29.5
01.01.11.01.01	Landing Gear (includes mechanisms)	0	5.0	0.0	30	0.0	0.0
01.01.11.01.02	EDS adaptor interface	1	25.0	25.0	18	4.5	29.5
01.01.11.01.03	Main Structure/Fuselage	0	25.0	0.0	30	0.0	0.0
01.01.11.02	Installation	-	----	0.0	0	0.0	0.0
01.01.11.03	Mechanisms	-	----	0.0	0	0.0	0.0
01.02	Lunar Descent Stage	-	----	266	0	1.1	266.8
01.03	Lunar Lander Stage	-	----	138	16.7	27.7	165.7
01.03.11	Structures & Mechanical Systems	-	----	15	15	2.8	18.2
01.03.11.01	Primary Structures	-	----	6	15	1.1	7.3
01.03.11.01.01	Landing Gear (includes mechanisms)	3	0.4	1.2	18	0.2	1.4
01.03.11.01.02	EDS adaptor interface	0	0.0	0.0	30	0.0	0.0
01.03.11.01.03	Main Structure/Fuselage	1	5.0	5.0	18	0.9	5.9
01.03.11.02	Installation	-	----	7.7	15	1.4	9.1
01.03.11.02.01	C&DH Installation	1	0.7	0.7	18	0.1	0.8
01.03.11.02.02	Communications Installation	1	0.5	0.5	18	0.1	0.6
01.03.11.02.03	GN&C Installation	1	1.7	1.7	18	0.3	2.0
01.03.11.02.04	Power Installation	1	2.2	2.2	18	0.4	2.6
01.03.11.02.05	Propulsion Installation	1	0.4	0.4	18	0.1	0.5
01.03.11.02.06	Propellant Storage Installation	1	0.7	0.7	18	0.1	0.8
01.03.11.02.07	Thermal Installation	1	1.5	1.5	18	0.3	1.8
01.03.11.03	Mechanisms	-	----	1.5	15	0.3	1.8
01.03.11.03.01	Solar array deployment mechanism	1	0.0	0.0	30	0.0	0.0
01.03.11.03.02	Radiator deployment mechanism (if applicable)	0	0.0	0.0	30	0.0	0.0
01.03.11.03.03	Separation mechanism (pyros)	1	1.5	1.5	18	0.3	1.8

4.4.4 Structures and Mechanisms Trades

Only metallic (aluminum-lithium, AL 2090) structure was considered in this design. No other trades on structural material were conducted.

4.4.5 Structures and Mechanisms Analytical Methods

The effect of the weight of the rest of the subsystems was considered in the structure.
Level and Approach of Analyses

- Geometry
  - Hexagonal bus with three legs. See Figure 4.6, Figure 4.7, and Figure 4.8.
  - Support structure made of 1.5 in.^{2} and 1.5-in.-diameter round tubes.
- Material
  - Aluminum lithium—Al 2090
- Loads:
  - 12-g acceleration in the axial direction (from Minotaur I & IV Users Guide).

- Mass effect of other subsystems were added
- Mass calculation
  - Run finite element analysis of current design for linear static failure and buckling. If positive margins were obtained use model mass, otherwise modify and repeat. Finite element analysis was done only for the main truss structure, the rest was added as a lumped mass.
  - Add other subsystem installation mass (10% of subsystem mass)
  - Separation mechanisms in MEL were scaled from Surveyor spacecraft databook.

Figure 4.5 is output from the finite element model (FEM) structural analysis program and represents the solid model of the Case 1 (baseline) chemical lander used in the FEM analysis.

For the FEM analysis, the following structural assumptions were used

- Maximum displacement = 1.1 in.
- Maximum stress = 3520 psi
- Minimum stress = 4500 psi
- Minimum margin of safety = 7.4 (ultimate allowable 38000 psi)

Figure 4.6, Figure 4.7, and Figure 4.8 are the results of the FEM analysis using the above assumptions on displacement and stress.

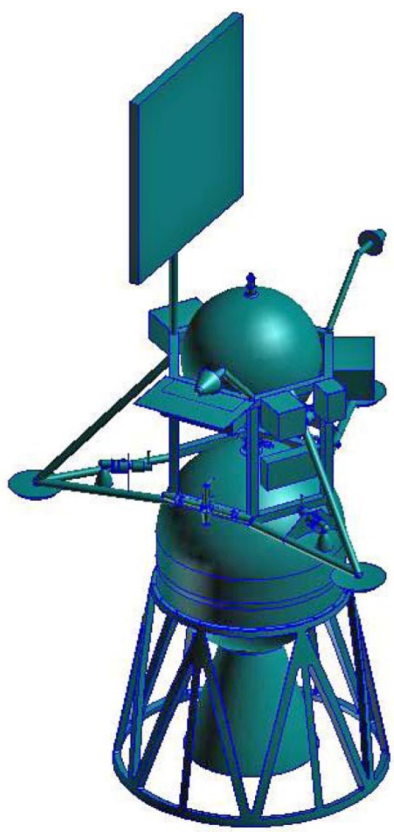


Figure 4.5—Case 1—Finite Element Model of Robotic Lunar Lander stack.

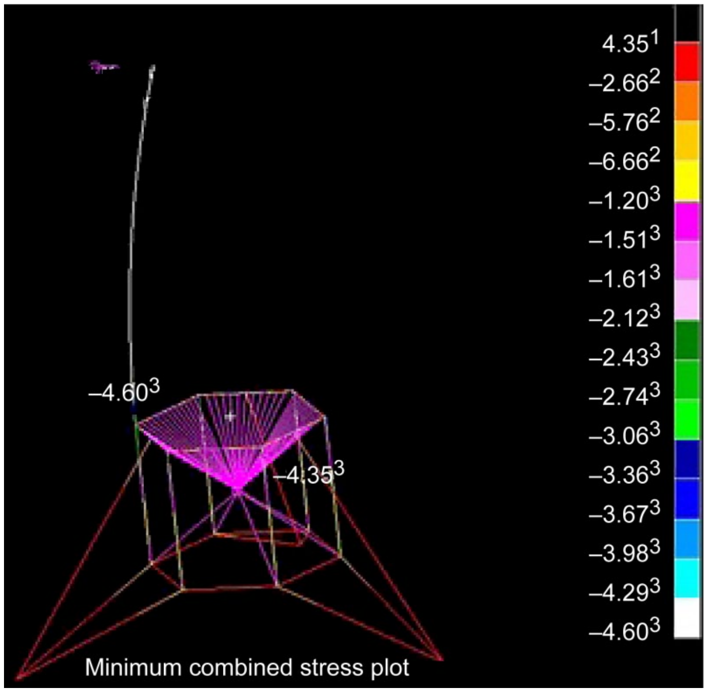


Figure 4.6—Case 1 minimum combined stress.

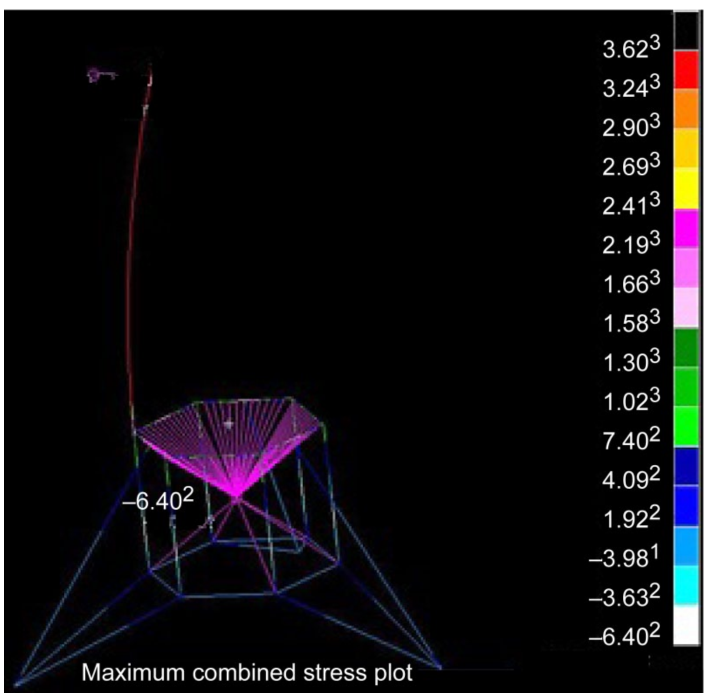


Figure 4.7—Case 1 maximum combined stress.

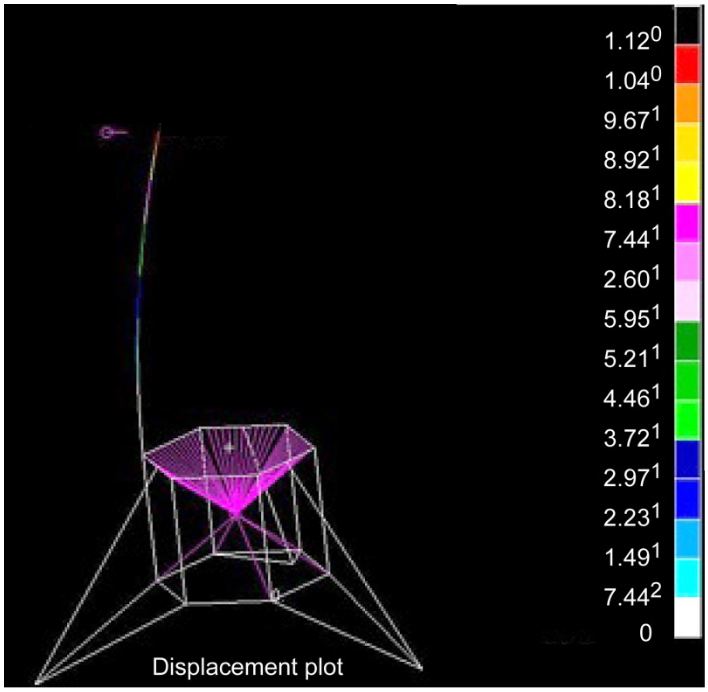


Figure 4.8—Case 1 displacement.

