NASA/TM—2012-217140

CD–2007–12

COMPASS Final Report:
Lunar Relay Satellite (LRS)

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April 2012

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COMPASS Final Report: Lunar Relay Satellite (LRS)

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1.0   Executive Summary

The Lunar Relay Satellite (LRS) COllaborative Modeling and Parametric Assessment of Space Systems (COMPASS) session was tasked to design a satellite to orbit in an elliptical lunar polar orbit to provide relay communications between lunar South Pole assets and the Earth. The design included a complete master equipment list, power requirement list, configuration design, and brief risk assessment and cost analysis.

The LRS is a half-TDRSS sized box spacecraft, which provides communications and navigation relay between lunar outposts (via Lunar Communications Terminals (LCT)) or Sortie parties (with user radios) and large ground antennas on Earth. The LRS consists of a spacecraft containing all the communications and avionics equipment designed by NASA Jet Propulsion Laboratory's (JPL) Team X to perform the relay between lunar-based assets and the Earth. The satellite design is a standard box truss spacecraft design with a thermal control system, 1.7 m solar arrays for 1 kWe power, a 1 m diameter Ka/S band dish which provides relay communications with the LCT, and a Q-band dish for communications to/from the Earth based assets. While JPL's Team X and Goddard Space Flight Center's (GSFC) I M Design Center (IMDC) have completed two other LRS designs, this NASA Glenn Research Center (GRC) COMPASS LRS design sits between them in terms of physical size and capabilities.
This LRS was used in conjunction with the LCT designed by COMPASS and documented in COMPASS final report CD-2007-11.

Figure 1 is a notional artist rending of the LCT and LRS systems in use at the lunar South Pole. Figure 2 shows the LRS concept design.

2.0   Study Background and Assumptions

2.1   Introduction

During crewed missions to the Moon, there is a need for redundant communications paths to Earth. For early, shorter duration missions, direct to Earth (DTE) and relay provides dual paths. For missions longer than the 2-wk two LRS satellites are required for continuous coverage.

During the crewed phase, communication and navigation services are primarily provided via relay, in conjunction with a surface LCT, in support of the outpost. This LCT also provides outpost communications interconnectivity using 802-network technology.

In all cases, the nominal communications operations include all outpost vicinity communications routed through the LCT to other local users or to the relay. The LRS and LCT will provide these basic communication and navigation services:

- Forward command
- Return mission data and telemetry
- One- and two-way ranging and Doppler tracking
- Beacon signals
- In situ routing

This design session concentrated on designing the LRS.
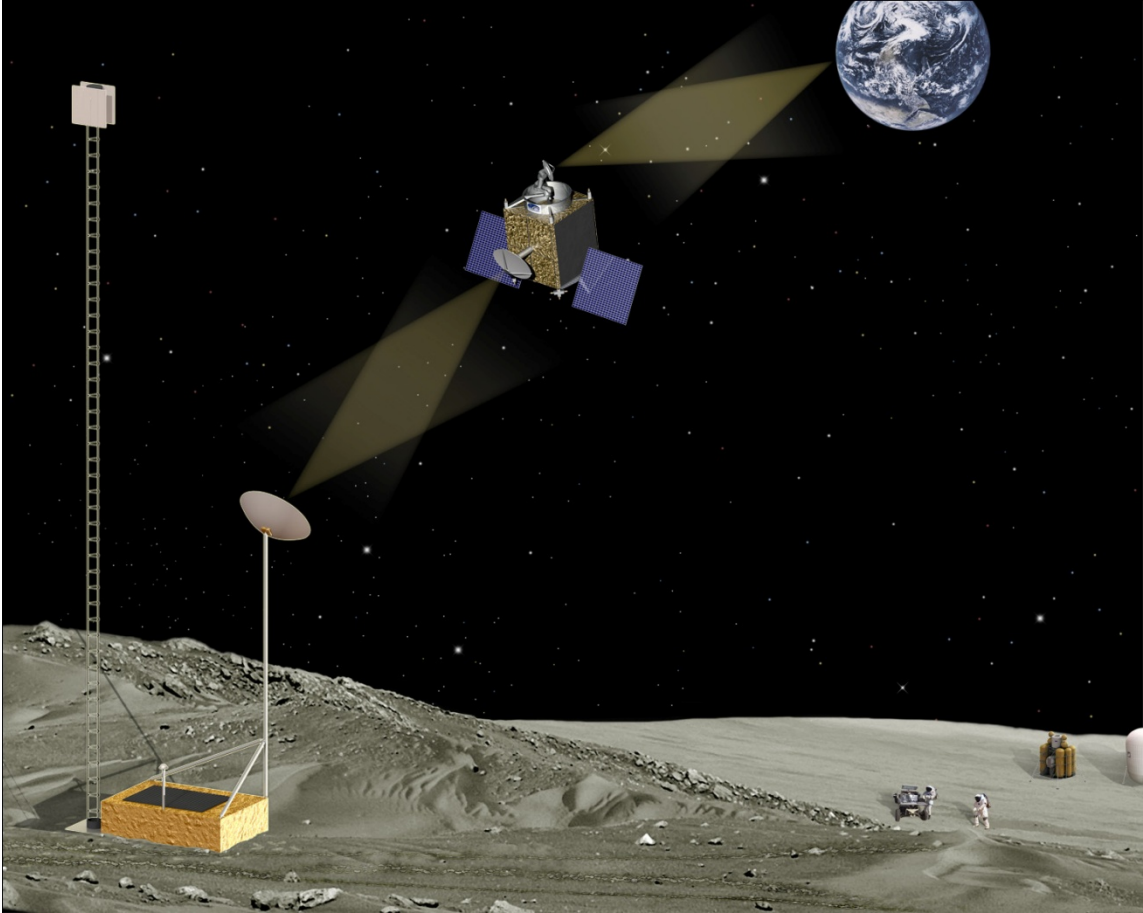
Figure 1.—An artists rending of the LCT and LRS systems in use at the lunar South Pole.
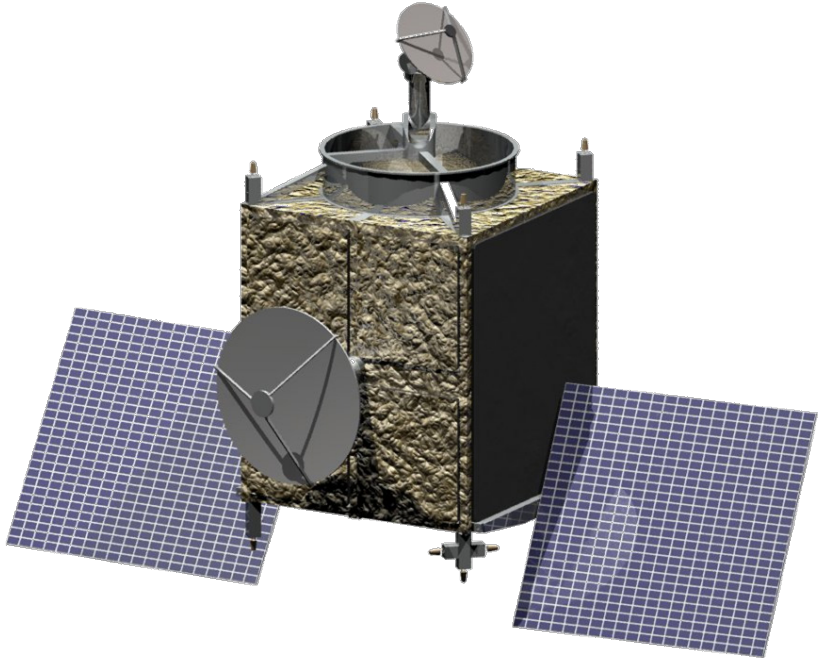
Figure 2.—LRS concept design.

2.1.1 Human Phase Relay (LAT 1 Baseline Concept)

For the Crewed Phase, Communication (Comm) and Navigation (Nav) services in the lunar vicinity will be provided primarily by LRS, backed up by Earth Based DTE antennas, which also provide service during Cis-lunar operations. The Surface Systems will provide wide area surface networking and surface links to the relay satellites.

2.1.2 Concept of Operations

Nominally, all communications during crew operations at the Moon will go through a relay. A single relay will provide about 60+ percent availability, or about 17 hr a day, from the baseline frozen orbit. When not in view of LRS or LCT, a user has the capability to communicate with Earth. (But for missions longer than 8 d, LRS is needed for daily Earth contact. LRS is also needed for redundant paths.) The LRS will broadcast a navigation signal and provide timing services. The LRS will have the capability to "locally" route communications in the lunar vicinity.

2.2 Design Assumptions and Requirements

2.3 Assumptions

Two LRS will be designed to provide Comm/Nav for piloted lunar missions, to be launched in 2018 and 2020, operating for a 7-yr mission with a design ground rule of a 10-yr lifetime.

2.3.1 System Level Requirements

The requirement for this design is to produce a satellite that meets the communications requirements. The LRS provides communications to all lunar surface elements (south pole outpost site focused), but also global coverage as necessary, robotic, rover, hab, ISRU in-situ resource utilization (ISRU), Space Power Unit (SPU)/Makeup Power Unit (MPU), 500-km diameter circle (S band), 20-km Ka band. **Figures of Merit (FOM)** for this study design were: Cost, Mass, Flexibility

2.4 Growth, Contingency and Margin Policy

Mass Growth: The COMPASS team uses the ANSI/AIAA R–020A–1999, *Recommended Practice for Mass Properties Control for Satellites, Missiles, and Launch Vehicles*. Table 1 shows the growth percentage based on design maturity and design area. Percent Mass Growth divided by design maturity and subsystem.

Once the percent growth factors are applied on a subsystem per subsystem basis, the total system growth of the design is calculated. The COMPASS system desired total growth to be 30 percent, and an additional growth is carried at the system level in order to add up to a total system growth of 30 percent on the dry mass of the system. Note that for designs requiring propellant, growth in propellant is either carried in the propellant itself or in the ΔV used to calculate the propellant necessary to fly a mission.

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

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

Power Growth: The COMPASS team uses a 30 percent growth percentage on power requirements.

TABLE 1.—COMPASS MASS GROWTH ALLOWANCE STANDARD

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

2.5 Mission Description

The LRS mission launches on an Atlas V 501 launch vehicle to a C_3 of approximately –2 km²/s². This places the LRS on a direct trajectory to the Moon with a trip time of approximately 4 d. The LRS places itself into a lunar frozen orbit for long duration operation with minimal station keeping required. From this orbit, with a long dwell time over the lunar south pole, the LRS completes its primary mission providing communications coverage to assets on the lunar surface and relaying that information back to ground stations on the Earth. Two LRS satellites are delivered to the 12 hr frozen orbit, out of phase with each other by 180°.

