NASA/TM—2012-217139 CD–2008–29

COMPASS Final Report: Lunar Network Satellite-High Rate (LNS-HR)

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

July 2012

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1.0 Executive Summary

Two design options were explored to address the requirement to provide lunar piloted missions with continuous communications for outpost and sortie missions. Two unique orbits were assessed, along with the appropriate spacecraft (S/C) to address these requirements (Figure 1). Both constellations (with only two S/C each) provide full time coverage (24 hr/7 d) for a south polar base and also provide continuous 7 day coverage for sorties for specified sites and periodic windows (for a top level comparison see Table 1). Thus a two-satellite system can provide full coverage for sorties for selected windows of opportunity without reconfiguring the constellation.

1.1 Frozen Orbit Case

The first design was a Lunar Network Satellite (LNS) system using a frozen orbit around the Moon. A frozen orbit has orbit parameters that change very little over time and therefore do not require station-keeping propellant (Figure 2). For the Moon, frozen orbits exist in which inclination and right ascension oscillate between acceptable limits. Extensive work was done using Satellite Orbit Analysis Program (SOAP) to assess: frozen orbit stability, coverage times, and S/C pointing and stabilization. The final design consists of two S/C evenly spaced in a 45° inclined, 24 hr, 0.4 eccentric nearly identical orbits. No major station keeping is needed for the orbits because they are in slightly different altitudes (5 km) to offset secular drift caused by Earth perturbations. The S/C will be Earth facing to eliminate previous designs needs for flipping the S/C to access Earth twice a month. The solar array (SA) axis is parallel to the Moon's axis. The two, 1 m lunar dishes will swing on an arm once an orbit to provide voice and high data rate communications for future crewed lunar outposts and sortie sites. The system would also be capable of servicing various science users. An optical trunk communications link back to Earth has also been made. The baseline design will launch two LNS S/C together on an Atlas 511 launcher and provide communications and navigation for 12 yr.

1.2 Lagrange Orbit Case

The second design utilized Earth/Moon L_1 and L_2 Lagrange orbits. A large selection of Lagrange orbits exist that could host lunar communications S/C. A 'figure eight' halo orbit at L_1 and one at L_2 were chosen for the design but further optimization may find more optimal orbits. The selected 16 day orbits would require around 60 m/s station keeping ΔV. Due to the requirement of providing communications to a 1 W astronaut suit radio combined with the distance of the S/C to the Moon, two 7-m dishes are required on the S/C. These will necessarily be deployable dishes similar to the early Tracking and Data Relay Satellite System (TDRSS) S/C. While an approach to launch both on an Atlas 521 launch vehicle would reduce launch costs, the cost for the two S/C network is 25 percent higher than the frozen orbit design due to the need for substantial station keeping propellant and large, deployable antennas. This report attempts to compare the two designs, along with an optical communications Earth link option.

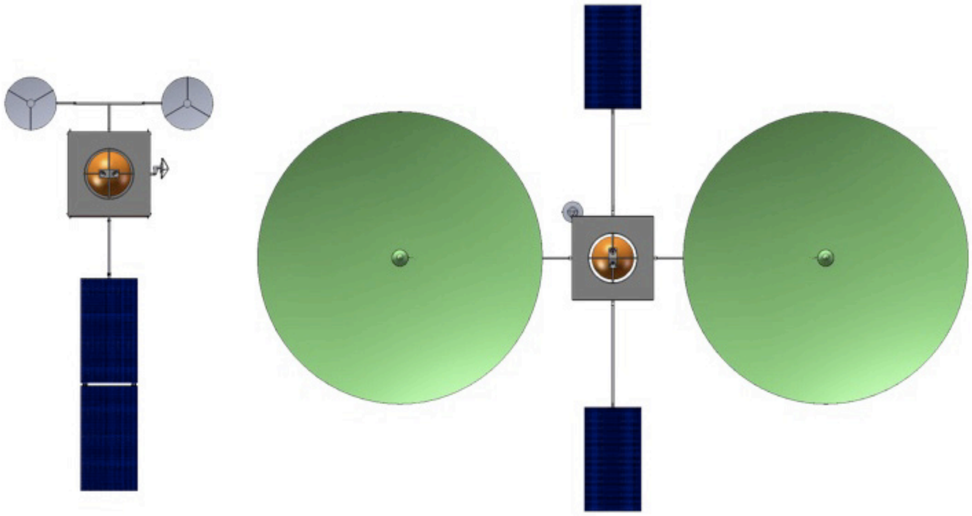


Figure 1.—LNS-HR frozen orbit (left) and Lagrange orbit (right) designs.

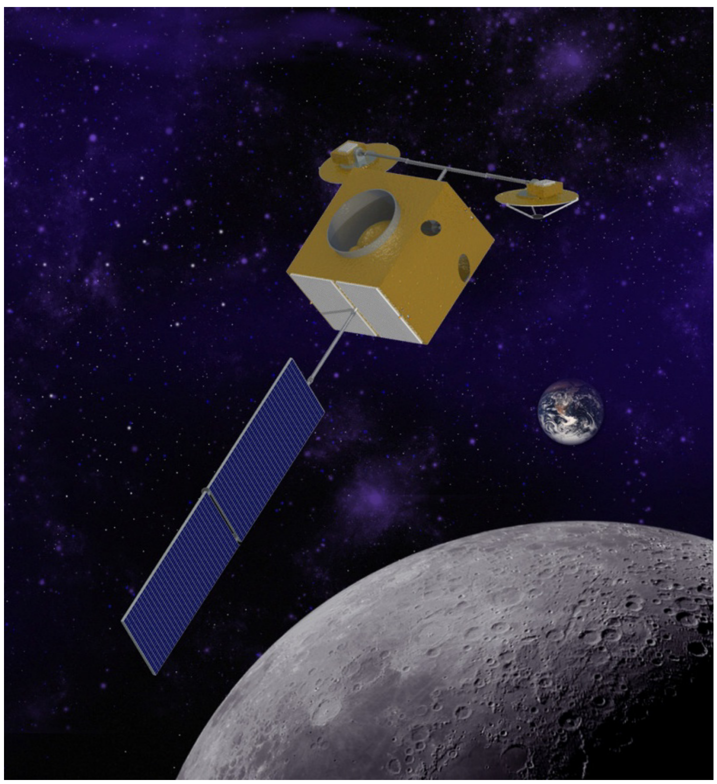


Figure 2.—An artist's rendition of the frozen orbit case.

TABLE 1.—LNS-HR FROZEN ORBIT VERSUS
LAGRANGE ORBIT DESIGN TOP LEVEL COMPARISON

Orbit type	Mass (kg)	Dimensions	Life cycle cost ($M)
Frozen Orbit (RF technology)	1490	5- by 10-m	899
Lagrange Orbit (RF technology)	2280	17.5- by 13-m	1,073

Table 2 collects the details of the subsystems at a top level in the frozen orbit Lunar Network Satellite (LNS) design using microwave technology for the communications links to the Moon and back to Earth.

This is also referred to as radio frequency (RF) communications throughout this report. All data are for a single satellite. Two satellites will be used in the constellation.

Table 3 collects the details of the subsystems at a top level in the Lagrange orbit LNS design. Note that some of the subsystems are the same as the design for the Frozen orbit RF case since they are unaffected by the choice of orbit. All data are for a single satellite. Two satellites will be used in the constellation.

TABLE 2.—MISSION AND S/C SUMMARY—FROZEN ORBIT RF SINGLE SATELLITE

Subsystem area	Details	Total mass with growth
Top level system	S/C to relay lunar communications and provide navigation support	1489 kg with growth per each satellite
Mission, Operations, Guidance, Navigation and Control (GN&C)	Two S/C in 24 hr, 45° inclined frozen elliptical lunar orbit: Minimal station-keeping 12 yr life, each LNS single fault tolerant	
Attitude Control System (ACS)	14 N-m-s momentum storage in four reactions wheels 12 Sun sensors, two star trackers, two inertial measurement units (IMUs)	51 kg
Launch	Atlas 511: >25% launch margin Dual launch capability	3915 kg performance to $C_3 = -2$ km^2/s^2
Power	1100 W average power load (including 30% margin) One single axis SA, 28% efficient triple-junction cells, 5.9 m^2 area, 1600 W beginning of life (BOL) Li-ion batteries sized for 2.5 hr lunar eclipse at full power, reduced ops for infrequent 7 hr Earth eclipses: 4.3 kW-hr	108 kg
Propulsion	Pressure fed hydrazine, two 100 lbf and sixteen 1 lbf thrusters	61 kg
Structures and mechanisms	Al-Li panel around a central thrust-tube, thrust-tube designed to carry second S/C above for launch	288 kg
Communications	Provides continuous voice communications to all users including suit radios 400 Mbps data from users, 50 Mbps from other lunar surface user using space-based router Support two-way ranging to up to five users simultaneously	138 kg
Command and Data Handling (C&DH)	Systems command, control, health management 24 hr communications storage array (0.3 TB)	97 kg
Thermal	Heat pipe—radiator system, heaters and multilayer insulation (MLI)	70 kg

TABLE 3.—MISSION AND S/C SUMMARY—LAGRANGE ORBIT CASE RF SINGLE SATELLITE DETAILS

Subsystem area	Details	Total mass with growth
Top level system	S/C to relay lunar communications and provide navigation support	2281 kg with growth per each satellite
Mission, Operations, GN&C		
ACS		94 kg
Launch	Atlas 521: >25% launch margin Dual launch capability	4710 kg performance to $C_3 = -2$ km^2/s^2
Power		125 kg
Propulsion		132 kg
Structures and mechanisms	Al-Li panel around a central thrust-tube, thrust-tube designed to carry second S/C above for launch	363 kg
Communications	Provides continuous voice communications to all users including suit radios 400 Mbps data from users, 50 Mbps from other lunar surface user using Space Based router Support two-way ranging to up to five users simultaneously	248 kg
C&DH	Systems command, control, health management 24 hr communications storage array (0.3 TB)	97 kg
Thermal	Heat pipe—radiator system, heaters and MLI	70 kg

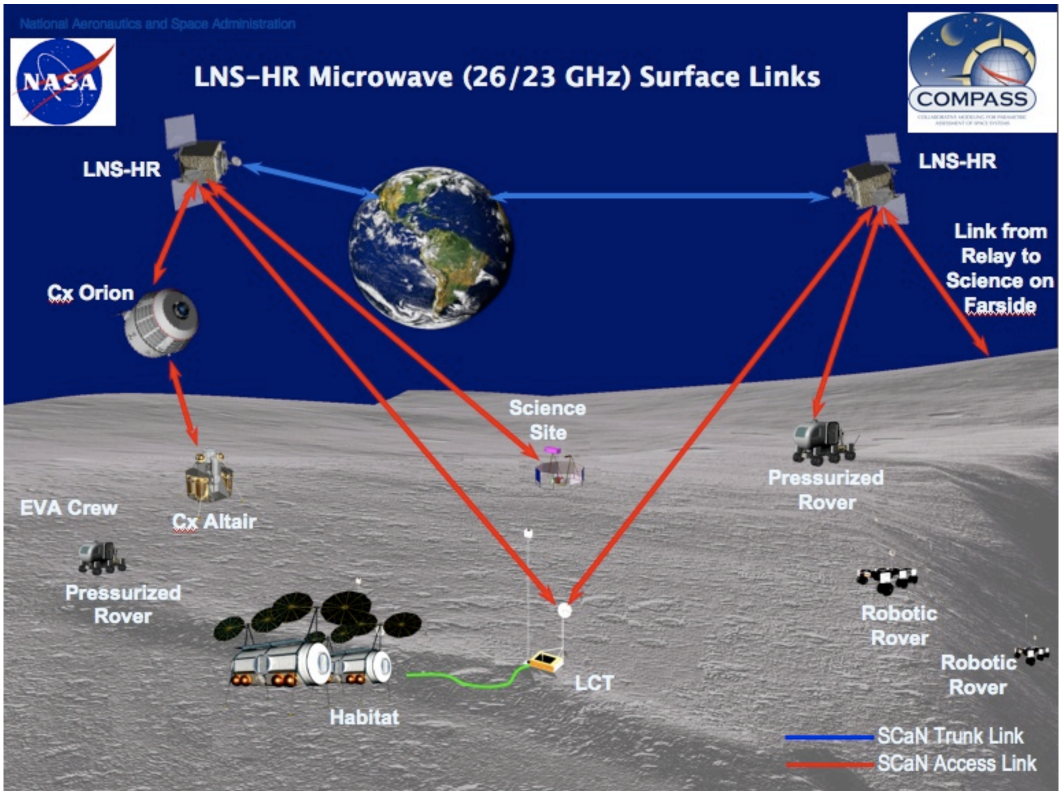


Figure 3.—Lunar architecture surface communications links.

2.0 Study Background and Assumptions

2.1 Introduction

COMPASS was engaged by the NASA Space Communications and Navigation (SCaN) office to provide a LNS concept designs as a follow-on for the Lunar Relay Satellite (LRS) design completed May 2007. The COMPASS team began the design of a LNS and constellation to provide voice and high data rate communications for future crewed lunar outposts and sortie sites (Figure 3). The system would also be capable of servicing various science users. Frozen lunar orbits and Lagrange orbits were traded in order to minimize propulsion requirements. A trade on microwave versus optical trunk communications back to Earth was also made.

A NASA Request for Information (RFI) was recently put out to the space communications community to solicit studies on the future technologies and ideas to accomplish the communications goals of NASA.

http://www.nasa.gov/home/hqnews/2008/aug/HQ_08207_RFI_Lunar_Comm.html

MEDIA ADVISORY: 08-207

NASA Seeks Input For Commercial Lunar Communications & Navigation
WASHINGTON -- NASA issued a Request for Information, or RFI, on Monday to gauge interest and solicit ideas from private companies in providing communications and navigation services that would support the development of exploration, scientific and commercial capabilities on the Moon over the next 25 yr.

Responses should be submitted to Barbara Adde, NASA Headquarters, Mail Suite 7L70, 300 E. St., SW, Washington, D.C. 20546-0001, by 4 p.m. EDT on Sept. 15, 2008.

To view the Request for Information, visit: http://www.spacecomm.nasa.gov

2.2 Assumptions

2.2.1 Coverage Assumptions/Requirements

NASA's Level 0 Requirement to the SCaN Program for communications and navigation coverage of the lunar surface is as follows: SCaN shall provide anytime/anywhere communication and navigation services as needed for lunar and Mars human missions. To satisfy this requirement with a two-satellite constellation of LNS, the system design will strive to provide continuous S-band coverage for a selected list of potential outpost and sortie sites. The expectation is to support two sites simultaneously—including the outpost continuously and another sortie site for 7 to 14 days. The selected list of sortie and outpost sites is described in Figure 4 and Table 4. Additionally, the LNS system will strive to provide periodic (8 hr/12 hr) high data rate coverage for the selected list of potential outpost and sortie sites. Table 5 lists the study assumptions and requirements and identified potential technology trade space. The third column is the areas of technology trades that could be done for each of the subsystems.

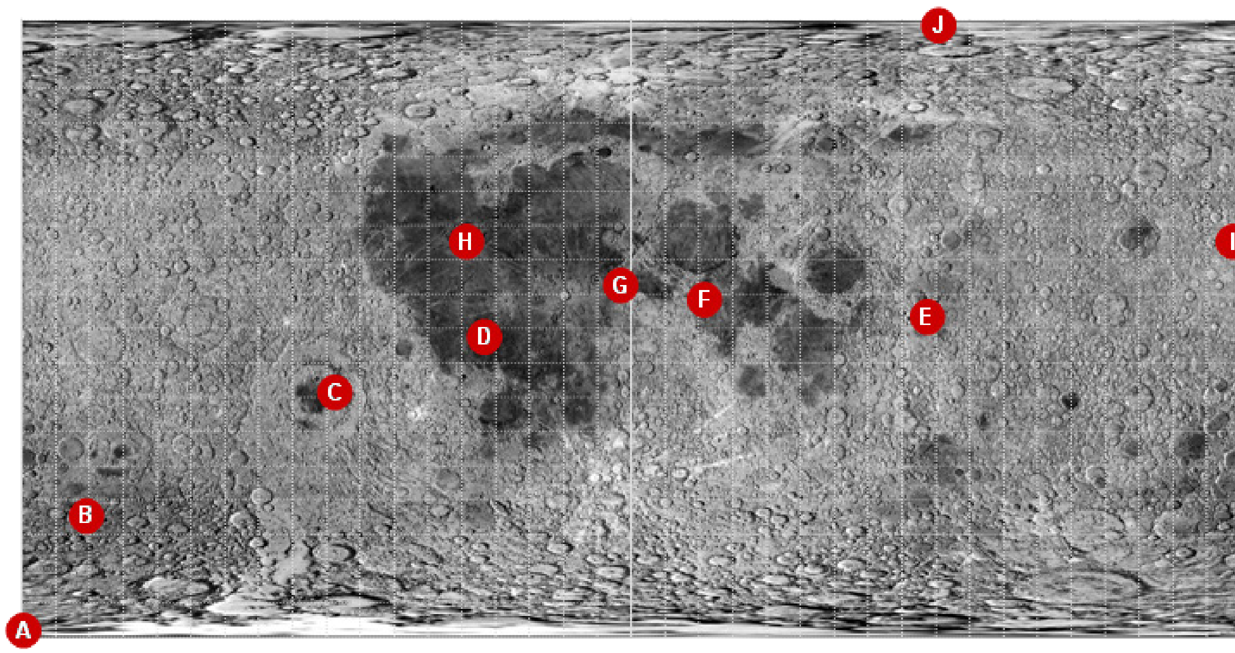


Figure 4.—Lunar Architecture SCaN Program Proposed Landing Sites

TABLE 4.—SELECTED LIST OF POTENTIAL LUNAR OUTPOST AND SORTIE SITES

Landing Site		Latitude	Longitude	Notes
A	South Pole	89.9 S	180.0 W	Rim of Shackleton
B	Far side SPA floor	54.0 S	162.0 W	Near Bose
C	Orientale basin floor	19.0 S	88.0 W	Near Kopff
D	Oceanus Procellarum	3.0 S	43.0 W	Inside Flamsteed P
E	Mare Smythii	2.5 S	86.5 W	Near Peek
F	W/NW Tranquilitatis	8.0 N	21.0 E	North of Arago
G	Rima Bode	13.0 N	3.9 W	Near Bode vent system
H	Aristarchus plateau	26.0 N	49.0 W	North of Cobra Head
I	Central far side highlands	26.0 N	178.0 E	Near Dante
J	North Pole	89.5 N	91.0 E	Rim of Peary B

TABLE 5.—STUDY ASSUMPTIONS

Subsystem area	Assumptions and study requirements	Critical trades
Top-Level	Two LRS provide comm/nav for piloted lunar missions, 2018 and 2020 launches. 12 yr mission for dual launch, all lunar surface elements (South Pole outpost site focused), *but*, also global coverage as necessary, robotic, rover, hab, in situ resource utilization (ISRU), Solar Power Unit/Makeup Power Unit (SPU/MPU), LCT RF power limit 40 to 80 W (determines antenna size of LRS), LCT 1 m Ka antenna, rover RF power limit 10 W and antenna size 0.3 m Ka band (modulation limited 2 bits/Hz)	12 hr Frozen orbit versus L_1/ L_2 (two S/C at each) for each configuration, S-Band and Ka-Band coverage circles, 7 yr option for separate launch approach
System	Figures of merit (FOMs): Cost, mass, flexibility, antenna sweep speed and excursion (LRS and lunar assets), Beam spot size of LRS Off-the-shelf (OTS) equipment where possible, Push technology to save mass/cost, Technology Readiness Level-6 (TRL-6) cutoff 2014, single fault tolerant , Mass Growth per ANSI/AIAA R-020A-1999 (add growth to make system level 30%)	
Communications	LRS to Earth: Ka band (400 Mbps mono, 800 Mbps dual), S-band (150 kbps), LRS to LCT: S-Band (150kbps), Ka band (400 Mbps,1200 Mbps), No cross-link with other LRS, gimbaled To Earth (110°): LRS to surface links: Ka 23/26 (25 Mbps) and TT&C S band, Processing for comm/nav in payload, storage in C&DH	Concentric ka/S downlink antenna(s), phase-array, trade LRS antennas, omni-S/Ka band antennas, number of Ka band antennas
Mission, Ops, ACS	18 m Earth antennas (3), constant one-way nav signal, two-way nav signal to Altair (descent/ascent)	12 hr frozen versus L_2 orbits disposal (is a ΔV of 50 m/s sufficient)
Launch vehicle (LV)	Atlas 501 or Delta IV equiv. Launch Loads: ~5gs	Dual launch manifest option
Power	1000 W triple junction gallium arsenide (GaAs), battery for (frozen orbit: 6 hr max, 2 hr avg. Earth eclipse ~once a year for L_2), 1 hr lunar eclipse (frequency), 30% power margin	Trade eclipse outage coverage
C&DH (including software)	Store of data (0.3 TB, single fault tolerant), storage, command and telemetry (C&T), nav, avionics processor boxes, 54 krad parts, –20 to 50 °C, 100 W	Redundant boxes, amount of processing/software between communications and C&DH packages
Thermal and Environment	Lowest temperature, (frozen orbit 6 hr Earth eclipse) , louvers, ~0.5 m radiator with heat pipes, 54 krad behind 100 mil Al	–20 to 50° C
Mechanical	Q Earth pt. antenna (110° gimbal(s), booms), K-, S-band LCT pointing antenna (dual axis, 90° from zenith gimbal where necessary), space frame (Al Li), thermal/micrometeoroid and orbital debris (MMOD)/dust cover. Antenna mount/deployment systems or fixed (launch loads) Secondary: 4% of stage components	Redundant gimbals, box location, configuration, antenna/boom locations, interface with Lunar Lander.

2.2.2 Communications Assumptions/Requirements

- No crosslinks to other LNS-HR

2.2.2.1 LNS-HR Microwave Trunk Link

- Ka-band spectrum: 37/40 GHz; bandwidth up to 500 MHz
- S-band spectrum: 2.2/2.0 GHz
- 400 Mbps from LNS-HR to Earth at Ka-band
- 100 Mbps from Earth to LNS-HR at Ka-band
- 150 kbps from LNS-HR to Earth at S-band
- Bit error rate: 10^{-8}
- One dual Ka-band/S-band trunk link antenna
- Radiometric tracking of LNS-HR from Earth at Ka-band (37/40 GHz)
- Command, control, telemetry, contingency voice, and safing on S-band
- Modulation limited to 2 bits/Hz

2.2.2.2 LNS-HR Optical Trunk Link

- 1.2 Gbps from LNS-HR to Earth
- 25 Mbps from Earth to LNS-HR

- Packet loss rate: 10^{-9}
- Optometric tracking of LNS-HR from Earth
- S-band spectrum: 2.2/2.0 GHz
- Command, control, telemetry, contingency voice, and safing on S-band
- S/C attitude stability: 0.3°
- S/C attitude knowledge: 1/10 of stability metric

2.2.2.3 LNS-HR Microwave Access Links

- Ka-band spectrum: 26/23 GHz
- S-band spectrum: 2.2/2.0 GHz
- Modulation limited to 2 bits/Hz
- 400 Mbps maximum from LNS-HR to lunar surface at Ka-band
- 150 Kbps maximum from LNS-HR to lunar surface at S-band
- Two Dual Ka-band/S-band access link antennas
- Navigation, TT&C, contingency voice, and safing on S-band link

2.2.2.4 Lunar Surface Assets

- Lunar Communications Terminal (LCT) antenna size: 1 m
- Rover antenna size: 0.3 m
- Extra Vehicular Activity (EVA) antenna: dipole
- EVA maximum transmit power: 1 W

2.2.3 Earth Based Telescopes for Laser Optical Trunk

- Station locations: Madrid, Canberra, Goldstone, three additional geographically separated sites
  - Antenna sizes: 18 m
- Aperture sizes: ~2 m (in order to preserve the same S/C pointing capability as RF)
- Transmit power: 5 W (TBR)
- Photon counting receivers: (TBD)
- State-of-the art (SOA) differential phase-shift keying-return-to-zero (DPSK-RZ) modem with 132 ms forward error correction (FEC)/interleaver
  - Most likely Pulse Position Modulation (PPM) instead of DPSK-RZ
- Tracking: 5 μrad coarse tracking, < 1 μrad fine tracking TBR
- Laser safety coordination: FAA and Laser Clearing House
- Near Sun pointing: 10° (TBR)
- Atmospheric monitoring: 15 dB fading (TBR)
- Data handling 2.5 Gbps (TBR); Boeing also has 10 and 40 Gbps archiving, and distribution due to high data rates
  - 10 to 20 Mbps up to the Moon

2.2.4 Derived Requirements

The following requirements are derived from the Level 0 constellation mission requirements. The wording choice of “***shall***” in the statements indicates that the statement is a requirement.

- The LNS System ***shall*** provide continuous voice coverage for the selected list of potential outpost and sortie sites.
  - Expected to support two sites continuously and simultaneously for 7 days
    - Outpost and another sortie site
  - Satellites may be moved but must complete move in < 3 months

    - It is assumed that no LNS-HR voice coverage is required for sites in view of the Earth
- The LRS system ***shall*** provide periodic (8 hr/12 hr) high data rate (400 Mbps/500 MHz BW) coverage for the list of potential outpost and sortie sites.
    - 400 Mbps for all RF system
    - 1.2 Gbps for optical (space trunk) system
- The LRS system ***shall*** provide communications to/from astronauts on EVA during contingency operations
    - 1 W max RF power from suit
    - Equivalent isotropically radiated power (EIRP) of –0.6 dB-W

2.3 Similar Past Missions

The LNS needs to be able to point to three (and perhaps four places) places simultaneously: the Sun (SA), the Earth (0.5 m trunk antenna), and down to the surface for lunar sites (perhaps up to two). The COMPASS design of a LRS performed in the spring of 2007 and documented in report CD–2007–12, designed a lunar relay satellite in support of human lunar base requirements. The requirements of this study have been further expanded since that study. The LRS design was used as one of the starting points for this study. In addition, a survey was done of current and recent examples of S/C that point to three things simultaneously. These examples also used as references in this study include:

- TDRSS (Figure 5)
    - Geosynchronous Earth orbit (GEO), points to Sun, Earth and objects to sides (other TDRSS)
- Hubble
    - Points to stars for long periods, SAs and two communications antennas
- Lunar Reconnaissance Orbiter (LRO) (Figure 6)
    - Polar orbit
    - Data return when on near side of Earth only
    - Earth antenna on long boom, dual gimbaled
    - Two axis gimbaled SA

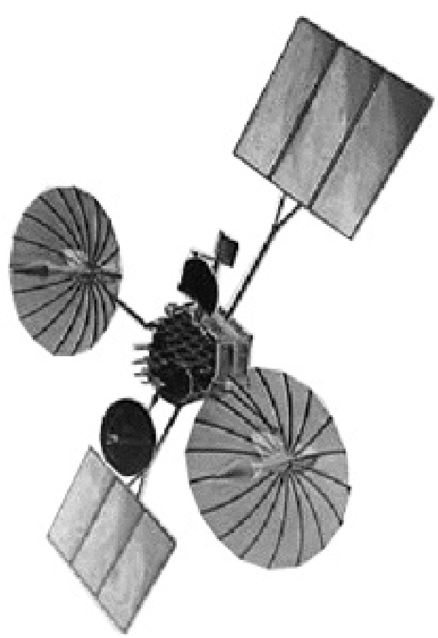


Figure 5.—TDRSS Satellite.

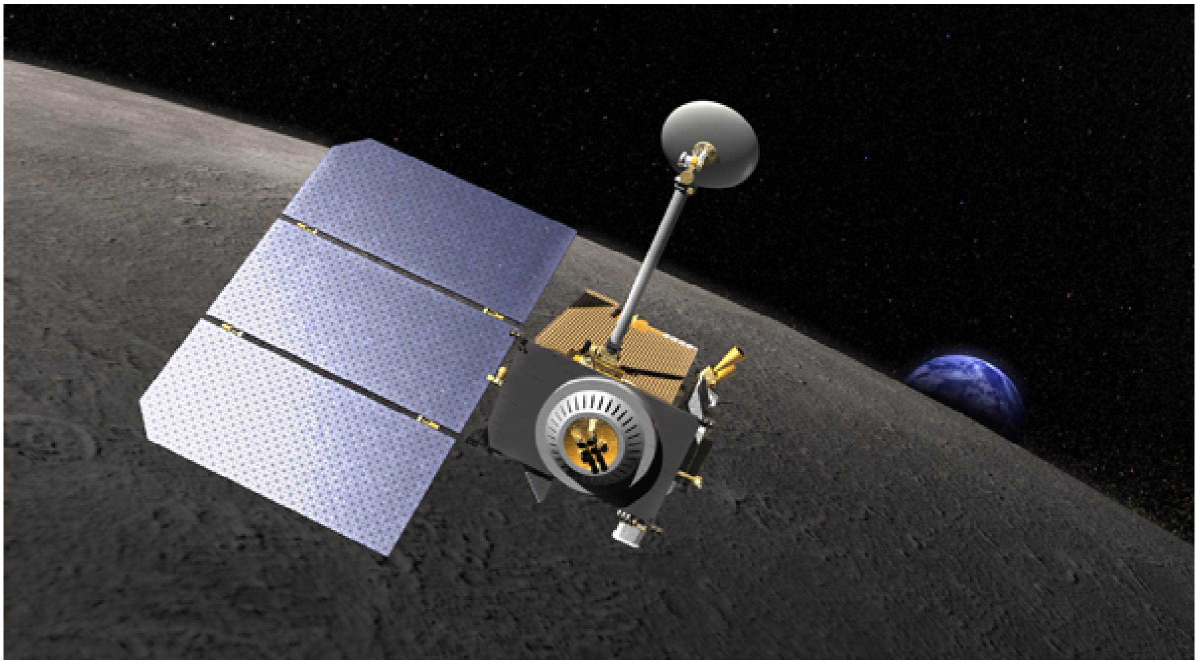


Figure 6.—An artists' rendition of the LRO mission.

2.4 Margin and Growth Policy

In order to clear up the margin and contingency (growth) policy of the MEL modeling, it helpful to define what is meant by the terms margin and contingency in the workings of the COMPASS design session.

2.4.1 Definition of Terms

For the purpose of the COMPASS designs, margin is the mass that is not flown (i.e., ΔV does is not applied to this mass in the rocket equation) to the final destination. This is most often referred to as LV margin. See Section 2.7 for a definition and further description of LV margin as applied to COMPASS designs. The margin percentage value is a per mission calculation. In some missions, the launch vehicle chosen to fly the mission will require more margin as a percentage of published launch vehicle performance than in others based on the propulsion system and mission risk(s).

For COMPASS, margin is not the same as Contingency or growth mass. Contingency or growth mass is a factor applied to the best estimate mass of a component in the S/C design, in order to bridge the gap between the current technology readiness level of that component and the final flight technology maturity mass of that component. This contingency is applied to the best guess mass to give a total mass calculation. It is this total mass that is flown to the mission's final destination and used with the rocket equation and mission ΔVs to calculate propellant mass. Each subsystem applies a growth percentage to each line item in their subsystem MEL. The system integration sheet rolls up these growth masses and calculate the overall S/C dry mass system contingency percentage (growth mass/total dry mass).

2.4.2 Definition of Mass Growth Allowance (MGA)

The COMPASS team uses the AIAA S–120–2006, *Standard Mass Properties Control for Space Systems* (Ref. 1). Table 6 shows the percent mass growth separated into a matrix specified by level of design maturity and specific subsystem. MGA is defined as the predicted change to the basic mass of an item based on an assessment of the design maturity and fabrication status of the item, and an estimate of the in-scope design changes that may still occur.

The schedule shown in Table 6 was used by the team as of September 2008. For most studies, the Major Category for all parts will be the E1 row, unless a component is significantly farther along in the technology development process.

The percent growth factors are applied to each subsystem, after which the total system growth of the design is calculated. The COMPASS design team designs to a total growth of 30 percent or less. An additional growth is carried at the system level in order to add up to a total system growth of a maximal 30 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.

Because the analysis performed by the COMPASS team has traditionally been in the pre-phase A study phase, things like customer reserve as illustrated in Figure 7, mass definition illustration, taken from the 2006 AIAA report and shown below are not taken into consideration in the design. A phase A design would begin by allocating customer reserve masses on top of the design developed in the pre-phase A study session. The purpose of a pre-phase A study is to come up with a design that will give the customer the *Mass requirement* or allowable mass as shown in Figure 7.

TABLE 6.—PERCENT MGA

Major category	Maturity code	Design maturity (Basis for mass determination)	Percent MGA											
			Electrical/electronic components			Structure	Brackets, clips, hardware	Battery	Solar array	Thermal control	Mechanisms	Propulsion	Wire harness	Instrumentation	ECSLSS, crew systems
			0-5 kg	5-15 kg	>15 kg										
E	1	Estimated (1) An approximation based on rough sketches, parametric analysis, or undefined requirements (2) A guess based on experience (3) A value with an unknown basis or pedigree	30	25	20	25	30	25	30	25	25	25	55	55	23
	2	Layout (1) A calculation or approximation based on conceptual designs (equivalent to layout drawings) (2) Major modifications to existing hardware	25	20	15	15	20	15	20	20	15	15	30	30	15
C	3	Preliminary design (1) Calculations based on a new design after initial sizing but prior to final structural or thermal analysis (2) Minor modifications of existing hardware	20	15	10	10	15	10	10	15	10	10	25	25	10
	4	Released design (1) Calculations based on a design after final signoff and release for procurement or production (2) Very minor modification of existing hardware (3) Catalog value	10	5	5	5	6	5	5	5	5	5	10	10	6
A	5	Existing hardware (1) Actual mass from another program, assuming that hardware will satisfy the requirements of the current program with no changes (2) Values based on measured masses of qualification hardware	3	3	3	3	3	3	3	2	3	3	5	5	4
	6	Actual mass Measured hardware	No mass growth allowance—use appropriate measurement uncertainty values												
	7	Customer furnished equipment or specification value	Typically a “not-to-exceed” value is provided; however, contractor has the option to include MGA if justified												

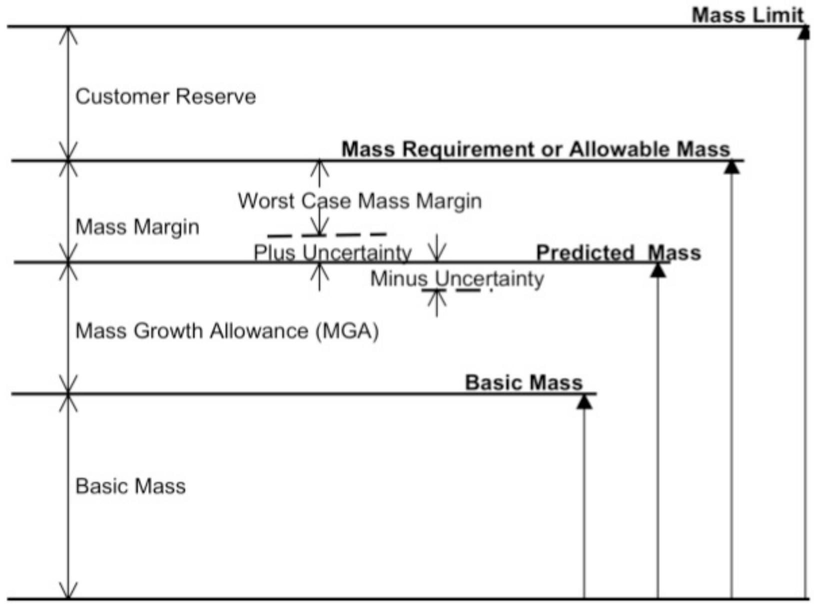


Figure 7.—Mass definition

Whether the total mass of the system designed by COMPASS becomes the requirement or the allowable mass depends on the requirements and goals of the customer in the outcome of the study. Some COMPASS studies provide the customer with the ceiling of what a certain launch vehicle would be able to send to a preferred destination. Other studies, with different goals, will give the customer various options of using different launch vehicles and provide the size and capabilities of the missions that can be performed if limitations are put on the launch vehicle, propulsion system, and/or reductions are made. The scope of allowable technology options in anything from launcher to propulsion system to power system will impact the total mass of the system achieved at the end of the design study.

2.4.3 From the AIAA Standard—Terms and Definitions as applied to Dry Mass.

The following definitions apply only to hardware and are taken from the AIAA 2006 document (Ref. 2). Appendix C also shows a series of MGA tables used in various NASA programs.

Allowable mass — *The limits against which margins are calculated after accounting for basic masses of flight hardware, MGA, and uncertainties*

NOTE*: Derived from the requirements early in the program, the allowable mass is intended to remain constant for the duration of the program.*

Basic mass — *The current mass data based on an assessment of the most recent baseline design.*

NOTE 1*: This design assessment includes the estimated, calculated, or measured (or actual) mass, and includes an estimate for undefined design details like cables, multi-layer insulation and adhesives.*

NOTE 2*: The MGA and uncertainties are not included in the basic mass.*

COMPASS has referred to this as current best estimate (CBE) in past mission design.

Mass — *The measure of the quantity of matter in a body*

MGA — *The predicted change to the basic mass of an item based on an assessment of the design maturity and fabrication status of the item, and an estimate of the in-scope design changes that may still occur*

Predicted mass	*The basic mass plus the mass growth allowance* *This is also referred to as the total mass by the COMPASS team.*
Mass margin	*The difference between the space system allowable mass and predicted mass* *For the COMPASS process, the total percentage on dry mass* *Mass Margin + Predicted Mass = CBE + 30 percent*CBE.* *Therefore, mass margin will be carried as a percentage of dry mass rather than as a hard-targeted number based on launch vehicle performance, etc. Because these studies are pre-Phase A and conceptual, unless the customer has other requirements for this total growth on the system, the 30 percent number is standard COMPASS operating procedure.*
Power growth	*The COMPASS team uses a 30 percent margin on the bottoms up power requirements in modeling the power system. See Sections 3.1.2, 3.2.2, 3.3.2, 3.4.2 for the power system assumptions.*

2.4.4 Launch Margin

Margin in terms of a LV from the S/C design perspective is that part of the LV performance that cannot be transferred to the S/C. This is uncertainty in the performance calculations. It is taken off the top of the launch vehicle's performance to the target orbit or C_3 (energy). The LV to S/C adaptor mass (i.e., a part left with the launch vehicle after separation) is typically taken out of this margin for the unusable LV performance capability. A conservative margin of 10 percent has been the usual rule of thumb. It not flown on the S/C, and not used in the inert mass calculations applied to trajectory parameters when sizing propellant requirements for the mission (but the adapter is part the full up LV mass of the pad and effects fuel, and performance). For this level of analysis, the COMPASS team uses a 10 percent factor on the quoted launch performance as margin.

$$\text{Launch Margin} = 10\% * \text{Specified LV Performance}$$

The launch margin is treated two ways in a design. It is either an established parameter that cannot be traded, or it is used as a parameter to establish the closure or goodness of a design. If a S/C design allows for better than 10 percent launch margin, it is considered to be a robust design. If it allows for less than 10 percent margin, then more work must be done, and the design is not closed.

2.4.5 Power Growth

The COMPASS team uses a 30 percent margin on the bottoms up power requirements in modeling the power system. See Section 5.6 for the power system assumptions and design.