4.4.6 Structures and Mechanisms Risk Inputs

The risk identified for the structures was the potential impact with foreign object or due to nearby operations.

4.4.7 Structures and Mechanisms Recommendation

Mass optimization needs to be done in the structure such that sizing is done for a minimum ultimate safety factor of 2 and a minimum buckling load factor of 1.4. In addition, it will be necessary to perform frequency/normal modes analysis especially on the rigid solar array panel.

4.5 Propulsion and Propellant Management

4.5.1 Propulsion and Propellant Management Requirements

The spacecraft propulsion subsystem was required for up to three propulsion operations

1. Orbit insertion
2. Descent to lunar surface
3. Surface landing

Depending on the study case, these operations were performed by different propulsion systems. Table 2.3 shows the performance requirements for the chemical and electric propulsion options.

4.5.2 Propulsion and Propellant Management Assumptions

The propulsion subsystems designed for these studies used only COTS components to mitigate development costs and time requirements. All solid rocket motors and chemical thrusters used in the design as well as the propellant management components and propellant tanks were from operating manufactures. The electric thrusters were assumed to come from known vendors or obtained directly from the NASA development teams. Electric thruster performance is based on demonstrated operation and does not require any new qualification testing.

4.5.3 Propulsion and Propellant Management Analytical Methods

Because the propulsion subsystems were assembled from existing components, the analysis required consisted primarily of maintaining a mass roll up for the various subassemblies. The first two propulsion operations were performed with solid rocket motors that were selected from the Star motor catalog. The selection was based primarily on total spacecraft mass and the propellant load fraction of the motor. The primary analysis that was actively performed was to determine the propellant tank sizes based on propellant conditions over the mission duration. The tank requirements were determined using propellant density and storage pressure through Hoop Stress Analysis. These requirements were then used to select the best match from the PSI and Arde Inc. storage tank catalogs.

Once the storage tank(s) were selected, the helium pressurization requirements were determined. A conventional He pressurization system configuration was used, based on our experience with previous lander and Orion Service Module studies.

4.5.3.1 LOI and Descent Propulsion System Modeling

For the study of the all-chemical propulsion case, the orbit insertion and descent burns were performed with solid rocket motors:

- Solid rocket motors were selected from existing catalog of flight-proven options to minimize development requirements. The motors were selected based on:
  - Thrust level and required propellant load, that were provided by Mission Analyses
  - Best match for solid motor performance was selected and motor weight was adjusted to account for off-loaded propellant (as required)
- Use adjusted propellant mass fraction to estimate propulsion subsystem final mass for spacecraft mass determination in order to perform mission analyses
- Known configuration of existing motor used by Structures and computer aided design (CAD) teams to establish spacecraft configuration

Figure 4.9 shows a pictorial illustration of the issues associated with choosing the appropriate solid rocket motor for a mission. The trade between the Star 27 and Star 24 motors is noted at the bottom of the figure. The Star 27 motor was chosen for the Case 1 (baseline) design. (Star 27 online data sheet http://www.atk.com/customer_solutions_missionsystems/documents/atk_catalog_may_2008.pdf.)

The summary of the propulsion technology choices made for the two cases that were investigated in this study are listed in Table 4.5 for the All-Chemical option and Table 4.6 for the Electric Propulsion option.

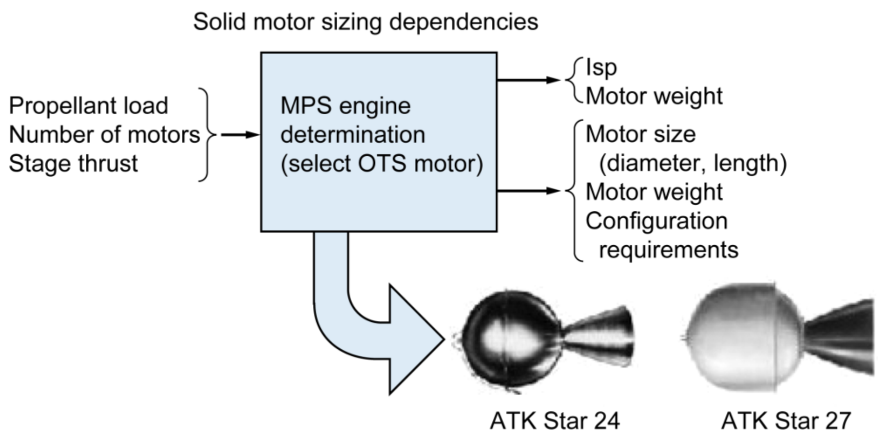


Figure 4.9—Solid motor sizing dependencies.

TABLE 4.5—ALL-CHEMICAL PROPULSION SUBSYSTEM BREAKDOWN

Stage	Engine type	Thrust level, N	Specific impulse, sec	Size, kg	Engine ID
Orbit insertion stage	Main: Solid rocket motor RCS: cold gas jets	---- ----	---- Direct trajectory ----	---- ----	---- ----
Descent stage	Main: Solid rocket motor RCS: cold gas jets	26996 ----	287.8 ----	265.5 ----	STAR 27 ----
Lander stage	Main: Liquid monoprop engine RCS: cold gas jets	220 2	222 229	0.63 0.35	MR–107K MR–111

TABLE 4.6—ELECTRIC PROPULSION OPTION ORBIT INSERTION SUBSYSTEM BREAKDOWN

Element	Type	Number	Mass, kg	Size, m	Operating power, W	Power duty cycle/ duration
Engines	HiVHAC Hall Effect thruster Thruster gimbal Thruster gimbal drive	2 2 2	3.6 4.6 2	---- TBD TBD	1350 ------ TBD	50% ---- TBD
Power management	Power processing unit	2	6	TBD	80	50%
Propellant storage	Xe high pressure storage tank—ATK–PS–80412–1	2	7	0.33 diam by 0.70	------	----
Propellant management	Feed system, fixed Feed system, per thruster	1 2	4 1	TBD TBD	TBD TBD	TBD TBD

For the study of the electric propulsion case, the orbit insertion and descent burns were performed with Hall Effect thrusters:

- The High Voltage Hall ACcelerator (HiVHAC) Hall Thrusters are under active development at GRC through the In Space Propulsion program. These were selected based on:
  - Optimum specific impulse capability allowed within available power budget
  - High efficiency operation of this thruster enabled this option to be competitive with the solid motor option while enabling increasing mission flexibility
- Xe propellant stored supercritically in high-pressure and light-weight Carbon-Overwrapped Pressure Vessels (COPV)

The liquid propellant storage tank was sized according to propellant amount and storage conditions (temperature and pressure):

- Used diaphragm tanks for blow down operation
- Selected tank from ATK-PSI catalog with necessary capacity after determining size based on propellant volume

The approach for determining the size and mass of the xenon propellant storage tank for the electric propulsion option case was similar however the xenon is stored at substantially higher pressures than the hydrazine.

Propellant management system (PMS) mass model derived from Aerojet's Hydrazine Auxiliary Propellant System (HAPS) configuration, illustrated in Figure 4.10.

- Used controlled database for component masses (valves, sensors, filters, etc)
- Assumed fixed feed line lengths

The PMS masses were composed of the component masses that were relatively static for each propellant option. Thermal control via insulation in addition to line and tank heaters was assumed to be included in the PMS mass estimate. The thermal requirements for Xe storage were significantly less than those of the hydrazine and were limited to insulation overwrap on the storage tank.

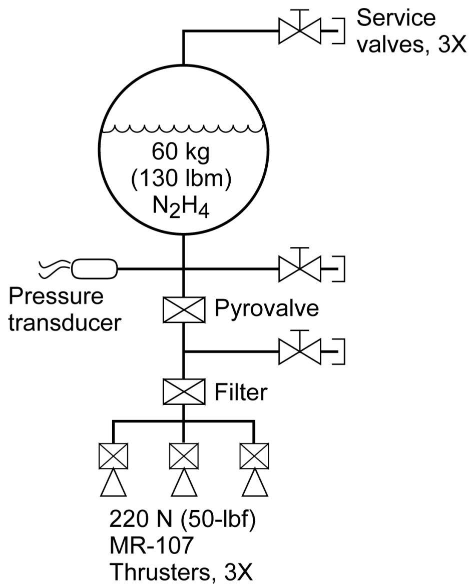


Figure 4.10—Aerojet's HAPS propulsion system schematic.

The Xe PMS was similarly modeled after historical flight systems and likewise was represented by a mass roll up of the individual components. Additionally, the power processing unit developed for the HiVHAC hall thruster was used for this model.

4.5.3.2 Lander Propulsion System Modeling

The propulsion subsystem on the Lander portion of the spacecraft performed two functions: first, controlled ("soft") landing which had traditionally required variable thrust engines, and second, reaction control propulsion for vehicle attitude maintenance and specialized maneuvers. The reaction control system (RCS) on the lander vehicle provided reaction and attitude control for the entire spacecraft over the mission.