2.5.1 Mission Requirements

- Deliver sufficient mass to the lunar frozen orbit.
- Provide as much communications coverage as possible to the lunar South Pole.
- Remain in orbit for a sufficiently long period of time.
- Dispose of spacecraft safely.

2.5.2 Launch Vehicle Data and Trades

Table 2 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.

TABLE 2.—LAUNCH SERVICE COSTS FROM KSC

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
4,711 to 5,400 kg	~$224M	Delta IV Medium +

2.5.3 Launch Vehicle Performance and Cost

Assumptions:

- Performance range based on a $C_3 = -2$ km^2/s^2 (lunar).
- Used a launch date of January 1, 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, which result in a slightly reduced amount of conservatism based on some recent acquisitions.
- All costs are estimated in real-year dollars 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 Launch Service Program.
- 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 (AF) infrastructure cost allocations, 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.

2.5.4 Mission Assumptions

The ground rules for the mission design were based on work done and assumptions levied on the JPL and GSFC design sessions on alternate LRS systems.

- Launch vehicle: Atlas V 401
- Lunar orbit parameters:
  - Periapse altitude: 718 km
  - Apoapse altitude: 8090 km
  - Inclination: 57.7°
  - Argument of periapse: 90°
- Frozen orbit allows the relay satellite to maintain its apoapse in the lunar southern hemisphere without the need for station keeping
  - Semi-major axis, inclination and argument of perigee will vary but they remain bounded

Figure 3 shows the direct injection trajectory as displayed in the lunar inertial coordinate frame.
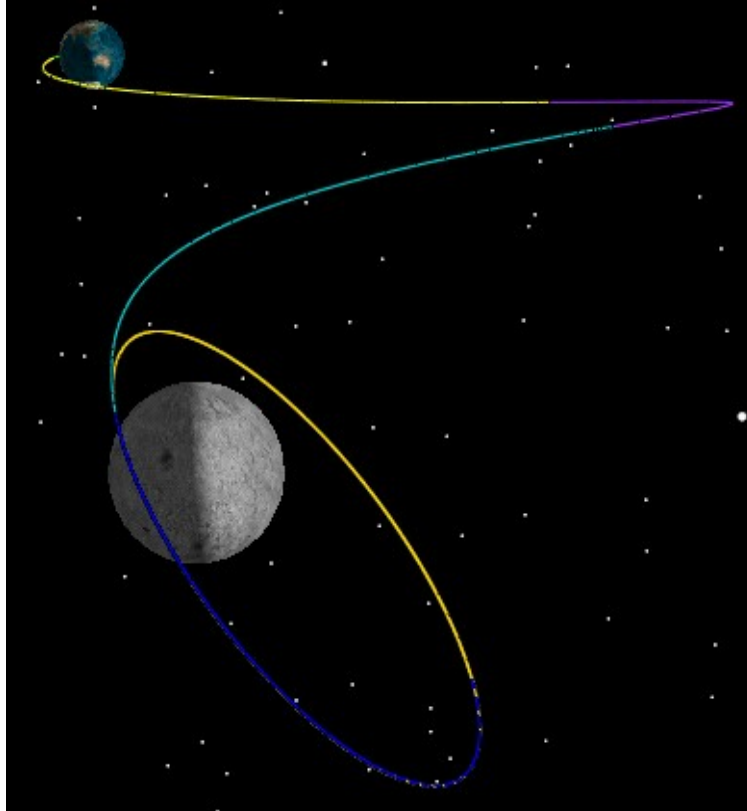
Figure 3—Direct injection—lunar inertial coordinate frame.

2.5.5 Mission Timeline

Table 3 outlines the mission events, magnitude and duration of ΔV burns for the LRS mission. The ΔV table was adapted from GSFC IMDC presentations on their LRS design, with the addition of:

- 50 m/s for attitude control in lunar orbit
- 50 m/s for orbit maintenance in lunar orbit
- 80 m/s for disposal on lunar surface near perilune

Phasing maneuver ΔV provides for:

- Critical event coverage ~4 missions, 45° mean anomaly change
  - ~4 m/s per 45° mean anomaly change with 2 wk to execute
- Did not include 180° mean anomaly changes to avoid long Earth eclipses
  - ~18 m/s with 2 wk to execute
- More analysis needed to assess
  - Stability of the phasing orbits
  - Extra station keeping ΔV (if any)
  - Effect to operations during the phasing time period

TABLE 3.—LRS MISSION TIMELINE AND BURN MAGNITUDES

Phase no.	Phase name	Main ΔV (m/s)	RCS ΔV (m/s)	Duration (hr)	Preburn mass (kg)	Main prop load (kg)	Main burn time (min)	RCS load (kg)	RCS burn time (min)
1	Launch from Earth	---	---	0.5	1165	0.0	0.0	0.0	0.0
2	Checkout	---	---	18.0	1165	0.0	0.0	0.0	0.0
3	Loiter to TLI window opening	---	---	70.5	1165	0.0	0.0	0.0	0.0
4	TLI opening to ignition	---	---	3.0	1165	0.0	0.0	0.0	0.0
5	TLI burn	---	---	0.1	1165	0.0	0.0	0.0	0.0
6	Correct for ELV dispersions	85	---	1.0	1165	42.5	3.6	0.0	0.0
7	Trans-Earth mid-course corrections	25	---	71.7	1122	12.2	1.0	0.0	0.0
8	Trans-lunar coast	---	---	0.5	1110	0.0	0.0	0.0	0.0
9	Lunar orbit capture burn	410	---	4.0	1110	182.3	15.6	0.0	0.0
10	lunar orbit adjustments	250	---	1.0	928	96.1	8.2	0.0	0.0
11	phasing maneuvers	50	---	1.0	831	18.0	1.5	0.0	0.0
12	Lunar telecom orbit	50	50	61320.0	813	17.6	1.5	18.0	15.1
13	Disposal	80	---	1.0	778	26.8	2.3	0.0	0.0
	Total ΔV	950.0	50.0			Total burn time	33.9		15.1

2.5.6 Mission Trades: Electric Propulsion (EP) Versus Chemical

The bottom line is that because of the excess of 60 percent launch margin using a chemical direct injection technique off of the available larger class launch vehicles for LRS, EP is not necessary for performance improvements.

Table 4 shows the comparison of the direct chemical injection mission versus a similar spacecraft configuration for the Solar Electric Propulsion (SEP) mission. Each delivers a similar final mass to the same lunar orbit.

TABLE 4.—COMPARISON OF DIRECT CHEMICAL AND SEP MISSIONS

	Direct Chemical Mission	SEP Mission
Launch vehicle and C_3	Atlas V 501 Launch to C_3 of ~–2 km^2/s^2	Atlas V 501 Launch to C_3 of ~–2 km^2/s^2
ΔV (m/s)	900	1200
Trip time	4 to 5 d	~85 d including a lunar flyby
Final orbit	Spacecraft placed in lunar frozen orbit	Spacecraft placed in lunar frozen orbit
Delivered spacecraft dry mass	~1300 kg	~1300 kg
Propellant	735 kg	80 kg
Power required	~1.5 kWe	~6 kWe

Figure 4 further points out the cost of the SEP performance over the direct injection chemical mission due to increased trip time.

2.5.7 Mission Analytical Methods

- Analysis performed using an internally developed mission architecture tool
  - This tool recursively calculates propellant masses based on propulsion system parameters, inert masses, and Δv schedule.
  - Can accommodate multiple vehicles and non-propulsive mass depletion

2.5.8 Mission Recommendation

- Because of the large launch vehicle, a significant amount of mass margin can be carried.
- A direct chemically propelled mission is the preferred architecture for this mission.
- This mission can meet all of the performance goals set forth.
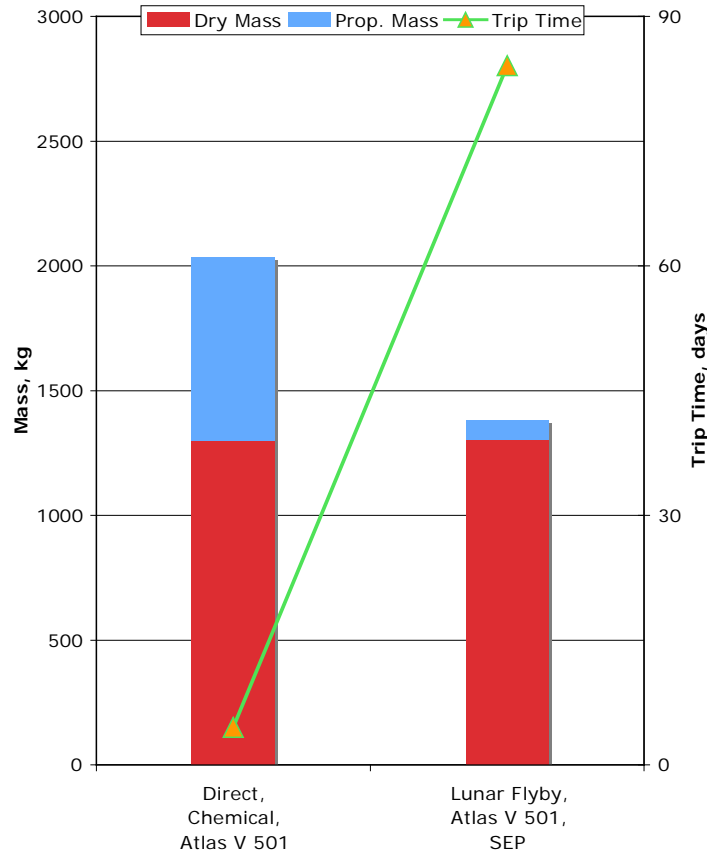
Figure 4.—Comparison of direct injection chemical to SEP mission performance.

2.5.9 Launch Vehicle Recommendations

LRS is ideally a Delta-II class payload, but Delta II's future is uncertain. Cost of keeping the Delta II pad operational may offset the savings in the cost of a Delta II launch. Moving up in EELV class, the Atlas V provides ~66 percent mass margin for LRS (i.e., it could launch almost three LRS) 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 5 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. LRS fits within a 4-m diameter payload envelope, so the 4-m fairing (Atlas V 401 series) is sufficient.

- Atlas V 401 and Delta IV Medium (EELV) provide excess capability for LRS
  - Performance excess mitigates risk of low performing launch vehicle
  - Atlas V is a NASA certified, flight proven launch vehicle
  - Excess launch vehicle performance can be provided for secondary payload opportunities using the ESPA adapter
  - Spacecraft can provide accommodations (mass, volume, power, etc.) for science mission payloads
- Savings could also be realized by launching LRS 1 and 2 on a single launch vehicle. See Figure 6 for an illustration of how two LRS vehicles could fit in the Atlas V fairing.

