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COMPASS Final Report: 2008 International Lunar Network (ILN)

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subject to revision as analysis proceeds.

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Contents

List of Tables

List of Figures

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1.0 Executive Summary

The Compass team developed three designs for a soft lander for the International Lunar Network (ILN) using different power systems. These systems included solar arrays and batteries and two radioisotope power system (RPS) options scaled down from previous designs. These are a half Stirling radioisotope generator and a one-eighth Radioisotope Thermoelectric Generator (RTG). The COMPASS design consists of a soft-lander capable of network science (seismic, magnetospheric, laser reflector, and thermal transfer) that can land anywhere on the near side of the Moon and operate for at least 6 years. The

* Currently retired.

baseline version, using only batteries and solar arrays, requires a *very* large battery (~80 kg) for the roughly 20 W night-time power requirements. The lander launch mass (with solid rocket breaking motor) is almost 900 kg. With such a large battery, the lander will require a launcher larger than the desired Minotaur V; a Taurus II (being developed for NASA Commercial Orbital Transportation Services (COTS)) or larger (Delta II or Atlas V) would be needed. The two RPS cases, half an Advanced Stirling Radioisotope Generators (ASRG) and one-eighth RTG have much smaller launch masses of 500 and 600 kg, respectively. (The one-eighth RTG case lacks the power for the full science package, so a solar array battery is still needed.) Unfortunately, the Minotaur V's launch capability is still not large enough (~430 kg to trans-lunar injection (TLI)) for the RPS systems. Consequently, a fourth case was developed to fly a limited science floor suite of instruments (seismometer only) which reduced the RPS Lander launch mass to fit on a Minotaur V.

Table 1.1 collects the details of the subsystems at a top level in the Baseline design as done in the trades studies, Case 2 (see Figure 1.1).

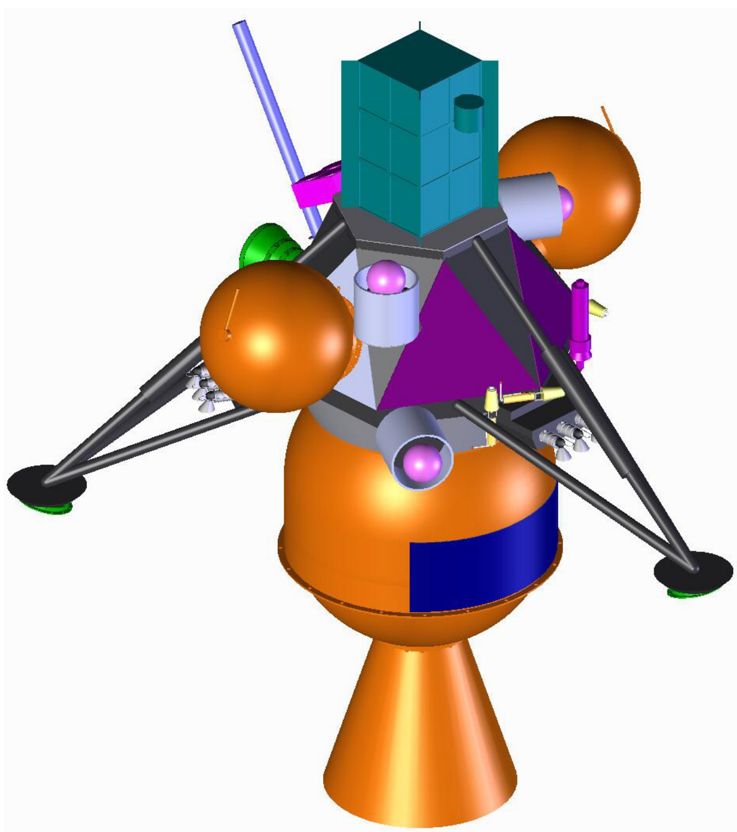


Figure 1.1.—Case 2 Baseline ASRG powered ILN Soft Lander.

TABLE 1.1.—MISSION AND SPACECRAFT (S/C) SUMMARY

Subsystem area	Details	Total mass with growth
Top Level System	Stack mass consists of expendable launch vehicle (ELV) adaptor (29.5 kg), lunar descent stage (Star motor and propellant (302.5 kg)), and ILN Lander with propellant to land softly on the lunar surface (159.7 kg) and science payload (20.2 kg). An additional system level growth (23.3 kg) is carried to bring the total growth on dry mass to 30%.	511.9 kg (with additional 23.3 kg at system)
Mission, Operations, Guidance, Navigation and Control (GN&C)	Star Tracker, sun sensors, mini-Inertial Measuring Unit (IMU), radar system for landing (Phoenix)	15.8 kg
Launch	Taurus II launch to TLI, direct lunar injection.	Delivered mass 1039 kg
Science	Seismometer (bottom deployed), magnetometer, laser reflector, subsoil thermal probe (Mole experiment deployed)	20.2 kg
Power	Half single ASRG for lunar night and solar arrays for transit power, battery backup for landing power	31 kg
Propulsion	Star 27 solid rocket motor for descent and disposed of before landing, pulsed hydrazine propulsion system for final landing using radar	Descent stage: 302.5 kg, Lander: 159.7 kg
Structures and Mechanisms	Predeployed, energy absorbing leg landing system, Al-Li hexagonal structure	31.5 kg
Communications	1 W radio frequency (RF) power S-band antenna (5 W DC power), pointable helical antenna	6.2 kg
Command and Data Handing (C&DH)	SCS750 main processor with 500 Mb of storage	4.7 kg
Thermal	Max heat rejection of 25 W, lunar surface operation (day and night), Radioisotope Heater Units (RHU) for heat during lunar night, Micrometeoroid and Orbital Debris (MMOD) shield on radiator, multilayer insulation (MLI), cold plates for electronics cooling	11.7 kg

2.0 Study Background And Assumptions

2.1 Introduction

Taking advantage of the current focus on lunar exploration, NASA is leading an international effort to establish a network of geophysical monitoring stations on the Moon. The venture, known as the ILN seeks to place between four and eight such bases at selected locations on the Moon in the next decade. Many of the nodes will be launched and operated by different national space agencies, but all will work together as a unified monitoring network. According to Jim Green, director of NASA’s Division of Planetary Science, this model of international cooperation could then serve as a template for a similar venture on Mars (Ref. 1).

At the Lunar and Planetary Science Conference in March 2008, the Associate Administrator for NASA’s Science Mission Directorate (SMD) announced NASA’s plans to build a Lunar Network with the help of international partners.

2.1.1 ILN Mission Subsystem Component

From the official request for Information, the ILN is described as follows (Ref. 1):

- Solicitation number: N/A
- Reference number: RFI–04–28–2008
- Posted date: April 29, 2008
- FedBizOpps posted date: April 29, 2008
- Original response date: May 14, 2008
- Current response date: May 14, 2008

> *"The ILN represents a series of U.S. and International Partner provided surface packages (sensing nodes), which act as common science nodes in a lunar geophysical network that will address Agency science goals. The NASA SMD expects that each node in ILN will provide a minimum core suite of two instruments and will include a lander to deliver these nodes to the lunar surface. Additional measurements and/or instruments may be accommodated provided adequate mass, power and budget margins exist. The mission addressed by this RFI encompasses two landers (i.e., anchor nodes). These first two nodes of the ILN will likely be placed at high lunar latitudes.*
>
> *The landers are expected to be small. For planning and RFI purposes, the following represent approximate anticipated payload/instrument accommodation considerations and constraints that will be used to assess lander capability and sizing applicability.*
>
> *Mass: 25-50 kg, Power: 1 W continuous, 2 W peak G-Load: 40 g Data Rate: 100 Mbits per Earth day (transmitted or stored)"*

2.2 Assumptions

Table 2.1 gathers the top-level assumptions and a listing of trades considered in each subsystem as the ILN design session was performed.

2.3 Growth, Contingency and Margin Policy

Mass Growth: The COMPASS team uses the ANSI/AIAA R–020A–1999, *Recommended Practice for Mass Properties Control for Satellites, Missiles, and Launch Vehicles* (Ref. 2). Table 2.2 shows the Percent Mass Growth Allowance (MGA) separated into a matrix specified by level of design maturity and specific subsystem.

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

From the Discovery Announcement of Opportunity (AO): Definitions of Contingency and Mass

Contingency (or Reserve), when added to a resource, results in the maximum expected value for that resource. Percent contingency is the value of the contingency divided by the value of the resource less the contingency.

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

Power Growth: The COMPASS team uses a 30 percent margin on the bottoms up power requirements in modeling the power system. See Section 3.1.1 for the power system assumptions.

TABLE 2.1.—ASSUMPTIONS AND STUDY REQUIREMENTS

Subsystem area	Assumptions and study requirements	Critical trades
Top-level	ILN of soft landers, four to eight at various lunar locations all Earth viewable: First two at high lunar latitudes Gather lunar/external science data for 6 years (or more) Figures of merit (FOM): Delivered mass, risk/location, science data, mission success probability, cost	Lifetime, location
System	Off-the-shelf (OTS) equipment where possible, [Technology Readiness Level (TRL) 6 cutoff 2011, 2014 launch year] Mass Growth per ANSI/AIAA R-020A-1999 (add growth to make system level 30%) zero-fault tolerant	Fault tolerance, launch year
Science	24/7 science operations once landed, seismometer, magnetometer, thermal probe (Mole), retroreflector, 100 Mb/day	GN&C cruise and landing instruments, hard landings, penetrators
Mission, operations, GN&C	Direct lunar trajectory (4 days) with crasher stage (surveyor-like), closed loop, pulsed landing system, assume landing azimuth can be kept to ± 15° and lands at beginning of lunar day, 350 m/s secondary (hydrazine), 2200 m/s primary (solid rocket), *constant* science operations on the Moon.	GN&C cruise and landing instruments, hard landings, penetrators
Launch vehicle	Taurus II, Minotaur V, $C_3 = -2$ km^2/s^2, 1190 kg, 466 kg respectively Adapter: TBD Launch Loads: Axial 12 ±1 g, Lateral 4 ± 1 g	Taurus II, Atlas, Delta II, Falcon 9, Athena II
Propulsion	Primary: Star 27 solid rocket Secondary (cruise, Attitude Control System (ACS), terminal landing): blow-down hydrazine, nine-15 lbf thrusters pulsed for landing, nine-1 lbf thrusters for ACS	Trade: secondary propulsion system (green)
Power	80 W cruise, 160 W landing, 20 to 30 W science operations, solar arrays and radiators *not* pointable Battery storage or RPS for 14 day eclipse, 28 V bus voltage, assume full brick (RPS systems), RPS	Array type, battery type, half Stirling, one-eighth thermoelectric, polar landing
Avionics/ communications	Science run from central controller, 1.4 GB data storage, 10 kb/s, single omni antenna, store/dump data return every 14 days, no relay satellite, single landing radar	Computer type, X band or Ka Band
Thermal and environment	Louvered radiator (transmitters, batteries, RPS) Tank heaters Deep space radiation level at 1 AU Cold plate/heat pipe heat transfer system S/C insulation and thermal control paints Phase change thermal energy storage	Trading RHU and heaters, louvers and variable heat pipes, radiator placement
Mechanisms	Minimize Science vibration interference	Deployment of science to surface
Structures	Primary: hexagonal, dimensions 0.5 by 0.5 m , truss, Al-Li, Secondary: 4% of stage components	Developing structure model, need ELV loads
Cost	Utilize Master Equipment List (MEL) and iterate with subsystems for new design, development, test, and evaluation (DDT&E)	New technology vs. existing heavier technologies
Risk	Identify major risks	Lifetime, location

TABLE 2.2.—PERCENT MGA

Code	Design maturity (basis for mass determination)	Percent MGA									
		Electrical/electronic components			Structure	Thermal control	Propulsion	Batteries	Wire harnesses	Mechanisms	Instrumentation
		0-5 kg	5-15 kg	>15 kg							
E	Estimated (preliminary sketches)	30	20	15	18	18	18	20	50	18	50
L	Layout (or major modification of existing hardware)	25	20	15	12	12	12	15	30	12	30
P	Prerelease drawings (or minor modification of existing hardware)	20	15	10	8	8	8	10	25	8	25
C	Released drawings (calculated values)	10	5	5	4	4	4	5	5	4	5
X	Existing hardware (actual mass from another program)	3	3	3	2	2	2	3	3	2	3
A	Actual mass (measured flight hardware)	0	0	0	0	0	0	0	0	0	0
CFE	Customer furnished equipment	0	0	0	0	0	0	0	0	0	0

2.4 Science Background

- Currently the mission means the first two U.S. nodes. They will be stations in a geophysical network, so potential instruments are
  - Seismometers
  - Heat-flow probes
  - Magnetic/electric field sensors
  - Retroreflectors
- The instruments will need to be continuously operational for (at least) 6 years, which justifies the need to consider “radioisotope power sources.”
- The mission is cost-capped at $200M for two stations, and they will likely need to be soft landers, so that is the constraint for “small.”
- Science objectives still in pre-phase-A formulation.
- ~30 W assumed for science payload power requirements (Ref. 3).

2.5 Science Objectives

Table 2.3 outlines the science objectives and the rationale and measurement requirements for each.

TABLE 2.3.—SCIENCE OBJECTIVES AND RATIONALE

	Science objectives	Science rationale	Measurement requirements
Seismometry	1. Understand the current seismic state and determine the internal structure of the Moon.	Seismic detection of lunar tectonic events will enable determination of the internal structure and composition of a differentiated planetary body. Understanding how strong Moonquakes are generated and where they occur has implications for the site of the lunar base.	Measure lunar seismicity using broad-band seismometry at multiple geometrically dispersed locations.
Heat flow	2. Measure the interior lunar heat flow to characterize the temperature structure of the lunar interior.	Heat flow measurements constrain the abundance of radiogenic elements, lateral variations in crustal and upper mantle composition, and the nature of the thermal evolution in a differential body.	Determine thermal conductivity by in situ heating at multiple depths. Determine the thermal gradient via long-term monitoring at multiple depths at each location.
EM Sounding	3. Use electromagnetic sounding to measure the electrical conductivity structure of the lunar interior.	Interior temperature and composition can be inferred from conductivity, allowing joint interpretation with seismology and heat flow. Enables additional measurements of crustal magnetization and the space-physics environment.	Measure ambient electric and magnetic fields at each station. Quantify contribution on EM measurements by local plasma field.
Retroreflectors	4. Increase ability to determine deep lunar structure and conduct tests of gravitational physics by laser ranging to the Moon by installing next-generation retroreflectors.	Highly accurate laser ranging to the Moon reveals small irregularities in the lunar rotation due to tidal changes of the Moon's shape and the effects of the lunar mantle and core. Ranging also enables tests of gravitational physics and improvement of the lunar orbit.	<2 cm range accuracy (measurements done from Earth).

2.6 Science Instrument Requirements

Table 2.4 outlines the instrument requirements in relationship to the mission requirements.

The science payload for Cases 1, 2, and 3 shown in the MEL in Table 2.5 was adapted from the descriptions in Table 2.4. All items but the first instrument (Work Breakdown Structure (WBS) 06.4.1), a seismometer, were removed to try to fit an ILN design onto a Minotaur V in Case 4.

To assess integration impacts, several representative science instruments were chosen based on the ILN science team's Objective Matrix and used in the science subsystem MEL. The instruments are:

- Seismometer design from Netlander
- Thermal experiment deployed by PLanetary Underground TOol (PLUTO) Mole
- Magnetometer from Rosetta Lander
- Laser retroreflector from Jason

TABLE 2.4.—SCIENCE INSTRUMENT REQUIREMENTS

	Mission requirements	Instrument requirements	Mass power, thermal, data, etc.
Seismometry	• Four simultaneously operating nodes • Continuous operation for 1 lunar tidal cycle (6 yr) • Inter-station timing accuracy ~5 msec • See separate site selection criteria • Instrument attached to ground and vibrationally isolated from S/C	• Three-axis very broad band (VBB) seismometers • Dynamics range of ~24 bit • For f > 1.0 Hz, sensitivity of 10 to 9 m/s. High frequency cutoff should be ~20 Hz • For 0.1 f > 1.0 Hz, sensitivity of 10 to 11 m • For 0.001 < f <0.1 Hz, sensitivity at least 2 x 10 to 11 m/s • Thermal stability of the instrument to ±5 • Thermally blanket the ground to ~1 m radius from the instrument • Attitude knowledge of deployed instrument	• Instrument-only mass = 6 kg • Instrument-only power = 2 W (peak), 1 W (continuous), 0.2 W (low power) • 100 Mb of data per Earth day; no downlink drivers • Plus power for an instrument heater (TBD W)
Heat flow	• Temperature sensor array must extend to at least 3 m depth • Thermal conductivity measurements require good contact with the regolith • Minimize the thermal effect of the S/C • Placed 200 to 300 km from major terrain boundaries • Continuous operation for 2 yr	• Each thermal conductivity measurements records thermal decay for ≥1 lunar day • Each sensor to measure T every 6 to 12 hr • Temperature sensor precision: 0.05 to 0.001 K • Minimize nine thermal conductivity measurements and nine temperature measurements • Each sensor spaced 30 cm apart	• Temperature sensor ~10g each • Instrument only power = 0.2 W (per sensor) • Sensor deployment mass and power unknown • Mole mass 2 kg, power 5 W (peak); includes full instrumentation
EM sounding	• Continuous operation for 1 yr • No contact requirements • Both magnetometers and one electrometers must be at least 2 m away from S/C • Magnetometers deployed in orthogonal directions • Langmuir probe must be 0.5 m vertically from S/C	• DC to 100 Hz • 2 x 3-component magnetometers (10 pT/rtHz) • 1 x 2-component electrometer (100 uV/m/rtHz) • 1 x Langmuir probe (on a vertical mast; 500 K, 10 e/cm^3) • Temperature sensor for calibration • Attitude knowledge of deployed instruments	• 2 to 5 kg for sensors, booms, cabling, avionics • 3 to 6 W continuous power • 10 to 100 Mb/day continuous • No downlink drivers
Retroreflectors	• Reflector array oriented within (3° to 15°—TBD) of direct Earth view • Station separation >90° in lat and/or Ion	• Retroreflector array of dimensions 10- by 10-cm	• Instrument-only mass ~1 kg • Deployment/hinge ~1 kg • Power = 0 W (deployment mech needs power) • Data = 0 bit

TABLE 2.5.—ILN SCIENCE PAYLOAD MEL

WBS Number	Description International Lunar Network (may 2008)	QTY	Unit Mass (kg)	CBE Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	**Robotic Lunar Lander - Landing ILN**			754.6	6%	47.0	801.6
06.1	ELV Adaptor			25.0	18%	4.5	29.5
06.2	Lunar Descent Stage			483.0	1%	4.3	487.3
06.3	Lunar Lander			231.1	15%	33.6	264.7
06.4	ILN Science Payload			15.5	30%	4.7	20.2
06.4.1	Passive Science Experiement (Seismometer)	1.0	6.0	6.0	30%	1.8	7.8
06.4.2	Magnetometer Experiment (includes sensors, booms, ca	1.0	5.0	5.0	30%	1.5	6.5
06.4.3	Reflector	1.0	2.0	2.0	30%	0.6	2.6
06.4.4	Heat Flow Experiment (Pluto MOLE)	1.0	2.5	2.5	30%	0.8	3.3

2.6.1 Science Instrument Details

2.6.1.1 Seismological Package—Example: NETLANDER Mission by European Space Agency (ESA)/NASA

- SEIS description:
  - https://www.seis-insight.eu/fr/public/l-instrument-seis/netlander
  - The overall mass of the SEIS experiment is 2.3 kg.
  - (Measured) power budget is about 3 W, but a power need reduction action is in progress (target is about 1 W). This budget includes all sensors and a dedicated (decentralized) avionics, based on a powerful LEON core.
  - Acquisition will be performed by a series of 24 bit A/D, while the thermal and drift control will be performed by a feedback generated by a 24 bit D/A.
- Industrial development:
  - Currently at the end of the B phase, with a breadboard of the axis already delivered by industry (EADS-Sodern) in July 2004.
  - Most critical parts have been tested, including shock tests (200g, 20 ms) for pivot, electronic components and displacement sensors.

The seismometer is deployed after landing, and is tethered to the S/C. The seismometer is dropped out of the bottom of the lander and into the ground.

2.6.1.2 Magnetometer

ROsetta Lander MAgnetometer and Plasma Monitor (ROMAP) consists of the Rosetta Lander-Magnetometer (ROLAND) and the Lander Plasma Monitor (SPM). The Fluxgate-Magnetometer, designed and built lead-managed by the Institute for Geophysics and extraterrestrial Physics, is situated in the center of the experiment. It consists of two entwined ringcores plus pick-up coils and Helmholtz coils for each sensor axis. With a weight of less than 40 g it can perform measurements between ± 2000 nT with a resolution of 10 pT. The IWF (Graz) and the MPE Garching joined the development and construction of the magnetometer and its electronics. More information is available at the ROSETTA URL: https://www.esa.int/Science_Exploration/Space_Science/Rosetta.

TABLE 2.6.—ROSETTA SPACECRAFT

ROMAP	Parameter
Sensor mass	35 g
Sensor volume	523 cm^3
Electronics mass	150 g
Resolution	10 pT
Dynamic range	4000 nT
Sensor noise (at 1 Hz)	10 pT/(Hz)1/2
Bandwidth	0 to 32 Hz
Sample rate	[1; 64] vec/s
Power (including plasma monitor)	1 W
Temperature range	–160 ... +120 °C

2.7 PLUTO Mole

The PLUTO Mole was developed and used on the Beagle 2 British led mission to Mars.
Beagle2 Mission websites:

- http://www.beagle2.com/index.htm
- http://www.beagle2.com/technology/mole.htm

"Managed by the Germany Aerospace Center (DLR) Cologne, the Mole will provide mobility to the stationary lander. Pluto, (planetary undersurface tool) as the mole is called, has the ability to crawl across the surface (Beagle 2 Mission) at the rate of 1cm every 5 sec, using a compressed spring mechanism to propel a drive mass. Samples are collected in a cavity in the tip which opens when the mole reaches a sampling location"

2.8 Lunar Landing Site Selection

Convolving suitable Moonquake tests with desires from heat flow and seismic access to the far side leads to the following approximate sites for the first two nodes:

- Station 1: –5° S, 75° W
- Station 2: 30° N, 75° E

Figure 2.1 illustrates the ILN soft lander landing sites.

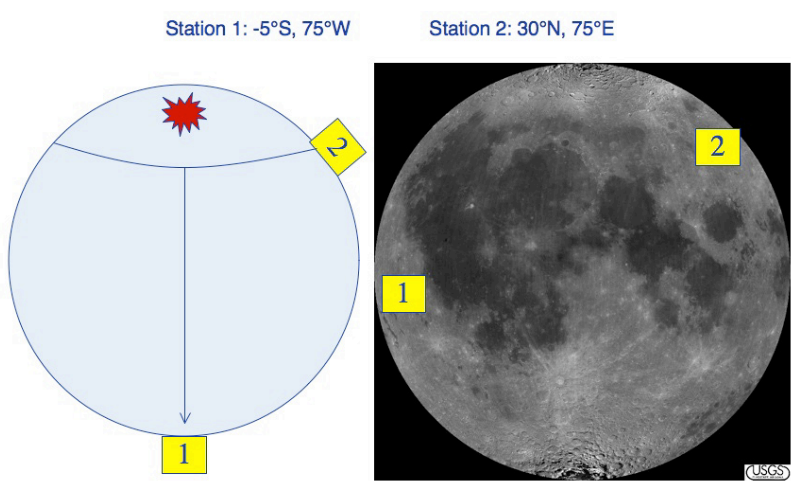


Figure 2.1.—ILN soft lander landing sites.

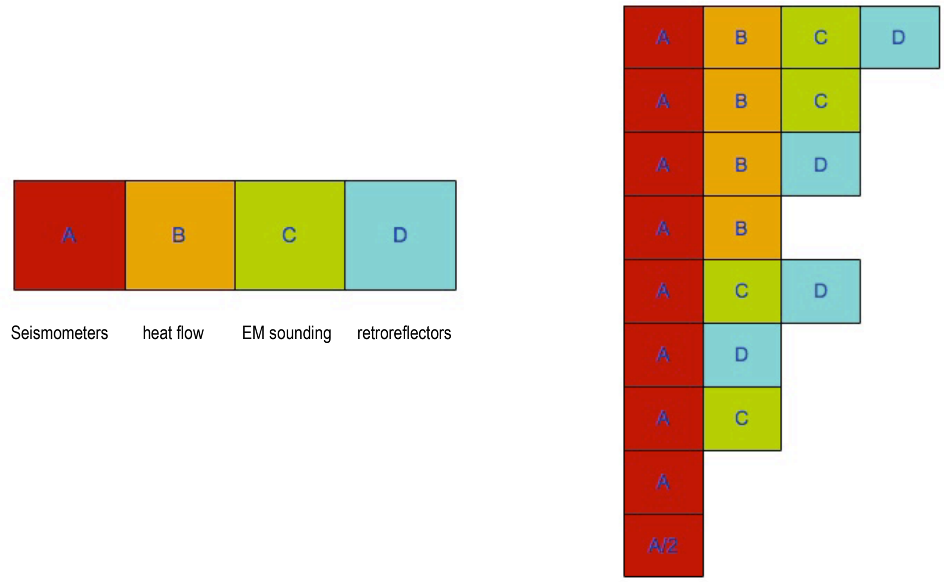


Figure 2.2.—Science descope tree.