For the main propulsion system (MPS), a liquid monopropellant engine was selected based on the performance requirements determined by trajectory analysis discussed earlier in this report that could be evolved to obtain the necessary throttling capability.

- Three MR-107K monoprop rockets were baselined for landing
  - Operating at: 257 N (57 lbf) thrust; 236 sec specific impulse
  - Operated with Hydrazine to reduce propellant management system complexity

For reaction/attitude control, small monoprop rockets were selected. This propulsion subsystem was derived from similar historical systems and was a conventional configuration:

- Three Aerojet MR-111E rockets were baselined
  - Operating at: 4.4 N (0.99 lbf) thrust, 229 sec specific impulse
  - Used Hydrazine propellant for reduced propellant management system complexity; common propellant for all-chemical option

The study scope was limited to Delta-V burn matching and did not require detailed analysis of reaction and attitude control performance. Hence, a simple mass model for the RCS was sufficient. All performance information for the rockets used in the propulsion subsystem model were obtained from either Aerojet product data sheets or from entries on the Astronautix website.

TABLE 4.7—LANDER PROPULSION SUBSYSTEM BREAKDOWN

Element	Type	Number	Mass, kg	Size, m	Operating power, W	Power duty cycle/ duration
Engines	Main: MR-107 MMH monoprop	3	0.63	0.66 diam by 2.2	60	1 min.
	RCS: MR-111 N2 Cold gas jets	6	0.35	0.36 diam by 0.17	12	5%
	Main Engine Gimbal	1	1	TBD	10	1 min.
Propellant storage	MMH (hydrazine) storage tank ATK-PSI 80259-1	2	6	0.48 Dia.	10	0.01 min.
	Valves, Service	3	0.28	TBD	TBD	7%
	Valve, Pyro	1	0.15	TBD	TBD	>1%
Propellant	Filter	7	0.05	TBD	Š	Š
Management	Transducer	2	0.1	TBD	TBD	TBD
	Valve, RCS Control	6	0.1	TBD	TBD	TBD
	Feed Lines	9	0.1	0.01	Š	Š

The liquid propellant storage tank was sized according to propellant amount and storage conditions (temperature and pressure):

- Used diaphragm tanks for blow down operation
- Selected tank from ATK-PSI catalog with necessary capacity after determining size based on propellant volume

PMS mass model derived from Aerojet’s Hydrazine Auxiliary Propellant System (HAPS) configuration

- Used controlled database for component masses (valves, sensors, filters, etc.)
- Assumed fixed feed line lengths

The PMS masses were composed of the component masses that were relatively static for each propellant option. Thermal control via insulation in addition to line and tank heaters was assumed to be included in the PMS mass estimate.

4.5.4 Propulsion and Propellant Management Design and MEL

Table 4.8 lists the line items in the Propulsion and Propellant Management MEL including the propellant. The MEL was created in a generic fashion for the all-chemical propulsion systems, but did include line items for the EP subsystems. Those systems are hidden in the table view in Table 4.6. *The growth percentage (growth mass/total mass) is only tracked and calculated on the line items*. Refer to the growth (kg) totals for details. The total growth percentage of the wet mass calculations needs to be removed to prevent confusion in future versions of the MEL tool. The total growth percentage using the wet mass number in the percentage calculation gives a deceptive percentage number. Typically growth percentage is calculated growth divided by total current best estimate (CBE) mass.

TABLE 4.8—PROPULSION AND PROPELLANT MANAGEMENT MEL

WBS	Description	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
01	Robotic Lunar Lander	-	----	429	7	33.3	462
01.01	Lunar Orbit Insertion Stage	-	----	25	15	4.5	29.5
01.02	Lunar Descent Stage	-	----	266	0	1.1	266.8
01.02.02	Propulsion (Chemical)	-	----	27	4	1.1	28.5
01.02.02.01	Main Engine	-	----	27.5	4	1.1	28.5
01.02.02.01.01	Main Engine	1	27.5	27.5	4	1.1	28.5
01.02.02.01.02	Main Engine Gimbal	0	0.0	0.0	4	0.0	0.0
01.02.02.02	Reaction Control System	-	----	0.0	0	0.0	0.0
01.02.02.02.01	RCS Engine	0	0.0	0.0	4	0.0	0.0
01.02.03	Propellant Management (Chemical)	-	----	0	0	0.0	0.0
01.02.04	Propellant Hardware (Chemical)	-	----	238	0	0.0	238.3
01.02.04.01	Main Engine Propellant	-	----	238	0	0.0	238.3
01.02.04.01.01	Fuel	-	----	238.3	0	0.0	238.3
01.02.04.01.02	Oxidizer	-	----	0.0	0	0.0	0.0
01.02.04.01.03	Pressurant	0	0.0	0.0	0	0.0	0.0
01.02.04.02	RCS Propellant	-	----	0.0	0	0.0	0.0
01.03	Lunar Lander Stage	-	----	138	16.7	27.7	165.7
01.03.02	Propulsion Hardware (Chemical)	-	----	5	5	0.3	5.6
01.03.02.01	Main Engine	-	----	3.3	4	0.1	3.5
01.03.02.01.01	Main Engine	3	0.8	2.3	4	0.1	2.4
01.03.02.01.02	Main Engine Gimbal	1	1.0	1.0	4	0.0	1.0
01.03.02.02	Reaction Control System	-	----	2.0	7	0.2	2.1
01.03.02.02.01	RCS Engine	6	0.3	2.0	8	0.2	2.1
01.03.03	Propellant Management (Chemical)	-	----	9	6	0.5	9.6
01.03.03.01	OMS Propellant Management	-	----	8	5	0.4	8.3
01.03.03.01.01	Fuel Tanks	1	6.0	6.0	4	0.2	6.3
01.03.03.01.02	Fuel Lines	1	0.3	0.3	18	0.1	0.3
01.03.03.01.03	Oxidizer Tanks	0	0.0	0.0	0	0.0	0.0
01.03.03.01.04	Oxidizer Lines	0	0.0	0.0	0	0.0	0.0
01.03.03.01.05	Pressurization System - tanks, panels, lines	0	0.0	0.0	0	0.0	0.0
01.03.03.01.06	Feed System - regulators, valves, etc	1	1.6	1.6	8	0.1	1.7
01.03.03.02	RCS Propellant Management	-	----	1.2	9	0.1	1.3
01.03.03.02.01	Fuel Tanks	0	0.0	0.0	0	0.0	0.0
01.03.03.02.02	Fuel Lines	1	0.3	0.3	18	0.1	0.3
01.03.03.02.03	Pressurization System - tanks, panels, lines	1	0.0	0.0	0	0.0	0.0
01.03.03.02.04	Feed System - regulators, valves, etc	1	0.9	0.9	8	0.1	1.0
01.03.04	Propellant (Chemical)	-	----	30	20	7.6	37.8
01.03.04.01	Main Engine Propellant	-	----	26	20	6.5	32.3
01.03.04.01.01	Fuel	-	----	25.8	20	6.5	32.3
01.03.04.01.01.01	Fuel Usable	1	25.8	25.8	25	6.5	32.3
01.03.04.01.01.02	Fuel Boiloff	0	0.0	0.0	0	0.0	0.0
01.03.04.01.01.03	Fuel Residuals (Unused)	1	0.0	0.0	0	0.0	0.0
01.03.04.01.02	Oxidizer	-	----	0.0	0	0.0	0.0
01.03.04.01.03	Pressurant	0	0.0	0.0	0	0.0	0.0
01.03.04.02	RCS Propellant	-	----	4.4	20	1.1	5.5
01.03.04.02.01	Fuel	-	----	4.4	20	1.1	5.5
01.03.04.02.01.01	Fuel Usable	1	4.4	4.4	25	1.1	5.5
01.03.04.02.01.02	Fuel Boiloff	0	0.0	0.0	0	0.0	0.0
01.03.04.02.01.03	Fuel Residuals (Unused)	1	0.0	0.0	0	0.0	0.0
01.03.04.02.02	Oxidizer	-	----	0.0	0	0.0	0.0
01.03.04.02.03	Pressurant	0	0.0	0.0	0	0.0	0.0

4.5.5 Propulsion and Propellant Management Trades

Three design trades for the propulsion subsystem were performed over the course of this study.

- Trade 1: Solid Rocket Motor Selection
  - Motor choices limited to selecting OTS motors
    - Advanced technology readiness and demonstrated heritage of these systems were required to ensure relatively low cost as well as availability
    - Mono- and bipropellant liquid rocket engine alternatives were examined but rejected because:
      - Propulsion subsystems heavier, much more complicated
      - Technology availability in the near-term
- Trade 2: Lander Rocket Engine Selection
  - Hydrazine-based engines were selected because
    - Technology mature and has extensive demonstrated heritage for similar applications
    - Engine class considered OTS (i.e., HAPS derivative)
    - Propellant management system less complex
  - Liquid propellant engine necessary for throttling performance required for landing on lunar surface
    - Cost of modification needs to be done
- Trade 3: Use of Electric Propulsion Insertion Stage (where applicable)
  - Electric thruster selection driven by lifetime requirements
    - HiVHAC thruster capability > 150 kg Xe throughput
    - SP-100 Hall thruster also considered because of match with performance requirements
      - Limited benefits because lifetime capability ~80 kg throughput so multiple thrusters will be needed

4.5.6 Propulsion and Propellant Management Risk Inputs

The only risk identified for this study was the availability of the electric propulsion technology and its associated costs.