At this time, the use of the excess capacity of the Atlas V class launch vehicle used in delivering the LRS as a cost saving solution has not been determined. It is offered up here as a suggestion.
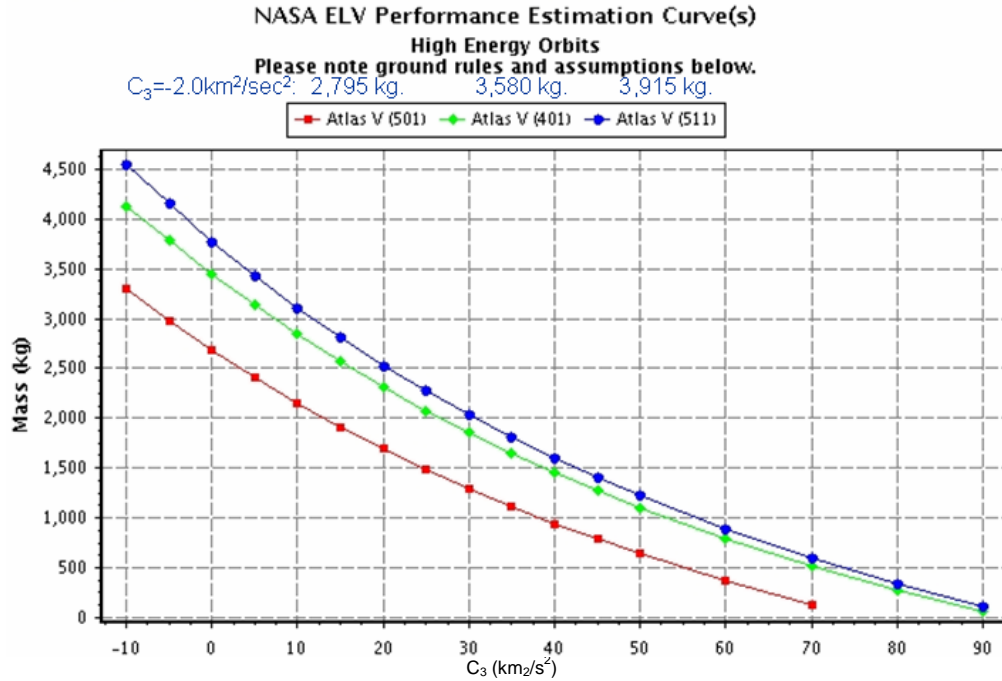
Figure 5.—Atlas V performance curve for $C_3 = -2$ km^2/sec^2.

Atlas V 401: Two Birds; One Launch
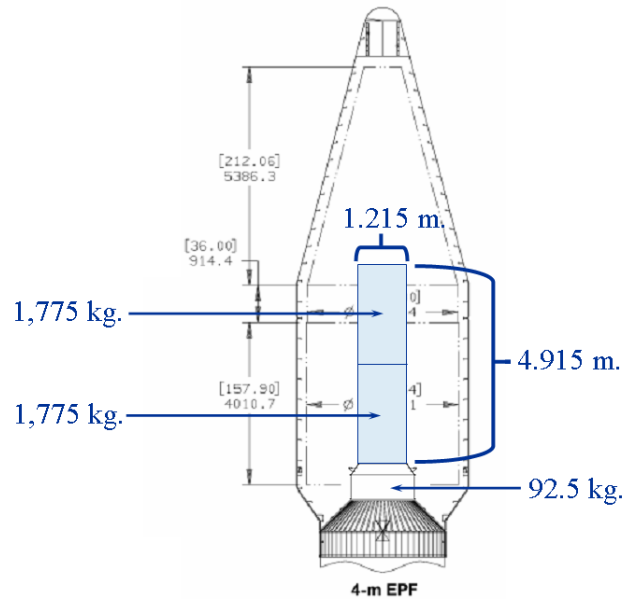
Figure 6.—Atlas V 401 fairing dimensions with notional two LRS launch packaging.

	Falcon 1	Falcon 5	Falcon 9	Falcon 9	Falcon 9 - S5	Falcon 9 - S9
LEO	570 kg	4,100 kg	9,300 kg	8,700 kg	16,500 kg	24,750 kg
GTO	--	1,050 kg	3,400 kg	3,100 kg	6,400 kg	9,650 kg
Fairing diameter	1.5 m	3.6 m	3.6 m	5.2 m	5.2 m	5.2 m
Price in millions *	$6.7	$18	$27	$35	$51	$78
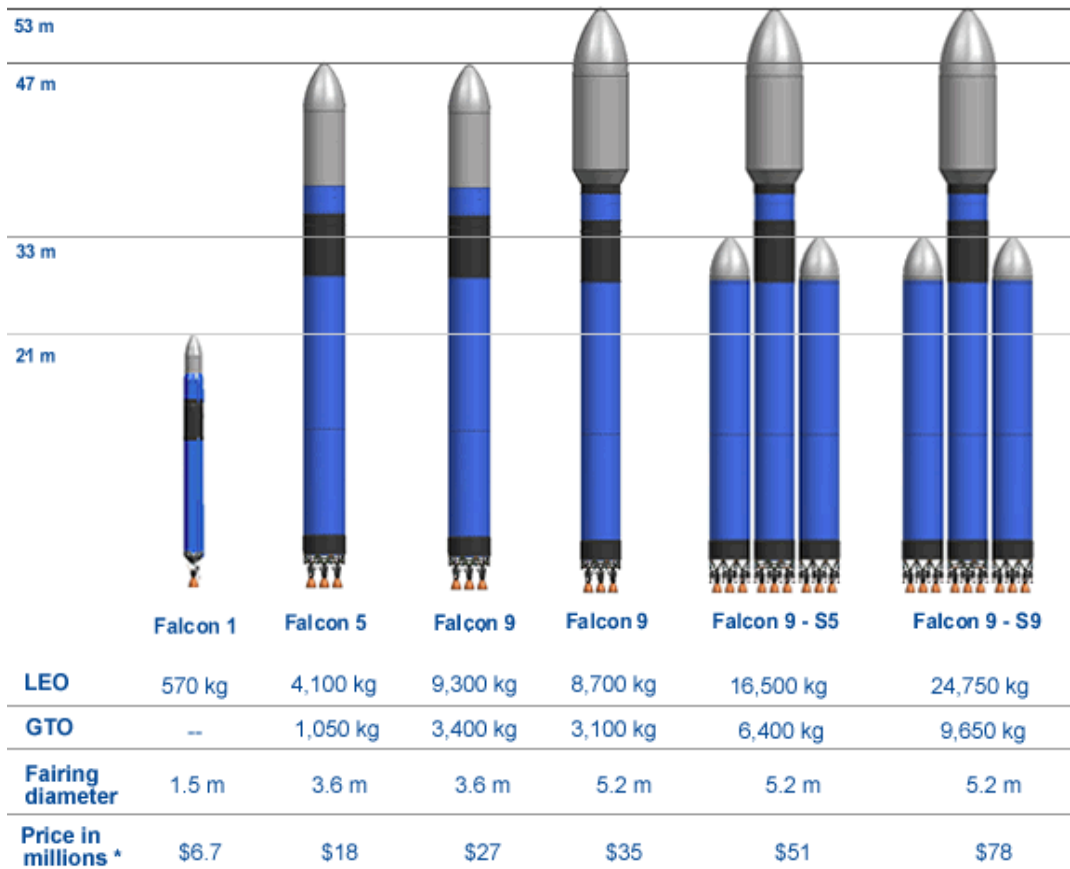
* Prices are all inclusive of launch range, third party insurance and standard payload integration costs.

Figure 7.—Falcon launch vehicle family.

Commercial alternative launch vehicles may be available in the LRS launch timeframe (2018). The Space-X Falcon 1 is too small, and the Falcon 5 will not be available, but Falcon 9 is appropriate, and at an advertised price of $50M, provides a 75 percent savings to LRS over the $175M Atlas V or Delta IV cost. Falcon 9 has the potential to provide a low cost launch option for LRS, should that service materialize. Figure 7 details the Falcon family of launch vehicles.

2.6   Design Trade Space

For the design of the LRS, the following major functional trades were identified.

2.6.1   One Versus Two DTE Antennas

- Previous (initial COMPASS payload design) had two 0.5-m Ka band DTE antennas to enable simultaneous Earth, Moon, Sun pointing
- Current design is one 0.5-m Ka band dish for DTE communications
- Impact: Reduces mass of one antenna and eliminates 3 dB loss in power due to Traveling Wave Tube Amplifier (TWTA)

2.6.2   One Versus Two Relay Antennas

- Previous: Relay link had two antennas: 1- by 1-m S band patch array + 0.5-m Ka band dish
- Current: Relay link has one 1-m dish with dual feed (S and Ka)

- Impact 1: Reduces required surface area on s/c bottom, but we lose flexibility of pointing Ka and S band antennas independently
- Impact 2: Decreases Ka band footprint from 60- to 30-km radius

2.6.3 Atomic Clocks

- Previous: None
- Current: 2 atomic clocks (1 + backup)
- Impact 1: Increase mass by 6 kg and DC power by 20 W
- Impact 2: Signal-to-noise (SN) signal pseudo noise (PN) code transmissions from LRS 2,3 and LCT are synchronous

3.0 Baseline Design

The simplicity of this spacecraft design rests in a 110° joint, which enables the Earth antenna to view more than a hemisphere, so that when Earth is edge-on to the LRS orbit, oscillating from slightly above to slightly below 90° relative to the Earth antenna, the Earth antenna can look down over the solar arrays to see Earth without requiring the spacecraft to flip over. In this design, a spacecraft flip is required twice a lunar month, every 14 d. The solar arrays are mounted to the rear, Russian style, to avoid blocking the Earth antenna view. The arrays have two degrees of freedom rotation to track the Sun. The lunar antenna is gimbaled. See Figure 8 for a design drawing with main components labeled.

