NASA/TP—2019–220391

NASA Lunar Lander Reference Design

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November 2019

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November 2019

Acknowledgments

The authors would like to thank the following people for their valuable contributions to the lunar lander concept development and the development of this Technical Publication:

Bobby Atkins
Naeem Ahmad
Evan Anzalone
Brian Bae
Juan Barragan
Wade Bostick
Darren Boyd
Ellen Braden
Shawn Breeding
Shawn Brechbill
Nathan Brown
Darren Boyd
Adam Burt
Mary Kate Butler
Monty Carroll
Bob Cataldo
Jack Chapman
Greg Chavers
Scott Craig
Kathryn Crowe
Ben Darby
Ken Deily
Gordon Deramus
Melanie Dervan
David Dominguez
Kim Ess
Jeff Farmer
William Foley
Greg Frady
Todd Freestone
Kenneth Goggans
Mike Hannan
Rick Hardy
Kathryn Horton-Mullins
Will Johnson
Valle Kauniste
Logan Kennedy
Tim Kibbey
Eric Lowery
Bebe Ly
Josh Moore
Steve Munday
Terrian Nowden
Nick Olson
Eric Ordonez
Juan Orphee
Mendy Peek
Tim Provin
Matt Pruitt
David Retherford
James Rice
Ed Robertson
Joao Seixal
Andy Singleton
Jim Sledd
George Story
Kevin Sykes
Huu Trinh
Madelyn Vandewalle
Kevin Varcak
Kosta Varnavas
Yvonne Villegas-Aguilera
Sharada Vitalpur
Andy Wayne
Parker Weide
Norma Whitehead
Kathryn Wiedow
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LIST OF ACRONYMS AND ABBREVIATIONS

ACS	attitude control system
AIAA	American Institute of Aeronautics and Astronautics
ARJ	active release joint
CAD	computer-aided design
CFS	core flight software
CG	center of gravity
CH_4	methane
CLV	commercial launch vehicle; crew launch vehicle
CMOS	complementary metal-oxide semiconductor
C/O	check out
Comm	Communications
COPV	composite overwrap pressure vessel
COTS	commercial off the shelf
CPU	central processor unit
CS	coordinate system
CSC	conical-shaped charges
C&T	command and telemetry
CTE	coefficient of thermal expansion
CW	continuous wave
DCD	data and command dictionary

LIST OF ACRONYMS AND ABBREVIATIONS (Continued)

DCS	descent control system
DE	descent engine
DoF	degrees of freedom
DPM	data processing unit
DSN	Deep Space Network
EEE	electrical, electronic, and electromechanical
EGSE	electrical ground support equipment
EMC	electro-mechanical compatibility
EMI	electro-mechanical interference
EoM	equations of motion
EPC	electronic power conditioner
EPS	electrical power system
FETA	flexible explosive transfer assembly
FSW	flight software
GHe	gaseous helium
GLASS	generalized lander simulation in Simulink
GNC	guidance, navigation, and control
GRAIL	Gravity Recovery and Interior Laboratory
GSE	ground support equipment
GSFC	Goddard Space Flight Center
IAU	integrated avionics unit

LIST OF ACRONYMS AND ABBREVIATIONS (Continued)

ICD	interface control documents
IMU	inertial measurement unit
I/O	input/output
I_{sp}	specific impulse
ITOS	Integrated Test and Operations System
JEOD	JSC engineering orbital dynamics
JSC	Johnson Space Center
LCH_4	liquid methane
LEO	low-Earth orbit
Li	lithium
LOS	line of sight
LOX	liquid oxygen
LPL	lunar pallet lander
LSC	linear-shaped charges
LSN	linear-shaped charges
MDP	maximum design pressure
MEOP	maximum expected operating pressure
MGA	medium gain antenna
MLI	multilayer insulation
MMH	monomethyl hydrazine
MON-25	mixed oxides of nitrogen-25

LIST OF ACRONYMS AND ABBREVIATIONS (Continued)

MSFC	Marshall Space Flight Center
MUSTANG	modular unified space technology avionics for next generation missions
NDL	Navigation Doppler LIDAR
NPR	NASA procedural requirement
OME	orbital maneuvering engines
OS	operating system
PDU	power distribution unit
PID	proportional integral derivative
PK	Peacekeeper
PMBT	propellant mean bulk temperature
PRM	payload release mechanism
Prop	propulsion
PSI	plume surface interaction
S&A	safe and arm
SAB	Simulink Aerospace Blockout
SEPIA	separation interface assembly
SLOC	source lines of code
SRM	solid rocket motor
SSW	simulation software
TBD	to be determined
TBR	to be resolved

LIST OF ACRONYMS AND ABBREVIATIONS (Continued)

TCM	trajectory correction maneuver
TCS	thermal control system
TDM	time division multiplexed
ThEL	thermal equipment list
TLI	translunar insertion
TP	Technical Publication
TRL	Technology Readiness Level
TRN	Terrain-Relative Navigation
TSE	Trick simulation environment
TSW	test software
TVC	thrust vector control
TWTA	traveling wave tube amplifier
ULA	United Launch Alliance
VSWR	voltage standing wave ratio

TECHNICAL PUBLICATION

NASA LUNAR LANDER REFERENCE DESIGN

1. BACKGROUND

Over the last few years, in preparation for potential robotic missions to the lunar surface, NASA has performed a number of concept studies to identify and examine technologies needed to get to the Moon. These studies have spawned development efforts in advanced propulsion, navigation and landing, and various other lander subsystems. NASA has also independently developed and terrestrially flown two vertical test beds: Morpheus (led by NASA Johnson Space Center (JSC)) and Mighty Eagle (led by NASA Marshall Space Flight Center (MSFC)). Morpheus and Mighty Eagle demonstrated vehicle-level integration capabilities using different propulsion architectures, navigation and landing systems, ground and flight software, and avionics. The success of these efforts led to the establishment of an integrated, cross-Agency lander community that now supports industry through several funded efforts.

2. PURPOSE

In one of the recent studies, a pallet lander concept (Volatiles Investigating Polar Exploration Rover (VIPER)) was introduced that used a 'pallet lander' to deliver a 300-kg robotic surface mobility system to the polar regions of the Moon equipped with an in situ resource utilization demonstration payload. This lander was designed to minimize cost and schedule, with its mission terminating once the surface payload was delivered. While still emphasizing simplicity and affordability, this lander design has been further evolved to develop a more general use, medium-class-payload lander and to investigate lander features potentially extensible to future human landers. While this design is not complete and lander subsystems are at various levels of maturity, this Technical Publication (TP) summarizes the design status, providing insight into ongoing design activities, trades, and challenges left to resolve.

This TP is intended to support a dialogue with industry on the lander technology investments NASA has made that are pertinent to this type of mission and on potential alternative robotic lunar lander concepts and technologies that might be proposed by industry. The information in this TP is for reference only and is not meant to promote one particular concept over any other. Also, specific hardware is identified in some areas, but this does not indicate selection or preference of a vendor, merely one possible solution that meets the current design approach.

3. INTRODUCTION

With increased emphasis on lunar exploration and scientific investigation, there is a desire to deliver a wide variety of payloads to the lunar surface. Many of these payloads will require the use of surface mobility capability such as a rover. NASA has combined spacecraft and subsystem engineers from across the Agency to develop a 'pallet' lander design intended to deliver and easily deploy a medium-sized payload (~300 kg) to the polar regions of the Moon. The lander provides power to the payload from transit soon after lunar landing. The lander is not intended to survive the lunar night. The design of the lander was based on a minimum set of level 1 requirements where traditional risk, mass, and performance trade parameters were weighed lower than cost. In other words, the team did not sacrifice 'good enough' for 'better' or 'best.' As a NASA class D spacecraft (as defined in NPR 8705.4, Risk Classification for NASA Payloads, the lander employs single-string (i.e., zero-fault-tolerant) systems as a baseline. The design utilizes existing technologies and components where possible, though some enhancements have been targeted in areas such as precision autonomous landing and low-cost structural design/fabrication. It is important to note that these and other derived technologies are extensible to other lander designs and missions.

This TP describes the requirements and approaches upon which the lander design is based; discusses key design decisions, analyses, and trades used to derive the design; provides a snapshot of each major subsystem; and identifies open items, issues, and challenges for which work is continuing.

4. MISSION DESIGN

4.1 System Description

The lander is designed to launch aboard a commercial launch vehicle (CLV) and fit within a standard 5-m fairing with adequate clearances. Table 1 shows a typical CLV mission launch and ascent timeline, and a notional mission timeline is illustrated in figure 1. The CLV contains the upper stage required to place the lander and layload—collectively referred to as the space vehicle—onto the translunar trajectory. The lander has been designed to sustain an envelope of existing CLV load environments. The launch platforms for which the lander is designed include the United Launch Alliance (ULA) Atlas V and the SpaceX Falcon 9. The lander can carry 300 kg of cargo to a location proximal to a permanently shadowed region near either lunar pole (85° to 90° latitude north or south).

Table 1. Mission: Typical CLV launch and ascent timeline.

Phase	Time
Liftoff	0:00:00
First stage main engine cutoff (MECO)	0:02:18
Second stage engine start-1 (SES-1)	0:02:29
Second stage engine cutoff-1 (SECO-1)	0:08:19
SES-2	0:44:21
SECO-2	0:45:10
Second stage separation (SCSEP)	0:49:24

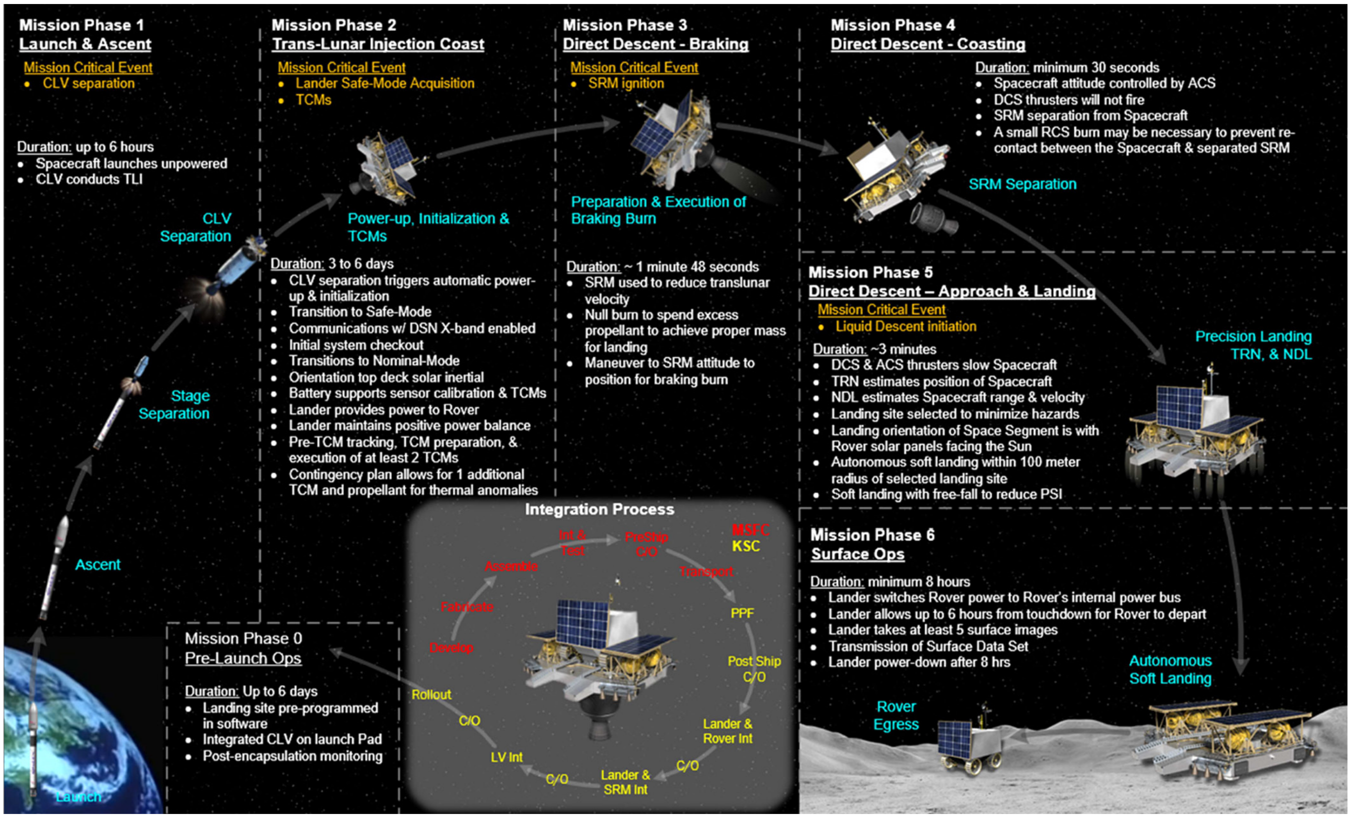

Figure 1. Mission: Notional lander mission timeline.

The CLV upper stage is launched into a short-term Earth parking orbit where it performs the translunar insertion (TLI) burn at the earliest opportunity following ascent.

While the SV launches unpowered, separation from the CLV triggers automatic power-up and initialization of the lander, which in turn provides 150 W of power to the rover. Transit to the lunar environment takes between 3 and 6 days during which time the lander guidance, navigation, and control (GNC) system will point the top of the lander toward the Sun where two solar panels will provide power generation. When pointed away from the Sun for sensor calibration or a trajectory correction maneuver (TCM), batteries will provide the needed power. Upon landing, the lander will provide power to the rover for 15 minutes, after which the rover will be under its own power.

The cruise phase is a 3- to 6-day direct Earth to Moon transfer, during which system checkout and notional payload instrument calibration will be performed in addition to two (nominal) TCMs. Contingency planning allows for one additional TCM.

The direct descent to the lunar surface is divided into three phases: braking, coast, and approach and landing. The braking phase is accomplished using a solid rocket motor (SRM), during which translunar velocity is reduced and the conditions for the approach and landing phase are set up. The coast phase is the transition from braking to approach and landing, during which the SRM separates from the lander. During coast, the attitude will be controlled and the descent engines will not fire. For the approach and landing phase, the lander uses its descent control system (DCS) and attitude control system (ACS) thrusters to further slow the vehicle, take out dispersions, and safely land within the required landing ellipse.

The lander utilizes Terrain-Relative Navigation (TRN) during the approach and landing phase of flight. TRN will be used to estimate the position of the lander relative to the lunar surface. While passive hazard avoidance is performed through landing site selection, active hazard avoidance technology is not baselined. This could be addressed for a future mission.

On the lunar surface, the lander will switch the payload to its internal power bus 15 minutes after landing via a mechanical switch. A rover will have 6 hours to perform all necessary checkouts and roll off the lander, during which time the lander will be transferring a critical dataset to the rover's long-term storage for eventual downlink. There will be no interaction between the lander and rover post-egress.

4.2 Assumptions

The following assumptions are made for the lander mission operations and parameters:

- Translunar cruise
  - Orientation: Lander top deck normal to the Sun (solar inertial)
  - Time: Up to 6 days
  - TCMs: Up to 3 days.

- Approach and landing
  - Landing phases: Braking, coast, and approach and landing
  - Landing date: Mission defined
  - Location(s): North or south lunar pole
  - Type: Direct descent
  - Hazard detection: Via site selection
  - Landing accuracy: 100-m radius (99% probability)
  - Engine cutoff altitude: 2 m (TBR).

4.3 Operations and Operational Limits

4.3.1 Solid Rocket Motor Braking Burn

In addition to system simplicity and lower cost/risk, using an SRM for the primary braking maneuver provides the best combination of performance (specific impulse (I_{sp})) and mass (a solid propulsion system has a higher propellant mass fraction (~0.9) compared to a liquid propulsion system (~0.7)). This is similar to the Surveyor landings in the 1960s.

After the SRM has completed its braking burn, the empty casing is separated and the lander uses the bipropellant liquid propulsion system to facilitate a controlled descent to the lunar surface.

The liquid propulsion system was sized by analyzing the trajectory with both hot and cold propellant mean bulk temperature (PMBT) of the SRM and designing for worst case.

4.3.2 Dynamics and Trajectories

To support the design, a delta-V breakdown for the phases is used and is detailed in table 2. A delta-V of 25 m/s is a conservative, historical-based assumption for the TCMs needed during transit. The actual delta-V needed for TCMs is largely dependent on the launch vehicle performance. The delta-V needed by the SRM is a worst-case value, which in this case is for a cold SRM. A 441 m/s delta-V is for a nominal powered descent and assumes that 10% of the liquid propulsion system is used for the roll ACS, which is a very conservative estimate. A delta-V of 21 m/s is allocated for a worst-case divert maneuver where the navigation system determines that the vehicle has a 6-km lateral error after SRM shutdown.

Table 2. Mission: Delta-V breakdown.