4.5.7 Propulsion and Propellant Management Recommendation

The chemical propulsion system option for trans-lunar injection was chosen. Electric propulsion will add capability but at the cost of a baseline vehicle mass that is at the limit of the Minotaur class launch vehicle capability. Additionally, the primary benefits of the electric propulsion technology are only realized when hardware reuse is required to enable or enhance mission return. One example of such a benefit is that the EP stage could remain in orbit as communications relay satellite, thereby extending the operational capability of the landing vehicle.

4.6 Thermal Control

4.6.1 Thermal Requirements

The thermal system must preserve acceptable temperature ranges on all systems from pad to launch to transit to landing to end of mission on the lunar surface. Each of these environments drives different parts of the thermal design, however the lunar surface environment was chosen as the driving case.

4.6.2 Thermal Assumptions

- Radiator panels are fixed (nondeployable) and located on the ascent vehicle.
- Radiators: Radiate heat to space. Low absorptive (a) and high emissivity (e) coatings are desirable to minimize solar input and maximize heat rejection to space.
- Various absorptivity (a) and emissivity (e) coatings are desirable for different locations within Lander.

- Heat Sinks: Placed in thermal contact with high dissipation components. Combined with radiators to dissipate the heat, which they conduct. Could use the structure of the lander itself as a heat sink.
- Heaters: Provide heating for temperature sensitive components (fuel, battery, etc.)
- MLI: blankets with low absorptivity and high emissivity. Reduce the heat flow rate of the system while preventing large heat flux. They can be used to wrap around sensors and payloads for thermal insulation and to reduce thermal requirements.
- Temperature Sensors: Resistive type thermal sensors. Monitor temperature of sensitive components. (Fuel, fuel line, batteries, etc.)

4.6.3 Thermal Design and MEL

The thermal design, shown in the Figure 4.11 schematic, for dissipating heat in the following major subsystem areas consisted of the following assumptions and design details:

Electronics:

- 15 W dissipation
- 270 K lunar sink
- 40 °C mounting plate maximum temperature
- Requires 0.085 m^2 radiator (45- by 19-cm) on the top cover. Box cover is 30- by 10-cm.
- Painted with Z-93 (e = 0.9, a = 0.2)

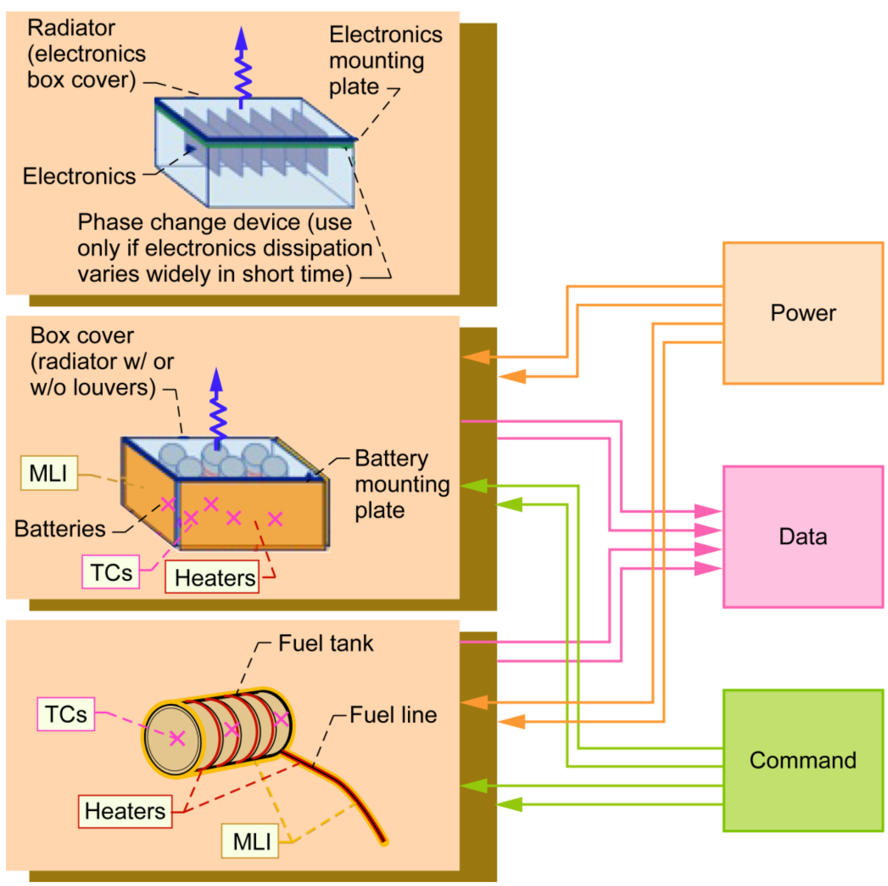


Figure 4.11—Thermal control schematic.

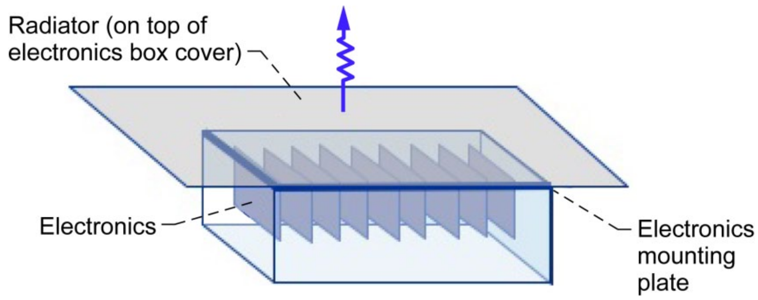


Figure 4.12—Thermal control for electronics.

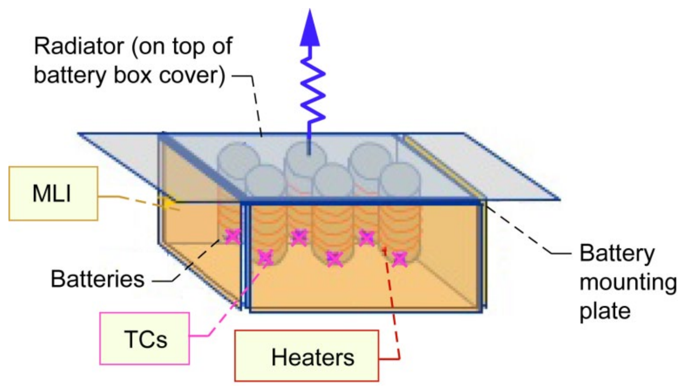


Figure 4.13—Thermal control for batteries.

Figure 4.12 shows a notional diagram of the thermal system designed to handle the heat of the electronics.

Batteries:

- 3 W dissipation
- 270 K lunar sink
- 20 °C mounting plate maximum temperature
- Requires 0.035 m^2 radiator (15- by 24-cm) on the top cover. Box cover is 15- by 15-cm.
- Painted with Z-93 (e = 0.9, a = 0.2)
- Heaters and MLI

Figure 4.13 shows a notional diagram of the thermal system designed to handle the heat of the batteries. Avionics (total of three boxes):

- 270 K lunar sink
- 40 °C mounting plate maximum temperature
- Radiators painted with Z-93 (e = 0.9, a = 0.2)
- RF assembly
  - 0.95 W dissipated
  - Requires 0.005 m^2 radiator. 10- by 8-cm top cover is sufficient.
- Transponder
  - 0.95 W dissipated
  - Requires 0.005 m^2 radiator. 20- by 16-cm top cover is sufficient.
- General avionics
  - 0.95 W dissipated
  - Requires 0.005 m^2 radiator. 13.35- by 11.4-cm top cover is sufficient.

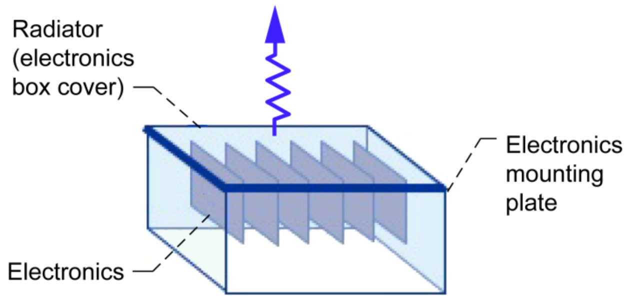


Figure 4.14—Thermal control for avionics.

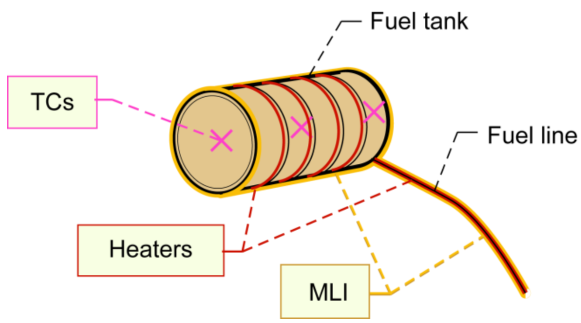


Figure 4.15—Thermal control for propulsion system.

Figure 4.14 shows a notional diagram of the thermal system designed to handle the heat of the avionics. Solid rocket motor, hydrazine fuel tank, and hydrazine fuel lines:

- Allowable temperature range: 7 to 35 °C
- 270 K lunar sink
- Heaters and MLI
- Hydrazine fuel: ~45 kg, tank is 53.3 cm (21 in.) diameter made of titanium.
- Solid Rocket: STAR 27, 238 kg total mass, roughly a 61-cm-diameter ball made of titanium.