3.1 Top Level Design (MEL and PEL)

The bottoms-up current best estimate (CBE) wet mass of the LRS satellite was 1030.5 kg. See Table 5 for detail breakdown per subsystem. With growth factors added at appropriate levels for technology development in the subsystems, the total bottoms up wet mass for the LRS was 1120.6 kg. Total propellant was 467.3 kg. This leaves the total CBE dry mass at 563.1 kg and the total dry mass with growth at 653.3 kg without the additional system level growth from Section 3.1.1.
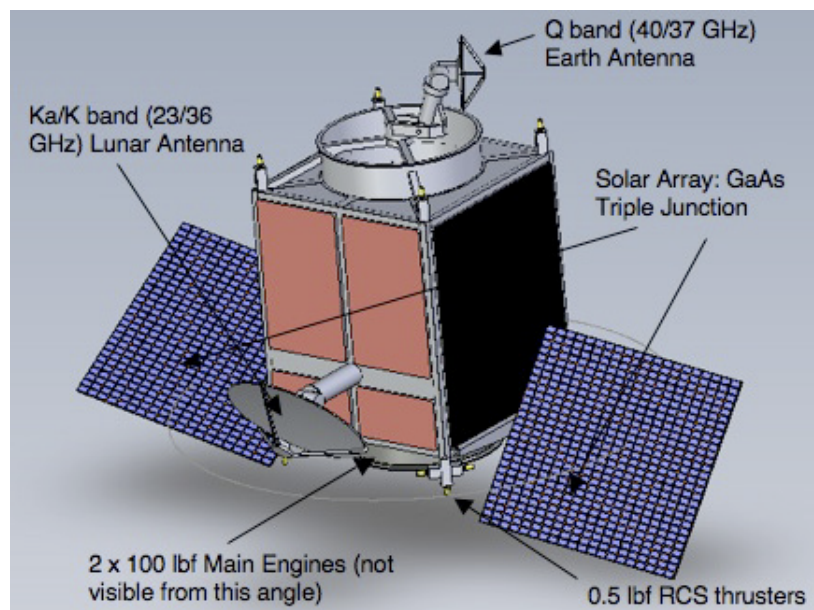
Figure 8.—LRS design with major subsystem components identified.

TABLE 5.—TOP LEVEL LRS MEL WITHOUT SYSTEM LEVEL GROWTH

Description	Quantity	Unit mass (kg)	CBE mass (kg)	Growth ()	Growth (kg)	Total mass (kg)	Nominal power (W)	Peak power (W)
Lunar COMM Relay Satellite (LRS)	----	----	1030.5	8.7	90.1	1120.6	683.6	259.6
LRS-2	----	----	1030.5	9	90.1	1120.6	683.6	259.6
Communications	----	----	78.4	3	2.4	80.8	494.0	0.0
Avionics	----	----	89.6	24	21.1	110.7	189.6	259.6
Structures & Mechanical Systems	----	----	180.5	14	25.1	205.6	0.0	0.0
Power System	----	----	72.6	31	22.2	94.8	0.0	0.0
Propulsion (Electric)	----	----	0.0	0	0.0	0.0	0.0	0.0
Propellant management (EP)	----	----	0.0	0	0.0	0.0	0.0	0.0
Propellant (EP)	----	----	0.0	0	0.0	0.0	0.0	0.0
Propulsion (Chemical)	----	----	21.8	8	1.7	23.5	0.0	0.0
Propellant management (Chemical)	----	----	72.9	15	10.9	83.9	0.0	0.0
Propellant (Chemical)	----	----	467.3	0	0.0	467.3	0.0	0.0
Thermal Control (Non-Propellant)	----	----	47.3	14	6.7	54.0	0.0	0.0

3.1.1 Master Equipment List (MEL)

Where the MEL (Table 6) captures the bottoms up estimation of CBE and growth percentage line item by item from the subsystem designer, Table 6 wraps up those total masses, CBE and total mass after applied growth percentage. The total growth from the bottoms up is 90.1 kg (16 percent of the dry mass). In order to reach the system level growth mass of 30 percent (168.9 kg) an additional 78.8 kg is carried at the system level.

TABLE 6.—LRS MASTER EQUIPMENT LIST WITH SYSTEM LEVEL GROWTH CALCULATIONS

LRS MEL	CBE (kg)	Growth ()	Growth (kg)	Total Mass (kg)	Percent of dry mass
Communications	78.4	3	2.4	80.8	12.4
Avionics & GN&C	89.6	24	21.1	110.7	16.9
Structures & Mechanisms	180.5	14	25.1	205.6	31.5
Power System	72.6	31	22.2	94.8	14.5
Propulsion (Chemical)	21.8	8	1.7	23.5	3.6
Propellant Management (Chemical)	72.9	15	10.9	83.9	12.8
Propellant (Chemical)	467.3	0	0.0	467.3	-----
Thermal Control	47.3	14	6.7	54.0	8.3
LRS dry mass without system growth	563.1	16.0	90.1	653.3	
Desired system growth (contingency)	563.1	30	168.9	732.1	
Added system level growth (contingency)	563.1	14.0	78.8	------	
LRS dry mass (with system growth)	563.1	30	168.9	732.1	
Total useable propellant	453.0	----	----	----	
Total residuals + pressurant	14.3	----	----	----	
LRS inert mass (with system growth)	577.5	N/A	168.9	746.4	
LRS total wet mass (with system growth)	1016.1	N/A	168.9	1185.1	

3.1.2 Power Equipment List (PEL)

The power listing for nominal loads is 684 W (see Table 7). This includes both the communications packages and the avionics. At the time of the study, propulsion and thermal had not yet gotten their power requirements into the system design. Therefore, the power system included a 30 percent margin to account for these unknowns.

TABLE 7.—LRS POWER EQUIPMENT LIST

LRS PEL	Nominal power (W)	Peak power (W)	Standby power (W)
LRS Subsystems			
Communications	494	0	0
Avionics	189.6	259.6	24.5
Power System	0	0	0
Structures & Mechanisms	0	0	0
Propulsion	0	0	0
Thermal Control	0	0	3.4
Total LRS	684	260	28

3.2 Design Concept Drawing and Description

The LRS in this design is a half-TDRSS sized box spacecraft, which provides communications and navigation relay between lunar outposts (via LCT) or Sortie parties (with user radios) and large ground antennas on Earth.

Figure 8 shows a front view of the LRS spacecraft. Called out are the two antenna, the solar arrays, and the two types of thrusters.

4.0 Subsystem Breakdown

4.1 Communications

The multi-center COMPASS design session worked to achieve a simple spacecraft design which allows the Earth antenna, lunar antenna, solar arrays and Roll Control System (RCS) thrusters to all remain pointed at their targets without blockage, as LRS traverses its elliptical orbit around the Moon over the lunar South Pole, while the Moon orbits the Earth, and Earth orbits the Sun. Prior LRS designs, developed by JPL's Team-X and GSFC's IMDC design teams, over three to four months, were unable to resolve this problem without requiring two spacecraft, spacecraft flips as often as four times a day, two Earth Ka-band antennas, or heavy mechanically complex gimbals to support antennas beyond the solar arrays.

4.1.1 Communications Requirements

The LRS provides the following communications for the manned lunar base.

- Provide connectivity between Habitat and
  - Surface users not on LCT network
  - Earth (Deep Space Network (DSN))
- Provide connectivity between LCT network and
  - Earth (DSN)
  - Surface users not on LCT network
- Provide connectivity between users not on LCT network and
  - Earth (DSN)
  - Habitat
  - Other users not on LCT network
- Support priority based demand access service for following types of IP-based connections
  - Two way full-duplex voice and data between
    - Surface user and surface user
    - Surface user and habitat
    - Surface user and Earth

- ○ One way video from
  - – Surface user to habitat and to Earth
  - – Habitat to Earth
  - – Earth to habitat

4.1.2 Communications Assumptions

- Physical layer and MAC layer protocol for satellite and LCT can be designed independently
- Users can carry separate radios for LCT and sat communications
- Satellite-based tracking services use SN signaling
  - ○ Communications and tracking services are not required to be integrated for data rates above 192 kbps
  - ○ Users requiring sat communications services above 192 kbps carry a separate Ka band transceiver in the 26/23 GHz band
- Users have the ability to point directional antennas to the satellite with sufficiently high accuracy
- 2 kbps emergency voice service is acceptable for communications with stranded astronauts
- No satellite services to lunar surface when satellite is in safe mode
  - ○ Safe mode/relay links operating frequencies are in reverse S-band configuration

4.1.3 Communications Design and MEL

On the top face, a Q-band Earth antenna (40/37 GHz) with a 0.5 m dish provides communication with Mission Control. A unique 110° joint in the gimbals allows the Earth antenna to look over the solar arrays to see Earth thus avoiding the need to flip the spacecraft over as often as four times a day (as previous designs by JPL and GSFC had required) during times in the orbit when Earth is sometimes above and sometimes below 90°. A low gain S band (2 GHz) antenna transmits/receives navigation from Network Control and telemetry from Mission Control. Figure 9 illustrates the viewing angle of the 110° joint gimbals on the Q-band antenna, and their ability to look over and around the two solar arrays mounted on the sides of the spacecraft. The brown cone is the cutout zone below which the Q-band antenna cannot see the Earth. This illustrates that the antenna can look more than over a 180° range, but slightly over each side. When the Earth enters into that brown cutout zone, the spacecraft will have to perform a maneuver to flip about that axis.

On the front face (velocity vector), a Ka/K band antenna (23/26 GHz) with a 1-m dish provides communications with the Moon. This antenna, from the Lunar Reconnaissance Orbiter, has an integral Cassegrainian S band (2 GHz) antenna for navigation. This antenna will be flight qualified in 2009 at the new Ka/K frequencies (23/26 GHz).

Table 8 captures the line items (quantity, mass and power) from the MEL of the communication system for LRS as used in the COMPASS design session. This system was provided by the communications consultants from JPL and adapted as the study progressed to the final version seen here.
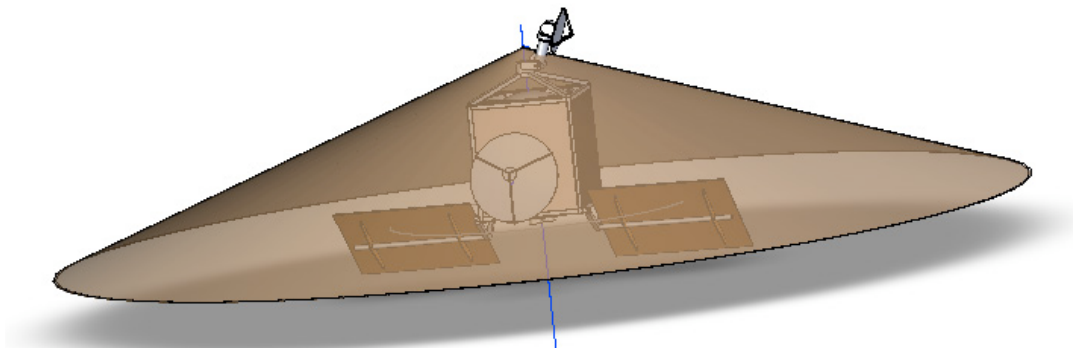

Figure 9.—Viewing angle of Q-band antenna on LRS.