2.5 Mission Description—Frozen Orbits

Any initial orbit around a central body will begin to change primarily due to the nonuniform gravity field of the central body and 3rd body effects. Some characteristics/orbit elements of the initial orbit can be fixed, i.e. frozen, or controlled by specifying values and/or rates of change for the other orbit elements. The goal, in theory, is to place two S/C into the same orbit, but located 180° apart. In order to maintain even coverage of the south-pole, the idea is to freeze the eccentricity and semi-major axis (SMA). For operation, it was discovered that a slight difference in SMA between the two orbits would keep the satellites a constant 180° apart in their orbits.

Several authors note the existence of a family of "Frozen" orbits around the Moon for a specific combination of orbital elements (see Refs. 10, 11, and 12). This family of orbits includes inclinations above 39.6°, arguments of perigee of 90° or 270°, and eccentricities (e) that correspond to inclinations (i) by the equation

$$e^2 + \frac{5}{3}\cos^2(i) = 1$$

For eccentricity of a circular orbit, $e = 0$.

$$e = 0$$

$$\frac{5}{3}\cos^2(i) = 1$$

$$\cos(i) = \sqrt{\frac{3}{5}} \cong 39.23°$$

It is easy to see from this equation that a circular orbit is 'frozen' at 39.6° with eccentricity increasing with inclination. In actuality these orbits do oscillate in eccentricity and inclination but repeat on a cycle of years. Thus, no station-keeping is needed if the extremes of inclination and eccentricity still meet the S/C's coverage requirement.

Despite these circulations such orbits are very advantageous for single polar coverage with the combination of arguments of perigee of 90° or 270° and high ellipticity. Thus an outpost at the South Pole would benefit from a high inclination and an argument of perigee of 90°. For example two S/C in a 12 hr orbit, spaced 180° apart with an inclination of 39° and SMA (9750.3 km and 9755.3 km offset to handle drift perturbations**)** gives good coverage of a South Pole outpost. However, such orbits do not provide good coverage of the northern regions of the Moon.

The requirements for this design study are to provide both continuous South Pole coverage and continuous sortie coverage, especially for the far side sites. Assuming that only one sortie occurs at any one time and that it is of a limited duration (less than 2 wk), a new frozen orbit could be designed which could provide continuous coverage of both the South Pole outpost and a specific far side sortie site for the 2 wk period. This can be achieved by using a higher frozen orbit (24 hr period) at a lower inclination to allow for a higher perigee to cover both the South Pole and the sortie site for at least a 2 wk period. The orbit inclination selected was 45° and an eccentricity of 0.4. The tool SOAP was used to ensure full coverage of each of the desired sortie sites during Sunlit periods (for power needs).

In addition to the oscillations up and down of the eccentricity and inclination over time the SMA also changes slightly up and down over time. For a single S/C this is of no issue but for a constellation of two or more, their spacing apart in an orbit will change and they will no longer maintain an equal spacing in argument of perigee. Thus the S/C could drift together and coverage would no longer be continuous. However, raising the SMA of one of the satellites slightly 5 km over the other has been shown to prevent this drifting of SMA from affecting the relative spacing of the two S/C. The orbits are maintainable without any orbit maintenance propellant necessary over the 12-yr lifetime requirement.

2.5.1 Frozen Orbit Mission Analysis Assumptions

While the orbits are 'Frozen' they do oscillate in SMA and eccentricity, but this oscillation is cyclic and repeats every ~ 2 yr. The effects on the orbit are:

- Argument of periapsis varies between 80° and 115°
- Eccentricity (between 0.35 and 0.5) will increase altitude and then decrease it again
- Inclination (between 39° and 55°) will change coverage

2.5.2 Frozen Orbit Mission Analysis Analytic Methods

Propagation of the orbits over time was done by two methods using two different trajectory analysis programs used by the COMPASS team. Orbit was designed using SOAP and checked with Spacecraft N-Body Analysis Program (SNAP).

The first method was via SOAP's low thrust propagation modeling capability. The orbital elements were input into SOAP and a propagation of those elements over time was run out for the expected 12-yr mission.

SOAP is used extensively by COMPASS to visualize orbits and communications link analysis as well as Sun angle for power analysis over the lifetime of a satellite in its orbit. SOAP, shown if Figure 8, is an interactive three-dimensional orbit visualization and analysis program for Windows, Macintosh, and Sun workstations. The software is capable of simultaneously propagating thousands of satellites, ground stations, aircraft, ships and planets using a diverse set of prediction methods SOAP was developed in 1988 internally by the Engineering Applications Department within the Aerospace Corporation. The purpose of SOAP was to provide visualization and analysis for Space missions. Users can also use SOAP's internal predefined variable types to perform analysis on a range of parameters in an orbit or architecture. Users can also import computer aided design (CAD) models of S/C and use visualization to gather further knowledge of pointing requirements for the various components: Antenna, arrays, thrusters.

To corroborate the behavior of the frozen orbits that were being produced by SOAP, a comparison of a representative frozen orbit was performed with SNAP. Developed at the NASA Glenn Research Center (GRC), SNAP is a high fidelity trajectory propagation program that can propagate the trajectory of a S/C about virtually any body in the solar system. The equations of motion can include the effects of central body gravitation with N x N harmonics, other body gravitation with N x N harmonics, solar radiation pressure, atmospheric drag (for Earth orbits) and S/C thrusting. The equations are solved using an eighth-order Runge-Kutta Fehlberg single step method with variable step size control. The input deck for this case is shown in Appendix D.

The representative orbit was chosen to be a 24 hr circular lunar orbit with an initial inclination of 40° and an initial right ascension of the ascending node (RAAN) of 50°. Including the Earth and the Sun as perturbing bodies, the orbits were propagated for 7 yr in both SNAP and SOAP and the time history of the orbital elements between the two programs was compared. As can be seen in Figure 9, while the initial comparison showed a similar trend in the behavior of the inclination, obviously the orbits produced by the programs differed greatly over time.

It has been seen that there are many different definitions of inertial frames, especially when it comes to lunar orbits, and it was decided to compare the definitions used by SNAP and SOAP. Upon investigation, it was discovered that the definition for the lunar inertial frame was indeed different, specifically the direction of the *X*-axis in inertial space. To obtain the direction of the *X* -axis for a Moon centered inertial frame, SNAP used the orbit normal of the Earth crossed with the Moon's North Pole whereas SOAP used a [0,0,1] vector in inertial space crossed with the Moon's North Pole. Since RAAN is measured from this *X*-axis, the initial orbits of the two programs were not actually aligned in inertial space even though the same initial conditions were used. The definition of the lunar inertial frame used by SOAP was implemented in SNAP, and the results of this change can be seen in Figure 10.

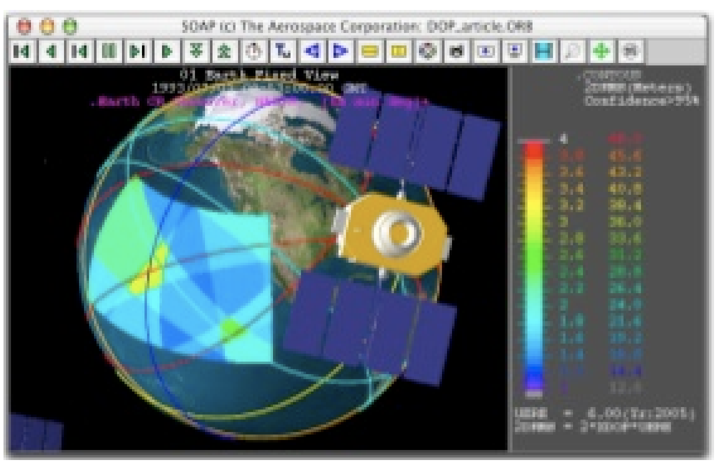


Figure 8.—SOAP graphic.

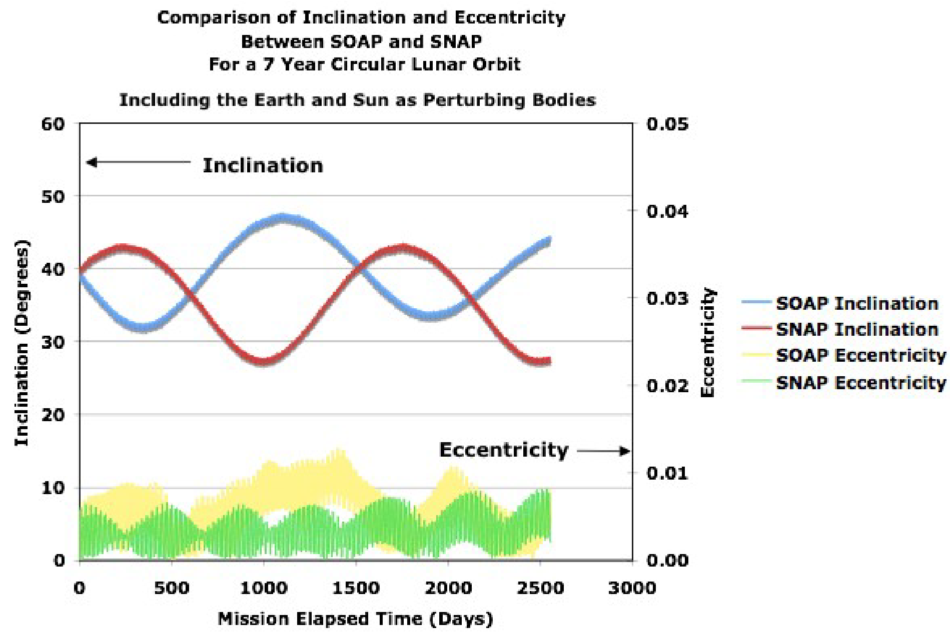


Figure 9.—Initial comparison of a representative frozen orbit between SNAP and SOAP.

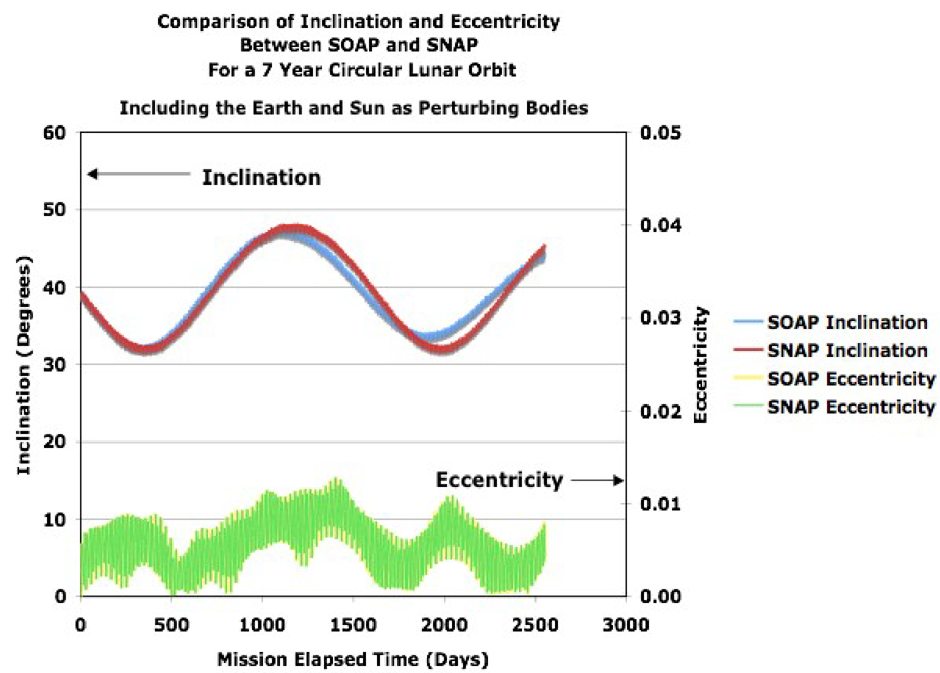


Figure 10.—Final comparison of a representative frozen orbit between SNAP and SOAP.

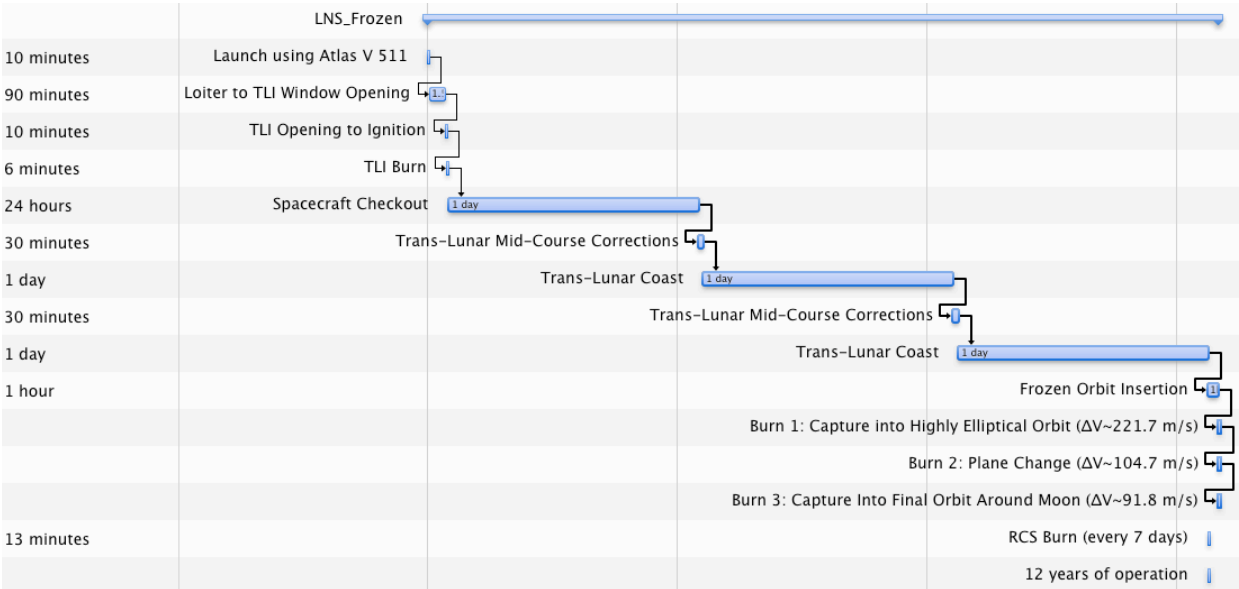


Figure 11.—LNS frozen orbit case CONOPs timeline.

After aligning the lunar inertial frames, the two programs showed very good agreement in the behavior of the representative frozen orbit.

2.5.3 Frozen Orbit Mission Analysis Event Timeline

Most of the ΔV is taken from the NASA Goddard Space Flight Center's (GSFC) I M Design Center (IMDC) study from the summer of 2006 for the sections of the mission.

Table 7 lists the ΔV assumptions used in sizing the propellant budget for performing the propulsion maneuvers throughout the frozen orbit mission. An additional allocation of 50 m/s has been added on for attitude control and station keeping maneuvers while in lunar orbit, and an additional 60 m/s has been added for end of life disposal of the S/C (Figure 11).

2.5.1 Frozen Orbit Delta V Schedule

TABLE 7.—LNS FROZEN ORBIT ΔV SCHEDULE

Phase name	OMS ΔV, m/s	RCS ΔV, m/s	Heritage
Launch from Earth Checkout Loiter to TLI window opening TLI opening to ignition TLI burn			
Correct for ELV Dispersions	85		GSFC IMDC
Trans-Earth mid-course corrections trans-lunar coast	25		GSFC IMDC
Lunar orbit capture burn	410		GSFC IMDC
Lunar orbit adjustments	250		GSFC IMDC
Phasing maneuvers	50		GSFC IMDC
Lunar telecom orbit	50	50	Added ^{2}v margin for attitude control and station keeping
Disposal	80		Estimate for disposal on lunar surface near Pericynthion
Total ΔV	950	50	

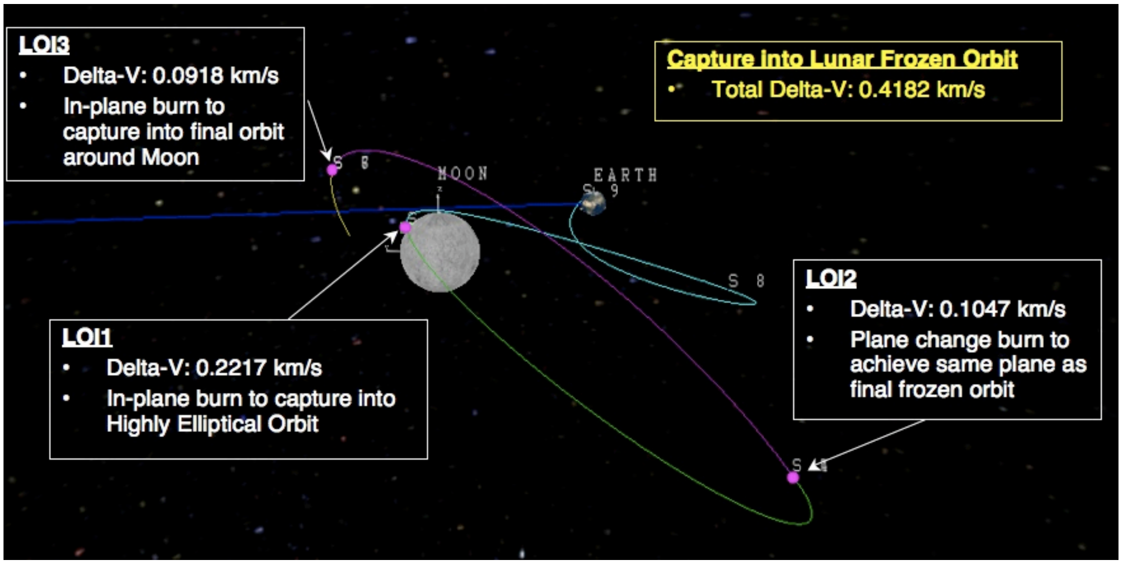


Figure 12.—LNS frozen orbit LOI maneuvers.

2.5.2 Frozen Orbit Mission Trajectory and Orbit Details

The LNS is injected to TLI from the Atlas V launch vehicle. It departs Earth on a coast trajectory in the (same state injected into by the launch vehicle on TLI). It coasts to the Moon and at the Moon three burns are completed to achieve the final frozen orbit. The first burn is lunar orbit insertion 1 (LOI 1), with the highest ΔV of the three burns. LOI 1 is an in-plane burn whose function is to capture the S/C into a highly elliptical orbit about the Moon. LOI 2 performs two functions. It is both a high plane change burn and also raises periapse to result in the desired final inclination. Next, there is another coast back to periapse where LOI 3 is performed (Figure 12). This third burn adjusts apoapse after LOI 2 in order to ensure reaching the final orbital parameters of the desired frozen orbit.

Verification of the LOI ΔV was performed using Copernicus. Copernicus is a high precision system for trajectory design and optimization where trajectories are modeled as a series of segments and was developed at the University of Texas at Austin by Dr. Cesar Ocampo.

2.5.3 Frozen Orbit Details

COMPASS design recommends a 24 hr orbit to provide better visibility the Moon. The 24 hr period orbits provide a simple duty cycle for ground controllers and users.

Using a frozen orbit design with two S/C 12 yr of constant South Pole coverage is possible with no station keeping. The SMA includes the radius of the Moon at 1738 km. The eccentricity of 0.4 yields an orbit that varied in altitude above the surface.

TABLE 8.—LNS FROZEN ORBIT ORBITAL PARAMETERS

Frozen orbit variable	Values
Date	January 1, 2018, 12:00 am
SMA	9750.3 and 9755.3 km (offset to handle drift perturbations)
Eccentricity	0.408248
Inclination	45°
Argument of periapsis	90°
RAAN	50°
Mean anomaly	135° and 315°

Figure 13 shows the two orbits described in Table 8 as illustrated using the trajectory tool, SOAP. The orbit planes are inclined 45° to the lunar equator.

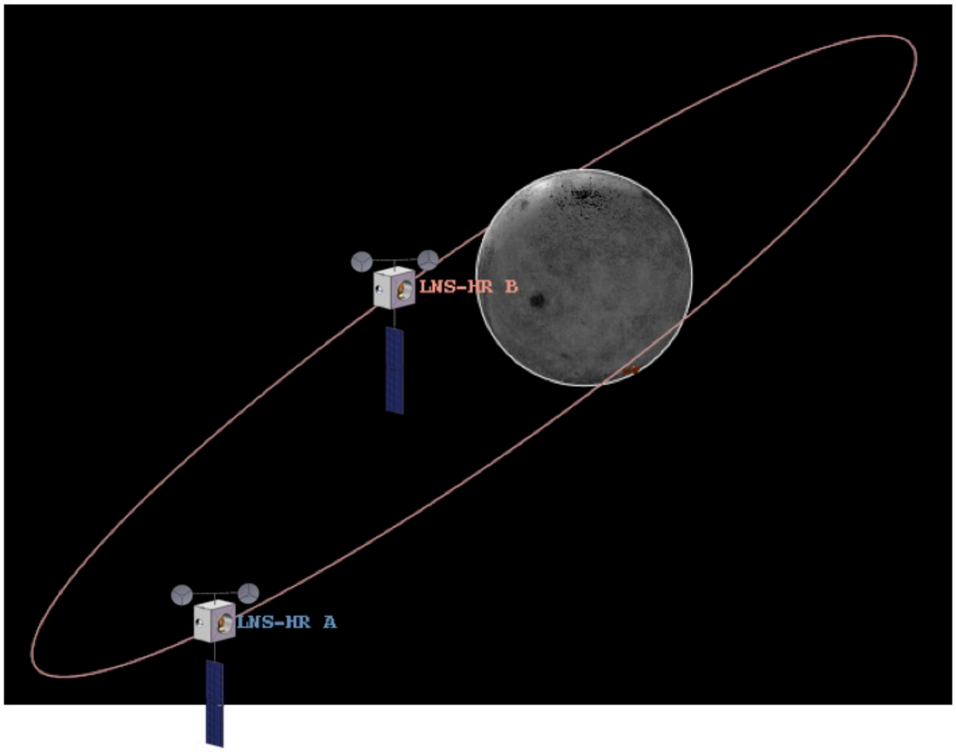


Figure 13.—Frozen orbit two satellite constellation illustration.

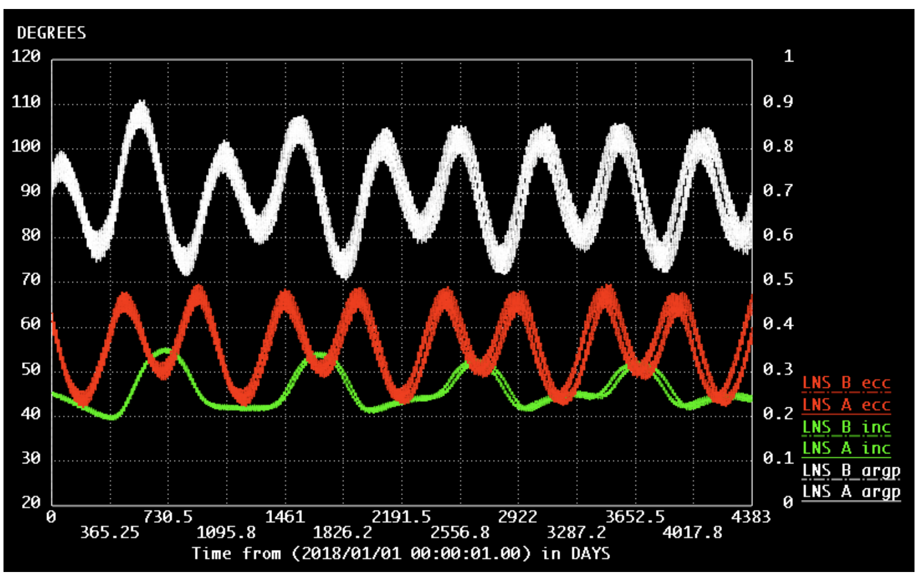


Figure 14.—Frozen orbit orbital element perturbation history.

2.5.4 Selected Design Frozen Orbit

Constellation spaced 180° in mean anomaly to provide constant coverage. Orbits for S/C LNS A and LNS B do oscillate but repeat roughly every 3 yr. ***No*** orbit maintenance is required.

2.5.5 Frozen Orbit Lunar Coverage

Requirements were established that 100 percent Sun visibility and constant coverage by at least one of the LNS-HR satellites was necessary at all times during long duration lunar outpost (South Pole) missions as well as 7 days continuous satellite communication and Sun visibility for sortie missions at the sites listed in Table 4. Coverage analysis for the LNS-HR frozen orbit was performed to verify that the lunar outpost received continuous coverage and that opportunities existed at each of the sortie sites for a 7-day mission.

The coverage analysis over a 4-wk period is presented in Figure 15. Four different Cartesian views of the lunar surface show contours of the percent coverage over a 7-day period. The color gradient bar corresponds to percentage coverage over each 7-day period with deep red representing 100 percent coverage, meaning at least one of the LNS-HR satellites and the Sun is visible during the entire 7 days. The beginning of this particular 4-wk period corresponds to the single opportunity available at which the Central Far Side Highlands site has continuous coverage for a full week, beginning August 30, 2021, 10:00:00. A further discussion of sortie opportunities is presented later in this section. In each view there is a concentrated area that receives full coverage surrounded by areas of less coverage and then an area that is not covered at all during the timeframe. The high percentage areas are located on regions of the lunar surface that is beneath the apocynthion of the frozen orbit. Since the satellites remain in this segment of the orbit for the longest period of time, longer coverage times are possible. The frozen orbit is inclined such that the LNS-HR satellites, when in the area of the apocynthion, are able to view the South Pole of the Moon. This orientation results in the southern sites on the Moon being more easily viewed by the satellites and in extended periods of coverage to be more numerous and frequent. The area of concentrated coverage moves approximately 180° in 14 days as the Moon rotates beneath the frozen orbit. While maintaining constant coverage of the outpost, this allows for periods of high coverage ratios at most of the sortie sites. Coverage ratio is defined as the total time the Sun and at least one of the LNS-HR satellites are in view during the total time of the opportunity (total time in view/total time of opportunity).

Although there is constant satellite coverage of the South Pole, there are occurrences when Sun visibility is interrupted. When there is full Sun visibility, 100 percent coverage lasts on average for 114 days. Figure 16 in which the opportunity for a 7-day 100 percent coverage mission to the Central Far Side Highlands is presented, the South Pole is being shaded for the entire month long period. Since the Central Far Side Highlands is the most northern sortie site examined, for the most part, longer periods of Sun visibility only occur when the position of the Sun is such that the South Pole received little to no visibility. However, there are limited opportunities for 7-day missions with above 99 percent coverage.

Most of the Moon is visible to the LNS-HR constellation, and the satellites are able to provide 66 percent coverage to the surface for high data rate access. The Sun and at least one of the frozen satellites is in view for 7-day periods. The Central Far Side Highlands has 100 percent coverage for 7 days beginning August 30, 2021, 10:00:00. Note in Figure 16 that the center of ridge moves ~180° in 14 days. This means that ***most*** sortie sites could be serviced for >95 percent of time for 14 days.

See Section 2.9 for S/C pointing analysis.

The orbital period drift is offset by increasing the SMA of one satellite with respect to the other by 5 km. Figure 17 shows the zenith view from selected lunar surface sites of interest as listed in Table 4. The lunar South Pole has Continuous coverage from the LNS-HR constellation. The Far Side SPA has long periods of continuous coverage. The Central Far Highlands have daily coverage with at least one set of continuous 7-day coverage. The North Pole has periods of daily coverage.

Continuing the analysis of the sites of interest from Table 4 and in Figure 17 are the coverage ratios from the LNS-HR constellation to the lunar surface.

Several of the sortie sites from Table 4 were examined to determine the opportunity availability based on their Sun-satellite coverage ratios. Although it is desirable to have 100 percent coverage during the entire 7 day sortie mission for communication and data transfer capabilities, because near side and LIM sortie sites can use the Deep Space Network (DSN) for voice communication that 66.6 percent coverage (8 hr/12 hr ratio) would be acceptable to perform the necessary data transmissions. For the two far side site selected, Far Side SPA (South Pole-Aitken) Floor and Central Far Side Highlands, 100 percent coverage is required because communication using the DSN network is not available.

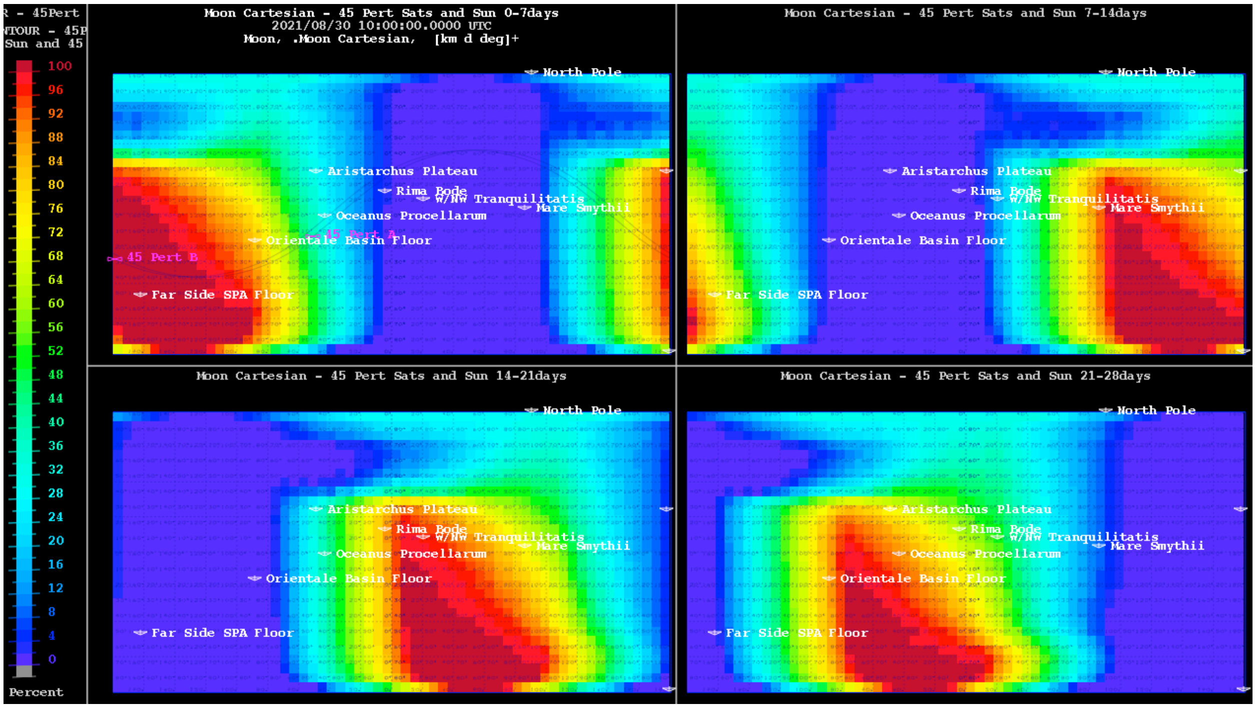


Figure 15.—Lunar surface coverage over 1 month.

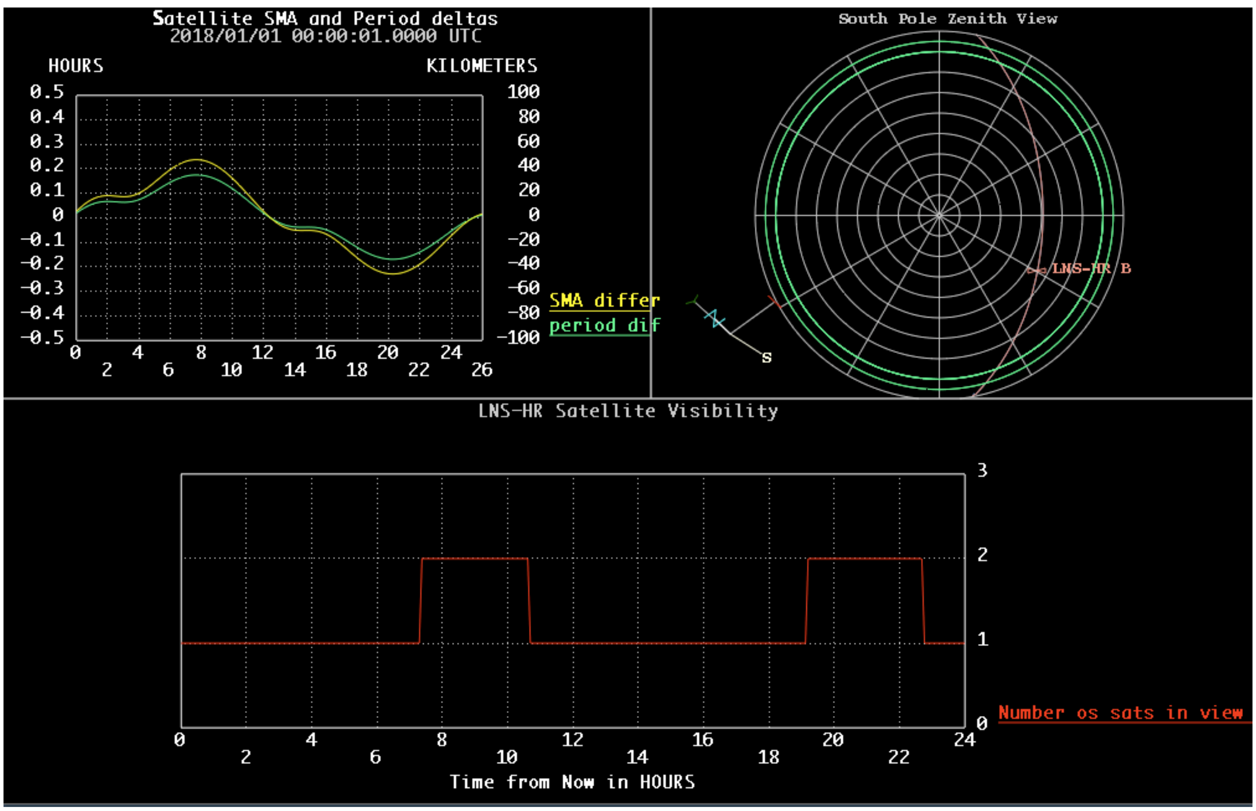


Figure 16.—Frozen orbit drift and South Pole coverage.

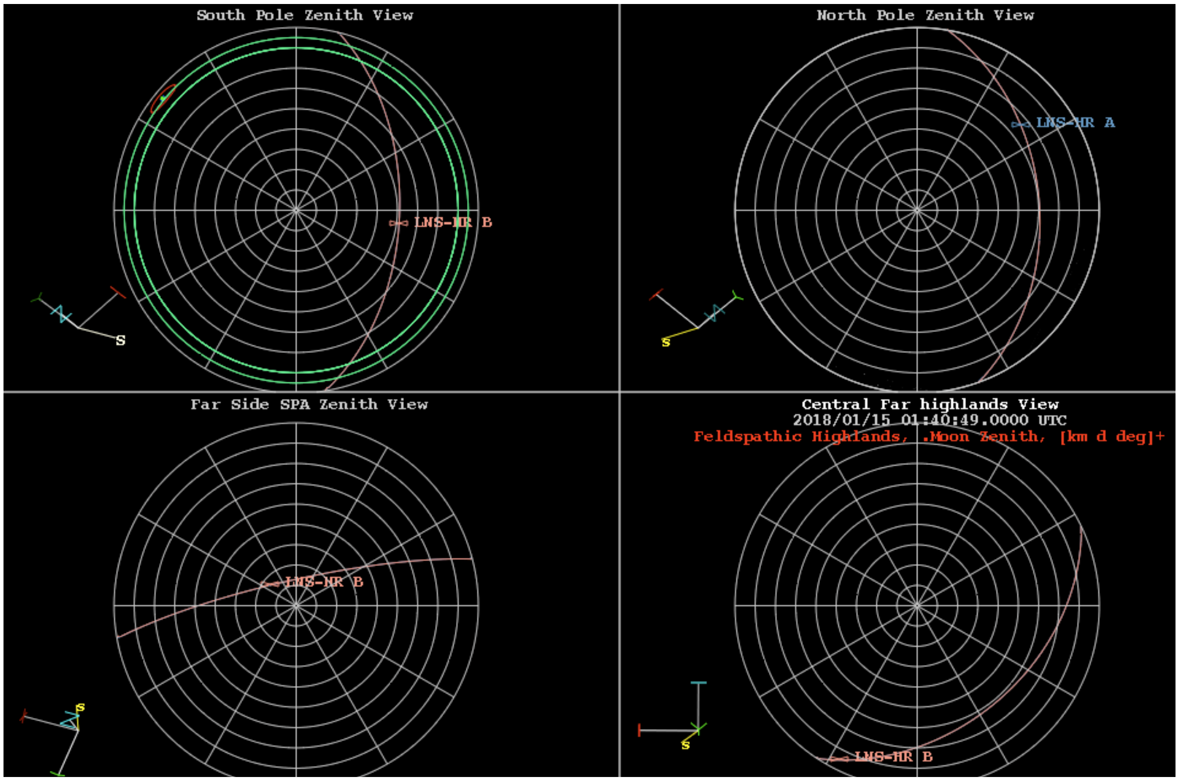


Figure 17.—Zenith view from selected sites.

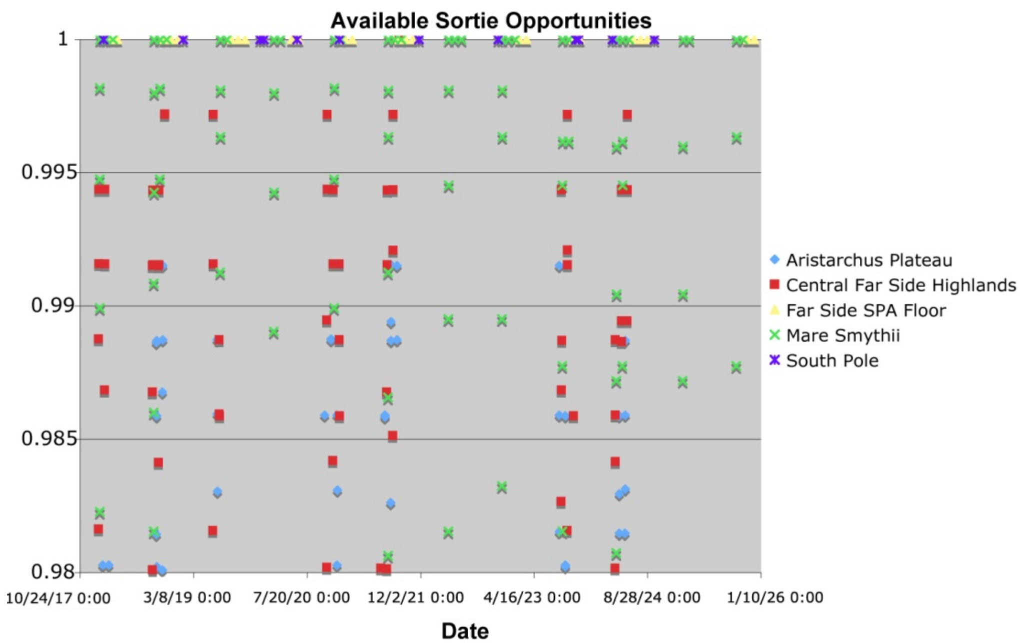


Figure 18.—Visibility coverage ratio of sortie sites.