Figure 4.15 shows a notional diagram of the thermal system designed to handle the heat of the propulsion system.

Table 4.9 lists the line items in the thermal control system MEL. These numbers are sent to the main system MEL for summary.

TABLE 4.9—THERMAL CONTROL SYSTEM MEL

WBS	Description	Qty	Unit Mass (kg)	CBE Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
01	Robotic Lunar Lander	---	--------	429	7	33.3	462
01.01	LOI Stage	---	--------	25	15	4.5	29.5
01.02	Lunar Descent Stage	---	--------	266	0	1.1	266.8
01.03	Lunar Lander Stage	---	--------	138	16.7	27.7	165.7
01.03.10	Thermal Control (Non-Propellant)	---	--------	18	15	3.3	21.7
01.03.10.01	Active Thermal Control	---	--------	9.5	15	1.7	11.2
01.03.10.01.01	Heaters	31	0.3	9.0	18	1.6	10.6
01.03.10.01.02	Thermal Control/Heaters Circuit	0	0.0	0.0	18	0.0	0.0
01.03.10.01.03	Data Acquisition	0	0.0	0.0	18	0.0	0.0
01.03.10.01.04	Thermocouples	50	0.0100	0.5	18	0.1	0.6
01.03.10.02	Passive Thermal Control	---	--------	8.9	15	1.6	10.5
01.03.10.02.01	Heat Sinks	1	0.3	0.3	18	0.1	0.3
01.03.10.02.02	Heat Pipes	0	0.0	0.0	18	0.0	0.0
01.03.10.02.03	Radiators	1	0.8	0.8	18	0.2	1.0
01.03.10.02.04	MLI	4	1.7	6.7	18	1.2	8.0
01.03.10.02.05	Temperature sensors	0	0.0	0.0	18	0.0	0.0
01.03.10.02.06	Phase Change Devices	0	0.0	0.0	18	0.0	0.0
01.03.10.02.07	Thermal Coatings/Paint	1	1.0	1.0	18	0.2	1.2
01.03.10.03	Semi-Passive Thermal Control	---	--------	0.0	0	0.0	0.0

4.6.4 Thermal Trades

No top-level trades were performed during this study. Additional ones on the value of using different thermal schemes are to be done in the future.

4.6.5 Thermal Analytical Methods

Microsoft Excel (Microsoft Corporation) based internal tools and C&R Technologies “Thermal Desktop” software were used in the analysis. The Thermal Desktop model used the simple model of the internal components shown in Figure 4.16. Figure 4.17 shows the results of applying the assumptions above in Section 4.6.5 to the Thermal Desktop simplified model.

Table 4.10 shows the results of the thermal desktop modeling through a snapshot of the Thermal Desktop modeling tool.

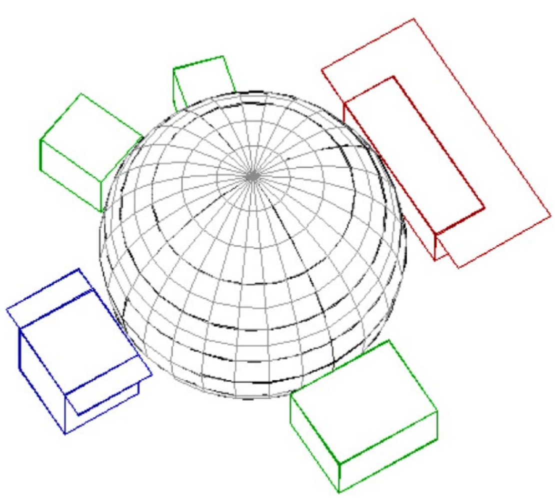


Figure 4.16—Thermal desktop simplified model.

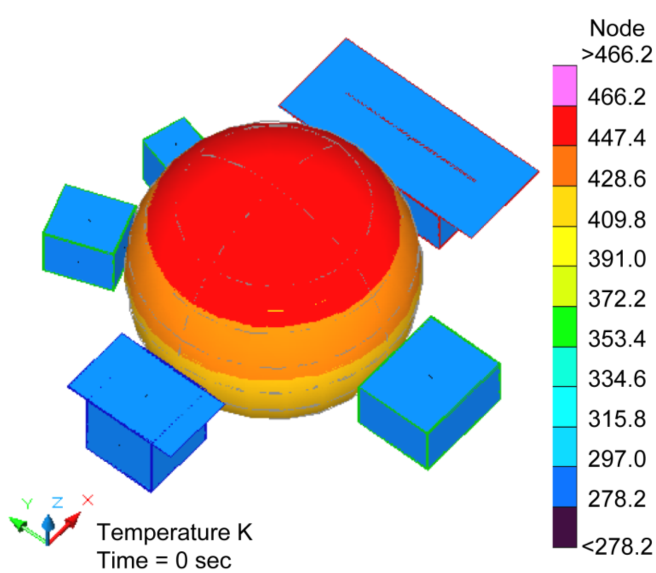


Figure 4.17—Thermal desktop simplified model analysis results.

TABLE 4.10—THERMAL CONTROL SYSTEM THERMAL DESKTOP MODELING

Radiators	Input Q, (W)	Input e, (ε)	Input Fin, (eff.)	Input σ, (W/m²K)	Input Ts, (K)	Input Tb, (C)	Box X, (cm)	Box Y, (cm)	Box Z, (cm)	Tb, (K)	(Tb^4–Ts^4)	Output Area, (m²)	Input Radiator X, (m)	Output Radiator Y, (m)	Input Radiator Z, (m)	Input Radiator density, (kg/m³)	Output Radiator mass, (kg)
Batteries	3	0.9	80%	5.67×10^{-8}	270	20	15	15	15	293.15	2.07×10^{9}	0.035	0.150	0.237	0.003	2770	0.295
Electronics	15	0.9	80%	5.67×10^{-8}	270	40	30	10	10	313.15	4.30×10^{9}	0.085	0.450	0.190	0.003	2770	0.710
RF assembly	0.95	0.9	80%	5.67×10^{-8}	270	40	10	8	12	313.15	4.30×10^{9}	0.005	0.100	0.054	0.003	2770	0.045
Transponder	0.95	0.9	80%	5.67×10^{-8}	270	40	20	16	11	313.15	4.30×10^{9}	0.005	0.200	0.027	0.003	2770	0.045
General avionics	0.95	0.9	80%	5.67×10^{-8}	270	40	13.35	11.4	11.4	313.15	4.30×10^{9}	0.005	0.134	0.041	0.003	2770	0.045
																	1.140

Heaters	Qty	Unit mass, (kg)	Total mass, (kg)	Notes
Fuel tank	1	5.0	5	Common Lunar Lander used 42 kg for 900 kg of fuel. For our 50 kg of fuel we need ~42/900*50~2.4*FS of 2~5 kg
Fuel line	20	0.1	2	JIMO estimate
Battery	10	0.2	2	
Totals	31	0.3	9	

MLI	Qty	Mass (kg)	Ti density (kg/m³)	Ti Cp (J/kg K)	Volume (m³)	Diam (cm)	Length (m)	Wall thickness (m)	Surface area (m²)	MLI layers	MLI specific weight (kg/m²)	MLI unit mass (kg)	Total MLI mass (kg)
Hydrazine fuel tank (sphere)	1	45	4500	520	0.0100	53.34			0.89383	100	2.856		2.55278
Solid rocket: Star 24C (sphere)	1	238	4500	520	0.0529	63.5			1.26677	50	1.428	1.8	1.80895
Hydrazine fuel lines	1	3.53	4500	520	0.0008	1	5	0.005	0.15708	30	0.8568	0.1	0.13459
Structure	1					5	10		1.5708	50	1.428	2.2	2.24310
Totals	4											1.7	6.7

MLI specific weight (kg/m²), JPL 10- by 10-cm, 15 layer sample measured, weight = 4.283 grams.

Heater power	Input Q, (W)	Input e, (ε)	Input n, (eff.)	Input σ, (W/m²K)	Input Ts, (K)	Input Tb, (C)	Tb, (K)	(Tb^4–Ts^4)	Output Area, (m²)	Output Radiator X, (m)	Output Radiator Y, (m)	Input Radiator Z, (m)	Input Radiator density, (kg/m³)	Output Radiator mass, (kg)
Batteries	3	0.9	80%	5.67×10^{-8}	4	–60	213.150	2.06×10^{9}	0.036	0.150	0.237	0.003	2770	0.295
Batteries + heaters	8.66	0.9	80%	5.67×10^{-8}	4	5	278.150	5.99×10^{9}	0.035	0.188	0.188	0.003	2770	0.295
Battery heater power	5.66													
Electronics	15	0.9	80%	5.67×10^{-8}	4	–17	256.150	4.31×10^{9}	0.085	0.292	0.292	0.003	2770	0.709
Electronics +heaters	16.7	0.9	80%	5.67×10^{-8}	4	–10	263.150	4.80×10^{9}	0.085	0.292	0.292	0.003	2770	0.709
Electronics heater power	1.7													
Avionics	3.8	0.9	80%	5.67×10^{-8}	4	–17	256.150	4.31×10^{9}	0.022	0.147	0.147	0.003	2770	0.180
Avionics + heaters	4.3	0.9	80%	5.67×10^{-8}	4	–10	263.150	4.80×10^{9}	0.022	0.148	0.148	0.003	2770	0.183
Avionics heater power	0.5													
										T initial (C)	Time (s)	Q req for heat up (W)		
[a]Hydrazine fuel tank	15.01	0.1	80%	5.67×10^{-8}	270	35	308.150	3.70×10^{9}	0.8938	20	36000	9.75		
Hydrazine fuel tank + heaters	24.97	0.1	80%	5.67×10^{-8}	4	7	280.150	6.16×10^{9}	0.8938					
Hydrazine fuel tank heater power	9.96													
[a]Fuel lines	2.64	0.1	80%	5.67×10^{-8}	270	36	308.150	3.70×10^{9}	0.1571	7	36000	1.43		
Fuel lines + heaters	4.39	0.1	80%	5.67×10^{-8}	4	7	280.150	6.16×10^{9}	0.1571					
Fuel lines heater power	1.75													