TABLE 8.—LRS COMMUNICATIONS SYSTEM MEL BREAKDOWN

Description	Quantity	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)	Nominal power (W)	Peak power (W)
Lunar COMM Relay Satellite (LRS)	--	----	1030.5	8.7	90.1	1120.6	683.6	259.6
LRS-2	--	----	1030.5	9	90.1	1120.6	683.6	259.6
Communications	--	----	78.4	3	2.4	80.8	494.0	0.0
Ka band (40/37) DTE Link	--	----	17.1	3.00	0.5	17.6	217.0	0.0
Ka DTE transponder	2	2.9	5.8	3.00	0.2	6.0	17.0	0.0
Ka TWTA (100W)	2	2.6	5.2	3.00	0.2	5.4	200.0	0.0
Ka-band diplexer	2	0.4	0.8	3.00	0.0	0.8	0.0	0.0
WG transfer switch	1	0.4	0.4	3.00	0.0	0.4	0.0	0.0
Coax switch	1	0.2	0.2	3.00	0.0	0.2	0.0	0.0
Attenuator	1	0.1	0.1	3.00	0.0	0.1	0.0	0.0
Ka band hybrid coupler	1	0.1	0.1	3.00	0.0	0.1	0.0	0.0
Ka band Wave Guide	9	0.5	4.5	3.00	0.1	4.6	0.0	0.0
Miscellaneous no. 1	0	0.0	0.0	3.00	0.0	0.0	0.0	0.0
Miscellaneous no. 2	0	0.0	0.0	3.00	0.0	0.0	0.0	0.0
Ka band (23/26) Relay Link	--	----	21.1	3.00	0.6	21.7	115.0	0.0
Ka band modulator	2	1.0	2.0	3.00	0.1	2.1	10.0	0.0
Ka band hybrid coupler	1	0.1	0.1	3.00	0.0	0.1	0.0	0.0
Ka TWTA (20W)	2	2.6	5.2	3.00	0.2	5.4	40.0	0.0
WG transfer switch	1	0.4	0.4	3.00	0.0	0.4	0.0	0.0
Ka-band diplexer	1	0.4	0.4	3.00	0.0	0.4	0.0	0.0
LNA-splitter	1	1.0	1.0	3.00	0.0	1.0	5.0	0.0
Ka receiver (up to four on, two spare)	6	2.0	12.0	3.00	0.4	12.4	60.0	0.0
Miscellaneous no. 1	0	0.0	0.0	3.00	0.0	0.0	0.0	0.0
Miscellaneous no. 2	0	0.0	0.0	3.00	0.0	0.0	0.0	0.0
Ka band (40/37) TT&C and Ranging	--	----	17.1	3.00	0.5	17.6	42.0	0.0
Ka DTE transponder	2	2.9	5.8	3.00	0.2	6.0	17.0	0.0
Ka TWTA (10 W)	2	2.6	5.2	3.00	0.2	5.4	25.0	0.0
Ka-band diplexer	2	0.4	0.8	3.00	0.0	0.8	0.0	0.0
WG transfer switch	1	0.4	0.4	3.00	0.0	0.4	0.0	0.0
Coax switch	1	0.2	0.2	3.00	0.0	0.2	0.0	0.0
Attenuator	1	0.1	0.1	3.00	0.0	0.1	0.0	0.0
Ka band hybrid coupler	1	0.1	0.1	3.00	0.0	0.1	0.0	0.0
Ka band wave guide	9	0.5	4.5	3.00	0.1	4.6	0.0	0.0
Miscellaneous no. 1	0	0	0.0	3.00	0.0	0.0	0.0	0.0
Miscellaneous no. 2	0	0	0.0	3.00	0.0	0.0	0.0	0.0
S band Relay Link	--	----	9.3	3.00	0.3	9.6	80.0	0.0
S band transceiver (10 W)	2	3.5	7.0	3.00	0.2	7.2	80.0	0.0
S band coax switch	1	0.3	0.3	3.00	0.0	0.3	0.0	0.0
S band coax	10	0.2	2.0	3.00	0.1	2.1	0.0	0.0
Data Router and SSR	--	----	4.0	3.00	0.1	4.1	20.0	0.0
Data router and SSR	1	4	4.0	3.00	0.1	4.1	20.0	0.0
Antennas (Booms excluded)	--	----	9.8	3.00	0.3	10.1	20.0	0.0
S band relay, 1- by 1-m patch array	0	2	0.0	3.00	0.0	0.0	0.0	0.0
Gimbal	0	1.4	0.0	3.00	0.0	0.0	5.0	0.0
S/Ka band relay, 1 m dia. reflector	1	4	4.0	3.00	0.1	4.1	0.0	0.0
Feed	1	0.5	0.5	3.00	0.0	0.5	0.0	0.0
Gimbal	1	1.4	1.4	3.00	0.0	1.4	5.0	0.0
Ka band DTE , 0.5 m dia. reflector	1	1	1.0	3.00	0.0	1.0	0.0	0.0
Feed	1	0.5	0.5	3.00	0.0	0.5	0.0	0.0
Gimbal	1	1.4	1.4	3.00	0.0	1.4	10.0	0.0
S band LGA (T/R) for safing	2	0.5	1.0	3.00	0.0	1.0	0.0	0.0

4.1.4 Communications Trades

In order to overcome the pointing issues with the LRS communications system, several design scenarios for the communications antenna were considered.

- **Options**: Single '110°' pointing or two Earth pointing antennas to eliminate 'flipping' the spacecraft (S/C) as often as previous studies had indicated was necessary
  - **Chosen**: Found design to 'point around' S/C, use single 0.5-m Q band antenna (see Figure 9 for viewing angle of the top antenna on LRS)
- **Options:** Two (1 S and 1 Ka band) or one (combined S and Ka band) antenna
  - **Chosen:** Single combined Ka and S 1-m antenna

4.2 Avionics

For the purposes of modeling the LRS system, the Avionics section included both Command and Data Handing (C&DH) and Guidance, Navigation and Control (GN&C) hardware.

4.2.1 Avionics Requirements

- Storage array for 24 hr of storage (0.3 TB)
- Satellite capacity requirements
  - Support the following peak offered load from the lunar surface to relay sat
    – 100 Mbps from Habitat
    – 100 Mbps from LCT
    – 50 Mbps from surfaces users not on the LCT network or in the Habitat
  - Support a forward link rate of up to 100 Mbps from relay sat to lunar surface
  - Support two-way ranging to up to five users simultaneously

4.2.2 Avionics Assumptions

The avionics shall be single fault tolerant.

4.2.3 Avionics Design and MEL

Avionics components are based on commercially available components come from British Aerospace (BAE). All avionics assume 3U-160 form factor cards.

4.2.4 GN&C Design

The GN&C subsystem line items were kept under the Avionics subsystem in the LRS MEL. The LRS GN&C subsystem provides navigation and attitude determination using

- Twelve coarse Sun sensor suites
- Two star tracking cameras
- Two inertial measurement units

LRS GN&C provides momentum control using

- Four reaction wheels with momentum storage capability of 20 Nms
- RW thrust assembly

LRS GN&C controls and logic are included in the Avionics system elements. Table 9 details the line items contained in the LRS Avionics MEL rolled up into the LRS design.

TABLE 9.—LRS AVIONICS DETAILED MEL

Description	Quantity	Unit mass (kg)	CBE Mass (kg)	Growth ()	Growth (kg)	Total mass (kg)	Nominal power (W)	Peak power (W)
Lunar COMM Relay Satellite (LRS)	---	-------	1030.5	8.7	90.1	1120.6	683.6	259.6
LRS-2	---	-------	1030.5	9	90.1	1120.6	683.6	259.6
Avionics	---	-------	89.6	24	21.1	110.7	189.6	259.6
Command & Data Handling	---	-------	53.7	23.22	12.5	66.1	157.6	157.6
General Avionics Processor	2	11	22.0	20.00	4.4	26.4	65.0	65.0
Time generation unit	2	0.4	0.8	3.00	0.0	0.8	0.6	0.6
Command and control harness (data)	1	4.88	4.9	50.00	2.4	7.3	0.0	0.0
Instrumentation & wiring	0	0	0.0	50.00	0.0	0.0	0.0	0.0
Data recorder	2	11	22.0	20.00	4.4	26.4	72.0	72.0
Atomic clock	2	2	4.0	30.00	1.2	5.2	20.0	20.0
Miscellaneous no. 2	0	0	0.0	0.00	0.0	0.0	0.0	0.0
Guidance, Navigation and ACS	---	-------	35.9	23.96	8.6	44.5	32.0	102.0
Reaction wheel assembly (RWA)	4	7	28.0	30.00	8.4	36.4	15.0	80.0
RWA mount	1	0.2	0.2	18.00	0.0	0.2	0.0	0.0
Star camera	2	3	6.0	2.00	0.1	6.1	5.0	10.0
Star camera mount	1	0.1	0.1	18.00	0.0	0.1	0.0	0.0
Inertial measurement units	2	0.75	1.5	2.00	0.0	1.5	12.0	12.0
Course Sun sensor suite	12	0.01	0.1	2.00	0.0	0.1	0.0	0.0

4.2.5 Avionics Risk Inputs

The 0.3 TB storage array is design 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.

4.3 Structures and Mechanisms

4.3.1 Structures and Mechanisms Requirements

Requirements include the following

- Provide sufficient space to house the necessary hardware for avionics, communications, propulsion, and power.
- Sustain maximum loads with minimum deflections without damage.
- Minimize weight.
- Minimize vibrations.
- Fit within the confines of the launch vehicle.

4.3.2 Structures and Mechanisms Assumptions

The primary structures material is aluminum (Al) 2090. It consisted of a tubular space frame construction with honeycomb core composite sandwich structure panels. A maximum axial load is 5.5 g and a maximum lateral load is 3 g.

4.3.3 Structures and Mechanisms Design and MEL

The material of the structure is Al 2090. The architecture of the structure consists of a tubular space frame with honeycomb core composite sandwich structure panels. Welds and threaded fasteners are used for the assembly.

Table 10 details the line items used in the Structures and Mechanical Systems subsystem used to build the LRS. CBE Mass, Growth and Total Mass are displayed.