2.9 Science Baseline, Descope and Floor

Science Baseline: Use seismometry, heat flow, electromagnetic sounding, and laser retroreflectors to obtain complementary geophysical data from a network of four nodes operating simultaneously and continuously for 6 years (1 lunar tidal cycle).

To provide minimum functionality in an ILN lander, the science mission goals were descoped to a science floor. (Figure 2.2 illustrates descoping from the baseline to the floor.) This was the minimal science mission required of the ILN Lander design. The science floor was used as the payload in Case 4. The science floor is defined as follows:

Science Floor: Determine the deep interior velocity structure of the Moon and place constraints on the core size/density of the science samples by operating two broadband seismometers placed in specific locations simultaneously and continuously for 2 years.

2.10 Mission Description

Each subsection below describes an aspect of the ILN mission.

2.10.1 Launch Vehicle Details

Initially, the ILN design cases used the Taurus II as the launch vehicle. Cases 1, 2, and 3 used Taurus to launch. A Minotaur V launch was the goal of the design in Case 4, and reductions in both payload and structural mass were made (as well as reductions in requirements on science) in order to fit onto the smaller launch vehicle performance of the Minotaur V. Case 4 was an attempt to reduce mission cost by fitting onto a less expensive launch vehicle than the Taurus II.

2.10.1.1Taurus II Performance

From the website, the description of the Taurus II reads as follows:

> *"Taurus II is a two-stage launch vehicle designed to provide responsive, low-cost, and reliable access to space for medium-class payloads weighing up to 5750 kg. Currently*

under development to demonstrate commercial re-supply of the International Space Station under a COTS contract, the Taurus II launch system utilizes identical management approaches, engineering standards, production and test processes common to Orbital's family of highly successful small-class Pegasus®, Taurus®, and Minotaur launch vehicles. These proven launch technologies, along with hardware from one of the world's leading launch vehicle integrators, combine to provide cost-effective access to a variety of orbits for civil, commercial and military Delta II-class payloads." (Ref. 4)

2.10.1.2 Minotaur V Performance

From the website the description of the Minotaur V reads as follows:

"Minotaur V is a five-stage evolutionary version of the Minotaur IV Space Launch Vehicle (SLV) to provide an extremely cost-effective capability to launch small S/C into high energy trajectories, including geosynchronous transfer orbits (GTO), as well as translunar and beyond. The Minotaur V concept leverages Orbital's flight proven heritage of the Minotaur family of launch vehicles, as well as the commercial Pegasus and Taurus SLVs, to create a low-risk, readily-developed system." (Ref. 5)

2.10.1.3 Minotaur V Configuration

From the website the description of the Minotaur V configuration reads as follows:

"The first three stages of the Minotaur V are the unmodified Peacekeeper solid rocket motors that are provided by the U.S. Air Force Rocket System Launch Program (RSLP). The fourth and fifth stages are commercial motors that can be selected to provide varying levels of performance. The stage four motor is a Star 48V configuration. The fifth stage can be either attitude controlled or spinning. For a spin- stabilized upper stage, a Star 37FM is used to provide maximum performance. A Star 37FMV, with gimbaled, flexseal nozzle, is used on a 3-axis stabilized version." (Ref. 5)

2.10.2 Mission Analysis Assumptions

The TLI burn is performed by the launch vehicle from a 300 km circular orbit around the Earth. A direct entry will be performed at the Moon with a final landing requirement that the velocity of the vehicle be 0.00 m/s at 10 ft above the lunar surface.

The direct entry maneuver onto the lunar surface is a split-descent: a lunar descent burn and a lunar landing burn. The lunar descent burn is performed using the solid rocket, Star 27/Star30BP/Star24C, and removes most of the vehicle velocity. The lunar landing burn is a chemical propulsion burn and removes the remaining velocity such that the 0.0 m/s 10 ft above the lunar surface requirement is met. Since the thrust-to-weight ratio of the vehicle in all four cases is relatively high, it is assumed that the gravity losses are negligible, and the impulsive Delta-V can be used.

The lunar insertion stage (Star27/Star30BP/Star24C) and the lunar lander are both active payload elements during the lunar descent burn. However, following the lunar descent burn the lunar insertion stage is jettisoned and only the lunar lander payload is active during the lunar landing burn.

An additional 10 m/s RCS Delta-V has been allocated for trans-Earth mid-course corrections and 11 m/s RCS Delta-V has been allocated for attitude control during the lunar descent burn.

2.10.3 Mission Analysis Analytic Methods

A table was compiled for a previous COMPASS study, CD-2006-03, "Low Cost Robotic Lunar Lander Study" (Ref. 6), that presents the integrated Delta-V required for descent to the lunar surface during a direct entry for a range of thrust-to-weight ratios. Above a thrust-to-weight ratio of 4, instantaneous Delta-V was assumed (see Table 2.7).

This Delta-V was further broken down to represent the split-descent with 88 percent being allocated for the lunar descent burn using the Star27/Star30BP/Star24C and the remaining 12 percent used during the lunar landing burn by the main chemical propulsion system. This is the same split that was assumed in Surveyor.

The mass history of the vehicle is tracked throughout all phases of the mission. The mass depleted as a result of the descent and landing burns is calculated using the known mass of the vehicle prior to the burns as well as the performance characteristics of each of the propulsion systems. The burn times associated with each maneuver are calculated in a similar manner. Included in the total mass of each element in addition to the required main engine propellant and Reaction Control System (RCS) propellant is an inert element mass. This mass includes boil-off and unusable propellant. The sum of the total element masses is then the delivered mass required by the launch vehicle.

2.10.4 Mission Analysis Event Timeline

- Mission event timeline from ILN mission (see Figure 2.3)
- Launch from Earth
- Checkout
- Loiter to TLI window opening
- TLI opening to ignition
- TLI burn
  - Case 1,2,3: Taurus II
  - Case 4: Minotaur V
- Earth Departure Stage (EDS) disposal
- Trans-Earth mid-course corrections
  - Performed by lunar lander RCS
- Trans-lunar coast
  - Approximately 3 days
- Lunar descent burn
  - Performed by solid rocket
    - Case 1: Star 30BP
    - Case 2: Star 27
    - Case 3: Star 27
    - Case 4: Star 24C
- Lunar landing burn
  - Performed by main chemical propulsion

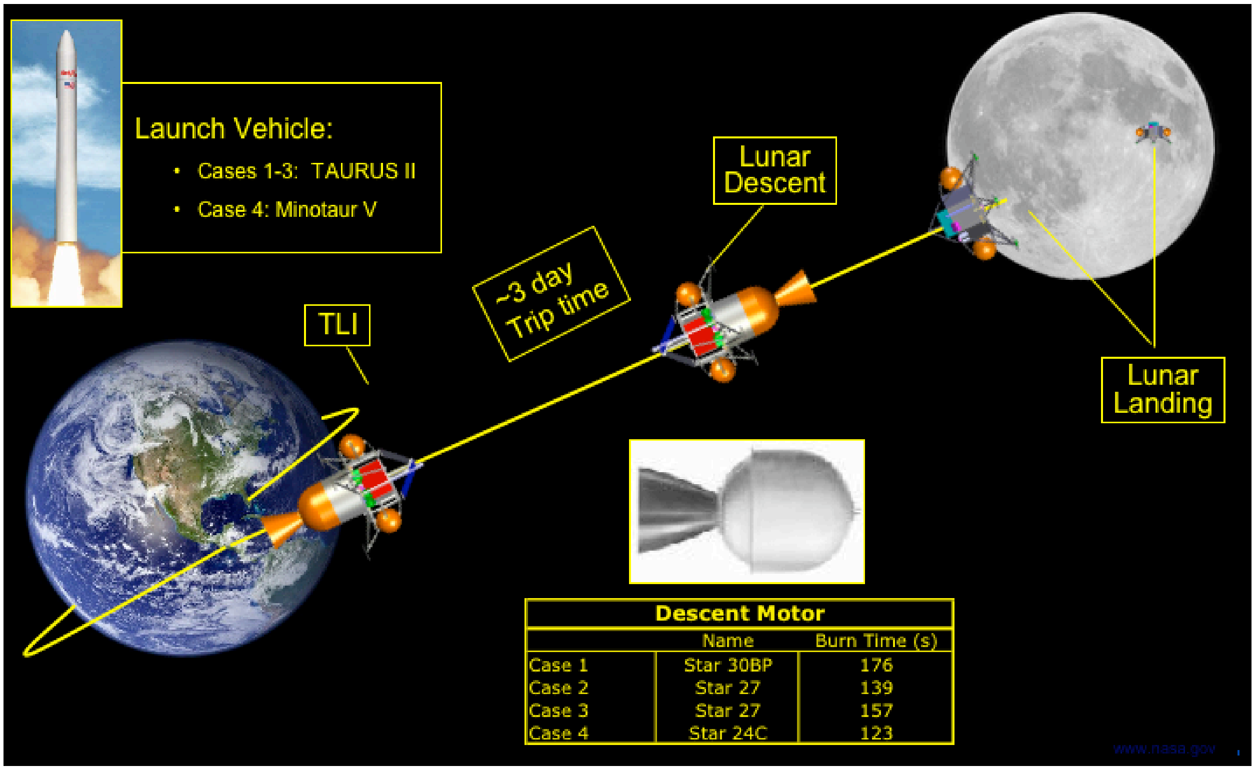


Figure 2.3.—Chemical translunar mission timeline.

TABLE 2.7.—DELTA-V REQUIRED FOR DESCENT DURING DIRECT ENTRY FOR A RANGE OF THRUST-TO-WEIGHT RATIOS

T/W	^{2}v decent, km/s	Starting altitude, km	Descent TOF, min
2.0	2.835	49.9	25.5
2.5	2.721	16.5	19.1
3.0	2.687	8.2	15.6
3.5	2.664	4.1	13.2
4.0	2.633	1.1	11.3
>4.0	2.490	0.5	5.0

2.10.5 Landing Azimuth

Assuming a landing azimuth with in ± 15° can be guaranteed (see Figure 2.4).

- The following can be prepointed
  - Radiators
  - Solar arrays
  - Communications antennas
  - Science
- Based on landed site

Impact on the GN&C package

- Presently have the ability to roll the vehicle
- The IMU should have the knowledge to guide the rolling of the vehicle during final descent <15° landing azimuth
- Ensures solar arrays, radiators, science and communication antennas pointed nearly optimal
- Communications antenna prepointed by latitude/longitude before launch

2.10.6 Mission Trajectory Delta V

Table 2.8 illustrates the trajectory delta V by mission phase.

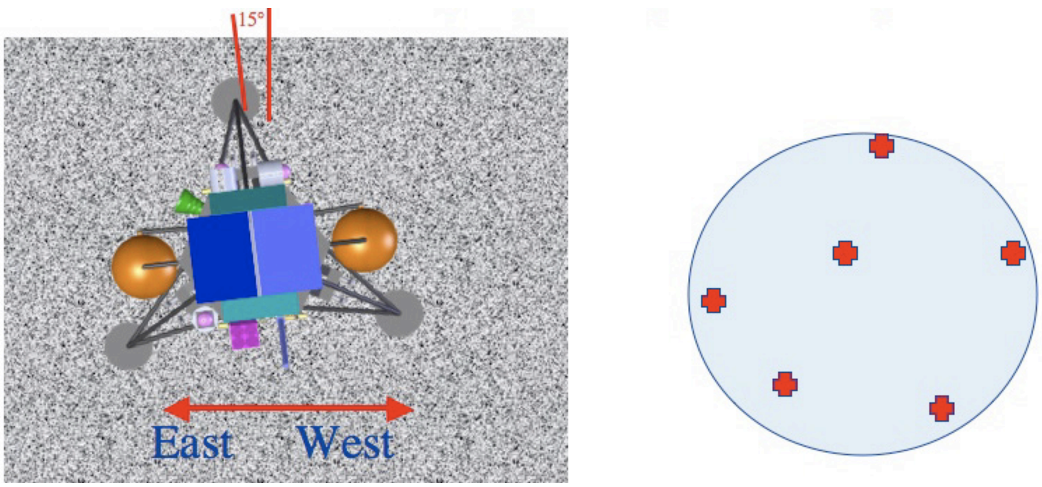


Figure 2.4.—Landing azimuth and lunar ILN sites of interest.

TABLE 2.8.—TRAJECTORY DELTA V BY MISSION PHASE

Case 1, 2, 3, and 4				
Phase name	Main DV, m/s	Main propulsion item	RCS DV, m/s	RCS propulsion item
Launch from earth				
Checkout				
Loiter to TLI window opening				
TLI opening to ignition				
TLI burn	0			
EDS disposal				
Trans-Earth midcourse corrections			10	Lander
Trans-lunar coast				
Lunar orbit capture burn	0			
Coast				
LOI plane change burn				
Coast				
Lunar orbit circularization burn				
Lunar descent burn	2191	Solid	11	Solid
Lunar landing burn	319	Lander		

2.10.7 ILN Visibility Analysis

Visualization analysis was performed within Satellite Orbit Analysis Program (SOAP) to determine that each of the two lunar landers has the capability to communicate with the ground stations of the U.S. for at least 9 to 14 hr per Earth day during lunar sunlit periods. Both of the landers targeted a ground station on the east and west coasts of the U.S. to best represent coverage throughout the country. As a result, the communication to the east coast lags the west coast by about 3 hr due to the rotation of the Earth.

The first lunar lander is assumed to be located at –5° S, 75° W, and the second at 30° N, 75° E. Figure 2.5 and Figure 2.6 show the communication availability between the Earth ground stations and lunar ground stations throughout different durations. Power assumptions such as lack of communication during lunar eclipse phases were not included in the analysis.

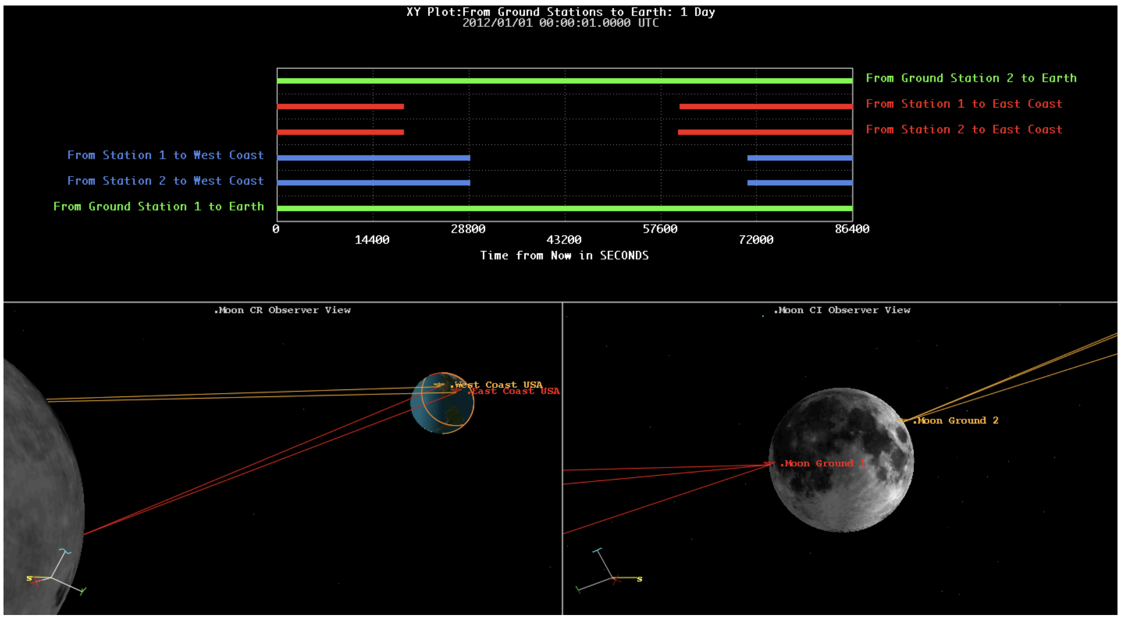


Figure 2.5.—ILN visibility analysis over a year.

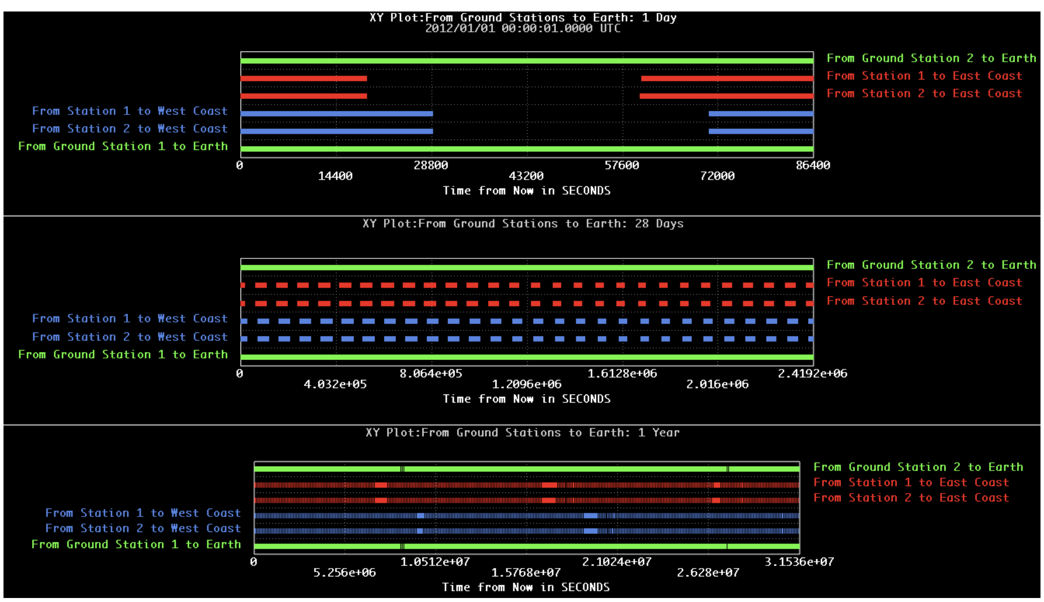
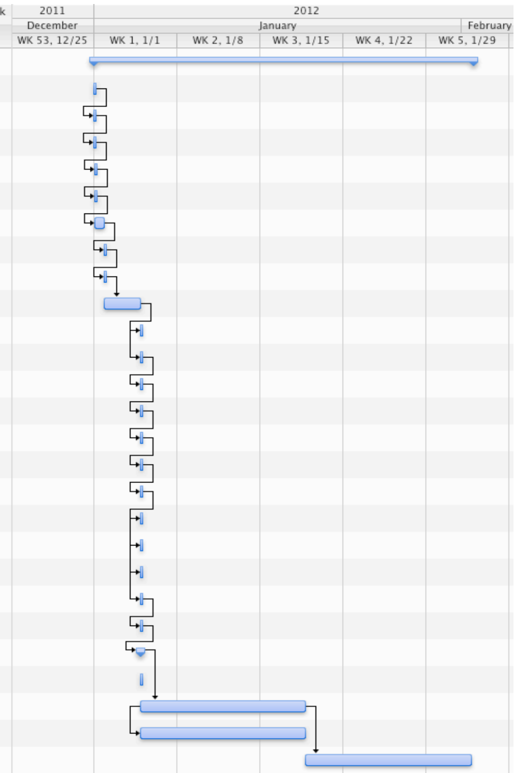


Figure 2.6.—ILN visibility analysis.

TABLE 2.9.—MISSION OPERATIONS EVENTS

#	Title	Given Planned Work
0	**ILN_Operations**	
1	Launch	10 minutes
2	Propellant Heater Elements Activated	
3	Loiter to TLI Window Opening	90 minutes
4	TLI Opening to Ignition	10 minutes
5	TLI Burn	6 minutes
6	Spacecraft Checkout	18 hours
7	Propellant Temperature Maintenance	
8	Trans-Earth Mid-Course Corrections	1 hour
9	Trans-Lunar Coast	72 hours
10	Reaction Control System Burn	566 seconds
11	Propellant Isolation Valves (open)	
12	Lunar Descent Burn	202.7 seconds
13	Solid Rocket Separation	
14	Landing Thrusters Activation	
15	Propellant Isolation Valves (open)	
16	Lunar Landing Burn	476.3 seconds
17	All Propellant Isolation Valves (closed)	
18	Propellant Temperature Maintenance (stopped)	
19	RCS, Landing Thrusters Powered Off (catalyst bed heaters)	
20	Vent Pressurant Gas from Tanks (minimize leakage)	
21	Deploy Science Instruments	
22	**Deliver Nodes to High Lunar Latitudes**	
23	(Each node provides a minimum core suite of two instruments, and one lander)	
24	Lunar Science Sunlit	14 days
25	Lunar Communications (8 hours per Earth day)	14 days
26	Lunar Science Eclipse	14 days



2.10.8 ILN Operations Timeline

There are about 78 cycles of lunar sunlight and lunar eclipse over the duration of the 6-year period of operation. Table 2.9 provides a listing of the major mission operations events.

2.11 System Design Trade Space—Study Starting Points

- Using a small class launch vehicle, design the architecture necessary to deliver a robotic lander to the Moon
- Objective: Provide a low cost (~$100) capability to place small payloads (10s of kg of science or tech demos) on the Moon
- Science sampler gathering requirements: at least 1 week duration (7 Earth days), through a Mars moon lunar day, with global access of the lunar surface
- Power for payload 25 W
- Launch vehicle options
    - Minotaur I, IV, V (Ref. 7)
    - Falcon 1
    - Taurus II
    - Atlas V
- Stages to design
    - TLI
    - Lunar capture and descent
    - Lunar Lander
- Propulsion options
    - Chemical: Solids, etc.

- Electric propulsion
- Combination of both

- Start with previous missions
    - Surveyor (Ref. 8) shown in Figure 2.7, etc.

2.12 Baseline System Design (Case 2)

After the analysis and design of Cases 1 to 4 using differing power technologies to power the ILN over the lunar night, Case 2 using the ½ ASRG as the power source was chosen as the baseline design for this report based on its performance. All subsystems documented in Section 5.0 will detail the subsystem designs as it applied to Case 2. Section 8.0 will provide top-level details of all four cases as well as the bottom line payload capabilities and design.

2.13 ELV Packaging

Figure 2.8 illustrates the Taurus II payload fairing and fairing envelope.

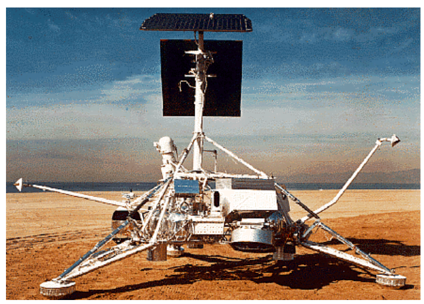


Figure 2.7.—Surveyor.

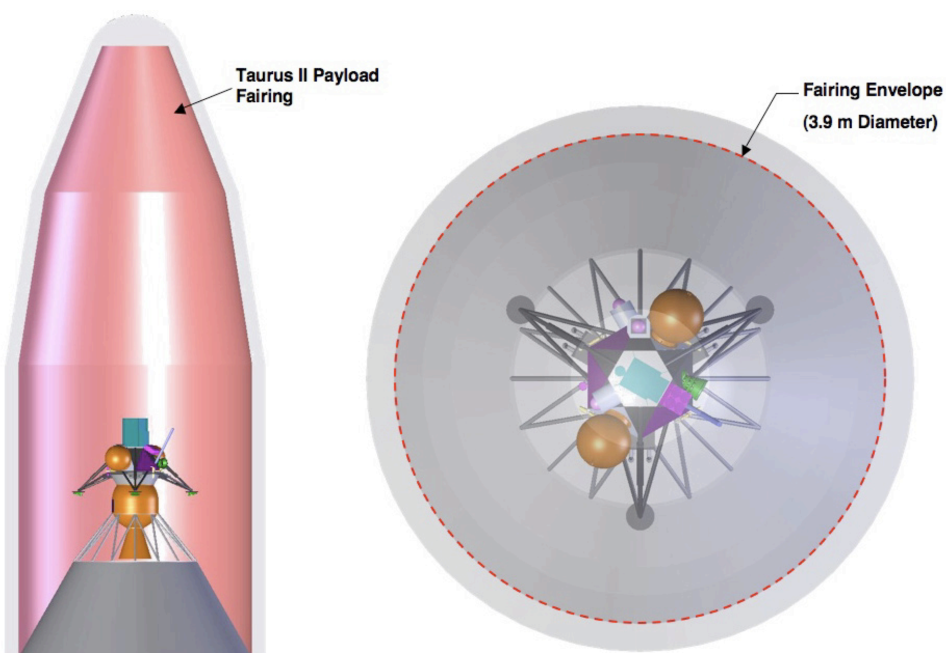


Figure 2.8.—Case 2, ½ ASRG Case in Taurus II payload fairing.