The visibility ratios for all sortie opportunities 7-days or longer for the sites investigated are shown in Figure 18. Below is a summary of opportunities for each site:

- Far-side sites
    - Central Far Side Highlands

        - 544 sortie opportunities with visibility ratios greater than 66.6 percent (8 hr/12 hr)
            - ♦ Maximum mission duration: 13.5 days
            - ♦ Average mission duration: 7.8 days
        - One opportunity with 100 percent Sun-sat visibility
    - Far Side SPA Floor
        - 25 sortie opportunities, all have 100 percent Sun-sat visibility
            - ♦ Maximum mission duration: 14.4 days
            - ♦ Average mission duration: 10.7 days
- LIM-sites
    - South Pole
        - 12 sortie opportunities, all have 100 percent Sun-sat visibility
            - ♦ Maximum mission duration: 171.2 days
            - ♦ Average mission duration: 114.2 days
    - North Pole
        - No opportunities for a 7-day period having a visibility ratio > 66.6 percent (8 hr/12 hr)
    - Mare Smythii
        - 226 sortie opportunities with visibility ratios > 66.6 percent (8 hr/12 hr)
            - ♦ Maximum mission duration: 14.5 days
            - ♦ Average mission duration: 8.9 days
        - 31 opportunities that have 100 percent Sun-sat visibility
- Near-side sites
    - Aristarchus Plateau
        - 553 sortie opportunities with visibility ratios > 66.6 percent (8 hr/12 hr)
            - ♦ Maximum mission duration: 13.0 days
        - Average mission duration: 7.7 days
        - No opportunities with 100 percent Sun-sat visibility, highest visibility ratio is 0.9943
        - 104 sortie opportunities with visibility ratios > 0.95 for 7-days

The LNS-HR Two-S/C Constellation fulfills 7-day Sortie requirement for far-side sites while providing 24/7 Outpost (South Pole) service. The constellation also provides alternate 8 /12 hr direct to Earth (DTE) link for voice and data for Near Side and LIM sites (except for North Pole).

The lack of 100 percent 7-day opportunities for Aristarchus Plateau and only one such opportunity for Central Far Side Highlands prompted an examination of whether 100 percent coverage could be achieved for slightly shorter sortie missions. Figure 2.17 shows the available coverage opportunities with durations greater than 2 days. Both 100 percent Sun and at least one of the LNS-HR Satellites are visible during these opportunities.

- Aristarchus Plateau
    - Forty-one 100 percent opportunities
        - Five opportunities with durations longer than 6 days
        - Maximum duration is 6.39 days
- Central Far Side Highlands
    - Forty 100 percent opportunities
        - One opportunity with duration 7 days or more (7.12)
    - Fourteen opportunities with durations between 6 and 7 days

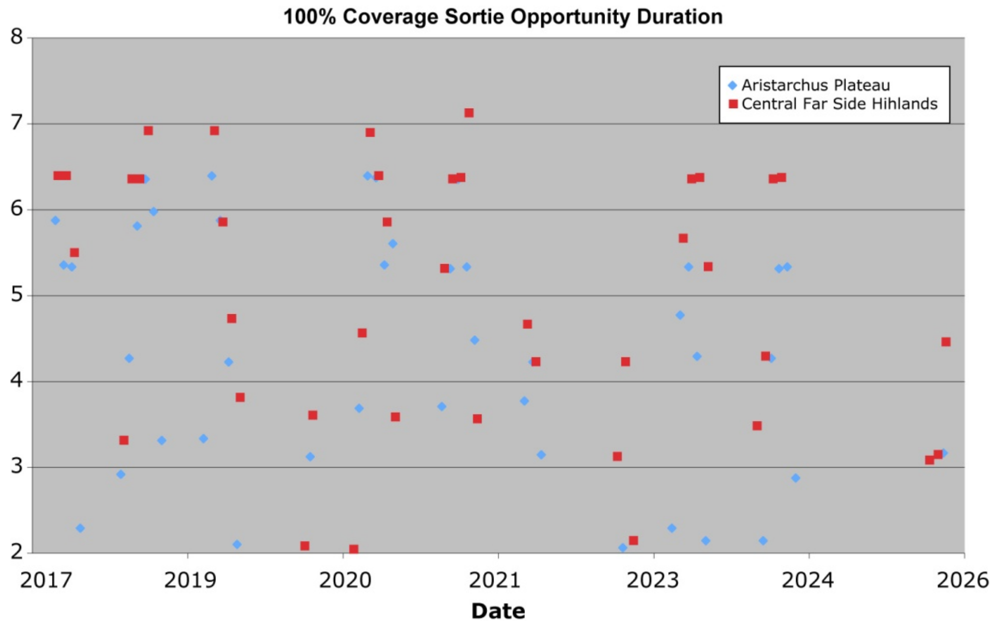


Figure 19.—One hundred percent coverage of sortie sites.

TABLE 9—SUN SHADE DURATION STATISTICS

Maximum	6.91 hr
Minimum	0.047 hr
Average	1.036 hr

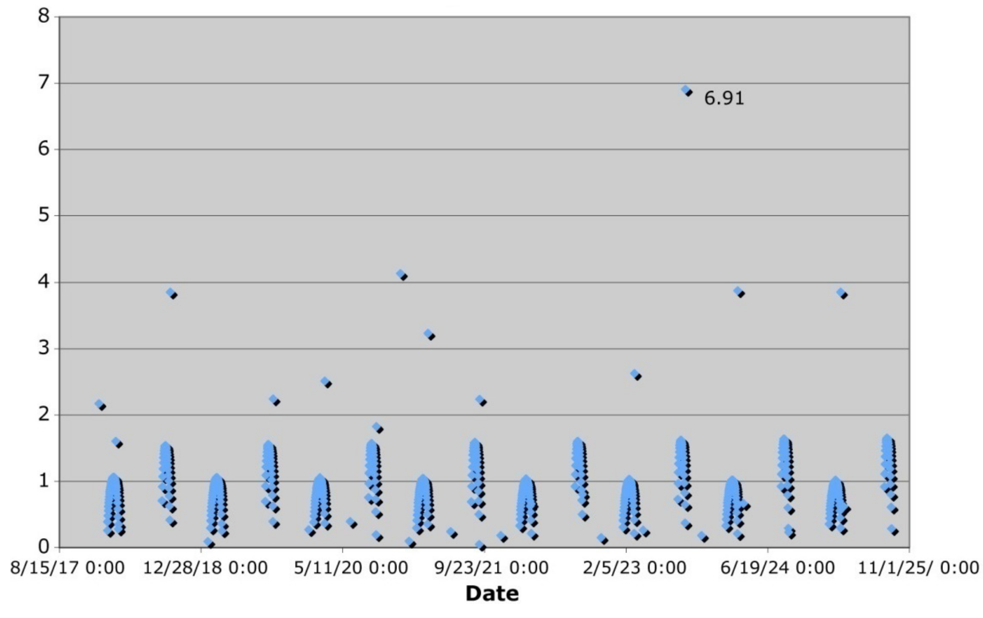


Figure 20.—Satellite Sun shading durations for frozen orbit satellites.

2.5.6 Frozen Orbit Solar Coverage

During the lifetime of the frozen orbit communications satellite, there are 569 Sun-Shade cycles. The periodically occurring short-duration Sun shading periods are caused by the LNS satellite being shaded by the Moon. These shading durations are less than 2 hr long, averaging slightly over an hour (Table 9 and Figure 20. On occasion both the Moon and the Earth shade the S/C resulting in longer shade durations.

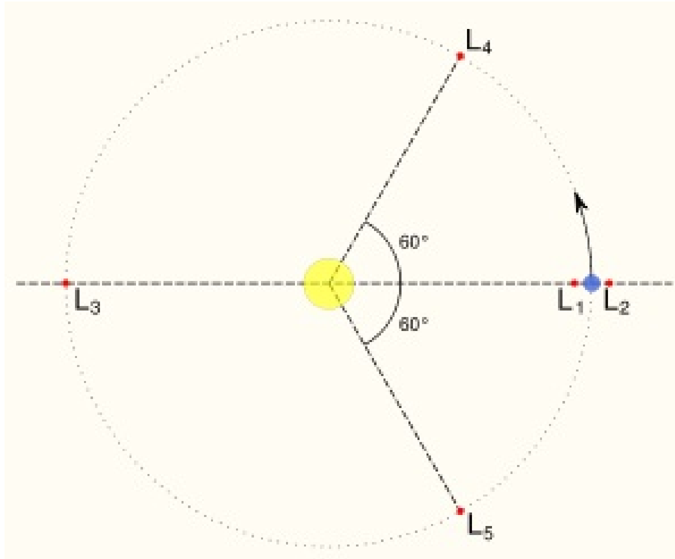


Figure 21.—Lagrange system illustration.

2.6 Mission Description—Earth/Moon Lagrange Orbits

The Lagrange points are the positions around a two body system where the gravitational forces of the two large masses (such as the Earth-Moon system or the Earth-Sun system) equal the centripetal force required to rotate with them. These provide relatively stable orbits where another, smaller body, may orbit and have the same stationary view of the two body system over time. These orbits will require stationkeeping propulsion to maintain the position over sometimes short and sometimes long periods of time (Refs. 8 and 9)

The L_1 point of the Earth-Sun system affords an uninterrupted view of the Sun and is currently home to the Solar and Heliospheric Observatory (SOHO) Satellite. The L_2 point of the Earth-Sun system is home to the Wilkinson Microwave Anisotropy Probe (WMAP) S/C and (perhaps by the year 2011) the James Webb Space Telescope.

Figure 21 shows the five Lagrange points in a two-body system with one body far more massive than the other (e.g., the Sun and the Earth or the Earth and the Moon). In such a system, L3–L5 will appear to share the secondary body's orbit, although in fact they are situated slightly outside it.

2.6.1 Earth/Moon Lagrange Orbit Mission Analysis Assumptions

A representative Lagrange orbit was chosen from the literature at the time of the study. An actual Lagrange orbit desired from trajectory analysis was not found by the time of the S/C design and is ongoing. Two Lagrange orbits, one at L_1 and one at L_2, were chosen from Reference 12 in order to provide a starting point for the power and communications system design. Complete lunar South-Pole coverage can be obtained with two S/C in the constellation.

2.6.2 Earth/Moon Lagrange Orbit Mission Analysis Analytic Methods

Due to the limited time for this study, rather than running an independent Lagrange trajectory, the Lagrange Orbit selection is from the technical report “Multibody Orbit Architectures for Lunar South Pole Coverage” Grebow, et al. Grebow looks for an orbit that would provide optimal South Pole/Shackelton crater coverage. The “optimal” was defined as continuous South Pole—and 7 day sortie site line of site coverage. Best Lagrange constellation option so far consists of a Vertical L_1 and a Vertical L_2 16 day halo orbit each. This is the only family that provides ~100 percent South Pole (North too) but also can provide extended (~7 day) coverage of sorties in northern lunar hemisphere.

2.6.3 Earth/Moon Lagrange Orbit Mission Analysis Event Timeline

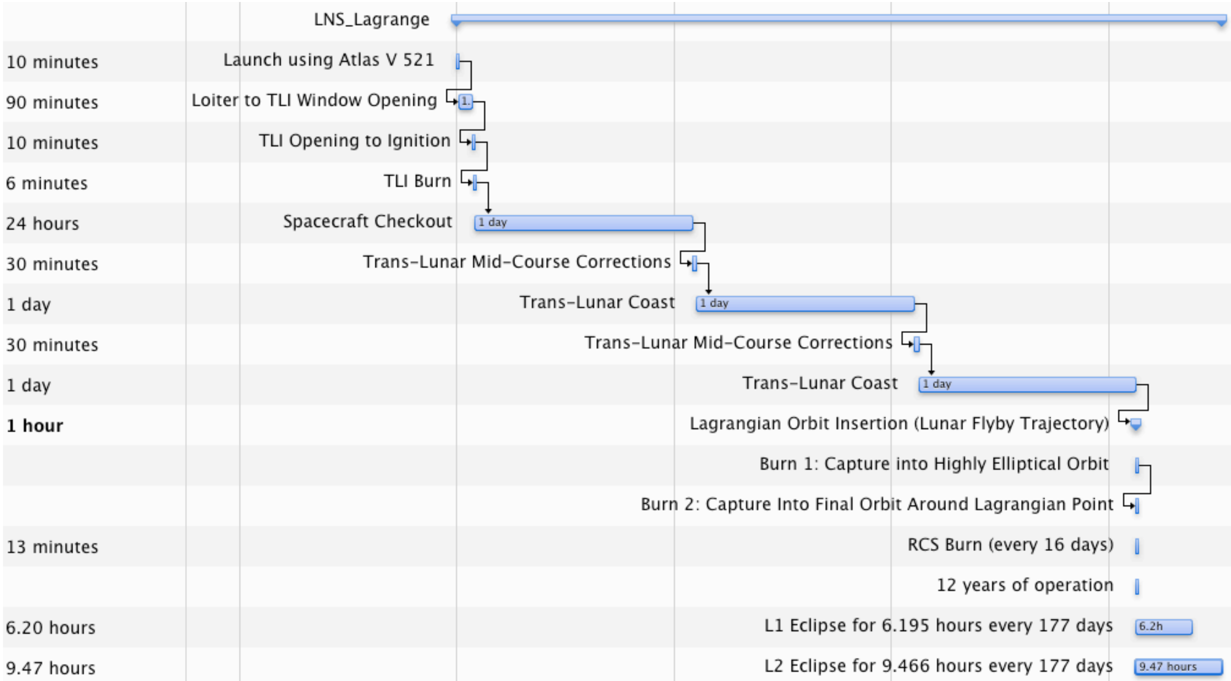


Figure 22.—Lagrange orbit constellation CONOPS.

2.6.4 Earth/Moon Lagrange Orbit Delta V Assumptions

The ΔV values used in Table 10 were taken from recent analysis published in technical literature on station-keeping for and orbit capture into Lagrange orbits. Approximately 500 m/s was allocated for orbit insertion with approximately 60 m/s/yr of station keeping for 12 yr, yields 720 m/s total station keeping ΔV. With larger launch vehicle for Lagrange orbits, propellant loading is, again, not a driving design parameter.

TABLE 10.—LAGRANGE ORBIT ΔV SCHEDULE BY PHASE

Phase Name	OMS ΔV (m/s)	RCS ΔV, m/s	Heritage
Launch from Earth Checkout Loiter to TLI window opening TLI opening to ignition TLI burn			
Correct for ELV dispersions	85		GSFC IMDC
Trans-Earth mid-course corrections trans-lunar coast	25		GSFC IMDC
Lunar orbit capture burn	500		Gordon Thesis, Purdue University
Lunar orbit adjustments	50		Estimate
Phasing maneuvers	720		Folta, Vaughn Paper, AIAA 2004-4741
Lunar telecom orbit	50	50	Added ²v margin for attitude control and station keeping
Disposal	80		Estimate for disposal on lunar surface near Pericynthion
Total ΔV	1550	50	

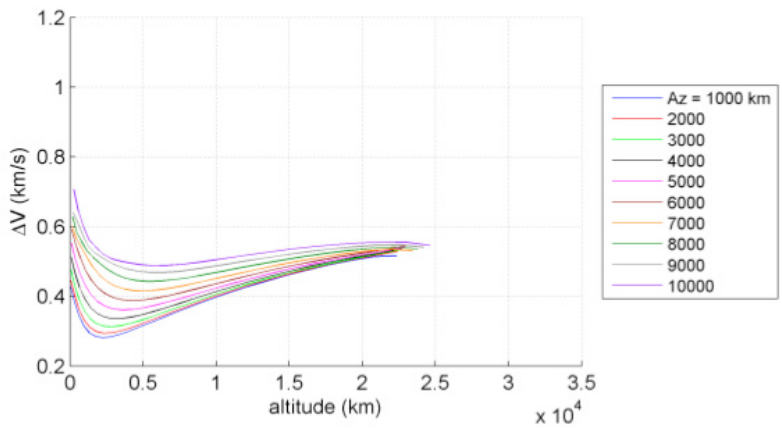


Figure 23.—Manifold insertion ΔV for short time of flight transfers to halo orbits between Az=1,000 and Az = 10,000 km.

2.6.5 Earth/Moon Lagrange Orbit Mission Trajectory Details

Table 11 captures the parameters used in assuming the Lagrange Orbit for the second type of the lunar communication satellite constellation examined in this study. Most of these parameters were obtained from literature.

TABLE 11.—LAGRANGE ORBIT PARAMETERS

Lagrange orbit parameters	Value
L_1 and L_2 Orbits	16 day
Station keeping	~60 m/s/yr
C_3	~ -2 km^2/s^2 lunar centric
Lagrange Points orbit insertion	~500 m/s
Distance to lunar users	~72,000 km
L_1 and L_2 S/C	Dual launch

2.6.6 Earth/Moon Lagrange Orbit Lunar Coverage

See Section 2.9 for S/C pointing analysis.

2.7 Launch Vehicle (LV) Details

2.7.1 LV Data and Trades

Table 12 lists the cost of launch vehicles available at the time of the LRS launch to lunar orbit. All available launch vehicles provide an excess of performance for the LRS mission. The range on the Delta IV medium is related to the choice of payload fairing diameter.

TABLE 12.—LAUNCH SERVICE COSTS FROM THE NASA KENNEDY SPACE CENTER (KSC)

ELV performance range	Cost	Launch vehicles
0 to 3,580 kg	~$175M	Atlas V 401, Delta IV Medium
0 to 2,795 kg	~$185M	Atlas V 501
0 to 4,711 to 5,400 kg	~$224M	Delta IV Medium +

2.7.2 LV Performance and Cost

Assumptions:

- Performance range of the ELV was based on a targeted C_3 = –2 km^2/s^2 (lunar).
- Used a launch date of 1 January 2018 and an eastern range.

- Per our PPBE07 assumptions, the prices reflect revised Launch Service Program (LSP) Pricing Strategy for the Evolved Expendable Launch Vehicles (EELV) Launch Service. Note that this is a slightly reduced amount of conservatism based on some recent acquisitions.
- All costs are estimated in real-year dollars (2008) based on current NASA Launch Services (NLS) contract information. Prices provided are for a complete launch service, which includes nominal allocation for mission unique launch vehicle modifications/services, mission integration, launch site payload processing, range safety, and telemetry/communications.
- There is no launch delay penalty cost assumed in the budget.
- The launch service prices are estimates and are not to be considered commitments from the LSP.
- Launch Service Task Order (LSTO) is awarded to the Contractor that provides the best value in launch services to meet the Government's requirements based on technical capability/risk, reasonableness of proposed price, and past performance.
- Due to uncertainty with U.S. Air Force (USAF) infrastructure cost allocations, the NASA Headquarters (HQ) Programs should carry threat to cover resulting unexpected price fluctuations.

Cost/performance range difference associated with the 0 to 3580 kg and 0 to 2795 kg scenarios is associated with the use of a larger fairing.

2.7.3 LV Details—Frozen Orbit Constellation

The Atlas V 401 was initially baselined as the launch vehicle choice for the LNS-HR mission based on the LRS design. The Atlas V provides ~66 percent mass margin for LNS-HR (i.e., it could launch almost three LNS-HR) with sufficient volume for a dual launch or secondaries. For lunar orbit [C_3 = –2.0 km^2/s^2], Atlas V 501 can launch 2,795 kg, Atlas V 401 3,580 kg, and Atlas V 511 3,915 kg (Figure 24 which was generated using KSC’s online ELV performance estimation tool on the Atlas class of launch vehicles.), compared to LRS mass of 1124 kg with growth. After sizing the LNS-HR S/C from bottoms up, in order to fit two LNS satellites onto one launch vehicle, the frozen orbit cases required the Atlas V 511 performance (the blue line in Figure 24).

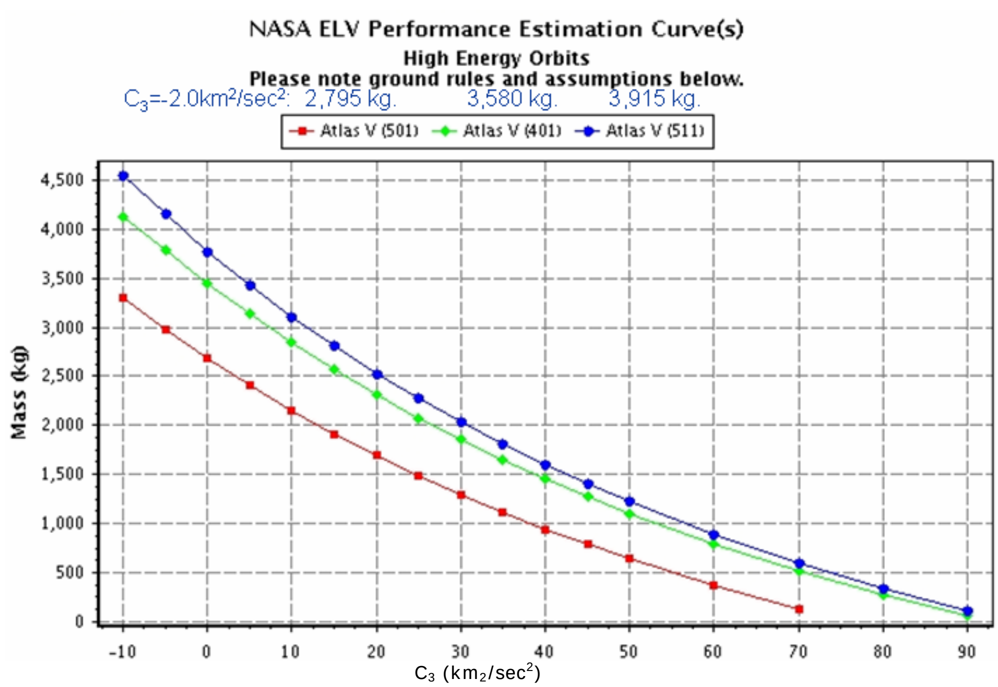


Figure 24.—Atlas V 501, 401, 511 performance curve for C_3 = –2 km^2/sec^2 (lunar).

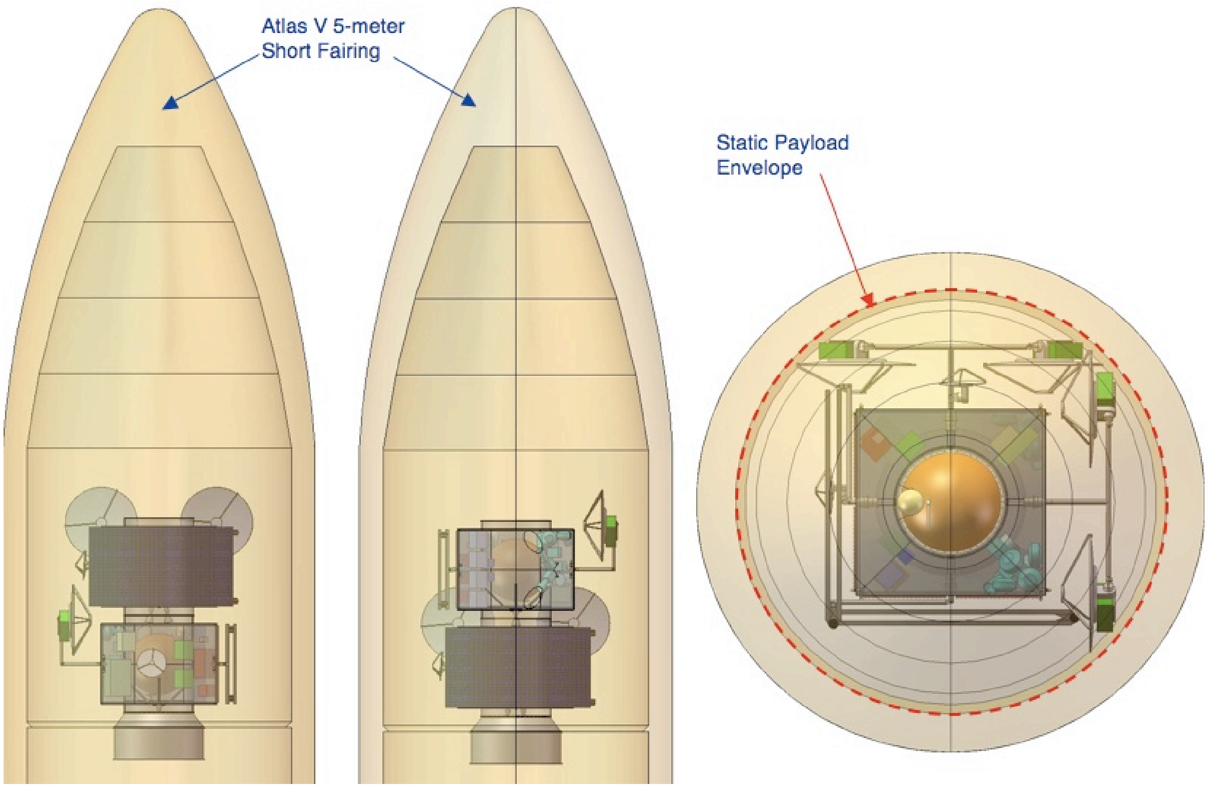


Figure 25.—Atlas V 511 fairing dimensions with notional two LNS launch packaging.

Figure 25 shows how two LNS-HR Frozen orbit satellites would fit onto the Atlas 511 launch vehicle.

2.7.4 LV Details—Lagrange Orbit Constellation

After sizing the LNS-HR S/C from bottoms up, in order to fit two LNS satellites onto one launch vehicle, the Lagrange orbit constellation required the Atlas V 521 performance (Figure 26 and Figure 27).

2.8 S/C Pointing

2.8.1 Frozen Orbit S/C Pointing

Four systems on the S/C must point continuously to four different locations (Figure 28)

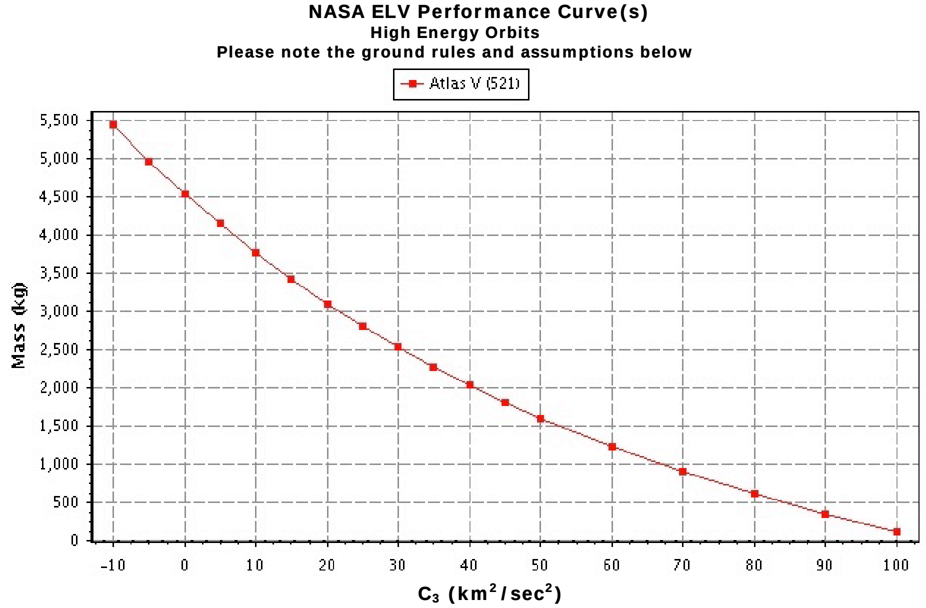
- Lunar antennas to two separate sites
- Earth antenna to Earth
- SA to Sun

The following are a series of S/C to target pointing options to provide pointing between the following: data from lunar surface to LNS, from LNS to Earth and from LNS SAs to the Sun. In all cases, the pointing refers to the communications system and how it is pointed.

- Point S/C at Moon
- Point S/C at Sun
- Point S/C at Earth

Performance plot results:

Important note: The data contained in these curves are based on ground rules and assumptions located below the plot. Please read this information carefully. This information is intended for NASA customers only.



Assumptions:
Atlas V (521)

- This performance does not include the effects of orbital debris compliance, which must be evaluated on a mission-specific basis. This could result in a significant performance impact for mission in which launch vehicle hardware remains in Earth orbit.
- 3-sigma mission required margin, plus additional reserves as determined by the LSP.
- Launch from SLC–41 at Cape Canaveral Air Force Station (CCAFS).
- Performance values assume harness, logo, reradiating antenna, three payload fairing doors.
- Payload mass greater than 9000 kg (19,841 lb) may require mission unique accommodations.
- Type B2 payload adapter plus type C2 spacer.
- 5-m Short Payload Fairing
- 185 km (100 n mi) minimum park orbit perigee altitude.
- 185 km (100 n mi) minimum escape orbit perigee altitude.
- Performance shown is applicable to declinations between 28.5° and –28.5°.

Last updated: May 22, 2006, 8:11:53 am.

Figure 26.—Atlas V 521 performance curve for $C_3 = -2\ km^2/sec^2$ (lunar).

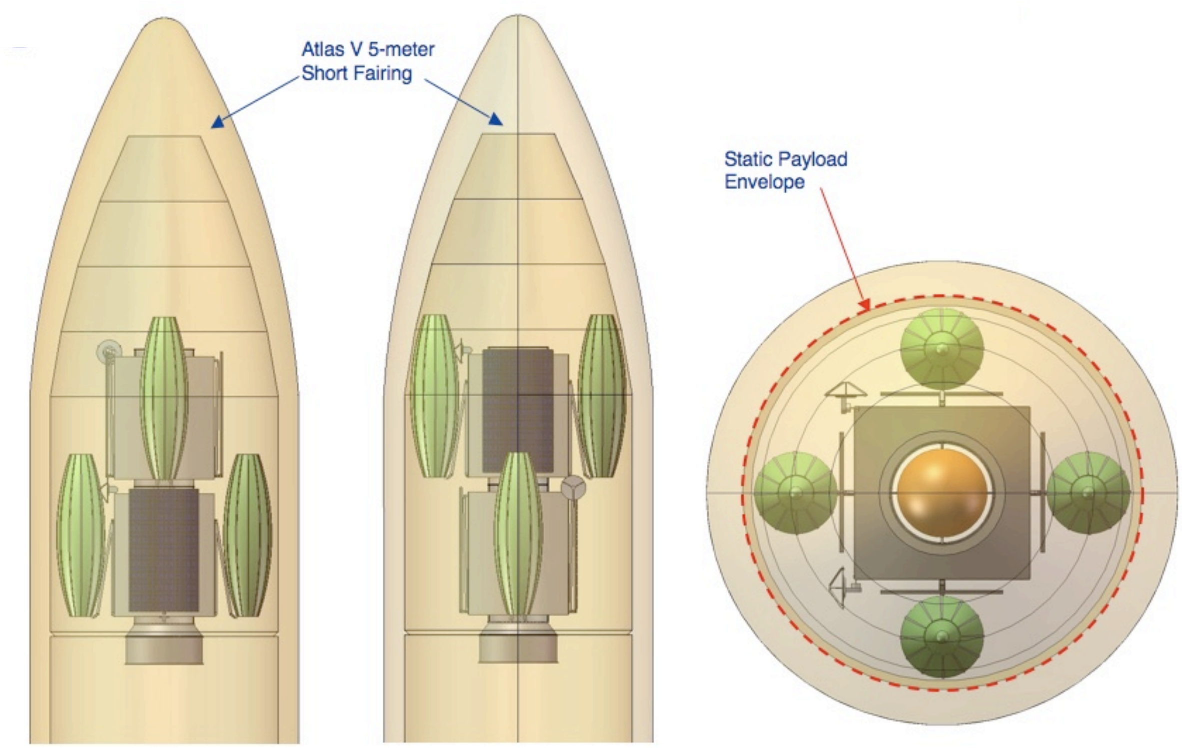


Figure 27.—Lagrange constellation S/C inside Atlas V 521 fairing.

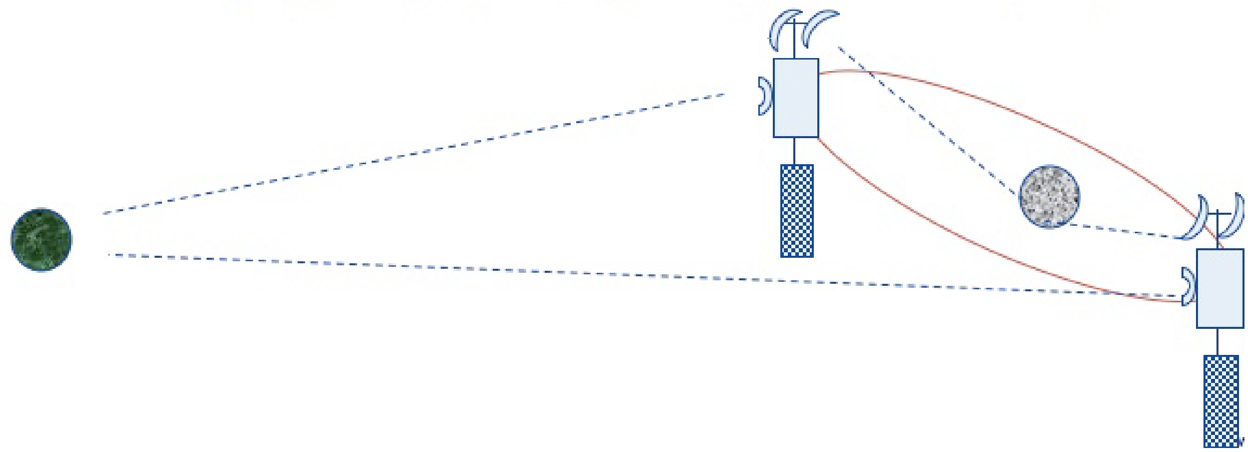


Figure 28.—Frozen orbit constellation pointing notional illustration.

Past designs have pointed the S/C at the Moon and provided an antenna on an arm to point at the Earth. At issue was the flipping that the S/C would need to perform to point the Earth antenna above or below the orbit plane based on the relation of the orbit plane to the Earth. Such flipping maneuvers would require propellant and would incur risks. One way to eliminate this flipping maneuver would be to point the S/C to the Earth. This has an additional benefit since the Sun is always very close to the Earth's orbit plane when viewed from an orbit around the Moon. Such an approach would require very little pointing for the Earth antenna, the SA will only require a single gimbal that will make one revolution each month (Figure 29). The lunar pointing antennas regardless of approach need to point to different portions of the Moon. With the Earth pointing S/C approach the antennas will need to rotate once a day and be able to point up and down 45° from the orbit plane.

The proposed pointing of LNS for the frozen orbit is summarized as follows

- Point 0.5 m antenna mounted on the S/C body at Earth
- Single (one axis gimbal) SA perpendicular to Moon/Earth orbit plane
- Point lunar antennas down/up to Moon (±45° elevation) and rotate about their axis 360° per orbit (once a day)
- The S/C will not have to flip over (as with past designs)
- GN&C
    - Satellite held in 'fixed' pointing to Earth
    - Lunar gravity gradient torques may be an issue (but should be periodic and can be taken out by wheels)

Since the Moon's orbit is only inclined ~6° from ecliptic, for the Moon, the Earth and the Sun are roughly in the same plane when viewed from an orbit about the Moon.

- Pointing approach
    - Align SA axis parallel to lunar axis, rotate SA about one axis of the S/C body once a month
    - Point the antenna mounted on the body of the S/C at Earth continuously
    - Rotate lunar antenna boom once an orbit
    - Gimbal the lunar antennas to individual lunar users
    - Eliminates S/C flip or additional Earth antenna from previous designs

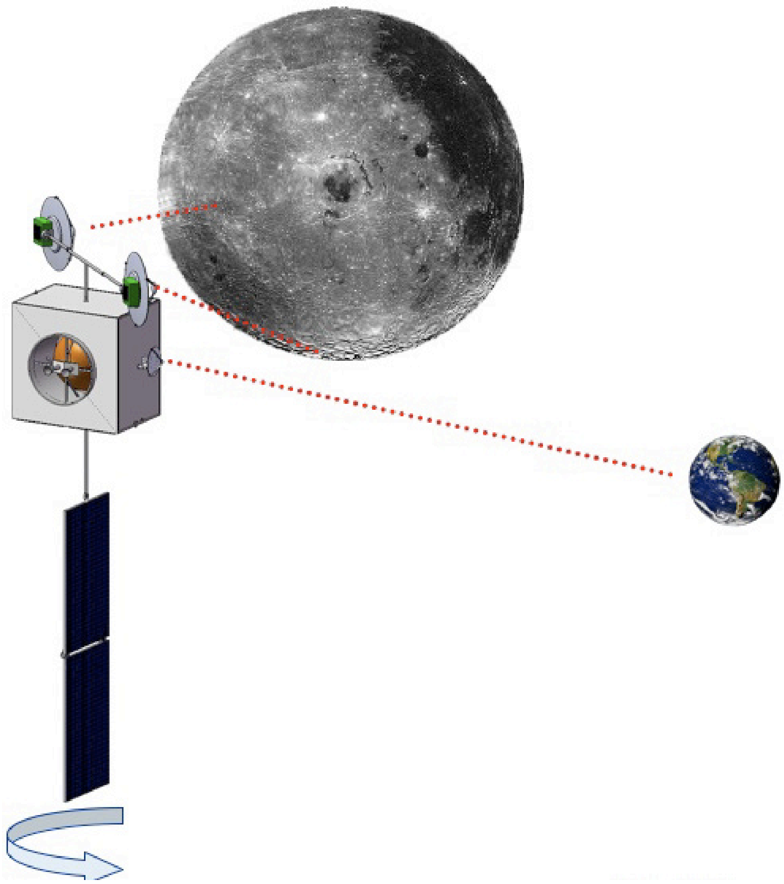


Figure 29.—Frozen orbit pointing notional illustration for single satellite.

The S-band and a Ka-band antenna are pointed at the lunar surface and rotate in order to continuously point to a desired site on the surface under the LNS as long as that point is in sight as shown in Figure 30. For example, the antenna points to the South Pole site (purple beam), until that site has gone out of range of the LNS through its orbit. A second LNS in the same orbit will pick up the line of site coverage with the South Pole as the first LNS goes out of range.

Figure 31 shows that the SAs rotate independently of the antenna in order to track the Sun. The yellow beam is the notional modeling of the line of site between the SA panels and the Sun. The purple beam is the antenna pointed at the South Pole of the lunar surface. The yellow beam representing viewing of the Sun from the S/C can also be seen in Figure 30. When looking from the point of view of the SAs, the Sun, the plane of the area of the SAs is perpendicular to the vector connecting the Sun to the arrays. From the graphic in Figure 30, the yellow line goes off to the top left where the Sun would be located with respect to the Moon.

The DTE antenna is located on the body of the LNS S/C bus, and is set to point at the Earth. This antenna can be seen in Figure 32 as the dish located on the Earth facing portion of the S/C bus. The Solar array is shown as the blue rectangular panel located at the bottom of the S/C bus. The purple beam is the link between the antenna and the lunar South Pole and the green beam is the link between the second antenna and a site chosen from the list of lunar surface sites of interest in Table 4.