TABLE 10.—LRS STRUCTURES AND MECHANICAL SYSTEMS DETAILED MEL

Description	Quantity	Unit mass (kg)	CBE mass (kg)	Growth ()	Growth (kg)	Total mass (kg)
Lunar COMM Relay Satellite (LRS)	--	-----	1030.5	8.7	90.1	1120.6
LRS-2	--	-----	1030.5	9	90.1	1120.6
Structures & Mechanical Systems	--	-----	180.5	14	25.1	205.6
Primary Structures	--	-----	164.1	14.43	23.7	187.7
Separation ring/flange	1	18.72	18.7	10.00	1.9	20.6
Top/bottom deck	2	5.735	11.5	15.00	1.7	13.2
Vertical inner panels	4	3.66	14.6	15.00	2.2	16.8
Side panels	4	13.09	52.4	15.00	7.9	60.2
Corner post	4	1.899	7.6	15.00	1.1	8.7
Top cover	0	0	0.0	15.00	0.0	0.0
Thrust tube	1	59.28	59.3	15.00	8.9	68.2
Miscellaneous clips/fasteners	0	0	0.0	15.00	0.0	0.0
Miscellaneous no. 1	0	0	0.0	15.00	0.0	0.0
Miscellaneous no. 2	0	0	0.0	15.00	0.0	0.0
Secondary Structures	--	-----	9.4	15.00	1.4	10.8
Antenna boom	1	5.126	5.1	15.00	0.8	5.9
Solar array boom(s)	2	2.148	4.3	15.00	0.6	4.9
EP thruster booms (N/A)	0	0.537	0.0	15.00	0.0	0.0
Chemical thruster booms	0	0.537	0.0	15.00	0.0	0.0
Miscellaneous no. 1	0	0	0.0	0.00	0.0	0.0
Miscellaneous no. 2	0	0	0.0	0.00	0.0	0.0
Installation	--	-----	7.0	0.00	0.0	7.0
C&DH installation	1	0.916	0.9	0.00	0.0	0.9
Communications installation	1	5.212	5.2	0.00	0.0	5.2
GN&C installation	1	0.916	0.9	0.00	0.0	0.9
Power installation	0	0	0.0	0.00	0.0	0.0
Propulsion installation	0	0	0.0	0.00	0.0	0.0
Propellant storage installation	0	0	0.0	0.00	0.0	0.0
Thermal installation	0	0	0.0	0.00	0.0	0.0
Miscellaneous no. 1	0	0	0.0	0.00	0.0	0.0
Miscellaneous no. 2	0	0	0.0	0.00	0.0	0.0
Miscellaneous no. 3	0	0	0.0	0.00	0.0	0.0
Mechanisms	--	-----	0.0	0.00	0.0	0.0
Solar array deployment mech (in power)	0	0	0.0	0.00	0.0	0.0
Radiator deployment mechanism (if applicable)	0	0	0.0	0.00	0.0	0.0
Separation mechanism (pyros)	0	0	0.0	0.00	0.0	0.0
Bi-axial antenna gimbal	0	0	0.0	0.00	0.0	0.0

4.3.4 Structures and Mechanisms Analytical Methods

A preliminary structural analysis using basic calculations with given launch loads was conducted.

4.3.5 Structures and Mechanisms Risk Inputs

Primary risks to maintaining structural integrity include the following

- Potential impact with foreign object.
- Excessive vibrations.
- Excess inertial loading from the launch vehicle.

4.3.6 Structures and Mechanisms Recommendation

A complete finite element analysis (FEA) to determine stresses and displacements along with a modal analysis for vibrations is necessary for further study.

4.4 Electrical Power System

4.4.1 Power Requirements

During eclipse periods longer than 2.5 hr, the power available must meet nominal power requirements.

4.4.2 Power Assumptions

The solar arrays are assumed to use GaAs triple-junction cells that are 28 percent efficient at the beginning-of-life. An off-point angle is assumed that accounts for 15° cosine losses between the Sun and solar array. The battery modules are assumed to be lithium-ion chemistry with a cell specific energy of 170 W-hr/kg and a module specific energy of 72 W-hr/kg. The spacecraft bus is assumed to be nominally 28 Vdc. A 30 percent power margin is assumed and 30 percent mass margin on everything except the wire harness, which is at 50 percent mass margin.

4.4.3 Power Design and MEL

Located on the sides of the LRS, the 1.7- by 1.7-m (area of 2.9 m^2) solar array panels rotate to face the Sun to generate a total of 1 kW (1040 We) useful power. The arrays are mounted to the base of the spacecraft to avoid blockage to the Earth antenna. Radiator panels on the sides provide heat rejection.

Two equal area solar array wings are used to have some fault tolerance and help with spacecraft balance. The arrays are sized to meet the peak power requirement plus 30 percent power margin. A single axis gimbal is assumed. The battery system is sized to meet the nominal power requirement for a 2.5 hr eclipse duration. For longer eclipses, a lower amount of power will be available.

Table 11 details the line items used in the Power Systems subsystem used to build the LRS. CBE Mass, Growth and Total Mass are displayed.

TABLE 11.—LRS POWER SYSTEM DETAILED MEL

Description	Quantity	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
Lunar COMM Relay Satellite (LRS)	--	------	1030.5	8.7	90.1	1120.6
LRS-2	--	------	1030.5	9	90.1	1120.6
Power System	--	------	72.6	31	22.2	94.8
Battery System	--	------	40.3	30.00	12.1	52.4
Battery assembly-secondary	1	40.3	40.3	30.00	12.1	52.4
Miscellaneous no. 1	0	0	0.0	0.00	0.0	0.0
Miscellaneous no. 2	0	0	0.0	0.00	0.0	0.0
Miscellaneous no. 3	0	0	0.0	0.00	0.0	0.0
Solar Array	--	------	26.1	30.00	7.8	33.9
Solar array panel	2	6.4	12.8	30.00	3.8	16.6
Solar array structure	2	0.64	1.3	30.00	0.4	1.7
Solar array drive assembly	2	3	6.0	30.00	1.8	7.8
Solar array interface (carried in structures)	0	0	0.0	0.00	0.0	0.0
Solar array deployment mech	2	3	6.0	30.00	1.8	7.8
Miscellaneous no. 2	0	0	0.0	0.00	0.0	0.0
Miscellaneous no. 3	0	0	0.0	0.00	0.0	0.0
Miscellaneous no. 4	0	0	0.0	0.00	0.0	0.0
Miscellaneous no. 5	0	0	0.0	0.00	0.0	0.0
Miscellaneous no. 6	0	0	0.0	0.00	0.0	0.0
Power Management & Distribution	--	------	6.2	37.10	2.3	8.5
Battery charge controller	1	4	4.0	30.00	1.2	5.2
Wire harness	2	1.1	2.2	50.00	1.1	3.3

4.4.4 Power Trades

One solar array wing would reduce structure and mechanism mass while potentially reducing fault tolerance and spacecraft balance. The battery sizing is sensitive to power level and eclipse duration. If a lower power load can be used during eclipse cycles, the battery mass will be reduced proportionally.

4.4.5　Power Analytical Methods

The battery and solar array system sizing is based on bottoms-up algorithms used in the System Power Analysis for Capability Evaluation (SPACE) code. The Solar Array Drive Assembly (SADA) system is based on Moog designs.

4.4.6　Power Recommendation

A detailed timeline of power loads versus time with the eclipse time period included would improve the sizing of both the solar array and battery systems. A more accurate peak and eclipse requirement would be identified that would directly affect the design.

4.5　Propulsion and Propellant Management

4.5.1　Propulsion and Propellant Management Requirements

There were two operational requirements for the propulsion subsystem:
- Two large burns for lunar orbit insertion and modification
- Several smaller burns required for orbit modifications
- The 100-lbf thruster was selected to perform these burns efficiently
- Attitude Control burns for momentum wheel desaturation.16 thrusters provide sufficient couples for rotating spacecraft on three axes and provide single fault (~1.5) tolerance

4.5.2　Propulsion and Propellant Management Assumptions

Assumption 1—Thruster Operation
- Thruster operation constant over duration of burn

Assumption 2—Propellant Storage
- Propellant delivery assumed to be isothermal
- Same with helium (He) pressurant into main tank

Assumption 3—Design for Minimum Risk (DMR) applied to propellant storage tank and propellant delivery lines, following Exploration Program lead. Additionally, DMR applied to passive elements only.

4.5.3　Propulsion and Propellant Management Design and MEL

The propulsion subsystem is comprised of the following main elements

- Two 100-lbf-thrust MR-104 hydrazine monopropellant engines—one operating, one spare
- Sixteen 0.5-lbf-thrust MR-111E hydrazine monopropellant engines—all operating
- One metallic titanium (Ti) hydrazine storage tank
- Two metallic Ti high pressure He storage tanks 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

Figure 10 shows the bottom view of the LRS highlighting the propulsion system.
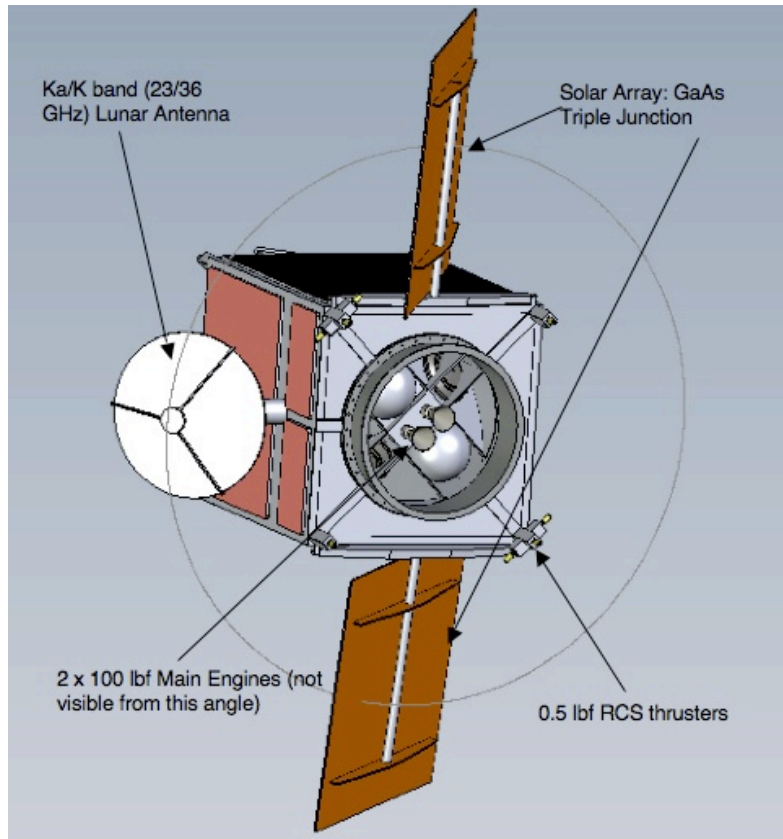
Figure 10.—Bottom view of main engines on LRS.