2.14 Internal COMPASS Details

COMPASS is a multidisciplinary collaborative engineering team whose primary purpose is to perform integrated vehicle systems analysis and provide trades and designs for both Exploration and Space Science Missions.

2.14.1 GLIDE Study Share

GLIDE (GLobal Integrated Design Environment) is a data collaboration tool that enables secure transfer of data between a virtually unlimited number of sites from anywhere in the world. GLIDE is the primary tool used by the COMPASS design team to pass data real-time between subsystem leads.

2.14.2 GLIDE Architecture

GLIDE details for the team are captured as follows.
Architecture: ILN

2.14.3 GLIDE Study Container(s)

Table 2.10 lists the GLIDE study containers for this report.

TABLE 2.10.—GLIDE STUDY CONTAINERS: TRADE SPACE CASES

Study case	Summary	Study container
Case 1	ILN lander using traditional solar arrays/battery system to sustain power over the lunar night, Taurus II launch vehicle and Star 27 motor	ILN_case1
Case 2	ILN lander using ½ an ASRG to sustain power over the lunar night, Taurus II launch vehicle and Star 27 motor	ILN_Case2_asrg
Case 3	ILN lander using RPS to sustain power over the lunar night, Taurus II launch vehicle and Star 27 motor	ILN_Case3_rps
Case 4	ILN lander using ½ an ASRG to sustain power over the lunar night, Minotaur V launch vehicle and Star 24 motor	ILN_Case4_asrg_minV

3.0 Baseline Design

Case 2 from the trade study summary was chosen as the baseline design for this report.

3.1 Top Level Design (MEL and PEL)

This section calls out the details of the chosen baseline for this study. Given the performance and complexity of the cases outlined in Section 2.14.3, the baseline was chosen to be Case 2, ½ ASRG for power through the lunar night. Section 8.0 details the other three cases and the details of their subsystems and overall performance.

Table 3.1 lists the top level of the MEL of the design with all the subsystem line elements hidden such that only the top level masses are shown. The total growth on the dry mass of the S/C is then rolled up to find a total growth mass and growth percentage. Engineers enter in the current best estimate (CBE) mass for each of their line elements, as well as quantity. Then the growth column is where each subsystem lists the recommended growth factor on each line items following the AIAA Weight Growth Allowance (WGA) schedule outlined in Table 2.3 in Section 2.5. The MEL takes all of the items and racks them up into totals and calculates a total CBE mass, a total mass and a total growth mass.

Table 3.1.—MEL

WBS Number	Description International Lunar Network (may 2008)	QTY	Unit Mass (kg)	CBE Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	**Robotic Lunar Lander - Landing ILN**			481.7	6%	30.2	511.9
06.1	ELV Adaptor			25.0	18%	4.5	29.5
06.1.1	C&DH, Communications GN&C			0	0%	0.0	0.0
06.1.2	Propulsion Hardware (Chemical)			0	0%	0.0	0.0
06.1.3	Propellant Management (Chemical)			0	0%	0.0	0.0
06.1.4	Propellant (Chemical)			0	0%	0.0	0.0
06.1.9	Electrical Power			0	0%	0.0	0.0
06.1.10	Thermal Control (Non-Propellant)			0	0%	0.0	0.0
06.1.11	Structures & Mechanical Systems			25	18%	4.5	29.5
06.2	Lunar Descent Stage			301.0	0%	1.5	302.5
06.2.1	C&DH, Communications GN&C			0	0%	0.0	0.0
06.2.2	Propulsion Hardware (Chemical)			27	4%	1.1	28.6
06.2.3	Propellant Management (Chemical)			0	0%	0.0	0.0
06.2.4	Propellant (Chemical)			272	0%	0.0	271.8
06.2.9	Electrical Power			0	0%	0.0	0.0
06.2.10	Thermal Control (Non-Propellant)			0	0%	0.0	0.0
06.2.11	Structures & Mechanical Systems			2	20%	0.4	2.1
06.3	Lunar Lander			140.2	14%	19.6	159.7
06.3.1	C&DH, Communications GN&C			22	21%	4.7	26.7
06.3.2	Propulsion Hardware (Chemical)			7	8%	0.6	7.5
06.3.3	Propellant Management (Chemical)			18	12%	2.1	19.7
06.3.4	Propellant (Chemical)			32	0%	0.0	31.6
06.3.9	Electrical Power			25	24%	5.9	31.0
06.3.10	Thermal Control (Non-Propellant)			10	18%	1.8	11.7
06.3.11	Structures & Mechanical Systems			27	17%	4.6	31.5
06.4	ILN Science Payload			15.5	30%	4.7	20.2

Where the MEL (Table 3.1) captures the bottoms up estimation of CBE and growth percentage line item by item from the subsystem designer. Table 3.2 wraps up those total masses, CBE and total mass after applied growth percentage. In order to meet the total of 30 percent at the system level, an allocation is necessary for system level growth. This additional system level mass of 23.3 kg as shown in Table 3.2, is assumed as part of the inert mass that is flown along the required trajectory. Therefore, the additional system level growth mass impacts the total propellant loading for the mission design.

The Total wet mass of the stack is 535.2 kg after the additional system level growth is applied. The available launch performance to TLI of the Taurus II was assumed to be 1038.5 kg. This leaves a launch margin of 535.8 kg. Out of this launch margin, the mass of the payload adaptor to the launch vehicle will be subtracted. This is the portion of the payload adaptor that stays with the launch vehicle. The adaptor is in the master MEL as WBS element 06.1 with a mass with growth of 29.5 kg.

The total inert mass of the landed payload (i.e., the lander on the Moon's surface) is calculated by taking the total wet mass of the stack, subtracting the sum of (1) ELV adaptor dry mass with the additional 13 percent growth mass (total 32.5 kg), (2) the Lunar Descent Stage dry mass with additional 13 percent system growth mass (total 38 kg) as well as the total used propellant for the descent state Star Motor (271.8 kg) and lander stage RCS propellant (31.6 kg). This calculation results in 161.4 kg landed onto the surface of the Moon. All of these values are shown in the Table 3.2 in the system summary.

TABLE 3.2.—CASE 2 SYSTEM SUMMARY SHEET

COMPASS study: Mars Moon Sampler Mission				**Study Date**	**6/18/08**
GLIDE container	*ILN: ILN_case2_asrg*				
Spacecraft Master Equipment List Rack-up (Mass)				**COMPASS S/C Design**	
WBS	**Main Subsystems**	**CBE Mass (lkg)**	**Growth (kg)**	**Total Mass (kg)**	**Aggregate Growth (%)**
06	Robotic Lunar Lander - Landing ILN	**481.7**	**30.2**	**511.9**	
06.1	*ELV Adaptor*	**25.0**	**4.5**	**29.5**	18%
06.1.11	Structures & Mechanical Systems	25.0	4.5	**29.5**	18%
06.2	*Lunar Descent Stage*	**301.0**	**1.5**	**302.5**	0%
06.2.1	C&DH, Communications GN&C	0.0	0.0	**0.0**	TBD
06.2.2	Propulsion Hardware (Chemical)	27.5	1.1	**28.6**	4%
06.2.3	Propellant Management (Chemical)	0.0	0.0	**0.0**	TBD
06.2.4	Propellant (Chemical)	271.8	0.0	**271.8**	0%
06.2.9	Electrical Power	0.0	0.0	**0.0**	TBD
06.2.10	Thermal Control (Non-Propellant)	0.0	0.0	**0.0**	TBD
06.2.11	Structures & Mechanical Systems	1.8	0.4	**2.1**	20%
06.3	*Lunar Lander Stage*	**140.2**	**19.6**	**159.7**	14%
06.3.1	C&DH, Communications GN&C	22.1	4.7	**26.7**	21%
06.3.2	Propulsion Hardware (Chemical)	6.9	0.6	**7.5**	8%
06.3.3	Propellant Management (Chemical)	17.7	2.1	**19.7**	12%
06.3.4	Propellant (Chemical)	31.6	0.0	**31.6**	0%
06.3.9	Electrical Power	25.1	5.9	**31.0**	24%
06.3.10	Thermal Control (Non-Propellant)	9.9	1.8	**11.7**	18%
06.3.11	Structures & Mechanical Systems	26.9	4.6	**31.5**	17%
06.4	*ILN Science Payload*	**15.5**	**4.7**	**20.2**	30%
	Estimated Stack Dry Mass	**178.4**	**30.2**	**208.5**	**17%**
	Estimated ELV Adaptor Inert Mass	**25**	**5**	**30**	**18%**
	Estimated Lunar Descent Stage Inert Mass	**29**	**1**	**31**	**5%**
	Estimated Lunar Lander Stage Inert Mass	**110**	**20**	**130**	**18%**
	Estimated Science Payload Inert Mass	**15.5**	**4.7**	**20.2**	**30%**
	Estimated Stack Wet Mass	**481.7**	**30.2**	**511.9**	
System LeveL Growth Calculations					**Total Growth**
	Total Estimated Growth		30.2		
	Dry Mass Desired System Level Growth	178.4	53.5	**231.9**	**30%**
	Additional Growth (carried at system level)		23.3		**13%**
	Total Wet Mass with Growth	482	53.5	**535.2**	
		Bottoms up (kg)	**Growth (kg)**	**Total Mass (kg)**	
	Total Spacecraft Dry Mass	178.4	53.5	231.9	
	Total ELV Adaptor Dry Mass	*25.0*	*7.5*	*32.5*	
	Total Lunar Descent Stage Dry Mass	*29.3*	*8.8*	*38.0*	53.5
	Total Lunar Lander Stage Dry Mass	*108.6*	*32.6*	*141.2*	0.0
	Total Science Payload Dry Mass	*15.5*	*4.7*	*20.2*	
	Estimated Stack Dry Mass (on pad)	178.4	53.5	**231.9**	
	Estimated Stack Inert Mass (for traj.)			**233.4**	
	Total S/C Wet Mass (on pad w/ adaptor)	482	53.5	**535.2**	
	Total SC Wet Mass w/ Growth (no adaptor)			**502.7**	
	Total Landed Spacecraft (inert)			**161.4**	
	Available Launch Performance to TLI (kg)			1038.5	375
	Launch margin available (kg)			**535.8**	

3.1.1 Power Equipment List (PEL)

The PEL gathers all of the power requirements from each subsystem over each phase of the mission (Table 3.3). The power requirements of the 4-day transit to the Moon required a solar array for all four cases examined in this study. The approximately 30 min landing power requirements resulted in the use of batteries for all options. The eclipse power is at a premium due to the 14-day duration (23 W).

TABLE 3.3.—PEL

	Notional time	Propulsion (W)	Avionics (W)	Communication (W)	Thermal (W)	GN&C (W)	Power (W)	Science (W)	CBE total (W)	30 % margin	Total (W)
Launch	10 m	20	7	5	6	22	7	0	67	20.175	87
S/C checkout	24 hr	20	7	5	6	30	7	6	81	24.3	105
Cruise	4 d	20	7	0	6	24	7	0	64	19.275	84
Cruise communications		20	7	5	6	23	7	0	68	20.475	89
Lunar landing	30 m	180	7	5	6	67	19	0	284	85.2	369
Lunar science sunlit	14 d	0	7	0	1	0	3	9	20	5.94	26
Lunar communications	8 hr/d	0	7	5	1	0	3	9	25	7.5	33
Lunar science eclipse	14 d	0	7	0	1	0	3	7	18	5.34	23

Eclipse power was minimized by

- Storing science data—no communications
- Thermal 'bottle' approach used (New Horizons) to utilize avionics waste heat and louvers on the radiators to keep interior of S/C warm
  - Losses estimated to be ~7 W through MLI, structure, wiring but could be larger with higher fidelity analysis
  - Alternative options include more heater power, RHUs, or use of RPS waste heat
- Science instruments external to the lander (e.g., seismometer) may need RHUs

The PEL was created as a result of numerous mission and design assumptions. The 4-day lunar transit requires solar arrays for power to the S/C. However, batteries are required during the landing phase. A power margin of 30 percent has also been added to each of the subsystems throughout each phase of the mission for assurance.

During the lunar eclipse phase, (approximately 14 days of no sunlight) the science data is stored, and all communications are shut down. After sunlight has returned, the science data will be transmitted back to Earth. The thermal system will utilize the waste heat of the avionics system and louvers on the radiators to keep the interior of the S/C warm. The combination of MLI, structure, and wiring creates an approximate 7 W loss of waste heat. Some possible options to minimize the losses include use of more heater power, RHUs, or RPS waste heat.

3.2 System Level Summary

Case 2 utilizing the ½ ASRG as shown in Figure 3.1 was selected as the baseline design. The basic design is centralized around a S/C bus with a lower hexagonal frame and deck and a smaller hexagonal top frame that is rotated 30° from the lower deck. The structural members that connect the upper and lower frames connect at the vertices of the hexagonal frames creating 12 triangular faces (six large and six smaller) around the bus. Two of the larger triangular faces provide the structural interface for the externally mounted tanks. Mounting the tanks externally helped reduce the overall bus size and provides some thermal isolation between the tanks and internal components. Three of the larger triangular faces provide the required radiator area, while the remaining large triangular face provides a location for the star tracker. This can be seen in Figure 3.2 that shows a top-down view of the S/C with all components labeled.

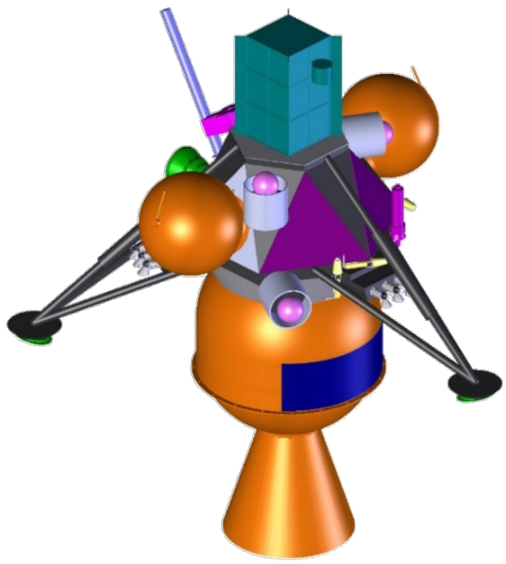


Figure 3.1.—Case 2 Lander, ½ ASRG Case, with Star 27 rocket motor attached.

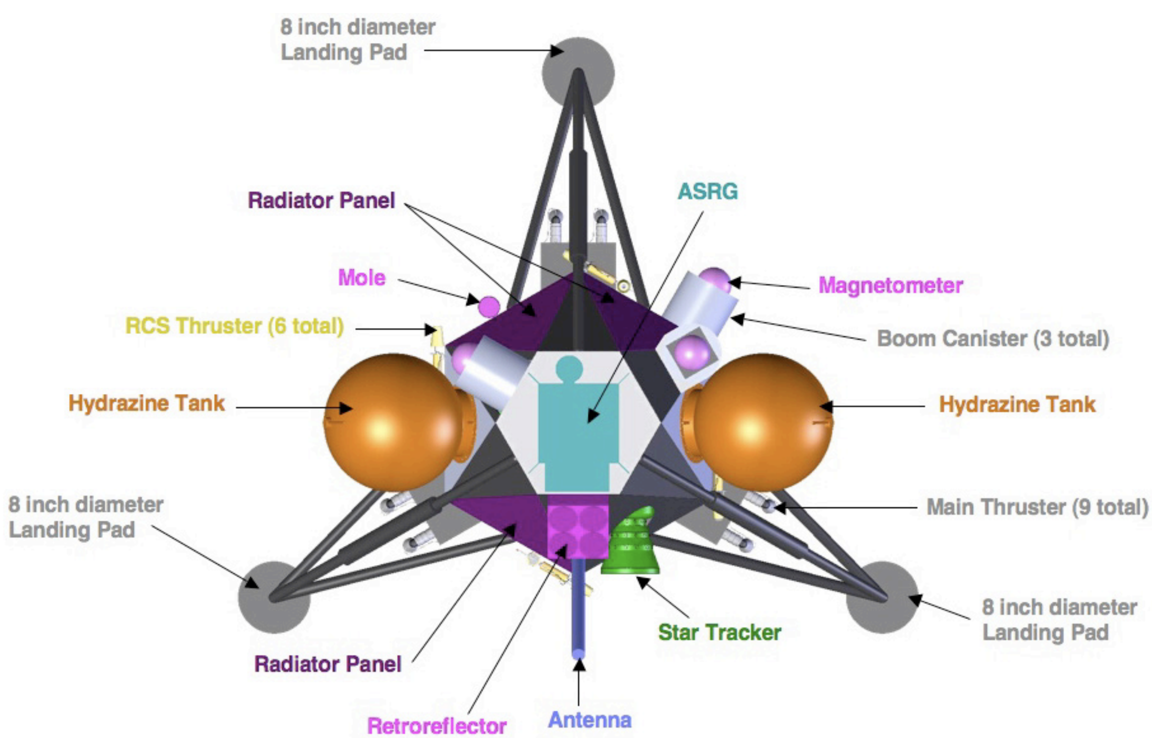


Figure 3.2.—Case 2 top-down view with component callouts.

In addition to the tanks and radiators, the other components that are externally mounted to the lander include the antenna, ½ ASRG, primary and secondary thrusters, radar altimeters, and science hardware. The antenna is mounted to the bottom plate of the bus structure and extends out of the lander at an angle 30° from vertical to help ensure a good view of Earth without the use of a gimbal mechanism. The science Retroreflector is mounted at the top of the bus and is pointed in the same direction as the antenna, again to obtain the best possible view of Earth without the use of a mechanism for pointing. Three boom canisters are mounted orthogonally from each other and contain the science hardware used for measuring the magnetic fields on the lunar surface. The Mole, used for measuring the soil temperatures below the lunar surface, is mounted along the bottom of the bus to allow for easy deployment and access to the surface. The top of the lander contains the ½ ASRG. This location was selected to allow the ASRG to radiate heat with minimal interference from other components. The main thrusters are composed of three pods of three thrusters each and are located within the landing support structure, while the RCS thrusters are located around the base of the bus structure. Three radar altimeters used for landing are located on the bottom of the three landing pads as shown in Figure 3.3.

The components internal to the lander are also shown in Figure 3.3. These components include the IMU, radar electronics box, avionics processor, battery, power management box, and a seismometer. All the internal components, with the exception of the seismometer are mounted to the bottom plate of the bus structure. Deployment of the seismometer will be performed by “dropping” it out from an opening in the bottom deck to the lunar surface. By locating the seismometer in the middle of the bus, the lander will be able to provide some shading to the seismometer and surrounding surface.

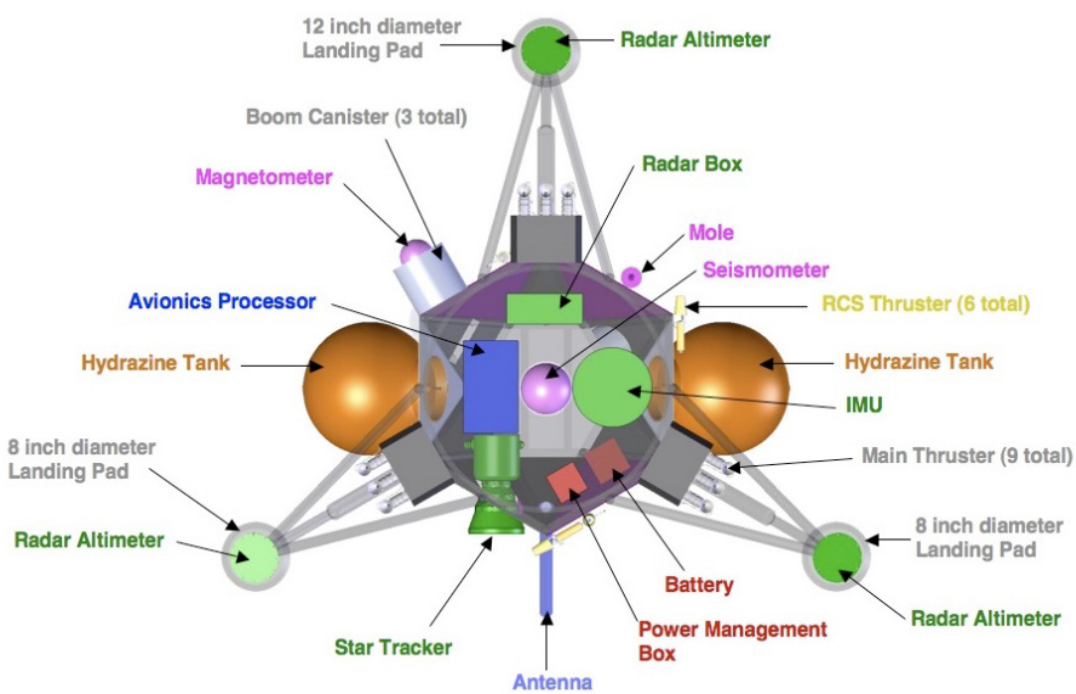


Figure 3.3.—Case 2 bottom-up view with component callouts.

3.3 Design Concept Drawing and Dimensions

The overall "prelanded" dimensions of the Case 2 Lander are shown in Figure 3.4. Of particular note is the 21.6 cm distance between the bottom of the landing pads and the bottom of the primary thruster pods. Once landed, this dimension decreases to about 8.9 cm after a displacement of 12.7 cm is incurred on the landing legs to reduce the landing force. This will in turn reduce the "prelanded" vehicle height to 99.1 cm after landing, producing a ratio of footprint diameter to Lander height of around 2:1.

Given that the ½ ASRG does not produce the amount of power required during the landing maneuver, a solar array is needed to boost the power. Since the solar array was no longer needed once landed, and given the shortage of space on the lander, the solar array was located on the STAR 27 solid rocket motor as shown in Figure 3.5.

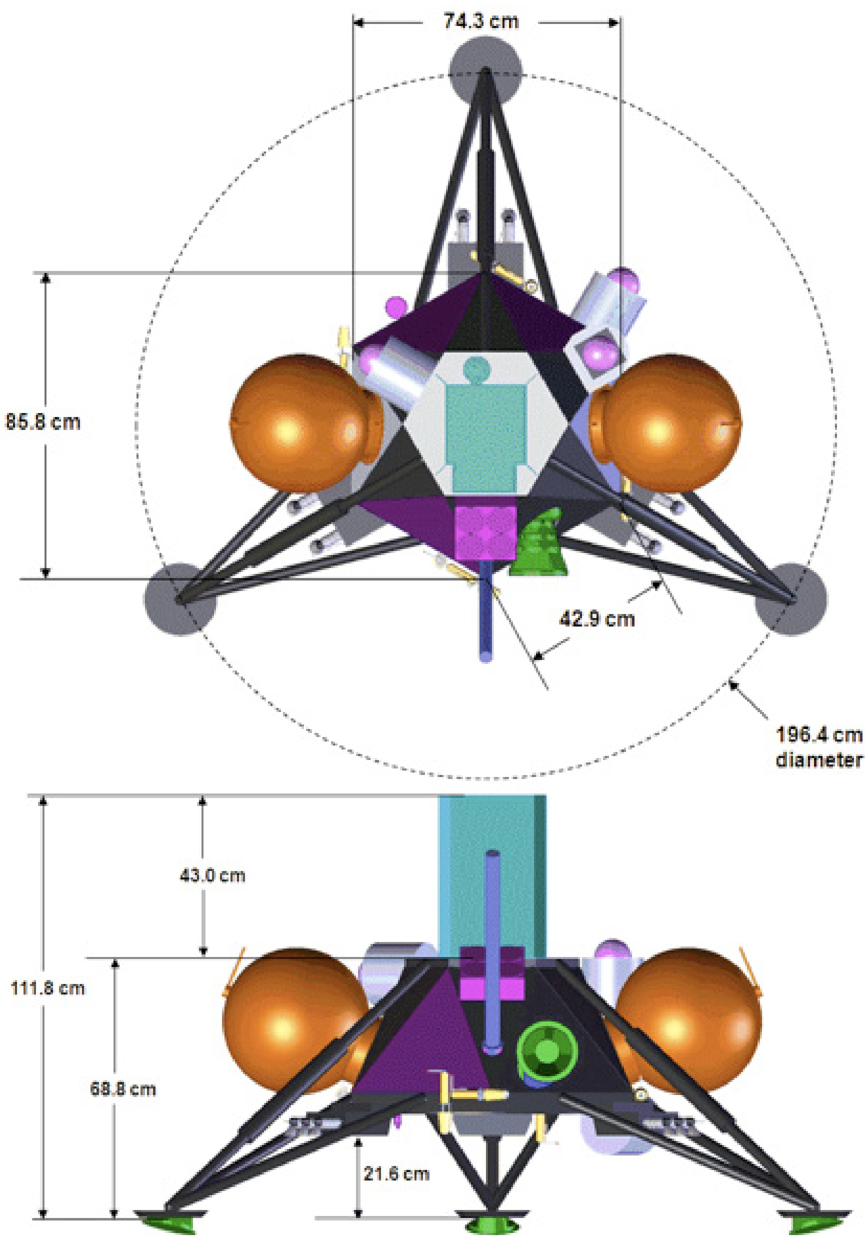


Figure 3.4.—Case 2 ASRG configuration with dimensions.