Flight Phase	Delta-V (m/s)
After separation from CLV	
+667-N descent thruster	25
+44.5-N ACS thruster (10%)	2.5
SRM operation	
+SRM operation	2,390
+44.5-N ACS thruster (25% duty)	0.24
Vertical descent by lander	
+667-N descent thruster	441
+44.5-N ACS thruster (10%)	41
+667-N descent thruster	21

The assessment of descent trajectories, masses, and SRM-lander separation dynamics are captured in other documents and may be available upon request.

5. LANDER SYSTEM DESIGN

5.1 System Description

The lander is comprised of a pallet-like sheet metal structure, a liquid bipropellant subsystem, a solid propellant subsystem, a thermal control subsystem (TCS), a GNC subsystem, an electrical power subsystem, an avionics subsystem, and flight software. The vehicle has two radiators that support all of the electronics and one vibration-isolated optical bench for GNC instrumentation.

Several trade studies were performed to reach the current lander architecture (i.e., primary structure, communications, power, thermal, propulsion, and avionics subsystems), each focusing on cost and mass as the primary figures of merit. The low-cost lander effort required a philosophy of selecting high Technology Readiness Level (TRL) components; thus, the majority of the components selected on the lander have flown on previous NASA missions (with exception of the mixed oxides of nitrogen-25 (MON-25) thrusters, the Navigation Doppler LIDAR (NDL), and TRN) and therefore required very limited new qualification data.

5.2 Lander Mass

Figure 2 illustrates the lander mass breakout. The current total mass at launch is ~4,250 kg, the majority of which is made up of propellant which is comprised of both the liquid and solid propulsion systems. The lander follows AIAA S-120A-2015 for mass margin guidance. Using these margins, the lander propellant mass fraction at launch is predicted to be 0.61.

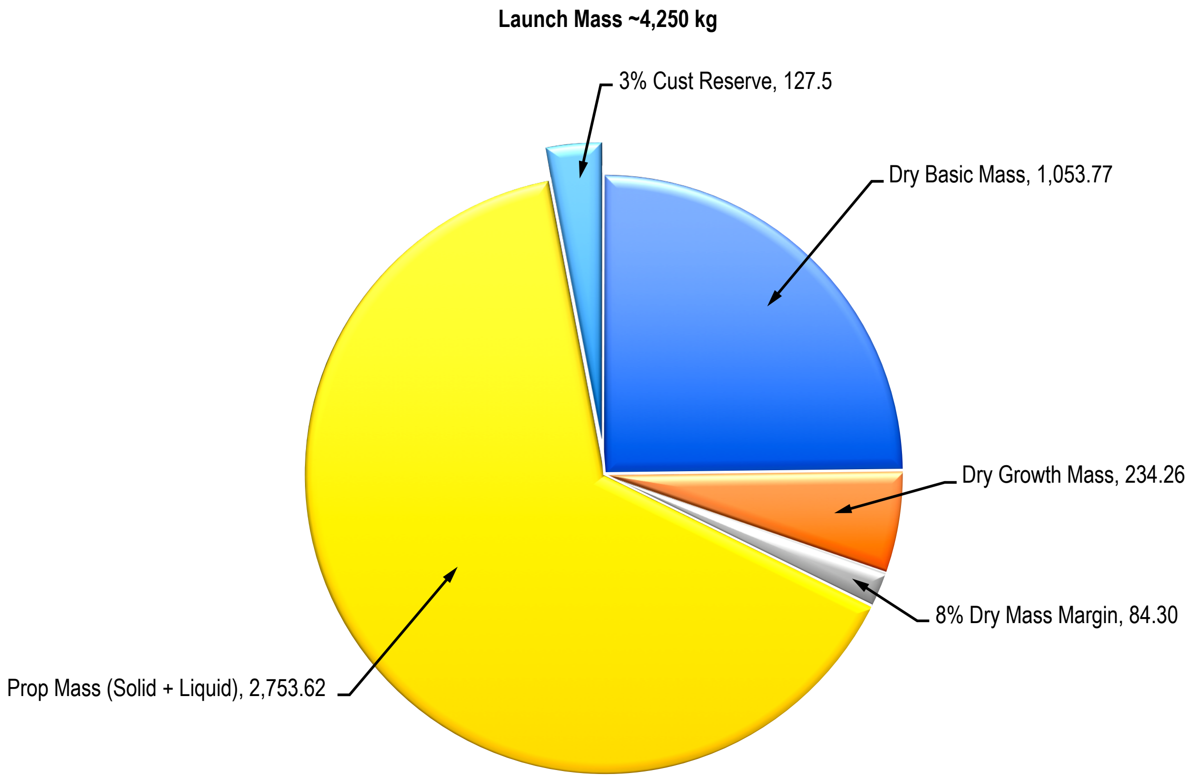

Figure 2. System: Lander mass properties.

5.3 Interface Development

Figure 3 illustrates all the lander external interfaces including external systems such as the launch vehicle and facilities, ground systems, mission operations, and Deep Space Network (DSN), in addition to the rover/payload being taken to the lunar surface. The Payload User's Guide will have the details of the services provided by the lander to payload, which will be used to help generate the interface control documents (ICD) between the two. All of these interfaces need to be taken into account early in the design process, as they will directly impact the lander design.

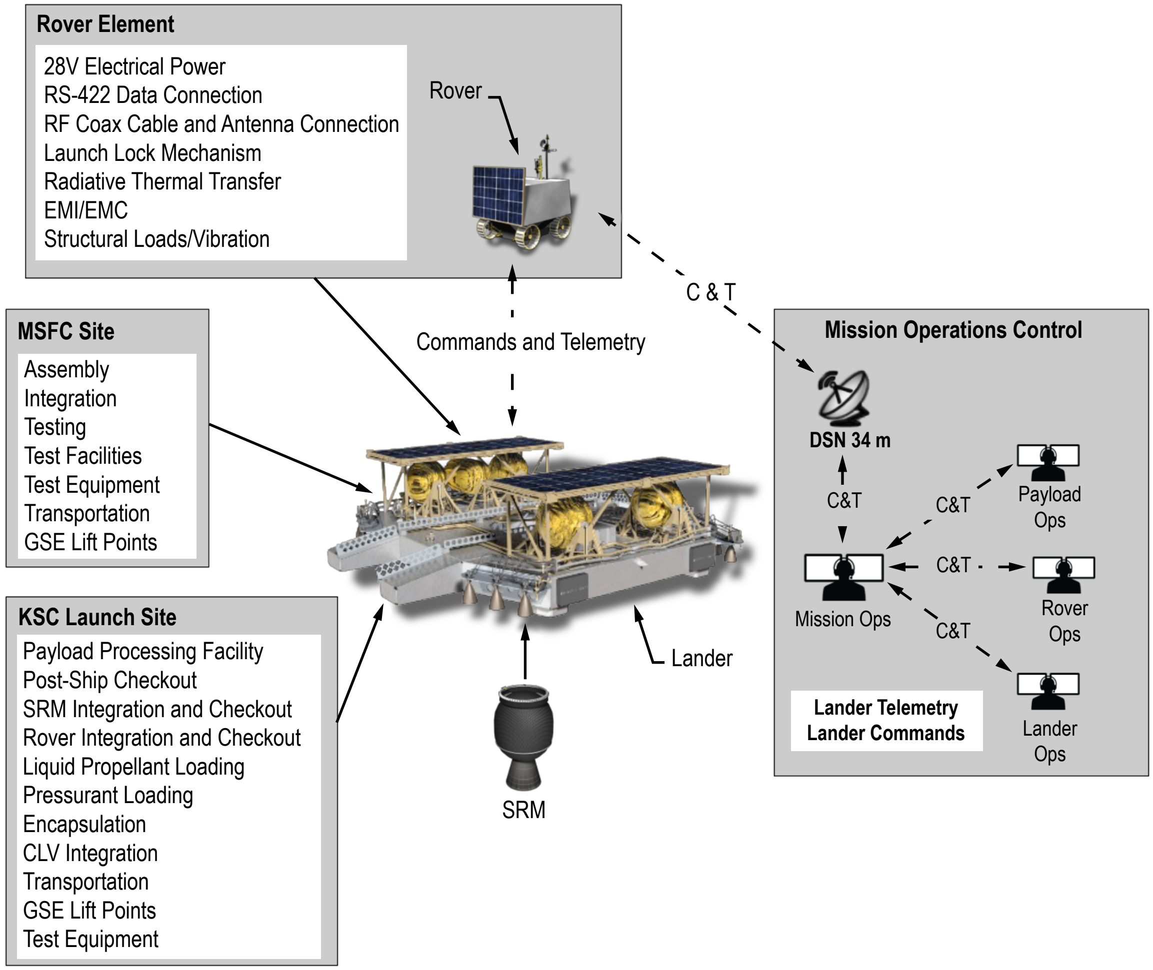

Figure 3. System: External interfaces.

5.4 Lander Integration Considerations

When designing a lander, several lander system integration products need to be generated with the inputs from all of the subsystems, in addition to tracking the designs of subsystems and components as they mature. Every subsystem and component impacts the overall integrated lander design. Following is a list of integrated products and areas of consideration when designing the lander as a whole:

- Integrated systems references:
  - Drawing tree
  - Master equipment list (MEL)/thermal equipment list (ThEL)/powered equipment list (PEL)
  - Design databook
  - Internal interface requirements and ICDs.

- Component integration considerations:
  - Component maturity level
  - Proximity—power source/thermal radiator
  - Placement affects center of gravity
  - Placement to reduce shadowing/view angles—cameras/Sun sensors.

- Integrated models—consistency throughout the team (below is not an inclusive list):
  - Metric units
  - Assigned material properties
  - ProE—Creo. 3.0 CAD models
  - Integrated thermal model.

5.5 Environments

The lander environmental specification, LPL-SPEC-001, defines the natural and self-induced environments that the lander will encounter during the various mission phases including ground operation and transportation, launch and ascent, lunar transfer, descent and landing, and lunar surface operations. The document does not define the requirements; it only serves as the factual or best estimate description of the environments.

6. STRUCTURES AND MECHANICAL

6.1 System Description

The primary structure of the lander has to carry all the components on the vehicle and survive all phases of the mission to safely deliver the payload to the lunar surface. Unlike many missions where minimum mass is the absolute goal, the overarching philosophy of the lander structure was to meet all mission objectives with the least overall cost in regards to design, material purchasing, manufacturing, testing, and verification.

Materials with known properties and availability were chosen to limit material property testing. Isotropic materials were chosen for their 'simplicity' of design and analysis. Use of tight tolerances on parts were minimized to lower inspection and associated part rejection costs. A 'match drill on assembly' approach was taken to reduce individual part tolerances and therefore cost. Tooling was eliminated as much as possible to lower manufacturing costs. The 'go/no go' inspection philosophy of riveted structure was used to lower inspection costs. As many mechanisms as possible were eliminated to lower the costs of testing, verification at operating conditions, and risk posture. The use of the smaller, periphery-mounted descent thrusters put the body of the vehicle close to the lunar surface, eliminating the need for a mechanism to get payloads to the surface. Home-built aircraft techniques were referenced to keep the design simple and cost effective for manufacturing. Common parts were used where possible when symmetry allowed.

During the initial conceptual design and sizing, it was noted that the analysis was favoring a machined deck with dimensions similar to those of sheet metal. Therefore, a decision was made to investigate using sheet metal and its associated construction techniques.

6.2 Structural Analysis Philosophy

The current lander sheet metal configuration has been assessed for strength capability. The sheet metal approach drives the lander to be a buckling critical structure. Because of this, modern-day linear finite element modeling approaches would be computationally expensive and inaccurate. In addition, non-linear assessments are not conducive for sizing and would need correlation to test. Therefore, a less traditional structural analysis approach was required to appropriately size the structure and determine margins of safety associated with ultimate, yield, buckling, crippling, and inter-rivet buckling. This method is described in the following sections. In addition, the following assumptions were used to support the structural analysis:

- Fracture control/analysis for structure-owned hardware has not been considered for this design.
- The structure is being designed to untested factors of safety per MSFC-HDBK-505B, consistent with NASA-STD-5001B as no static strength testing is planned. Any form of buckling is considered to be a failure. Post-buckling strength is not assessed or accounted for when sizing the structure.

• Design loads, payload frequency requirements, and environments used are an envelope of the Atlas V and Falcon 9.
• Secondary structure is being designed utilizing a system-level loads model. Therefore, no frequency requirement needs to be levied on any secondary structure.

6.3 Stress Analysis Method

This section provides an overview of the methodology used to size the primary structure (fig. 4). In general, the design of the lunar lander is governed by buckling rather than by the ultimate or yield allowable of the material. This complicates the typical stress analysis process and requires extensive post-processing of results to determine how to reinforce the sheet metal to increase the critical load capability to be above design loads. This process has been automated to recover the enveloping loading across all load cases to determine the sizing load for each beam. Using this analysis tool, the structural analysis has the following flow to determine appropriate sizing and corresponding margins of safety.

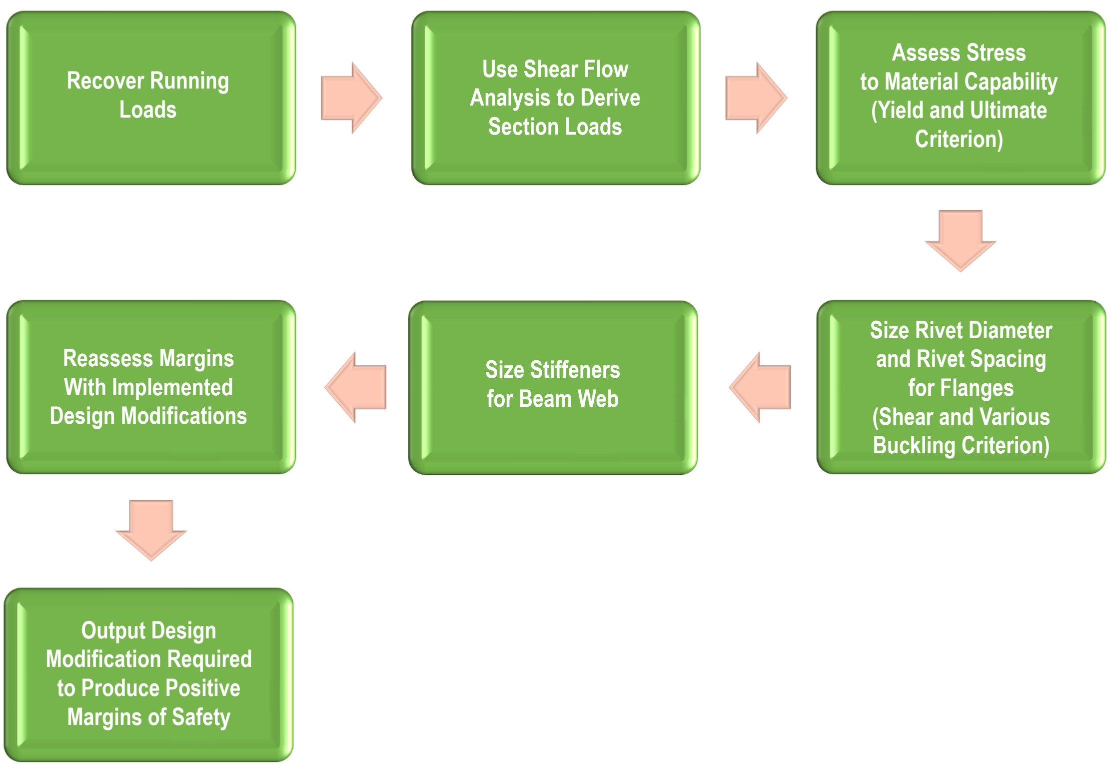

Figure 4. Structural analysis tool showing flow for appropriate size and margins of safety.

The structural strength assessment currently shows positive margins of safety with the exception of crippling. Additional modifications to the baseline design will have to be made for crippling to have an adequate margin of safety.

6.4 Alternate Structural Concepts Trade

A 'pallet' design was driven by trying to get the payload as close as possible to the surface so the ramps could be simple and static. This has additional benefits of increased surface stability and insensitivity to landing hazards such as craters of 1 m diameter or less and lunar rocks less than 30 cm (TBR) on the lunar surface. There were five potential design options traded for the primary structure: isogrid/grid stiffened, flanged grid stiffened, open grid stiffened, composite (metal-core-metal), and a baseline of sheet metal. The figures of merit for evaluating these potential designs were the first modal frequency of the structure, the mass, and the cost.

The mass, frequency, and ranking of cost for each design option are listed in table 3. There was no cost provided for the open grid design option because it may be function-prohibitive for both thermal and thruster plume reasons. The composite design option was the only one that met the minimum required 25-Hz natural frequency and had the lightest mass, but it was the most expensive option. Therefore, the sheet metal design was chosen due to the relatively low cost, ease of manufacturing, and flexibility for future design changes.

Table 3. Structure: Results of alternate structural concepts trade.