On the bottom face, the OMS main engine provides hydrazine-fueled propulsion. Four RCS thruster pods at the corners provide yaw, pitch, and roll.

The main propulsion thruster selected for the LRS was the **Aerojet MR-104** Hydrazine thruster. A monoprop engine was selected to simplify the propulsion subsystem. The primary characteristics of the main thruster are:

- Thrust: 441 N (100 lbf); 204.6 to 572.5 N (46 to 128.7 lbf)
- Specific Impulse: 239; 239-223 s
- Mass: 1.86 kg
- Power: 43.1 W
- Lifetime: 2,654 s (cumulative)

The thruster selected for the reaction control system on LRS was the **Aerojet MR-111E** Hydrazine engine. This is also a monoprop device with the following characteristics:

- Thrust: 2.2 N (0.5 lbf); 0.5 to 2.2 N (0.11 to 0.5 lbf)
- Specific Impulse: 224; 224-213 s
- Mass: 0.33 kg
- Power: 13.64 W
- Lifetime: 26.7 hr (cumulative)

TABLE 12.—LRS PROPULSION HARDWARE DETAILED MEL

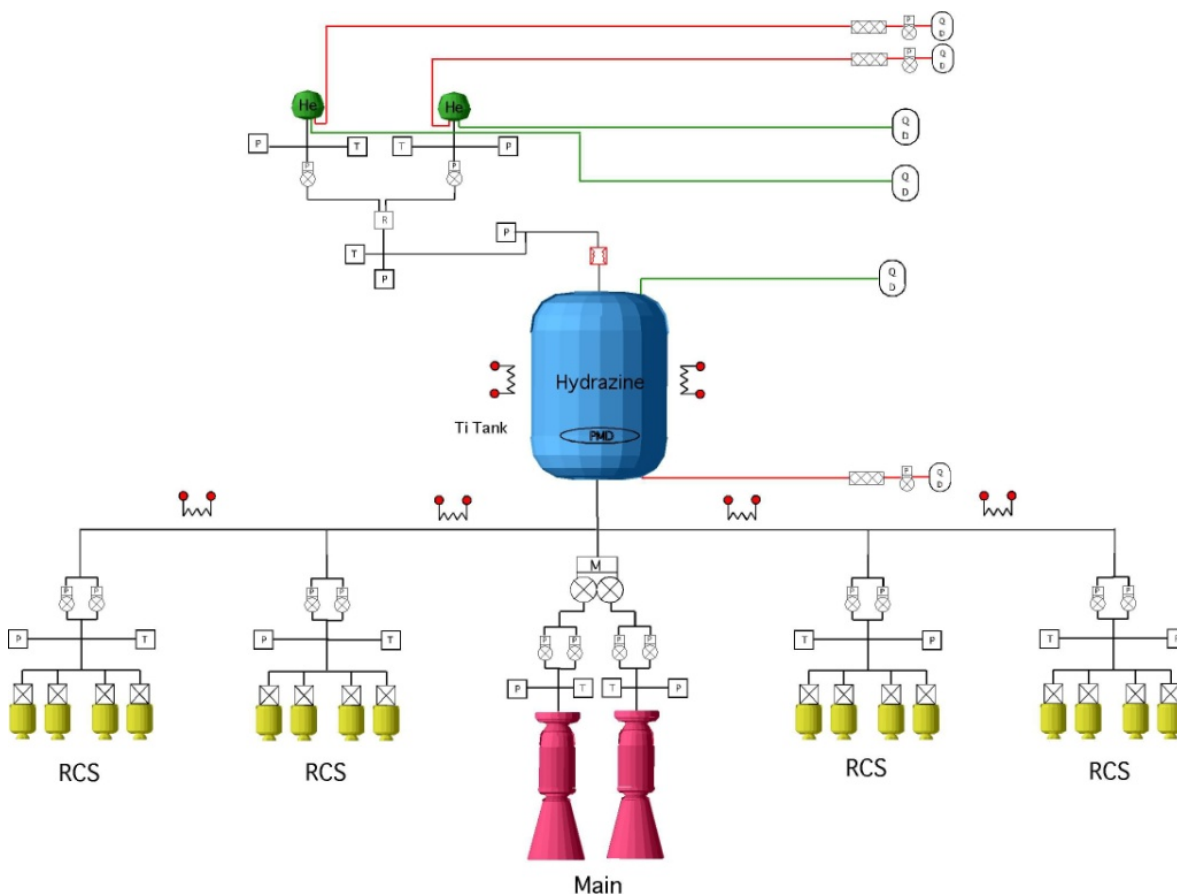| Description	Quantity	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
Lunar COMM Relay Satellite (LRS)	--	-------	1030.5	8.7	90.1	1120.6
LRS-2	--	-------	1030.5	9	90.1	1120.6
Propulsion (Chemical)	--	-------	21.8	8	1.7	23.5
Main Engine	--	-------	5.5	8.00	0.4	6.0
Main engine	1	5.543	5.5	8.00	0.4	6.0
Main engine gimbal	0	0	0.0	0.00	0.0	0.0
Reaction Control System	--	-------	16.3	8.00	1.3	17.6
RCS engine	4	4.064	16.3	8.00	1.3	17.6

Figure 11.—Propulsion system schematic diagram.

Table 12 lists the MEL for the main hardware of the propulsion system (main engine, RCS system, etc.). The propellant and pressurant distribution systems are comprised of gas/fluid delivery lines, the control components (isolation/latching valves, control valves, filters, and service valves). The instrumentation for pressure and temperature are included in the distribution subassemblies. Finally, hardware for attachment/positioning are included here. The propellant subsystem schematic shown in Figure 11 provides a notional drawing of the physical components included in it.

The hydrazine propellant for main and reaction control propulsion was stored in a Ti pressure vessel based on a COTS unit from ATK-PSC, Inc., Model No. 80356-1. The baseline tank dimensions were:

- Size: 0.91- by 1.1-m L (3.0 ft by 3.6 ft L)

- Internal volume: 0.59 m^3 (20.8 ft^3)

Table 13 details the propellant management system line item breakdown including tanks and fuel lines. Table 14 lists the MEL for the propellant used to perform the LRS Mission.

TABLE 13.—LRS PROPELLANT MANAGEMENT DETAILED MEL

Description	Quantity	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
Lunar COMM Relay Satellite (LRS)	--	--------	1030.5	8.7	90.1	1120.6
LRS-2	--	--------	1030.5	9	90.1	1120.6
Propulsion (Chemical)	--	--------	21.8	8	1.7	23.5
Propellant Management (Chemical)	--	--------	72.9	15	10.9	83.9
Main Propellant Management	--	--------	69.2	15.00	10.4	79.6
Fuel tanks	1	41.35	41.4	15.00	6.2	47.6
Fuel lines	0	0	0.0	0.00	0.0	0.0
Oxidizer tanks	0	0	0.0	0.00	0.0	0.0
Oxidizer lines	0	0	0.0	0.00	0.0	0.0
Pressurization system: tanks, panels, lines	1	23.67	23.7	15.00	3.6	27.2
Feed system: regulators, valves, etc	1	4.201	4.2	15.00	0.6	4.8
RCS Propellant Management	--	--------	3.7	15.00	0.6	4.3
Fuel tanks	0	0	0.0	0.00	0.0	0.0
Fuel lines	0	0	0.0	0.00	0.0	0.0
Pressurization system: tanks, panels, lines	0	0	0.0	0.00	0.0	0.0
Feed system: regulators, valves, etc	1	3.705	3.7	15.00	0.6	4.3
Propellant (Chemical)	--	--------	467.3	0	0.0	467.3

TABLE 14—LRS PROPELLANT DETAILED MEL

Description	Quantity	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
Lunar COMM Relay Satellite (LRS)	--	---------	1030.5	8.7	90.1	1120.6
LRS-2	--	---------	1030.5	9	90.1	1120.6
Propellant (Chemical)	--	---------	467.3	0	0.0	467.3
Main Engine Propellant	--	---------	464.4	0.00	0.0	464.4
Fuel	--	---------	464.4	0.00	0.0	464.4
Fuel usable	1	453.1	453.1	0.00	0.0	453.1
Fuel boiloff	0	0	0.0	0.00	0.0	0.0
Fuel residuals (unused)	1	11.33	11.3	0.00	0.0	11.3
Oxidizer	--	---------	0.0	0.00	0.0	0.0
Oxidizer usable	0	0	0.0	0.00	0.0	0.0
Oxidizer boiloff	0	0	0.0	0.00	0.0	0.0
Oxidizer residuals (unused)	0	0	0.0	0.00	0.0	0.0
Pressurant	0	0	0.0	0.00	0.0	0.0
RCS Propellant	--	---------	2.9	0.00	0.0	2.9
Fuel	--	---------	0.0	0.00	0.0	0.0
Fuel usable	0	0	0.0	0.00	0.0	0.0
Fuel boiloff	0	0	0.0	0.00	0.0	0.0
Fuel residuals (unused)	0	0	0.0	0.00	0.0	0.0
Oxidizer	--	---------	0.0	0.00	0.0	0.0
Oxidizer usable	0	0	0.0	0.00	0.0	0.0
Oxidizer boiloff	0	0	0.0	0.00	0.0	0.0
Oxidizer residuals (unused)	0	0	0.0	0.00	0.0	0.0
Pressurant	1	2.929	2.9	0.00	0.0	2.9

Minor size changes were to made to match propellant load determined by the in-house developed Mission analysis tool mentioned in the mission assumptions section. These changes consisted primarily of

increasing the tank length but keeping the tank diameter fixed. A propellant management device is included in tank mass roll-up.

A He pressurization system is necessary to maintain the propellant delivery pressure to all of the engines. This system is comprised to two spherical metallic Ti tanks and a pressurant distribution and control feed system. The pressurization subsystem is derived from current configuration designs or existing hardware to minimize or eliminate technology development activity. For example, the pressurant tanks are based on an Arde, Inc., high-pressure tank that is readily available.

All components of propulsion subsystem are at a high technology readiness level, so little or no development costs required.

4.5.4 Propulsion and Propellant Management Trades

The primary trade study performed within the propulsion subsystem was to consider a Blowdown propellant storage and delivery approach versus an inert gas pressurization subassembly. Preliminary trade on Blowdown system found the tanks required for storage of the hydrazine propellant to be heavier and require more volume within the spacecraft. While more complex, the He pressurization system was selected in order to save both weight and volume. Single metallic tank selected for storage of the hydrazine propellant. While this configuration is zero-fault tolerance, it has been the approach of Exploration to consider passive elements to have innately lower risk; consequently the approach for the tank, as well as the propellant feed lines is acceptable.