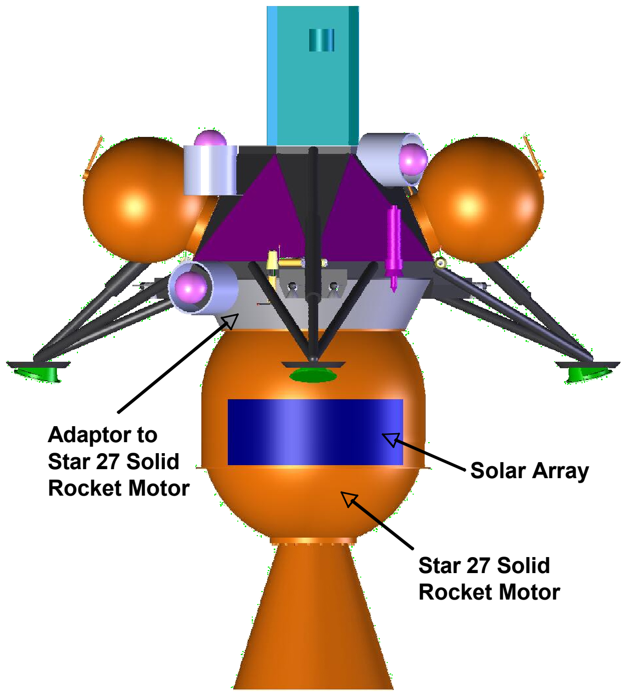


Figure 3.5.—Case 2 launch stack.

4.0 Challenges, Conclusion and Areas For Future Study

This section describes observations during the design of the system.

4.1 Challenges

The primary challenge for the ILN was the ability to gather all of the desired science data (and therefore science instruments) during the lunar night (14 days) and still launch the S/C on a small launcher (Minotaur V). This challenge can be successfully met by

1. Utilizing a radioisotope power source for constant power
2. Minimizing Science to only a seismometer

4.2 Conclusions and Future Work

- Non-RPS (solar array-battery) design requires almost twice the launch mass of the best RPS design
- 4 day transit and ~30 min landing power will require solar arrays and batteries regardless of concept
    - Transit and landing power levels much larger than on-Moon science phase
- All designs can fit on a Taurus II (Delta II class) with good margin
- Only a science floor, seismometer only, with a half ASRG can fit on a Minotaur V
    - While 30 percent growth and 10 percent launch margin assumed, design is limited to these growth allowances
- Dual launches on Atlas V (Falcon 9) a possibility

- Half ASRG provides a stand-alone power source for surface operations at >60 W with good margin at this point in the design
  - ASRG vibrations impact on science still needs addressed
- One-eighth RTG will need solar arrays and substantial battery power unless night-time power requirements reduced to below 15 W
  - One option may be to shut-down main S/C to keep-alive and have a single instrument store its own data for 14 day eclipse
- Costs show that a two S/C cost will be over $400M (using two Taurus 2's) unless each could fit on a Minotaur V
  - Updates on Taurus II and Minotaur V costs are needed

5.0 Subsystem Breakdown

The major subsystems of the ILN design are described in this section. All details are specific to the baseline, Case 2, used in this study.

5.1 Communications

This section describes the telecommunications subsystem of the ILN lander, dealing specifically with communications equipment on board the craft. Major telecommunications subsystem components have been chosen for the ILN lander in response to the science mission requirements and design considerations such as anticipated maximum distances, desired data rates, onboard power and mass limitations.

5.1.1 Communications Requirements

The general requirements on the telecommunications subsystem are to provide the best signal possible in terms of available onboard electrical power, accuracy, reliability, and quality assurance, with constraints on mass, size and costs. Specific requirements are provided in Table 5.1.

TABLE 5.1.—COMMUNICATIONS REQUIREMENTS

Requirements	Description
Data rates	Downlink: 10.6 kbps at 2275 to 2280 MHz Uplink: 1 kbps at 2119 MHz
Daily data volume	Average science/mission data rate: 100 Mb/day
Data storage	Estimated storage capacity: 100 Mb/day or 1.4 Gb of data to be stored up to 14 days.
Frequency	Shall support ILN at S-band and use 2119 MHz for uplink and 2275 to 2280 MHz for downlink
Housekeeping and overhead	Housekeeping and any overhead shall include in stated data rates in this table
Available onboard power	Total estimated onboard power: 5 W DC average during sunlight periods (1 W RF) for S-band receiver and transmitter
Effective isotropic radiated power (EIRP)	EIRPs shall be as required in order to achieve 10 bit error rate (BER) for given data rates

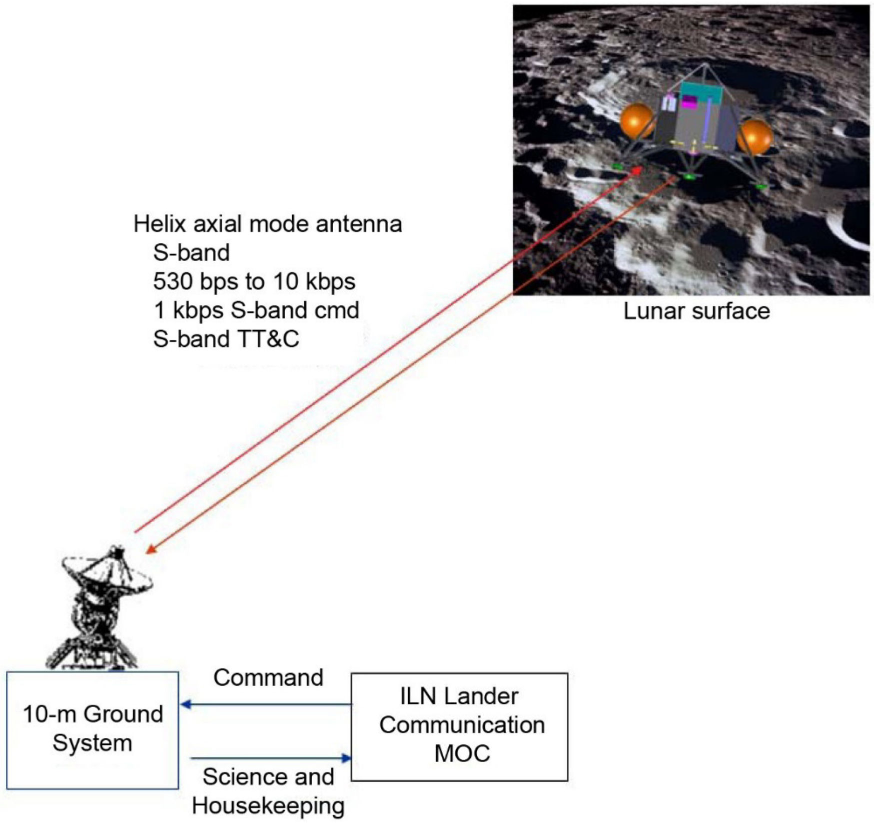


Figure 5.1.—Communications path assumptions.

5.1.2 Communications Assumptions

This study makes no assumptions with regard to data management and ground operation or distribution of scientific data after they are collected and transmitted to Earth. However, a general functional configuration is shown in Figure 5.1 to illustrate the communication path between the ILN craft on the Moon surface and Earth ground network. The lander is assumed to land within a 15° azimuth band as mentioned in Section 2.10.5. Thus it is assumed that the antenna will be pre-pointed relative to the S/C based on landing site latitude and longitude. This will eliminate the need for a pointing mechanism on the antenna but will necessitate accepting off-pointing of the antenna of up to a half-beamwidth.

5.1.3 Communications Design and MEL

Telecom Subsystem

- RF telecommunication subsystem consists of one axial mode helix antenna, one transmitter, one receiver and two diplexers
- S-band Telemetry, Tracking and Command (TT&C)

Estimated Data Storage and Transmit to Earth

- Science data per day: 5.3 Mb/day
  - Passive science experiment (Seismometer): 2.5 Mb/day
  - Magnetometer experiment (ROMAP): 0.2 Mb/day
  - Heat flow experiment (PLUTO Mole): 2.6 Mb/day (or 30 bit/sec)
- Assume 14 days Lunar surface operation, a total of 74.2 Mb of science data to be collected

- At downlink data rate of 1 kbps, 7.2 Mb can be transmitted to Earth in 2 hr
- At data rate of 10 kb, 72.0 Mb can be transmitted to Earth in 2 hr

(Note that the MEL for the telecom subsystem is included in the Section 5.2 Avionics in Table 5.2.)

5.1.4 Communications Trades

Antenna Trade

- Axial mode helix antenna
- Known parameters (ALSEP Report (Ref. 9))
  - Antenna length, L = 58 cm
  - Diameter, D = 3.8 cm
  - Pitch angle, α = 15°
  - Mass, M = 580 g
  - Downlink at 2275 and 2280 MHz
  - Uplink at 2119 MHz
  - Half-power beamwidth, is 26.8°
  - Downlink at 2280 MHz
  - λ = 13 cm
  - Ten-turn axial mode
- Calculated parameters
  - Circumference of helix

$$C = 3.1416 * D = 3.1416 * 3.8\text{ cm} = 11.94\text{ cm}$$

  - Space between turns

$$S = C\tan(\alpha) = 11.94\text{ cm}\tan(15°) = 3.2\text{ cm}$$

  - Peak antenna gain

$$\text{Gain}_{\text{peak}} = 10.3 = 10\log\left(\frac{C * 2 * L}{\lambda * 3}\right) = 16.1\text{ dBi}$$

  - Antenna gain G = 11.1 dBi, which gives us a link margin of 5.0 dB (See Figure 5.2).

Without antenna aiming mechanism, attitude control must be within ±15° in all three directions.

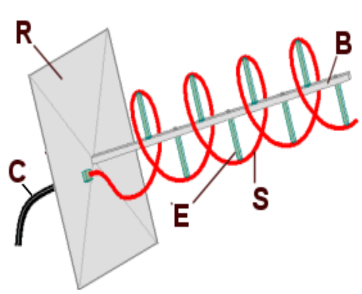


Figure 5.2.—Axial mode helix antenna.

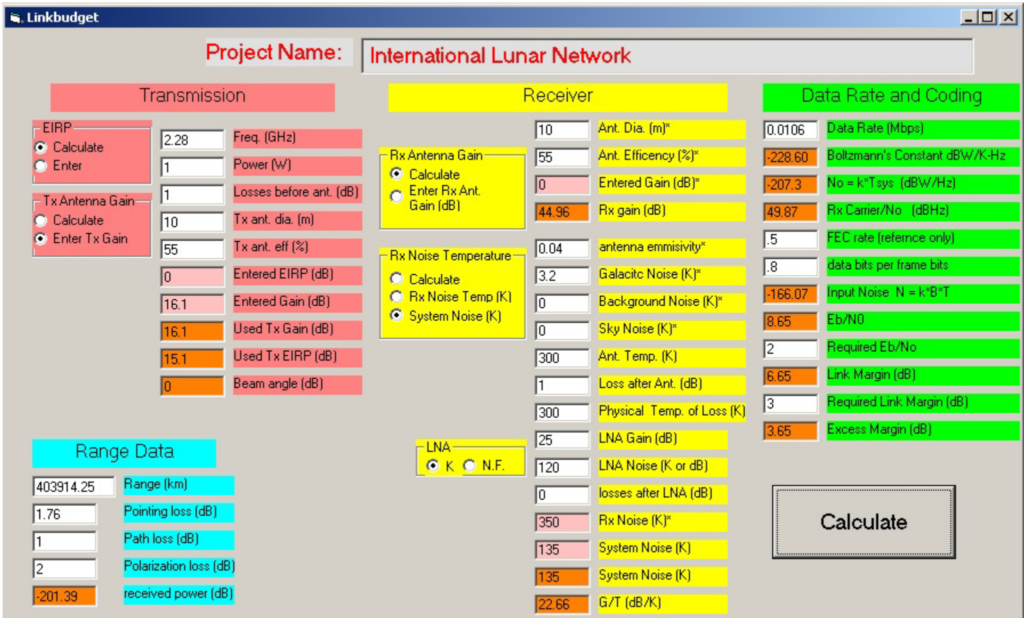


Figure 5.3.—Link Budgets for ILN Lander.

5.1.5 Communications Analytical Methods

Figure 5.3 provides the link budgets for the ILN lander.

5.1.6 Communications Risk Inputs

None

5.1.7 Communications Recommendation

- The link budgets analysis is based on a 10-m ground antenna, for better reception a 30-m or larger antenna should be used.
- An antenna aiming mechanism on the ILN lander is recommended, should there be a light-weight antenna steering system is available.

5.2 Avionics

The C&DH subsystem generally provides all telemetry acquisition and processing, and forwards telemetry to the TT&C subsystem for transmission to Earth. Experimental data will be stored on the onboard data recorder up to 14 days.

5.2.1 Avionics Requirements

- Science data per day: 100 Mb/day
- Assume 14 days Lunar surface operation, a total of 1.4 Gb of science data to be collected
    - 16 Gb storage drive
- Assume downlink at 10 kbps, 100 Mb can be transmitted to Earth in about 3 hr
- View-ability window shows 9 to 14 hr ground site access per day.
- Storage/downlink approach:
    - Store 14 days of science
    - Downlink each sunlit day science and one stored eclipsed day science in 6 hr

5.2.2 Avionics Assumptions

Avionics, C&DH Design Approach

- Meet the ILN Lander's science and mission requirements
- Provide support for communications, C&DH, GN&C, propulsion, power and thermal management
- Include software size and throughput estimates for all onboard applications and operating system

5.2.3 Avionics Design and MEL

Design Description

- Avionics subsystem consists of a single board computer, time generation unit, a data recorder, uplink and downlink electronics cards, and science payload interfaces (see the Table 5.2)
- Single board computer is Maxwell SCS750D, capable of supporting up to 1800 Dhrystone MIPS at 800 MHz with 512 KB L2 Cache, 256 MB SDRAM, and software selectable power consumption
- Data recorder unit is a 16 GB Intel Z-P140 solid-state drive
- Connectivity and communication interfaces: 1553, PCI, FireWire, and 32-bit programmable I/O
- SCS750D Specifications listed in Table 5.3

TABLE 5.2.—AVIONICS SUBSYSTEM MEL (CASE 2)

WBS Number	Description International Lunar Network (may 2008)	QTY	Unit Mass (kg)	CBE Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	**Robotic Lunar Lander - Landing ILN**			481.7	6%	30.2	511.9
06.1	ELV Adaptor			25.0	18%	4.5	29.5
06.2	Lunar Descent Stage			301.0	0%	1.5	302.5
06.3	Lunar Lander			140.2	14%	19.6	159.7
06.3.1	**C&DH, Communications GN&C**			22	21%	4.7	26.7
06.3.1.a	**Command & Data Handling**			3.7	28%	1.0	4.7
06.3.1.a.a	General Avionics Processor	1	1.5	1.5	30%	0.5	2.0
06.3.1.a.b	Time Generation Unit	1	0.3	0.3	20%	0.1	0.4
06.3.1.a.c	Command and Control Harness (data)	1	1.2	1.2	25%	0.3	1.5
06.3.1.a.d	Instrumentation & Wiring	1	0.7	0.7	30%	0.2	0.9
06.3.1.a.e	Data Recorder	1	0.001	0.0	50%	0.001	0.002
06.3.1.a.f	Misc 2	0	0.0	0.0	0%	0.0	0.0
06.3.1.b	**Communication**			5.2	20%	1.0	6.2
06.3.1.b.a	**S-Band Comm System**			4.2	16%	0.7	4.8
06.3.1.b.a.a	Receiver	1	1.2	1.2	15%	0.2	1.4
06.3.1.b.a.b	Transmitter	1	1.2	1.2	15%	0.2	1.4
06.3.1.b.a.c	Processing Module	0	0.0	0.0	0%	0.0	0.0
06.3.1.b.a.d	Antenna	1	0.6	0.6	20%	0.1	0.7
06.3.1.b.a.e	Coaxial Cable	0	0.0	0.0	0%	0.0	0.0
06.3.1.b.a.f	Diplexer	2	0.6	1.2	15%	0.2	1.4
06.3.1.b.b	**Communications Instrumentation**			1.0	36%	0.4	1.4
06.3.1.b.b.a	Coaxial Cable	1	0.3	0.3	50%	0.2	0.5
06.3.1.b.b.b	Installation - Mounting & Circuitry	1	0.7	0.7	30%	0.2	0.9
06.3.1.c	**Guidance, Navigation, & Control**			13.2	20%	2.6	15.8
06.3.1.c.a	Inertial Measurement Units	1	4.7	4.7	20%	0.9	5.6
06.3.1.c.b	misc	0	0.0	0.0	0%	0.0	0.0
06.3.1.c.c	Sun Sensors	8	0.01	0.1	20%	0.0	0.1
06.3.1.c.d	Star Tracker	1	3.4	3.4	20%	0.7	4.1
06.3.1.c.e	Attitude control system	0	0.0	0.0	30%	0.0	0.0
06.3.1.c.f	Flight Control Electronics	0	0.0	0.0	0%	0.0	0.0
06.3.1.c.g	Radar System	1	5.0	5.0	20%	1.0	6.0
06.3.1.c.h	Visualization System	0	0.0	0.0	0%	0.0	0.0
06.4	ILN Science Payload			15.5	30%	4.7	20.2

TABLE 5.3.—SCS750D SPECIFICATIONS

Technical Specifications	
Radiator Tolerance	**Mechanical**
* One Board Upset Every 3000 years in LEO or GEO Orbit * TID: >100 krad (Si) - Orbit dependent * SEL (th): >80 MeV-cm^2/mg - All parts except SDRAM ≈ 50 MeV-cm^2/mg - SDRAM	* 6u x 160 mm * 1.5 kg (3.3 lb) Max
Processors	**Models**
(3) **FULLY TMR PROTECTED PROCESSORS** * PowerPC 750FX™ on Silicon on Insulator (SOI), 0.13 µm * 2.32 Dhrystone MIPS/MHz * > 1800 Dhrystone MIPS at 800 MHz * 400 to 800 MHz - Software Selectable Core Clock Rate	**SCS750F - FLIGHT CONFIGURATION** * Rad-Tolerant, Class "S" or Equivalent Components * Conduction Cooled * 2 ACTEL RTAX FPGAs
L1 CACHE * 32 KB Instruction with Parity * 32 KB Data with Parity **L2 CACHE** *512 KB on-chip with ECC at CPU Core Clock Rate	**SCS750E - ENGINEERING CONFIGURATION** * Parts Identical to Flight (Not screened to flight level) * Conduction Cooled * 2 ACTEL RTAX FPGAs
Memory **VOLATILE** * 256 MB SDRAM - Reed-Solomon Protected - Double Device Correction	**SCS750d - ENGINEERING DEVELOPMENT CONFIGURATION** * Commercial Components, ACTEL FPGAs * Full Hardware & Software Compatibility w/ E & F Models * Convection Cooled
NON-VOLATILE * 8 MB EEPROM - ECC Protected - 7.0 MB EEPROM available to user - 0.5 MB Primary SuROM - 0.5 MB Secondary SuROM (Autoswap on Primary Failure)	**SCS750P - PROTOTYPE CONFIGURATION** * Commercial Components, Xilinx FPGAs * Similar Functionality to D, E & F Models * Convection Cooled
Interfaces	**All Models are available with an optional 1553 interface**
cPCI BUS * 6u * 3.3v * 32 bit, 33 MHz * Master/Target & Syscon/Peripheral	
1553 * BC/RT/MT * SEU Immune	**Board Support Package**
SERIAL * UART (Asynchronous), LVDS * (2) USRTs (Synchronous), LVDS **PROGRAMMABLE I/O** * 32 Programmable General Purpose I/O (GPIO) **Power** * 7 - 25 W (typical) dependent on clock rate/MIPS requirements * 5 V for 1553 interface, 3.3 V for rest of board **Operating System** * VxWorks, Tornado **Temperature** * -30 °C to +65 °C (Acceptable Levels) * -40 °C to +70 °C (Qualification Levels)	* Detailed Specification * User Manual - Interface Control Documents - Software User's Manual (SUM) * VxWorks® Runtime License * Certificate of Conformance * Startup ROM Source Code * Functional Test Procedure - Test Plan - Test Log * Functional Test Report * Environmental Test Procedure (Flight Only) - Test Plan - Test Log * Environmental Test Report (Flight Only)

5.2.4 Avionics Trades

None.

5.2.5 Avionics Risk Inputs

None.

5.2.6 Avionics Recommendation

None.

5.3 Guidance, Navigation and Control (GN&C)

5.3.1 GN&C Requirements

The GN&C subsystem shall provide full 6-degrees of freedom (DOF) control of the vehicle from launch through end of mission. This includes stabilization of the vehicle after launch vehicle separation, attitude control throughout the cruise, commanding and controlling all slews, and performing the automated descent and landing.

The GN&C system shall land the vehicle within ± 15° of the desired azimuth. This allows for the prepointing of the radiators, solar arrays, communications antennae, and science instruments before launch dependent on the landing site.

5.3.2 GN&C Assumptions

The GN&C system design relies heavily on the designs of other missions that have flown. First, the mission design was based on the Surveyor missions to the Moon, using a solid rocket for the majority of the descent burn, and onboard propulsion for the remainder. Second, the Mars Phoenix instruments and pulsed propulsion system were used as design references for this lander design.

5.3.3 GN&C Design and MEL

The GN&C system hardware is made up of:

- 1 Star Tracker (Goodrich HD-1003)
- 1 IMU (Honeywell MIMU)
- Radar altimeter for landing (Honeywell HG9550, Phoenix lander, C-17J heritage)
- Sun sensors to aide in Earth acquisition (Barnes, 2.0° one axis)
- GN&C Software run on main C&DH computers

The detailed mass accounting for the GN&C system can be seen in Table 5.4. The block diagram can be seen in Figure 5.4.

TABLE 5.4.—GN&C MEL (CASE 2)

Description	QTY	Unit Mass	CBE Mass	Growth	Growth	Total Mass
International Lunar Network (may 2008)		(kg)	(kg)	(%)	(kg)	(kg)
Guidance, Navigation, & Control			13.2	17%	2.6	15.8
Inertial Measurement Units	1	4.7	4.7	20%	0.9	5.6
Sun Sensors	8	0.01	0.1	20%	0.0	0.1
Star Tracker	1	3.4	3.4	20%	0.7	4.1
Radar System	1	5.0	5.0	20%	1.0	6.0

5.3.4 GN&C Trades

No specific trades were completed. A feasible design was constructed and minor changes were made to help close the mission.

5.3.5 GN&C Analytical Methods

The GN&C subsystem mainly utilized the Rocket Equation to calculate propellant masses for the mapping, landing, and ascent burns. Again, because of the short duration of the study, no in-depth dynamic analyses were completed.

5.3.6 GN&C Risk Inputs

Figure 5.4 illustrates the layout of the GN&C subsystem. Table 5.5 outlines the GN&C risk input.

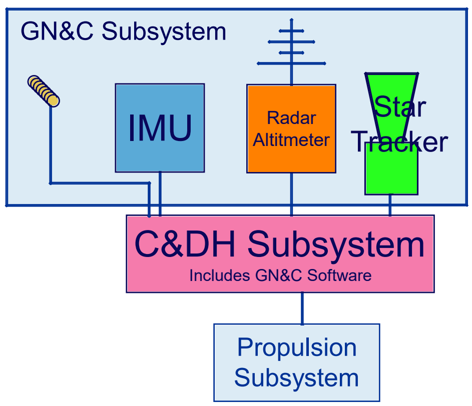


Figure 5.4.—GN&C system block diagram.

TABLE 5.5.—GN&C RISK INPUT

ID	Risk information sheet		Date identified: April 9, 2008
Likelihood: 3 Consequences: Cost: 4 Schedule: 4 Performance: 4 Safety: 1	Risk title: Launch Vehicle Performance and Cost Risk Risk statement: Given the cost limits imposed on this mission; the possibility exists that an unproven launch vehicle with unproven performance and costs will be chosen that will either not meet performance or require substantially higher costs for mission success.		
Timeframe:	Initiation: Doug Fiehler	Classification: Launch Vehicle	Assigned to:
Context: Because of the cost limits imposed on this mission, unflown/unproven launch vehicles become attractive options to stay within the cost limits. These launch vehicles quote low costs and offer the performance required, but have not flown. If these launch vehicles do not meet the performance goals, either the cost will increase for that particular launch vehicle, or a different, more expensive launch vehicle will have to be chosen for this mission.			
Approach: Research/Accept/Watch/Mitigate Watch: Watch the different low-cost launch vehicle development programs to assure that they meet performance and cost requirements for this mission.			