Data cannot be compared with current baseline design due to significant design fidelity differences.			
Option	**First Frequency (Hz)**	**Mass (kg)**	**Cost**
No. 1: Grid stiffened	17.7	180	1: Lowest
No. 2: Flanged grid stiffened	20.2	174	2: Mid
No. 3: Open grid stiffened	15.1	270	N/A
No. 4: Composite metal-core-metal	35.0	60	4: Highest
No. 5 Baseline sheet metal	~35.0	83	3: Mid

Note: This study looked at the deck only. Also, higher frequencies and lower mass can be achieved with local geometry refinement.

6.5 Hardware

6.5.1 Solid Rocket Motor Separation Mechanism

The Planetary Systems Corporation lightband is the two-part, spring-loaded release mechanism responsible for the jettison of the SRM at the end of the braking stage of flight (fig. 5). The 'active' half of the mechanism remains with the SRM; the 'passive' half of the mechanism remains with the lander.

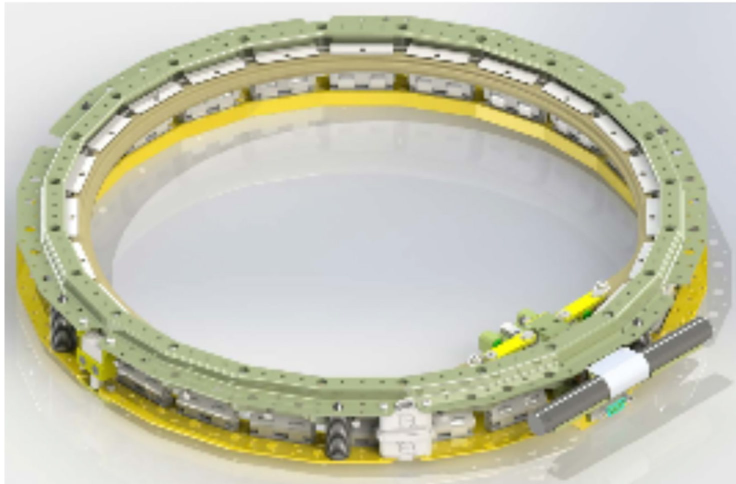

Figure 5. Structure: SRM separation system.

6.5.2 Payload Release Mechanism

The payload release mechanism (PRM) structurally attaches payloads to the lander (fig. 6). The primary function of the mechanism is to structurally secure the given payload during flight and to release it once landed on the Moon. The payload is mounted on the PRM via three attachment locations, with an active release joint (ARJ) at each location. The ARJ is also used for disconnecting multiple electrical connections. Each ARJ comprises of two assemblies. The main assembly is a part of the PRM and the mating assembly is installed on the payload prior to payload integration with the lander.

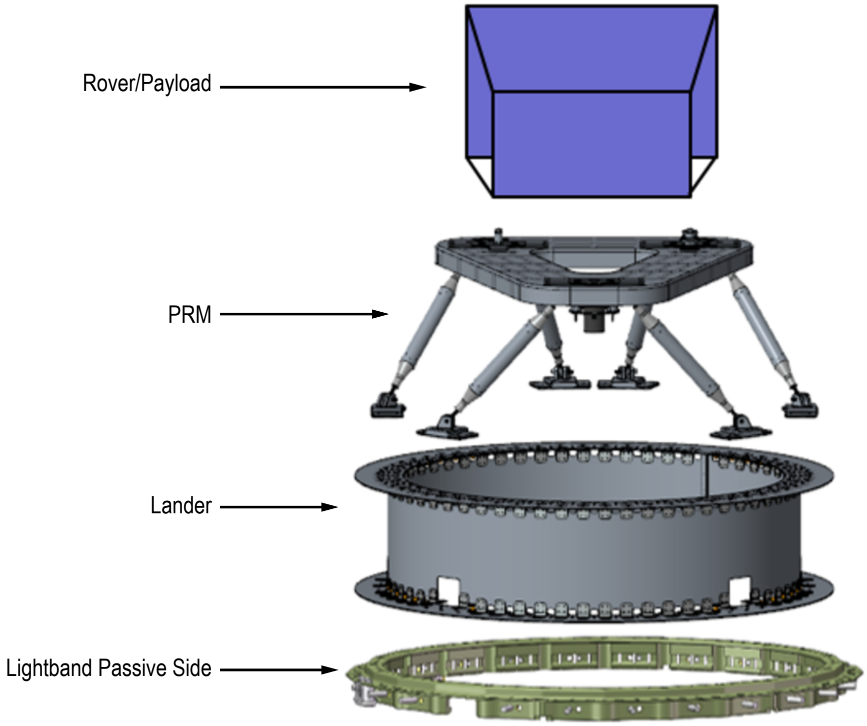

Figure 6. Structure: PRM vehicle stack.

The baseline system supports a 300-kg payload. Upon the release command, the system structurally separates the payload at the three attachment locations and disconnects up to six electrical connectors (i.e., 38999 series III inserts with zero force extraction shell configuration, shell size 21).

The ARJ uses a single release actuator. This mechanism is responsible for initiating the release of the ARJ. The main assembly of the ARJ remains on the lander side; the mating assembly of the ARJ is released with the primary payload.

The ARJ uses two separate features to transfer both axial and lateral loads. Axial loads are transferred through the tension fastener that is released upon command. Lateral loads are transferred through the shear pins until the shear pins are totally retracted once the rover/payload rises.

The ARJ system is modular and can be tailored to a variety of payload configurations. For instance, the system can be downscaled to support payloads with smaller footprints and different electrical connector arrangements. See figure 7.

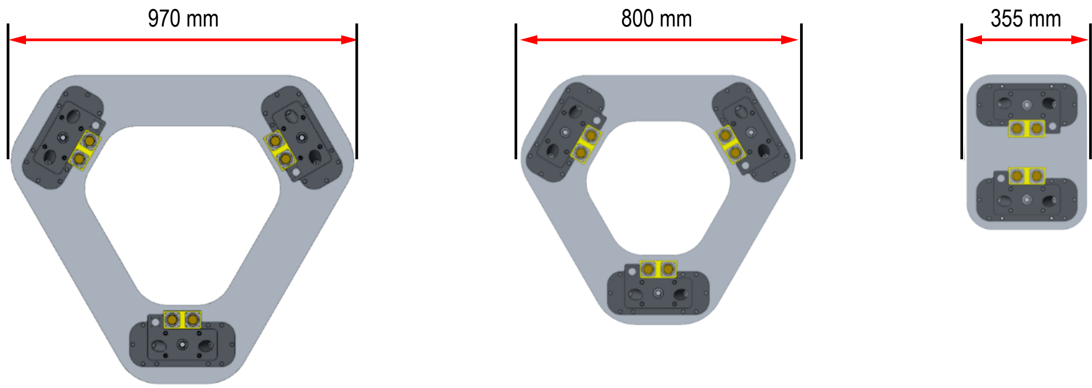

Figure 7. Structure: PRM vehicle stack.

7. POWER

7.1 System Description

The electrical power subsystem (EPS) provides power to required structural, TCS, propulsion, GNC, and avionics subsystems, as well as to any payload. The EPS consists of solar arrays for power generation, batteries for power storage, and associated power equipment such as an integrated avionics unit (IAU) and cable harness. The IAU will control the lander's battery state of charge and provide array string switching. Temperature sensors will record battery and array temperature data available for control logic and telemetry.

The EPS is turned on when the lander separates from the CLV via payload adaptor separation switches and mechanical break wires by connecting the battery to the spacecraft bus. During spacecraft assembly and while on the launch pad, special circuits provide pyro device inhibits in compliance with range safety protocols. After the lander has booted up and the arrays are Sun pointing, it will fully power lander components and full flight required functionality. The EPS will also be capable of supplying an average of 150 We with peak current up to 14 A at a nominal 28 Vdc to the payload, including up to 15 minutes after lunar landing.

7.2 Assumptions

It is assumed that arrays may not be pointed to the Sun during TCMs, so power will be supplied by the battery. During cruise phases between TCMs, full array power is available to operate the lander and recharge the lander batteries. No TCM will last more than 60 minutes, including the time to orient to required vehicle attitude, perform thrust maneuver, and reorient arrays to the Sun.

The lander maintains the rover battery at $95\%^{+}$ state-of-charge via a dedicated channel supplying trickle-charge capability at sufficient voltage. Fifteen minutes after landing, the lander avionics system switches the rover's power to its internal bus and then dead-faces the power between the lander and payload.

7.3 Operations and Operational Limits

A breakout of the power allocations for the various subsystems/components during each mission phase can be seen in table. 4. It includes average and peak power at the different mission phases. Maximum power occurs during the braking and approach and landing phases of flight. Peak power load capability is 3,000 We for 6 minutes. The average wattage is 355 We. As dictated by lithium (Li) ion battery characteristics, maximum battery, and hence bus voltage, ranges between 33.6 Vdc (4.2 Vdc per cell) at full charge and minimum discharge voltage of 20 Vdc (2.5 Vdc per cell).

Table 4. EPS: Breakout of power allocations for the various subsystems/components.

	Mode Power (W)									
Load Power Allocation	**Cruise**		**TCM**		**Braking**		**Landing**		**Post-Landing**	
Subsystem	**Avg**	**Peak**	**Avg**	**Peak**	**Avg**	**Peak**	**Avg**	**Peak**	**Avg**	**Peak**
Avionics	26.00	33.00	26.00	51.00	51.00	51.00	51.00	51.00	21.00	28.00
Electrical power (PDU)	10.00	23.00	10.00	23.00	10.00	23.00	10.00	23.00	10.00	23.00
Flight software										
Guidance, navigation, and control	23.00	47.00	23.00	52.00	23.00	52.00	23.00	52.00	–	–
Harness/cables/distribution										
Mission design	–	–	–	–	–	–	–	–	–	–
Propulsion	30.00	30.00	30.00	2,700.00	2,700.00	2,700.00	2,700.00	2,700.00	–	–
RF communication	65.00	100.00	65.00	100.00	65.00	100.00	65.00	100.00	65.00	100.00
Rover/payload	150.00	150.00	150.00	150.00	150.00	150.00	150.00	150.00	100.00	100.00
Structures and mechanisms	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	2.00	28.00
Terrain relative navigation	–	–	–	–	–	–	85.00	85.00	–	–
Thermal	100.00	100.00	100.00	150.00	100.00	100.00	100.00	100.00	100.00	150.00
Total:	**355.00**	**434.00**	**355.00**	**3,177.00**	**3,050.00**	**3,127.00**	**3,135.00**	**3,212.00**	**298.00**	**429.00**
Systems engineering (growth %)	408.25	499.10	408.25	3,653.55	3,507.50	3,596.05	3,605.25	3,693.80	342.70	493.35
Lander management (uncertainty %)	449.08	549.01	449.08	4,018.91	3,858.25	3,955.66	3,965.78	4,063.18	376.97	542.69

7.4 Hardware

7.4.1 Solar Arrays

The lander design assumes two transit solar array panels that will provide power generation from launch vehicle separation until surface landing. The following assumptions were used in determining solar array sizing:

- 630 We
- Worst-case solar flux (1.015 AU, 1,367 We/m^2/1.0152 = 1,326W/m^2)
- Assume 80% packing factor (ratio of cell area to panel area)
- Areal mass body mounted solar panel = 2.6 kg/m^2 (cells, coverglass, substrate, and harness)
- Mass of panel = 7.8 kg.

7.4.2 Batteries

The lander design utilizes the high capacity LG MJ1 Li ion cell. The driving energy requirement that sizes the battery capacity is lunar orbit insertion to landing and post-landing when no solar array output is expected. Battery characteristics follow:

• Type: Li ion
• Battery case size: 295 mm × 355 mm × 180 mm
• Cell configuration: 8 s × 24 p (192 total)
• Nominal mass: 20.7 kg
• Nominal voltage: 28 Vdc
• Capacity: 78 Ahr.

A battery validation test was performed on the LG MJ1 cell over the entire lunar insertion to landing phase. Current rates were scaled to represent conditions of the full-capacity battery and discharge loads. The cell was taper charged to 4.2 V and test chamber temperature was 20 °C. Battery voltage performance of the MJ1 cell remained above 28 V for the entire discharge (fig. 8).

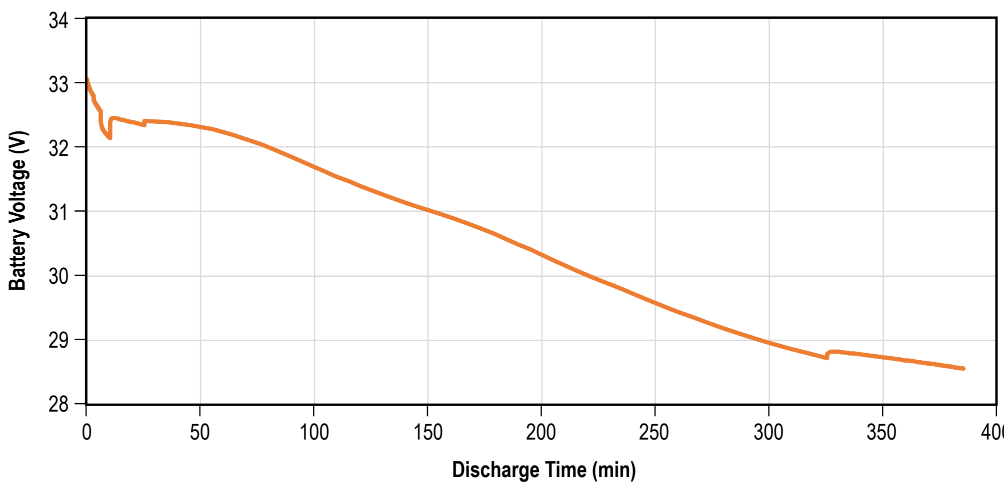

Figure 8. Power: Battery voltage versus time.

8. THERMAL

8.1 System Description

The lander TCS maintains component temperatures within operating and survival ranges from launch through mission end. The TCS attempts to provide or tailor thermal environments suitable to the payload while it is attached to the lander, allowing it to operate from transit through release on the lunar surface. The lander TCS design is largely passive and consistent with the requirements of a class D spacecraft (NPR 8705.4).

During translunar cruise, the lander maintains a top deck solar inertial attitude. Flying at this attitude and locating the lander solar arrays on the top deck reduces the required solar array area by approximately a factor of 3 compared to the previous side-mounted solar array configurations that performed a thermally-driven 'barbecue' roll (spin axis perpendicular to the solar vector). This solar inertial configuration also results in some heater power savings for some key components and protects the solar arrays from DCS thruster plumes.

Other than sensors (e.g., star trackers, landing cameras, altimeter, and the inertial measurement unit (IMU)), the majority of the lander avionics and GNC electronics are placed on two radiators embedded in the top of the deck near the forward and aft ends of the payload. The radiators reject heat from the electronics and allow the heat generated by one component to be shared with other components on the same radiator. This sharing of heat attempts to minimize the number of heaters and the power levels. Because the radiators are pointed at the Sun during nominal transit conditions, optical solar reflector surfaces (i.e., surfaces with very low solar absorptivity/infrared emissivity ratios) and coatings are used to provide a favorable heat rejection capability. These radiators also serve as structural support/mounting for the electronics and provide for convenient bench-top integration prior to insertion/attachment to the lander structure.

As described in section 13, the propulsion subsystem is based on the use of monomethyl hydrazine and MON-25. The TCS provides temperature control of the propulsion system during all phases of the mission and protects key propulsion elements from over-temperature conditions.

It is required that the lander remain in a safe state during payload proximity operations on the lunar surface. The primary concern is the prevention of rupture/leakage of the propellant leading to a combustion event or contamination of sensitive rover components. Lunar surface thermal analysis has shown that no freeze/thaw cycles are likely to happen with a lunar sunrise landing timeline for ~10 hours. Therefore, instead of active safing such as venting the system or heating to prevent propellants from freezing, the lander will prove itself safe through analysis.

Thermal control of the lander power generation and storage system components (solar arrays, batteries) utilizes a passive approach.

The GNC components are located on the deck-embedded radiators, the optical bench near the aft ramp, and in other locations. The early design state of the optical bench presents potentially severe thermal challenges for components in this area. Subjected to environments similar to the top of the deck, the optical bench is also subject to potentially severe plume loading from the DCS thrusters as well as the SRM, and optical bench sensors need to remain operational during thruster firings. Protecting these sensitive components from high heat loads is a key part of the thermal challenge. During the non-thrusting phases of translunar cruise, these components must be kept warm, and the optical bench structure must be kept reasonably isothermal to prevent excessive differential thermal expansion that can impact the relative locations of the GNC components and negatively impact the accuracy of the GNC solution.

In the solar inertial orientation, the SRM is pointed away from the Sun, thus eliminating passive solar heating as an option for maintaining SRM temperatures. Therefore, the temperatures and temperature gradients for the SRM are kept within design targets.