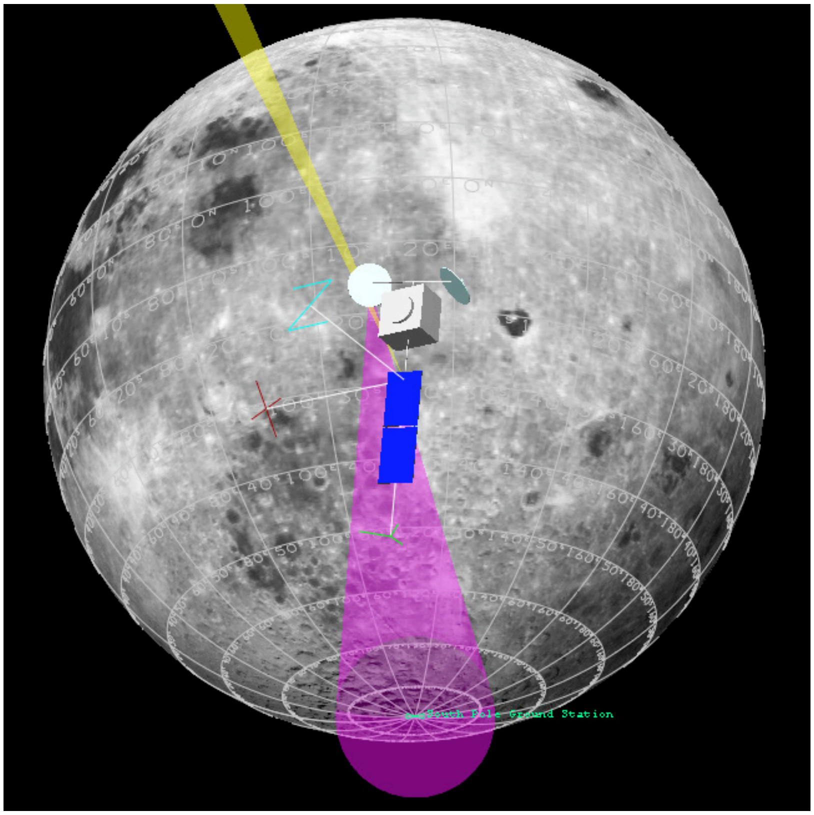


Figure 30.—Frozen orbit pointing SOAP modeling.

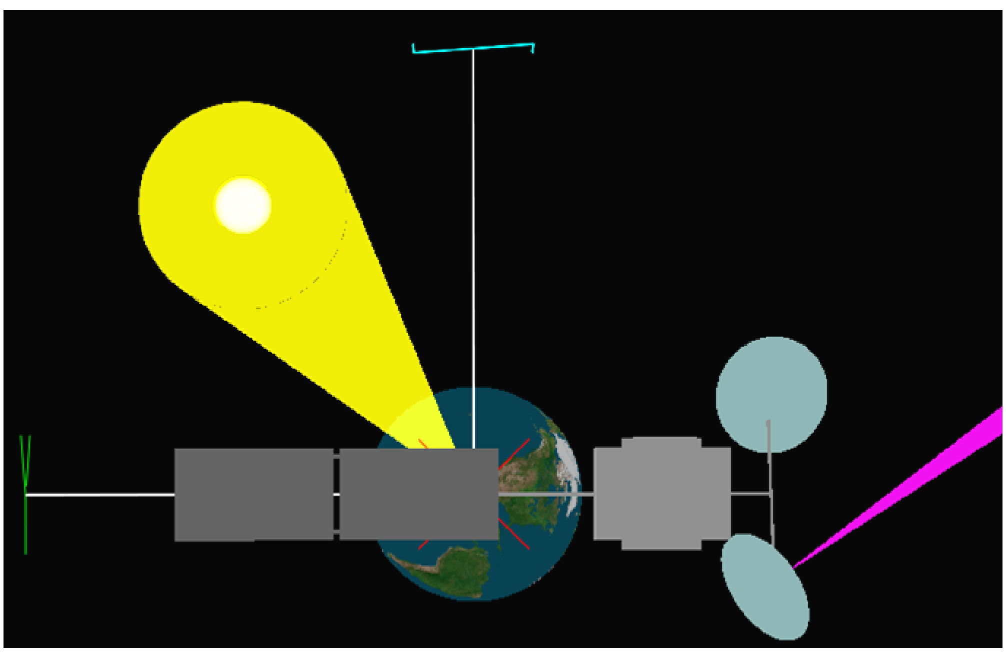


Figure 31.—Frozen orbit pointing modeled in SOAP.

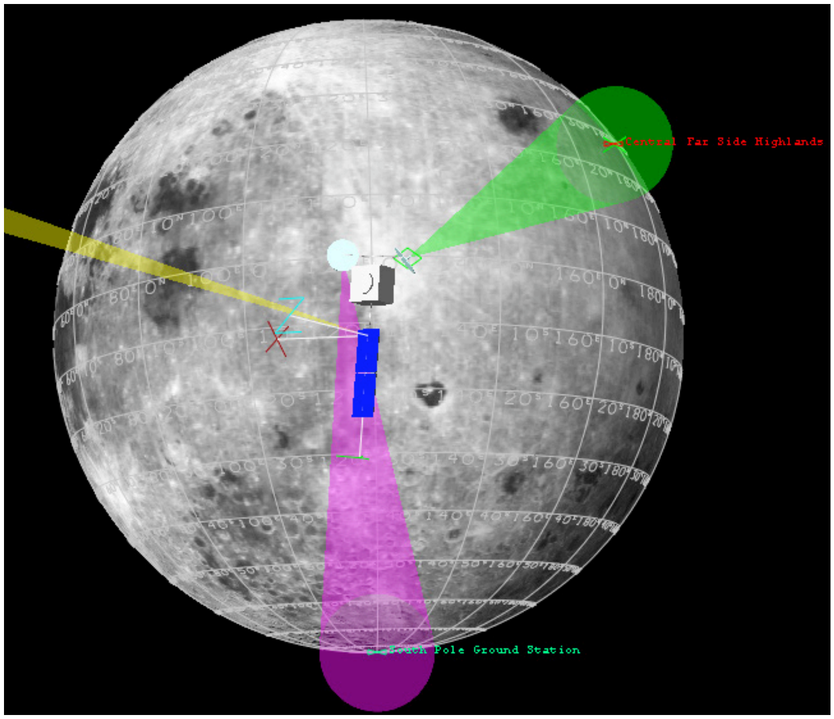


Figure 32.—Frozen orbit pointing modeled in SOAP.

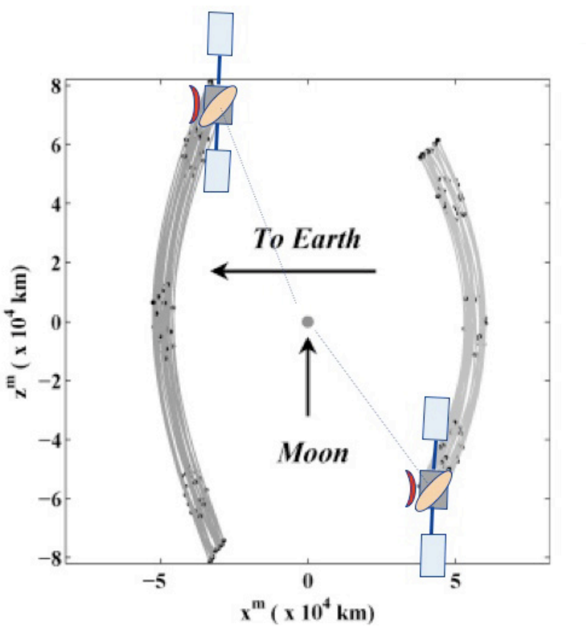


Figure 33.—Lagrange orbit pointing notional diagram.

2.8.2 Lagrange Orbit S/C Pointing

Options considered for pointing

- Earth pointing S/C
- Moon Pointing S/C

Having the S/C point parallel to chosen Earth-Moon vector was easier on both Earth antenna and the SAs. The 7 m lunar antennas need to gimbal ~ 160° along the *y*-axis. The S/C was slewed about the *z*-axis to point to Moon. For this maneuver, the 0.5 m Earth antenna needed to gimbal ~ 15° in *z* and *y*-axis (Figure 33).

2.9 System Design Trade Space

Two major trades were made in the design session. First, the trade was made between constellation orbits: frozen circular lunar orbits versus Lagrange orbit constellations. Second, the trade was made in each constellation on type of communications technology: microwave (RF) versus Optical.

3.0 Baseline Design

3.1 Top Level Design (MEL and PEL)—Microwave LNS

3.1.1 Master Equipment List (MEL)—Microwave LNS (Frozen Orbit)

Table 13 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 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 WGA schedule outlined in Table 6 in Section 2.4. 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.

Where the MEL (Table 13) captures the bottoms up estimation of CBE and growth percentage line item by item from the subsystem designer, Table 15 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 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.

TABLE 13.—MASTER EQUIPMENT LIST—LNS RF FROZEN ORBIT CASE

WBS no.	Description LNS-HR (September 2008) Frozen, RF	Quantity	Unit mass, kg	Basic mass, kg	MGA, %	Growth, kg	Predicted mass, kg
6.0	Lunar Network Satellite-High Rate	-	-	1288.1	11	142.8	1430.9
6.0.1	RF Communications Package	-	-	123.1	12	14.9	138.0
6.0.2	Optical Communications Package	-	-	0.0	0	0.0	0.0
6.0.3	C&DH	-	-	77.2	26	20.2	97.4
6.0.4	GN&C	-	-	42.5	20	8.5	51.0
6.0.5	Structures & Mechanical Systems	-	-	233.7	23	54.7	288.4
6.0.6	Power System	-	-	83.0	30	24.9	107.9
6.0.10	Propulsion (Chemical)	-	-	10.6	10	1.1	11.7
6.0.11	Propellant Management (Chemical)	-	-	43.9	13	5.7	49.5
6.0.12	Propellant (Chemical)	-	-	617.6	0	0.0	617.6
6.0.13	Thermal Control (Non-Propellant)	-	-	56.5	23	13.0	69.5

TABLE 14.—SYSTEM INTEGRATION—LNS RF FROZEN ORBIT CASE

GLIDE container:	*LunarRelaySat: LNS_Frozen_RF*				
	Spacecraft Master Equipment List Rack-up (Mass)			COMPASS S/C Design	
WBS	Main Subsystems	Basc (CBE) Mass (kg)	Growth (kg)	Predicted (Total) Mass (kg)	Aggregate Growth (%)
6.0	Lunar Network Satellie-High Rate	1288.1	142.8	1430.9	
6.0.1	RF Communications Package	123.1	14.9	138.0	12%
6.0.2	Optical Communications Package	0.0	0.0	0.0	TBD
6.0.3	Command & Data Handling (C&DH)	77.2	42.5	97.4	55%
6.0.4	Guidance, Navigation and Control System (GN&C)	42.5	8.5	51.0	20%
6.0.5	Structures & Mechanical Systems	233.7	54.7	288.4	23%
6.0.6	Power System	83.0	24.9	107.9	30%
6.0.7	Propulsion (Electric)	0.0	0.0	0.0	TBD
6.0.8	Propellant Management (EP)	0.0	0.0	0.0	TBD
6.0.9	Propellant (EP)	0.0			
6.0.10	Propulsion (Chemical)	10.6	1.1	11.7	10%
6.0.11	Propellant Management (Chemical)	43.9	5.7	49.5	13%
6.0.12	Propellant (Chemical)	617.6			
6.0.13	Thermal Control (Non-Propellant)	56.5	13.0	69.5	23%
	Estimated Spacecraft Dry Mass	670	143	813	21%
	Estimated Spacecraft Wet Mass	1288	165	1431	
	Estimated Spacecraft Inert Mass (for traj.)	689	201	889.7	
System LeveL Growth Calculations					Total Growth
	Dry Mass Desired System Level Growth	670	201	871.6	30%
	Mass Margin (carried at system level)			58	9%
	Total Wet Mass with Growth	1288	201	1489.2	
	Available Launch Performance to C3 (kg)			3915.0	kg
	Total Number of LNS-HR Satellites			2	
	Total mass of Two LNS-HR satellites			2978.5	kg
	Launch margin available (kg)			936.5	kg
	Launch margin available (%)			23.9	%

3.1.2 Power Equipment List (PEL)—Microwave LNS (Frozen Orbit)

The power listing for nominal loads is shown in Table 15.

- Power system tracks its own power
- Power system must provide:
  - ~1250 W (with 30 percent growth) during communications operations
  - ~1000 W (with 30 percent growth) during S/C shadowing (seasonal) assuming Ka not used
- Generated thermal waste heat includes:
  - ~700 W (with 30 percent growth) during communications operations
  - ~550 W (with 30 percent growth) during S/C shadowing (seasonal) assuming Ka not used

TABLE 15.—PEL—FROZEN ORBIT—RF

	Notional time	Propulsion, W	Avionics, W	Comm. , W	Thermal, W	GN&C, W	Power , W	Science, W	CBE Total, W	30 % Margin, W	Total, W
Launch	10 m	20	25	20	5	48	0.0	0.0	118	35.34	153
S/C checkout	24 h	75	152	589	30	112	0.0	0.0	958	287.43	1246
Cruise	4 d	75	152	20	30	112	0.0	0.0	389	116.73	506
Cruise Communications		75	152	387	30	112	0.0	0.0	756	226.83	983
Orbit Injection	30 m	75	152	387	30	168	0.0	0.0	812	243.63	1056
Lunar Sunlit Communications	Continuous	75	152	589	30	112	0.0	0.0	958	287.43	1246
Lunar Eclipse Communications	2h/d (max.)	75	152	387	60	112	0.0	0.0	786	235.83	1022
Waste Heat											
Launch		0.0	25	10.0	0.0	32.0	0.0	0.0	66.5	20.0	86.5
S/C checkout		0.0	152	294.5	0.0	92.0	0.0	0.0	538.6	161.6	700.2
Cruise		0.0	152	10.0	0.0	92.0	0.0	0.0	254.1	76.2	330.3
Cruise Communications		0.0	152	193.5	0.0	92.0	0.0	0.0	437.6	131.3	568.9
Orbit Injection		0.0	152	193.5	0.0	92.0	0.0	0.0	437.6	131.3	568.9
Lunar Sunlit Communications		0.0	152	294.5	0.0	92.0	0.0	0.0	538.6	161.6	700.2
Lunar Eclipse Communications		0.0	152	193.5	0.0	92.0	0.0	0.0	437.6	131.3	568.9

3.2 Top Level Design (MEL and PEL)—Optical LNS (Frozen Orbit)

3.2.1 MEL—Optical LNS (Frozen Orbit)

Table 16 lists the top level of the MEL the frozen orbit case using optical communications as well as microwave. Figure 17 wraps up those total masses, CBE and total mass after applied growth percentage for the frozen orbit optical communications case.

TABLE 16.—MEL—FROZEN ORBIT—OPTICAL

WBS no.	Description LNS-HR (September 2008) Frozen, Optical	Quantity	Unit mass, kg	Basic mass, kg	MGA, %	Growth, kg	Predicted mass, kg
6.0	Lunar Network Satellite-High Rate	-	-	1261.9	11	144.2	1406.1
6.0.1	RF Communications Package	-	-	92.3	11	10.5	102.8
6.0.1.1	Ka band (40/37) DTE Link	-	-	0.0	0	0.0	0.0
6.0.1.2	Ka band (23/26) Relay Link	-	-	15.5	10	1.6	17.1
6.0.1.3	Ka band (40/37) TT&C and Ranging	-	-	0.0	0	0.0	0.0
6.0.1.4	S band Relay Link	-	-	24.4	10	2.4	26.8
6.0.1.5	Data Router and SSR	-	-	4.0	10	0.4	4.4
6.0.1.6	Antennas (Booms excluded)	-	-	48.4	13	6.1	54.5
6.0.2	Optical Communications Package	-	-	18.1	32	5.8	23.8
6.0.2.1	Optical Head	-	-	6.7	34	2.3	9.0
6.0.2.2	Electronics Box	-	-	10.8	30	3.2	14.0
6.0.2.3	Optical Head-to-E.Box Interface	-	-	0.5	55	0.3	0.8
6.0.2.4	Interface to Spacecraft	-	-	0.1	30	0.0	0.1
6.0.3	C&DH	-	-	77.2	26	20.2	97.4
6.0.3.1	C&DH Hardware	-	-	74.2	25	18.5	92.7
6.0.3.2	Instrumentation & Wiring	-	-	3.0	55	1.7	4.7
6.0.4	GN&C	-	-	42.5	20	8.5	51.0
6.0.4.1	GN&C Hardware	-	-	42.5	20	8.5	51.0
6.0.5	Structures & Mechanical Systems	-	-	225.0	23	52.5	277.5
6.0.5.1	Primary Structures	-	-	199.1	25	49.8	248.9
6.0.5.2	Secondary Structures	-	-	7.3	25	1.8	9.1
6.0.5.3	Installation	-	-	3.6	25	0.9	4.4
6.0.5.4	Mechanisms	-	-	15.0	0	0.0	15.0
6.0.6	Power System	-	-	83.0	30	24.9	107.9
6.0.6.1	Battery System	-	-	43.0	30	12.9	55.9
6.0.6.2	Solar Array	-	-	25.0	30	7.5	32.5
6.0.6.3	Power Management & Distribution	-	-	15.0	30	4.5	19.5
6.0.10	Propulsion (Chemical)	-	-	10.6	10	1.1	11.7
6.0.10.1	Main Engine	-	-	4.6	10	0.5	5.1
6.0.10.2	Reaction Control Sys tem	-	-	6.0	10	0.6	6.6
6.0.11	Propellant Management (Chemical)	-	-	51.5	15	7.8	59.2
6.0.11.1	OMS Propellant Management	-	-	0.0	0	0.0	0.0
6.0.11.2	RCS Propellant Management	-	-	51.5	15	7.8	59.2
6.0.12	Propellant (Chemical)	-	-	605.4	0	0.0	605.4
6.0.12.1	Main Engine Propellant	-	-	0.0	0	0.0	0.0
6.0.12.2	RCS Propellant	-	-	605.4	0	0.0	605.4
6.0.13	Thermal Control (Non-Propellant)	-	-	56.5	23	13.0	69.5
6.0.13.1	Active Thermal Control	-	-	4.2	21	0.9	5.1
6.0.13.2	Passive Thermal Control	-	-	41.4	23	9.5	50.9
6.0.13.3	Semi-Passive Thermal Control	-	-	10.9	24	2.6	13.5

TABLE 17.—MEL—FROZEN ORBIT—OPTICAL

COMPASS study: Lunar Network Satellite - High Rate (LNS-HR)				Study Date	9/22/08
GLIDE container:	LunarRelaySat: LNS_Frozen_Optical				
Spacecraft Master Equipment List Rack-up (Mass)				COMPASS S/C Design	
WBS	Main Subsystems	Basc (CBE) Mass (kg)	Growth (kg)	Predicted (Total) Mass (kg)	Aggregate Growth (%)
6.0	**Lunar Network Satellie-High Rate**	**1261.9**	**144.2**	**1406.1**	
6.0.1	RF Communications Package	92.3	10.5	**102.8**	11%
6.0.2	Optical Communications Package	18.1	5.8	**23.8**	32%
6.0.3	Command & Data Handling (C&DH)	77.2	42.5	**97.4**	55%
6.0.4	Guidance, Navigation and Control System (GN&C)	42.5	8.5	**51.0**	20%
6.0.5	Structures & Mechanical Systems	225.0	52.5	**277.5**	23%
6.0.6	Power System	83.0	24.9	**107.9**	30%
6.0.7	Propulsion (Electric)	0.0	0.0	**0.0**	TBD
6.0.8	Propellant Management (EP)	0.0	0.0	**0.0**	TBD
6.0.9	Propellant (EP)	0.0			
6.0.10	Propulsion (Chemical)	10.6	1.1	**11.7**	10%
6.0.11	Propellant Management (Chemical)	51.5	7.8	**59.2**	15%
6.0.12	Propellant (Chemical)	605.4			
6.0.13	Thermal Control (Non-Propellant)	56.5	13.0	**69.5**	23%
	Estimated Spacecraft Dry Mass	**657**	**144**	**801**	**22%**
	Estimated Spacecraft Wet Mass	1262	167	1406	
	Estimated Spacecraft Inert Mass (for traj.)	674	197	**871.3**	
System LeveL Growth Calculations					**Total Growth**
	Dry Mass Desired System Level Growth	657	197	**853.5**	**30%**
	Mass Margin (carried at system level)		**53**		**8%**
	Total Wet Mass with Growth	1262	197	**1458.9**	
	Available Launch Performance to C3 (kg)			3915.0	kg
	Total Number of LNS-HR Satellites			2	
	Total mass of Two LNS-HR satellites			2917.8	
	Launch margin available (kg)			**997.2**	

3.2.2 PEL—Optical LNS (Frozen Orbit)

The power listing for nominal loads is listed in Table 18.

- Power system tracks its own power
- Power system must provide:
  - ~1250 W (with 30 percent growth) during communications operations
  - ~1000 W (with 30 percent growth) during S/C shadowing (seasonal) assuming Ka not used
- Generated thermal waste heat includes:
  - ~700 W (with 30 percent growth) during communications operations
  - ~550 W (with 30 percent growth) during S/C shadowing (seasonal) assuming Ka not used

TABLE 18.—PEL—FROZEN ORBIT—OPTICAL

	Notional Time	Propulsion, W	Avionics, W	Comm. , W	Thermal, W	GN&C, W	Power , W	Science, W	CBE Total, W	30 % Margin, W	Total, W
Launch	10 m	20	76	20	5	48	0.0	0.0	169	50.79	220
S/C checkout	24 h	75	150	589	30	112	0.0	0.0	956	286.65	1242
Cruise	4 d	75	150	20	30	112	0.0	0.0	387	115.95	502
Cruise Communications		75	150	387	30	112	0.0	0.0	754	226.05	980
Orbit Injection	30 m	75	150	387	30	168	0.0	0.0	810	242.85	1052
Lunar Sunlit Communications	Continuous	75	150	589	30	112	0.0	0.0	956	286.65	1242
Lunar Eclipse Communications	2 h/d (max.)	75	150	387	60	112	0.0	0.0	784	235.05	1019
Waste Heat											
Launch		0.0	76	10.0	0.0	32.0	0.0	0.0	118.0	35.4	153.4
S/C checkout		0.0	150	294.5	0.0	92.0	0.0	0.0	536.0	160.8	696.8
Cruise		0.0	150	10.0	0.0	92.0	0.0	0.0	251.5	75.5	327.0
Cruise Communications		0.0	150	193.5	0.0	92.0	0.0	0.0	435.0	130.5	565.5
Orbit Injection		0.0	150	193.5	0.0	92.0	0.0	0.0	435.0	130.5	565.5
Lunar Sunlit Communications		0.0	150	294.5	0.0	92.0	0.0	0.0	536.0	160.8	696.8
Lunar Eclipse Communications		0.0	150	193.5	0.0	92.0	0.0	0.0	435.0	130.5	565.5

3.3 Top Level Design (MEL and PEL)—Microwave LNS (Lagrange Orbit)

3.3.1 MEL—Microwave LNS (Lagrange Orbit)

Table 19 lists the top level of the MEL the Lagrange orbit case using optical communications as well as microwave. Figure 20 wraps up those total masses, CBE and total mass after applied growth percentage for the Lagrange orbit microwave communications case.

TABLE 19.—MEL—LAGRANGE ORBIT—MICROWAVE

WBS no.	Description LNS-HR (September 2008) Lagrange RF	Quantity	Unit mass, kg	Basic mass, kg	MGA, %	Growth, kg	Predicted mass, kg
6.0	Lunar Network Satellite-High Rate	-	-	2005.2	10	207.9	2213.1
6.0.1	RF Communications Package	-	-	204.1	22	44.1	248.2
6.0.1.1	Ka band (40/37) DTE Link	-	-	17.9	10	1.8	19.7
6.0.1.2	Ka band (23/26) Relay Link	-	-	15.4	10	1.5	16.9
6.0.1.3	Ka band (40/37) TT&C and Ranging	-	-	14.9	18	2.7	17.6
6.0.1.4	S band Relay Link	-	-	24.4	10	2.4	26.8
6.0.1.5	Data Router and SSR	-	-	4.0	10	0.4	4.4
6.0.1.6	Antennas (Booms excluded)	-	-	127.5	28	35.2	162.7
6.0.2	Optical Communications Package	-	-	0.0	0	0.0	0.0
6.0.2.1	Optical Head	-	-	0.0	0	0.0	0.0
6.0.2.2	Electronics Box	-	-	0.0	0	0.0	0.0
6.0.2.3	Optical Head-to-E.Box Interface	-	-	0.0	0	0.0	0.0
6.0.2.4	Interface to Spacecraft	-	-	0.0	0	0.0	0.0
6.0.3	C&DH	-	-	77.2	26	20.2	97.4
6.0.3.1	C&DH Hardware	-	-	74.2	25	18.5	92.7
6.0.3.2	Instrumentation & Wiring	-	-	3.0	55	1.7	4.7
6.0.4	GN&C	-	-	78.5	20	15.7	94.2
6.0.4.1	GN&C Hardware	-	-	78.5	20	15.7	94.2
6.0.5	Structures & Mechanical Systems	-	-	291.0	25	71.8	362.8
6.0.5.1	Primary Structures	-	-	248.3	25	62.1	310.4
6.0.5.2	Secondary Structures	-	-	9.4	25	2.4	11.8
6.0.5.3	Installation	-	-	18.3	25	4.6	22.9
6.0.5.4	Mechanisms	-	-	15.0	18	2.8	17.8
6.0.6	Power System	-	-	96.0	30	28.8	124.8
6.0.6.1	Battery System	-	-	43.0	30	12.9	55.9
6.0.6.2	Solar Array	-	-	35.0	30	10.5	45.5
6.0.6.3	Power Management & Distribution	-	-	18.0	30	5.4	23.4
6.0.10	Propulsion (Chemical)	-	-	18.7	10	1.9	20.5
6.0.10.1	Main Engine	-	-	9.3	10	0.9	10.3
6.0.10.2	Reaction Control System	-	-	9.3	10	0.9	10.2
6.0.11	Propellant Management (Chemical)	-	-	98.6	13	12.6	111.2
6.0.11.1	OMS Propellant Management	-	-	0.0	0	0.0	0.0
6.0.11.2	RCS Propellant Management	-	-	98.6	13	12.6	111.2
6.0.12	Propellant (Chemical)	-	-	1084.6	0	0.0	1084.6
6.0.12.1	Main Engine Propellant	-	-	0.0	0	0.0	0.0
6.0.12.2	RCS Propellant	-	-	1084.6	0	0.0	1084.6
6.0.13	Thermal Control (Non-Propellant)	-	-	56.5	23	13.0	69.5
6.0.13.1	Active Thermal Control	-	-	4.2	21	0.9	5.1
6.0.13.2	Passive Thermal Control	-	-	41.4	23	9.5	50.9
6.0.13.3	Semi-Passive Thermal Control	-	-	10.9	24	2.6	13.5

TABLE 20.—MEL—LAGRANGE ORBIT—MICROWAVE LNS

Spacecraft Master Equipment List Rack-up (Mass)				COMPASS S/C Design	
WBS	Main Subsystems	Basc (CBE) Mass (kg)	Growth (kg)	Predicted (Total) Mass (kg)	Aggregate Growth (%)
6.0	Lunar Network Satellie-High Rate	2005.2	207.9	2213.1	
6.0.1	RF Communications Package	204.1	44.1	248.2	22%
6.0.2	Optical Communications Package	0.0	0.0	0.0	TBD
6.0.3	Command & Data Handling (C&DH)	77.2	20.2	97.4	26%
6.0.4	Guidance, Navigation and Control System ((	78.5	15.7	94.2	20%
6.0.5	Structures & Mechanical Systems	291.0	71.8	362.8	25%
6.0.6	Power System	96.0	28.8	124.8	30%
6.0.7	Propulsion (Electric)	0.0	0.0	0.0	TBD
6.0.8	Propellant Management (EP)	0.0	0.0	0.0	TBD
6.0.9	Propellant (EP)	0.0			
6.0.10	Propulsion (Chemical)	18.7	1.9	20.5	10%
6.0.11	Propellant Management (Chemical)	98.6	12.6	111.2	13%
6.0.12	Propellant (Chemical)	1084.6			
6.0.13	Thermal Control (Non-Propellant)	56.5	13.0	69.5	23%
	Estimated Spacecraft Dry Mass	921	208	1129	23%
	Estimated Spacecraft Wet Mass	2005	208	2213	
	Estimated Spacecraft Inert Mass (for traj.)	952	276	1228.1	
System LeveL Growth Calculations					Total Growth
	Dry Mass Desired System Level Growth	921	276	1196.8	30%
	Mass Margin (carried at system level)		68		7%
	Total Wet Mass with Growth	2005	276	2281.4	
	Available Launch Performance to C3 (kg)			4710.0	kg
	Total Number of LNS-HR Satellites			2	
	Total mass of Two LNS-HR satellites			4562.7	
	Launch margin available (kg)			147.3	103

3.3.2 PEL—Microwave LNS (Lagrange Orbit)

The following requirements were used in sizing the power for the Lagrange orbit constellation using RF communications (Table 21).

- Power system tracks its own power
- Power system must provide:
    - ~1200 W (with 30 percent growth) during communications operations
    - ~800 W (with 30 percent growth) during S/C shadowing (seasonal) assuming Ka not used
- Generated thermal waste heat includes:
    - ~850 W (with 30 percent growth) during communications operations
    - ~650 W (with 30 percent growth) during S/C shadowing (seasonal) assuming Ka not used

TABLE 21.—PEL—MICROWAVE LNS-HR LAGRANGE ORBIT—RF

	Notional Time	Propulsion, W	Avionics, W	Comm. , W	Thermal, W	GN&C, W	Power , W	Science, W	CBE Total, W	30 % Margin, W	Total, W
Launch	10 m	20	79	20	3	48	0.0	0.0	170	51.12	222
S/C checkout	24 h	75	152	531	30	112	0.0	0.0	900	270.03	1170
Cruise	2 d	75	152	20	30	112	0.0	0.0	389	116.73	506
Cruise Communications		75	152	385	30	112	0.0	0.0	754	226.23	980
Orbit Injection	1 h	75	152	385	30	168	0.0	0.0	810	243.03	1053
Lunar Sunlit Communications	Continuous	75	152	531	30	112	0.0	0.0	900	270.03	1170
Lunar Eclipse Communications	2 h/d (max.)	75	152	203	60	112	0.0	0.0	602	180.63	783
Waste Heat											
Launch		0.0	79	10.0	0.0	32.0	150.0	0.0	270.7	81.2	351.9
S/C checkout		0.0	152	265.5	0.0	92.0	150.0	0.0	659.6	197.9	857.5
Cruise		0.0	152	10.0	0.0	92.0	150.0	0.0	404.1	121.2	525.3
Cruise Communications		0.0	152	192.5	0.0	92.0	150.0	0.0	586.6	176.0	762.6
Orbit Injection		0.0	152	192.5	0.0	92.0	150.0	0.0	586.6	176.0	762.6
Lunar Sunlit Communications		0.0	152	265.5	0.0	92.0	150.0	0.0	659.6	197.9	857.5
Lunar Eclipse Communications		0.0	152	101.5	0.0	92.0	150.0	0.0	495.6	148.7	644.3

3.4 Top Level Design (MEL and PEL)—Optical LNS (Lagrange Orbit)

3.4.1 MEL—Optical LNS (Lagrange Orbit)

Table 22 lists the top level of the MEL the Lagrange orbit case using optical communications as well as microwave. Table 23 wraps up those total masses, CBE and total mass after applied growth percentage for the Lagrange orbit optical communications case.

TABLE 22.—MEL—LAGRANGE ORBIT—OPTICAL

WBS no.	Description LNS-HR (September 2008) Lagrange , Optical	Quantity	Unit mass, kg	Basic mass, kg	MGA, %	Growth, kg	Predicted mass, kg
6.0	Lunar Network Satellite-High Rate	-	-	1915.4	11	206.3	2121.7
6.0.1	RF Communications Package	-	-	147.0	25	37.1	184.1
6.0.1.1	Ka band (40/37) DTE Link	-	-	0.0	0	0.0	0.0
6.0.1.2	Ka band (23/26) Relay Link	-	-	15.4	10	1.5	16.9
6.0.1.3	Ka band (40/37) TT&C and Ranging	-	-	0.0	0	0.0	0.0
6.0.1.4	S band Relay Link	-	-	12.6	10	1.3	13.9
6.0.1.5	Data Router and SSR	-	-	4.0	10	0.4	4.4
6.0.1.6	Antennas (Booms excluded)	-	-	115.0	29	33.9	148.9
6.0.2	Optical Communications Package	-	-	18.1	32	5.8	23.8
6.0.3	C&DH	-	-	77.2	26	20.2	97.4
6.0.3.1	C&DH Hardware	-	-	74.2	25	18.5	92.7
6.0.3.2	Instrumentation & Wiring	-	-	3.0	55	1.7	4.7
6.0.4	GN&C	-	-	78.5	20	15.7	94.2
6.0.4.1	GN&C Hardware	-	-	78.5	20	15.7	94.2
6.0.5	Structures & Mechanical Systems	-	-	289.3	25	71.3	360.6
6.0.5.1	Primary Structures	-	-	248.3	25	62.1	310.4
6.0.5.2	Secondary Structures	-	-	9.4	25	2.4	11.8
6.0.5.3	Installation	-	-	16.5	25	4.1	20.7
6.0.5.4	Mechanisms	-	-	15.0	18	2.8	17.8
6.0.6	Power System	-	-	96.0	30	28.8	124.8
6.0.6.1	Battery System	-	-	43.0	30	12.9	55.9
6.0.6.2	Solar Array	-	-	35.0	30	10.5	45.5
6.0.6.3	Power Management & Distribution	-	-	18.0	30	5.4	23.4
6.0.10	Propulsion (Chemical)	-	-	18.7	10	1.9	20.5
6.0.10.1	Main Engine	-	-	9.3	10	0.9	10.3
6.0.10.2	Reaction Control System	-	-	9.3	10	0.9	10.2
6.0.11	Propellant Management (Chemical)	-	-	98.6	13	12.6	111.2
6.0.11.1	OMS Propellant Management	-	-	0.0	0	0.0	0.0
6.0.11.2	RCS Propellant Management	-	-	98.6	13	12.6	111.2
6.0.12	Propellant (Chemical)	-	-	1035.6	0	0.0	1035.6
6.0.12.1	Main Engine Propellant	-	-	0.0	0	0.0	0.0
6.0.12.2	RCS Propellant	-	-	1035.6	0	0.0	1035.6
6.0.13	Thermal Control (Non-Propellant)	-	-	56.5	23	13.0	69.5
6.0.13.1	Active Thermal Control	-	-	4.2	21	0.9	5.1
6.0.13.2	Passive Thermal Control	-	-	41.4	23	9.5	50.9
6.0.13.3	Semi-Passive Thermal Control	-	-	10.9	24	2.6	13.5

TABLE 23.—MEL—LAGRANGE ORBIT—OPTICAL LNS

Spacecraft Master Equipment List Rack-up (Mass)				COMPASS S/C Design	
WBS	Main Subsystems	Basc (CBE) Mass (kg)	Growth (kg)	Predicted (Total) Mass (kg)	Aggregate Growth (%)
6.0	Lunar Network Satellie-High Rate	1915.4	206.3	2121.7	
6.0.1	RF Communications Package	147.0	37.1	184.1	25%
6.0.2	Optical Communications Package	18.1	5.8	23.8	32%
6.0.3	Command & Data Handling (C&DH)	77.2	20.2	97.4	26%
6.0.4	Guidance, Navigation and Control System ((	78.5	15.7	94.2	20%
6.0.5	Structures & Mechanical Systems	289.3	71.3	360.6	25%
6.0.6	Power System	96.0	28.8	124.8	30%
6.0.7	Propulsion (Electric)	0.0	0.0	0.0	TBD
6.0.8	Propellant Management (EP)	0.0	0.0	0.0	TBD
6.0.9	Propellant (EP)	0.0			
6.0.10	Propulsion (Chemical)	18.7	1.9	20.5	10%
6.0.11	Propellant Management (Chemical)	98.6	12.6	111.2	13%
6.0.12	Propellant (Chemical)	1035.6			
6.0.13	Thermal Control (Non-Propellant)	56.5	13.0	69.5	23%
	Estimated Spacecraft Dry Mass	880	206	1086	23%
	Estimated Spacecraft Wet Mass	1915	206	2122	
	Estimated Spacecraft Inert Mass (for traj.)	910	264	1173.8	
System LeveL Growth Calculations					Total Growth
	Dry Mass Desired System Level Growth	880	264	1143.7	30%
	Mass Margin (carried at system level)		58		7%
	Total Wet Mass with Growth	1915	264	2179.3	
	Available Launch Performance to C3 (kg)			4710.0	kg
	Total Number of LNS-HR Satellites			2	
	Total mass of Two LNS-HR satellites			4358.7	
	Launch margin available (kg)			351.3	246

3.4.2 PEL—Optical LNS (Lagrange Orbit)

- Power system tracks its own power
- Power system must provide (Table 24):
  - ~1200 W (with 30 percent growth) during communications operations
  - ~900 W (with 30 percent growth) during S/C shadowing (seasonal) assuming Ka not used
- Generated thermal waste heat includes:
  - ~850 W (with 30 percent growth) during communications operations
  - ~700 W (with 30 percent growth) during S/C shadowing (seasonal) assuming Ka not used

TABLE 24.—PEL—ORBITAL LNS-HR LAGRANGE ORBIT—OPTICAL

	Notional time	Propulsion, W	Avionics, W	Comm. , W	Thermal, W	GN&C, W	Power , W	Science, W	CBE Total, W	30 % Margin, W	Total, W
Launch	10 m	20	79	20	3	48	0.0	0.0	170	51.12	222
S/C checkout	24 h	75	152	531	30	112	0.0	0.0	900	270.03	1170
Cruise	2 d	75	152	20	30	112	0.0	0.0	389	116.73	506
Cruise Communications		75	152	337	30	112	0.0	0.0	706	211.68	917
Orbit Injection	1 h	75	152	337	30	168	0.0	0.0	762	228.48	990
Lunar Sunlit Communications	Continuous	75	152	531	30	112	0.0	0.0	900	270.03	1170
Lunar Eclipse Communications	2h/day (max)	75	152	294	60	112	0.0	0.0	693	207.93	901
Waste Heat											
Launch	10 m	0.0	79	10.0	0.0	32.0	150.0	0.0	270.7	81.2	351.9
S/C checkout	24 h	0.0	152	265.5	0.0	92.0	150.0	0.0	659.6	197.9	857.5
Cruise	2 d	0.0	152	10.0	0.0	92.0	150.0	0.0	404.1	121.2	525.3
Cruise Communications		0.0	152	168.3	0.0	92.0	150.0	0.0	562.4	168.7	731.1
Orbit Injection	1 h	0.0	152	168.3	0.0	92.0	150.0	0.0	562.4	168.7	731.1
Lunar Sunlit Communications	Continuous	0.0	152	265.5	0.0	92.0	150.0	0.0	659.6	197.9	857.5
Lunar Eclipse Communications	2h/d (max)	0.0	152	147.0	0.0	92.0	150.0	0.0	541.1	162.3	703.4

3.5 System Level Summary

3.5.1 Frozen Orbit RF (Microwave) Concept Configuration

The desire to launch two S/C in one launch without the use of a Dual Payload Attach Fitting (DPAF) was the primary driver in the sizing and layout of the S/C bus. By utilizing a thrust tube structure that extends beyond the faces of the S/C bus, two S/C can be stacked on top of one another, with only the need for a separation mechanism, allowing the launch and stacked loads to be carried primarily through the thrust tube. The stowed and stacked configuration of the Frozen Orbit RF S/C is shown back in Figure 25.