5.4 Electrical Power System

The power system shall provide sufficient power for the S/C and Lander system throughout the mission including launch from Earth, transfer to the Moon, and the surface of the Moon. In addition, the power system must be sized to supply sufficient power to the other subsystems throughout the mission.

5.4.1 Power Requirements

Three power systems were considered for the ILN. Analysis from the COMPASS team when analyzing the ILN shows power requirements of 26 W during the day and 23 W during the night. For this design we assumed an unregulated bus operating nominally at 28 V with allowable voltages ranging from 26 to 36 V.

- Power Requirements:
  - 26 W day
  - 23 W night
  - Peak loads during transit
  - Batteries capable of providing power during landing

5.4.2 Power Assumptions

See Figure 5.5 for solar power as a function of time of day on the Moon.

5.4.3 Power Design and MEL

Single general purpose heat source (GPHS) half ASRG providing

- 63 W day (excess of 37 W)
- 70 W night (excess of 40 W)

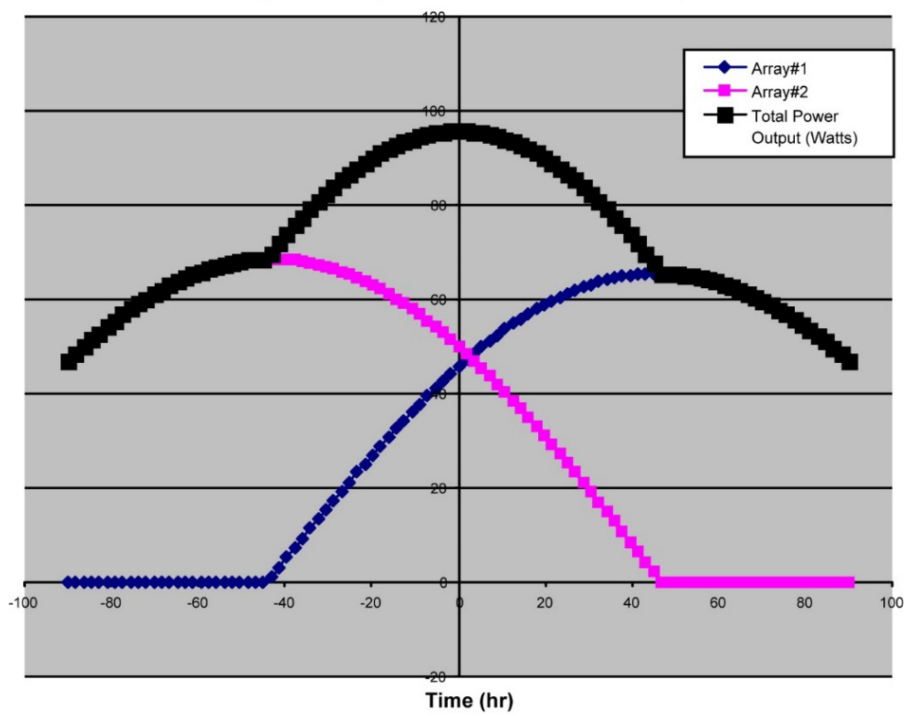
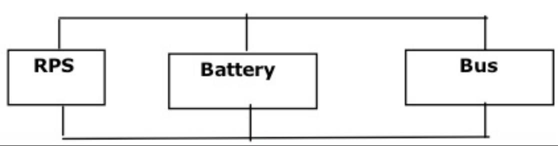


Figure 5.5.—Solar power as a function of time of day.

One half an ASRG (1 GPHS) is designed to provide 70 W to power the ILN lander at beginning of life (BOL). The system is designed to provide 59 W or better (daytime) to the ILN lander at end of life (EOL)/6 years. There are negligible thermal interactions between the ASRGs. Figure 5.6 shows a typical ASRG with the main components called out in the graphic. The eight are connected together via a shunt regulator/bus protection (RBI) assembly. This RBI isolates the ASRGs from S/C bus and each other and follows load demands from S/C bus. There is an approximately 6 percent loss through the RBI and monitoring circuitry (94 percent of power flows through to loads) with 53 W used for fault detection/monitoring. Included in this system is a bus Capacitance of 3000 µf which provides some bus rigidity. Power cabling and harness systems design assumes a 1 percent line loss. Figure 5.6 and Figure 5.7 illustrate the ASRG power system design and components.

- Two fixed panels (east/west) solar arrays
    - Triple junction solar cells
    - 30 percent conversion efficiency
    - Power cabling, wiring harness, power storage during coast operations assumptions, etc.
    - Li-ion batteries for night operations
    - RTG reduces battery energy by 50 percent
    - Battery dimensions: 14 by 14 by 14 cm
    - Half ASRG dimensions: 30 cm H by 46 cm W by 46 L
    - Solar arrays: 2X (9 cm by 9 cm)
- Technology status
    - Solar array, TRL-9
    - GPHS RTG TRL-6
    - Batteries TRL-6
        - Only needed for landing phase
        - Assumed ABSL 18650 hard carbon (HC) based battery
    - Solar arrays used during flight from Earth to Moon



Description	Quantity	Unit mass (kg)	Growth (%)	Total
½ ASRG	1	14	18	16.5
Battery	1	1.5	20	1.8
Solar arrays	1	1.6	18	1.9
Power Management and Distribution (PMAD)	1	8	26	10.8
			Totals	**31.0**

Figure 5.6.—Power system design schematic.

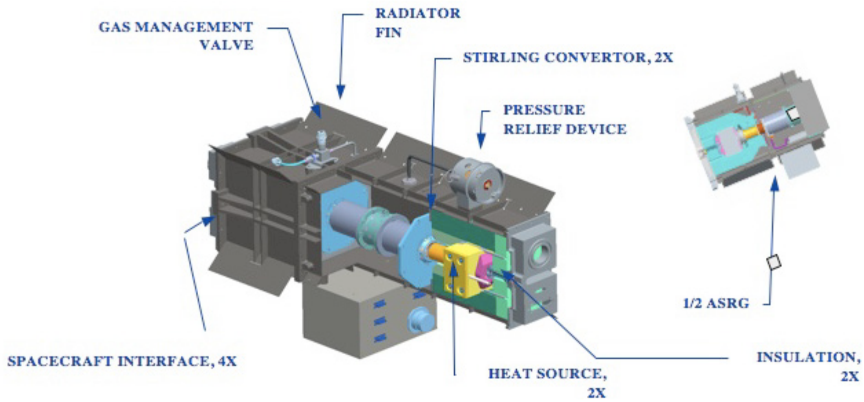


Figure 5.7.—ASRG with components labeled.

5.4.4 Power Trades

The first power system option (Case 1) considered used a traditional solar arrays/battery system. The solar arrays consisted of triple junction solar cells with a baseline efficiency of 28 percent and a battery based upon state-of-the-art Li-ion ABSL/Sony HC 18650 cells, which have flown in numerous S/C.

The second option (Case 3) considered was a single GPHS Multi-Mission Radioisotope Thermoelectric Generator (MMRTG) or RTG system producing 15 W coupled to a solar arrays and battery system to make up the additional power required by the ILN. This system would require the development of a new structure/housing for either the MMRTG or RTG to allow the use of a single GPHS. A voltage boost is required (DC/DC convertor) as the output from the RTG is about 3 V and the power system bus operates at 28 V.

The final candidate (Case 2—Baseline) was half of an ASRG. NASA and the Department of Energy are developing the ASRG as a high efficiency candidate for potential future space missions. The ASRG higher efficiency allows a fourfold reduction in PuO_2 fuel when compared with the RTG used in past NASA missions while producing 140 W of electrical power. It is proposed that half an ASRG be used to provide 70 W to the ILN.

The current configuration of the ASRG has two Stirling convertors operating in a dual opposed fashion to cancel out the vibrations generated from the moving piston and displacer. Because we are now using only a single convertor the vibration cancellation will be handled by a passive balancer which effectively replaces the moving parts of the out of phase second Stirling convertor. The Stirling convertor puts out single phase DC at 28 V nominally but can operate from 24 to 36 V.

At the close of the study, a science floor configuration was modeled using the same power system as modeled in Case 2 (1/2 ASRG) , (Case 4), with a minimal science package in order to reduce the total mass of the S/C enough to fit within the performance of a Minotaur V launch vehicle.

5.4.5 Power Analytical Methods

For the solar array battery system the design assumed two fixed arrays, tilted at 45° one facing east and the other west. Using the lunar declination and S/C orientation offsets (1.5° and 15°, respectively) estimates were made of the total energy output of the arrays during a lunar day and then matched to that required by the batteries for the lunar night power requirements. Battery charge/discharge efficiency, line losses and power electronics were included.

For the RTG/solar array/battery system the identical process described above was used with the exception of assuming a constant electrical power input of 13 W into the bus from the RTG/MMRTG.

Single GPHS/RTG mass was found by estimating the endcap housings from the RTG program and then adding to it the mass from the GPHS and surrounding thermocouple and structure mass.

For the half ASRG, because of the lower operating temperature, the system is more sensitive to temperature variations of the environment. Using previous studies it was found that the sink temperature variations on the lunar surface change the power output of the system by about 10 percent. Estimates for the half ASRG power output were 63 W day/70 W night. Estimates for the mass were made using the existing ASRG, removing the one convertor and then adding back the passive balancer mass.

5.4.6 Power Risk Inputs

- The ASRG technology could achieve TRL-9 by 2012 to 2013.
- Vibration levels may be acceptable for science instruments, but more data is needed.
- If allowable, a single GPHS module may be sufficient for the mission.

5.4.7 Power Recommendation

Options to reduce mass and recommendations: limit night operations and power as the majority of mass comes from energy storage subsystem.

5.5 Structures and Mechanisms

The main backbone of the S/C is the primary structure. The structure bears the main loads imparted on the vehicle. Also, it contains all the components required to operate the vehicle and execute the intended mission. The following sections describe the requirements and the overall construction of the structure for the cases studied.

5.5.1 Structures and Mechanisms Requirements

The structure of the ILN Lander must contain necessary hardware for research instrumentation, avionics, communications, propulsion and power. The structure must be able to withstand applied loads from launch vehicle and provide minimum deflections, sufficient stiffness, and vibration damping. In addition, there is a need to accommodate loads from landing on the lunar surface. Weight is to be kept to a minimum while still meeting the previously mentioned requirements. The structure must fit within confines of launch vehicle.

5.5.2 Structures and Mechanisms Assumptions

Initial assumptions affecting the design of the structure and mechanisms include the following

- Launch vehicle max longitudinal load: 11 g
- Launch vehicle max lateral load: 4 g
- Landing conditions:
  - Approach velocity 3 m/s
  - Max pad to surface pressure of 69 kPa
  - Using Surveyor I data from report NASA SP–126 (Ref. 10)
- Structural Design Assumptions
  - Material: Al 2090-T3
  - Space frame with square and round tubular members
  - Composite sandwich structure deck
  - Welded and threaded fastener assembly
  - Mounting hardware mass ~4 percent of hardware mass

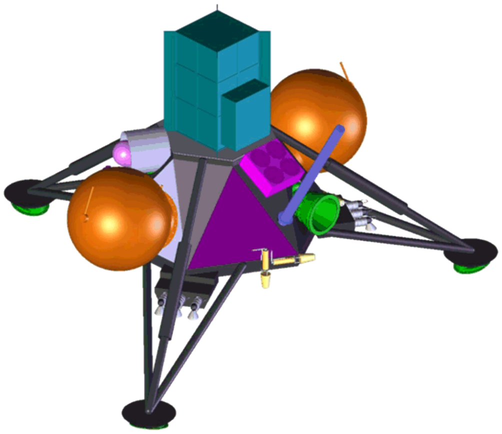


Figure 5.8.—Case 2 lander design.

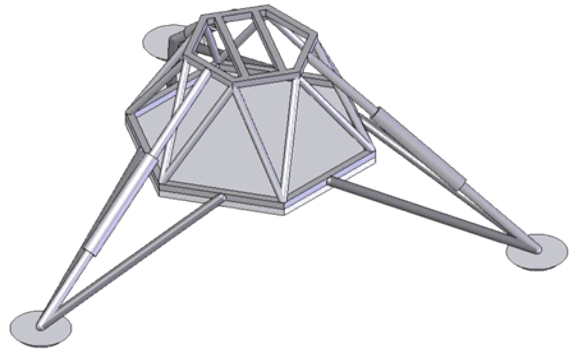


Figure 5.9.—Structural design landing legs and space frame.

5.5.3 Structures and Mechanisms Design and MEL

Design Description

Figure 5.8 and Figure 5.9 illustrate the Case 2 lander design.

- Tubular space frame in hexagonal configuration
  - Plate mounted, between vertical struts, external propellant tanks
- Deck of composite sandwich architecture to mount hardware
  - Al 2090 face sheets
  - Al honeycomb core
- Spring struts to support landing hardware
  - 218 kg vehicle landing mass, 3 m/s approach velocity
  - 30.5 cm D landing pads (8 in.)
  - 13 cm vertical displacement
  - Damping: friction or ratcheting mech.

5.5.4 Structures and Mechanisms Trades

Four different cases were run in this design session, and the structure was designed in each case to fit the vehicle designed and accommodate the different power system. Cases 1 to 3 landed roughly the same mass, so the tubular space frame and landing legs were built to handle the landing loads of the same mass (roughly 200 kg). Case 4, however, was a lighter mass lander vehicle, so the structures mass was also reduced because it only had to accommodate landing roughly a 100 kg mass.

5.5.5 Structures and Mechanisms Analytical Methods

Preliminary calculations were performed to check stresses for members in the main structure under maximum longitudinal launch loads. Landing hardware was sized according to the approach velocity and the maximum pressure applied to the lunar surface.

5.5.6 Structures and Mechanisms Risk Inputs

Excessive g loads, impact from a foreign object, or excessive landing velocity may cause too much deformation, vibrations, or fracture of sections of the support structure. Consequences include lower performance from mounted hardware to loss of mission.

An excessive deformation of the structure can misalign components dependent on precise positioning, therefore, diminishing their performance. Internal components may be damaged or severed from the rest of the system resulting in diminished performance or incapacitation of the system. Excessive vibrations may reduce instrumentation performance and/or potentially lead to structural failure due to fatigue. Overall, the mission may not be completed in an optimum manner or it can be terminated in the worst case.

As part of the risk mitigation process the structure is to be designed to NASA standards to withstand expected g loads, a given impact, and to have sufficient stiffness and damping to minimize issues with vibrations. Trajectories are to be planned to minimize the probability of impact with foreign objects and prevent any damage upon landing.

5.5.7 Structures and Mechanisms Recommendation

A thorough structural finite element analysis should be conducted to optimize strut sizes to minimize weight while accommodating anticipated loads. A modal analysis should be conducted to avoid natural frequencies near any excitation frequencies that may exist.

5.6 Propulsion and Propellant Management

5.6.1 Propulsion and Propellant Management Requirements

The S/C propulsion subsystem was required for three propulsive operations:

1. Orbit transfer and insertion
2. Reaction/attitude control
3. Surface landing

5.6.2 Propulsion and Propellant Management Analytical Methods

Because the propulsion subsystems were assembled from existing components, the analysis required consisted primarily of maintaining a mass roll up for the various subassemblies. The first two propulsion operations were performed with solid rocket motors that were selected from the Star motor catalog. The selection was based primarily on total S/C mass and the propellant load fraction of the motor.

The primary analysis that was actively performed was to determine the propellant tanks sizes based on propellant conditions over the mission duration. The tank requirements were determined using propellant density and storage pressure through Hoop Stress Analysis. These requirements were then used to select the best match from the PSI and Arde, Inc storage tank catalogs.

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

5.6.3 Propulsion and Propellant Management Assumptions

The propulsion subsystems designed for these studies used only commercially available (OTS) components to mitigate development costs and time requirements. All solid rocket motors and chemical thrusters used in the design as well as the propellant management components and propellant tanks were from operating manufactures.

5.6.4 Propulsion and Propellant Management Design and MEL

5.6.4.1 LOI and Descent Propulsion System Modeling

For this mission, orbit insertion and descent burns were performed with a solid rocket motor:

- Solid rocket motors were selected from existing catalog of flight-proven options to minimize development requirements. The motors were selected based on:
  - Thrust level and required propellant load provided by Mission Analyses

    - Best match for solid motor performance was selected and motor weight was adjusted to account for off-loaded propellant (as required)
- Use adjusted propellant mass fraction to estimate propulsion subsystem final mass for S/C mass determination in order to perform mission analyses
- Known configuration of existing motor used by Structures and CAD teams to establish S/C configuration

The Star 27 motor was chosen for the Case 2 baseline design.

5.6.4.2 Chemical Propulsion System Modeling for Landing and Reaction Control

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

For the *landing propulsion system*, a system similar to the landing system used by the Mars Phoenix lander was developed to build off that controlled landing experience. For the landing system model, a series of liquid monopropellant engines were selected to meet the performance requirements determined by trajectory analysis. The engines were clustered and operated in pulsed mode to provide an aggregated throttleable thrust for a soft landing.

- Nine Aerojet MR-120 hydrazine rockets were baselined for landing
    - Operating at 90 N (20 lbf) thrust; 229 sec specific impulse
    - Rockets arranged in three clusters (pods) of three thrusters each

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

- Nine Aerojet MR-111E (Ref. 11) rockets were baselined
    - Operating at: 4.4 N (0.99 lbf) thrust, 229 sec specific impulse
    - Rockets arranged in three pods of three engines each
    - Used Hydrazine propellant for reduced propellant management system complexity; common propellant for landing and RCS

The study scope was limited to delta-V burn matching and did not require detailed analysis of reaction and attitude control performance. Hence, a simple mass model for the RCS was sufficient.

Propellant Management System

The Hydrazine propellant for RC propulsion was stored in two spherical carbon-overwrapped Al-Li pressure vessels. These tanks based on a COTS unit from ATK-PSC Inc., Model No. 80273-7. The baseline tank dimensions were:

- Size: 0.48 m diam. (19.1 in. diam.)
- Internal volume = 0.60 m^3 (20.8 ft^3)

Gaseous nitrogen pressurization via tank blowdown was used for this study to simplify the propulsion system and reduce costs. Diaphragm storage tanks were oversized to allow room for the He pressurant. The storage tank was pressurized with He to approximately 700 psi at launch to ensure sufficient propellant could be delivered by the time that the minimum tank pressure for proper thruster operation was reached at approximately 150 psi. Minor size changes to made to match propellant load determined by the Mission analysis tool. A propellant management device is included in tank mass roll-up.

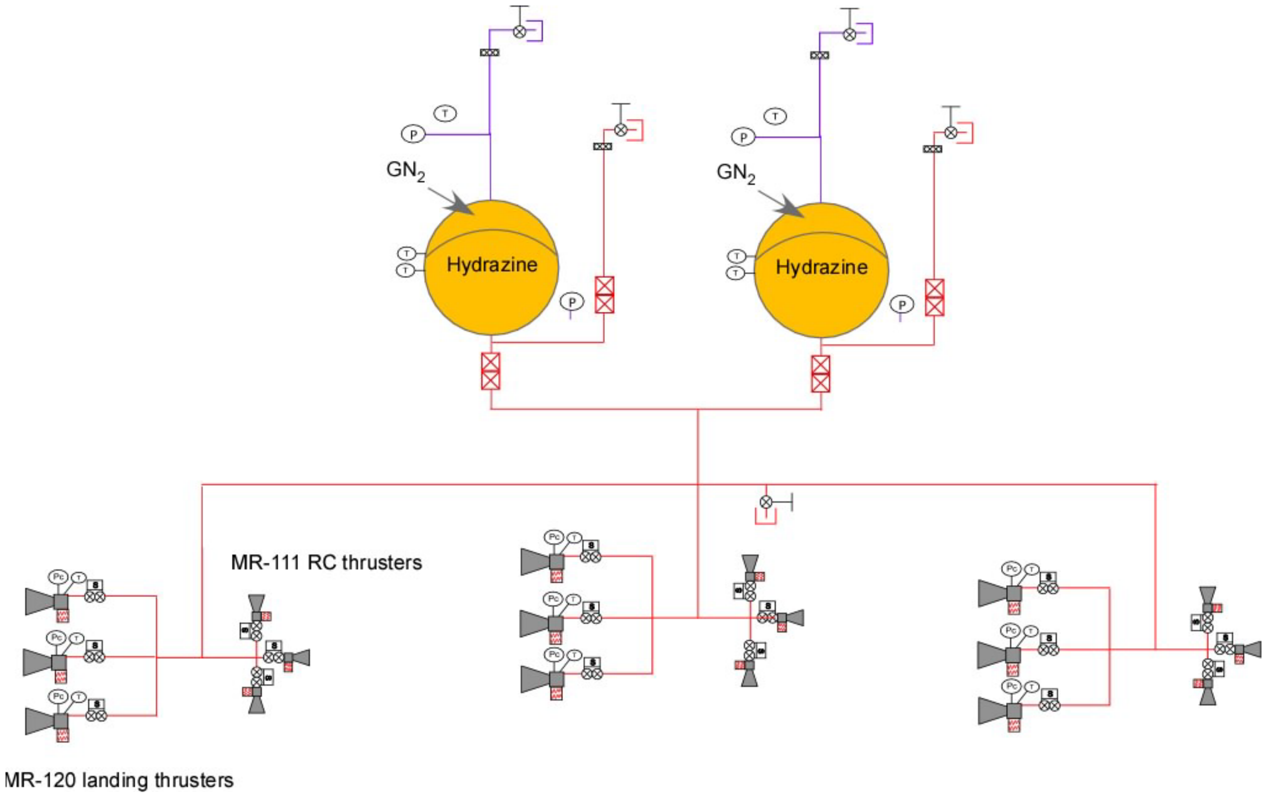


Figure 5.10.—Chemical propulsion systems combined schematic.

The propellant distribution system is comprised of gas/fluid delivery lines, the control components (isolation/latching valves, control valves, filters, and service valves). All components are assumed to be mature commercial off the shelf (COTS) technologies, so little or no development costs required. The propellant distribution is illustrated in Figure 5.10.

Weight estimates for propellant storage tank and lines heaters, along with required power estimates, are incorporated into the propulsion system mass roll-up. Additionally, MLI for tanks are lines are incorporated into the system mass. The heater mass and power estimates and MLI mass estimates are obtained from the Thermal system model based on tank and line size data that is provided by the propulsion subsystem model.

Table 5.6 lists the line items in the Propulsion and Propellant Management MEL including the propellant. Typically, growth percentage is calculated growth divided by total CBE mass.