[a]Assume allowable temperature ranges of 7 to 35 °C
Q = power dissipated
A = radiator area
e = emissivity of box top surface
n = fin effectiveness of box top surface
σ = Stefan-Boltz constant
Tb = temperature of box top surface
Ts = Moon sink temperature for horizontal surface (radiator at equator and angle of the sun above the horizon is 90°)
$A = Q/(e*n*\sigma*(Tb^4 – Ts^4))$
Assume 40 °C Tb for electronics
Assume 20 °C Tb for batteries
Assume horizontal radiators at lunar equator

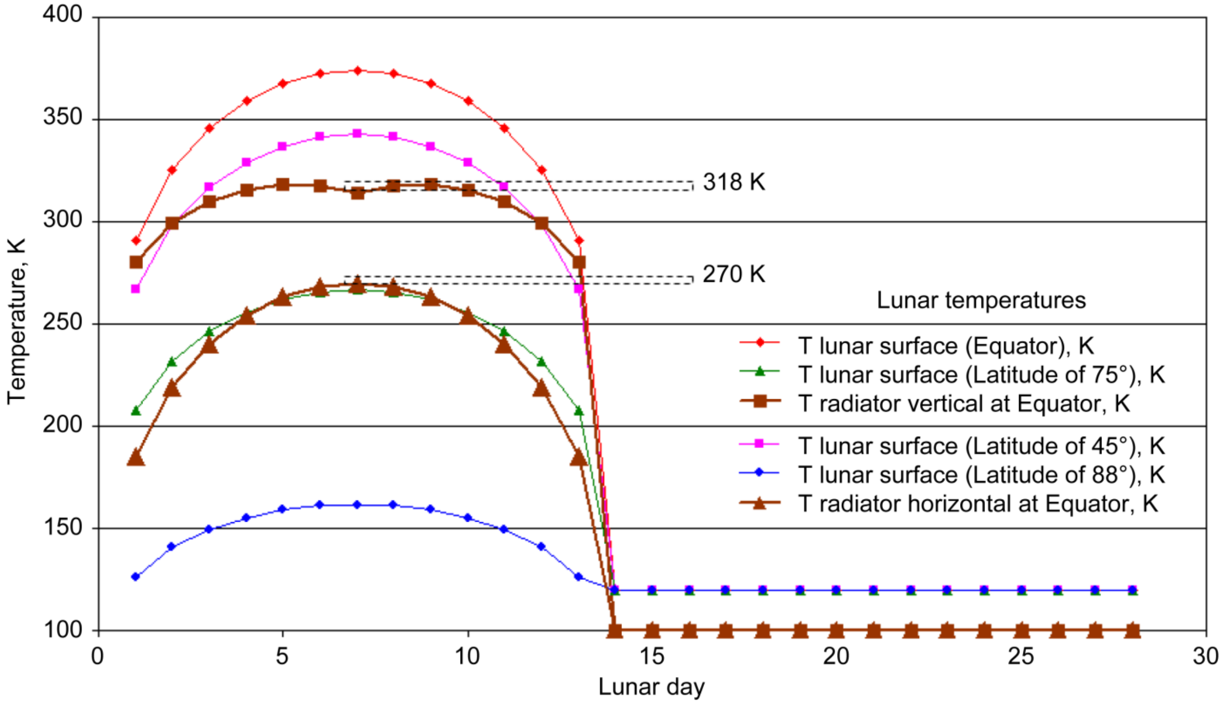


Figure 4.18—Thermal desktop simplified model.

Figure 4.18 shows a graphical representation of the thermal environment modeled over a lunar day/night cycle.

4.6.6 Thermal Risk Inputs

To be completed in the future.

4.6.7 Thermal Recommendation

None.

5.0 Cost, Risk and Reliability

5.1 Costing: Baseline Chemical Lunar Lander

5.2 Cost Modeling Assumptions

Cost estimates were based on the Case 1 (baseline) MEL received on September 29, 2006. The scope of the estimates is prime contractor costs for DDT&E and first-unit production. Costs for contractor fee, government support and operations are not included. Launch vehicle costs are not included. Estimates are based on a mixture of subsystem level estimates (Avionics, Thermal Control, and Structure) and estimates developed at the component level (Propulsion and Power). The estimates provided represent the mean of the probability distribution and are in FY07 dollars. A protoflight approach was assumed for development and testing.

5.3 Cost Modeling Approach

The costs were estimated in Microsoft Excel (Microsoft Corporation) using an approach similar to NAFCOM, i.e., the elements of a product-oriented WBS were estimated and the sum of these product costs were used to estimate system integration costs (“wraps”). Estimates generated account for all prime contractor costs; however they do not break out labor and material costs. @Risk v. 4.5.5, a Monte Carlo simulation application that integrates with Excel, was used for probabilistic risk assessment. Wherever possible, the Cost Estimating Relationships (CER) were developed using as many relevant data points as available so the standard errors used to develop the risk model would have a strong statistical basis. All in-house CERs were developed using DataFit v. 8.1.69 with the dual priorities of minimizing standard error and maximizing the probability that the selected independent variables have significant correlation with regards to cost. Table 5.1 shows the Phase C/D Cost Estimates.

TABLE 5.1—PHASE C/D COST ESTIMATES

FY07 $M:	D&D	Flight HW	Total
LOI Stage			(incl. with structures and mechanisms)
Lunar Descent Stage	0.7	0.5	1.2
Avionics, Instrumentation, GN&C	24.4	12.0	34.0
Propulsion & Propellant Management (Chemical)	3.5	1.3	4.9
Power	1.4	2.9	4.3
Thermal Control	3.4	1.5	4.9
Structures and Mechanisms	10.5	5.6	16.1
Subsystem Subtotal	44.0	23.9	65.4
Systems Integration			
Integration, Assembly & Checkout	5.1	1.9	7.0
System Test Operations	2.7		2.7
Ground Support Equipment	3.5		3.5
Systems Engineering & Integration	2.9	1.4	4.2
Project Management	6.1	2.9	9.0
Launch Operations & Orbital Support	4.6		4.6
Systems Integration Subtotal	24.9	6.2	31.1
Total Prime Contractor	68.9	30.0	99.0

6.0 Trade Space Iterations

6.1 Case 1: Off-the-Shelf Chemical Propulsion

The baseline case (case 1) used all chemical propulsion systems to perform burns during the mission. The two trades done on the baseline involved the application of electric propulsion thrusters and low thrust mission analysis to the Lunar Lander problem. See Table 3.3 for the Case 1 system summary chart.

6.2 Case 2: Off-the-Shelf Electric Propulsion

This design used OTS Electric propulsion as the lunar propulsion system. The mission is launched on a Minotaur V to TLI, uses SEP to spiral to LLO, and a Solid/Monoprop propulsion system on the Lander. Unfortunately, this trade yielded no useable payload to the lunar surface. See the summary in Table 6.1.

TABLE 6.1—CASE 2 SYSTEM SUMMARY CHART

Case 2—Robotic Lander Stage masses (kg)	Current Values with growth margin (kg)			Totals	Percent of dry mass
Subsystems	EDS stage	Descent stage (solid)	Lander stage (mono-p)	Total Robotic Lander	Total Robotic Lander
Avionics	4.29	0.00	36.79	41	11
Propulsion (chemical)	0	14.37	5.61	20	5
Prop Management (chemical)	0	0.00	9.20	9	3
Propellant (chemical)	0	105.72	29.79	136	N/A
Propulsion (EP)	37.51	0.00	0	38	10
Prop Management (EP)	21.60	0	0	22	6
Propellant (EP)	142.49	0	0	142	N/A
Power Processing (EP)	0	0	0	0	0
Power	88.77	0	30.85	120	33
Thermal Control	26	0	21.69	48	13
Structure	52.38	0	17.69	70	19
Total Mass (Wet)	373	120	152	645	N/A
Lander Total Wet Mass	645				
Lander Total Dry mass (kg)	367	110			
Lander Total Inert Mass (kg)	391				
Payload (left over)	–6				

6.3 Case 3: Advanced Direct Drive Electric Propulsion

The summary of the Case 3 design components is as follows: “Advanced” Direct-Drive Electric Propulsion, Minotaur V to TLI, and spiral to LLO, solid/monoprop lander. The case yielded a 1-yr transfer to the Moon and 27 kg useable payload on lunar surface.