The 1.1 m diameter flange on the propellant tank drove the diameter of the thrust tube. The height was selected to completely enclose the propellant and pressurant tanks, allow the main thrusters to stay outside of the S/C yet allow for stacking in the fairing, and to maintain the proper clearances near the Launch Vehicle Adaptor (LVA). A conical S/C adaptor is required to mate the thrust tube to the standard LVA interface. The overall bus size was a resultant of the thrust tube dimensions, and the ability to completely enclose all internal components within the bus. These subsystem components are mounted on structural panels that radiate from the thrust tube diagonally out the bus edges, and are grouped together by subsystem to allow for easier assembly. Layout of the bus structure and subsystem components is show in Figure 34 and Figure 35. Overall deployed dimensions are shown in Figure 36.

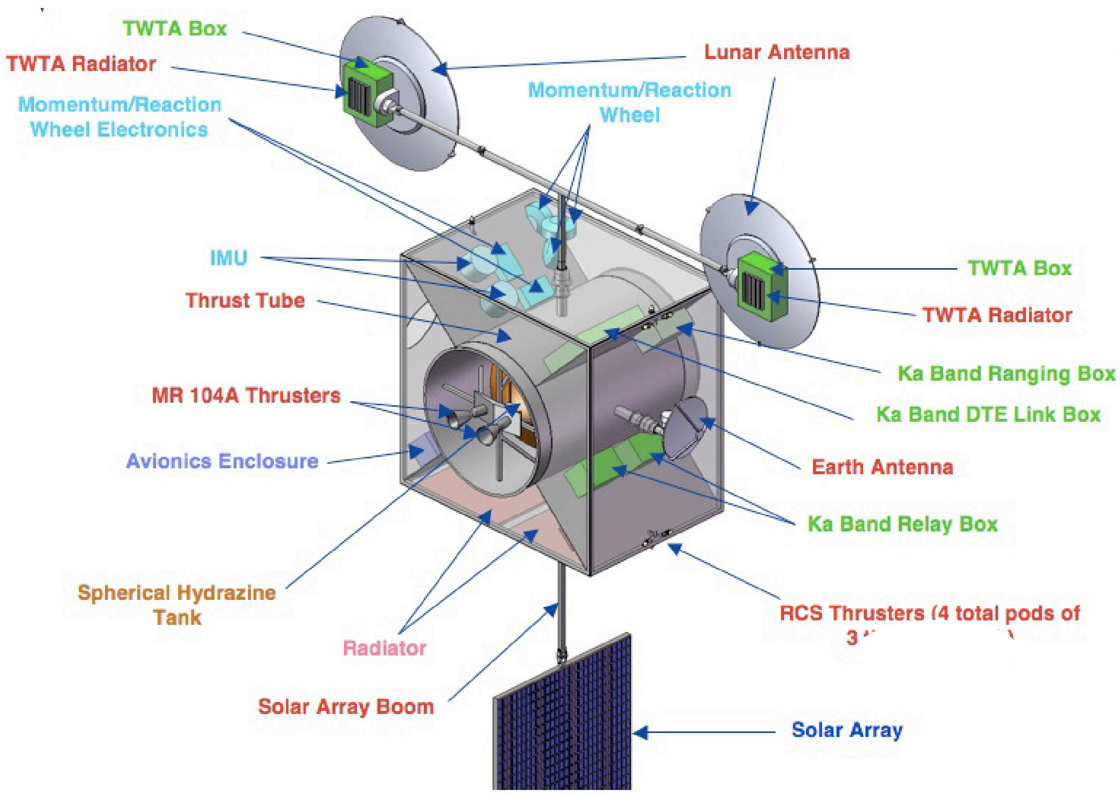


Figure 34.—Frozen orbit RF S/C components.

Locations of the lunar antennas, Earth antenna, SA, and radiators were driven by the requirement to have 24-hr communication capabilities between either of the two S/C in the constellation and Earth, lunar outpost at the South Pole, another lunar sortie site, as well as the ability to track the Sun with the arrays. In order to minimize gimbal requirements for tracking, the Earth antenna was placed on the side of the S/C bus that is always pointed back towards Earth. This "fixed" orientation resulted in minimum two-axis gimbal angle requirements for the Earth antenna and allowed the SA to have a single axis gimbal since the Sun and Earth are essentially in the same plane. The array is located on a boom extended from the face of the S/C perpendicular to the face containing the Earth antenna, and opposite the face which contains the two lunar pointing antennae, to allow for the single axis Sun tracking. The radiators are located on the face containing the SA boom to ensure they will maintain a view that is perpendicular to both Sun and Earth thus increasing their effectiveness in thermal rejection. A T-shaped boom is used for the two lunar antennas to allow for proper tracking of the lunar surface. With the antennas located on each end of the T-shaped boom, the entire boom can utilize a single axis gimbal rotating 360° per orbital period to ensure that both antennas are pointed towards the lunar surface. Each antenna then has a two-axis gimbal to allow each to track two separate locations on the lunar surface. The Traveling Wave Tube Amplifier (TWTA) for each lunar antenna is mounted to the back of the dish and contains its own radiator for heat rejection. Locations of all components are shown in Figure 34 and Figure 35.

Star trackers are internally mounted and have a view out of the remaining open face of the S/C. They are oriented in such a way that they are pointed 90° from each other and angled 45° from the face of the bus in the direction of the SA. This orientation ensures that the star trackers never have a direct view of the Sun, Earth, or lunar surface. Figure 35 illustrates the location and orientation of the star trackers.

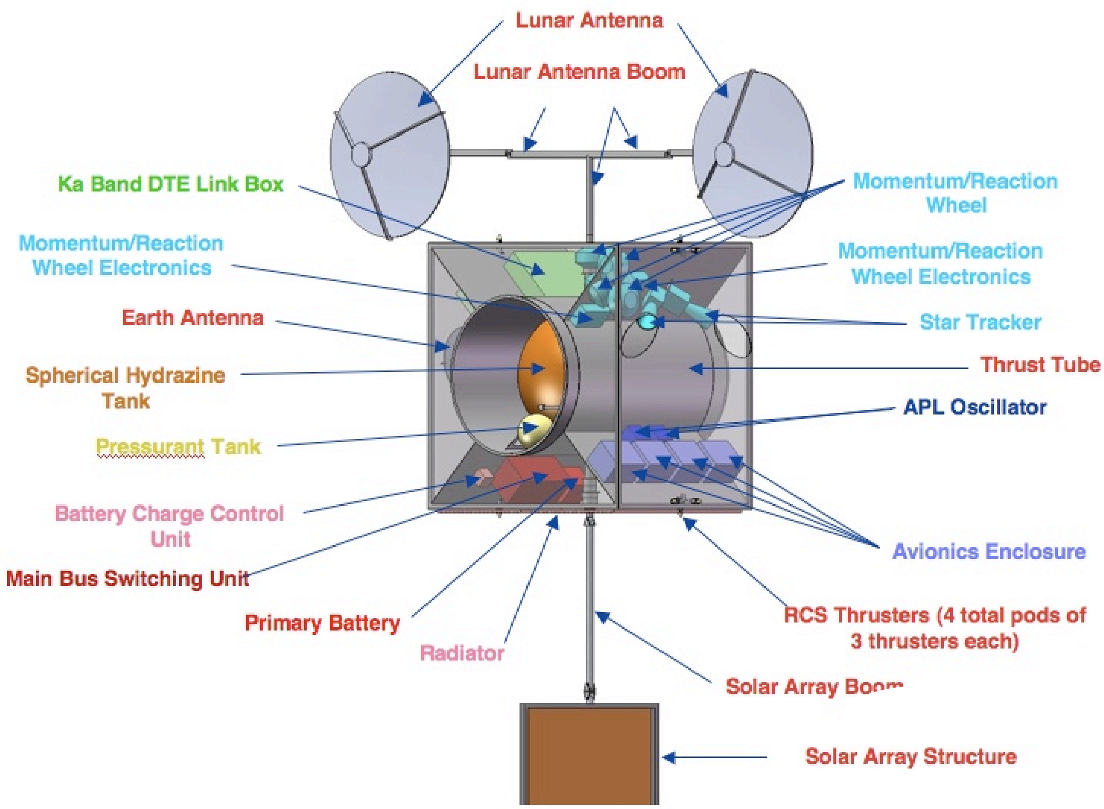


Figure 35.—Frozen Orbit RF S/C components.

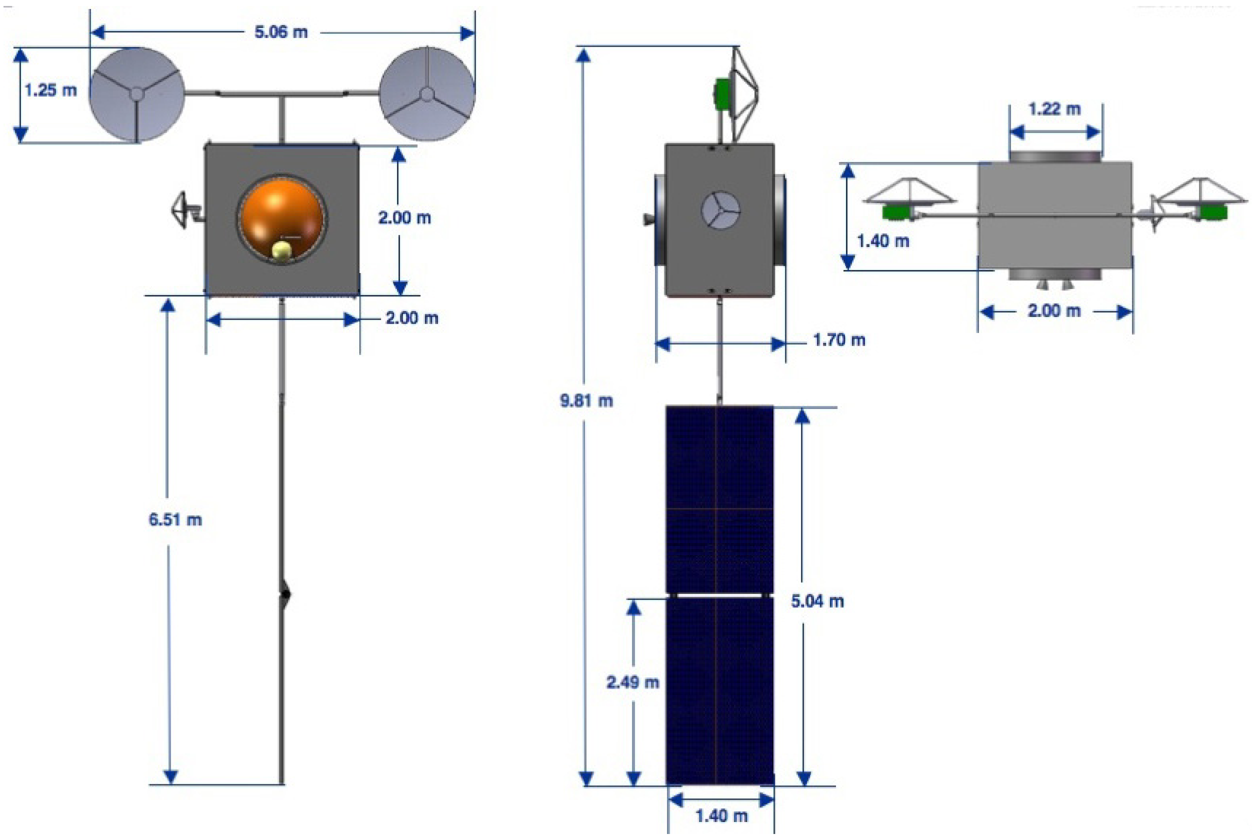


Figure 36.—LNS RF frozen orbit S/C dimensions.

3.5.2 Lagrange Orbit RF Concept Configuration

Similarly to the Frozen Orbit configuration, a thrust tube was implemented in the bus design to allow for stacking two S/C in the payload fairing without the use of a DPAF. Again, the diameter of the thrust tube was driven by the propellant tank diameter, while the length was increased to include an additional propellant tank. All internal components were mounted in the same fashion as the Frozen Orbit configuration.

With the S/C in a Lagrange Orbits (one about L_1 and one about L_2, see Figure 22 in Section 2.6.3 timeline), the Moon and Earth are located on opposite sides of the S/C. This dictated the location and pointing of both lunar antennas and the Earth antenna. The lunar antennas are located on booms placed on opposite sides of the S/C to help balance out the forces from solar pressure considering their large diameter. Each has a two-axis gimbal to allow for accurate and individual pointing. The Earth antenna is pointed opposite of the lunar antennas and has a two-axis gimbal for accurate pointing. The SAs are located on booms perpendicular to the lunar antenna booms, and perpendicular to the direction of the Earth antenna, again to allow for single axis tracking of the Sun, which is essentially in the same plane as the Earth. In this case, the arrays are located on two separate booms on opposite sides of the bus. This location was selected to help balance out the force from the solar pressure. Dimensions of the fully deployed Lagrange Point S/C are shown in Figure 37.

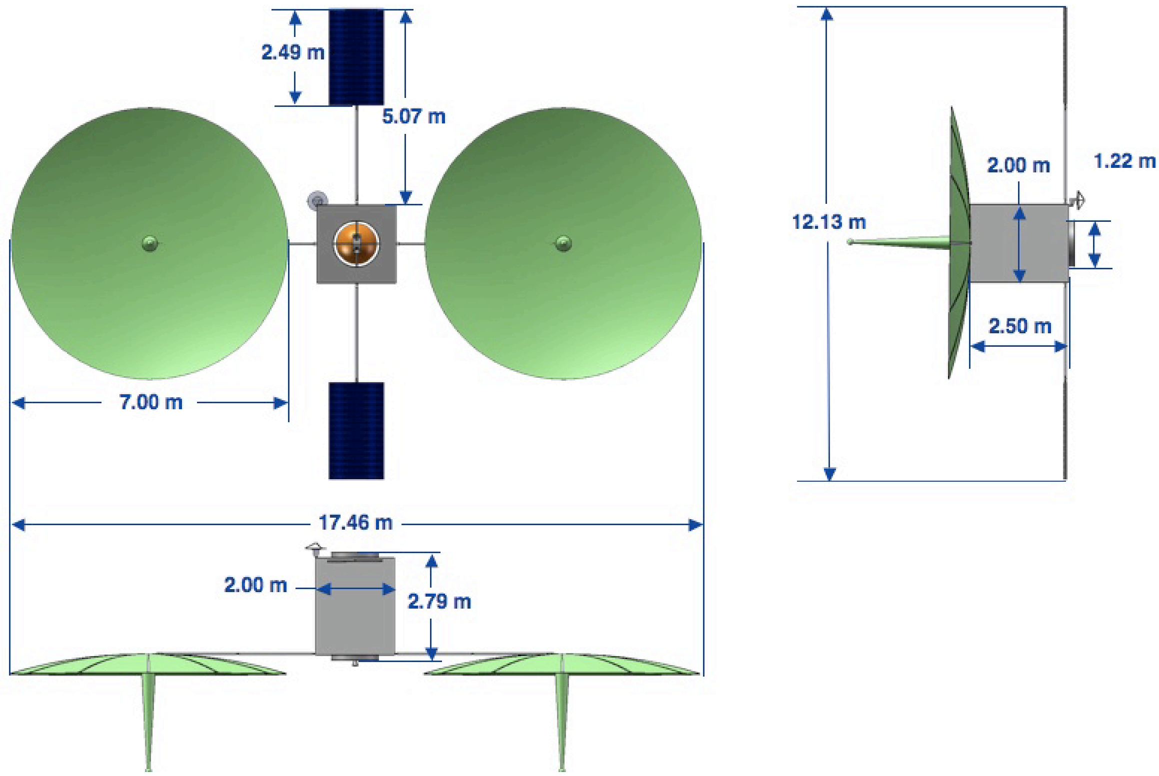


Figure 37.—LNS RF Lagrange orbit S/C dimensions.

4.0 Conclusions, Lessons Learned, and Areas for Future Study

4.1 Conclusions

Both the Frozen orbit and Lagrange orbit S/C can be dual launched on an atlas class S/C. The Lagrange S/C is much larger due to its large antennas and larger station keeping requirements. The antennas are very large solely based upon the 1 W suit radio and the distance to the Lagrange point. This points to making the frozen orbit constellation preferred over the Lagrange point network.

A trade of laser communications for the Earth link was made. The pointing accuracy of the S/C was made the same as the RF option by utilizing larger ground telescopes. Thus the differences in the S/C design were minimal. A complete trade including the ground stations would need to be made to demonstrate the advantages and disadvantages of the laser communications.

The frozen orbit S/C was half the height of the Lagrange S/C due to much less propellant (much smaller station keeping requirements). The Frozen orbit S/C also requires a much smaller lunar antenna (1.2 m versus 7 m) due to much lower lunar orbit.

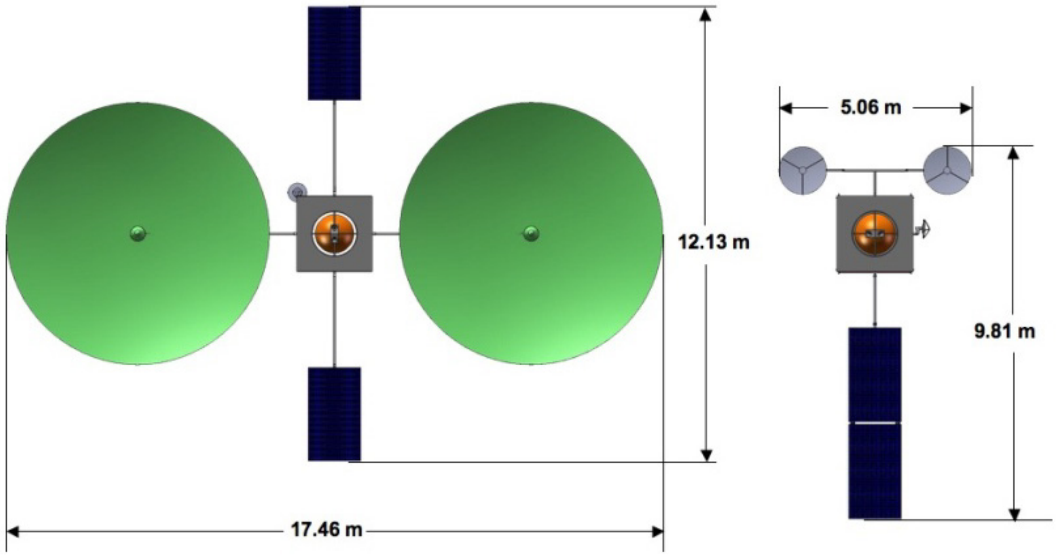


Figure 38.—LNS RF Lagrange versus frozen S/C dimensions—Front view.

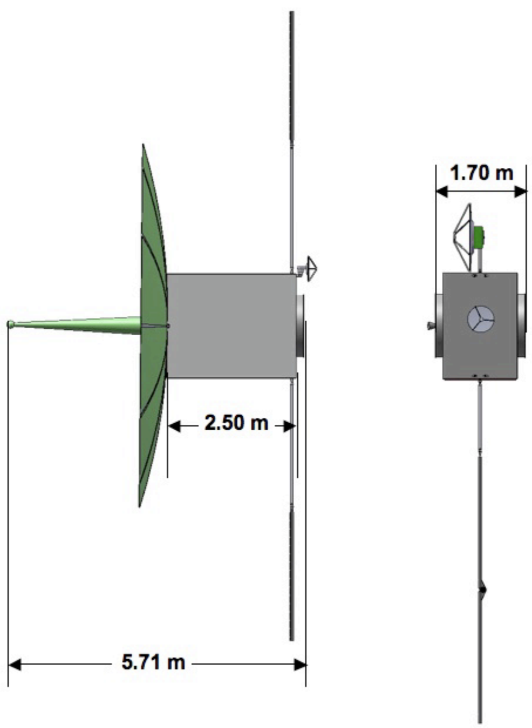


Figure 39.—LNS RF Lagrange versus frozen S/C dimensions—Side view.

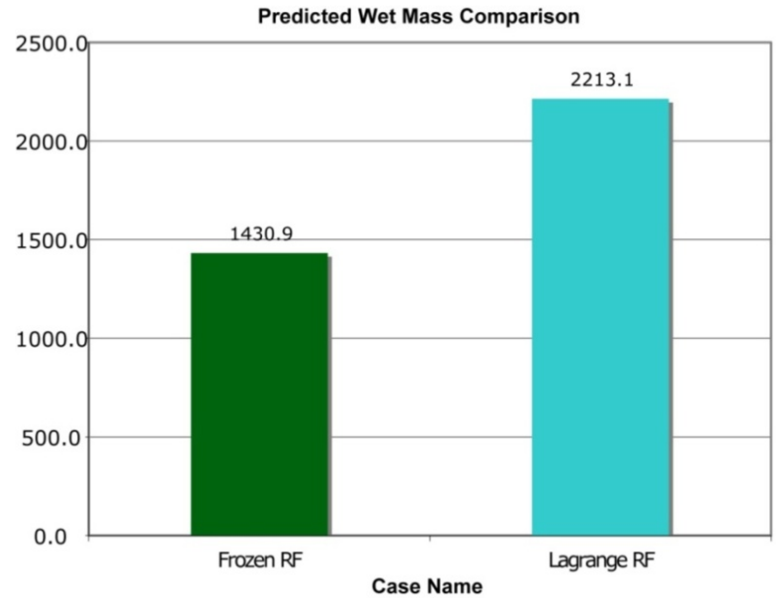


Figure 40.—LNS RF frozen versus Lagrange S/C predicted wet mass.

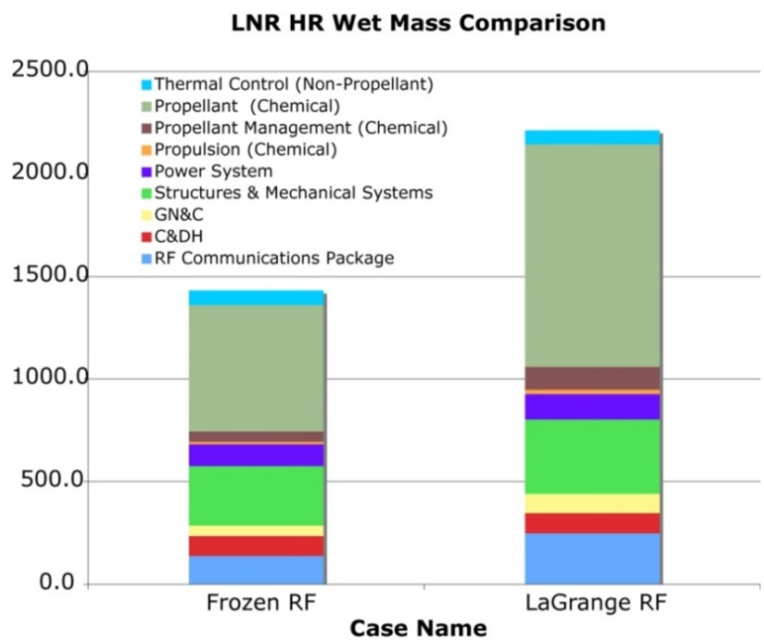


Figure 41.—LNS RF frozen versus Lagrange S/C predicted wet mass by subsystem.

4.2 Areas for Future Study

The following areas were identified during this study as future work or other items that should be examined in more detail.

- Trajectory analysis on actual halo Lagrange orbit in order to model the RCS propellant loading and pointing rather than an assumed orbit.
- Perform Risk and Reliability analysis of the Frozen orbit S/C (both RF and Optical).
- Refine the Frozen orbits and S/C design aspects

- Verify the SOAP frozen orbit 12-yr perturbations with SNAP high fidelity integration code (orbital modeling).

5.0 Subsystem Breakdown

5.1 Microwave Communications

5.1.1 Microwave Communications Requirements

All options must provide 100 percent 24/7 coverage to outpost, 7 day of both Sunlight and communications coverage at other sortie sites, and voice and global coverage as necessary. DTE communication for voice, available 24/7, and LNS using Ka Band will have about 28 min of communication time per orbit.

5.2 Microwave Communications Assumptions

Links

- LNS to Earth: Ka band (400 Mbps mono, 800 Mbps dual), S-band (150 kbps)
- LNS to outpost: S-Band (150 kbps), Ka band (400 Mbps,1200 Mbps)
- LNS to surface links: Ka 23/26 (25 Mbps) and Telemetry, Tracking and Command (TT&C) S-band, processing for communication/navigation in payload, storage in C&DH

LNS-HR Microwave Trunk Line

- LNS-HR to Earth link (Ka-band)
  - Trunk data rate: 400 Mbps per polarization
  - Spectrum: 37/40 GHz
  - Bandwidth: 500 MHz
  - Modulation: QPSK
  - Coding: Reed-Solomon or LDPC (TBR)
  - Code rate: ½ (TBR for each orbit)
  - Trunk antenna size: 0.5 m (TBR)
  - Bit error ratio (BER): 10×10^{-8}
- LNS-HR to Earth link (S-band)
  - Data Rate: 150 kbps

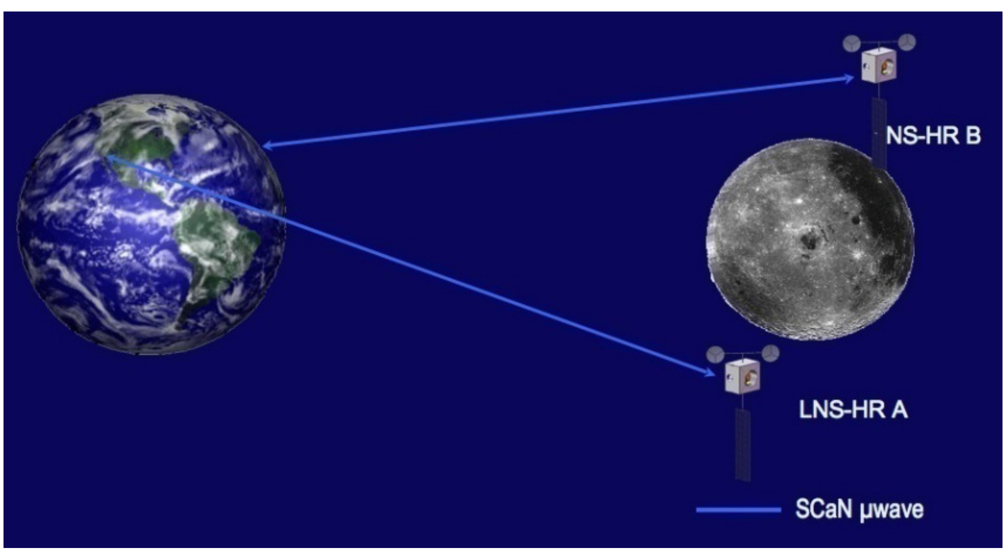


Figure 42.—Frozen orbit communication trunk line.

Earth-Based Ground Systems

- Antennas: 18 m
- Locations: DSN sites (Goldstone, Canberra, Madrid)

TT&C of LNS-HR From Earth

- Radiometrics for position/velocity knowledge using S-band
- Star tracking for orientation

General RF Aspects

- The highest modulation allowed for S-band and K/Ka-band is a form of quadrature phase-shift keying (QPSK).
- The forward error correction code is Reed-Solomon with Viterbi rate ½ (204, 255).
- The maximum K/Ka-band data rate is 400 Mbps uncoded.
- The maximum S-band data rate is 150 kbps uncoded.
- The digital part of the radio is software designed radio with an average power need of 37.5 W
- The high power amplifier is a TWTA for RF transmit power of 10 W or more, and solid state for less than 10 W of power.
- The efficiency of the TWTA is 40 percent for transmit powers less than 25 W and 50 percent for transmit power more than 25 W.
- The efficiency of the solid-state is 10 percent for powers greater than 4 W and 20 percent for power less than 5 W. (This reflects that above 4 W one generally needs to do power combining to achieve the power needed and below 4 W on three-stage power amplifier is all that is needed.)
- The mass and size of the TWTA does not change with power level up to several watts of power.
- The LRS be a full network node with a high-speed router, and memory for store-and-forward and for file serving.

Lunar Links

- The links from the surface of the Moon to the LNS in terms of frequency and protocols are to be the same as from the Moon to Earth.
- The size of the antenna is sized to receive 8 kbps from the EVA suit. The maximum transmit power for the EVA suit is 1 W and the EVA suit uses a simple dipole antenna. The suits transmit linearly polarized signals and the LRS receives circular polarized signals. (Increase margin from 3 to 6 dB.)
- The size of the antenna on the lunar communication terminal is 1 m.
- The size of the antenna on the rover is 0.3 m.
- The two lunar coverage antennae can be backup of each other; therefore additional redundant RF components are not necessary

Earth Links

- The antennae looking down to Earth is a dual band capability receiving at 25.5 to 27 GHz and at 2.20 to 2.29 GHz, and receiver at 23 to 23.5 GHz and at 2.00 to 2.09 GHz
- The system noise temperature of the antenna and the LNA system is 306 K at 25.5 GHz and 139 K at 2.2 GHz. (This is based on a G/T of 26.5 dB/K for the 25.5 to 27 GHz band, and a G/T of 28.8 dB/K for the LRS 18.3 m antenna.)

5.2.1 Microwave Communications Design and MEL

The RF signals at intermediate frequency (IF) will be used to modulate individual optical wavelengths. The IF frequency is below 5 GHz and may differ for the K/Ka band, Ka band, and S-band signals. These signals will be multiplexed together and sent up a single optical fiber. A space qualified optical fiber rotary joint will be used in each gimbal assembly. For the antenna systems looking down at the lunar system with two antennas, the optical wavelength at the “T” will be split with the appropriate wavelength going to each antenna. At the antenna, the signals will be de-multiplexed and converted back to the IFs. These intermediate frequencies will be upconverted to the frequencies needed.

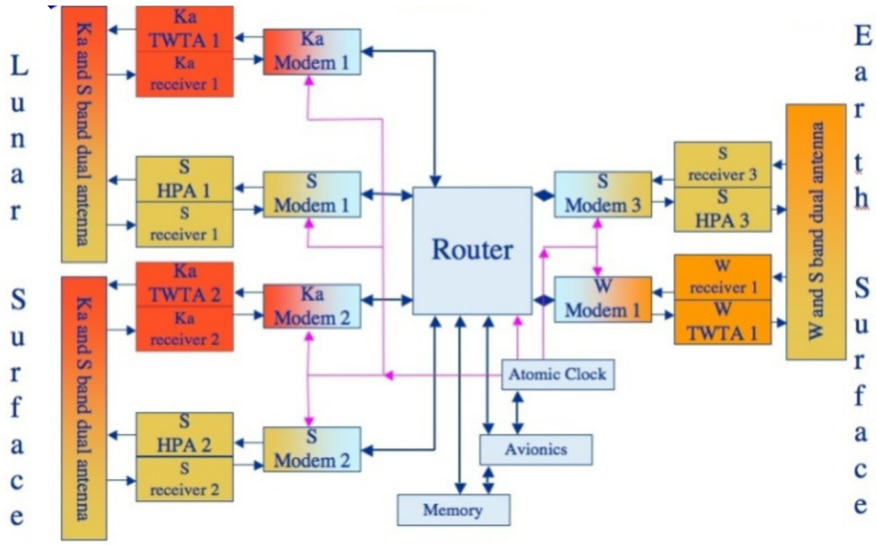


Figure 43.—Microwave communications flow diagram.

5.2.2 Microwave Communications Analytical Methods

5.2.2.1 Lunar Communication From Frozen Orbit to Lunar Surface

Table 25 gives the power needed to complete the links for the different forward error correction technique. As you can see the transmitted power is the same for the two forward error correction techniques, the reason being, the antenna size was adjusted to give the same bit error rate.

The antenna size to give a gain of 28.5 dBi for the receive frequency of 2.2 is 1.56 m, while for 26.2 dBi it is 1.2 m. These antenna sizes are chosen to allow astronauts on EVA to be able to talk using voice over IP to the satellite.

TABLE 25.—LRS TO ELLIPTICAL ORBIT LINK BUDGET—LPDC

LRS in Elliptical Orbit d = 12400 km LPDC								
Direction	Rx = 26 GHz Tx = 23 GHz for LRS				Rx = 2.2 GHz Tx = 2.0 GHz for LRS			
	Gain, db		Data rate, Mbps	Tx power, W	Gain, db		Data rate, Mbps	Tx power, W
	Tx	Rx			Tx	Rx		
LCT→DTE	46.1	71.4	200	7.4	24.7	50	150/3000	0.22/4.4
Rover→DTE	35.6	71.4	200	9.0	14.2	50	150	2.5
EVA→LRS					1	26.2	8	1
LRS→EVA					25.4	1	150	14
LRS→ LCT	46.7	45	100	0.65	25.4	23.8	3000	0.98
LRS→Rover	46.7	34.6	10	0.72	25.4	13.4	3000	10.9

TABLE 26.—LRS TO ELLIPTICAL ORBIT LINK BUDGET—VITERBI

LRS in Elliptical Orbit d = 12400 km Viterbi								
Direction	Rx = 26 GHz Tx = 23 GHz for LRS				Rx = 2.2 GHz Tx = 2.0 GHz for LRS			
	Gain, db		Data rate, Mbps	Tx power, W	Gain, db		Data rate, Mbps	Tx power, W
	Tx	Rx			Tx	Rx		
LCT→DTE	46.1	71.4	200	7.4	24.7	50	150/3000	0.22/4.4
Rover→DTE	35.6	71.4	200	9.0	14.2	50	150	2.5
EVA→LRS					1	28.5	8	1
LRS→EVA					27.7	1	150	14
LRS→ LCT	48.9	45	100	0.65	27.7	23.8	3000	0.98
LRS→Rover	48.9	34.6	10	0.72	27.7	13.4	3000	10.9

When choosing the power of the transmitter on LNS to communicate with the surface, chose the maximum power needed. As one can see the power needed to communicate with the rover is the maximum amount of power. This assumes the LCT has a 1 m antenna on it and the rover a 1/3 m antenna.

The transmit data rate from LNS to the LCT and rover is estimated but needs to be reviewed. The transmit frequency is 23 GHz. The data rate from LRS to LCT is 100 Mbps. This is data rate is higher than that for the International Space Station (ISS) and will support the activities in the habitat and in the habitat area. The data rate from LNS to a rover away from the habitat area is 10 Mbps. For clarity purpose the needed power to close those links when the temperature of the lunar surface at the equator is 425 K is shown for both the LCT and the rover. But in building the satellite, the large transmit power at 23 GHz is chosen.

Lunar Communication From Lagrange Orbit to Lunar Surface

Going out to the L_2 point and allowing the astronauts to communicate at 8 kbps only changes the antenna size on the S/C if the losses between the antenna and the LNA and high power antenna (HPA) remains constant. The antenna size for the Viterbi FEC case is 9.06 m in diameter and for the LPDC (low-density parity check) the size is 6.95 m in diameter (Table 27). All the output power from the HPAs at either the L_1 and L_2 halo orbits as well as for the elliptical frozen orbit on the lunar surface assets remain the same.

TABLE 27.—LRS TO LAGRANGE ORBIT LINK BUDGET—LPDC AND VITERBI

LRS in L_2 Orbit d = 7202400 km LPDC								
Direction	Rx = 26 GHz Tx = 23 GHz for LRS				Rx = 2.2 GHz Tx = 2.0 GHz for LRS			
	Gain, db		Data rate, Mbps	Tx power, W	Gain, db		Data rate, Mbps	Tx power, W
	Tx	Rx			Tx	Rx		
LCT→DTE	46.1	71.4	200	7.4	24.7	50	150/3000	0.22/4.4
Rover→DTE	35.6	71.4	200	9.0	14.2	50	150	2.5
EVA→LRS					1	41.5	8	1
LRS→EVA					40.7	1	150	14.0
LRS→ LCT	61.9	45	100	0.65	40.7	23.8	3000	0.69
LRS→Rover	61.9	34.6	10	0.72	40.7	13.4	3000	10.9
LRS in L_2 Orbit d = 72000 km Viterbi								
Direction	Rx = 26 GHz Tx = 23 GHz for LRS				Rx = 2.2 GHz Tx = 2.0 GHz for LRS			
	Gain, db		Data rate, Mbps	Tx power, W	Gain, db		Data rate, Mbps	Tx power, W
	Tx	Rx			Tx	Rx		
LCT→DTE	46.1	71.4	200	7.4	24.7	50	150/3000	0.22/4.4
Rover→DTE	35.6	71.4	200	9.0	14.2	50	150	2.5
EVA→LRS					1.0	43.8	8	1.0
LRS→EVA					43.0	1.0	150	14.0
LRS→ LCT	64.2	45.0	100	0.65	43.0	23.8	3000	0.69
LRS→Rover	64.2	34.6	10	0.72	43.0	13.4	3000	10.9

In conclusion on the power needed by the HPA for 23 GHz is at least 72 W while for the 2.0 GHz HPA the power needed is 14 W. In all cases the power can be adjusted if the assumed data rates change (Table 28).