5.6.5 Propulsion and Propellant Management Trades

Solid Rocket Motors

- Motor choices limited to selecting an OTS motor
- Because technology readiness and demonstrated heritage of these systems ensure relatively low cost as well as readily used in vehicle systems analysis

TABLE 5.6.—PROPULSION SUBSYSTEM AND PROPELLANT MEL (CASE 2)

Lunar Descent Stage			**299.3**	**0%**	**1.1**	**300.4**
C&DH, Communications GN&C						
Propulsion Hardware (Chemical)			**27**	**4%**	1.1	**28.6**
Main Engine			27.5	4%	1.1	28.6
Main Engine	1	27.5	27.5	4%	1.1	28.6
Main Engine Gimbal	0	0.0	0.0	0%	0.0	0.0
Reaction Control System			0.0	0%	0.0	0.0
RCS Engine	0	0.0	0.0	0%	0.0	0.0
Propellant Management (Chemical)			**0**	**0%**	0.0	**0.0**
OMS Propellant Management			**0**	**0%**	0.0	**0.0**
RCS Propellant Management			0.0	0%	0.0	0.0
Propellant (Chemical)			**272**	**0%**	0.0	**271.8**
Main Engine Propellant			**272**	**0%**	0.0	**271.8**
Fuel			271.8	0%	0.0	271.8
Oxidizer			0.0	0%	0.0	0.0
Pressurant	0	0.0	0.0	0%	0.0	0.0
RCS Propellant			0.0	0%	0.0	0.0
Propulsion Hardware (EP)			**0**	**0%**	0.0	**0.0**
Propellant (EP)			**0**	**0%**	0.0	**0.0**
Propellant Management (EP)			**0**	**0%**	0.0	**0.0**
Power Processing (EP)			**0**	**0%**	0.0	**0.0**
Electrical Power						
Thermal Control (Non-Propellant)						
Structures & Mechanical Systems						
Lunar Lander			**56.2**	**4%**	**2.6**	**58.8**
C&DH, Communications GN&C						
Propulsion Hardware (Chemical)			**7**	**7%**	0.6	**7.5**
Main Engine			3.3	7%	0.3	3.5
Main Engine	9	0.4	3.3	8%	0.3	3.5
Main Engine Gimbal	0	0.0	0.0	0%	0.0	0.0
Reaction Control System			3.7	7%	0.3	4.0
RCS Engine	2	1.8	3.7	8%	0.3	4.0
Propellant Management (Chemical)			**18**	**10%**	2.1	**19.7**
OMS Propellant Management			**0**	**0%**	0.0	**0.0**
RCS Propellant Management			17.7	10%	2.1	19.7
Fuel Tanks	2	3.1	6.3	0%	0.0	6.3
Fuel Lines	0	0.0	0.0	18%	0.0	0.0
Pressurization System - tanks, panels, lines	1	2.0	2.0	18%	0.4	2.4
Feed System - regulators, valves, etc	1	9.4	9.4	18%	1.7	11.1
Propellant (Chemical)			**32**	**0%**	0.0	**31.6**
Main Engine Propellant			**0**	**0%**	0.0	**0.0**
Fuel			0.0	0%	0.0	0.0
Oxidizer			0.0	0%	0.0	0.0
Pressurant	0	0.0	0.0	0%	0.0	0.0
RCS Propellant			31.6	0%	0.0	31.6
Fuel			31.2	0%	0.0	31.2
Oxidizer			0.0	0%	0.0	0.0
Pressurant	1	0.3	0.3	0%	0.0	0.3

Lander Rocket Engine

- Monopropellant Hydrazine-based engines selected because of technology maturation and demonstrated heritage
    - Engine class considered OTS
- Preliminary engine operating mode: Pulsed

- ○ Average performance and lifetime considerations need to be assessed for these engines to determine the operating parameters necessary for vehicle landing
    - – Pulse rate; effective thrust, specific impulse
- ○ Other chemical engines may be considered if necessary

5.6.6 Propulsion and Propellant Management Risk Inputs

There were three risk issues that were identified for detailed risk analysis. These issues are:

1. Impingement of RCS thruster plumes on sensitive S/C surfaces. The plumes may degrade the surfaces via material deposition or erosion which could jeopardize mission capability. Surfaces seeing the plumes need to be assessed.
2. Freezing of chemical propellant in lines or tanks. Without propellant reaching the thrusters, mission capability would be critically compromised to the point of complete failure.
3. Additionally, recovery procedures to return propellant to liquid state are problematic and suspect.

5.6.7 Propulsion and Propellant Management Recommendation

None applicable.

5.7 Thermal Control

The thermal modeling provides power and mass estimates for the various aspects of the vehicle thermal control system based on several inputs related to the vehicle geometry, flight environment and component size.

5.7.1 Thermal Requirements

Objective: To provide spreadsheet-based models capable of estimating the mass and power requirements of the various thermal systems.

The thermal requirements for the ILN were to provide a means of cooling and heating of the S/C equipment during operation on the lunar surface in order to remain within their maximum and minimum temperature requirements.

The maximum heat load to be rejected by the thermal system was 25 W, and the desired operating temperature for the electronics was 300 K and 250 K for S/C structure. The S/C was required to operate through the lunar nighttime.

5.7.2 Thermal Assumptions

The assumptions utilized in the analysis and sizing of the thermal system were based on the operational environment. It was assumed that operation would take place within the lunar surface environment. The following assumptions were utilized to size the thermal system.

- Lunar surface operation: Day and night
- Radiator designed to see deep space with minimal view factors to Earth, lunar surface and to the solar array. The radiator was positioned at a 30° angle to the horizontal based on minimizing the view factors and available vehicle location.
- The maximum angle of the radiator to the Sun was 15°.
- The radiator temperature was 320 K.
- A redundant radiator was used to account for vehicle orientation on the surface and to increase overall reliability.

5.7.3 Thermal Design and MEL

The thermal system is used to remove excess heat from the electronics and other components of the system as well as provide heating to thermally sensitive components throughout the lunar nighttime.

Excess heat is collected from a series of four Al cold plates located throughout the interior of the S/C. These cold plates have heat pipes integrated into them. The heat pipes transfer heat from the cold plates to the radiator, which radiates the excess heat to space. The portions of the heat pipes that extend from the communications box are integrated to the radiator and protected with a micrometeor shield. The system utilizes a louver system on the radiators to regulate the internal temperature and to insulate the radiators during the lunar nighttime.

Two radiators were used to provide redundancy and margin as well as account for the unknown landing orientation of the S/C. This added margin insures against unforeseen heat loads, degradation of the radiator due to lunar dust buildup and increased view factor toward any other thermally hot body not accounted for in the analysis.

To provide internal heating for the electronics an option to use a series of RHUs was considered. Each of these heaters would provide a continuous 1 W of thermal power to maintain sufficient internal temperature during the nighttime. These heaters may be needed for any of the external science packages. MLI is also utilized on the S/C exterior to regulate and maintain the desired internal temperatures.

5.7.4 Thermal Trades

A comparison was made between utilizing electric heaters and the RHUs. Due to the long nighttime period, the mass of the energy storage required to operate the electric heater system was prohibitive. Therefore, RHUs were chosen as the method for providing heat to the critical components during the nighttime.

5.7.5 Thermal Analytical Methods

The analysis performed to size the thermal system is based on first principle heat transfer from the S/C to the surroundings. This analysis takes into account the design and layout of the thermal system and the thermal environment to which heat is being rejected to or insulated from. For more detailed information on the thermal analysis a summary white paper titled “Preliminary Thermal System Sizing” was produced.

Environmental Models

Lunar surface environment represents worst case operating condition. All components were sized to operate within this environment.

Systems Modeled

- Micrometeor shielding on radiator
- Radiator panels (placement, sizing)
- Thermal control of propellant lines and tanks
- S/C insulation (layers of MLI, MLI placement)
- Avionics and PMAD cooling (number of cold plates, heat pipe length)
- Component heating (electric heaters and/or RHUs)

Radiator Sizing

The radiator panel area has been modeled along with an estimate of its mass. The model was based on first principles analysis of the area needed to reject the identified heat load to space. From the area, a series of scaling equations were used to determine the mass of the radiator within the lunar surface environment. Table 5.9 lists radiator sizing assumptions.

Thermal Analysis Propellant Lines and Tanks

Power requirements and mass have been modeled. This modeling included propellant tank MLI and heaters and propellant line insulation and heaters.

The model was based on a first principles analysis of the radiative heat transfer from the tanks and propellant lines through the S/C structure to space. The heat loss through the insulation set the power requirement for the tank and line heaters. The near lunar space thermal environment was used to calculate the heat loss. Assumptions used are listed in Table 5.10.

Thermal Analysis—S/C Insulation

The mass of the S/C MLI insulation was modeled to determine the mass of the insulation and heat loss. The model was based on a first principles analysis of the heat transfer from the S/C through the insulation to space. Nighttime lunar surface thermal environment was used to size the insulation. Two types of heaters were considered, RHU, and electrical heaters. Table 5.7 shows the thermal system inputs and outputs. Table 5.8 shows the thermal system radiator sizing assumptions, Table 5.9 shows the thermal system tank insulation sizing assumptions, and Table 5.10 shows the thermal system tank insulation sizing assumptions.

TABLE 5.7.—THERMAL SYSTEM INPUTS AND OUTPUTS DATA PASSING

Thermal system input	Thermal modeling output
S/C dimensions (length, diameter)	Heat pipe length and mass
Power management and electronics dimensions	Cold plate size and mass
Waste heat load to be rejected View factor to the Earth, lunar surface and solar array Solar flux	Radiator size and mass S/C insulation mass and thickness Thermal system components mass
Propellant tank dimensions and operating temperature	Propellant tanks insulation mass and heater power level
Propellant line lengths and operating temperature	Propellant line insulation mass and heater power level
Component minimum temperature requirements	Heating mass and power requirement

TABLE 5.8.—THERMAL SYSTEM RADIATOR SIZING ASSUMPTIONS

Variable	Value
Radiator solar absorptivity	0.14
Radiator emissivity	0.84
Radiator Sun angle	75°
Radiator operating temperature	320 k
Total radiator dissipation power	25 W
Lunar surface temperature	400 K (worst case)
Radiator angle to the horizontal	30°

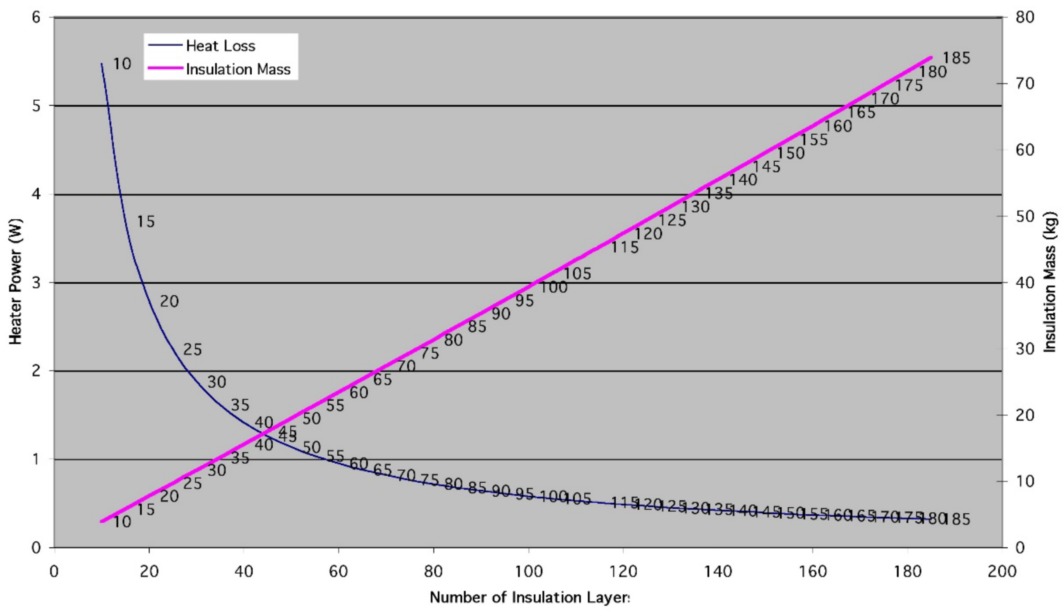


Figure 5.11.—MLI.

TABLE 5.9.—THERMAL SYSTEM TANK INSULATION SIZING ASSUMPTIONS

Variable	Value
Tank surface emissivity (ε_t)	0.1
MLI emissivity (ε_i)	0.07
MLI material	Al
MLI material density (ρ_i)	2,770 kg/m^3
Internal tank temperature (T_i)	300 K
MLI layer thickness (t_i)	0.025 mm
Number of insulation layers (n_i)	10
MLI layer spacing (d_i)	1.0 mm
Tank immersion heater mass and power level	1.02 kg at up to 1,000 W
S/C inner wall surface emissivity	0.98
S/C outer wall surface emissivity	0.93
Line foam insulation conductivity	0.0027 W/m K
Line foam insulation emissivity	0.07
Propellant line heater specific mass and power	0.143 kg/m at up to 39 W/m
Line foam insulation density	56 kg/m^3

TABLE 5.10.—THERMAL SYSTEM TANK INSULATION SIZING ASSUMPTIONS

Variable	Value
S/C MLI material	Al
S/C MLI material density (ρ_{isc})	2,770 kg/m^3
MLI layer thickness (t_i)	0.025 mm
Number of insulation layers (n_i)	100
MLI layer spacing (d_i)	1.0 mm

Thermal Analysis—PMAD Cooling

Thermal control of the electronics and Active Thermal Control System (ATCS) is accomplished through a series of cold plates and heat pipes to transfer the excess heat to the radiators. The model for sizing these components was based on a first principles analysis of the area needed to reject the identified heat load to space. From the sizing a series of scaling equations were used to determine the mass of the various system components. Assumptions used are listed in Table 5.11.

The Thermal System MEL for Case 2 (the ½ ASRG for power) is shown in Table 5.12. There is only a thermal system on the ILN Lander and not on the Star motor.

TABLE 5.11.—THERMAL SYSTEM PMAD
COOLING SIZING ASSUMPTIONS

Variable	Value
Cooling plate and lines material	Al
Cooling plate and lines material density	2,770 kg/m^3
Number of cooling plates	4
Cooling plate lengths	0.1 m
Cooling plate widths	0.1 m
Cooling plate thickness	5 mm
Heat pipe specific mass	0.15 kg/m

TABLE 5.12.—THERMAL SUBSYSTEM MEL (CASE 2)

WBS Number	Description International Lunar Network (may 2008)	QTY	Unit Mass (kg)	CBE Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	**Robotic Lunar Lander - Landing ILN**			**481.7**	**6%**	**30.2**	**511.9**
06.1	**ELV Adaptor**			**25.0**	**18%**	**4.5**	**29.5**
06.2	**Lunar Descent Stage**			**301.0**	**0%**	**1.5**	**302.5**
06.3	**Lunar Lander**			**140.2**	**14%**	**19.6**	**159.7**
06.3.10	**Thermal Control (Non-Propellant)**			**10**	**18%**	1.8	**11.7**
06.3.10.a	**Active Thermal Control**			2.0	18%	0.4	2.4
06.3.10.a.a	Heaters	4	0.1	0.6	18%	0.1	0.7
06.3.10.a.b	Thermal Control/Heaters Circuit	1	0.2	0.2	18%	0.0	0.2
06.3.10.a.c	Data Acquisition	1	1.0	1.0	18%	0.2	1.2
06.3.10.a.d	Thermocouples	25	0.0100	0.3	18%	0.0	0.3
06.3.10.b	**Passive Thermal Control**			6.1	18%	1.1	7.1
06.3.10.b.a	Heat Sinks	4	0.1	0.6	18%	0.1	0.7
06.3.10.b.b	Heat Pipes	1	0.3	0.3	18%	0.1	0.4
06.3.10.b.c	Radiators	2	1.0	2.0	18%	0.4	2.4
06.3.10.b.d	MLI (Multi Layer Insulation)	1	2.1	2.1	18%	0.4	2.4
06.3.10.b.e	Temperature sensors	25	0.010	0.3	18%	0.05	0.3
06.3.10.b.f	Phase Change Devices	2	0.4	0.8	18%	0.1	0.9
06.3.10.b.g	Thermal Coatings/Paint	1	0.055	0.1	18%	0.010	0.06
06.3.10.c	**Semi-Passive Thermal Control**			1.8	18%	0.3	2.2
06.3.10.c.a	Louvers	2	0.5	1.0	18%	0.2	1.2
06.3.10.c.b	Thermal Switches	4	0.2	0.8	18%	0.1	0.9
06.4	**ILN Science Payload**			**15.5**	**30%**	**4.7**	**20.2**

5.7.6 Thermal Risk Inputs

The risks associated with the thermal system are based mainly on the failure of a component or multiple components of the system. The majority of the system operation is passive and therefore has a fairly high reliability. Some of the major failure mechanisms are listed below.

- Heat pipe failure. This can be due to cracking due to thermal stresses, micrometeor impact or design defect. This likelihood of this type of failure is low. The impact of this failure would be a loss of all or a portion of the S/C's capability.
- Louver failure. A louver on the radiator panel can fail limiting the control of the heat flow from the radiator. The louvers on the radiator panels are individually actuated through a passive thermally activated spring. Failure of a single louver will still leave the remaining portion of the radiator functioning. The redundant radiator system provides sufficient margin to reject heat to account for any single or multiple louver failures in the closed position. If a louver fails in the open position it could provide a significant heat loss during the lunar nighttime. This heat loss might not be capable of being offset by the RHUs to maintain the desired internal temperature.
- Radiator failure. This can occur if there is significant buildup of lunar dust onto the radiator surface. This buildup can be caused by activity near and within the vicinity of the S/C. The likelihood of this type of failure is remote. The impact of this failure would be a gradual decrease in the thermal performance of the system over time with the final result being a failure of a portion of the electronics due to overheating.

5.7.7 Thermal Recommendation

To improve the reliability of the system and compensated for the identified failure risks the following system design changes can be made.

- Since redundant radiators are being utilized, any heat pipe failure can be offset by the other radiator system.
- To reduce the risk of a louver failure in the open position, it is suggested that the heat pipes going to each radiator have a pyro-activated valve that will vent the internal fluid. In the case of a louver failure in the open position the heat pipes to that radiator can be disabled thereby significantly reducing the heat transfer to that radiator.
- To reduce the risk of the radiator being contaminated with lunar dust, the radiator can be place as high above the lunar surface as possible. The greater the height the less lunar dust contamination will occur. Since the lunar dust particle motion follows a ballistic trajectory, a shield can be place along the edge of the radiator panel. This type of shield should significantly reduce the dust buildup on the radiator.

6.0 Software Cost Estimation

6.1 Assumptions

- SCS750D single board computer (SBC)
- Ada as baseline
- S/C bus functions:
    - ACS: 5500 source lines of code (SLOC)

    - This estimate includes: Sun sensor, Star Tracker, rate gyros, complex ephemeris, kinematic integration, error determination, thruster control, reaction wheel control, orbit propagation
  - C&DH: 550 SLOCs
    - Command and telemetry processing
  - Onboard autonomy (complex assumed): 4100 SLOCs
  - Fault detection (onboard systems monitoring and correction): 1650 SLOCs
  - Power management and thermal control: 500 SLOCs
- Payload functions:
  - Sensor processing: 670 SLOCs
  - Data Reduction and Transmission: 200 SLOCs
- Table 6.1 listed the SLOC estimation by function

6.2 Approach

- Programming language is Ada
- One SCS750 SBC
- Complex autonomy
- Real-time operating system (RTOS) on SCS750 SBC
- Ground software not included

7.0 Risk And Reliability

7.1 Risk Analysis and Reduction

The management of risk is a foundational issue in the design, development and extension of technology. Risk management is used to innovate and shape the future. Identifying risks early in the design cycle is a change to do better than planned. Each subsystem was tasked to write a risk statement regarding any concerns, issues and ‘ah ha’s’. Mitigation plans would focus on recommendations to alleviate, if not eliminate the risk.

TABLE 6.1.—SLOC ESTIMATION BY FUNCTION

S/C bus function	Code size, K words	Code size × 4, K words	SLOC
C&DH (command and telemetry processing)	2.0	8.0	550
Attitude determination and control (Sun sensor, Star Tracker, rate gyros, complex ephemeris, kinematic integration, error determination, thruster control, reaction wheel control, orbit propagation)	20.2	80.8	5500
Onboard autonomy	15.0	60.0	4100
Complex autonomy			
Fault detection			
Onboard systems monitoring	4.0	16.0	1100
Fault correction	2.0	8.0	500
Power management and thermal control	1.8	7.2	500
Residing in onboard computers assumed			
Subtotal	45.0	180.0	12300

Total (including payload functions): 13170 SLOCs.

7.2 Risk Assumptions

- Risk attributes are based on Crew Exploration Vehicle (CEV) risk values
- Risk list is not based on trends
- Some mitigation plans are offered as suggestions

7.2.1 Risk List

Table 7.1 provides the list of risks evaluated by the Compass team. Figure 7.1 provides the risk summary matrix.

TABLE 7.2.—RISK LIST

No. and criticality	Team and attribute values	Risk title, statement, and context
1 **Med**	Launch Vehicle Likelihood: 3 Consequences: Cost: 4 Schedule: 4 Perform: 4 Safety: 1	Launch Vehicle Performance and Cost. ➢ Given the unproven launch vehicle with cost limits imposed on this mission; there is a possibility that the vehicle will not meet performance requirements or require substantially higher costs for mission success. *Context*: Because of the cost limits imposed on this mission, unflown/unproven launch vehicles become attractive options to stay within the cost limits. These launch vehicles quote low costs and offer the performance required, but have not flown. If these launch vehicles don't meet performance goals, either the cost will increase for that particular launch vehicle, or a different, more expensive launch vehicle will have to be chosen for this mission. Mitigation Strategy: Watch: Watch the different low-cost launch vehicle development programs to assure that they meet performance and cost requirements for this mission.
2 **Med**	Communications Likelihood: 3 Consequences: Cost: 3 Schedule: 3 Perform: 4 Safety: 2	Antenna Aiming Mechanism ➢ Given the current ILN Lander design does not include an antenna aiming mechanism; there is a possibility that it will be difficult to accurately point the antenna to Earth that will have the best possible signals. *Context*: The ILN Lander design does not include an antenna steering/aiming mechanism due to mass constraint. As a result of mass constraint, the ILN lander will rely only on the altitude control for pointing the antenna to Earth within ±15° in all three directions. Without the antenna aiming mechanism, it will be difficult to accurately point the antenna to Earth that will have the best signals.
3 **Med**	Thermal Likelihood: 2 Consequences: Schedule: 2 Perform: 5 Cost: 2 Safety: 4	Heat Pipe (Louver) Failure ➢ Given the failure of one or both of the louvers that cover the radiators; there is a possibility that this will cause a failure in the drive mechanism for the louvers or its control system. *Context*: If this failure occurs it would limit the heat transfer from the S/C. If the failure occurred while the louvers were closed the electronics and interior components cooled by the radiator would overheat during the daytime. If it occurred while the louvers were open excessive heat would be lost during the nighttime, significantly reducing the component temperature and possibly damaging the electronics and other systems. Mitigation Strategy: The mitigation approach is to provide a redundant radiator/louver system. This could compensate for any problem that is caused by one of the louver failures.

TABLE 7.1.—Continued.

No. and criticality	Team and attribute values	Risk title, statement, and context
4	Thermal	Heat Pipe Failure
Med	Likelihood: 2 Consequences: Perform: 5 Schedule: 2 Cost: 2 Safety: 4	➢ Given the stress cracking, weld failure or micro meteor hit of the heat pipes; there is a possibility that of one or more heat pipes will fail due to an internal fluid leak; *Context*: If this failure occurs it would most likely lead to a failure of the electronics which are being cooled by the heat pipe system. Mitigation Strategy: The mitigation approach is to utilize micro meteor shielding on any exposed heat pipes (those going to the radiator), inspect any welds made in the pipes and design the system to minimize stress on the heat pipes.
5	Propulsion	RCS thruster plume impingement
Med	Likelihood: 3 Consequences: Safety: 3 Cost: 2 Perform: 3 Schedule: 3	➢ Given the number of RC thrusters mounted on the body of the S/C, there is a possibility that RCS thruster plume may impinge on sensitive surfaces thereby degrading performance. Additionally, plume impingement could significantly compromise science return capability. *Context*: The RC thrusters need to be mounted on several locations on the S/C surface, some of which are in the proximity of critical features such as the solar arrays and communications antennas. While no plume directly impinges on S/C surfaces, these plumes could result in deposition of reaction products on nearby surfaces. The degree of deposition and resulting impact of that deposition is not presently known and requires further study to quantify. Additionally, RCS plume could impinge on the Moon's surface during landing and contaminate the surface material with propellant reaction products that will likely interfere with planned science activities. This is highly undesirable for the principal investigator. Mitigation Strategy: The primary approach to mitigating degradation due to deposition is to phase positioning of the movable systems in the proximity of the operating thruster to ensure maximum separation from its plume. To mitigate surface contamination, a cold gas rocket is being investigated for the final landing burn.
6	GNC	Single String System Design
Med	Likelihood: 2 Consequences: Perform: 4	➢ Given the single string system design for propulsion and GN&C; there is a possibility that GN&C or propulsion will experience a failure in the delivery system.
7	Thermal	Heater Failure
Med	Likelihood: 2 Consequences: Perform: 4 Safety: 3 Cost: 2 Schedule: 2	➢ Given the failure of one or more internal heaters, used to maintain a minimum electronics and component temperature; there is a possibility that the electronics or components will not function properly and could jeopardize mission success. *Context*: If this failure occurs it could lead to the failure of the electronics or component and jeopardize all or part of the mission Mitigation Strategy: The mitigation approach is to utilize redundant heaters and MLI in order to minimize any effects of a heater failure.

TABLE 7.1.—Concluded.