8.2 Assumptions

The following assumptions were made:

- Lander and the payload are powered during transit. Payloads control their own heaters with lander-provided power.
- Single-string heater, sensor, and controllers are implemented consistent with class D expectations, although some survival heaters may be included for mission-critical components.
- No lander mechanisms require active thermal control, with the exception of the SRM, and payload separation mechanisms.
- During TCMs, only one DCS thruster per module is operated.
- During approach and landing, all DCS thrusters from each module may be operated simultaneously and will be pulsed.

8.3 Environments

The lander encounters a variety of thermal environments. The thermal model incorporates the following environments:

- Low-Earth orbit (LEO) (post-launch and prior to TLI).
- Translunar cruise (mostly driven by cold space and orientation toward the Sun)
  - Nominal (solar inertial orientation)
  - Attitude changes due to TCMs, calibration holds, etc.
- Lunar orbit/braking/descent/landing (includes thermal contributions from the Sun and the Moon).
- Polar, high-latitude lunar surface daytime operations (unique environments depending on latitude, longitude, surface tilts, topographical features, time of day, and lander orientation).
- Plume loading
  - SRM (convective and radiative)
  - DCS (convective plume and radiative from hot nozzle)
  - ACS (convective plume and radiative from hot nozzle).

8.4 Schematics

The primary TCS components include the heaters that are distributed throughout the lander. As an example, the lander TCS utilizes heating zones to maintain proper propellant temperatures during translunar cruise. These zones are tailored to the functions and predicted environments associated with each component. The heater circuits for these zones are grouped together in parallel and series to fit within current and voltage allocations of the power distribution system. This figure is only representative of what the system may look like. Changes to the propulsion system layout will have a direct bearing on the number and nature of these zones.

To maintain allowable temperature variations of propellant in each volume (tank, lines, thruster manifolds/heads), the heater system can utilize tighter proportional integral derivative (PID) control or lower fidelity mechanical thermostats. PID control architecture has been found to increase avionics hardware volume requirements under the lander deck. Mechanical thermostats can alleviate lander deck overcrowding of hardware by providing hard-wired control of propellant heaters where the propulsion system allows greater variability.

8.5 Open Design Issues

While thermal analyses show portions of the thermal design to be effective, some portions of the thermal design remain to be formulated and evaluated:

- Thermal and vibration isolation and structural design of the radiators, mass optimization of the radiator, and an assessment of the necessity and efficacy of a heat rejection, turn-down approach to minimize heater power needs to be worked out.
- Work with **S&MA** to complete a failure modes and effect analysis to determine potential consequences of any failures in the heater circuits. This will dictate the need for any safety-critical redundant heaters or controllers that are not currently required as part of the class D mission risk posture.
- Provide preliminary heater circuit layouts to avionics team for integration into the detailed vehicle circuit design.
- Updated payload thermal requirements and payload thermal model integration are needed to support the assessment of how the thermal design of the lander impacts the payload and vice versa, as well as to identify and evaluate any steps (e.g., lander operational constraints or design changes) to remediate these impacts, if necessary.
- Thermal design of the GNC optical bench and associated optics and electronics to ensure nominal or survival temperatures during all mission phases and modes of operations.

8.6 Hardware

8.6.1 Insulation

Most lander components are insulated with standard multilayer insulation to minimize heat transfer with the environment. The design and materials are tailored to the specific needs of each component.

8.6.2 Heaters and Heater Control

The current TCS utilizes PID control heaters to minimize power requirements resulting from large bang/bang control variabilities during flight. This control mechanism is driven by maximum power availability constraints and lander under deck volume constraints. This control is currently planned for implementation in the IAU. There are also a number of components that do not require precise temperature control. Depending on the redundancy assessments to come, additional heaters with mechanical thermostats may be added for survival heating.

9. AVIONICS

9.1 System Description

The avionics subsystem is all of the electronics required for the control of the lander. This includes the separation detection and activation device, the flight computer, the solar array charge controller, the power distribution unit, the propulsion controller, and the thermal non-mechanical thermostat heater controller. The flight computer also interfaces to all the GNC sensors. The services the avionics subsystem provides are as follows:

- Command and data handling for the GNC, thermal, and propulsion subsystems
- Control of propulsion system valves
- Pressure regulation of the gaseous helium (GHe) pressurant
- Pyrotechnic electronic safing, arming, and firing
- SRM initiation
- SRM steering and control
- Operation of separation mechanisms (except the launch vehicle separation)
- Detection of CLV separation
- Spacecraft initialization
- Spacecraft real-time systems clock
- Data and telemetry interfaces between the lander and payload while it is integrated on the lander
- Instrumentation and non-GNC sensors
- Housekeeping data storage
- Limited fault protection by way of Watchdog and Command-Loss timers.

The avionics are in three physical modules: an integrated avionics unit (IAU), a propulsion controller, and a separation interface assembly (SEPIA).

- The IAU serves as the flight computer and power manager for the spacecraft. The flight software (FSW) resides on the IAU's primary microprocessor board. The flight computer also contains interface and signal conditioning boards that interface to the propulsion controller, the SEPIA, the GNC subsystem, and all the non-propulsion sensors. The IAU contains the solar array and charge interfaces and the power distribution unit (PDU) portions of the EPS. The IAU interfaces with and provides power to the rover, while the PDU manages power. The latest concept for payload release is to have both the lander and the payload to have the capability to release the payload.

- The propulsion controller commands all propulsion-related functions as directed by the GNC algorithms in the flight computer.

- The SEPIA is a commerical off-the-shelf (COTS) item with flight heritage. Its primary function is to make sure the lander and its payload remains off during ascent and that it does not power up until after vehicle separation. It accomplishes this by keeping the battery off of the lander's bus. It also enables charging of the lander battery while in the launch vehicle on the pad, allows for check-out of the lander while in the launch vehicle through ground support equipment (GSE), and allows monitoring of key components through GSE.

9.2 Assumptions

The following assumptions were determined:

- The electrical subsystem electronics are inside the IAU with the flight computer.

- The propulsion controller controls everything associated with the propulsion system, including electronically-controlled heaters. The propulsion controller is powered directly off of the battery. The IAU controls when the latching power relays are turned on and thus powers up the propulsion controller.

- The GNC TRN camera and associated electronics will have a separate processor from the IAU.

9.3 Environments

The electrical, electronic, and electromechanical (EEE) parts need to meet the EEE parts requirements described in LPL-REQ-001, EEE Parts Control Requirements. In addition to the EEE parts requirements, the lander avionics needs to operate through the varying radiation environments they will see. These environments vary from launch vehicle separation, transit to the Moon, landing, through lunar operations. The avionics subsystem must also meet the outgassing, thermal, electromagnetic interference, vibration, and structural requirements per LPL-SPEC-0001 Environment Specification Documents.

9.4 Operations and Operational Limits

9.4.1 Power and Thermal

Current power and thermal estimates can be found in LPL-CI-001, LPL MEL/PEL/ThEL.

9.5 Open Design Issues

The effects of lunar regolith on the avionics components is unclear and could drive component selection and design. Studies will be required to determine if the component boxes can be vented or need to be sealed. Other requirements for avionics/propulsion controller are still to be determined.

9.6 Hardware

9.6.1 Integrated Avionics Unit

The IAU provides the following capabilities:

- Million instructions per second >400
- At least 32 GB flash memory with dual access
- Capability for 18 devices with RS-422 or RS-485
- Two 1,553 interfaces
- Three Spacewire interfaces
- 24 PID heater controllers
- 17 AD590 temperature readings
- Capable of controlling 20 solar array sections
- 14-A current limited switch for the primary payload
- Two 15-A switches
- 35 PDU switches
- 17 discrete 28-V commands
- 60-A battery bus
- Real-time clock (1-ms accuracy, 1-s drift/day)
- Radiation tolerance, LET 37 or higher
- Function through **SEE**
- 3-krad total dose.

The selected hardware for the IAU baseline design is comprised of an SCS750 Maxwell processor board and 11 other modular Goddard Space Flight Center (GSFC) designs (known as MUSTANG (modular unified space technology avionics for next generation missions) avionics) (fig. 9). All MUSTANG avionics are build-to-print with no change. The processors are procured items with a connector modification, housed in an existing chassis design (in-house design). One of the processor boards will be adapted to a memory board, which will also provide the Spacewire capability for communicating with the other 11 modules. Shown in figure 9 is the MUSTANG modularity with a picture of MUSTANG flight hardware built for an in-house program.

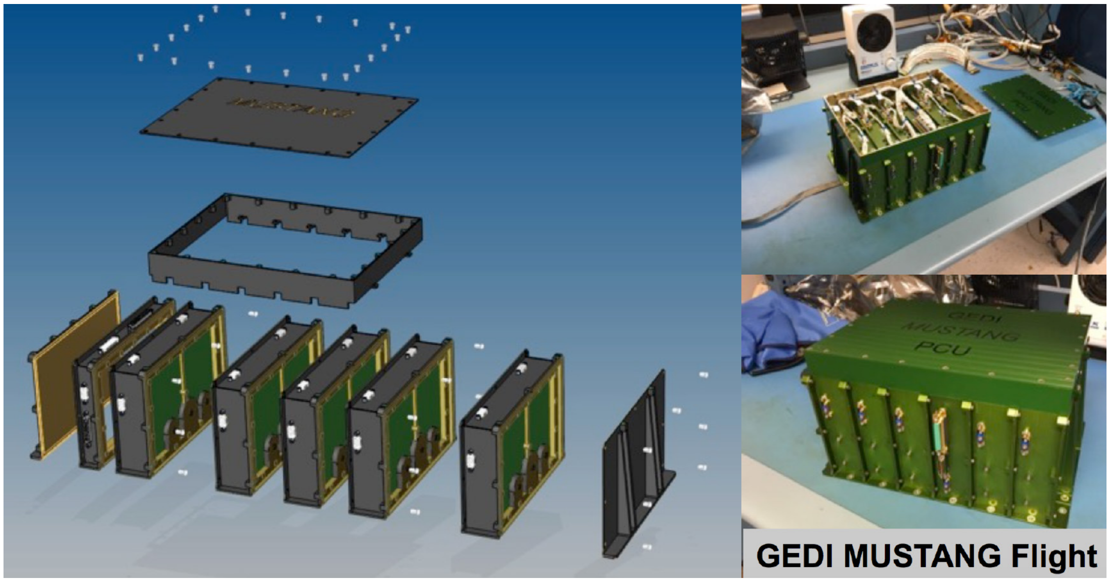

Figure 9. Avionics: MUSTANG hardware.

9.6.2 Propulsion Controller

The propulsion controller design provides the following capabilities:

• Drives eight descent thrusters
• Drives 12 attitude thrusters
• Initiates and drives an articulated SRM
• Controls seven pyrotechnic valves
• Controls solenoid isolation valves
• Provides closed-loop pressure control using a bang-bang pressure-regulating valve
• Provides closed-loop temperature control of the different elements within the propulsion system
• Reads the propulsion system sensors
• Controls the separation sequence for the SRM.

The propulsion controller is an MSFC in-house design. The propulsion controller, like the IAU, is reconfigurable to meet the changing needs of the mission.

9.6.3 Separation Interface Assembly

The SEPIA's primary function is to make sure the lander and its payload remains off during ascent and that it does not power up until after vehicle separation. In addition, it also enables charging of the lander battery while in the launch vehicle on the pad, allows for checkout of the lander while in the launch vehicle through GSE, and allows monitoring of key components through GSE. See figure 10.

Supplier: MBR

Model: SEPIA

Performance Characteristics	
Keeps battery, unswitched and lararus solar array strings disconnected from IAU before separation	
Autonomously connects battery, unswitched and lazarus solar array strings to IAU upon separation	
Provides battery voltage sense inputs to IAU	
Provides battery trickle charge path without powering IAU	
Provides IAU 28 V Input without connecting battery for test	
Provides IAU and EGSE battery monitor board power	
Saves power by deenergizing relay coils upon IAU power-up	
Provides 28 V bus voltage sense to EGSE	
EGSE verification of relay status	
Independent IAU current telemetry to EGSE	
Relays resettable via EGSE command only	
Size	6.18 in × 3.43 in × 2.16 in
Mass	0.8 g
Power	300 W wc

Figure 10. Avionics: SEPIA.

10. FLIGHT SOFTWARE

10.1 System Description

The lander software suite consists of FSW, simulation software (SSW), and test software (TSW). The FSW provides closed-loop control of all dynamic functions of the lander during the operational phase of the mission, except for power-up, which is electromechanically controlled by the SEPIA. The SSW supports the development and verification of the FSW by simulating the lander and its environment. The TSW supports the testing and verification of the FSW by providing a data-and-control interface to the FSW.

The C-based lander FSW is built upon a reusable software library developed by GSFC called core flight software (CFS). This library provides common and necessary services and utilities typically needed by spaceflight applications.

The CFS uses the publish/subscribe model for the communication between software applications. One of the CFS core services provides and manages the software bus. The inter-task software messages are published and subscribed by registered applications. Lander FSW uses the CFS software bus to exchange data.

The lander FSW runs on the IAU main central processor unit (CPU) and relies on the electronic interface provided by the IAU to communicate with the hardware it manages and controls. The software interfaces with GNC hardware (e.g., IMU, star tracker) via the IAU's interface board. It also monitors and controls other sensors through the IAU's signal-conditioning board.

10.2 Assumptions

The following assumptions were determined:

- The lander FSW will use either RTEMS™ or VxWorks® as the lander operating system.
- The lander will use CFS as the infrastructure for lander FSW applications.

10.3 Flight Software Components

The lander FSW is grouped into four major components:

(1) Core flight executive core services:

- Part of the CFS, this component can be reused as-is without additional reconfiguration. It is a set of core services that are functional building blocks to create and host FSW applications.

(2) CFS configurable applications:

- Telemetry output—Responsible for sending telemetry packets to the ground.
- Command ingest—Responsible for receiving and processing commands from the ground.
- Housekeeping—Responsible for building combined telemetry messages containing data from system applications.
- Schedule—Responsible for time division multiplexed (TDM) operation of applications via software bus messages.
- **CCSDS** file delivery protocol—Responsible for transmitting and receiving files to and from the ground.
- File manger—Responsible for providing ground operators with file/directory management commands for the onboard file system.
- Data storage—Responsible for storing table-defined data packages into files for later transmission to the ground.
- Health and status—Responsible for providing and managing health and status of the CPU and its application along with the watchdog.
- Store command—Responsible for providing services to execute preloaded command sequences at predetermined absolute or relative time intervals.

(3) Mission-specific software:

- Autonomous flight manager—Coordinates modes and flight phase-dependent commands to GNC.
- Navigation apps—Processes sensor information to provide estimates of vehicle translation and attitude motion in various frames.
- Guidance—Determines required acceleration needed to achieve trajectory targets. Cycles through powered flight trajectory milestones.
- Control—Orients vehicle and manages requested thrust from propulsion to track the acceleration and orientation request from guidance.

(4) Mission-specific devices input/output (I/O) software:

- Analog/digital interface—Provides software interface to analog and/or digital I/O board.
- PCIO—Provides software interface to the propellant controller hardware.
- IMU interface—Provides software interface to IMUs.
- Sun sensor interface—Provides software interface to the Sun sensor.
- Star tracker interface—Provides software interface to the Doppler LIDAR.
- Altimeter interface—Provides software interface to the Doppler LIDAR.
- TRNIO—Provides software interface to the TRN system.

The mission-specific devices I/O software initializes the devices and sensors (e.g., IMU), retrieves the data generated by them, and processes the data for use by other software. The processed data are published to the software bus and made available to subscribed software, such as navigation software.

10.4 Simulation Software

The SSW supports the development and verification of the FSW by simulating the lander and its environment.

The C++-based lander SSW is built upon a generic simulation toolkit developed by JSC called Trick (fig. 11). The Trick Simulation Environment (TSE) is a set of software utilities that allows users to rapidly develop, integrate, and operate simulations based on specific requirements of their applications. In addition, the lander simulation development leverages existing dynamic and generic subsystem models (JSC engineering orbital dynamics (JEOD), Valkyrie). The JEOD package produces celestial mechanics and vehicle body dynamics. The CFS, embedded in the simulation executable, runs synchronized with the Trick job scheduler. The LPL_SIM is a suite of simulation models that takes FSW commands as input and produces FSW sensor readings as output using environments and vehicle states. The S-modules define simulation jobs as FSW scheduler calls and simulation model function calls.