In an attempt to make EP work, mass needs to be removed from the EP stage. Some options to reduce mass are as follows:

- Utilize ‘direct-drive’ technology
  - 500 V solar array directly fed to Hall thruster(s)
  - Thruster ancillaries fed by lander 28 V power
  - Eliminates power converters, and most of Power Processing Units (PPUs), most of radiators, (>60 kg savings)
- Start in Geostationary Transfer Orbit (GTO) off an Minotaur V (case transportable to secondary payload consideration)
  - More starting mass (710 kg)
  - Faster trip time
  - Less Radiation on spacecraft and array
  - Similar to SMART-1 trajectory

Table 6.2 is the system summary results of the Case 3 design using direct drive EP.

TABLE 6.2—CASE 3 SYSTEM SUMMARY CHART

Case 3—Robotic Lander Stage Masses (kg)	Current values with growth margin (kg)			Totals	Percent of dry mass
Subsystems	EDS stage	Descent stage (solid)	Lander stage (mono-p)	Total Robotic Lander	Total Robotic Lander
Avionics	0	0.00	44.45	44	15
Propulsion (chemical)	0	19.06	5.61	25	9
Prop Management (chemical)	0	0.00	9.20	9	3
Propellant (chemical)	0	147.81	36.65	184	N/A
Propulsion (EP)	15.40	0.00	0	15	5
Prop Management (EP)	21.60	0	0	22	7
Propellant (EP)	139.57	0	0	140	N/A
Power Processing (EP)	0	0	0	0	0
Power	47.71	0	33.73	81	28
Thermal Control	6	0	21.69	28	10
Structure	45.13	0	18.56	64	22
Total Mass (Wet)	275	167	170	612	N/A
Lander Total Wet Mass	612				
Lander Total Dry mass (kg)	288	86			
Lander Total Inert Mass (kg)	314				
Payload (left over)	26.9				

The mission timeline for Case 3 is as follows:

- Minotaur V to GTO
- EP Spiral from GTO to LLO (a.k.a. SMART-1)
- Direct drive Hall thruster system to minimize mass
- Solid rocket motor for descent
- Hydrazine propulsion system for final landing

This provides for the delivery of a science payload onto the lunar surface capable of 1 week operation, landing in a sunlit area on the lunar surface, Global Access, 25 W for payload on lunar surface. The Case 3 vehicle consisted of three stages:

Stage 1: Chemical Lander Stage (Similar to the Case 1 Chemical Lander)

- High Gain Antenna
- 125 W GaAs Solar Array (two-axis pointing)
- Blow-down monoprop system for final landing and pointing
- Avionics on top for improved cooling
- Fixed leg landing system using Doppler radar (antennas on feet)

Stage 2: Descent Stage (Similar to the Case 1 Chemical Lander)

- Star 24 Solid Rocket (30% offload) (different motor than case 1 but same class, disposed before landing)

Stage 3: EP Stage (This Replaces the TLI Performed by the Minotaur V in Case1)

- 3 kW Ultraflex Solar Array
- Xe Tanks (130 kg Xe)
- HiVHAC Hall Thrusters
- Launch vehicle adapter re-used for EP stage structure

Figure 6.1 shows the stacked version of the Case 3 vehicle with the Ultraflex solar array in stowed position. Figure 6.2 shows the Case 3 stack rendered with the Ultraflex solar array deployed.

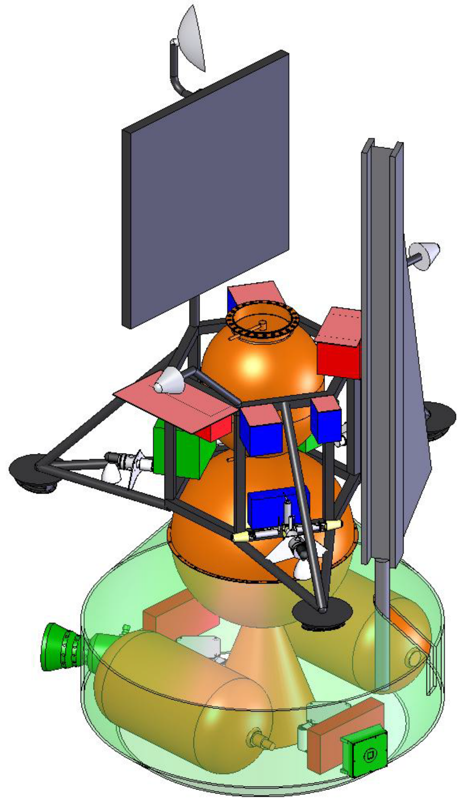


Figure 6.1—Case 3 Robotic Lunar Lander design.

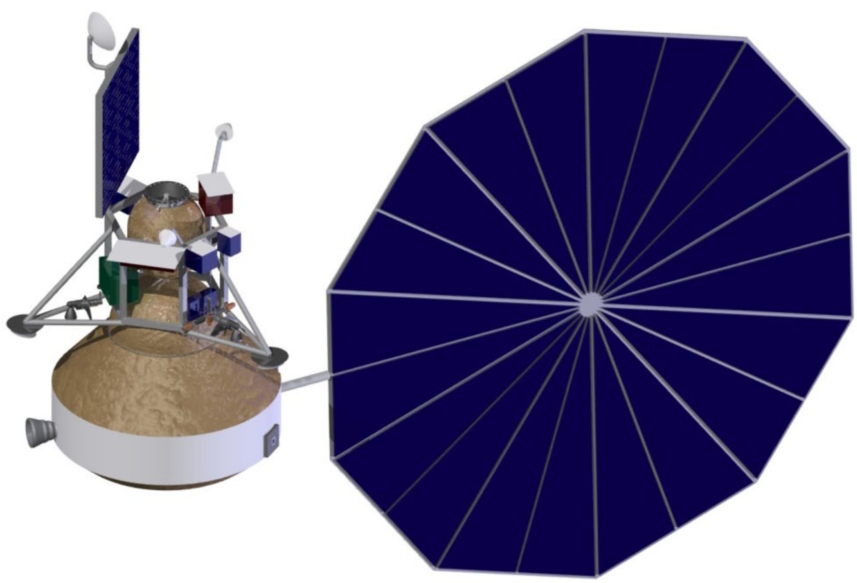


Figure 6.2—Case 3 Robotic Lander, solar panel for EP system deployed.

EP system better choice when system can be reused after lunar arrival as a Relay/com/nav station, for additional power or as a station keeping propulsion system. Very low lunar orbit observer station-keeping. Further collaboration with Relay Sat study and ARC planned.

Appendix A.—Acronyms and Abbreviations

ARC	Ames Research Center
C&DH	Command and Data Handing
CAD	computer aided design
CBE	current best estimate
CER	Cost Estimating Relationship
CEV	Crew Exploration Vehicle
Comm	Communications
COMPASS	COllaborative Modeling and Parametric Assessment of Space Systems
COPV	Carbon-Overwrapped Pressure Vessels
COTS	Commercial off the Shelf
DMR	Design for Minimum Risk
DSN	Deep Space Network
DTE	direct to Earth
DDT&E	Design Development Testing and Engineering
EELV	Evolved Expendable Launch Vehicle
EP	Electric Propulsion
FEA	finite element analysis
FEM	Finite Element Model
FOM	figure of merit
GLIDE	GLobal Integrated Design Environment
GN&C	Guidance, Navigation and Control
GRC	NASA Glenn Research Center
GTO	Geostationary Transfer Orbit
HAPS	Hydrazine Auxiliary Propellant System
HQ	NASA Headquarters
HiVHAC	High Voltage Hall ACcelerator)
IP	internet protocol
LAT	Lunar Architecture Team
LEO	low Earth orbit
LLO	low lunar orbit
LOI	Lunar Orbit Insertion
LPRP	Lunar Precursor and Robotic Program
LSP	Launch Service Program
LSTO	Launch Service Task Order
MAC	Media Access Control
MEL	Master Equipment List
MLI	Multilayer Insulation
MPU	Makeup Power Unit
NASA	National Aeronautics and Space Administration
Nav	navigation
NLS	NASA Launch Services
OTS	Off the Shelf
PEL	Power Equipment List
PMS	Propellant Management System
PN	pseudo-noise
PPU	Power Processing Units
RF	radio frequency
RCS	Reaction Control System
S/C	spacecraft
SADA	Solar Array Drive Assembly
SEP	Solar Electric Propulsion
SN	signal-to-noise
SPU	Solar Power Unit
TDRSS	Tracking and Data Relay Satellite System
TLI	trans-lunar injection
TOF	time of flight
TWTA	Traveling Wave Tube Amplifier
WSB	weak stability boundary

Appendix B.—Case 1 Rendered Design Drawings

Figure B.1 shows the total Case 1 design with Star 47 motor, adaptor, and lander stack. Figure B.2 shows the bottom view of the total Case 1 design with Star 47 motor, adaptor, and lander stack as looking up the Star 47 Rocket motor.

Figure B.1—Case 1 Robotic Lander, stack rendered.

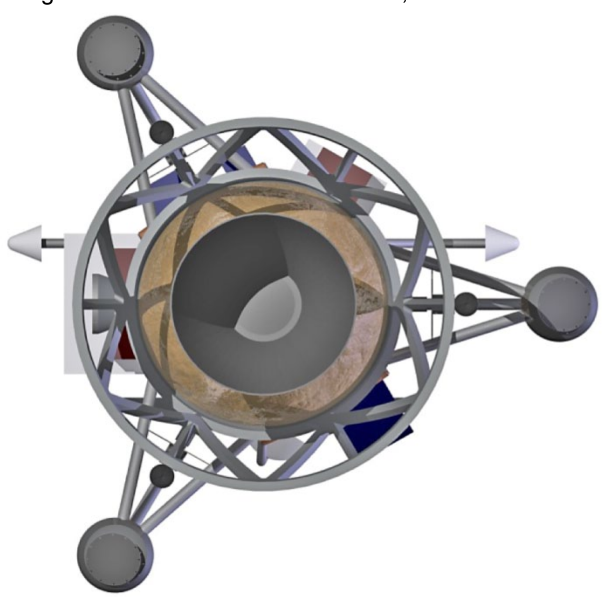


Figure B.2—Case 1 Robotic Lander, bottom view.