TABLE 28.—LUNAR COMMUNICATION LINK BUDGET ASSUMPTIONS AND CALCULATIONS

	Vertibi suit to LRS	LPDC suite to LRS
Tx frequency, GHz	2.2	2.2
Tx wavelength, $\lambda = c/f$, mm	136.27	136.27
Antenna efficiency, η, percent	55	55
Efficiency loss, $L_\eta = 10\log_{10}(\eta/100)$, dB	2.60	2.60
Tx antenna diameter, D_{TX}	0.30	0.30
Tx antenna half-power beamwidth, $\alpha(°) = 72\lambda/(D_{TX})$, degree	**32.70**	**32.70**
Tx antenna gain, $20\log_{10}[\pi(D_{TX})/\lambda] - L_\eta$, dBi	1.00	1.00
TWTA output, W	1.00	1.00
TWTA out, dB	0.00	0.00
Output power delivery loss, dB	1.5	1.5
TWTA out, W	0.71	0.71
Tx antenna feedhorn output power, dBW	–1.50	–1.50
Pointing (gimbal) loss, dB	0.1	0.1
EIRP, dBW	–0.60	–0.60
Path length/range, R_b, km	12,400	12,400
Path loss, $P_L = 20\log_{10}[4\pi(R_b)/\lambda]$, dB	181.16	181.16
Receive pointing (gimbal) loss, dB	0.1	0.1
Rain fade, **26 GHz**, dB	0.00	0.00
Rain fade, **23 GHz**, dB	0.00	0.00
Received power, dBW	–181.86	–181.86
Rx illuminating flux density, dBW/m^2	–153.46	–153.46
Rx antenna, D_{RX}	1.56	1.20
Rx antenna gain, $20\log_{10}[\pi(D_{RX})/\lambda] - L_\eta$, dBi	28.52	26.22
Rx antenna half-power beamwidth, $\alpha(°) = 72\lambda/(D_{RX})$	**6.29**	**8.20**
Rx LNA gain, dB	23	23
Rx LNA noise figure, **NF**, (dB) wrt To=290 K, dB	0.00	0.00
Rx LNA noise temperature, $T_{\mathrm{LNA}} = (F - 1)*To$, K	0.00	0.00
Rx diplexer loss, dB	0.00	0.00
Rx waveguide/mismatch loss, dB	0.00	0.00
Rx waveguide temperature, T_L, K	0.00	0.00
SLR 1 m Rx antenna (26 GHz) at Equator, K	0.00	0.00
SLR 1 m Rx antenna (2.26 GHz) at South Pole, K	425	425
Antenna galactic noise temperature, T_{GAL}, K	0.00	0.00
Antenna resistive loss, $T_{\mathrm{A,RES}}$ (0.4 emissivity), K	0.00	0.00
Combined input loss in front of LNA, dB	0.00	0.00
Input loss (lumped lossy elements), L	1.00	1.00
$(L - 1)*T_L$, K	0.00	0.00
LNA noise contribution, $L*T_{\mathrm{LNA}}$, K	140.00	140.00
System temperature, $T_{\mathrm{SYS}} = T_{\mathrm{A,RES}} + T_{\mathrm{GAL}} + T_L + L*T_{\mathrm{LNA}}$, K	373.75	373.75
G/T (dBi/K), dB/K	2.80	0.49
Boltzmann's constant, k, dBW/K-Hz	–228.60	–228.60
No = $k*T_{\mathrm{SYS}}$, dBW/Hz	–202.88	–202.88
Power level at Rx antenna feedhord, dBW	–153.34	–155.64
Rx carrier/no, dBHz	49.53	47.23
Rx bit rate, Mbps	0.008	0.008
Rx noise bandwidth, B = 2.0*Rx bit rate, MHz	0.016	0.016
Rx noise BW, dBHz	42.04	42.04
Input noise, $N = k*B*T$, dBW	–160.83	–160.83
LRS input SNR, C/N, dB	7.49	5.19
Required Eb/No OQPSK 10^{-8} BER, dB	4.5	2.2
Required SNR OQPSK 10^{-8} BER, dB	1.49	–0.81
Link margin, dB	6.00	6.00
Required link margin, dB	6.00	6.00
Excess, dB	0.00	0.00

5.2.3 Microwave Communications Risk Inputs

5.2.4 Microwave Communications Recommendation

The Lagrange orbit configuration levies large requirements for communications distances from the Moon to the LNS as well as from the LNS at L_2 to the Earth.

5.3 Optical Communications

5.3.1 Optical Communications Requirements

All options must provide 100 percent 24/7 coverage to outpost, 7-day of both Sunlight and communications coverage at other sortie sites, and voice and global coverage as necessary. DTE communication for voice, available 24/7, and LNS using Ka Band will have about 28 min of communication time per orbit.

5.3.2 Optical Communications Assumptions

- Data rates
  - Downlink from LNS-HR: 1.2 Gbps (TBR)
  - Uplink to LNS-HR: 25 Mbps (TBR)
- Packet loss rates: 1×10^{-9} (from the NASA Jet Propulsion Laboratory (JPL))
- Size and mass
  - Mass: ~ 20 kg (from JPL)
    - Includes entire package, e.g., pointing mechanisms, tracking, etc.
    - Does not include redundancy
    - For single-fault tolerance—redundant diodes, redundant amplifier chain, backup processor
  - Physical size and dimensions: TBR
    - (6 in. diameter cylinder—4 in. pure aperture)
    - Electronics box
      - Processors, lasers, command and control, I/F to S/C bus
      - Optical fiber and electrical connections to transceiver
- GEO Optical Relay
  - Station locations: GEO
  - Aperture sizes: 50 cm (TBD)
    - Arrayed versus single aperture
  - Transmit power: TBD
  - Photon counting receivers: TBD
    - Signaling format: PPM
  - Tracking of LRS: TBD
  - Near Sun pointing: TBD
  - Satellite station keeping
    - Platform stability and attitude for acquiring and pointing optical link
  - RF between Earth and optical relay
  - Data handling, archiving, and distribution due to high data rates
  - Some outages with one optical relay

5.3.3 Optical Communications Design and MEL

- Power
  - Total power allocation: 40 to 60 W (from JPL)
  - Survival power: TBD
  - Standby power: TBD

    - Transmit power: ~ 1 to 2 W radiated power (TBR)
- Power efficiency:
    - Laser: 8 to 10 percent (typical) for 1550 nm (from JPL)
    - Overall system: TBD
- Optical transceiver
    - Aperture size: 5 to 10cm (from JPL)
    - Receive/transmit isolation: TBD
    - Receive/transmit wavelength: ~1550 nm with 30 nm separation between receive and transmit (from JPL)
    - Tracking sensor
    - Uplink detector
    - Mounted on two-axis gimbal
    - Fine pointing control
    - Solar protection filter

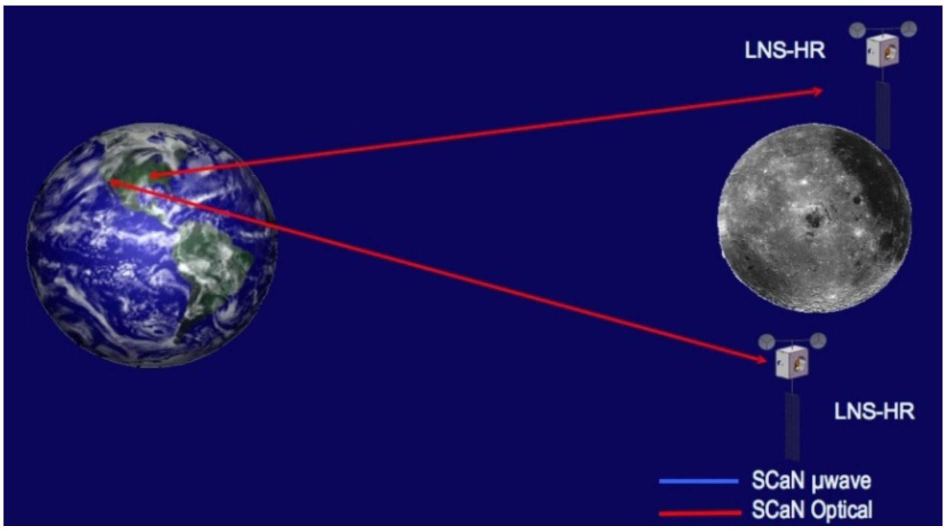


Figure 44.—LNS optical payload.

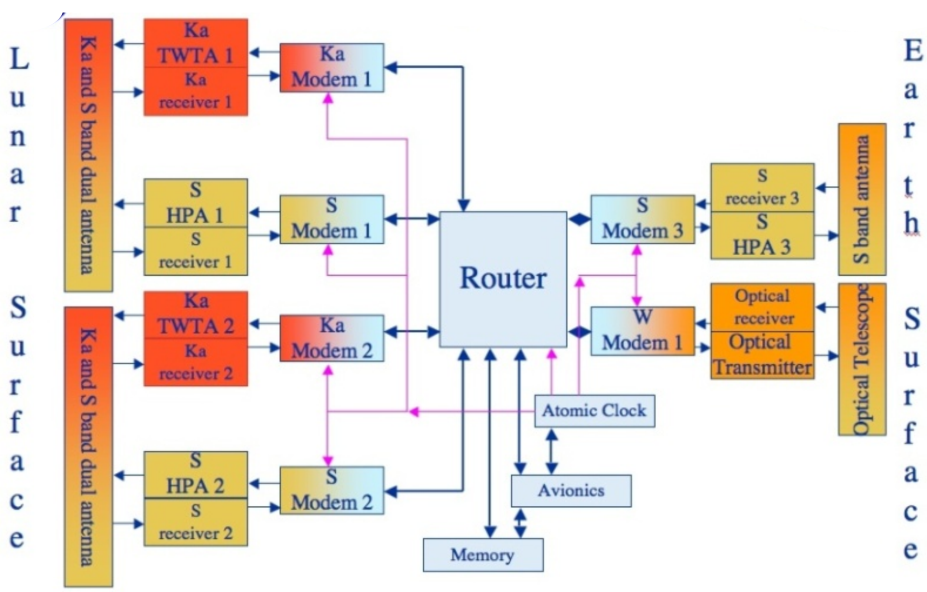


Figure 45.—Optical communication flow diagram.

5.3.4 Optical Communications Trades

The use of the optical communications system itself was the trade performed.

5.3.5 Optical Communications Analytical Methods

Optical communications system designed from requirements and all details are experimental.

5.4 Command and Data Handling (C&DH)

The C&DH system is also often referred to as the Avionics system.

5.4.1 C&DH Requirements

- Storage array for 24 hr of storage (0.38 TB or better)
- Avionics for systems command, control, and health management
- Use of highly stable oscillators in conjunction with atomic clocks
- Functionality of the Avionics Systems is provided on a following chart
- S/C GN&C—includes interfacing with IMUs, star trackers, and Sun sensors

5.4.2 C&DH Assumptions

- Single fault tolerant avionics
- Storage array will use next generation radiation hardened storage, 128 GB or better per card
- 100 krad avionics provides operation for ~ 12 yr
- Cabling is estimated as 10 percent of the avionics hardware
- All spares are cold spares, except data recorder, which is a hot spare

5.4.3 C&DH Design and MEL

Model Summary

- One FT redundant system design
- Routine avionics cabling
- Growth factor chosen based on AIAA table
- Each C&DH Processor can access up to 32 (TBR) channels of I/O based on presently available space rated hardware

Avionics Box contains a radiation tolerant PowerPC 750 processor and storage card for general LNS command and control. A communications card is included for communication with the LNS router. The package includes any necessary DC-DC converters, filter, and EMI shielding. There are two independent avionics strings for single fault tolerance. All avionics components based on COTS components. All avionics assume 3U-160 form factor cards.

An ultra-stable oscillator (USO) is included in the avionics package. There are two independent USOs for single fault tolerance. Oscillator drift nominally $< 7\times10^{-13}$/day.

Figure 46.—General Avionics Processor.

A rubidium atomic clock is included in the avionics package. Clock drift assumed at nominally 5×10^{-14}/d.

Data Recorder is based on 128 GB or better storage cards. A processor is included for managing the storage array. A comm. card is included for communications to the LNS router.

- General Avionics Processor (Figure 46)
  - System initialization
  - Antenna deployment
  - Antenna positioning
  - SA deployment
  - SA positioning
  - Satellite navigation—includes interfacing with IMUs, star trackers, and Sun sensors
  - Satellite guidance
  - Propulsion system control
  - Systems health and status management
  - Power management, control, distribution, and load shedding
  - Battery regulation and management
  - Thermal system management – includes control of pumps, valves, and heaters
  - System fault detection and correction
  - Time synchronization via atomic clock
  - Time stamping
  - Router management
  - Communications system management
- Data Recorder
  - Storage array monitoring and health management, including fault detection and correction
  - Synchronization of data between redundant data recorders
  - Buffering of data
  - Power down during eclipses if necessary
  - Risk assumed: technology advancements will have progressed far enough to permit 128 GB or better per PMC card
- Time generation unit
  - System time generation, used by the General Avionics Processor for time synchronization

Both an atomic clock and a time generation unit were used in the design per recommendations from JPL and the Optical Communications system engineers.

The components mentioned above of the C&DH system are listed in Table 29.

TABLE 29.—LNS-HR FROZEN ORBIT—C&DH MEL

WBS no.	Description LNS-HR (September 2008) Frozen, RF	Quantity	Unit mass, kg	Basic mass, kg	MGA, %	Growth, kg	Predicted mass, kg
6.0	Lunar Network Satellite-High Rate	-	---	1288.1	11	142.8	1430.9
6.0.1	RF Communications Package	-	---	123.1	12	14.9	138.0
6.0.2	Optical Communications Package	-	---	0.0	0	0.0	0.0
6.0.3	C&DH	-	---	77.2	26	20.2	97.4
6.0.3.1	C&DH Hardware	-	---	74.2	25	18.5	92.7
6.0.3.1.a	General Avionics Processor	2	4	8.0	10	0.8	8.8
6.0.3.1.b	Time Generation Unit	2	0.4	0.8	20	0.2	1.0
6.0.3.1.c	Command and Control Harness (data)	2	4	8.0	55	4.4	12.4
6.0.3.1.d	Data Recorder	2	4	8.0	20	1.6	9.6
6.0.3.1.e	Atomic Clock	2	6.6	13.2	20	2.6	15.8
6.0.3.1.f	Command and Control Harness (data)	0	0	0.0	0	0.0	0.0
6.0.3.1.g	Shared DPU (From APL Science Instruments)	0	0	0.0	0	0.0	0.0
6.0.3.1.h	Avionics enclosure	4	8.8	35.2	25	8.8	44.0
6.0.3.1.i	APL Ultra Stable Oscillator	2	0.4	0.8	10	0.1	0.9
6.0.3.1.j	Router Printed Circuit Board	1	0.2	0.2	10	0.0	0.2
6.0.3.1.k	Miscellaneous 3	0	0	0.0	0	0.0	0.0
6.0.3.2	Instrumentation & Wiring	-	---	3.0	55	1.7	4.7
6.0.3.2.a	Operational Instrumentation, sensors	8	0.3	2.4	55	1.3	3.7
6.0.3.2.b	Data Cabling	2	0.3	0.6	55	0.3	0.9
6.0.3.2.c	Miscellaneous 1	0	0	0.0	0	0.0	0.0
6.0.3.2.d	Miscellaneous 2	0	0	0.0	0	0.0	0.0
6.0.4	GN&C	-	---	42.5	20	8.5	51.0
6.0.5	Structures & Mechanical Systems	-	---	233.7	23	54.7	288.4
6.0.6	Power System	-	---	83.0	30	24.9	107.9
6.0.10	Propulsion (Chemical)	-	---	10.6	10	1.1	11.7
6.0.11	Propellant Management (Chemical)	-	---	43.9	13	5.7	49.5
6.0.12	Propellant (Chemical)	-	---	617.6	0	0.0	617.6
6.0.13	Thermal Control (Non-Propellant)	-	---	56.5	23	13.0	69.5

5.4.4 C&DH Trades

No trades performed.

5.4.5 C&DH Risk Inputs

The 0.3 TB storage array is designed around future technology. If the technology for the storage array is not yet available when the LRS is built, this component could be larger than expected.

The functions of all the avionics systems are not detailed to all lower levels. Exact packaging details have not been completely documented.

5.5 Guidance, Navigation and Control (GN&C)

5.5.1 GN&C Requirements

The GN&C system provides full 6-DOF control of the vehicle from launch through end of mission.

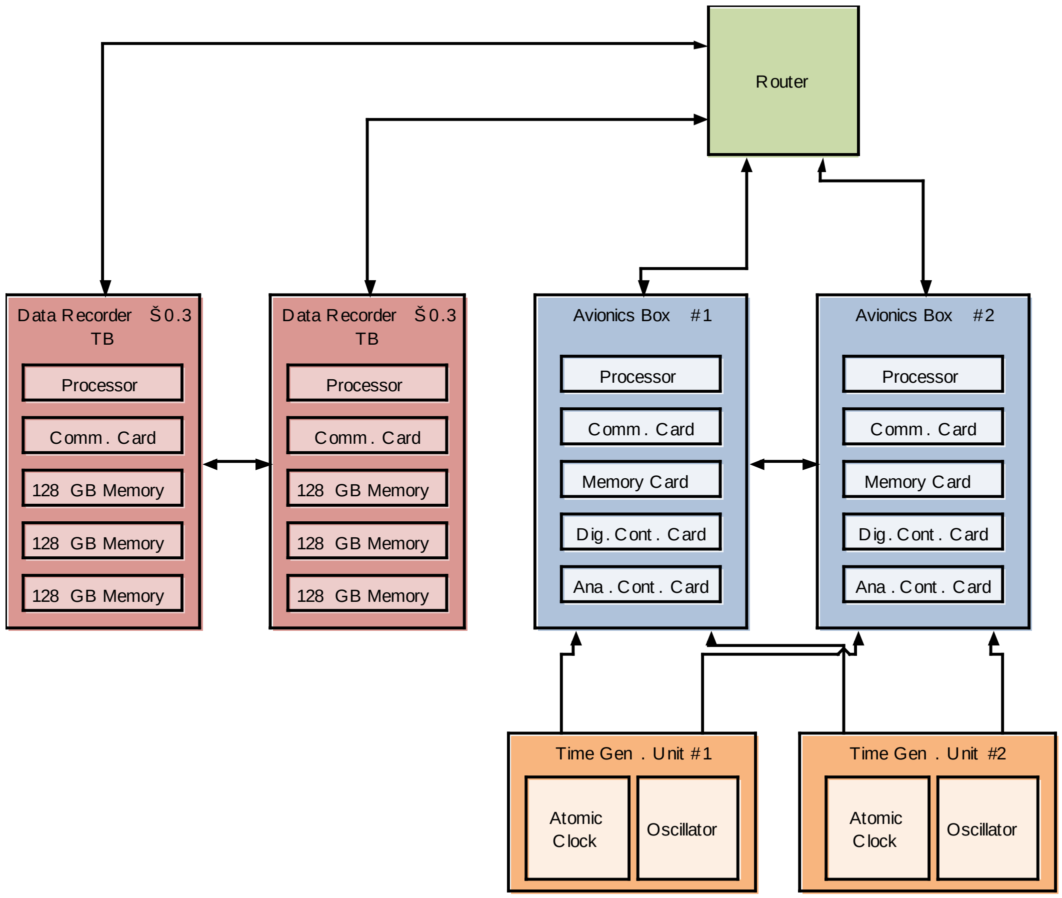


Figure 47.—LNS avionics block diagram.

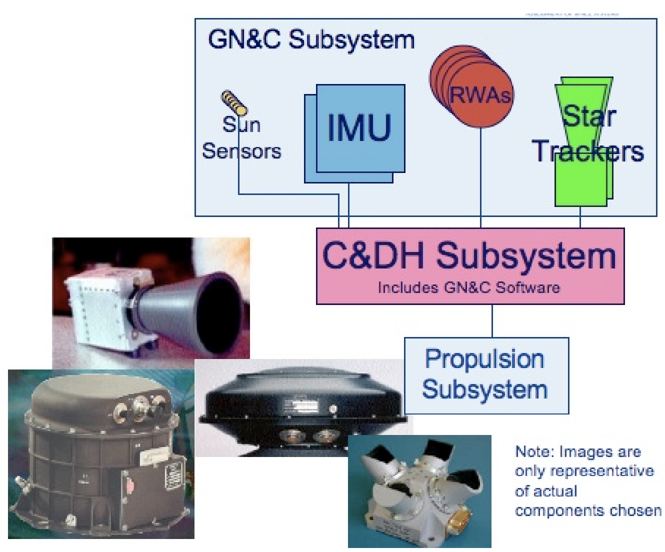


Figure 48.—GN&C flow diagram.

5.5.2 GN&C Assumptions

Parts assembled from off the shelf (OTS) hardware and components.

5.5.3 GN&C Design and MEL

- Model summary
    - Two Star Trackers
        - Goodrich HD-1003
    - Two IMUs
        - Honeywell MIMU
    - Sun sensors to aide in Earth acquisition
        - Adcole Sun sensors, two units, three sensor heads, each
    - Four reaction wheel assemblies for attitude control and fine pointing
        - 14 Nms of momentum storage for ~0.3° pointing accuracy
    - GN&C Software run on main C&DH computers

The components of the GN&C design of the Frozen orbit spacecraft is listed in Table 30.

TABLE 30.—LNS-HR FROZEN ORBIT—GN&C MEL

WBS no.	Description LNS-HR (September 2008) Frozen, RF	Quantity	Unit mass, kg	Basic mass, kg	MGA, %	Growth, kg	Predicted mass, kg
6.0	Lunar Network Satellite-High Rate	-	------	1288.1	11	142.8	1430.9
6.0.1	RF Communications Package	-	------	123.1	12	14.9	138.0
6.0.2	Optical Communications Package	-	------	0.0	0	0.0	0.0
6.0.3	C&DH	-	------	77.2	26	20.2	97.4
6.0.4	GN&C	-	------	42.5	20	8.5	51.0
6.0.4.1	GN&C Hardware	-	------	42.5	20	8.5	51.0
6.0.4.1.a	Reaction Wheel Assembly	4	5	20.0	20	4.0	24.0
6.0.4.1.b	RWA Mount	1	0.2	0.2	15	0.0	0.2
6.0.4.1.c	Star Camera	2	3.402	6.8	20	1.4	8.2
6.0.4.1.d	Star Camera Mount	1	0.1	0.1	15	0.0	0.1
6.0.4.1.e	Inertial Measurement Units	2	4.7	9.4	20	1.9	11.3
6.0.4.1.f	Course Sun Sensor Suite	6	1	6.0	20	1.2	7.2
6.0.5	Structures & Mechanical Systems	-	------	233.7	23	54.7	288.4
6.0.6	Power System	-	------	83.0	30	24.9	107.9
6.0.10	Propulsion (Chemical)	-	------	10.6	10	1.1	11.7
6.0.11	Propellant Management (Chemical)	-	------	43.9	13	5.7	49.5
6.0.12	Propellant (Chemical)	-	------	617.6	0	0.0	617.6
6.0.13	Thermal Control (Non-Propellant)	-	------	56.5	23	13.0	69.5

5.5.4 GN&C Trades

No trades performed.

5.5.5 GN&C Analytical Methods

Station-keeping Budget—Lagrange Orbit

From Table 31 the vertical cases L_1, L_2 are not as stable but provide better global coverage of the lunar surface.

For budgeting the ΔV for station keeping, use the details in the Folta, AIAA-2004-4741 technical report in Table 32. This gives a more general station keeping solution. Total ΔVs of ~60 m/s/yr can be expected using dLQR (design linear-quadratic regulator) control method expressed in Table 32. Station keeping burns need to be done on the order of every orbit.

TABLE 31.—LAGRANGE ORBIT STATION-KEEPING

Orbit type	Liberation point	Period, day	Stability index	Average $3\sigma\lVert\delta V_i\rVert$, cm/s	Number of maneuvers	Average time between, days	Average $\lVert\Delta v_i\rVert$, m/s	Average $\lVert\Delta v\rVert$, m/s
Near-rectilinear halo	L_2	7.0	1.00	2.06	86	4.20	0.057	4.82
Near-rectilinear halo	L_1	8.0	1.25	1.52	55	6.40	0.101	5.54
Near-rectilinear halo	L_2	8.0	1.00	2.18	55	6.40	0.086	4.69
Halo	L_1	12.0	60	3.82	60	6.00	1.106	66.33
Halo	L_2	14.0	115	2.77	156	2.33	0.183	28.47
Vertical	L_1	14.0	690	3.13	68	5.25	2.527	171.82
Butterfly	L_2	14.0	11.3	9.78	78	4.67	0.409	31.86
Vertical	L_1	16.0	370	2.81	91	4.00	0.347	31.55
Vertical	L_2	16.0	515	2.75	60	6.00	1.472	88.32

TABLE 32.—DLQR STATIONKEEPING YEARLY COST (M/S)

	Small Lissajous	Small Halo	Large Lissajous	Large Halo
L_1 no errors	6.41	6.11	5.61	5.99
L_1 with errors	61.26	61.13	60.22	60.48
L_2 no errors	5.37	5.38	5.38	5.61
L_2 with errors	60.87	61.00	59.88	59.86

5.6 Electrical Power System

5.6.1 Power Requirements

- 12 yr operational life
- Solar arrays provide 1100 W end of life (EOL) net power including 30 percent margin

5.6.2 Power Assumptions

- Eclipse power is limited to 605 W net S/C power, including 30 percent margin, for maximum eclipse time of nearly 7 hr

5.6.3 Power Design and MEL

- Solar array
  - 5.9 m^2 total area
  - 28.5 percent SOA triple-junction solar cells
  - Rectangular array structure
  - 15° cosine loss assumed
  - Single axis gimbal
  - 1600 W BOL net power including 30 percent margin
- Battery
  - 4.3 kWh total net energy
  - SOA Li-ion battery cells
- Power Management and Distribution
  - Switching unit
  - Battery charge control unit
  - Cabling
- Technology status
  - No special technology requirements
  - SOA SA, batteries, Power Management and Distribution (PMAD)
- SA mass: 32 kg
- Battery system mass: 56 kg
- Additional power system mass: 20 kg

The components of the power system design of the Frozen orbit spacecraft is listed in Table 33.

TABLE 33.—LNS-HR FROZEN ORBIT—POWER SYSTEM MEL

WBS no.	Description LNS-HR (September 2008) Frozen, RF	Quantity	Unit mass, kg	Basic mass, kg	MGA, %	Growth, kg	Predicted mass, kg
6.0	Lunar Network Satellite-High Rate	-	---	1288.1	11	142.8	1430.9
6.0.1	RF Communications Package	-	---	123.1	12	14.9	138.0
6.0.2	Optical Communications Package	-	---	0.0	0	0.0	0.0
6.0.3	C&DH	-	---	77.2	26	20.2	97.4
6.0.4	GN&C	-	---	42.5	20	8.5	51.0
6.0.5	Structures & Mechanical Systems	-	---	233.7	23	54.7	288.4
6.0.6	Power System	-	---	83.0	30	24.9	107.9
6.0.6.1	Battery System	-	---	43.0	30	12.9	55.9
6.0.6.1.a	Battery Assembly-Primary	1	43	43.0	30	12.9	55.9
6.0.6.1.b	Battery Assembly-Secondary	0	0	0.0	0	0.0	0.0
6.0.6.2	Solar Array	-	---	25.0	30	7.5	32.5
6.0.6.2.a	Solar Array Panel	1	18	18.0	30	5.4	23.4
6.0.6.2.b	Solar Array Structure	1	3	3.0	30	0.9	3.9
6.0.6.2.c	Solar Array Drive Assembly	1	4	4.0	30	1.2	5.2
6.0.6.2.d	Solar Array Interface	0	0	0.0	30	0.0	0.0
6.0.6.3	Power Management & Distribution	-	---	15.0	30	4.5	19.5
6.0.6.3.a	Main Bus Switching Unit	1	4	4.0	30	1.2	5.2
6.0.6.3.b	Battery Charge Control Unit	1	1	1.0	30	0.3	1.3
6.0.6.3.c	Power Cabling	1	10	10.0	30	3.0	13.0
6.0.6.3.d	Miscellaneous 1	0	0	0.0	0	0.0	0.0
6.0.6.3.e	Miscellaneous 2	0	0	0.0	0	0.0	0.0
6.0.10	Propulsion (Chemical)	-	---	10.6	10	1.1	11.7
6.0.11	Propellant Management (Chemical)	-	---	43.9	13	5.7	49.5
6.0.12	Propellant (Chemical)	-	---	617.6	0	0.0	617.6
6.0.13	Thermal Control (Non-Propellant)	-	---	56.5	23	13.0	69.5

5.6.4 Power Trades

More S/C eclipse power versus more battery mass.

5.6.5 Power Analytical Methods

Figure 49 shows the variation of eclipse time for the frozen orbit spacecraft throughout the 8 year mission lifetime. The eclipse times are used to size the battery systems.

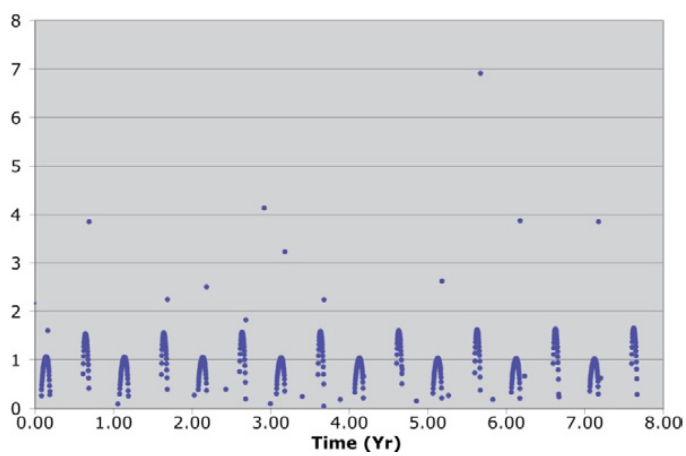


Figure 49.—Eclipse time for frozen orbit power system modeling.

5.6.6 Power Recommendation

- Further safe-mode-like operations for longest (but rare) eclipses will reduce battery mass further
- Ultraflex SA may reduce mass at potentially higher cost

5.7 Structures and Mechanisms

5.7.1 Structures and Mechanisms Requirements

- Contain necessary hardware for research instrumentation, avionics, communications, propulsion and power
- Withstand applied loads from launch vehicle and provide minimum deflections, sufficient stiffness, and vibration damping
- Maximum longitudinal loads: 6g
- Maximum lateral loads: 4g
- Ability to have two probes stacked in launch vehicle
- Minimize weight
- Fit within confines of launch vehicle
- Accommodate landing and takeoff on terrestrial body

5.7.2 Structures and Mechanisms Assumptions

- Material: Aluminum
- Central thrust tube
- Space frame with square tubular members
- Shear panels
- Welded and threaded fastener assembly

5.7.3 Structures and Mechanisms Design and MEL

The thrust tube carries another S/C above as well as propellant tanks. The S/C volume was sufficient for all systems. Each system populates a single side of a stiffener panel.

Frozen Orbit Main Structures

- Installation mass was calculated as 4 percent of the CBE mass of mounted unit masses
- Negligible stress, 140 psi, in thrust tube with 4g lateral load and stacked

Figure 50 shows the preliminary wire model of the LNS Frozen orbit spacecraft primary structural elements. This preliminary model was used to determine the size of the primary structures of the spacecraft bus. The structural components and masses are listed in Table 34. Figure 51 shows the preliminary wire model of the LNS Lagrange orbit spacecraft primary structural elements. This preliminary model was used to determine the size of the primary structures of the spacecraft bus.

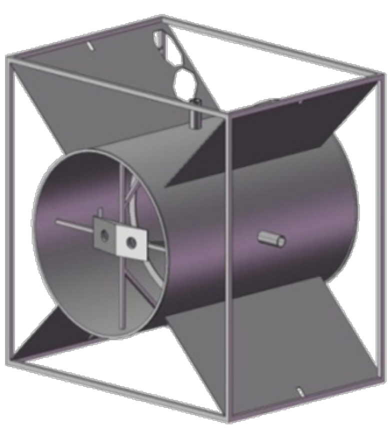


Figure 50.—LNS frozen orbit structure wire model.

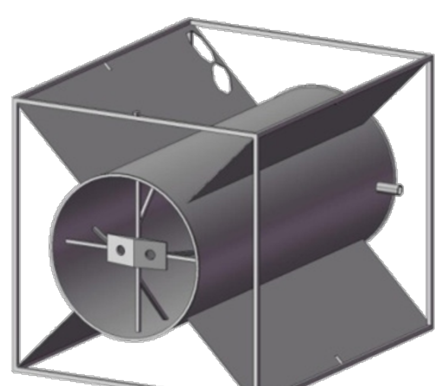


Figure 51.—LNS Lagrange orbit structure wire model.

TABLE 34.—LNS-HR FROZEN ORBIT—STRUCTURES MEL

WBS no.	Description LNS-HR (September 2008) Frozen, RF	Quantity	Unit mass, kg	Basic mass, kg	MGA, %	Growth, kg	Predicted mass, kg
6.0	Lunar Network Satellite-High Rate	----	-------	1288.1	11	142.8	1430.9
6.0.1	RF Communications Package	----	-------	123.1	12	14.9	138.0
6.0.2	Optical Communications Package	----	-------	0.0	0	0.0	0.0
6.0.3	C&DH	----	-------	77.2	26	20.2	97.4
6.0.4	GN&C	----	-------	42.5	20	8.5	51.0
6.0.5	Structures & Mechanical Systems	----	-------	233.7	23	54.7	288.4
6.0.5.1	Primary Structures	----	-------	199.1	25	49.8	248.9
6.0.5.1.a	Separation Ring	1	18.72	18.7	25	4.7	23.4
6.0.5.1.b	Top/Bottom Deck	2	15.54	31.1	25	7.8	38.9
6.0.5.1.c	Vertical Inner Panels	4	3.942	15.8	25	3.9	19.7
6.0.5.1.d	Side Panels	4	14.09	56.4	25	14.1	70.5
6.0.5.1.e	Corner post	4	0.668	2.7	25	0.7	3.3
6.0.5.1.f	Top Cover	0	0	0.0	25	0.0	0.0
6.0.5.1.g	Thrust tube	1	74.51	74.5	25	18.6	93.1
6.0.5.1.h	Miscellaneous clips/fasteners	0	0	0.0	25	0.0	0.0
6.0.5.1.i	Miscellaneous 1	0	0	0.0	15	0.0	0.0
6.0.5.1.j	Miscellaneous 2	0	0	0.0	15	0.0	0.0
6.0.5.2	Secondary Structures	----	-------	7.3	25	1.8	9.1
6.0.5.2.a	Antenna Boom	1	5.126	5.1	25	1.3	6.4
6.0.5.2.b	Solar Array Boom (s)	1	2.148	2.1	25	0.5	2.7
6.0.5.2.c	EP Thruster Boom s	0	0.537	0.0	25	0.0	0.0
6.0.5.2.d	Chemical Thruster Boom s	0	0.537	0.0	25	0.0	0.0
6.0.5.2.e	Miscellaneous 1	0	0	0.0	0	0.0	0.0
6.0.5.2.f	Miscellaneous 2	0	0	0.0	0	0.0	0.0
6.0.5.3	Installation	----	-------	12.3	25	3.1	15.4
6.0.5.3.a	RF Communications Installation	1	4.804	4.8	25	1.2	6.0
6.0.5.3.b	Optical Communications Installation	0	0	0.0	25	0.0	0.0
6.0.5.3.c	C&DH Installation	1	0.916	0.9	25	0.2	1.1
6.0.5.3.d	GN&C Installation	on 1	1.7	1.7	25	0.4	2.1
6.0.5.3.e	Power Installation	1	2.92	2.9	25	0.7	3.7
6.0.5.3.f	Propulsion (EP) Installation	0	0	0.0	25	0.0	0.0
6.0.5.3.g	Propellant (EP) Storage Installation	0	0	0.0	25	0.0	0.0
6.0.5.3.h	Propulsion (Chemical) Installation	1	0.48	0.5	25	0.1	0.6
6.0.5.3.i	Propellant (Chemical) Storage Installation	0	1.663	0.0	25	0.0	0.0
6.0.5.3.j	Thermal Installation	1	1.494	1.5	25	0.4	1.9
6.0.5.3.k	Miscellaneous 1	0	0	0.0	0	0.0	0.0
6.0.5.3.l	Miscellaneous 2	0	0	0.0	0	0.0	0.0
6.0.5.3.m	Miscellaneous 3	0	0	0.0	0	0.0	0.0
6.0.5.4	Mechanisms	----	-------	15.0	0	0.0	15.0
6.0.5.4.a	Solar array deployment mechanism	1	5	5.0	0	0.0	5.0
6.0.5.4.b	radiator deployment mechanism (if applicable)	0	0	0.0	0	0.0	0.0
6.0.5.4.c	Separation mechanism (pyros)	0	0	0.0	0	0.0	0.0
6.0.5.4.d	Bi-axial antenna gimbal	1	6	6.0	0	0.0	6.0
6.0.5.4.e	Antenna deployment	1	4	4.0	0	0.0	4.0
6.0.5.4.f	Miscellaneous 3	0	0	0.0	0	0.0	0.0
6.0.6	Power System	----	-------	83.0	30	24.9	107.9
6.0.10	Propulsion (Chemical)	----	-------	10.6	10	1.1	11.7
6.0.11	Propellant Management (Chemical)	----	-------	43.9	13	5.7	49.5
6.0.12	Propellant (Chemical)	----	-------	617.6	0	0.0	617.6
6.0.13	Thermal Control (Non-Propellant)	----	-------	56.5	23	13.0	69.5

Lagrange Point Orbit Main Structures

- Structure width and thrust tube diameter maintained relative to frozen orbit structure
- Length increased relative to frozen orbit structure
- Installation mass, 4 percent of mounted unit mass
- Negligible stress, 182 psi, in thrust tube with 4g lateral load and stacked

5.7.4 Structures and Mechanisms Analytical Methods

Main structure is designed using launch loads and rough stresses and analysis at a top level. Preliminary structural analysis with given launch loads. Scaling equations are used for some mass numbers. Modeling is done in close coordination with configuration in order to get accurate dimensions for the main structure for mass calculations.

5.7.5 Structures and Mechanisms Risk Inputs

- Potential impact with foreign object or due to nearby operations.

5.7.6 Structures and Mechanisms Recommendation

- Current configuration based on heritage design
- Accommodate longitudinal and lateral launch loads
- Thrust tube bears majority structural loads

5.8 Propulsion and Propellant Management

5.8.1 Propulsion and Propellant Management Requirements

- Subsystem is single-fault tolerant, excluding DMR (Design for Minimum Risk) elements such as the propellant tank and main propellant lines.

5.8.2 Propulsion and Propellant Management Assumptions

- Thruster Operation
    - Thruster operation constant over duration of burn
- Propellant Storage
    - Propellant delivery assumed to be isothermal
    - Same with He pressurant into main tank
- Design for Minimum Risk Applied to Tanks and Feed Lines
    - Following Exploration lead
    - Limit DMR to Passive elements

5.8.3 Propulsion and Propellant Management Design and MEL

The monoprop propulsion subsystem used for station-keeping is comprised of (Figure 52):

- Two x 100-lbf-thrust MR-104 Hydrazine Monopropellant Engines—one operating, one spare
- Four pods of three each 0.5-lbf-thrust MR-111E Hydrazine Monopropellant Engines—all operating
- One metallic Ti hydrazine storage tank

- One metallic Ti high pressure He storage tank for pressurant gas
- Propellant distribution system to control propellant delivery from storage tank to thrusters
- Pressurant distribution system to control pressurant delivery to hydrazine storage tank

The components of the Propulsion system design and Propellant of the Frozen orbit spacecraft is listed in Table 35.