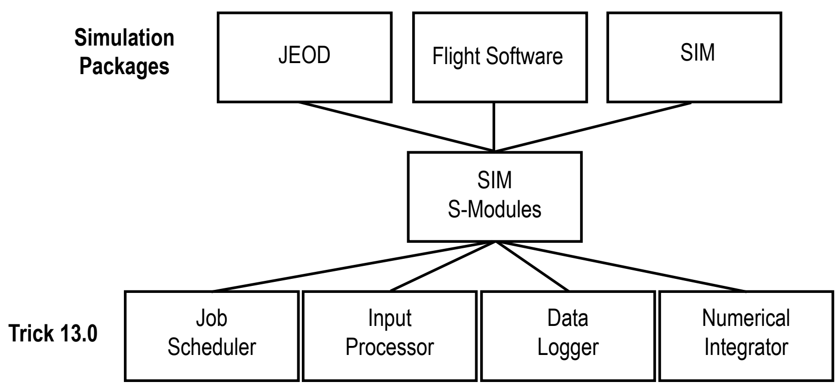

Figure 11. Software: Flight dynamics Sim architecture.

10.5 Test Software

The TSW supports the testing and verification of the FSW by providing a data-and-control interface to the FSW. The Java-based lander TSW is developed by using the Integrated Test and Operations System (ITOS) developed by GSFC.

10.6 Lander Simulator

As part of the software development process, a software prototype and a 6-degrees-of-freedom (6-DoF) integrated vehicle simulation with active guidance and control and perfect navigation were developed. The prototype simulates braking, coast, and approach and landing flight dynamics.

10.7 Data and Command Dictionary

All of the data and commands exchanged between the onboard applications and the ground are contained in the Data and Command Dictionary (DCD). The DCD is used to define, manage, and record all data elements that interface with the core avionics software and hardware. Those data elements include, but are not limited to:

- Unique identification
- Channelization (e.g., bus mapping, vehicle wiring mapping, end-to-end hardware channelization, from sensor/transducer, through signal condition, and multiplexing, to final destination of measurement, etc.)
- I/O variables
- Rate group data
- Telemetry format
- Data recorder format
- Command definition
- Effector command information
- Operational limits (e.g., minimum/maximum values)
- Applicable polynomial coefficients required for data conversion.

10.8 Schematics

The FSW runs on the CPU of the IAU and depends on the electronic interface provided by the IAU to communicate with the hardware it manages and/or controls. The software interfaces with GNC hardware (e.g., IMU, star tracker) through the IAU's interface board and also monitors and controls other sensors through the IAU's signal condition board.

10.9 Lander Source Lines of Code

The estimated source lines of code (SLOC) for lander software is shown in table 5.

Table 5. Software: Estimated SLOC.

Flight Software	**Estimated Source Lines of Code (k)**
CFS (reuse)	60
OS/driver (procure/reuse)	15
Vehicle-specific FSW (new)	85
Simulation and Test Software	**Estimated Source Lines of Code (k)**
Trick, Valkyrie, JEOD (reuse)	373
ITOS (reuse)	300
Vehicle-specific simulation software (new)	45
Vehicle-specific test software (new)	15

10.10 Operations and Operational Limits

The operational limits are to be determined (TBD) for FSW.

11. COMMUNICATIONS

11.1 System Description

In November 2018, we were directed to remove the radio frequency (RF) communication system from the lander and rely on the payload for communication. A multimission lander may want to retain communication capability, so this section is left in for reference only.

The lander communications subsystem works hand-in-hand with the lander avionics subsystem to provide lander-to-Earth communications and data transfer from the time of CLV separation through the end of mission. At a minimum, the lander communication system must be active during mission-critical events.

It is assumed that payloads will have their own communication system to handle their needs on the lunar surface. However, during the transit phase to the Moon, the payload will have a data interface to the lander avionics system to transfer data between the payload and the lander. The lander will relay the data to/from Earth through the lander communication system.

The lander communication subsystem data rates are currently being worked by trying to understand what the subsystem data needs are as well as the reserve needs for the payload. The current reference downlinks data rates are 20 kbps (TBR) during transit and descent phases of flight and 4 kbps (TBR) during post-landing surface operations using the X-band (8,450–8,500 MHz) spectrum. The lander communication subsystem uplink rates are currently 4 kbps (TBR) during all mission phases post-CLV separation using the X-band (7,190–7,235 MHz) spectrum. The lander communication subsystem is designed with a link margin of at least 3 dB during all mission phases.

The lander communication provides near-spherical coverage from CLV separation through landing, and near-hemispherical coverage while on the lunar surface.

11.2 Assumptions

The following assumptions were determined:

- As with all lander subsystems, the lander communication subsystem is zero fault tolerant unless a mission assessment determines that some amount of redundancy is needed.
- The lander communications uses the DSN 34-m dish for analysis and system sizing. Use of other NASA or commercial networks, depending on availability and ability to meet missions, will also be considered.
- Relative navigation is handled by the lander GNC subsystem.
- 100% communications availability during all critical events.
- The lander communication provides lander GNC with range and range rate measurements for navigation during the transit phase.
- Payload communication is provided by the lander communication subsystem as a service during transit.

11.3 Schematic/Block Diagram

A notional lander communication subsystem diagram is shown in figure 12. The number of antenna and type of antennas required, antenna placement, and traveling wave tube amplifier (TWTA) versus **SSP**A are all still being worked.

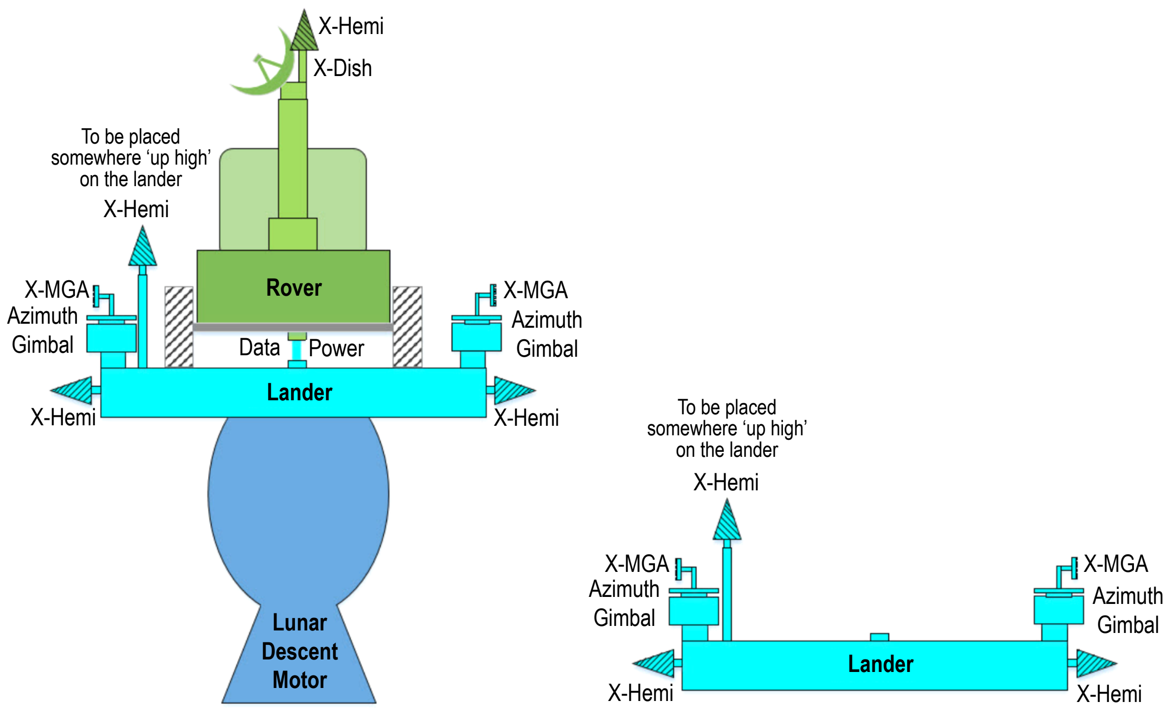

Figure 12. Comm: Antenna configuration (with notional payload).

11.4 Operations and Operational Limits

During translunar cruise and descent, the lander communicates continuously (20 kbps downlink; 4 kbps uplink) with the Earth-based DSN 34-m dish via two opposed, hemispherical, X-band antennas. Once on the lunar surface, the pallet-mounted, gimballed, medium-gain antenna provides nominal communication with the DSN, while a third pallet-mounted hemispherical X-band antenna provides low-rate contingency communication.

The lander is designed to function on up to a 15° slope relative to the lunar surface. If the lander happens to be tilted 15° away from Earth when it lands, the hemispherical contingency antenna gain will be reduced, and the best estimate downlink rate will be ~140 bits/s (assuming 16 dB multipath degradation).

Rover communication is provided by the lander starting at upper stage separation through landing.

11.5 Open Design Issues

Surface antenna locations are being reevaluated to determine whether to have the antenna fixed or 'deployed' when on the lunar surface.

A fixed Omni versus a two-axis gimballed medium gain antenna are also being evaluated in order to support lunar surface operations.

11.6 Hardware

11.6.1 X-Band Conical Spiral Antenna

A typical communications X-band conical spiral antenna is given in figure 13.

Weight: 1 kg
Dimensions: 10 in × 10 in

Figure 13. Comm: X-band conical spiral antenna (typical).

11.6.2 Two-Way Switch No. 1

Mass: 0.5 kg

11.6.3 Two-Way Switch No. 2

Mass: 0.5 kg

11.6.4 X-Band Combiner/Splitter

Figure 14 shows the communications X-band combiner/splitter.

Model: 4311B-2

Mass: 0.6 kg

Figure 14. Comm: X-band combiner/splitter (four-way model shown).

11.6.5 X-Band Diplexer

Mass: 1.2 kg

11.6.6 Communications Transponder

The communications transponder is shown in figure 15.

Power in: ~20 W DC
Power out: 1 to 100 mW RF
Heat out: 12.5 W
Mass: 3 kg

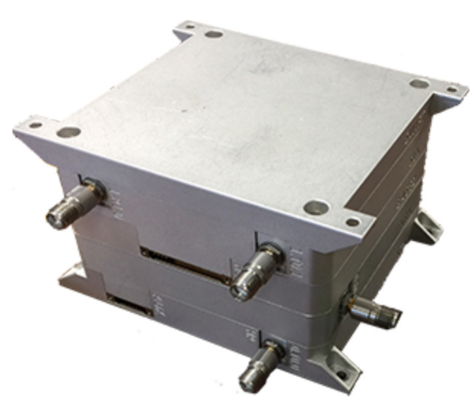

Figure 15. Comm: Communications transponder.

11.6.7 X-Band Traveling Wave Tube Amplifier and Electric Power Conditioner

A typical communications X-band traveling wave tube (TWT), amplifer, and electronic power conditioner (EPC) are shown in figure 16.

Power in: 102 W + 8 W DC = 110 W
Power out: 61 W RF
Heat out: 49 W
Mass: 1 kg + 1.6 kg = 2.6 kg

Figure 16. Comm: X-band TWT, amplifier, and EPC (typical).

11.6.8 Waveguide (WR-137 or WR-112)

Typical electrical specifications are given in figure 17.

Mass: 2.7 kg

Typical Electrical Specifications	
Size	WR 137
Frequency	5.85–8.20 GHz
CW power in watts MCK/MSB	5,000
CW power in watts	2,000
PEAK power in kilowatts	500
VSWR* MCKS	1.09
Insertion loss** MCKS	0.05
VSWR MTPS	1.09
Insertion loss** MTPS	0.07
VSWR MSBS	1.10
Insertion loss**	0.05
VSWR MSDS	1.10
Insertion loss**	0.05

* VSWR is per 2-ft section
** Insertion loss is in dB per foot with silver-plated waveguide

Typical Mechanical Specifications	
Size	WR 137
Frequency	5.85–8.20 GHz
E-Plane (w/ jacket)	2.38
H-Plane (w/jacket)	3.38
E-Plane (w/o jacket)	1.50
H-Plane (w/o jacket)	2.07

Figure 17. Comm: Waveguide (typical).

11.6.9 Coaxial Cables

Figure 18 shows typical communications coaxial cables.

Mass: 2.5 kg

Figure 18. Comm: Coaxial cables (typical).

12. GUIDANCE, NAVIGATION, AND CONTROL

12.1 System Description

The lander GNC subsystem consists of several sensors and a set of algorithms executed on the flight computer that flies the vehicle autonomously from upper stage separation to lunar touchdown. It issues commands to the 44.5-N (10-lb_f) ACS thrusters, the 667-N (150 lb_f) DCS thrusters, and the SRM thrust vector control (TVC) actuators. The lander GNC manages the lander attitude required for thermal control and power generation during the operational phase of flight. A detailed description of the GNC design can be found in reference 2. Also, a detailed discussion on navigation sensor suite and guidance and navigation trades can be found in reference 3.

12.2 Assumptions

The lander GNC system must:

- Perform attitude control during all operational phases of flight.
- Perform up to three TCMs during translunar cruise.
- Perform a braking burn using the SRM.
- Perform a controlled and precision lunar landing using the 667-N (150-lb_f) and 44.5-N (10-lb_f) thrusters, with rapid pulse capability (minimum impulse bit of 8.01 N-s, and startup response time of 12 ms and shutdown response time of 12 ms).
- Declare landing.
- The SRM provides TVC in pitch and yaw, but not roll.
- The lander ACS thrusters provide roll control during the SRM burn.
- A 25% duty cycle is assumed for the 44.5-N (10-lb_f) thrusters for the nominal duration of the SRM burn.
- GNC reserves 2.5% of the total loaded propellant mass.
- SRM performance margin: 5% of descent delta-V.
- The delta-V budget for the TCMs is 25 m/s.
- The attitude control budget for the TCMs is 10% of the TCM delta-V, or 2.5 m/s.

12.3 Mission Design

The mission design is discussed in detail in reference 1 and a short description is provided below. The mission analysis was conducted using the Copernicus n-body trajectory optimization software tool, originally developed at The University of Texas at Austin and now maintained by JSC.

For the preliminary analysis, some simplifying assumptions were made on the mission design to approximate a CLV trajectory. The ascent to LEO was not modeled. Instead, a circular parking orbit was set up with an inclination of 28.5° that was representative of a parking orbit that could be achieved from Cape Canaveral Air Force Station. From this orbit, a simple, impulsive delta-V was used to model the TLI burn. This was done to get a realistic trajectory to the Moon from Earth and will be fine-tuned in future revisions.

After separation from the CLV, the lander will perform up to three ground-calculated TCMs prior to the ignition of the SRM. The delta-V budget for these TCMs has been set at 25 m/s until a more detailed analysis can be completed. Based on historical data, the 25-m/s delta-V budget is considered conservative. If the propellant actually required for the TCMs is significantly less than that budgeted, the excess will be non-propulsively burnt off prior to SRM ignition.

The braking burn for the mission will be performed with an SRM. All CLV analysis so far has concentrated on a cold SRM for conservatism. Future analysis will include the nominal and hot SRM motors in the optimization. The attitude for the burn is set to be within 2° of horizontal in a lunar local vertical, local hold frame, and the yaw angle is optimized to target the landing site.

After SRM burnout, there will be a fair amount of error in the navigated state. Thirty seconds is set aside after burnout for a state and attitude update from the TRN camera and the star tracker. Previous analysis on this mission profile showed that this error in navigated state could be up to 6 km in downrange and crossrange. This means that a total of five post-burnout descent trajectories needed to be modeled in Copernicus to properly assess the effects of this uncertainty (fig. 19). The blue line represents the nominal trajectory, while the red lines represent the four dispersed trajectories that are ±6 km in either downrange or crossrange. This allows the optimization to bias the nominal trajectory so that the propellant required for the redirect maneuvers from the navigation dispersion is minimized. The SRM performs best when it burns out at a low altitude, but the lander needs higher altitudes to minimize the redirect propellant requirements. By varying the burnout altitude parametrically in Copernicus, the optimal burnout altitude for the SRM was determined to be approximately 10 km.

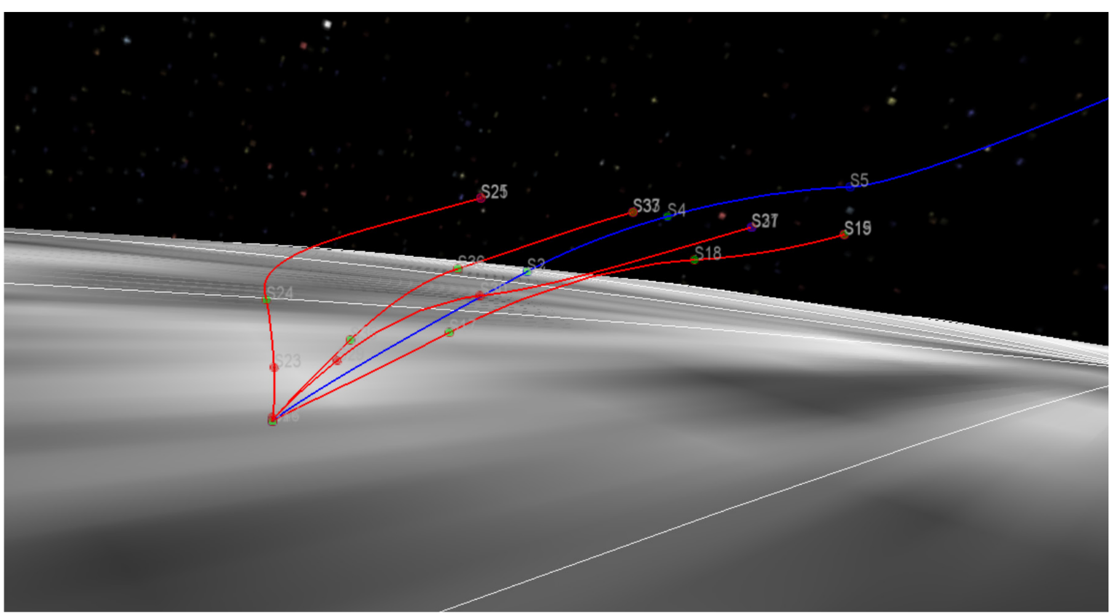

Figure 19. GNC: Copernicus view of the dispersed trajectories.