Figure B.3 is the landed portion of the Robotic Lunar Lander as it would appear on the Moon's surface with the solar array and antenna deployed.

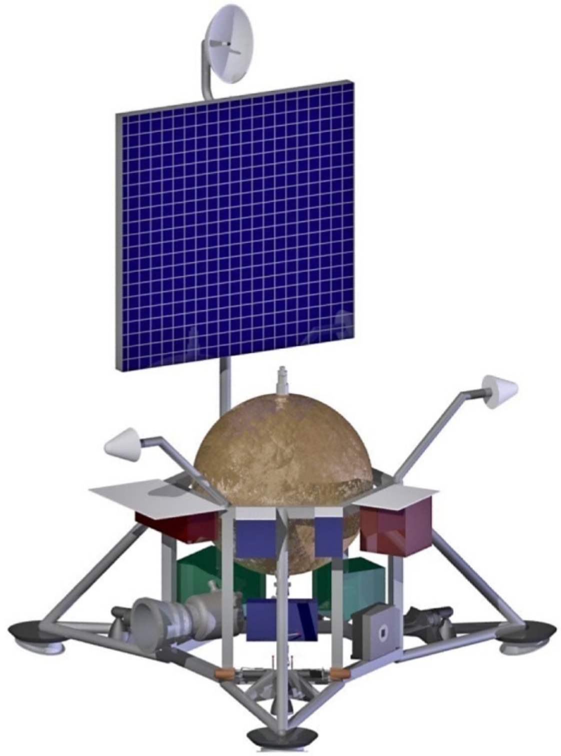


Figure B.3—Case 1 landed portion of Robotic Lunar Lander.

Appendix C.—Study Participants

Low Cost Robotic Lunar Lander Design Session			
Subsystem	Name	Center	Email
Customer	ARC		
Customer	Lunar Architecture Team (LAT)	HQ	
Lead	Steve Oleson	GRC	Steven.R.Oleson@nasa.gov
Mission Design	Doug Fiehler	GRC	Douglas.I.Fiehler@nasa.gov
Mission Design	Craig Kluever	UMo	KlueverC@missouri.edu
Concept Design Integration Lead, MEL and Final Report Documentation	Melissa McGuire	GRC	Melissa.L.Mcguire@nasa.gov
ELV, Integration and Test, Operations	Jeff Woytach	GRC	Jeffery.M.Woytach@nasa.gov
System Integration	Dave Grantier	GRC	David.T.Grantier@nasa.gov
Structures and Mechanisms	Nelson Morales	GRC	Nelson.Morales-1@nasa.gov
Structures and Mechanisms	John Gyekenyesi	GRC	John.P.Gyekenyesi@nasa.gov
Thermal	John Siamidis	GRC	John.Siamidis-1@nasa.gov
Power	James Fincannon	GRC	Homer.J.Fincannon@nasa.gov
Communications and Avionics	Rich Oeftering	GRC	Richard.C.Oeftering@nasa.gov
Propulsion	Tim Sarver-Verhey	GRC	Timothy.R.Verhey@nasa.gov
Guidance, Navigation and Control	Jeffrey Lee	GRC	Jeffrey.R.Lee@nasa.gov
Configuration	Tom Packard	GRC	Thomas.W.Packard@nasa.gov
Cost	Tom Parkey	GRC	Thomas.J.Parkey@nasa.gov
Risk/Reliability	Anita Tenteris	GRC	Anita.D.Tenteris@nasa.gov
Risk/Reliability	Bill Strack	GRC	bstrack@wowway.com

Bibliography

Aerojet engines and propellant systems (http://www.aerojet.com/capabilities/spacecraft.php)
Arde Inc tank Catalogs: http://www.ardeinc.com
Astronautix Web site: http://www.astronautix.com
ATK PSI tank catalogs: http://www.psi-pci.com
Beagle 2 official website: http://beagle2.open.ac.uk/index.htm
Belbruno, E. A., and J. P. Carrico (2000), Calculation of Weak Stability Boundary Ballistic Lunar Transfer Trajectories, AIAA/AAS Astrodynamics Specialist Conference, American Institute of Aeronautics and Astronautics, Denver, CO, abstract AIAA 2000-4142, August 14-17.
Orbital Science Corporation, Minotuar V Fact Sheet: http://www.orbital.com/NewsInfo/Publications/Minotaur_V_Fact.pdf.
Parker, J.S. "Families of Low Energy Lunar Halo Transfers," AAS/AIAA Spaceflight Dynamics Conference, Paper No. AAS 06-132, Tampa, FL, January 22-26, 2006.
Recommended Practice for Mass Properties Control for Satellites, Missiles, and Launch Vehicles, ANSI/AIAA R-020A-1999, 1999.
Space Mission Analysis and Design, 3rd edition (Space Technology Library), Wiley J. Larson and James R. Wertz (eds.), October 1999.
Star Motor reference pages from ATK. (http://www.atk.com/starmotors/starmotors_star48v.asp)
Surveyor Web site: http://www.lpi.usra.edu/expmoon/surveyor/surveyor.html.

REPORT DOCUMENTATION PAGE					*Form Approved* *OMB No. 0704-0188*

The public reporting burden for this collection of information is estimated to average 1 hour per response, including the time for reviewing instructions, searching existing data sources, gathering and maintaining the data needed, and completing and reviewing the collection of information. Send comments regarding this burden estimate or any other aspect of this collection of information, including suggestions for reducing this burden, to Department of Defense, Washington Headquarters Services, Directorate for Information Operations and Reports (0704-0188), 1215 Jefferson Davis Highway, Suite 1204, Arlington, VA 22202-4302. Respondents should be aware that notwithstanding any other provision of law, no person shall be subject to any penalty for failing to comply with a collection of information if it does not display a currently valid OMB control number.
PLEASE DO NOT RETURN YOUR FORM TO THE ABOVE ADDRESS.

1. REPORT DATE *(DD-MM-YYYY)* 01-12-2010	**2. REPORT TYPE** Technical Memorandum	**3. DATES COVERED** *(From - To)*
4. TITLE AND SUBTITLE COMPASS Final Report: Low Cost Robotic Lunar Lander		**5a. CONTRACT NUMBER**
		5b. GRANT NUMBER
		5c. PROGRAM ELEMENT NUMBER
6. AUTHOR(S) McGuire, Melissa, L.; Oleson, Steven, R.		**5d. PROJECT NUMBER**
		5e. TASK NUMBER
		5f. WORK UNIT NUMBER WBS 069256.03.09.01.03
7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES) National Aeronautics and Space Administration John H. Glenn Research Center at Lewis Field Cleveland, Ohio 44135-3191		**8. PERFORMING ORGANIZATION REPORT NUMBER** E-17199
9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES) National Aeronautics and Space Administration Washington, DC 20546-0001		**10. SPONSORING/MONITOR'S ACRONYM(S)** NASA
		11. SPONSORING/MONITORING REPORT NUMBER NASA/TM-2010-216218

12. DISTRIBUTION/AVAILABILITY STATEMENT
Unclassified-Unlimited
Subject Categories: 17, 18, and 91
Available electronically at http://gltrs.grc.nasa.gov
This publication is available from the NASA Center for AeroSpace Information, 443-757-5802

13. SUPPLEMENTARY NOTES

14. ABSTRACT
The COllaborative Modeling for the Parametric Assessment of Space Systems (COMPASS) team designed a robotic lunar Lander to deliver an unspecified payload (greater than zero) to the lunar surface for the lowest cost in this 2006 design study. The purpose of the low cost lunar lander design was to investigate how much payload can an inexpensive chemical or Electric Propulsion (EP) system deliver to the Moon's surface. The spacecraft designed as the baseline out of this study was a solar powered robotic lander, launched on a Minotaur V launch vehicle on a direct injection trajectory to the lunar surface. A Star 27 solid rocket motor does lunar capture and performs 88 percent of the descent burn. The Robotic Lunar Lander soft-lands using a hydrazine propulsion system to perform the last 10% of the landing maneuver, leaving the descent at a near zero, but not exactly zero, terminal velocity. This low-cost robotic lander delivers 10 kg of science payload instruments to the lunar surface.

15. SUBJECT TERMS
Low cost; Robotics; Electric propulsion; Chemical propulsion; Lunar surface; Launch vehicles; Solid propellant

16. SECURITY CLASSIFICATION OF:			**17. LIMITATION OF ABSTRACT**	**18. NUMBER OF PAGES**	**19a. NAME OF RESPONSIBLE PERSON** STI Help Desk (email:help@sti.nasa.gov)
a. REPORT U	**b. ABSTRACT** U	**c. THIS PAGE** U	UU	56	**19b. TELEPHONE NUMBER** *(include area code)* 443-757-5802

Standard Form 298 (Rev. 8-98)
Prescribed by ANSI Std. Z39-18