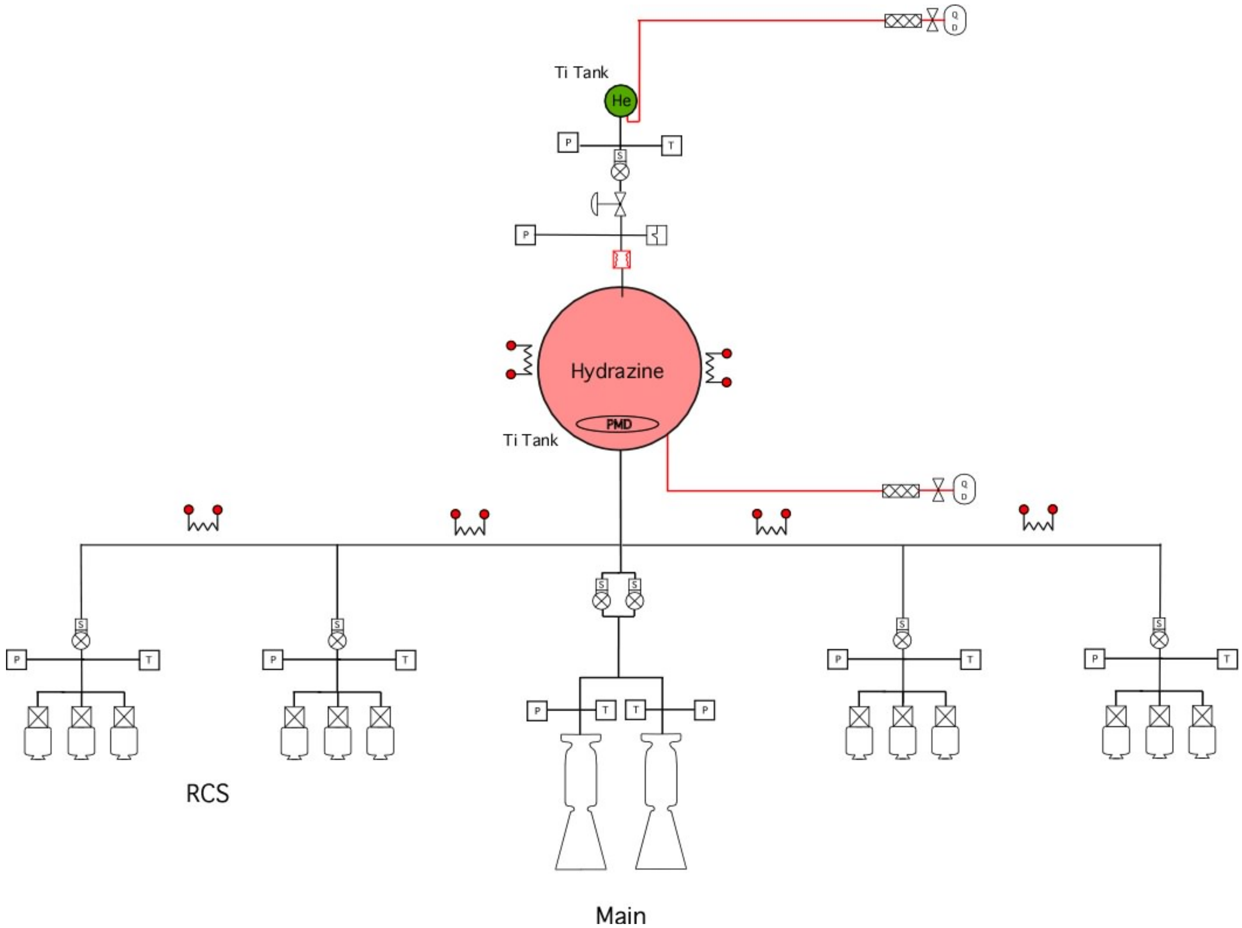


Figure 52.—Mono-Prop Propulsion System schematic.

TABLE 35.—LNS-HR FROZEN ORBIT—PROPULSION AND PROPELLANT MEL

WBS no.	Description LNS-HR (September 2008) Frozen, RF	Quantity	Unit mass, kg	Basic mass, kg	MGA, %	Growth, kg	Predicted mass, kg
6.0	Lunar Network Satellite-High Rate	-	-------	1288.1	11	142.8	1430.9
6.0.1	RF Communications Package	-	-------	123.1	12	14.9	138.0
6.0.2	Optical Communications Package	-	-------	0.0	0	0.0	0.0
6.0.3	C&DH	-	-------	77.2	26	20.2	97.4
6.0.4	GN&C	-	-------	42.5	20	8.5	51.0
6.0.5	Structures & Mechanical Systems	-	-------	233.7	23	54.7	288.4
6.0.6	Power System	-	-------	83.0	30	24.9	107.9
6.0.10	Propulsion (Chemical)	-	-------	10.6	10	1.1	11.7
6.0.10.1	Main Engine	-	-------	4.6	10	0.5	5.1
6.0.10.1.a	Main Engine	2	2.313	4.6	10	0.5	5.1
6.0.10.1.b	Main Engine Gimbal	0	0	0.0	10	0.0	0.0
6.0.10.2	Reaction Control System	-	-------	6.0	10	0.6	6.6
6.0.10.2.a	RCS Engine	4	1.495	6.0	10	0.6	6.6
6.0.11	Propellant Management (Chemical)	-	-------	43.9	13	5.7	49.5
6.0.11.1	OMS Propellant Management	-	-------	0.0	0	0.0	0.0
6.0.11.1.a	Fuel Tanks	0	0	0.0	10	0.0	0.0
6.0.11.1.b	Fuel Lines	0	0	0.0	25	0.0	0.0
6.0.11.1.c	Oxidizer Tanks	0	0	0.0	0	0.0	0.0
6.0.11.1.d	Oxidizer Lines	0	0	0.0	0	0.0	0.0
6.0.11.1.e	Pressurization System - tanks , panels , lines	0	0	0.0	25	0.0	0.0
6.0.11.1.f	Feed System - regulators, valves, etc.	0	0	0.0	25	0.0	0.0
6.0.11.2	RCS Propellant Management	-	-------	43.9	13	5.7	49.5
6.0.11.2.a	Fuel Tanks	1	26.33	26.3	10	2.6	29.0
6.0.11.2.b	Fuel Lines	0	0	0.0	0	0.0	0.0
6.0.11.2.c	Pressurization System - tanks, panels, lines	1	13.63	13.6	15	2.0	15.7
6.0.11.2.d	Feed System - regulators, valves, etc.	1	3.92	3.9	25	1.0	4.9
6.0.12	Propellant (Chemical)	-	-------	617.6	0	0.0	617.6
6.0.12.1	Main Engine Propellant	-	-------	0.0	0	0.0	0.0
6.0.12.1.a	Fuel	-	-------	0.0	0	0.0	0.0
6.0.12.1.a.a	Fuel Usable	0	0	0.0	0	0.0	0.0
6.0.12.1.a.b	Fuel Boiloff	0	0	0.0	0	0.0	0.0
6.0.12.1.a.c	Fuel Residuals (Unused)	0	0	0.0	0	0.0	0.0
6.0.12.1.b	Oxidizer	-	-------	0.0	0	0.0	0.0
6.0.12.1.b.a	Oxidizer Usable	0	0	0.0	0	0.0	0.0
6.0.12.1.b.b	Oxidizer Boiloff	0	0	0.0	0	0.0	0.0
6.0.12.1.b.c	Oxidizer Residuals (Unused)	0	0	0.0	0	0.0	0.0
6.0.12.1.c	Main Engine Pressurant	0	0	0.0	0	0.0	0.0
6.0.12.2	RCS Propellant	-	-------	617.6	0	0.0	617.6
6.0.12.2.a	Fuel	-	-------	614.5	0	0.0	614.5
6.0.12.2.a.a	Fuel Usable	1	599.5	599.5	0	0.0	599.5
6.0.12.2.a.b	Fuel Boiloff	0	0	0.0	0	0.0	0.0
6.0.12.2.a.c	Fuel Residuals (Unused)	1	14.99	15.0	0	0.0	15.0
6.0.12.2.b	Oxidizer	-	-------	0.0	0	0.0	0.0
6.0.12.2.b.a	Oxidizer Usable	0	0	0.0	0	0.0	0.0
6.0.12.2.b.b	Oxidizer Boiloff	0	0	0.0	0	0.0	0.0
6.0.12.2.b.c	Oxidizer Residuals (Unused)	0	0	0.0	0	0.0	0.0
6.0.12.2.c	RCS Pressurant	1	3.123	3.1	0	0.0	3.1
6.0.13	Thermal Control (Non-Propellant)	-	-------	56.5	23	13.0	69.5

5.8.4 Propulsion and Propellant Management Analytical Methods

Hydrazine Thrusters

- Aerojet MR-104 thruster characteristics (Figure 53)
  - Thrust: 441 N (100 lbf); 204.6 to 572.5 N (46 to 128.7 lbf)
  - Specific impulse: 239; 239 to 223 sec
  - Mass: 1.86 kg
  - Power = 43.1 W
  - Lifetime = 2,654 sec (cumulative)
- Aerojet MR-111E thruster characteristics (Figure 54)
  - Thrust: 2.2 N (0.5 lbf); 0.5 to 2.2 N (0.11 to 0.5 lbf)
  - Specific Impulse: 224; 224 to 213 sec
  - Mass: 0.33 kg
  - Power = 13.64 W
  - Lifetime = 26.7 hr (cumulative)
- Storage tank based on COTS unit from ATK-PSC, Inc. (Figure 55)
  - Titanium metallic tank
  - Derived from Model No. 80352-1
    - Size: 1.06 m diam. spherical (42 in. diam.)
    - Internal volume = 0.59 m^3 (20.3 ft^3)
  - Minor size changes to match propellant load
  - Propellant management device included in tank mass

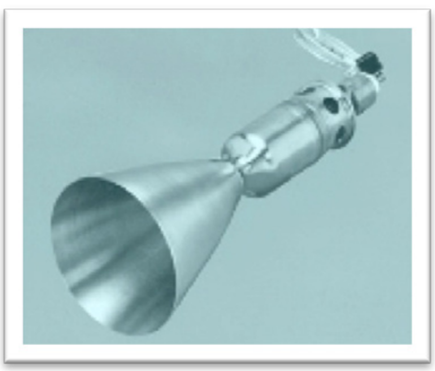


Figure 53.—Aerojet MR-104 Thruster.

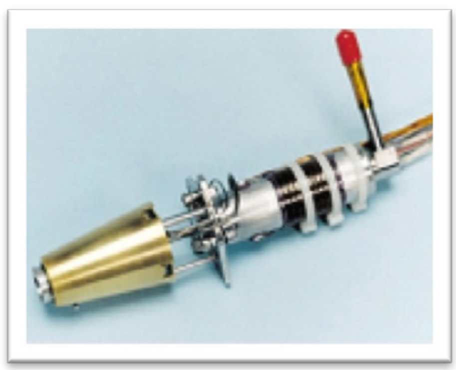


Figure 54.—Aerojet MR-111E Thruster.

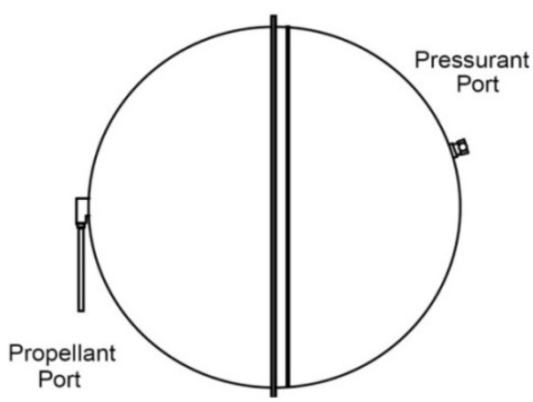


Figure 55.—Propellant Storage Tank.

- Helium Pressurization System
    - One metallic Ti spherical tank, storing pressurant at 2800 psia
- Propellant Distribution System Components
    - Derived from existing technology
    - Did not need to include technology in development
    - Minimal redundancy to reduce system mass
- All Components of Propulsion Subsystem at High TRL
    - Little or no development costs required

5.8.5 Propulsion and Propellant Management Risk Inputs

- Potential risks
    - RCS thruster plume impingement (**Low**)
        - Potential degradation due to deposition on sensitive surfaces
    - Propellant Freezing within fluid lines (**Low-Medium**)
        - Freezing could lead to line ruptures

5.8.6 Propulsion and Propellant Management Recommendation

- Propellant storage and delivery
    - Single metallic tank selected for hydrazine storage
    - He pressurization selected over blowdown approach to save mass
        - Preliminary trade on blowdown system found the tanks required for storage of the hydrazine propellant to be significantly larger and therefore heavier
    - He pressurization and propellant distribution systems are configured to be single-fault tolerant
        - Excluding tanks and lines which are single string

5.9 Thermal Control

5.9.1 Objective

To provide spreadsheet based models capable of estimating the mass and power requirements of the various thermal systems. The thermal modeling provides power and mass estimates for the various aspects of the vehicle thermal control system based on a number of inputs related to the vehicle geometry, flight environment and component size. The system consists of the following elements

- Electric heaters
- MLI
- Thermal paint
- Radiator with louvers
- Thermal Control System (sensors, switches, data acquisition)

5.9.2 Thermal Requirements

The thermal requirements for the LNS were to provide a means of cooling the S/C during operation as well as provide heat to vital components and systems to maintain a minimum temperature throughout the mission. Figure 56 shows the Frozen orbit configuration with the thermal components identified.

The maximum heat load to be rejected by the thermal system was 656 W, and the desired operating temperature for the electronics and propellant was 300 K.

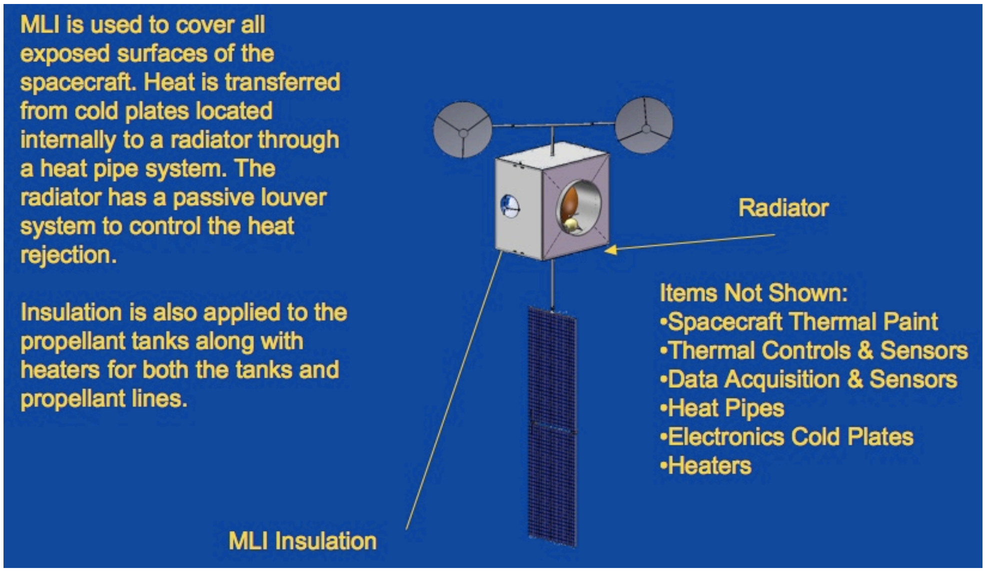


Figure 56.—LNS Frozen Orbit Thermal Systems.

5.9.3 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 orbital environment. The following assumptions were utilized to size the thermal system.

- The view factors for the radiator to the Earth, lunar surface and SA were assumed to be 0.1, 0.25 and 0.1 respectively.
- The maximum angle of the radiator to the Sun was 30°.
- The radiator temperature was 320 K.

5.9.4 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 during periods of inactivity.

Excess heat is collected from a series of aluminum 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 S/C body and are integrated to the radiator are protected with a micro meteor shield. The radiator has exterior louvers on it to provide some control over its heat transfer capability.

The radiator was sized with approximately 75 percent margin in its heat rejection area. This added margin insures against unforeseen heat loads, degradation of the radiator and increased view factor toward the Sun or other thermally hot body not accounted for in the analysis.

To provide internal heating for the electronics and propulsion systems a series of electric heaters are utilized. These heaters are controlled by an electronics controller, which reads a series of thermocouples through a data acquisition system.

MLI is also utilized on the S/C, and propellant system to regulate and maintain the desired temperatures.

The components of the Thermal system design of the Frozen orbit spacecraft is listed in Table 36.

TABLE 36.—LNS-HR FROZEN ORBIT—THERMAL MEL

WBS no.	Description LNS-HR (September 2008) Frozen, RF	Quantity	Unit mass, kg	Basic mass, kg	MGA, %	Growth, kg	Predicted mass, kg
6.0	Lunar Network Satellite-High Rate	--	--------	1288.1	11	142.8	1430.9
6.0.1	RF Communications Package	--	--------	123.1	12	14.9	138.0
6.0.2	Optical Communications Package	--	--------	0.0	0	0.0	0.0
6.0.3	C&DH	--	--------	77.2	26	20.2	97.4
6.0.4	GN&C	--	--------	42.5	20	8.5	51.0
6.0.5	Structures & Mechanical Systems	--	--------	233.7	23	54.7	288.4
6.0.6	Power System	--	--------	83.0	30	24.9	107.9
6.0.10	Propulsion (Chemical)	--	--------	10.6	10	1.1	11.7
6.0.11	Propellant Management (Chemical)	--	--------	43.9	13	5.7	49.5
6.0.12	Propellant (Chemical)	--	--------	617.6	0	0.0	617.6
6.0.13	Thermal Control (Non-Propellant)	--	--------	56.5	23	13.0	69.5
6.0.13.1	Active Thermal Control	--	--------	4.2	21	0.9	5.1
6.0.13.1.a	Heaters	10	0.143	1.4	15	0.2	1.6
6.0.13.1.b	Thermal Control/Heaters Circuit	2	0.2	0.4	25	0.1	0.5
6.0.13.1.c	Data Acquisition	1	1	1.0	25	0.3	1.3
6.0.13.1.d	Thermocouples	25	0.01	0.3	15	0.0	0.3
6.0.13.1.e	Radiator MMOD Shielding	1	1.109	1.1	25	0.3	1.4
6.0.13.2	Passive Thermal Control	--	--------	41.4	23	9.5	50.9
6.0.13.2.a	Heat Sinks	4	3.463	13.9	20	2.8	16.6
6.0.13.2.b	Heat Pipes	1	1.98	2.0	25	0.5	2.5
6.0.13.2.c	Radiators	1	17.49	17.5	25	4.4	21.9
6.0.13.2.d	MLI	1	5.497	5.5	25	1.4	6.9
6.0.13.2.e	Temperature sensors	50	0.01	0.5	15	0.1	0.6
6.0.13.2.f	Phase Change Devices	0	0	0.0	0	0.0	0.0
6.0.13.2.g	Thermal Coatings /Paint	1	0.907	0.9	15	0.1	1.0
6.0.13.2.h	Antenna Radiator	2	0.586	1.2	25	0.3	1.5
6.0.13.3	Semi-Passive Thermal Control	--	--------	10.9	24	2.6	13.5
6.0.13.3.a	Louvers	1	10.07	10.1	25	2.5	12.6
6.0.13.3.b	Thermal Switches	4	0.2	0.8	15	0.1	0.9

5.9.5 Thermal Trades

None

5.9.6 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.

5.9.6.2 Environmental Models

Solar intensity modeling was based on S/C location. Components were sized for worst case operating conditions, heat rejection: near Earth, minimum temperature: lunar orbital location.

5.9.6.3 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 radioisotope heater units (RHU))

Table 37 captures the inputs and outputs used in the design of the thermal system.

TABLE 37.—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 SAs 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

5.9.6.4 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 environment. Lunar orbit 1 AU thermal environment was used to size the radiator.

Louvers are active or passive devices that regulate the amount of heat rejected by the radiator. Active controlled louvers use temperature sensors and actuators to control the louver position. Passive controlled louvers commonly use a bimetallic spring that opens and closes the louver based on temperature. The louver specific mass is 4.5 kg/m^2. Table 38 captures the assumptions used in the design of the radiator thermal system. Figure 57 shows an artist's representation of a louver prototype.

TABLE 38.—THERMAL SYSTEM RADIATOR SIZING ASSUMPTIONS

Variable	Value
Radiator solar absorptivity	0.14
Radiator emissivity	0.84
Radiator Sun angle	70°
Radiator operating temperature	320 k
Total radiator dissipation power	656.5 W
View Factor to Solar Array	0.10
View Factor to Earth	0.10
View Factor to Moon	0.25

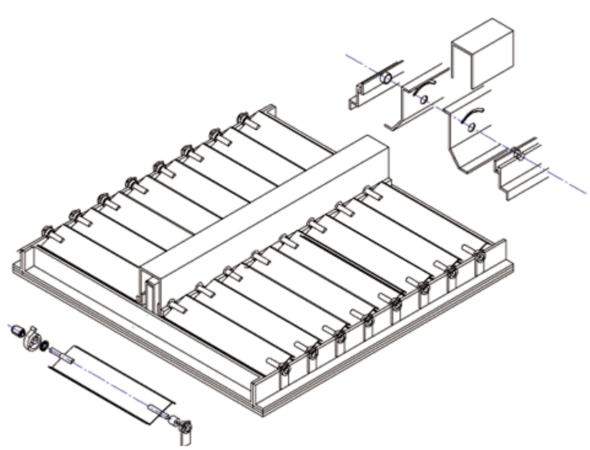


Figure 57.—Schematic view of the louver prototype.

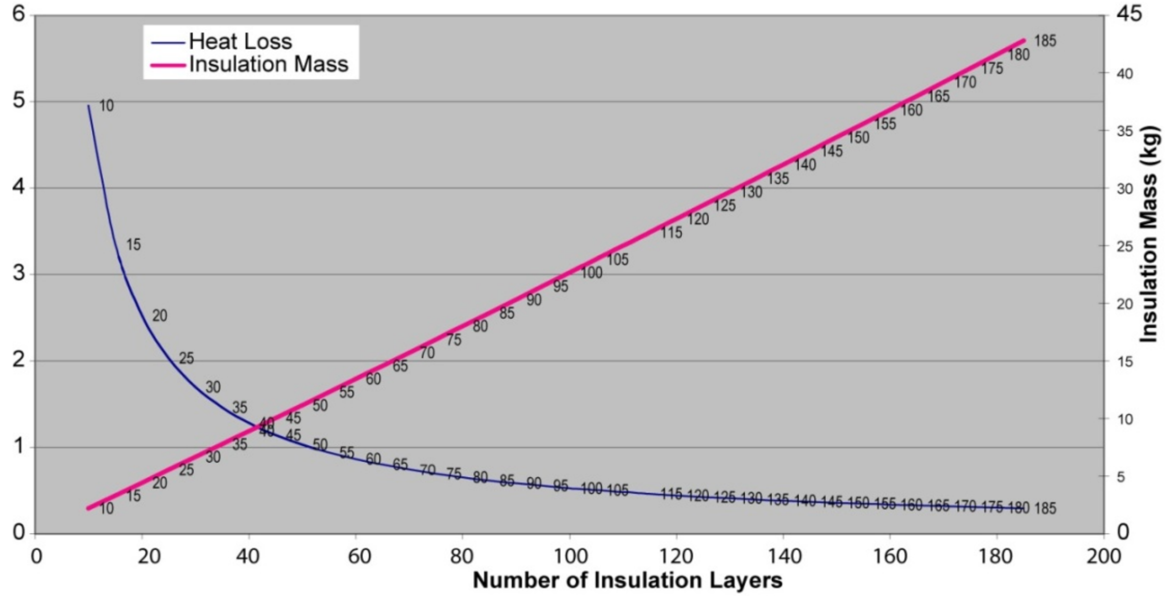


Figure 58.—Heat loss (heater power) and MLI mass as a function of MLI layers for a 1 m diameter propellant tank.

5.9.6.5 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 (Figure 58).

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 1 AU thermal environment was used to calculate the heat loss. Table 39 shows the assumptions used.

TABLE 39.—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

5.9.6.6 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 40 shows the assumptions used.

TABLE 40.—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
S/C Radius (rsc)	0.825 m

5.9.6.7 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. Table 41 shows the assumptions used.

TABLE 41.—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	0.5 m
Cooling plate widths	0.5 m
Cooling plate thickness	5 mm
Heat pipe specific mass	0.15 kg/m

5.9.7 Thermal Risk Inputs

The risks associated with the thermal system are based mainly on the failure of a component of 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, micro-meteor 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.
- ***Heater system failure***.—This would most likely be due to wire breakage or a controller failure. The likelihood of this type of failure is low. The impact of this failure would be a loss of certain components or propulsion capability once the vehicle is exposed to an extended period of cold
- ***Radiator louver failure***.—A failure of the louvers used to regulate the heat dissipation of the radiators can cause the radiator to partially or fully fail. The louvers are however, built of separate slats that are individually controlled by a passive temperature sensitive spring mechanism. Failure of one or more of these spring mechanisms would degrade the performance of the radiator. The margin built into the radiator sizing can accommodate some louver failure without impacting the required heat dissipation capability of the radiator.

5.9.1 Thermal Recommendation

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

- Redundant heat pips can be utilized for each cold plate. The heat pipes can be individually run to the radiator to provide independent cooling paths. The radiator can be separated into two independent units providing additional redundancy.
- Redundant heating system controllers can be utilized. The heaters can be wired individually so that a single heater failure does not bring down any additional heaters. Additional insulation can be added to the S/C to insure that the interior components do not drop below their desired minimum temperature based on the known shadow period of operation.
- Building margin into the radiator sizing will provide a means of accommodating any failure that may occur with the louver system.

6.0 Software Cost Estimation

6.1 Objectives

- To understand the functional requirements that the LRS is designed to meet (software is strongly affected by hardware design)
- To develop software cost estimates based on the LRS functional requirements, especially the LRS avionics functionality.

6.2 Assumptions

- Real-time operating system (RTOS) (such as VxWorks or RT Linux) for a single-board computer or a multipurpose computer to be ported to the LRS system/subsystems
- Inter-network Operating System (Similar to Cisco IOS) to be ported to LRS router
- IP-based networks to be supported
- Comply with all security protocols defined by Constellation C3I
- Any new software to be developed will be C/C++ or Ada or a combination of these languages for mission critical such as avionics. For this software cost estimate, the study team assumed C.
- Source lines of code (SLOC) is a count of the text of the source code including comment lines, neither "physical" nor "logical" is considered as the types of SLOC measures. Software sizing and estimation are based on mainly on the LRS hardware functionality for avionics and communication and navigation. Other sources and references are also used for estimating number of SLOC. For this study, see Table 42 for the SLOC estimate is used based on inputs from the COMPASS team
- The variation of the provided SLOC from –10 to 30 percent is consistent with other software estimates generated for COMPASS.

TABLE 42.—ASSUMED SLOC FOR LNS

	LRS Avionics Functionality	SLOC	–10%	Provided	30%	Application
General Avionics	System initialization	400	360	400	520	System Device Utilities
Processor	Antenna deployment and positioning	800	720	800	1,040	Flight Systems
	Solar array deployment and positioning	800	720	800	1,040	Flight Systems
	Satellite navigation	11,400	10,260	11,400	14,820	Flight Systems
	Satellite guidance	2,000	1,800	2,000	2,600	Flight Systems
	Propulsion system control	300	270	300	390	Flight Systems
	Systems health and status management	3,000	2,700	3,000	3,900	Network Management
	Power management and monitoring	500	450	500	650	Diagnostics
	Battery regulation and management	500	450	500	650	Diagnostics
	Thermal system management	300	270	300	390	System Device Utilities
	System fault detection and correction	2,500	2,250	2,500	3,250	Diagnostics
	Time synchronization and stamping	200	180	200	260	Process Control
	Router management (assuming no new software has to be written)	0	------	------	------	OS/Executive
	Communication system management	800	720	800	1,040	Communications
Data Recorder	Storage array monitoring and health management	300	270	300	390	Network Management
	Synchronization of data between redundant data recorders	200	180	200	260	Testing Software
	Buffering data from WAN and LAN for transmission to Earth	300	270	300	390	Communications
	Estimated total SLOC—Avionics	24,300	------	------	------	
Communications	Command processing	400	360	400	520	Command/Control
	Telemetry processing	400	360	400	520	Communications
Navigation	Attitude Determination and Control	11,000	9,900	11,000	14,300	Flight Systems
	Estimated total SLOC—Comm and nav	11,800	------	------	------	
	Total SLOC	36,100	32,490	36,100	46,930	

6.3 Approach

The System Evaluations and Estimation of Resources-Software Estimation Model (SEER-SEM), Version 7.7.21 was used to estimate the cost of software development. The SLOC estimate from the previous section was used as the major input for this model. SLOC includes all executable LOC, non-executable declarations and compiler directives, but excludes comments, banners, blank lines, and non-blank spacers. Additional SEER-SEM specific assumptions for this study are as follows:

- Platform: Unmanned space
- Application: Shown in Table 42
- Development method: No knowledge
- Development standard: No knowledge
- Class: No knowledge

Using the inputs and assumptions listed above, the detailed breakdown of the software cost estimate is shown in Table 43. Monte Carlo risk analysis is performed on the estimate based on the SLOC range for each subprogram modeled in SEER-SEM. The risk analysis results for this estimate are as Table 44.

TABLE 43.—SEER-SEM RESULTS: SOFTWARE COST BREAKDOWN FOR LNS

SEER-SEM (TM) Software Schedule, Cost & Risk Estimation Version 7.2.21

Project : Lunar Network Satellite — 9/25/08

PROJECT : 1: Lunar Network Satellite — 3:34:54 PM

Basic Estimate

	Development Cost	Maintenance Cost	Total Cost
+ 1: Lunar Network Satellite	13,562,408	0	13,562,408
+ 1.1: General Avionics	9,283,919	0	9,283,919
- 1.1.1: System Initialization	114,741	0	114,741
- 1.1.2: Antenna Deployment & Positioning	212,665	0	212,665
- 1.1.3: Solar array deployment & Positioning	212,665	0	212,665
- 1.1.4: Satellite Navigation	5,155,534	0	5,155,534
- 1.1.5: Satellite Guidance	638,593	0	638,593
- 1.1.6: Propulsion System Control	65,544	0	65,544
- 1.1.7: Systems Health & Status Management	1,299,100	0	1,299,100
- 1.1.8: Power Management & Monitoring	138,245	0	138,245
- 1.1.9: Battery Regulation & Management	138,245	0	138,245
- 1.1.10: Thermal System Management	81,244	0	81,244
- 1.1.11: System Fault Detecction & Correction	953,701	0	953,701
- 1.1.12: Time Synchronization & Stamping	34,181	0	34,181
- 1.1.13: Router Management	0	0	0
- 1.1.14: Communication System Management	239,461	0	239,461
+ 1.2: Data Recorder	248,770	0	248,770
- 1.2.1: Storage Array Monitoring & Health Management	81,968	0	81,968
- 1.2.2: Synchronization of data between Data Recorders	65,088	0	65,088
- 1.2.3: Buffering data from WAN & LAN to Transmission to Earth	101,715	0	101,715
+ 1.3: Communications	214,205	0	214,205
- 1.3.1: Command Processing	109,973	0	109,973
- 1.3.2: Telemetry Processing	104,232	0	104,232
- 1.4: Navigation - Attitude Determination & Control	3,815,514	0	3,815,514

TABLE 44.—SEER-SEM RESULTS: SOFTWARE COST RISK ANALYSIS FOR LNS

SEER-SEM (TM) Software Schedule, Cost & Risk Estimation Version 7.2.21

Project : Lunar Network Satellite 9/30/08
PROJECT : 1: Lunar Network Satellite 12:46:35 PM

Project/Rollup Cost Risk

Confidence Level	Dev Cost (Independent)	Dev Cost (Dependent)
10%	11,208,956	5,687,606
20%	12,637,557	7,683,784
30%	13,942,940	9,697,728
40%	16,510,281	11,994,174
50%	18,076,553	13,938,745
60%	19,441,541	16,592,209
70%	20,654,653	20,577,633
80%	23,793,667	25,152,793
90%	29,375,258	33,237,655
Mean	18,661,549	18,085,395

(Based on 100 iteration sampling)

WBS Allocation Of Most Likely Development Cost

	Dev Cost	% of Total	(StdDev)
+ 1: Lunar Network Satellite	18,076,553		(6,598,573)
+ 1.1: General Avionics			
- 1.1.1: System Initialization	143,202	0.79%	(106,448)
- 1.1.2: Antenna Deployment & Positioning	277,612	1.54%	(197,349)
- 1.1.3: Solar array deployment & Positioning	249,471	1.38%	(147,035)
- 1.1.4: Satellite Navigation	6,682,797	36.97%	(5,126,674)
- 1.1.5: Satellite Guidance	756,550	4.19%	(490,686)
- 1.1.6: Propulsion System Control	85,712	0.47%	(57,646)
- 1.1.7: Systems Health & Status Management	1,572,621	8.70%	(1,141,517)
- 1.1.8: Power Management & Monitoring	206,260	1.14%	(178,483)
- 1.1.9: Battery Regulation & Management	182,439	1.01%	(147,796)
- 1.1.10: Thermal System Management	112,677	0.62%	(84,280)
- 1.1.11: System Fault Detecction & Correction	1,286,693	7.12%	(1,072,027)
- 1.1.12: Time Synchronization & Stamping	45,532	0.25%	(31,952)
- 1.1.13: Router Management	0	0.00%	(0)
- 1.1.14: Communication System Management	316,345	1.75%	(242,558)
+ 1.2: Data Recorder			
- 1.2.1: Storage Array Monitoring & Health Management	97,295	0.54%	(75,460)
- 1.2.2: Synchronization of data between Data Recorders	76,708	0.42%	(53,483)
- 1.2.3: Buffering data from WAN & LAN to Transmission to Ear	126,295	0.70%	(98,031)
+ 1.3: Communications			
- 1.3.1: Command Processing	126,209	0.70%	(74,032)
- 1.3.2: Telemetry Processing	141,114	0.78%	(109,302)
- 1.4: Navigation - Attitude Determination & Control	5,591,020	30.93%	(4,760,789)

7.0 Cost, Risk and Reliability

7.1 Costing

The S/C cost and life cycle cost of a mission are important products developed by the COMPASS team. Just as each subsystem lead performs a risk assessment on their system, cost risk must also be addressed. In order to evaluate uncertainties in a mission that affect cost, the COMPASS team uses the following steps to develop the cost estimate and address cost risk for that mission:

1. Take the final MEL and Map each element of the MEL to a cost spreadsheet.

2. Apply an appropriate Cost Estimating Relationship (CER) to each element and/or subsystem of the MEL. Currently, the cost estimation office has a repository of mass-based, parametric CERs that were developed using historical costs. Additional cost methodologies may be developed and applied to address and unique elements or subsystems for a given mission.
3. Discuss with the COMPASS team if any of the subsystems have inheritance from previous missions, determine the flight sparing philosophy, determine the development approach (proto-type, proto-flight, etc.), and decide if there are any other factors that may affect cost.
4. Once each element or subsystem is appropriately modeled with a CER, quantitative risk analysis is performed on the S/C cost using Monte Carlo simulation based on mass and CER uncertainties.
5. Life Cycle Costs for the mission may also be developed by adding launch services, mission operations, project management, reserves and other relevant costs for that mission.

In order to provide a cost estimate for each of the four various trades in this report, the following assumptions apply:

- A proto-type development approach is assumed for each option, except for some payload development items, which include both an engineering model and a qualification article. For a proto-flight development, 50 percent of the hardware cost for each subsystem/component is added to the development cost for hardware refurbishment.
- The cost for a single flight spare is included where appropriate.
- The life cycle cost estimates are for two satellites and assume that both satellites are launched on a single launch vehicle.
- Mission operations costs and the cost for any ground system necessary to support these satellites are not included in the estimates provided in this section.

Table 45 is a life cycle cost comparison for each of the four cases. The cost for two satellites is included for each of the cases. All costs are in FY08$M.

As seen in Table 45, NASA Project Office/Technical Oversight includes 5 percent of all other costs. Phase A costs are calculated at 5 percent of the S/C cost and fee. The satellite cost is for two satellites and is the mean estimate base on the stochastic simulation results. For the second satellite, a learning curve of 95 percent is assumed. S/C Prime Contractor Fee is 10 percent of the S/C cost. The launch services costs are provided by KSC and assume that both satellites are launched on a single platform. Reserves are calculate at 25 percent and are not applied to launch services.

TABLE 45.—LIFE CYCLE COST COMPARISON OF THE FOUR OPTIONS

Lunar Network Satellite				
FY08$M	Frozen RF	Frozen Opt	Lagrange RF	Lagrange Opt
NASA Project Office/Technical Oversight	36	36	42	43
Phase A	27	27	33	33
Two Satellites	482	495	597	605
Spacecraft Prime Contractor Fee (10%)	48	49	60	61
Launch Services	158	158	158	158
Reserves (25%)	148	152	183	185
Life Cycle Cost	899	918	1,073	1,085

*Life cycle cost if for two satellites

Table 46 to Table 49 show breakdowns of costs at the subsystem level for each of the various trades. In each case, the costs provided are in FY08$M and only include the cost of a single satellite.

TABLE 46.—SUBSYSTEM COST BREAKDOWN FOR A SINGLE SATELLITE FOR THE FROZEN RF OPTION IN FY08$M

WBS	Description	DDT&E total by $M	Flight hardware by $M	Mfg/DDT&E total by $M
6.0.1	RF Communications Package	$53.1	$23.3	$76.4
6.0.3	C&DH	$42.0	$13.2	$55.2
	Flight software	$19.8	$0.0	$19.8
6.0.4	GN&C	$15.2	$10.5	$25.7
6.0.5	Structures and Mechanical Systems	$20.4	$9.7	$30.1
6.0.6	Power System	$12.5	$3.0	$15.5
6.0.10	Propulsion (Chemical)	$1.4	$0.6	$2.0
6.0.11	Propellant Management (Chemical)	$9.6	$4.0	$13.6
6.0.12	Propellant (Chemical)	$0.0	$0.0	$0.0
6.0.13	Thermal Control (Non-Propellant)	$0.0	$1.6	$15.3
	Contingency Mass	$6.8	$2.7	$9.4
	Subtotal	$174.6	$66.0	$243.3
	Systems Integration	$95.6	$42.7	$138.3
1.6.2.2	IACO	$23.4	$32.0	$55.4
1.6.2.2	STO	$23.4	$0.0	$23.4
1.6.2.2	GSE Hardware	$8.9	$0.0	$8.9
1.6.2.2	SE&I	$19.9	$5.4	$25.3
1.6.2.1	PM	$19.9	$5.4	$25.3
1.6.2.2	LOOS	$0.0	$0.0	$0.0
	Total Prime Cost	$270.2	$108.7	$378.9

TABLE 47.—SUBSYSTEM COST BREAKDOWN FOR A SINGLE SATELLITE FOR THE FROZEN OPTICAL OPTION IN FY08$M

WBS	Description	DDT&E total by $M	Flight hardware by $M	Mfg/DDT&E total by $M
6.0.1	RF Communications Package	$32.4	$16.8	$49.2
6.0.2	Optical Communications Package	$38.1	$10.1	$48.3
6.0.3	C&DH	$36.2	$6.5	$42.8
	Flight software	$19.8	$0.0	$19.8
6.0.4	GN&C	$17.2	$10.5	$27.7
6.0.5	Structures and Mechanical Systems	$20.2	$9.6	$29.8
6.0.6	Power System	$14.9	$3.0	$18.0
6.0.10	Propulsion (Chemical)	$1.3	$0.6	$2.0
6.0.11	Propellant Management (Chemical)	$12.4	$4.6	$17.0
6.0.12	Propellant (Chemical)	$0.0	$0.0	$0.0
6.0.13	Thermal Control (Non-Propellant)	$0.0	$1.2	$10.8
	Contingency Mass	$6.7	$2.3	$8.9
	Subtotal	$189.1	$63.0	$254.4
	Systems Integration	$102.5	$41.2	$143.7
1.6.2.2	IACO	$25.0	$30.9	$55.9
1.6.2.2	STO	$25.0	$0.0	$25.0
1.6.2.2	GSE Hardware	$9.8	$0.0	$9.8
1.6.2.2	SE&I	$21.4	$5.2	$26.6
1.6.2.1	PM	$21.4	$5.2	$26.6
1.6.2.2	LOOS	$0.0	$0.0	$0.0
	Total Prime Cost	$291.6	$104.2	$395.8

TABLE 48—SUBSYSTEM COST BREAKDOWN FOR A SINGLE SATELLITE FOR THE LAGRANGE RF OPTION IN FY08$M

WBS	Description	DDT&E total by $M	Flight hardware by $M	Mfg/DDT&E total by $M
6.0.1	RF Communications Package	$73.3	$29.0	$102.3
6.0.3	C&DH	$34.4	$13.0	$47.4
	Flight software	$19.8	$0.0	$19.8
6.0.4	GN&C	$16.3	$12.7	$29.0
6.0.5	Structures and Mechanical Systems	$34.3	$16.1	$50.4
6.0.6	Power System	$15.4	$5.7	$21.1
6.0.10	Propulsion (Chemical)	$2.7	$2.4	$5.1
6.0.11	Propellant Management (Chemical)	$20.1	$6.2	$26.3
6.0.12	Propellant (Chemical)	$0.0	$0.0	$0.0
6.0.13	Thermal Control (Non-Propellant)	$0.0	$1.7	$16.6
	Contingency Mass	$6.6	$2.7	$9.3
	Subtotal	$211.4	$86.7	$307.5
	Systems Integration	$113.8	$52.8	$166.6
1.6.2.2	IACO	$27.5	$39.2	$66.8
1.6.2.2	STO	$27.5	$0.0	$27.5
1.6.2.2	GSE Hardware	$11.2	$0.0	$11.2
1.6.2.2	SE&I	$23.7	$6.8	$30.5
1.6.2.1	PM	$23.7	$6.8	$30.5
1.6.2.2	LOOS	$0.0	$0.0	$0.0
	Total Prime Cost	$325.2	$139.5	$464.8

TABLE 49—SUBSYSTEM COST BREAKDOWN FOR A SINGLE SATELLITE FOR THE LAGRANGE OPTICAL OPTION IN FY08$M

WBS	Description	DDT&E total by $M	Flight hardware by $M	Mfg/DDT&E total by $M
6.0.1	RF Communications Package	$40.7	$27.2	$67.9
6.0.2	Optical Communications Package	$38.1	$10.1	$48.3
6.0.3	C&DH	$36.2	$6.5	$42.8
	Flight software	$19.8	$0.0	$19.8
6.0.4	GN&C	$19.6	$12.7	$32.3
6.0.5	Structures and Mechanical Systems	$30.4	$14.1	$44.5
6.0.6	Power System	$18.2	$5.7	$23.9
6.0.10	Propulsion (Chemical)	$1.3	$0.6	$2.0
6.0.11	Propellant Management (Chemical)	$21.4	$6.2	$27.6
6.0.12	Propellant (Chemical)	$0.0	$0.0	$0.0
6.0.13	Thermal Control (Non-Propellant)	$0.0	$1.3	$13.1
	Contingency Mass	$5.9	$2.3	$8.2
	Subtotal	$223.7	$84.4	$310.3
	Systems Integration	$116.2	$51.7	$167.8
1.6.2.2	IACO	$28.1	$38.4	$66.5
1.6.2.2	STO	$28.1	$0.0	$28.1
1.6.2.2	GSE Hardware	$11.5	$0.0	$11.5
1.6.2.2	SE&I	$24.2	$6.6	$30.9
1.6.2.1	PM	$24.2	$6.6	$30.9
1.6.2.2	LOOS	$0.0	$0.0	$0.0
	Total Prime Cost	$339.8	$136.1	$475.9

7.2 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. Risks are 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.