No. and criticality	Team and attribute values	Risk title, statement, and context
8	Structures	Potential Structural Failure
Med	Likelihood: 2 Consequences: Safety: 1 Perform: 4 Cost: 4 Schedule: 4	➢ Given the excessive g loads, impact from a foreign object, or excessive landing velocity may cause too much deformation, vibrations, or fracture of sections of the support structure; there is a possibility that lower performance from mounted hardware could result leading to loss of mission. *Context*: Excessive deformation of the structure can misalign components dependent on precise positioning, therefore, diminishing their performance. Internal components may be damaged or severed from the rest of the system resulting in diminished performance or incapacitation of the system. Excessive vibrations may reduce instrumentation performance and/or potentially lead to structural failure due to fatigue. Overall, the mission may not be completed in an optimum manner or it can be terminated in the worst case. Mitigation Strategy: The structure is to be designed to NASA standards to withstand expected g loads, a given impact, and to have sufficient stiffness and damping to minimize issues with vibrations. Trajectories are to be planned to minimize the probability of impact with foreign objects and prevent any damage upon landing..
9	Propulsion	Freezing of liquid propellant in distribution system
Low	Likelihood: 2 Consequences: Perform: 2 Schedule: 2 Cost: 3	➢ Given the relatively high temperatures required to keep hydrazine liquid, a freezing event of the fuel within the propellant distribution system could result in loss of a portion of the thrusting capability and, if the delivery lines are compromised, the potential release of hydrazine into the S/C interior *Context*: The ILN S/C will be in space for the majority of the operating lifetime for the propulsion subsystem and therefore will be exposed to extremely low temperatures. In the event of power loss, even a short one, the propellant temperature may drop below its freezing point of approximately 1.2 °C and solidify within the lines. Upon resumption of operation, the frozen propellant may melt such that the line is ruptured or the propellant may remain frozen thereby resulting in the loss of the thrusters downstream of the obstruction. Mitigation Strategy: Ensure that adequate insulation installed on all propellant lines and consider using a redundant line and tank heater configuration for fault tolerance. In the event of suspected freezing of the chemical propellant, all scientific data should be recovered to a ground station. If possible, the S/C should be reoriented to exposure the suspected location of the freezing to as much solar/thermal input as possible to promote melting of the propellant. Evaluation of the feed system sensors should be performed to determine if related hardware problems were responsible (i.e., failing line heater). If possible, the propulsion system should be tested to determine performance capability.

No.	L×C	Team	Risk title
1	3×4	Launch vehicle	Launch vehicle performance and cost
2	3×4	Communications	Antenna airing mechanism
3	2×5	Thermal	Heat pipe (louver) failure
4	2×5	Thermal	Heat pipe failure
5	3×3	Propulsion	RCS thruster plume impingement
6	2×4	GN&C	Single string system design
7	2×4	Thermal	Heater failure
8	2×4	Structures	Potential structural failure
9	2×3	Propulsion	Freezing liquid propellant in distribution system

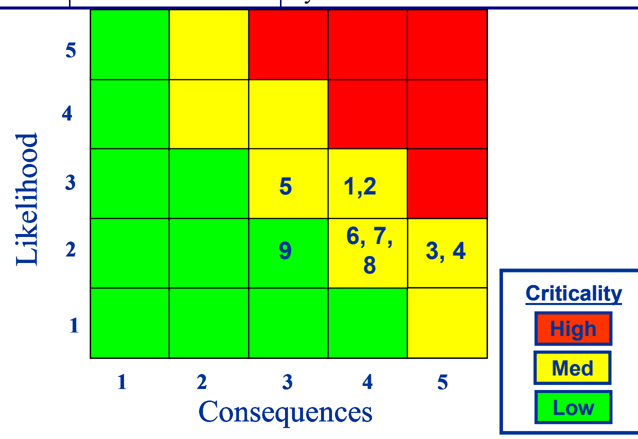


Figure 7.1.—Risk summary matrix.

7.2.2 Risk Summary

These risks, with proper pro-active planning can be mitigated early to avoid becoming problems late in the development life cycle.

8.0 Trade Space Iterations

Cases 1 through 3 are all designed to be launched on a Taurus II. After reducing requirements for science and power, Case 4 was able to fit onto a Minotaur V with minimal launch margin available. Launch margin available is calculated as the difference between the total wet mass (with system level growth applied) of the stack (adaptor, star, lander and payload) and the performance to TLI of the launch vehicle.

8.1 Trade Space Summary

Table 8.1 shows the mission summary for all four cases. Figure 8.1 illustrates Cases 1 to 3.

Table 8.2 shows the differences in the four concept designs between the top-level details of the different major components that make up the ILN lander system: ELV adaptor, Lunar Descent Stage, Lunar Lander Stage and Science payload.

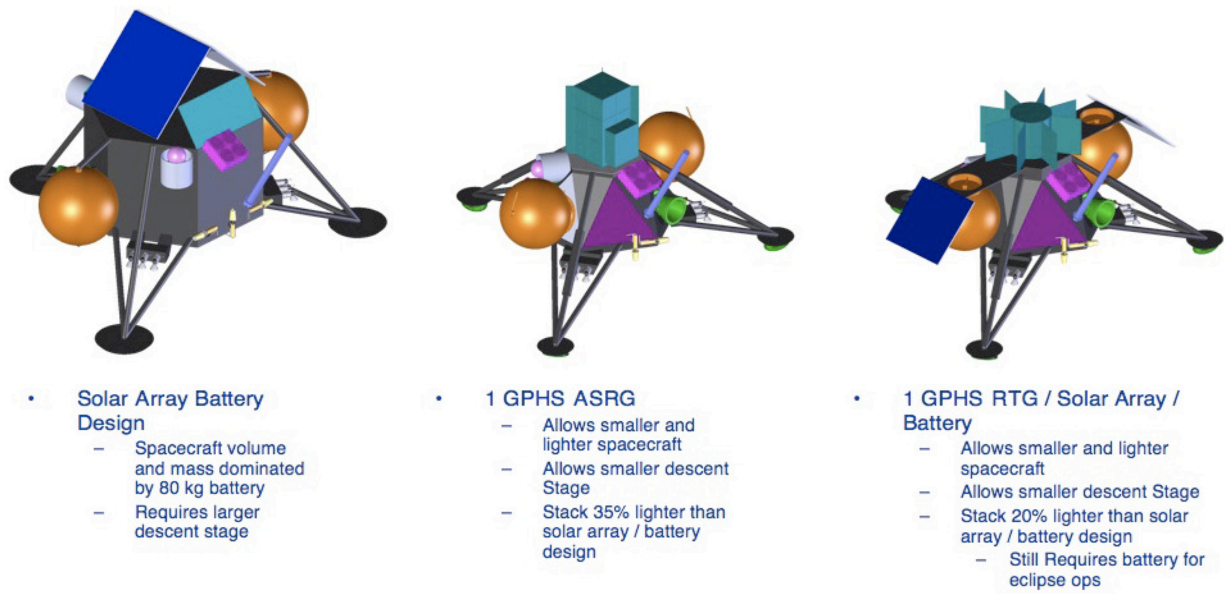


Figure 8.1.—Cases 1 to 3 trade designs.

TABLE 8.1.—MISSION SUMMARY FOR CASES 1 TO 4

Case name	Case 1	Case 2	Case 3	Case 4
Mission Details	**ILN_case1**	**ILN_case2_asrg**	**ILN_case3_rtg**	**ILN_case4_asrg_minV**
Architecture Details	**Mass (kg)**	**Mass (kg)**	**Mass (kg)**	**Mass (kg)**
Launch Vehicle	**Taurus II**	**Taurus II**	**Taurus II**	**Minotaur V**
Solid Rocket Descent Motor	Star 30BP	Star 27	Star 27	Star 24C
Performance to TLI	1039	1039	1039	387
ELV Margin	0.1	0.1	0.1	0.1
Total ELV performance	1190.00	1190.00	1190.00	466.00
Calculated Spacecraft Margin	238.63	532.87	425.18	6.30
Spacecraft Adaptor to ELV Mass	**30**	**30**	**30**	**30**
SC Wet Mass w/ System Growth	**804**	**503**	**612**	**378**
Calculated Launch Margin Available	**235**	**536**	**426**	**9**

TABLE 8.2.—TOP LEVEL SYSTEM SUMMARY FOR CASES 1 TO 4

Case Name	Case 1	Case 2	Case 3	Case 4
GLIDE Study Case name	**ILN_case1**	**ILN_case2_asrg**	**ILN_case3_rtg**	**ILN_case4_asrg_minV**
	Total Mass (kg)	**Total Mass (kg)**	**Total Mass (kg)**	**Total Mass (kg)**
Total ELV Adaptor Dry Mass	*32.5*	*32.5*	*32.5*	*32.5*
Total Lunar Descent Stage Dry Mass	*67.0*	*38.0*	*39.8*	*29.8*
Total Lunar Lander Stage Dry Mass	*234.4*	*141.2*	*179.8*	*112.7*
Total Science Payload Dry Mass	*20.2*	*20.2*	*20.2*	*7.8*
Estimated Stack Dry Mass (on pad)	**354.0**	**231.9**	**272.3**	**182.7**
Estimated Stack Inert Mass (for traj.)	**356.4**	**233.4**	**274.1**	**183.8**
Total S/C Wet Mass (on pad w/ adaptor)	**836.3**	**535.2**	**644.9**	**410.9**
Total SC Wet Mass w/ Growth (no adaptor)	**803.8**	**502.7**	**612.4**	**378.4**
Total Landed Spacecraft (inert)	**254.5**	**161.4**	**200.0**	**120.5**
Available Launch Performance to TLI (kg)	1038.5	1038.5	1038.5	386.9
Launch margin available (kg)	**234.7**	**535.8**	**426.1**	**8.5**

Figure 8.2 shows the relative sizes of the first four case designs including the Lunar Descent Stage (the star motor), and the Lander itself.

The ILN system was divided into four major WBS items:

- ELV adaptor
  - dropped with the ELV
- Lunar descent stage
  - The Star Motor, adaptor and propellant
- Lunar lander stage
- ILN science payload

The COMPASS team uses the ANSI/AIAA R-020A-1999, Recommended Practice for Mass Properties Control for Satellites, Missiles, and Launch Vehicles (Ref. 2). The percent MGA factors are applied to each subsystem, after which the total system growth of the design is calculated. All masses are presented in Table 8.3 with the total masses calculated after the subsystem MGA factors are applied. Inert mass is dry mass plus any trapped residuals left onboard that the landing propulsion system has to land on the lunar surface. In order to meet the goal of 30 percent growth on dry mass at the system level, an allocation is necessary for system level growth in each case. Total wet mass with growth is the mass of the four major WBS systems, with propellant, and subsystem MGA factors applied as well as the additional system level growth carried to bring the growth on dry mass up to 30 percent.

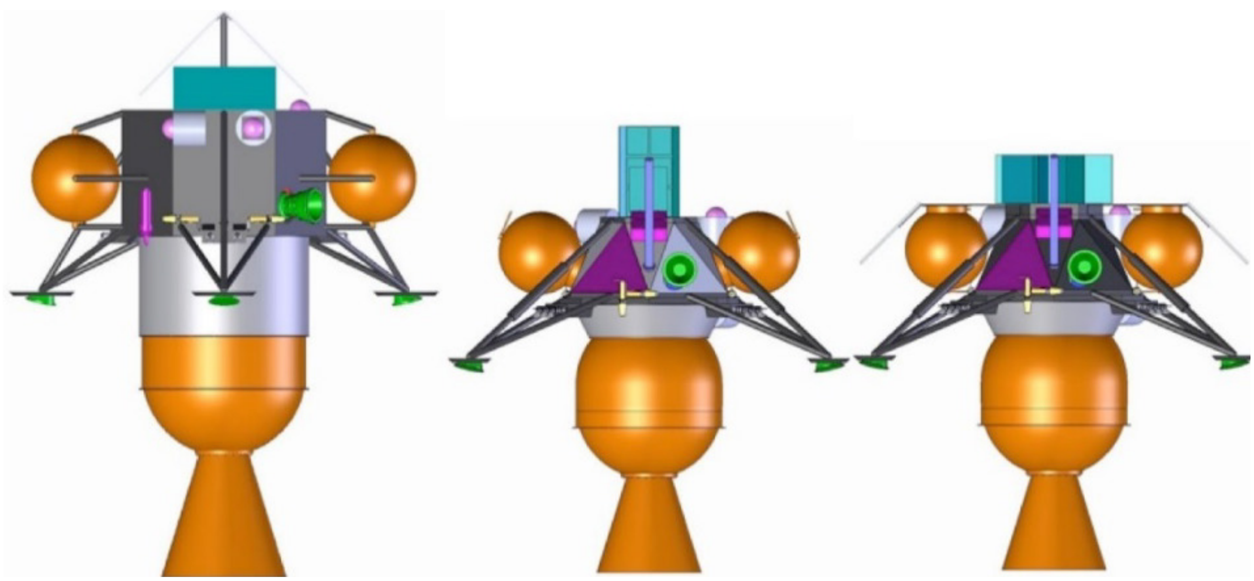


Figure 8.2.—Cases 1 to 3 full stack concept designs.

TABLE 8.3.—SUBSYSTEM TOTAL MASS SUMMARY FOR CASES 1 TO 4

	Case Name	Case 1	Case 2	Case 3	Case 4
	GLIDE Study Case name	ILN_case1	ILN_case2_asrg	ILN_case3_rtg	ILN_case4_asrg_minV
WBS	Main Subsystems	Total Mass (kg)	Total Mass (kg)	Total Mass (kg)	Total Mass (kg)
06	Robotic Lunar Lander - Landing ILN	801.6	511.9	618.2	392.6
06.1	*ELV Adaptor*	29.5	29.5	29.5	29.5
06.1.11	Structures & Mechanical Systems	29.5	29.5	29.5	29.5
06.2	*Lunar Descent Stage*	487.3	302.5	364.8	229.1
06.2.1	C&DH, Communications GN&C	0.0	0.0	0.0	0.0
06.2.2	Propulsion Hardware (Chemical)	39.2	28.6	30.0	20.5
06.2.3	Propellant Management (Chemical)	0.0	0.0	0.0	0.0
06.2.4	Propellant (Chemical)	431.4	271.8	332.7	204.8
06.2.11	Structures & Mechanical Systems	16.6	2.1	2.1	3.8
06.3	*Lunar Lander Stage*	264.7	159.7	203.7	126.2
06.3.1	C&DH, Communications GN&C	26.7	26.7	26.7	26.7
06.3.2	Propulsion Hardware (Chemical)	8.3	7.5	7.5	5.8
06.3.3	Propellant Management (Chemical)	22.5	19.7	19.2	15.0
06.3.4	Propellant (Chemical)	50.8	31.6	39.9	23.3
06.3.9	Electrical Power	110.3	31.0	66.9	27.7
06.3.10	Thermal Control (Non-Propellant)	11.7	11.7	11.7	11.7
06.3.11	Structures & Mechanical Systems	34.3	31.5	31.8	15.9
06.4	*ILN Science Payload*	20.2	20.2	20.2	7.8
	Estimated Stack Dry Mass	319.3	208.5	245.6	164.5
	Estimated ELV Adaptor Inert Mass	30	30	30	30
	Estimated Lunar Descent Stage Inert Mass	56	31	32	24
	Estimated Lunar Lander Stage Inert Mass	217	130	166	104
	Estimated Science Payload Inert Mass	20.2	20.2	20.2	7.8
	Estimated Stack Wet Mass	801.6	511.9	618.2	392.6
System LeveL Growth Calculations					
	Total Estimated Growth	47.0	30.2	36.1	23.9
	Dry Mass Desired System Level Growth	354.0	231.9	272.3	182.7
	Additional Growth (carried at system level)	34.7	23.3	26.7	18.3
	Total Wet Mass with Growth	836.3	535.2	644.9	410.9

8.2 Trade 1—Case 1—Battery Powered ILN Soft Lander)

Case one used a battery to provide power through the lunar night. This lander version required heavier and stronger structures and landing legs to support the larger mass of the batter system. The foot pads on the landing legs were 12 in. in diameter.

Science

- Seismometer (bottom deployed), magnetometer, laser retroreflector, subsoil thermal probe (Mole deployed)

Mission Summary

Table 8.4 provides the system summary for Case 1.

- Chemical direct injection mission (aka Surveyor)
- Taurus II to TLI and direct lunar injection (~80 W for transit)
- Star 30 Solid Rocket motor for descent (~350 W for descent)
- Pulsed hydrazine propulsion system for final landing using radar (aka Phoenix)
- >6-year life, anywhere landing nearside of Moon, ~23 W for payload/housekeeping during lunar night
- Batteries during the lunar night and solar arrays in the day for power.

TABLE 8.4.—SYSTEM SUMMARY FOR CASE 1

Spacecraft Master Equipment List Rack-up (Mass)				COMPASS S/C Design	Case 1
WBS	**Main Subsystems**	**CBE Mass (lkg)**	**Growth (kg)**	**Total Mass (kg)**	**Aggregate Growth (%)**
06	Robotic Lunar Lander - Landing ILN	**754.6**	**47.0**	**801.6**	
06.1	*ELV Adaptor*	**25.0**	**4.5**	**29.5**	18%
06.2	*Lunar Descent Stage*	**483.0**	**4.3**	**487.3**	1%
06.3	*Lunar Lander Stage*	**231.1**	**33.6**	**264.7**	15%
06.3.1	C&DH, Communications GN&C	22.1	4.7	**26.7**	21%
06.3.2	Propulsion Hardware (Chemical)	7.7	0.6	**8.3**	8%
06.3.3	Propellant Management (Chemical)	20.3	2.1	**22.5**	11%
06.3.4	Propellant (Chemical)	50.8	0.0	**50.8**	0%
06.3.9	Electrical Power	91.0	19.3	**110.3**	21%
06.3.10	Thermal Control (Non-Propellant)	9.9	1.8	**11.7**	18%
06.3.11	Structures & Mechanical Systems	29.2	5.0	**34.3**	17%
06.4	*ILN Science Payload*	**15.5**	**4.7**	**20.2**	30%
	Estimated Stack Dry Mass	**272.3**	**47.0**	**319.3**	**17%**
	Estimated ELV Adaptor Inert Mass	**25**	**5**	**30**	**18%**
	Estimated Lunar Descent Stage Inert Mass	**52**	**4**	**56**	**8%**
	Estimated Lunar Lander Stage Inert Mass	**183**	**34**	**217**	**18%**
	Estimated Science Payload Inert Mass	**15.5**	**4.7**	**20.2**	**30%**
	Estimated Stack Wet Mass	**754.6**	**47.0**	**801.6**	
System LeveL Growth Calculations					**Total Growth**
	Total Estimated Growth		47.0		
	Dry Mass Desired System Level Growth	272.3	81.7	**354.0**	**30%**
	Additional Growth (carried at system level)		34.7		**13%**
	Total Wet Mass with Growth	755	81.7	**836.3**	

Chemical Lander

Figure 8.3 illustrates the Case 1 battery powered soft ILN lander.

- Power: 100 W GaAs solar arrays (fixed) and 80 kg Li-ion batteries for lunar night operations
- Propulsion: Blow-down pulsed monocrop system for transit and solid rocket burn ACS and final landing and pointing
- C&DH: SCS750 main processor with 1.4 Gb storage
- Communications: 5 W DC S-band communications through helical antenna (average during sunlight periods)
- GN&C: Star-Tracker, Sun sensors, mini-IMU, landing radar (Phoenix)
- Structures: Predeployed, energy absorbing leg landing system (~15 cm displacement, Al-Li hexagonal structure

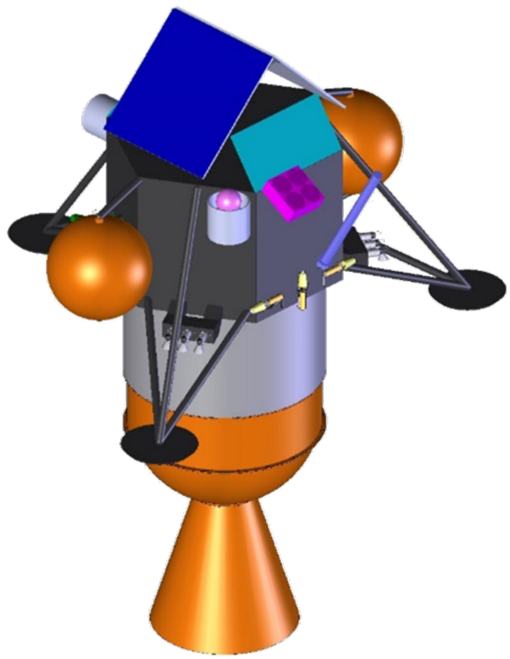
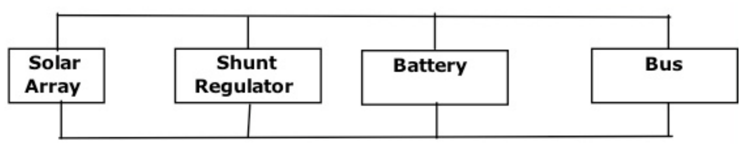


Figure 8.3.—Case 1 battery powered soft ILN lander.



Description	Quantity	Unit mass (kg)	Growth (%)	Total
Battery	1	79.0	17	94.8
Solar array	1	4	16	4.8
PMAD	1	8	26	10.8
			Totals	**110.8**

Figure 8.4.—Case 1 SA and battery system details.

Power System Design Highlights

Figure 8.4 provides the Case 1 solar array and battery system details.

- Two fixed panels (east/west) solar arrays
    - Triple junction solar cells
    - Conversion efficiency is 30 percent
    - Power cabling, wiring harness, power storage during coast operations assumptions, etc.
    - Li-ion batteries for night operations
    - Battery dimensions: 40- by 40- by 40-cm
    - Solar arrays: two 54- by 54-cm
- Technology status
    - Solar array: TRL-9
    - Batteries: TRL-6
        - Batteries flown before but not in the lunar environment

– Assumed ABSL 18650 HC based battery
  - Solar arrays mass: two 0.5 kg = 1 kg
  - Battery mass: 79 kg
  - Total power system mass with margin 110.3 kg

8.3 Trade 2—Case 2—Half ASRG Powered ILN Soft Lander—Taurus II ELV

Science

- Seismometer (bottom deployed), magnetometer, laser retroreflector, subsoil thermal probe (Mole deployed)

Mission Summary

Table 8.5 provides the top level system summary for Case 2.

- Chemical direct injection mission (aka Surveyor)
- Taurus II to TLI and direct lunar injection (~80 W for transit)
- Star 27 Solid Rocket motor for descent (~350 W for descent), disposed of before landing
- Pulsed hydrazine propulsion system for final landing using radar (aka Phoenix)
- >6-year life, anywhere landing nearside of Moon, ~23 W for payload/housekeeping during a lunar night

TABLE 8.5.—TOP LEVEL SYSTEM SUMMARY FOR CASE 2

Spacecraft Master Equipment List Rack-up (Mass)				COMPASS S/C Design	Case 2
WBS	Main Subsystems	CBE Mass (lkg)	Growth (kg)	Total Mass (kg)	Aggregate Growth (%)
06	Robotic Lunar Lander - Landing ILN	481.7	30.2	511.9	
06.1	*ELV Adaptor*	25.0	4.5	29.5	18%
06.2	*Lunar Descent Stage*	301.0	1.5	302.5	0%
06.3	*Lunar Lander Stage*	140.2	19.6	159.7	14%
06.3.1	C&DH, Communications GN&C	22.1	4.7	26.7	21%
06.3.2	Propulsion Hardware (Chemical)	6.9	0.6	7.5	8%
06.3.3	Propellant Management (Chemical)	17.7	2.1	19.7	12%
06.3.4	Propellant (Chemical)	31.6	0.0	31.6	0%
06.3.9	Electrical Power	25.1	5.9	31.0	24%
06.3.10	Thermal Control (Non-Propellant)	9.9	1.8	11.7	18%
06.3.11	Structures & Mechanical Systems	26.9	4.6	31.5	17%
06.4	*ILN Science Payload*	15.5	4.7	20.2	30%
	Estimated Stack Dry Mass	**178.4**	**30.2**	**208.5**	**17%**
	Estimated ELV Adaptor Inert Mass	**25**	**5**	**30**	**18%**
	Estimated Lunar Descent Stage Inert Mass	**29**	**1**	**31**	**5%**
	Estimated Lunar Lander Stage Inert Mass	**110**	**20**	**130**	**18%**
	Estimated Science Payload Inert Mass	**15.5**	**4.7**	**20.2**	**30%**
	Estimated Stack Wet Mass	**481.7**	**30.2**	**511.9**	
System LeveL Growth Calculations					**Total Growth**
	Total Estimated Growth		30.2		
	Dry Mass Desired System Level Growth	178.4	53.5	231.9	30%
	Additional Growth (carried at system level)		23.3		13%
	Total Wet Mass with Growth	482	53.5	535.2	

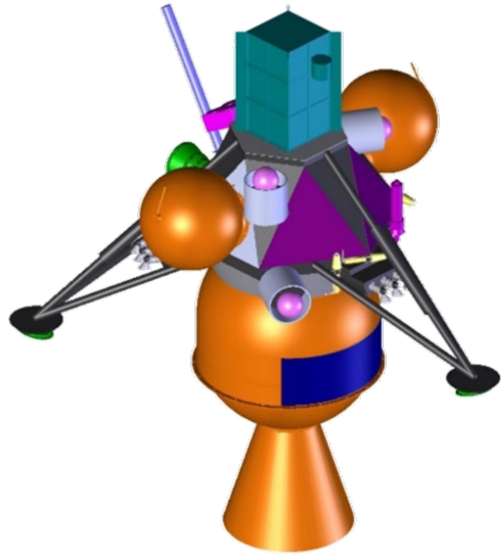


Figure 8.5.—Case 2 ASRG powered soft ILN lander.