The final descent phase for the lander begins after the 30-s coast and uses the lander's liquid propulsion system. The descent phase is divided into four phases in Copernicus:

(1) The first phase completes the braking burn of the SRM, and the thrust vector is flown using the Copernicus velocity vector control frame with an optimized pitch rate and constant yaw attitude.

(2) A short coast period is inserted in the second phase to give the lander a chance to reorient for the next phase of flight.

(3) The third thrusting segment opens up both pitch and yaw controls in an east-north-radius control frame so that the descent is fully vertical just prior to landing.

(4) The final thrusting phase concludes <1 m above the surface and is constrained to be vertical at least 2 s prior to touchdown.

Though the current design of the lander is for direct descent to the lunar surface, its GNC subsystem is capable of orbital operations. However, such operations would require significant modification to other lander subsystems.

12.3.1 Design Constraints

In addition to mission and system requirements, the lander design constrains the design and performance of the integrated GNC subsystem (table 6). This is particularly true of the propulsion system components that have been baselined.

Table 6. GNC: Constraints on GNC design and performance.

LPL Design	Constraints	Effect on GNC Design/Performance
SRM performs the braking burn	Gimbal range, gimbal rate limit	Software limits, gimbal margin
Bipropellant, hypergolic liquid propulsion system	Trajectory design, mass properties	The I_{sp} of the liquid propulsion system constrains the trajectory design and the lander mass properties
Solar array pointing	ACS performance	The need to point the solar arrays toward the Sun within some angular range affects the control system design

12.3.2 Integrated Guidance, Navigation, and Control System

The integrated GNC system overview is described in detail in reference 2. The navigation and guidance trade studies and navigation sensor suite performance are documented in detail in reference 3. The GNC is a set of algorithms that will be deployed as part of the FSW. The launch vehicle upper stage is responsible for the TLI burn that sends the lander on its way to the Moon. A short time after separation from the upper stage, the lander GNC system will use its Sun sensors and star trackers to first determine the lander's attitude, and then to orient the lander so that the solar arrays face the Sun. During the 4- to 5-day transit to the Moon, two to three TCMs may be performed, such that the lander arrives at the Moon with the desired navigation state.

The SRM will ignite at a time determined by the guidance system. During the SRM burn, the only active navigation sensor is the IMU. Attitude in pitch and yaw is controlled by the TVC system of the SRM. Roll is controlled by the ACS roll thrusters. After SRM burnout, the SRM will separate and the lander will enter a coast period of approximately 30 s. During this coast period, the ACS will reorient the lander in preparation for ignition of the liquid descent engines. Also, during the coast period, the first TRN position measurements will be made and the star tracker will refine the navigation state attitude. After the 30-s coast, powered descent starts using the DCS engines. Once the altimeter is within range (approximately 4-km altitude), range and velocity data become available and contribute to the navigation solution. TRN will also be active during this liquid burn phase of the mission. At some altitude below 1 km, TRN will no longer be useful and navigation will be dependent on the IMU and the altimeter. At approximately 30 m, the DCS thrusters will start kicking up lunar regolith, and any sensor that cannot 'see' through the dust will become unreliable and will be ignored, making navigation 100% inertial (i.e., the altimeter is no longer used). The flight 'sensor ignore' altitude due to dust kick-up will be determined pre-launch.

12.3.3 Design Updates and Potential Risks

The propulsion system has been upgraded from 12×100-lb_f descent engines with 12×5-lb_f attitude thrusters to 8×150-lb_f descent engines and 10-lb_f attitude control thrusters. While GNC detailed design has not incorporated this propulsion change, it is believed that it will not be a significant impact on the vehicle control and flying performance.

The 150-lb_f engine is currently under development, so GNC design is based on predicted engine performance. GNC is very sensitive to minimum impulse bit capability of the engine in order to pulse and off-pulse rapidly. If minimum impulse does not meet the expected performance, the vehicle may not be controllable in this configuration and may require significant redesign; e.g., larger attitude thrusters.

A high-level description of the risk that plume-surface interaction (PSI) presents to a lunar lander is given in appendix A. PSI can damage the lander, the payload, and potentially loss of mission. GNC makes the recommendation that the nominal descent engine shutdown altitude be set to 2 m. A lander that shuts its descent engines down at precisely 2-m altitude with an initial velocity of –1 m/s will reach the lunar surface at –2.74 m/s. Analysis indicates that with the baseline sensor suite that the navigation system altitude error at a targeted shutdown altitude of 2 m is approximately ±0.5 m 3σ. The precise values depend on numerous factors (sensor alignment error, the altitude at which NDL is shut down, NDL noise, …). Details on this analysis are available from the authors listed in reference 3.

Critical GNC sensors are long lead items and thus present a schedule risk. They include:

- Terrain relative navigation system.
- NDL.
- Deep-space/radiation-hardened IMUs.

12.3.4 Guidance, Navigation, and Control Simulation

The lander GNC team has created a digital simulation of the lander. That simulation was used to design the GNC algorithms and to verify most of the GNC requirements.

12.3.4.1 Generalized Lander Simulation in Simulink. It was desirable to create a modern, modular, user-friendly, 6-DoF, Simulink-based simulation for use in control system design and analysis of landers. Support for unit testing of components was another design requirement for the lander simulation. Generalized lander simulation in Simulink (GLASS) is being developed to fill this roll. GLASS is built using the latest version (2019a) of MATLAB®/Simulink in order to take advantage of the newest features and improvements found in this software. GLASS was constructed from the ground up to utilize several of these features, including Simscape Multibody™, autocoding functionality, and data logging.

12.3.4.2 Generalized Lander Simulation in Simulink Modeling Capabilities. While GLASS is currently being utilized for the simulation of landers, GLASS is capable of simulating a variety of spacecraft. The core dynamics engine of the simulation can be easily modified to model a new craft. As the simulation is 6-DoF, the craft being modeled can be a lander, an ascent vehicle, a satellite, a rover, etc. The use of Simscape Multibody is what gives GLASS the flexibility to model a variety of vehicles without huge modifications to the simulation. This also allows GLASS to model multiple spacecraft within the same simulation.

GLASS is capable of simulating landers in a variety of environments. The currently implemented environments include flat and spherical planet modes, and Earth-Moon and Sun-Earth-Moon systems. These modes allow the user to choose a level of fidelity appropriate to the current design. The Earth-Moon and Sun-Earth-Moon systems utilize JPL's SPICE Toolkit to calculate the planetary ephemeris data used by GLASS to drive the positions of the celestial bodies. While only combinations of the Sun, Earth, and Moon are currently modeled, other planetary bodies can be modeled as well.

GLASS is also capable of running different gravity models depending on the desired level of fidelity required. The standard gravity model in GLASS is a pure spherical gravity model; however, the user may also select a spherical harmonic gravity model.

12.3.4.3 Generalized Lander Simulation in Simulink Spherical Harmonic Gravity Model. By default, GLASS uses a simple spherical gravity model based on the distance of the spacecraft from the center of the Moon. This method assumes constant gravitational acceleration for equidistant points from the Moon's center, which is not as valid an approximation for the Moon as it is for the Earth. This is because of various mass concentrations under the surface of the Moon which cause relatively large gravitational anomalies. For increased accuracy, a spherical harmonic gravity model was formulated, implementing data from NASA's Gravity Recovery and Interior Laboratory (GRAIL) satellites. The data are provided in the form of normalized Stokes coefficients, along with a reference altitude and gravitational constant to characterize the data. The coefficients are provided between 2° and 270°.

The gravitational acceleration at any point above the center of the Moon is calculated via a Cartesian reformulation of the Pines spherical harmonic gravity model, which uses the Stokes coefficients and Legendre functions to model the gravity field. The degree of the model is chosen to balance accuracy against computation time. Testing has shown that no significant increases to accuracy, albeit significant reduction in simulation speed, occur after degree 120. Therefore, degrees 80 or 120 are often used when modeling gravity in GLASS with a spherical harmonic model. The original MATLAB code for the spherical harmonic gravity model was compiled into an S-function to increase speed.

Orbital tests have shown that a higher degree spherical harmonic model matched the reference trajectory generated by Copernicus much better than a simple spherical model. The Simulink Aerospace Blockset™ (SAB) also includes a spherical harmonic gravity model block. However, when run using the same GRAIL data, the SAB does not compare as well to the reference trajectory as the custom spherical harmonic gravity model built for GLASS. In fact, when using larger time steps, the SAB actually diverges from the other gravity modeling methods.

12.3.4.4 Simscape Multibody and Autocoding. Simscape Multibody is the most noticeable feature within GLASS (fig. 20). Simscape Multibody breaks from the standard method of having to deal directly with the equations of motion. Instead, the equations of motion are formulated and solved within Simscape Multibody. This allows the user to develop the core dynamics (degrees of freedom, gravity model, inertial frame, etc.) one time, regardless of which spacecraft is implemented in the simulation. This allows the core of the simulation to be divided between the 6-DoF dynamics and the Simscape Multibody components related to the modeled craft. This reinforces the modular nature of GLASS and allows new spacecraft or variants to be modeled quickly.

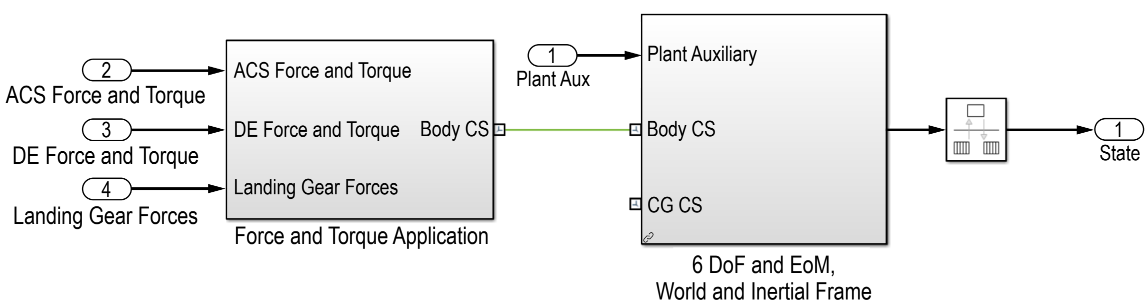

Figure 20. GNC: GLASS core dynamics separation.

Additionally, splitting the core dynamics from the spacecraft-specific portions of the model allows the core dynamics to be common between multiple simulations while disallowing proprietary and sensitive data to be shared between the various simulations.

Simscape Multibody supports the built-in autocoding functionality of Simulink. With the assistance of MathWorks, the core Simscape components of GLASS were rearranged along with several other tweaks to allow the entire core of the simulation to be autocoded into an S-function. This

S-function allows nearly any block parameters (such as initial conditions) to be tweaked prior to running the simulation. Only one S-function is required to be included with the simulation to provide the same functionality as the full Simscape version. Once autocoded into an S-function, GLASS no longer requires a Simscape Multibody license for users to utilize the simulation. Once the core dynamics and spacecraft-specific blocks are added to the Simscape core, very few changes will occur within the subsystem, so GLASS can be delivered to the various groups working on the project, and Simscape-based license issues are eliminated. Depending on the configuration of the spacecraft simulation, the S-function version of the Plant can show performance improvements over the full model.

12.4.4.5 Simulink Projects and Data Dictionaries. GLASS utilizes a relatively new functionality within Simulink called Simulink Projects. Simulink Projects assists in the componentization and modularization of the model by breaking the model down into multiple 'projects.' A project contains everything that a particular model needs to run, including libraries, parameters, models, and documentation. Simulink Projects automatically handles the paths to all necessary files along with integrating with various source control software. Projects can also handle running initialization or termination scripts for a project, simplifying the code associated with models.

Projects can also be contained within larger projects. For example, a DCS engine project would be contained within the larger Plant project. The top-level project requires the sub-level project to run, while the sub-level project can run independently. This allows groups to work independently on different sub-level projects, ensuring that the top-level framework into which the sub-level projects incorporate will operate correctly.

Along with Projects, GLASS incorporates data dictionaries into the file infrastructure. Data dictionaries contain all of the relevant parameters and variables that a model requires. Data dictionaries work similarly to Projects in that top-level dictionaries can reference multiple sub-level dictionaries. Typically, these dictionaries mirror the project hierarchy and are included with their respective project. Data dictionaries can also contain Simulink parameters that can contain additional information for variables beyond just the data type and value, such as units or even notes on what the variable represents. Data dictionaries can also contain documents that belong with the simulation. As with Simulink Projects, data dictionaries help make GLASS modular and allow for unit testing of various models within the simulation.

12.4 Operations and Operational Limits

The lander GNC subsystem utilizes the end effectors (thrusters) provided by the propulsion subsystem to control the spacecraft, which is impacted by the locations and orientations of those end effectors.

12.5 Hardware

Notional hardware options are listed in sections 12.5.1 through 12.5.6.

12.5.1 Optical Bench

Quantity: 1
Unit Mass: TBD kg

12.5.2 Star Tracker

The HYDRA-M star tracker is shown in figure 21.

Supplier: Sodern
Model: HYDRA-M

Optical Head
Quantity: 2
Unit Mass: 1.4 kg
Electronics Unit
Quantity: 1
Unit Mass: 1.35 kg

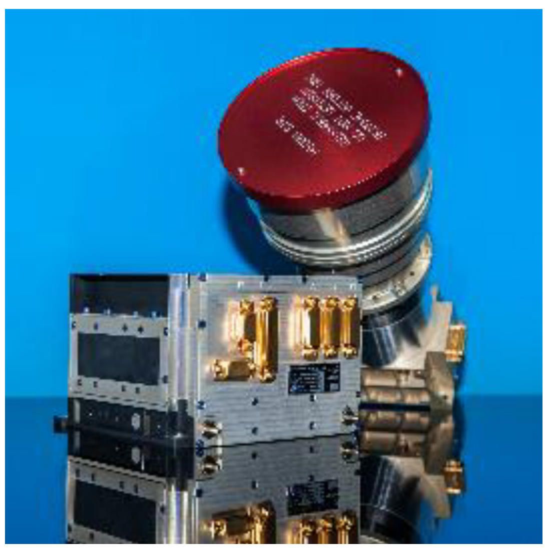

Figure 21. GNC: HYDRA-M star tracker.

12.5.3 Sun Sensor

The sun sensor and facts are given in figure 22.

Supplier: NewSpace Systems
Model: SS411
Quantity: 6
Unit Mass: 0.04 kg

	Sun Sensor
Functional Characteristics	
Field of view	140°
Update rate	5 Hz
Accuracy	≤0.1° rms error over the FOV
Physical Characteristics	
Dimensions	34 mm × 32 mm × 20 mm
Mass	35 g
Power	7.5 mA average, 26 mA peak
Enviromental Characteristics	
Thermal (operational)	–25 to 75 °C
Vibration (qualification)	14 g rms random (10 g acceptance), 1,000 shock
Radiation (TID)	10 krad total dose (component level)
Interfaces	
Power supply	5 V DC nominal (5 to 50 V)
Data	RS485 UART
Connector	9-way female Micro D
Mechanical	No. 2 screw

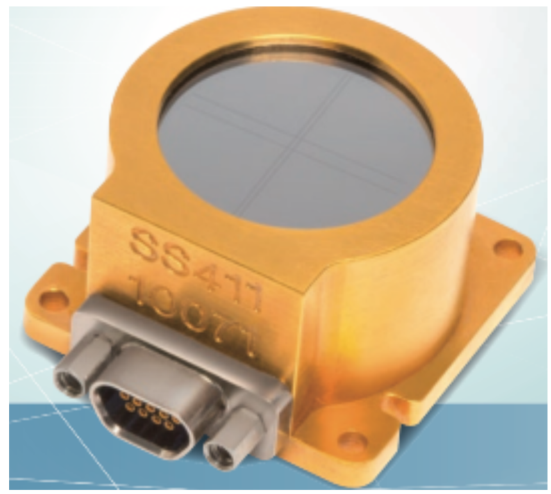

Figure 22. GNC: Sun sensor.