7.2.1 Assumptions

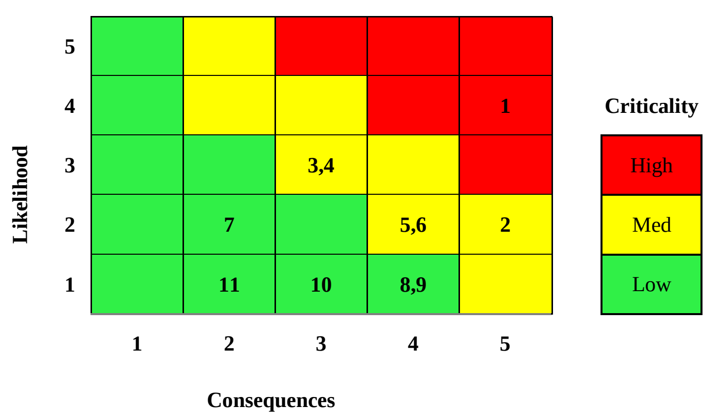
Risk attributes are based on CEV risk values. Risk List is not based on trends or criticality. Some mitigation plans are offered as suggestions

7.2.2 Risk List and Summary

These risks, with proper pro-active planning can be mitigated early to avoid becoming problems late in the development life cycle. The risks gathered by the design team are captured in Table 50.

TABLE 50.—RISK SUMMARY TABLE

No.	L×C	Team	Risk title
1	4×5	Communications	Multiple data channel feed through
2	2×5	Structures	Potential structural failure
3	3×3	Propulsion	RCS thruster plume impingement
4	3×3	Communications	LNS antenna pointing
5	2×4	Operations	Lunar insertion burn accuracy
6	2×4	Avionics	Failure of avionics to recover from switch to backup
7	2×2	Power	Science data memory overflow
8	1×4	Avionics	Major solar burst radiation damage
9	1×4	Operations	Dual launch impacts on Constellation
10	1×3	GN&C	Navigation sensor view blockage
11	1×2	Power	Freezing of liquid propellant in distribution system



7.2.3 Trade Space Iterations

Two different orbital constellation options, Frozen orbit and Lagrange orbit, of the available trade space of orbit types were investigated in order to weigh the cost and benefits of the communications coverage available to a lunar South Pole Constellation human lunar landing site. The summary masses by subsystem are shown in Figure 59. For each of these orbital options, two communications technology options were measured against one another: an RF communications system and an Optical communications system as summarized in Table 51.

Figure 60 shows the comparison of the four LNS spacecraft designs by total wet mass. Each subsystem is called out in a separate color. It can be seen that the Lagrange RF spacecraft is the greatest wet mass of the four designs.

Table 52 is the top level MEL and the system summary rack up of the case in order to show top level masses, total propellant, additional growth carried at the system level for each case. In the optical cases, the optical comm. only provided the DTE communications link while RF communications were still used to the lunar surface.

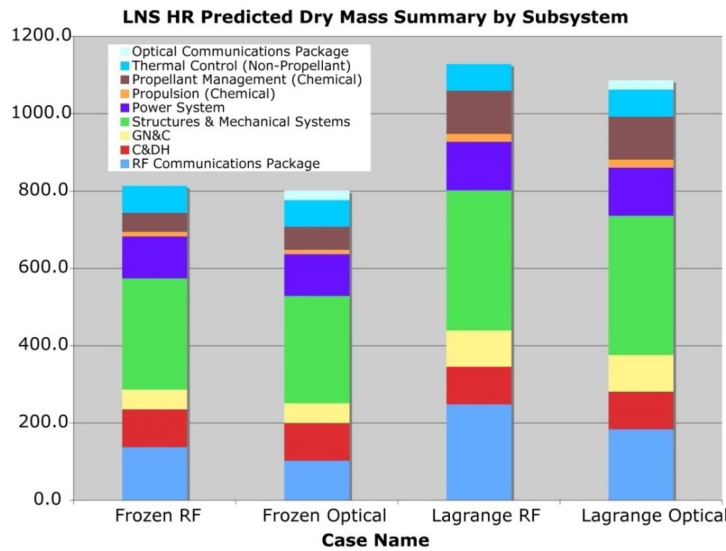


Figure 59.—LNS trade space total dry mass comparison by subsystem.

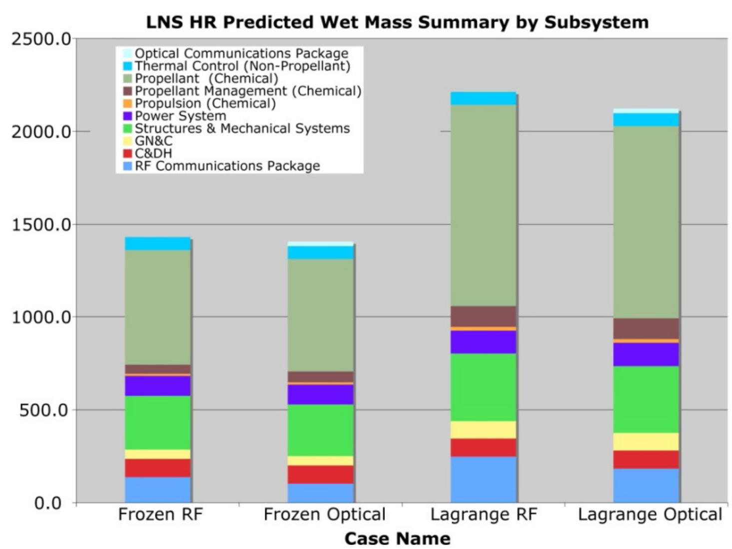


Figure 60.—LNS trade space wet mass comparison by subsystem.

TABLE 51.—LNS-HR TRADE SPACE SUMMARY

WBS no.	Description Case—LNS-HR	Frozen RF	Frozen optical	Lagrange RF	Lagrange optical
	Mass Type	Predicted mass, kg	Predicted mass, kg	Predicted mass, kg	Predicted mass, kg
6.0	Lunar Network Satellite-High Rate	1430.9	1406.1	2213.1	2121.7
6.0.1	RF Communications Package	138.0	102.8	248.2	184.1
6.0.2	Optical Communications Package	0.0	23.8	0.0	23.8
6.0.3	C&DH	97.4	97.4	97.4	97.4
6.0.5	GN&C	51.0	51.0	94.2	94.2
6.0.5	Structures & Mechanical Systems	288.4	277.5	362.8	360.6
6.0.6	Power System	107.9	107.9	124.8	124.8
6.0.10	Propulsion (Chemical)	11.7	11.7	20.5	20.5
6.0.11	Propellant Management (Chemical)	49.5	59.2	111.2	111.2
6.0.12	Propellant (Chemical)	617.6	605.4	1084.6	1035.6
6.0.13	Thermal Control (Non-Propellant)	69.5	69.5	69.5	69.5

TABLE 52.—LNS TRADE SPACE MASS RACK-UP

Spacecraft Master Equipment List Rack-up (Mass)

	Orbit Type	Frozen Orbit		Lagrange Orbit	
	Communications Type	RF	Optical	RF	Optical
WBS	Main Subsystems	Predicted (Total) Mass (kg)	Predicted (Total) Mass (kg)	Predicted (Total) Mass (kg)	Predicted (Total) Mass (kg)
6.0	Lunar Network Satellie-High Rate	1431	1406	2213	2122
6.0.1	RF Communications Package	138	103	248	184
6.0.2	Optical Communications Package	0	24	0	24
6.0.3	Command & Data Handling (C&DH)	97	97	97	97
6.0.4	Guidance, Navigation and Control System (GN&C)	51	51	94	94
6.0.5	Structures & Mechanical Systems	288	277	363	361
6.0.6	Power System	108	108	125	125
6.0.7	*Propulsion (Electric)*	*0*	*0*	*0*	*0*
6.0.8	*Propellant Management (EP)*	*0*	*0*	*0*	*0*
6.0.9	*Propellant (EP)*	*0*	*0*	*0*	*0*
6.0.10	Propulsion (Chemical)	12	12	21	21
6.0.11	Propellant Management (Chemical)	50	59	111	111
6.0.12	Propellant (Chemical)	618	605	1085	1036
6.0.13	Thermal Control (Non-Propellant)	69	69	69	69
	Estimated Spacecraft Dry Mass	813	801	1129	1086
	Estimated Spacecraft Wet Mass	1431	1406	2213	2122
	Estimated Spacecraft Inert Mass (for traj.)	890	871	1228	1174
	Dry Mass Desired System Level Growth	872	854	1197	1144
	Mass Margin (carried at system level)	58	53	68	58
	Total Wet Mass with Growth	1489	1459	2281	2179
	Launch Vehicle	*Atlas V 511*	*Atlas V 511*	*Atlas V 521*	*Atlas V 521*
	Available Launch Performance to C3 (kg)	3915	3915	4710	4710
	Total Number of LNS-HR Satellites	2	2	2	2
	Total mass of Two LNS-HR satellites	2978	2918	4563	4359
	Launch margin available (kg)	937	997	147	351

7.3 Case 1—Frozen RF

Case 1 designed an RF communications driven S/C in frozen orbit (Table 53). Launching two LNS-HR satellites on an Atlas 511 with a performance to $C_3 = -2\ \text{km}^2/\text{sec}^2$ (lunar) of 3915 kg allows for a 937 kg margin. An additional 58 kg are carried at the system level in order to reach a dry mass growth of 30 percent.

TABLE 53.—LNS FROZEN RF SYSTEM SUMMARY

COMPASS study: Near Earth Asteroid Sampler Mission				**Study Date**	**9/19/08**
GLIDE container:	*LunarRelaySat: LNS_Frozen_RF*				
Spacecraft Master Equipment List Rack-up (Mass)				**COMPASS S/C Design**	
WBS	**Main Subsystems**	**Basc (CBE) Mass (kg)**	**Growth (kg)**	**Predicted (Total) Mass (kg)**	**Aggregate Growth (%)**
6.0	**Lunar Network Satellie-High Rate**	**1288.1**	**142.8**	**1430.9**	
6.0.1	RF Communications Package	123.1	14.9	**138.0**	12%
6.0.2	Optical Communications Package	0.0	0.0	**0.0**	TBD
6.0.3	Command & Data Handling (C&DH)	77.2	42.5	**97.4**	55%
6.0.4	Guidance, Navigation and Control System (GN&C)	42.5	8.5	**51.0**	20%
6.0.5	Structures & Mechanical Systems	233.7	54.7	**288.4**	23%
6.0.6	Power System	83.0	24.9	**107.9**	30%
6.0.7	Propulsion (Electric)	0.0	0.0	**0.0**	TBD
6.0.8	Propellant Management (EP)	0.0	0.0	**0.0**	TBD
6.0.9	Propellant (EP)	0.0			
6.0.10	Propulsion (Chemical)	10.6	1.1	**11.7**	10%
6.0.11	Propellant Management (Chemical)	43.9	5.7	**49.5**	13%
6.0.12	Propellant (Chemical)	617.6			
6.0.13	Thermal Control (Non-Propellant)	56.5	13.0	**69.5**	23%
	Estimated Spacecraft Dry Mass	**670**	**143**	**813**	**21%**
	Estimated Spacecraft Wet Mass	1288	165	1431	
	Estimated Spacecraft Inert Mass (for traj.)	689	201	**889.7**	
System LeveL Growth Calculations					**Total Growth**
	Dry Mass Desired System Level Growth	670	201	**871.6**	**30%**
	Mass Margin (carried at system level)		**58**		**9%**
	Total Wet Mass with Growth	1288	201	**1489.2**	
	Available Launch Performance to C3 (kg)			3915.0	kg
	Total Number of LNS-HR Satellites			2	
	Total mass of Two LNS-HR satellites			2978.5	
	Launch margin available (kg)			**936.5**	656

7.4 Case 2—Frozen Optical

Case 2 designed an Optical communications driven S/C with RF communications, where necessary, in frozen orbit. Launching two LNS-HR satellites on an Atlas 511 with a performance to $C_3 = –2\ km^2/sec^2$ (lunar) of 3915 kg allows for a 997 kg margin (Table 54). An additional 53 kg are carried at the system level in order to reach a dry mass growth of 30 percent.

TABLE 54.—LNS FROZEN OPTICAL SYSTEM SUMMARY

COMPASS study: Near Earth Asteroid Sampler Mission				**Study Date**	**9/22/08**
GLIDE container:	*LunarRelaySat: LNS_Frozen_Optical*				
Spacecraft Master Equipment List Rack-up (Mass)				**COMPASS S/C Design**	
WBS	**Main Subsystems**	**Basc (CBE) Mass (kg)**	**Growth (kg)**	**Predicted (Total) Mass (kg)**	**Aggregate Growth (%)**
6.0	**Lunar Network Satellie-High Rate**	**1261.9**	**144.2**	**1406.1**	
6.0.1	RF Communications Package	92.3	10.5	**102.8**	11%
6.0.2	Optical Communications Package	18.1	5.8	**23.8**	32%
6.0.3	Command & Data Handling (C&DH)	77.2	42.5	**97.4**	55%
6.0.4	Guidance, Navigation and Control System (GN&C)	42.5	8.5	**51.0**	20%
6.0.5	Structures & Mechanical Systems	225.0	52.5	**277.5**	23%
6.0.6	Power System	83.0	24.9	**107.9**	30%
6.0.7	Propulsion (Electric)	0.0	0.0	**0.0**	TBD
6.0.8	Propellant Management (EP)	0.0	0.0	**0.0**	TBD
6.0.9	Propellant (EP)	0.0			
6.0.10	Propulsion (Chemical)	10.6	1.1	**11.7**	10%
6.0.11	Propellant Management (Chemical)	51.5	7.8	**59.2**	15%
6.0.12	Propellant (Chemical)	605.4			
6.0.13	Thermal Control (Non-Propellant)	56.5	13.0	**69.5**	23%
	Estimated Spacecraft Dry Mass	**657**	**144**	**801**	**22%**
	Estimated Spacecraft Wet Mass	1262	167	1406	
	Estimated Spacecraft Inert Mass (for traj.)	674	197	**871.3**	
System LeveL Growth Calculations					**Total Growth**
	Dry Mass Desired System Level Growth	657	197	**853.5**	**30%**
	Mass Margin (carried at system level)		**53**		**8%**
	Total Wet Mass with Growth	1262	197	**1458.9**	
	Available Launch Performance to C3 (kg)			3915.0	kg
	Total Number of LNS-HR Satellites			2	
	Total mass of Two LNS-HR satellites			2917.8	
	Launch margin available (kg)			**997.2**	698

7.5 Case 3—Lagrange RF

Case 3 designed an RF communications driven S/C in a Lagrange orbit. Launching two LNS-HR satellites on an Atlas 521 with a performance to $C_3 = -2\ \text{km}^2/\text{sec}^2$ (lunar) of 4710 kg allows for a 147 kg margin (Table 55). An additional 68 kg is carried at the system level in order to reach a dry mass growth of 30 percent.

TABLE 55.—LNS LAGRANGE ORBIT RF SYSTEM SUMMARY

COMPASS study: Near Earth Asteroid Sampler Mission				Study Date	9/29/08
GLIDE container:	LunarRelaySat: LNS_Lagrange_RF				
Spacecraft Master Equipment List Rack-up (Mass)				COMPASS S/C Design	
WBS	Main Subsystems	Basc (CBE) Mass (kg)	Growth (kg)	Predicted (Total) Mass (kg)	Aggregate Growth (%)
6.0	Lunar Network Satellie-High Rate	2005.2	207.9	2213.1	
6.0.1	RF Communications Package	204.1	44.1	248.2	22%
6.0.2	Optical Communications Package	0.0	0.0	0.0	TBD
6.0.3	Command & Data Handling (C&DH)	77.2	20.2	97.4	26%
6.0.4	Guidance, Navigation and Control System (GN&C)	78.5	15.7	94.2	20%
6.0.5	Structures & Mechanical Systems	291.0	71.8	362.8	25%
6.0.6	Power System	96.0	28.8	124.8	30%
6.0.7	Propulsion (Electric)	0.0	0.0	0.0	TBD
6.0.8	Propellant Management (EP)	0.0	0.0	0.0	TBD
6.0.9	Propellant (EP)	0.0			
6.0.10	Propulsion (Chemical)	18.7	1.9	20.5	10%
6.0.11	Propellant Management (Chemical)	98.6	12.6	111.2	13%
6.0.12	Propellant (Chemical)	1084.6			
6.0.13	Thermal Control (Non-Propellant)	56.5	13.0	69.5	23%
	Estimated Spacecraft Dry Mass	921	208	1129	23%
	Estimated Spacecraft Wet Mass	2005	208	2213	
	Estimated Spacecraft Inert Mass (for traj.)	952	276	1228.1	
System LeveL Growth Calculations					Total Growth
	Dry Mass Desired System Level Growth	921	276	1196.8	30%
	Mass Margin (carried at system level)		68		7%
	Total Wet Mass with Growth	2005	276	2281.4	
	Available Launch Performance to C3 (kg)			4710.0	kg
	Total Number of LNS-HR Satellites			2	
	Total mass of Two LNS-HR satellites			4562.7	
	Launch margin available (kg)			147.3	103

7.6 Case 4—Lagrange Optical

Case 4 designed an Optical communications driven S/C with RF communications, where necessary, in a Lagrange orbit (Table 56). Launching two LNS-HR satellites on an Atlas 521 with a performance to C_3 = –2 km^2/sec^2 (lunar) of 4710 kg allows for a 351 kg margin. An additional 58 kg is carried at the system level in order to reach a dry mass growth of 30 percent.

TABLE 56.—LNS LAGRANGE ORBIT OPTICAL SYSTEM SUMMARY

COMPASS study: Near Earth Asteroid Sampler Mission				**Study Date**	**9/30/08**
GLIDE container:	*LunarRelaySat: LNS_Lagrange_Optical*				
Spacecraft Master Equipment List Rack-up (Mass)				**COMPASS S/C Design**	
WBS	**Main Subsystems**	**Basc (CBE) Mass (kg)**	**Growth (kg)**	**Predicted (Total) Mass (kg)**	**Aggregate Growth (%)**
6.0	**Lunar Network Satellie-High Rate**	**1915.4**	**206.3**	**2121.7**	
6.0.1	RF Communications Package	147.0	37.1	**184.1**	25%
6.0.2	Optical Communications Package	18.1	5.8	**23.8**	32%
6.0.3	Command & Data Handling (C&DH)	77.2	20.2	**97.4**	26%
6.0.4	Guidance, Navigation and Control System (GN&C)	78.5	15.7	**94.2**	20%
6.0.5	Structures & Mechanical Systems	289.3	71.3	**360.6**	25%
6.0.6	Power System	96.0	28.8	**124.8**	30%
6.0.7	Propulsion (Electric)	0.0	0.0	**0.0**	TBD
6.0.8	Propellant Management (EP)	0.0	0.0	**0.0**	TBD
6.0.9	Propellant (EP)	0.0			
6.0.10	Propulsion (Chemical)	18.7	1.9	**20.5**	10%
6.0.11	Propellant Management (Chemical)	98.6	12.6	**111.2**	13%
6.0.12	Propellant (Chemical)	1035.6			
6.0.13	Thermal Control (Non-Propellant)	56.5	13.0	**69.5**	23%
	Estimated Spacecraft Dry Mass	**880**	**206**	**1086**	**23%**
	Estimated Spacecraft Wet Mass	1915	206	2122	
	Estimated Spacecraft Inert Mass (for traj.)	910	264	**1173.8**	
System LeveL Growth Calculations					**Total Growth**
	Dry Mass Desired System Level Growth	880	264	**1143.7**	**30%**
	Mass Margin (carried at system level)		**58**		**7%**
	Total Wet Mass with Growth	1915	264	**2179.3**	
	Available Launch Performance to C3 (kg)			4710.0	kg
	Total Number of LNS-HR Satellites			2	
	Total mass of Two LNS-HR satellites			4358.7	
	Launch margin available (kg)			**351.3**	246

Appendix A.—Acronyms and Abbreviations

ACS	Attitude Control System
AIAA	American Institute for Aeronautics and Astronautics
Al	aluminum
ANSI	American National Standards Institute
ATCS	Active Thermal Control System
BER	bit error ratio
BOL	beginning of life
C&DH	Command and Data Handing
C&T	Command and Telemetry
CAD	computer aided design
CBE	current best estimate
CCAFS	Cape Canaveral Air Force Station
CER	Cost Estimating Relationship
Comm	Communications
COTS	commercial off-the-shelf
dLQR	design linear-quadratic regulator
DMR	design for minimum risk
DPAF	Dual Payload Attach Fitting
DPSK-RZ	differential phase-shift keying-return-to-zero
DSN	Deep Space Network
DTE	direct to Earth
EELV	Evolved Expendable Launch Vehicles
EIRP	equivalent isotropically radiated power
EOL	end of life
FEC	forward error correction
FOM	figure of merit
GaAs	gallium arsenide
GEO	geosynchronous Earth orbit
GN&C	Guidance, Navigation and Control
GRC	NASA Glenn Research Center
GSFC	NASA Goddard Space Flight Center
hab	habitat
He	helium
HQ	NASA Headquarters
IMDC	I M Design Center
IMU	inertial measurement unit
ISRU	in situ resource utilization
ISS	International Space Station
JPL	NASA Jet Propulsion Laboratory
KSC	NASA Kennedy Space Center

LAN	local area network
LCT	Lunar Communications Terminal
Li	lithium
LNS	Lunar Network Satellite
LOI	lunar orbit insertion
LPDC	low-density parity check
LRO	Lunar Reconnaissance Orbiter
LRS	Lunar Relay Station
LSP	Launch Service Program
LSTO	Launch Service Task Order
LV	launch vehicle
LVA	launch vehicle adaptor
MGA	mass growth allowance
MLI	multilayer insulation
MMOD	Micrometeoroid and Orbital Debris
MPU	Makeup Power Unit
Nav	navigation
NLS	NASA Launch Services
OMS	Orbital Maneuvering System
OTS	off-the-shelf
PMAD	Power Management and Distribution
PPM	Pulse Position Modulation
QPSK	quadrature phase-shift keying
RAAN	right ascension of the ascending node
RF	radio frequency
RFI	Request for Information
RHU	radioisotope heater unit
RTOS	real-time operating system
S/C	spacecraft
SA	solar array
SCAN	NASA Space Communications and Navigation
SLOC	source lines of code
SMA	semimajor axis
SNAP	Spacecraft N-Body Analysis Program
SOA	state-of-the-art
SOAP	Satellite Orbit Analysis Program
SOHO	Solar and Heliospheric Observatory
SPA	South Pole-Aitken
SPU	Solar Power Unit
TBD	to be discussed
TBR	to be resolved

TDRSS	Tracking and Data Relay Satellite System
Ti	titanium
TLI	translunar injection
TRL	technology readiness level
TT&C	Telemetry, Tracking and Command
TWTA	traveling wave tube amplifier
USAF	U.S. Air Force
USO	ultra-stable oscillator
WAN	wide area network
WMAP	Wilkinson Microwave Anisotropy Probe

Appendix B.—Rendered Images

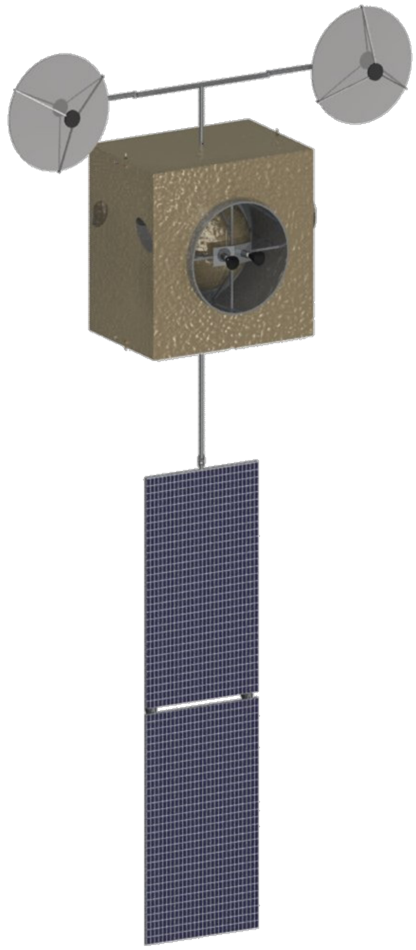


Figure B.1—LNS-HR satellite, frozen orbit,
deployed configuration—thruster view.

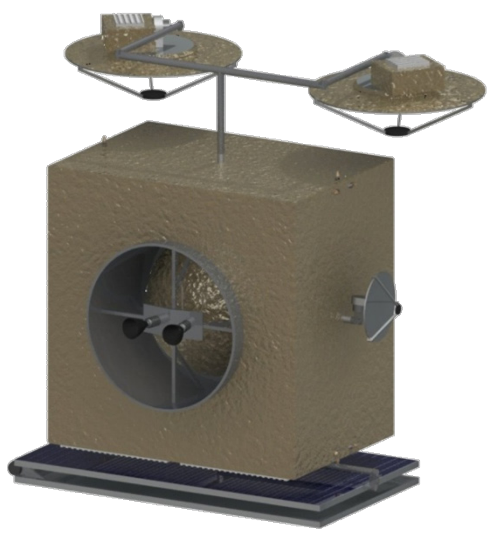


Figure B.2—LNS-HR satellite, frozen orbit,
stowed configuration.

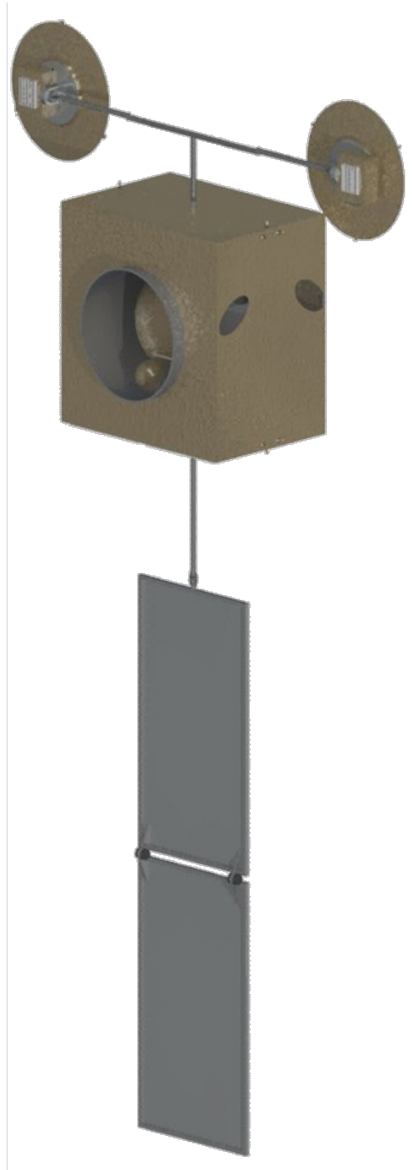


Figure B.3—LNS-HR satellite, frozen orbit,
deployed configuration—antenna view.

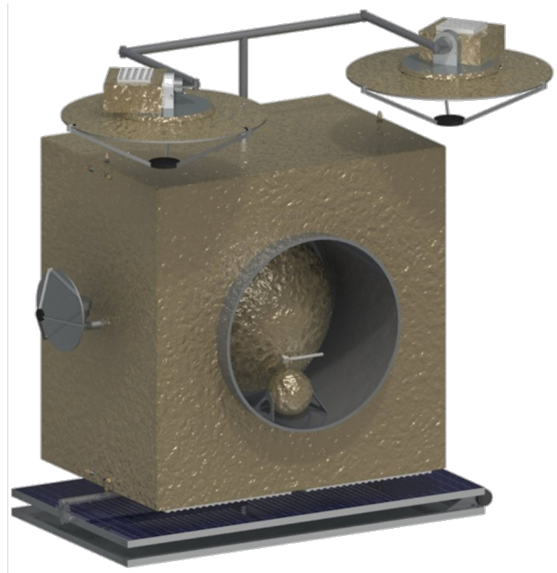


Figure B.4—LNS-HR Satellite, frozen orbit, stowed configuration—tank face.

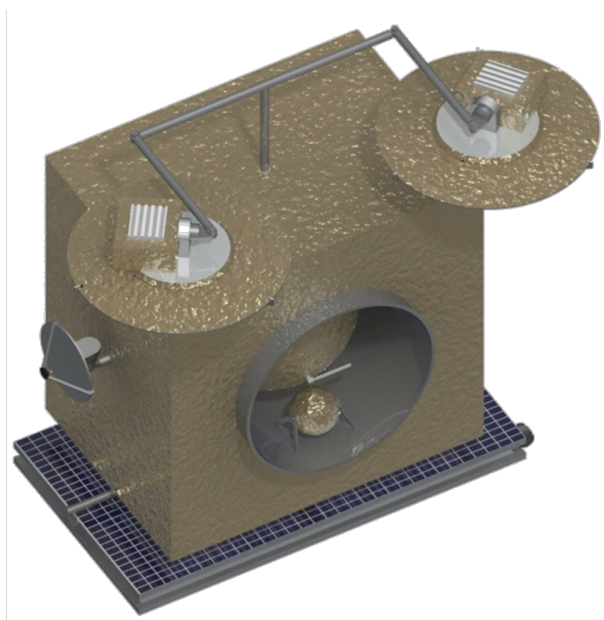


Figure B.5—LNS-HR satellite, frozen orbit, stowed configuration—top surface.

Appendix C.—LNS Lagrange Orbit Rendered Design Drawings

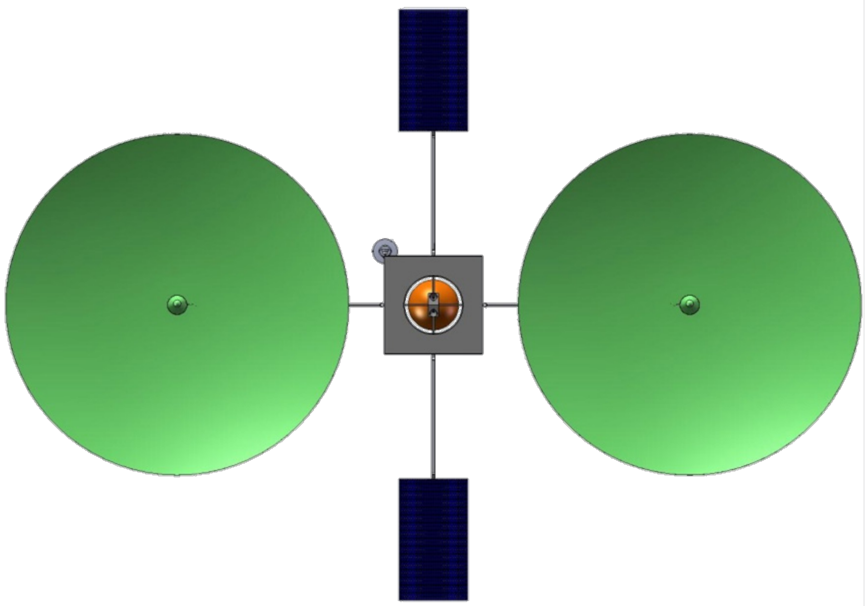


Figure C.1—LNS-HR satellite, Lagrange orbit, deployed configuration.

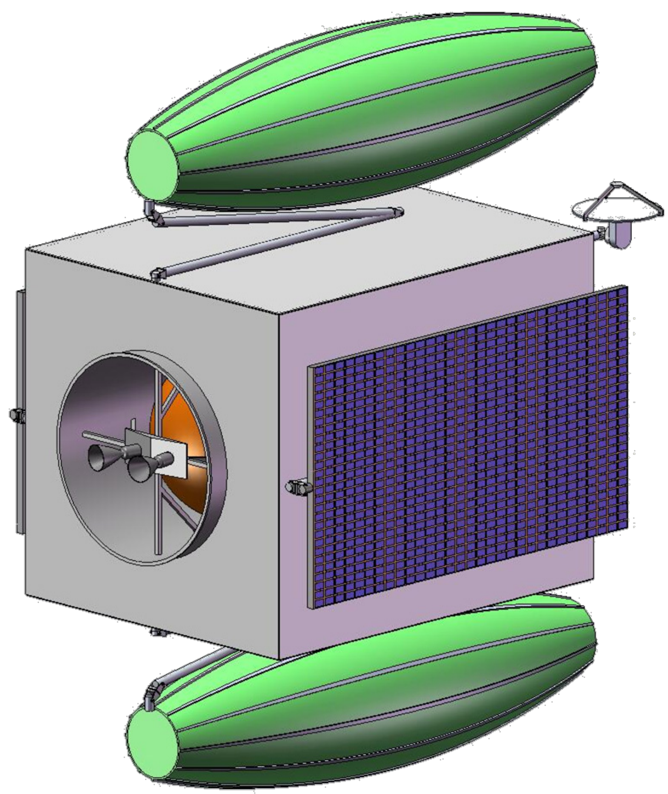


Figure C.2—LNS-HR satellite, Lagrange orbit, stowed configuration.

Appendix D.—SNAP Input File for Frozen Orbit Trajectory Propagation Analysis

The following text is the input file for the SNAP trajectory propagation code used to model the frozen lunar orbit in Section 2.5.2. This input file should work with snap version xx.

---------begin input deck here, do not include this line------------------

```
&input

@ There is only 1 phase in this example

nphase=1

precision='double'

@ Input state
 tin(1)=20180101.
 tin(2)=000001.0
 tintyp = 'UTC'
 dut = 65.184d0

 xintyp(1)='kepler'
 xintyp(2)='tod'
 xin(1) = 9751.0d0
 xin(2) = 1.d-7
 xin(3) = 39.2315d0
 xin(4) = 50.0d0
 xin(5) = 1.707547293e-6
 xin(6) = 44.9999996d0
 xin(7) = 10000.d0

@----Print Options--------------------------------------
 printflg(1)='on'
 printflg(2)='on'
 printflg(3)='on'
 printflg(4)='on'
 printflg(5)='on'
 printflg(6)='on'
 printflg(7)='on'
 printflg(26)='on'
 printflg(27)='on'
 printflg(28)='on'
@----End Print Options----------------------------------

 scrflg='on'
 subsurface_stop = 'on'
 Moon_equinox_method = 1

@=====Begin Phase 1===================================

 title(1)='Propagate in LEO'
 strflg(1)='lvlh'
 cbody(1) = 'Moon'
 obody(1,1) = 'Earth'
 obody(2,1) = 'Sun'
@ obody(3,1) = 'jupiter barycenter'

@--- Stop Condition
@-- Propagate for 1 day
```

```
 stopphs(0,1)='dtime'
 stopval(1,0,1)=2557.0d0
 stopval(2,0,1)=86400.0d0
@ stopval(2,0,1)=0.0d0
 stoptol(0,1)=1.0d-9

@--- Integration Method

 inttyp(1)='cart'
 relerr(1)=1.d-10
 abserr(1)=1.d-10

@-- Output Conditions

 stopphs(1,1)='dtime'
 stopval(1,1,1)=1.d0
 stopval(2,1,1)=86400.d0
 stopflg(1,1)='on'

@ lharm(1)=70
@ mharm(1)=70
@ nrmlzd(1)='yes'

@=====End Phase 1=====================================

@-----Load PCK Files------------------------------------------------
 data_fnum=2
 data_file(1)='pck_files/planets-Moon.tpc'
 data_file(2)='pck_files/Moon_cs.tpc'

@-----End loading PCK files------------------------------------------


@-----Load Ephemeris Files------------------------------
 eph_fnum=1
 ephfil(1)='bsp_files/de405.bsp'
@-----End loading Ephemeris Files-----------------------

 &end
```
---------end input deck here, do not include this line-----------------

Appendix E.—Study Participants

Lunar Network Satellite–High Rate (LNS–HR)—Design Session			
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13. SUPPLEMENTARY NOTES

14. ABSTRACT
Two design options were explored to address the requirement to provide lunar piloted missions with continuous communications for outpost and sortie missions. Two unique orbits were assessed, along with the appropriate spacecraft (S/C) to address these requirements. Both constellations (with only two S/C each) provide full time coverage (24 hr/7 d) for a south polar base and also provide continuous 7 day coverage for sorties for specified sites and periodic windows. Thus a two-satellite system can provide full coverage for sorties for selected windows of opportunity without reconfiguring the constellation.

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