Chemical Lander

- Figure 8.5 illustrates the Case 2 ASRG powered soft ILN lander. Power: 1 GPHS, single Stirling ASRG for landed operations, SA for transit power, battery for landing power
- Propulsion: Blow-down pulsed monoprop system for transit and solid rocket burn ACS and final landing and pointing
- C&DH: SCS750 Main processor with 1.4 Gb storage
- Communications: 5 W DC S-band comm. through helical antenna (average during sunlight periods)
- GN&C: Star-Tracker, Sun sensors, mini-IMU, landing radar (Phoenix)
- Structures: Predeployed, energy absorbing leg landing system (~15 cm displacement, Al-Li hexagonal structure

8.4 Trade 3—Case 3—RTG Powered ILN Soft Lander

Science

- Seismometer (bottom deployed), magnetometer, laser retroreflector, subsoil thermal probe (Mole deployed)

Mission Summary

Table 8.6 provides the top-level system summary for Case 3.

- Chemical direct injection mission (aka Surveyor)
- Taurus II to TLI and direct lunar injection (~80 W for transit)
- Star 27 Solid Rocket motor for descent ((~350 W for descent) disposed of before landing)
- Pulsed hydrazine propulsion system for final landing using radar (aka Phoenix)
- >6-year life, anywhere landing nearside of Moon, ~23 W for payload/housekeeping during a lunar night

TABLE 8.6.—TOP LEVEL SYSTEM SUMMARY FOR CASE 3

Spacecraft Master Equipment List Rack-up (Mass)				S/C Design	Case 3
WBS	Main Subsystems	CBE Mass (lkg)	Growth (kg)	Total Mass (kg)	Aggregate Growth (%)
06	Robotic Lunar Lander - Landing ILN	582.1	36.1	618.2	
06.1	*ELV Adaptor*	25.0	4.5	29.5	18%
06.2	*Lunar Descent Stage*	363.3	1.5	364.8	0%
06.3	*Lunar Lander Stage*	178.2	25.5	203.7	14%
06.3.1	C&DH, Communications GN&C	22.1	4.7	26.7	21%
06.3.2	Propulsion Hardware (Chemical)	6.9	0.6	7.5	8%
06.3.3	Propellant Management (Chemical)	17.2	1.9	19.2	11%
06.3.4	Propellant (Chemical)	39.9	0.0	39.9	0%
06.3.9	Electrical Power	55.0	11.9	66.9	22%
06.3.10	Thermal Control (Non-Propellant)	9.9	1.8	11.7	18%
06.3.11	Structures & Mechanical Systems	27.2	4.6	31.8	17%
06.4	*ILN Science Payload*	15.5	4.7	20.2	30%
	Estimated Stack Dry Mass	209.4	36.1	245.6	17%
	Estimated ELV Adaptor Inert Mass	25	5	30	18%
	Estimated Lunar Descent Stage Inert Mass	31	2	32	5%
	Estimated Lunar Lander Stage Inert Mass	141	25	166	18%
	Estimated Science Payload Inert Mass	15.5	4.7	20.2	30%
	Estimated Stack Wet Mass	582.1	36.1	618.2	
System LeveL Growth Calculations					Total Growth
	Total Estimated Growth		36.1		
	Dry Mass Desired System Level Growth	209.4	62.8	272.3	30%
	Additional Growth (carried at system level)		26.7		13%
	Total Wet Mass with Growth	582	62.8	644.9	

Chemical Lander

- Figure 8.6 illustrates the Case 3 RTG powered soft ILN lander. Power: 1 GPHS, RTG, 30 kg battery and SA for landed operations, SA for transit power, battery for landing power
- Propulsion: Blowdown pulsed monoprop system for transit and solid rocket burn ACS and final landing and pointing
- C&DH: SCS750 Main processor with 1.4 Gb storage
- Communications: 5 W DC S-band Comm. Through helical antenna (average during sunlight periods)
- GN&C: Star-Tracker, Sun sensors, mini-IMU, landing radar (Phoenix)
- Structures: Predeployed, energy absorbing leg landing system (~15 cm displacement, Al-Li hexagonal structure

Power Assumptions

- 26 W day
- 23 W night
- Peak loads during transit
- Batteries capable of providing power during landing

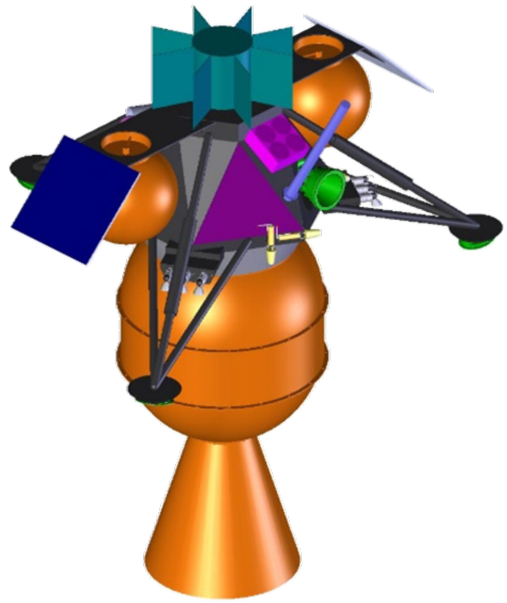


Figure 8.6.—Case 3 RTG powered soft ILN lander.

Design Highlights

Figure 8.7 and Figure 8.8 illustrate the RTG and the RTG solar power system schematic.

- Single GPHS RTG providing 14 W day/night
  - Material of TE PbTe, 3 V output
- Two fixed panels (east/west) solar arrays
  - Triple junction solar cells
  - Conversion efficiency is 30 percent
  - Power cabling, wiring harness, power storage during coast operations assumptions, etc.
  - Li-ion batteries for night operations
  - RTG reduces battery energy by 50 percent
  - Battery dimensions : 14- by 14- by 14-cm
  - RTG dimensions : 23- by 57-cm diameter
  - Solar arrays: two 44- by 44-cm
- Technology status
  - Solar array: TRL-9
  - GPHS RTG: TRL -6
  - Batteries: TRL-6
    - Batteries flown before but not in the lunar environment. Assumed ABSL 18650 HC based battery

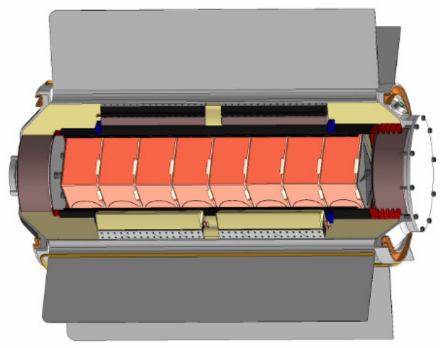
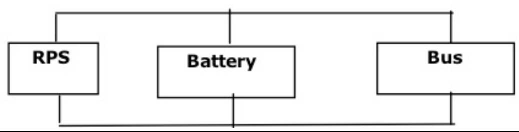


Figure 8.7.—RTG.



Description	Quantity	Unit mass (kg)	Growth (%)	Total
RTG	1	10.9	18	12.9
Battery	1	34.0	20	40.8
Solar array	1	2.1	16	2.5
PMAD	1	8	26	10.8
			Totals	**66.9**

Figure 8.8.—RTG—Solar array power system schematic.

8.5 Trade 4—Case 4—Science Floor ASRG Powered ILN Soft Lander—Minotaur V ELV

In order to design a version of the ILN that would fit within the performance of the Minotaur V launch vehicle, the lander from Case 2 using the ½ ASRG power system was redesigned. The science package was reduced to the minimum science floor as scoped out in Section 2.9 and the solid rocket motor used for lunar descent was reduced from the Star 27 in case 2 to a Star 24. Table 8.7 lists the top level system summary for Case 4.

Science

- Seismometer (bottom deployed) only

Mission Summary

- Chemical direct injection mission (aka Surveyor)
- Minotaur V to TLI and direct lunar injection (~80 W for transit)
- Star 24 Solid Rocket motor for descent (~350 W for descent)
- Pulsed hydrazine propulsion system for final landing using radar (aka Phoenix)
- >6-year life, anywhere landing nearside of Moon, ~15 W for payload/housekeeping during a lunar night

TABLE 8.7.—TOP LEVEL SYSTEM SUMMARY FOR CASE 4

Main Subsystems	CBE Mass (lkg)	Growth (kg)	Total Mass (kg)	Aggregate Growth (%)
Robotic Lunar Lander - Landing ILN	**368.7**	**23.9**	**392.6**	
ELV Adaptor	**25.0**	**4.5**	**29.5**	18%
Lunar Descent Stage	**227.7**	**1.4**	**229.1**	1%
Lunar Lander Stage	**110.0**	**16.2**	**126.2**	15%
C&DH, Communications GN&C	22.1	4.7	**26.7**	21%
Propulsion Hardware (Chemical)	5.4	0.4	**5.8**	8%
Propellant Management (Chemical)	13.5	1.5	**15.0**	11%
Propellant (Chemical)	23.3	0.0	**23.3**	0%
Electrical Power	22.3	5.4	**27.7**	24%
Thermal Control (Non-Propellant)	9.9	1.8	**11.7**	18%
Structures & Mechanical Systems	13.5	2.4	**15.9**	18%
ILN Science Payload	**6.0**	**1.8**	**7.8**	30%
Estimated Stack Dry Mass	**140.6**	**23.9**	**164.5**	**17%**
Estimated ELV Adaptor Inert Mass	**25**	**5**	**30**	**18%**
Estimated Lunar Descent Stage Inert Mass	**23**	**1**	**24**	**6%**
Estimated Lunar Lander Stage Inert Mass	**88**	**16**	**104**	**18%**
Estimated Science Payload Inert Mass	**6.0**	**1.8**	**7.8**	**30%**
Estimated Stack Wet Mass	**368.7**	**23.9**	**392.6**	
. Growth Calculations				**Total Growth**
Total Estimated Growth		23.9		
Dry Mass Desired System Level Growth	140.6	42.2	**182.7**	**30%**
Additional Growth (carried at system level)		18.3		**13%**
Total Wet Mass with Growth	369	42.2	**410.9**	

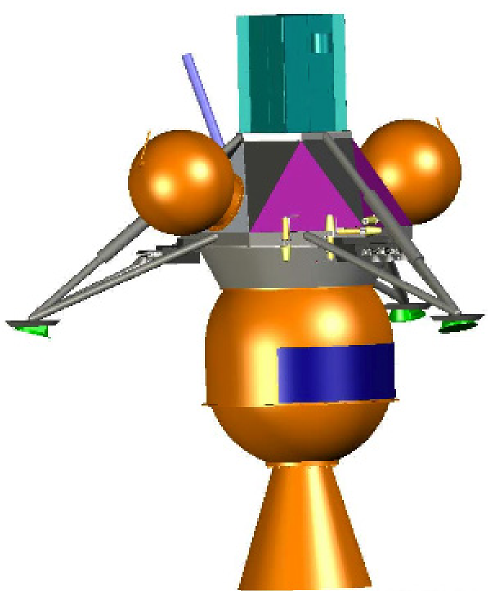


Figure 8.9.—Case 4 ASR powered soft ILN lander, on Minotaur V launch.

Chemical Lander

Figure 8.9 illustrates the Case 4 ASR powered soft ILN lander.

- Power: 1 GPHS, single Stirling ASRG for landed operations, solar array for transit power, battery for landing power

- Propulsion: Blow-down pulsed monoprop system for transit and solid rocket burn ACS and final landing and pointing
- C&DH: SCS750 main processor with 1.4 Gb storage
- Communications: 5 W DC S-band communications through helical antenna (average during sunlight periods)
- GN&C: Star-Tracker, Sun sensors, mini-IMU, landing radar (Phoenix)
- Structures: Predeployed, energy absorbing leg landing system (~15 cm displacement, Al-Li hexagonal structure

Appendix A.—Acronyms and Abbreviations

ACS	Attitude Control System
AIAA	American Institute for Aeronautics and Astronautics
Al	aluminum
ALSEP	Apollo Lunar Surface Experiments Package
ANSI	American National Standards Institute
AO	Announcement of Opportunity
APL	Applied Physics Laboratory
ASRG	Advanced Stirling Radioisotope Generators
ATCS	Active Thermal Control System
BER	bit error rate
BOL	beginning of life
C&DH	Command and Data Handing
CBE	current best estimate
CER	cost estimation relationship
COMPASS	COllaborative Modeling and Parametric Assessment of Space Systems
COTS	NASA Commercial Orbital Transportation Services
DDT&E	design, development, test, and evaluation
DOF	degree of freedom
DLR	Germany Aerospace Center
EDS	Earth departure stage
EIRP	effective isotropic radiated power
ELV	expendable launch vehicle
EOL	end of life
ESA	European Space Agency
FOM	figure of merit
GaAs	gallium arsenide
GLIDE	GLobal Integrated Design Environment
GN&C	Guidance, Navigation and Control
GPHS	general purpose heat source
GRC	NASA Glenn Research Center
GTO	geosynchronous transfer orbits
HC	Hard Carbon
ILN	International Lunar Network
IMU	Inertial Measuring Unit
Li	lithium
MEL	Master Equipment List
MGA	Mass Growth Allowance
MLI	multilayer insulation
MMOD	Micrometeoroid and Orbital Debris
MMRTG	Multi-Mission Radioisotope Thermoelectric Generators
MSFC	NASA Marshall Spaceflight Center
NASA	National Aeronautics and Space Administration
O	oxygen
OTS	off-the-shelf
Pb	lead
PEL	Power Equipment List
PLUTO	PLanetary Underground Tool
PMAD	Power Management and Distribution
Pu	plutonium
RBI	regulator/bus protection
RC	reaction control
RCS	Reaction Control System
RF	radio frequency
RHU	Radioisotope Heater Unit
ROLAND	Rosetta Lander-Magnetometer
ROMAP	ROsetta Lander MAgnetometer and Plasma Monitor
RPS	Radioisotope Power System
RSLP	U.S. Air Force Rocket System Launch Program
RTG	Radioisotope Thermoelectric Generators
RTOS	real-time operating system
SBC	single board computer
SLOC	source lines of code
SLV	Space Launch Vehicle
SMD	NASA's Science Mission Directorate
SOAP	Satellite Orbit Analysis Program
SPM	Lander Plasma Monitor
TBD	to be discussed
Te	tellurium
TLI	translunar injection
TRL	Technology Readiness Level

TT&C	Telemetry, Tracking and Command
VBB	very broad band
WBS	Work Breakdown Structure
WGA	Weight Growth Allowance
Xe	xenon

Appendix B.—Study Participants

International Lunar Network Design Session			
Subsystem	**Name**	**Center**	**Email**
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Mission	Laura Burke	GRC	Laura.M.Burke@nasa.gov
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Risk/Reliability	Anita Tenteris-Noebe	GRC	Anita.D.Tenteris@nasa.gov
Risk/Reliability	Joe Hemminger	GRC	Joseph.A.Hemminger@nasa.gov

Appendix C.—Rendered Design Drawings

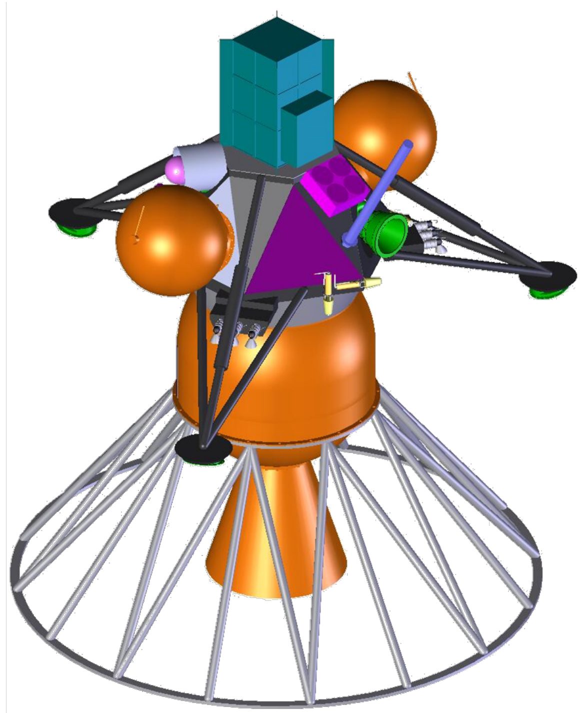

Figure C.1.—Case 2—ASRG powered soft ILN Lander on ELV adaptor.

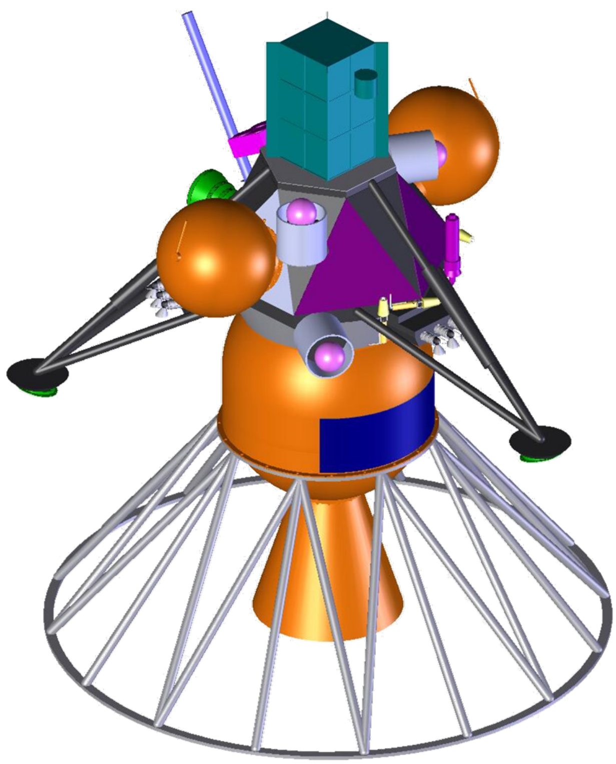

Figure C.2.—Case 2—ASRG powered soft ILN Lander on ELV adaptor.

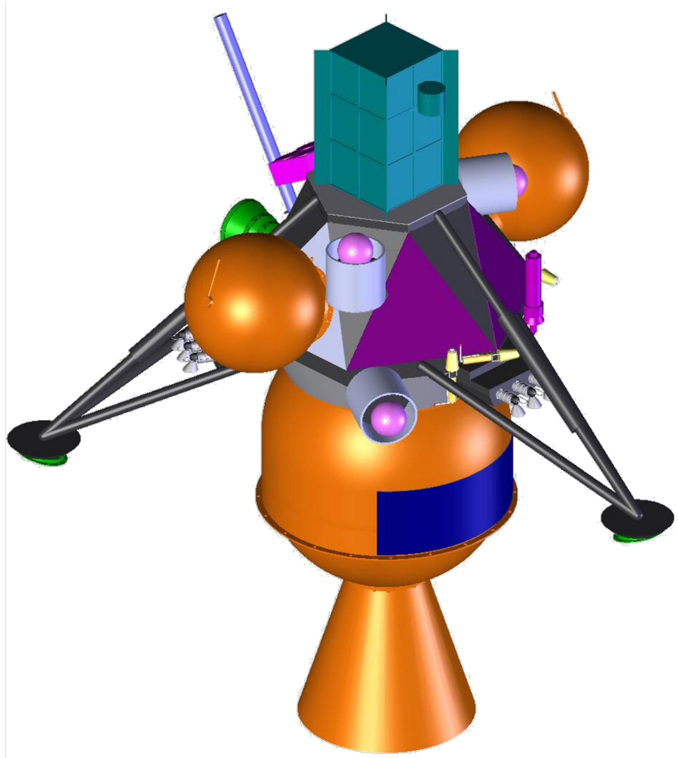

Figure C.3.—Case 2—ASRG powered soft ILN Lander on Star Motor—solar array view.

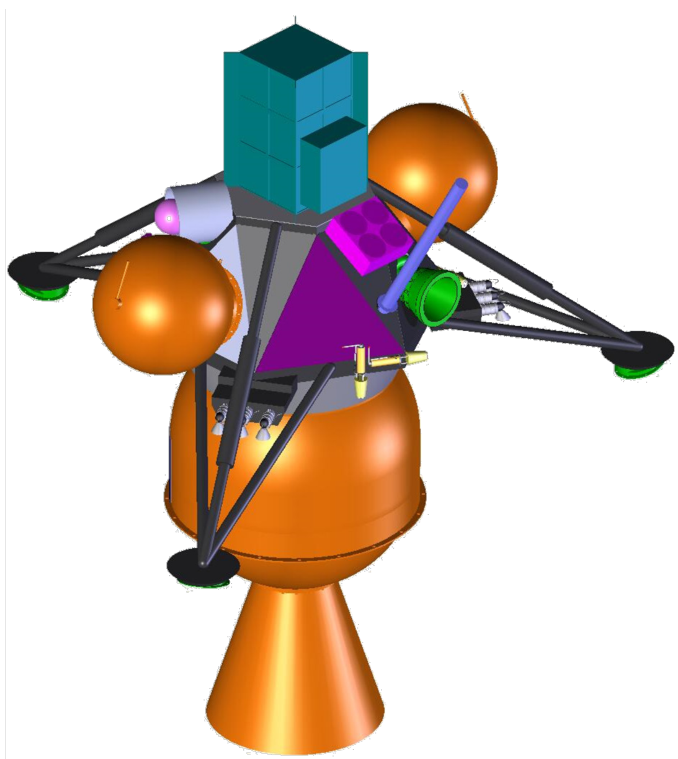

Figure C.4.—Case 2—ASRG powered soft ILN Lander on Star Motor.

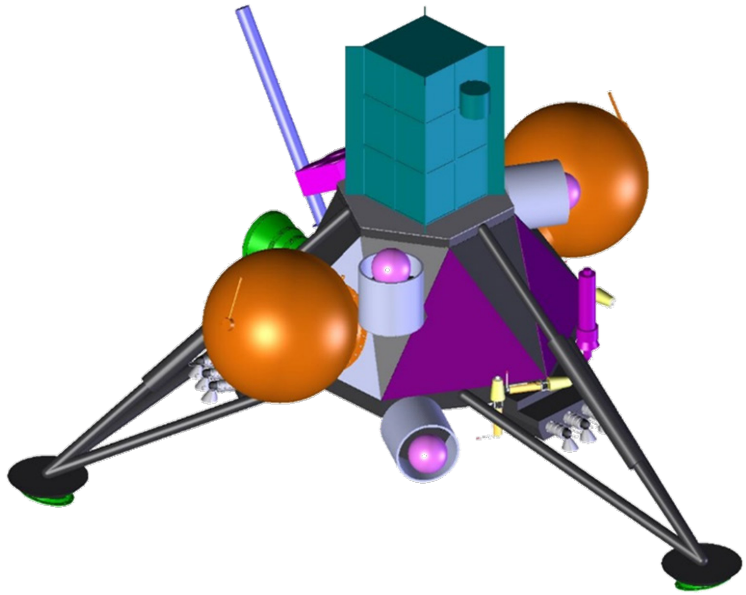

Figure C.5.—Case 2—ASRG powered soft ILN Lander—view 1.

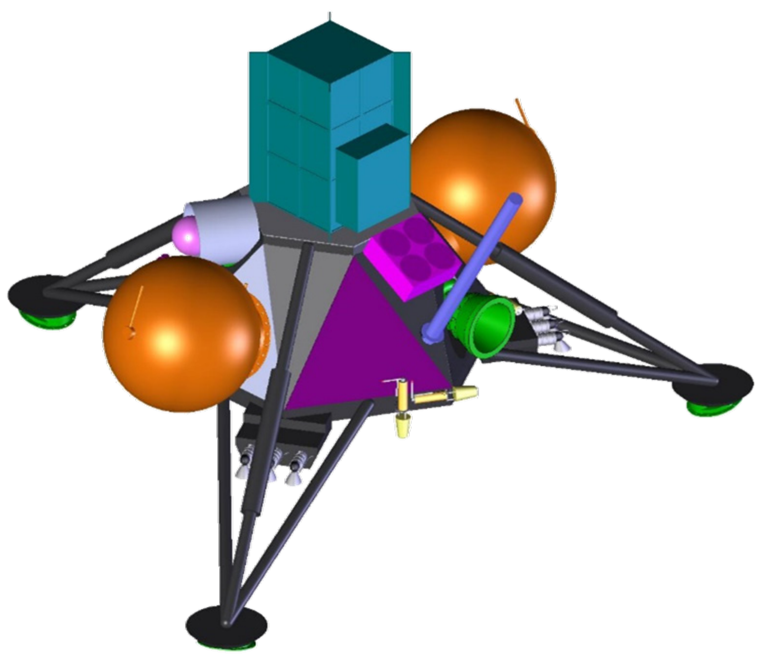

Figure C.6.—Case 2—ASRG powered soft ILN Lander—view 2.

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