12.5.4 Terrain-Relative Camera

Figure 23 shows the TRN camera.

Supplier: Malin Space Science Systems
Model: ECAM-C50
Optical Head
Quantity: 2
Unit Mass: 0.53 kg
Digital Video Recorder (DVR)
Quantity: 1
Unit Mass: 1.11 kg

Note: The ECAM-L50 currently in development for the Mars 2020 mission is being considered for the lander.

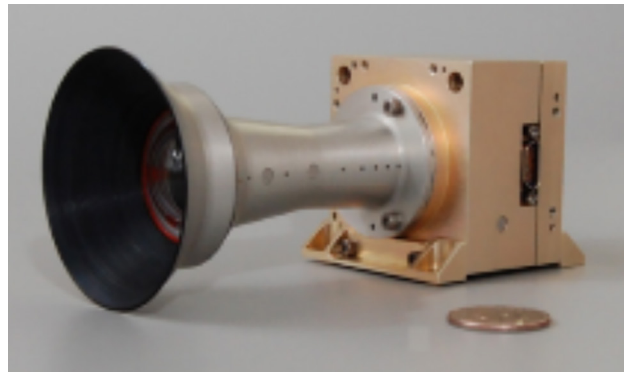

Figure 23. GNC: TRN camera.

12.5.5 Altimeter/Velocimeter

The NDL, currently under development, is the baseline for the lander and shows great promise. It can provide accurate line-of-sight range and velocity for all three vehicle axes in a compact package. It is currently at TRL-5 and expected to be at TRL-6 by the end of 2019. A description of the NDL is given in figure 24.

Maximum LOS range*		>4,000 m
Maximum LOS velocity		200 m/s
LOS velocity error**		0.2 cm/s
LOS range error**		30 cm
Data rate		20 Hz
Dimensions	Electronic chassis	28 × 22 × 20 cm
	Optical head	34 × 33 × 21 cm
Mass	Electronic chassis***	8.7 kg
	Optical head	5 kg
Power (28 VDC)***		

* Dependent on atmosphere and surface albedo.
** Errors dominated by the vehicle's vibration and angular motions.
*** Heatsink and fans module for terrestrial operation adds 1.5 kg and 10 W.

Figure 24. GNC: NDL GEN 3.

12.5.6 Inertial Measurement Unit

The IMU, characteristics, and performance are given in figure 25.

Supplier: Northrop Grumman
Model: LN-200S
Quantity: 1
Unit Mass: 0.75 kg

Characteristics	
Power	12 W nominal regulated at ±5 Vdc and ±15 Vdc
Dimensions	Diameter: 3.5 in (8.89 cm) Height: 3.35 in (8.51 cm) (plus connector)
Weight	1.65 lb (748 g)
Volume	35 in^3
Temperature range	–54 °C (–65.2 °F) to 71 °C (159.8 °F)
Survival temp range	–62 °C (–79.6 °F) to 85 °C (185 °F) Perf
Shock	400 g/100 Hz; 1,500 g/1,000 Hz
Vibration (survival)	15 g rms random
Radiation tolerant	10 krad
Electrical interface protocol	RS-422/485 serial data
Enclosure	Hermetically sealed

Performance	
Accelerometer (1σ)	
Bias repeatability	300 μg, 1σ
Noise	35 μg/√Hz
Scale factor accuracy	300 ppm, 1σ
Input axis alignment	0.1 mrad
Max input accel	40 g
Gyro (1σ)	
Bias repeatability	1°/hr, 1σ
Scale factor stability	100 ppm
Angle random walk	<0.07°/√hr
Input axis alignment	0.1 mrad
Dynamic range (max)	1,000°/s
Bandwidth	200 Hz @ 400 Hz data rate

Figure 25. GNC: IMU.

13. PROPULSION

13.1 System Description

The lander propulsion subsystem provides end-effectors (thrust) required by the GNC subsystem to control the spacecraft from the time of upper stage separation through lunar landing. The propulsion subsystem is comprised of a solid propulsion stage and a liquid propulsion stage. The solid propulsion stage is used for braking at the end of translunar cruise, whereas the liquid propulsion stage is used for TCMs, nutation damping, attitude control, and lunar descent.

The baseline lander propulsion subsystem is predominantly based on either COTS or existing design components. However, while some propulsion components are operational, other key components such as propellant tanks and thrusters are in development. The need for requalification of these operational components has not been fully assessed.

The solid propulsion system consists of a single SRM with a TVC system that brakes the lander out of its translunar coast trajectory. The SRM is jettisoned after motor burnout as the lander enters its terminal descent phase.

13.2 Assumptions

Assumptions for the liquid and solid propellent stages follow:

- Liquid propellant (lander) stage:
  - This design activity focuses on the integration of currently existing and available components, with the exception of the thrusters. Development of the thrusters is an ongoing, separate activity that is expected to be completed in the near future. The integrated lander design assumes a functional set of thrusters (ACS and DCS) that meet all performance and interface requirements identified in this TP.

- Solid propellant (braking) stage:
  - Any increase in solid motor propellant will operate within the capabilities already demonstrated by the vendor.
  - The anticipated nominal temperature environment will be within the demonstrated parameters for the solid propellant and specified propulsion components.

13.3 Trade Study

13.3.1 Propulsion Configuration

Based on an initial concept for a previous configuration, the propulsion configuration has two stages: a separable SRM for the braking function and a bipropellant liquid propellant stage to perform all other GNC functions. Because of mission cost constraints, the trade study was performed to find ways of reducing cost with technical risk, with system mass and technology advancement requirements taken into consideration.

For the lander stage, the trade study included COTS hardware, Aerojet R-4D, LEROS2b, and the like. In addition, there is a design that heavily leveraged propulsion hardware from fourth stage of the Peacekeeper intercontinental ballistic missile. Table 7 summarizes the rationale for each concept.

Table 7. Prop: Stage concepts.

Concept	Rationale
Braking Stage	
SRM	Already qualified and operational, system simplicity, high propellant mass ratio
LOX/CH_4	Non-toxic, high performance, provides opportunity to demonstrate technology for future exploration
Hypergolic bipropellant	Flight qualified, low cost (government-owned, such as fourth stage PK and space shuttle OME)
Lander Stage	
COTS	Already qualified and operational
Available PK hardware	Flight qualified, low cost (government-owned, such as fourth stage PK missile)

Several criteria were established for the propulsion system downselection. The cost of developing the propulsion system should be within the projected allocation and the lander mass should be within the launch capability. Based on the described criteria, four options for the propulsion configuration were downselected from 11 configurations. The liquid oxygen (LOX)/liquid methane (LCH_4) was not considered due to it being the most expensive option from a flight hardware development, system integration, and ground handling.

13.3.2 Command Destruct System

A quick trade was done to determine the best command destruct system to use on the lander braking SRM.

The lander would not require an FTS on the deorbit stage SRM, because the motor is considered inert until the safe and arm (S&A) device is activated. Instead, the range may require that there is a command destruct system for the SRM. After launch, if the launch vehicle alters course off its expected trajectory past a specified range, its internal flight termination system would abort the launch. Depending on the debris analysis conducted by the range, they may request that the lander SRM is broken into smaller pieces to minimize damage. This command destruct system would be located on the motor and would receive a signal to destruct from the launch vehicle's flight termination system when it activates.

There are two primary options for the command destruct charge for the SRM. Linear-shaped charges (LSC) could be used, expanding the length of the motor. The alternative option is conical-shaped charges (CSC) mounted in proximity to the fore or aft motor domes of the motor. It was determined that the preferred option is the usage of CSCs as the destruct charges. The advantage being flexibility for the design and construction for incorporating other components to the motor, taking up less space than the LSCs.

13.4 Configuration

13.4.1 Solid Propulsion Stage

An SRM has been downselected for the braking stage. SRMs have a high demonstrated I_{sp}, mass fraction, and are commercially available (fig. 26). The assumed SRM includes a titanium case, solid propellant, igniters, a vectorable nozzle, and an actuator system. Changes to a commercial product may be driven by a trade, to include heater/power weights versus a lower temperature operation of the motor, below the current temperature limits.

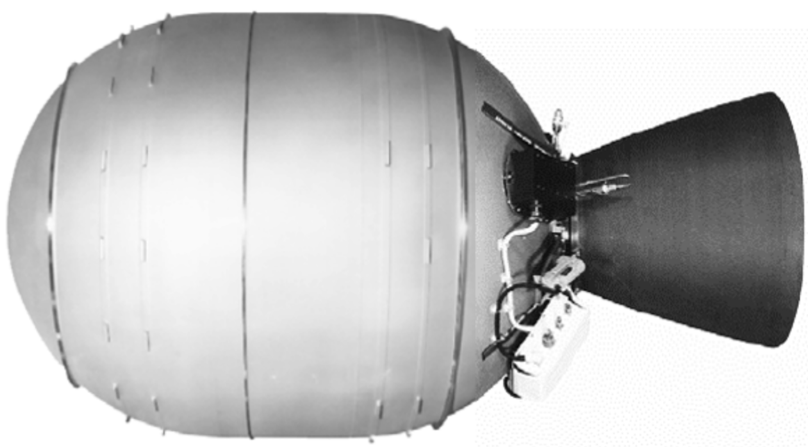

Figure 26. Prop: Example SRM (braking stage).

The SRM mates to the lander via the Mark II motorized lightband separation mechanism. The lower ring of the lightband bolts to an adapter ring, which then bolts to the SRM. The lightband separates internally, allowing the motor to separate.

The SRM has a vectorable nozzle and an integrated, single-use power supply for the actuators. The power supply can be bypassed and the system fed with ground-supplied power to ensure proper connections to the thrust vector actuators and to demonstrate their function during ground checkout. Vector commands during flight will come from the GNC subsystem.

13.5 Operations and Operational Limits

The SRM provides the braking necessary to depart translunar cruise and reach the coast velocity prior to approach and landing. The SRM is jettisoned at the conclusion of the SRM burn, which nominally takes ~87 s.

The planned PMBT is 70 °F. The GNC and TCS subsystems are designed for this PMBT.

A coast period follows the braking burn, after which approach and landing commences, which nominally takes ~101 s.

In the current phase of development, 288.37 kg of liquid propellants are required; estimate unusable residuals (feedlines and tanks): 23.31 kg.

13.6 Hardware

13.6.1 Solid Rocket Motor

The propulsion braking stage is shown in figure 27 and the typical SRM thrust profile in figure 28.

Supplier: TBD
Model: TBD
Quantity: 1
Safety: S&A Device
TVC: Required

Figure 27. Prop: Braking stage.

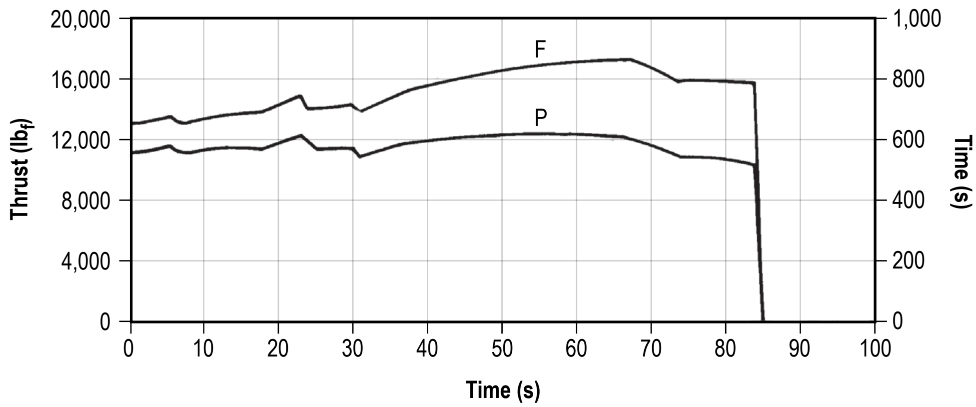

Figure 28. Prop: SRM thrust profile (typical).

13.6.2 Safe-and-Arm Device

The propulsion SRM S&A device is shown in figure 29.

Supplier: TBD
Model: TBD
Quantity: 1

Figure 29. Prop: SRM S&A device.

13.6.3 Explosive Transfer Assembly (ETA)

Figure 30 shows the typical propulsion flexible explosive transfer assembly.

Supplier: TBD
Model: TBD
Quantity: 1

Figure 30. Prop: Typical FETA.

13.6.4 Descent Control System Thrusters

The lander platform utilizes eight 667-N DCS thrusters arranged in linear parallel groups of two at the four corners of the structure. The configuration affords sufficient control of descent during the approach and landing phase of flight. A direct-acting valve assembly is integrated to the thruster.

13.6.5 Attitude Control System Thrusters

The lander platform utilizes twelve 44.5-N ACS thrusters arranged in triorthogonal groups of three at the four corners of the structure. The configuration affords full pitch, roll, and yaw authority throughout all stages of the mission post-upper stage separation.

13.6.6 Gaseous Helium Latch Isolation Valve

Table 8 lists the lander design specifications for the GHe positive isolation valve. An informal vendor survey has indicated that an existing design is available with either delta-qualification effort or specification adjustments, particularly for internal leakage at low temperatures. The notional specifications and part are listed in table 8.

Table 8. Prop: GHe latch isolation valve specifications.

Title	Requirement
Item definition	Unidirectional, latching gas isolation valve
Interface Definitions	
Mechanical interface	
Mounting	Panel mounted, bolted to vehicle
I/O ports	3/8 in OD, 304 L welded tube stub
Electrical interface	
Operating voltage	27 ± 2 Vdc
Connection	Electrical connector TBS
Fluids	
Operating fluid	GHe per MIL-PRF-27407
Compatible fluid	GN_2 per MIL-PRF-27401 IPA per TT-I-735
Pressure	
Operating pressure range	0 to 4,500 psia
Maximum expected operating pressure (MEOP)	4,500 psia
Maximum design pressure (MDP)	4,500 psia
Performance Characteristics	
Pressure	
Proof pressure	$1.5 \times$ MDP
Burst pressure	$2.5 \times$ MDP
Flow and pressure drop	
Flow and pressure drop	ESEOD ≥ 0.10 in based on $Cd = 0.65$
Response time	
Opening response time	≤100 ms
Closing response time	≤25 ms
Leakage	
Internal leakage	$\leq 1 \times 10^{-3}$ sccs GHe @ 0 psia to MEOP and over operating temperature range
External leakage	$\leq 1 \times 10^{-6}$ sccs GHe @ 0 psia to MEOP and over operating temperature range
Life	
Shelf life	≥10 years
Service life	≥10 years
Cycle life	≥250 cycles (500 cycles for qualification)
Physical characteristics	
Envelope	TBS
Weight	≤5 lbm
Component orientation	Operate in any orientation
Nameplate	Each valve shall have a part number identifier and unique serial number
Environments	
Operating fluid temperature range	–120 °F to 140 °F
Non-operating temperature range	20 °F to 120 °F
External pressure	0 to 15 psia
Humidity	0 to 100% relative humidity

Note: To Be Specified (TBS) designates a value that will be defined by the supplier and agreed to by the purchaser.

13.6.7 Gaseous Helium Pressure Regulation ('Bang-Bang') Solenoid Valve

Table 9 lists the lander design specifications for the 'bang-bang' valve. An informal vendor survey has indicated that an existing design is available with either delta-qualification effort or specification adjustments, particularly for internal leakage at low temperatures. The notional specifications and part are listed in table 9.

Table 9. Prop: GHe latch pressure regulation valve specifications.

Title	Requirement
Item definition	Unidirectional, normally closed valve
Interface Definitions	
Mechanical interface	
Mounting	Panel mounted, bolted to vehicle
I/O ports	3/8 in OD, 304 L welded tube stub
Electrical interface	
Opening voltage	27 ± 2 Vdc
Opening power	≤100 W
Holding voltage	≤12 Vdc
Holding power	≤14 W @ 12 Vdc
Connection	Electrical connector TBS
Fluids	
Operating fluid	GHe per MIL-PRF-27407
Compatible fluid	GN_2 per MIL-PRF-27401 IPA per TT-I-735
Pressure	
Operating pressure range	0 to 4,500 psia
MEOP	4,500 psia
MDP	4,500 psia
Performance Characteristics	
Pressure	
Proof pressure	1.5 × MDP
Burst pressure	2.5 × MDP
Flow and pressure drop	
Flow and pressure drop	ESEOD ≥ 0.10 in based on Cd = 0.65
Response time	
Opening response time	≤100 ms
Closing response time	≤50 ms
Leakage	
Internal leakage	≤1×10^{-3} sccs GHe @ 500 psia to MEOP and over operating temperature range
External leakage	≤1×10^{-6} sccs GHe over operating pressure and temperature ranges
Life	
Shelf life	≥10 years
Service life	≥10 years
Cycle life	≥1,000 cycles

Table 9. Prop: GHe latch pressure regulation valve specifications (Continued).

Title	Requirement
Physical characteristics	
Envelope	TBS
Weight	≤4 lbm
Component orientation	Operate in any orientation
Nameplate	Each valve shall have a part number identifier and unique serial number
Environments	
Operating fluid temperature range	–120 °F to 140 °F
Non-operating temperature range	20 °F to 120 °F
External pressure	0 to 15 psia
Humidity	0 to 100% relative humidity

Note: To Be Specified (TBS) designates a value that will be defined by the supplier and agreed to by the purchaser.

13.6.8 Pyrotechnic Valves

Table 10 lists the lander design specifications for the 3/8-in pyrotechnic valves. An informal vendor survey has indicated that currently available designs exist that will largely meet these specifications. The notional specifications and part are listed in table 10.

Table 10. Pyrotechnic valve specifications.

Title	Requirement
Item definition	Normally closed, pyro-actuated isolation valve. Low pressure (–1) and high pressure (–2) configurations.
Interface Definitions	
Mechanical interface	
Mounting	Panel mounted, bolted to vehicle
I/O ports	–1: 0.750 in OD welded tube stub –2: 0.375 in OD welded tube stub
Initiator	Two NASA Standard initiator 1A initiator 3/8-24 UNJF
Electrical interface	
Connection	MS3116E8-2S
Fluids	
Operating fluid	–1: MON-25 per MIL-PRF-26539, MMH per MIL-PRF-27404 –2: GHe per MIL-PRF-27407, MMH per MIL-PRF-27404, MON-25 per MIL-PRF-26539
Compatible fluid	GN_2 per MIL-PRF-27401, IPA per TT-I-735A, GHe per MIL-PRF-27407
Pressure	
Operating pressure range	–1: 0 to 450 psia –2: 0 to 4,500 psia
MEOP	–1: 450 psia –2: 4,500 psia
MDP	–1: 450 psia –2: 4,500 psia

Table 10. Pyrotechnic valve specifications (Continued).

Title	Requirement
Performance Characteristics	
Pressure	
Proof pressure	1.5 × MDP
Burst pressure	4.0 × MDP
Flow and pressure drop	
Flow and pressure drop	–1: ESEOD ≥0.70 in based on *Cd* = 0.65
	–2: ESEOD ≥0.10 in based on *Cd* = 0.65
Leakage	
Internal forward leakage	≤1×10^{-6} sccs GHe prior to actuation across operating pressure and temperature
External leakage	≤1×10^{-6} sccs GHe before and after actuation across operating pressure and temperature
Design requirements	
Opening response time	≤10 ms
Minimum energy margin	80% of the nominal output charge
Maximum energy margin	120% of the nominal output charge
Manufacturing and test requirements	Per NASA-SPEC-5022
Life	
Shelf life	≥10 years
Service life	≥10 years
Physical characteristics	
Envelope	TBS
Weight	–1: ≤1.50 lbm including NSI –2: ≤1.00 lbm including NSI
Component orientation	Operate in any orientation
Nameplate	Each valve shall have a part number identifier and unique serial number
Environments	
Operating fluid temperature range	–1: –40 °F to 140 °F –2: –120 °F to 140 °F
Non-operating temperature range	20 °F to 120 °F
External pressure	0 to 15 psia
Humidity	0 to 100% relative humidity

Note: To Be Specified (TBS) designates a value that will be defined by the supplier and agreed to by the purchaser.

13.6.9 Common Service Valves

The propulsion common service value specifications are given in table 11.

Table 11. Prop: Common service valve specifications.

Title	Requirement
Item definition	Manually operated fill and drain service valve. High pressure (–1) and low pressure (–2) configurations.
Interface Definitions	
Mechanical interface	
Mounting	Panel mounted, bolted to vehicle
I/O ports	–1: 0.375 in OD welded tube stub –2: 0.375 in OD welded tube stub
GSE interface	–1: 0.3125” male fitting per SAE-AS4395-05 –2: 0.25” male fitting per SAE-AS4395-04
Operating fluid	
Operating fluid	–1: GHe per MIL-PRF-27407 –2: GHe per MIL-PRF-27407, MON-25 per MIL-PRF-26539, MMH per MIL-PRF-27404
Compatible fluid	GN_2 per MIL-PRF-27401, IPA per TT-I-735
Pressure	
Operating pressure range	–1: 0 – 4,500 psia –2: 0 – 450 psia
MEOP	–1: 4,500 psia –2: 450 psia
MDP	–1: 4,500 psia –2: 450 psia
Performance Characteristics	
Pressure	
Proof pressure	$1.5 \times$ MDP
Burst pressure	$2.5 \times$ MDP
Flow and pressure drop	
Flow and pressure drop	ESEOD ≥ 0.05 in based on $Cd = 0.65$
Leakage	
External leakage valve open	$\leq 1 \times 10^{-6}$ sccs GHe @ 5 psia to MEOP, over operating temperature range
External leakage valve closed and capped	$\leq 1 \times 10^{-6}$ sccs GHe @ 5 psia to MEOP, over operating temperature range
Life	
Shelf life	≥10 years
Service life	≥10 years
Open/close cycles	≥50 cycles
Mate/demate cycles	≥50 cycles
Physical characteristics	
Envelope	TBS
Weight	≤1.0 lbm
Component orientation	Operate in any orientation
Nameplate	Each valve shall have a part number identifier and unique serial number

Table 11. Prop: Common service valve specifications (Continued).

Title	Requirement
Environments	
Operating fluid temperature range	–1: –120 °F to 140 °F –2: –40 °F to 140 °F
Non-operating temperature range	20 °F to 120 °F
External pressure	0 to 15 psia
Humidity	0 to 100% relative humidity

Note: To Be Specified (TBS) designates a value that will be defined by the supplier and agreed to by the purchaser.

13.6.10 Fluid Filters

Table 12 gives GHe/propellant filter specifications.

Table 12. GHe/propellant filter specifications.

Title	Requirement
Item definition	Filters. High pressure (–1) and low pressure (–2) configurations.
Interface Definitions	
Mechanical interface	
Mounting	Panel mounted via a P-clamp
I/O ports	–1: 3/8 in OD, 304 L welded tube stub –2: 3/4 in OD, 304 L welded tube stub
Fluids	
Operating fluid	–1: GHe per MIL-PRF-27407 –2: MON-25 per MIL-PRF-26539, MMH per MIL-PRF-27404
Compatible fluid	GN_2 per MIL-PRF-27401 IPA per TT-I-735 Demineralized water per ASTM D1193, type 2 GHe per MIL-PRF-27407
Pressure	
Operating pressure range	–1: 0 – 4,500 psia –2: 0 – 450 psia
MEOP	–1: 4,500 psia –2: 450 psia
MDP	–1: 4,500 psia –2: 450 psia
Performance Characteristics	
Pressure	
Proof pressure	1.5 × MDP
Burst pressure	2.5 × MDP
Flow and pressure drop	
Flow and pressure drop	–1: ESEOD ≥ 0.15 in based on *Cd* = 0.65 –2: ESEOD ≥ 0.50 in based on *Cd* = 0.65

Table 12. GHe/propellant filter specifications (Continued).

Title	Requirement
Filtration	
Holding capacity	–1: 0.10 g AC coarse dust –2: 1.00 g AC coarse dust
Filtration rating	≤10 µm absolute
Leakage	
External leakage	$\leq 1 \times 10^{-6}$ sccs GHe @ 0 psia to MEOP and over operating temperature range
Life	
Shelf life	≥10 years
Service life	≥10 years
Physical characteristics	
Envelope	TBS
Weight	–1: ≤3.50 lbm –2: ≤2.50 lbm
Component orientation	Operate in any orientation
Nameplate	Each valve shall have a part number identifier and unique serial number
Environments	
Operating fluid temperature range	–1: –120 °F to 140 °F –2: –40 °F to 140 °F
Non-operating temperature range	20 °F to 120 °F
External pressure	0 to 15 psia
Humidity	0 to 100% relative humidity

Note: To Be Specified (TBS) designates a value that will be defined by the supplier and agreed to by the purchaser.

13.6.11 Pressure Transducer

Characteristics of the pressure transducer are given in table 13.

Table 13. Pressure transducer characteristics.

Title	Requirement
Item definition	Pressure transducers. Low pressure (–1) and high pressure (–2) configurations.
Interface Definitions	
Mechanical interface	
Mounting	Panel mounted, bolted to vehicle
I/O ports	1/4 in OD, 304 L welded tube stub
Electrical interface	
Excitation voltage	16 – 40 Vdc unregulated
Output voltage	5.00 Vdc ± 0.5% full-scale output at 70 °F
Connection	Electrical connector TBS
Fluids	
Operating fluid	–1: GHe per MIL-PRF-27407, MMH per MIL-PRF-27404, MON-25 per MIL-PRF-26539 –2: GHe per MIL-PRF-27407
Compatible fluid	GN_2 per MIL-PRF-27401 IPA per TT-I-735

Table 13. Pressure transducer characteristics (Continued).

Title	Requirement
Pressure	
Operating pressure range	–1: 0 to 450 psia –2: 0 to 4,500 psia
MEOP	–1: 450 psia –2: 4,500 psia
MDP	–1: 450 psia –2: 4,500 psia
Performance Characteristics	
Pressure	
Proof pressure	$1.5 \times$ MDP
Burst pressure	$2.5 \times$ MDP
Leakage	
External leakage	$\leq 1 \times 10^{-6}$ sccs GHe across operating pressure and temperature.
Accuracy	
Linearity	±0.20% full-scale output
Hysteresis	±0.20% full-scale output
Repeatability	±0.10% full-scale output
Life	
Shelf life	≥10 years
Service life	≥10 years
Physical characteristics	
Envelope	TBS
Weight	≤0.63 lbm
Component orientation	Operate in any orientation
Nameplate	Each valve shall have a part number identifier and unique serial number
Environments	
Operating fluid temperature range	–1: –40 °F to 140 °F –2: –120 °F to 140 °F
Non-operating temperature range	20 °F to 120 °F
External pressure	0 to 15 psia
Humidity	0 to 100% relative humidity

Note: To Be Specified (TBS) designates a value that will be defined by the supplier and agreed to by the purchaser.

APPENDIX A—PLUME SURFACE INTERACTION RISK

Rocket PSI (fig. 31) describes the lander environment due to the impingement of hot rocket exhaust on regolith of planetary bodies. This environment is characterized by the plume flow physics, cratering physics, and ejecta dynamics.

Because the lander design has multiple pulsed engines and because the nozzle exit is so close to the lunar surface, the PSI risk is especially high. The lander and propulsion designs may:

- Lead to extensive cratering and exceed tilt requirements.
- Result in heavy ejecta that can damage the lander, instrumentation, and limit the range that NDL can operate over.
- The multiple-plume physics can lead to high aerothermal environments and destabilizing aerodynamics.

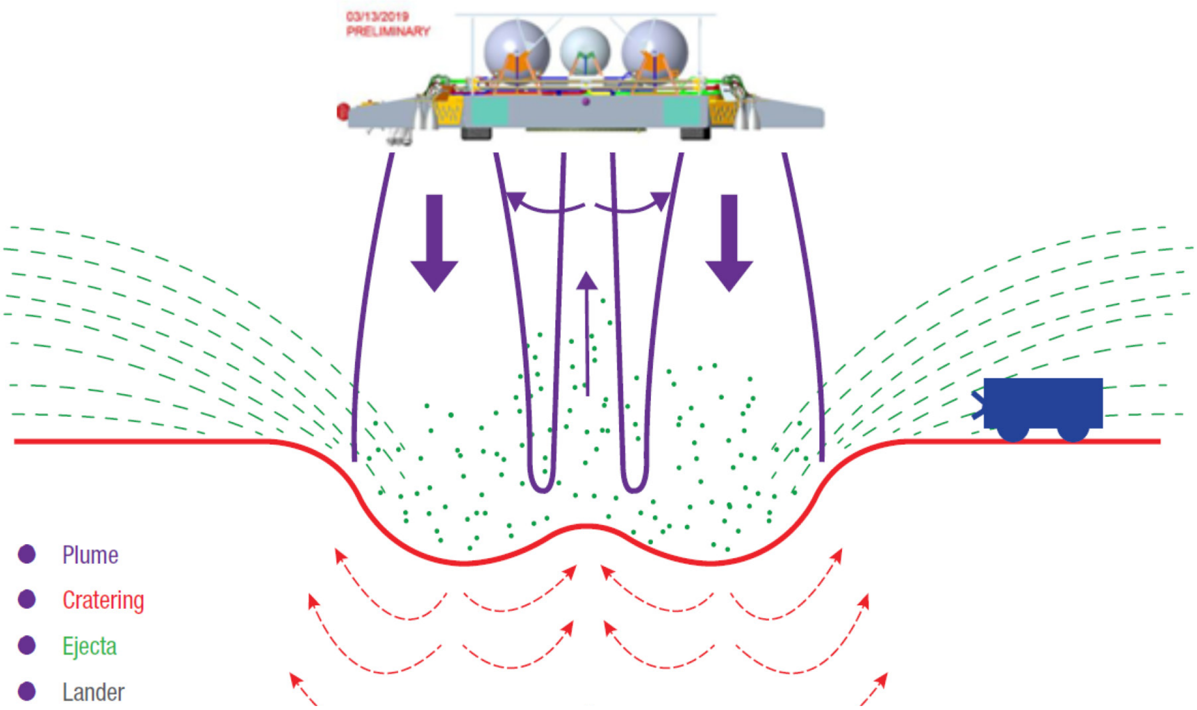

Figure 31. Plume-surface interaction.

REFERENCES

1. Craig, A.S.; Holt, J.P.; Hannan, M.R.; and Orphee, J.I.: "Mission Design for the Lunar Pallet Lander," AAS 19-330, https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20190001933.pdf.

2. Orphee, J.; Hannan, M.; Braden, E.; et al.: "Guidance, Navigation, and Control for NASA Lunar Pallet Lander," AAS 19-033, https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20190002111.pdf.

3. Anzalone, E.; Braden, E.; Ahmad, N.; et al.: "Guidance and Navigation Design Trades for the Lunar Pallet Lander," AAS 19-091, https://ntrsreg.nasa.gov/archive/sti.nasa.gov/20190002112.pdf.

The following referenced documents are available upon request:

- NASA Procedural Requirements (NPR) 8705.4: Risk Classification for NASA Payloads
- Ames Procedural Requirements (APR) 8070.2: Class D Spacecraft Design and Environmental Test
- AIAA Mass Properties Control for Space Systems: S-120A
- LPL Master Equipment List: LPL-CI-001
- LPL Instrumentation List: LPL-CI-003
- LPL Command and Data Dictionary: LPL-CI-004
- LPL EEE Parts Control Requirements: LPL-REQ-001
- LPL Level 2 Requirements: LPL-REQ-002
- LPL Level 3 Requirements: LPL-REQ-003
- LPL Environmental Specification: LPL-SPEC-001

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

1. REPORT DATE *(DD-MM-YYYY)* 01–11–2019	**2. REPORT TYPE** Technical Publication	**3. DATES COVERED** *(From - To)*

4. TITLE AND SUBTITLE
NASA Lunar Lander Reference Design

5a. CONTRACT NUMBER
5b. GRANT NUMBER
5c. PROGRAM ELEMENT NUMBER

6. AUTHOR(S)
L.D. Kennedy

5d. PROJECT NUMBER
5e. TASK NUMBER
5f. WORK UNIT NUMBER

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
George C. Marshall Space Flight Center
Huntsville, AL 35812

8. PERFORMING ORGANIZATION REPORT NUMBER
M–1492

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/TP—2019–220391

12. DISTRIBUTION/AVAILABILITY STATEMENT
Unclassified-Unlimited
Subject Category 18
Availability: NASA STI Information Desk (757–864–9658)

13. SUPPLEMENTARY NOTES
Prepared by the Exploration Systems Development Branch, Space Transportation Directorate

14. ABSTRACT
In preparation for robotic missions to the lunar poles and building on lander development history around the Agency, NASA created a 'pallet lander' concept to deliver a 300-kg rover to the surface to search for and characterize volatiles. The lander was designed with simplicity and low cost in mind, generally employing single-string systems, minimal mechanisms, and existing technologies, though some enhancements were targeted in areas such as precision landing. This Technical Publication describes the lander, its mission, and its subsystems, specifically relating to interfaces and support to the payload. Notional hardware is shown where applicable.

15. SUBJECT TERMS
lander, lunar pallet lander

16. SECURITY CLASSIFICATION OF: a. REPORT	b. ABSTRACT	c. THIS PAGE	17. LIMITATION OF ABSTRACT	18. NUMBER OF PAGES	19a. NAME OF RESPONSIBLE PERSON / 19b. TELEPHONE NUMBER *(Include area code)*
U	U	U	UU	86	STI Help Desk at email: help@sti.nasa.gov / STI Help Desk at: 757–864–9658

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

National Aeronautics and
Space Administration
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George C. Marshall Space Flight Center
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