4.5.5 Propulsion and Propellant Management Analytical Methods

The analytical techniques used in the propulsion subsystem model are as follows:

1. Mass roll-up of the known components and assemblies within the model. This included the thrusters, the feed system control components, and instrumentation.
2. Hoop Stress analysis with material characteristics derived from commercial-off-the-shelf (COTS) high-pressure tanks were used to properly size the propellant storage and pressurant tanks in response to changes in propellant loads.
3. Feed lines were estimated from expected run lengths within the spacecraft and lineal density calculations developed for hypergol propellants.
4. Pressurant gas estimates were made with gas density estimates obtained with National Institute of Standards and Technology (NIST) databases.

4.5.6 Propulsion and Propellant Management Risk Inputs

Two potential risks were identified:

1. RCS Thruster Plume Impingement (LOW)
   a. Degradation of spacecraft performance due to deposition on sensitive surfaces, such as the solar arrays or the radiators may be possible due to the RCS thruster plumes. Careful examination of potential interferences should be undertaken.
2. Propellant freezing within fluid lines (LO-MED)
   a. Freezing of the hydrazine within the feed lines, particularly the lines going to the RCS thrusters, could lead to line ruptures or other performance failures. Careful modeling of the spacecraft thermal environment to understand where cold spots can form and proper configuration of thermal insulation is necessary to protect the integrity of the feedlines.

4.5.7 Propulsion and Propellant Management Recommendation

The propulsion subsystem used in the LRS vehicle analysis is a conventional assembly of mature and proven components. No new technologies are needed for this vehicle. Therefore, the best

recommendation to be made is that proper modeling and investigation of the thermal conditions on the spacecraft be performed to insure that propulsion subsystem can operate properly.

4.6   Thermal Control

4.6.1   Thermal Requirements

The thermal requirements for the LRS were to provide a means of cooling the spacecraft during operation as well as provide heat to vital components and systems to maintain a minimum temperature throughout the mission.

The maximum heat load to be rejected by the thermal system was 480 W, and the desired operating temperature for the electronics and propellant was 300 K.

4.6.2   Thermal Assumptions

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

- The view factors for the radiator to the Earth, lunar surface and solar array were assumed to be 0.1, 0.25, and 0.1, respectively.
- The maximum angle of the radiator to the Sun was 15°.
- The radiator temperature was 320 K.

4.6.3   Thermal Design and MEL

The thermal system is used to remove excess heat from the electronics and other components of the system as well as provide heating to thermally sensitive components during periods of inactivity. Figure 12 shows an illustration of the major items in the thermal system.

Excess heat is collected from a series of Al cold plates located throughout the interior of the spacecraft. 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 spacecraft 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.

Multilayer insulation is also utilized on the spacecrafts, and propellant system to regulate and maintain the desired temperatures.

Figure 12 is a notional block diagram of the separate components of the LRS thermal subsystem. Table 15 lists the MEL for the thermal control system sketched in Figure 12 used to perform the LRS Mission.
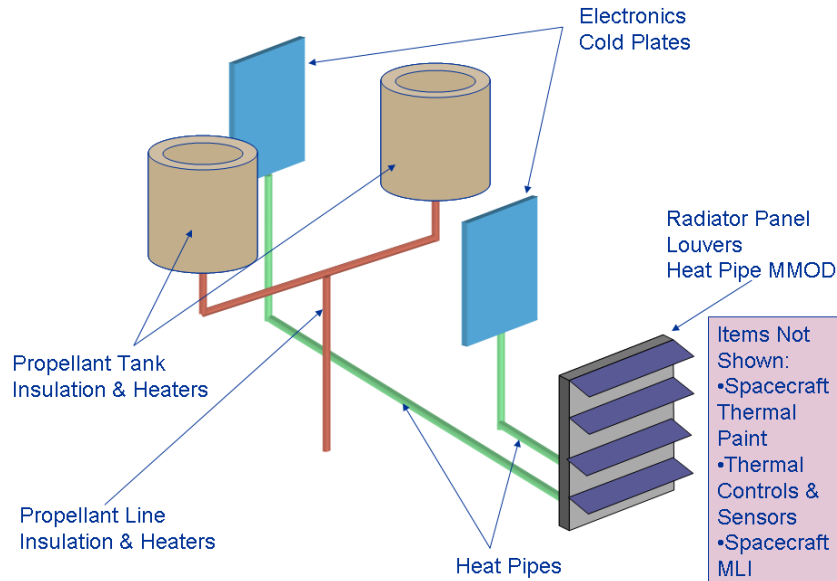
Figure 12.—LRS thermal control system.

TABLE 15.—LRS THERMAL CONTROL SYSTEM DETAILED MEL

Description	Quantity	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
Lunar COMM Relay Satellite (LRS)	-------	---------	1030.5	8.7	90.1	1120.6
LRS-2	-------	---------	1030.5	9	90.1	1120.6
Thermal Control (Non-Propellant)	-------	---------	47.3	14	6.7	54.0
Active Thermal Control	-------	---------	4.7	12.77	0.6	5.3
Heaters	10	0.143	1.4	10.00	0.1	1.6
Thermal control/heaters circuit	2	0.2	0.4	10.00	0.0	0.4
Data acquisition	1	1	1.0	15.00	0.2	1.2
Thermocouples	25	0.01	0.3	10.00	0.0	0.3
Radiator MMOD shielding	2.07	0.769	1.6	15.00	0.2	1.8
Miscellaneous no. 2	0	0	0.0	0.00	0.0	0.0
Passive Thermal Control	-------	---------	34.9	14.32	5.0	39.8
Heat sinks	4	3.463	13.9	15.00	2.1	15.9
Heat pipes	1	1.98	2.0	15.00	0.3	2.3
Radiators	1	12.12	12.1	15.00	1.8	13.9
MLI	1	5.497	5.5	12.00	0.7	6.2
Temperature sensors	50	0.01	0.5	10.00	0.1	0.6
Phase change devices	0	0	0.0	0.00	0.0	0.0
Thermal coatings/paint	1	0.907	0.9	10.00	0.1	1.0
Miscellaneous no. 1	0	0	0.0	0.00	0.0	0.0
Miscellaneous no. 2	0	0	0.0	0.00	0.0	0.0
Semi-Passive Thermal Control	-------	---------	7.8	14.49	1.1	8.9
Louvers	1	6.977	7.0	15.00	1.0	8.0
Thermal switches	4	0.2	0.8	10.00	0.1	0.9

4.6.4 Thermal Analytical Methods

The analysis performed to size the thermal system is based on first principle heat transfer from the spacecraft 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 or the vehicle is being insulated from. For more detailed information on the thermal analysis a summary white paper titled "Preliminary Thermal System Sizing" was produced.

4.6.5　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, micrometeor impact or design defect. The likelihood of this type of failure is low. The impact of this failure would be a loss of all or a portion of the spacecraft'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.

4.6.6　Thermal Recommendation

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

- Redundant heat pipes 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 spacecraft to insure that the interior components do not drop below their desired minimum temperature based on a known shadow period of operation.

5.0　Cost and Risk

5.1　Costing

Costing results and assumptions will be documented in a subsequent report.

5.2　Risk Analysis and Reduction

- Risk Management Requirements for LRS
  - The management of risk is a foundational issue in the design, development and extension of technology.
  - 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.

5.2.1　Risk Assumptions

- Assumptions for Risk Management
  - Risk attributes are based on Crew Exploration Vehicle (CEV) risk values.
  - Risk List is not based on trends or criticality.
  - Some mitigation plans are offered as suggestions.
  - Based on the derived model of this study only.

5.2.2　Risk Matrix and Risk List

TABLE 16.—LRS RISK MATRIX
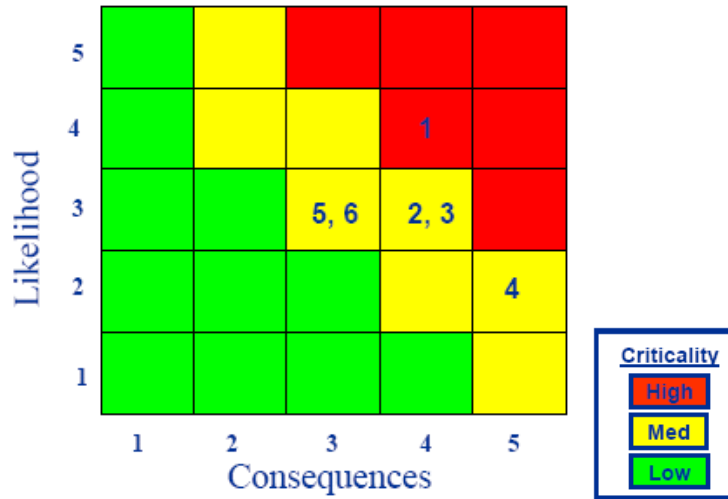
No.	L×C	Team	Risk title
1	4×4	Design	Overly Complex Spacecraft Design
2	3×4	Avionics	Avionics System Functions Not Completely Defined
3	3×4	Avionics	Data Recorder Design Based on Future Technology
4	2×5	Structure	Potential Structure Failure
5	3×3	Avionics	Atomic Clock at Low TRL
6	3×3	Communications	Health Management

TABLE 17.—LRS RISK LIST AND DETAILED DATA TABLE

No. & Criticality	Team & Attribute Values	Risk Title and Statement
1	Design	**Overly Complex Spacecraft Design**
High	22 Lk: 4 Consq; Cost: 3 Sched: 3 Safety: 4 Perform: 3	Given the overly complex spacecraft design; there is a possibility that the pointing three parts (lunar antennas, Earth antennas, and solar arrays) will degrade or fail during operation *Context*: The overly complex design has a higher probability of degradation or failure than is normal for a communications satellite. Cost is high to supplement LRS with ground system **Mitigation Strategy**: Ensure sufficient resources for design and test the system
2	Avionics	**Avionics System Functions not completely defined**
Med	19 Lk: 3 Consq; Cost: 4 Sched: 3 Safety: 2 Perform: 4	➢ Given the functions of the avionics system are not well defined; there is a possibility that the avionics hardware may be considerably over or under designed. *Context*: The avionics system components are currently spec'ed out for what is perceived to be their functionality. Given the immaturity of the LRS design, the size, complexity, and mass of the avionics components could be significantly different from this initial estimate **Mitigation Strategy**: Design the avionics system with the largest reasonable growth factor to account for changes in the system's functionality.
3	Avionics	**Data Recorder Design based on future technology**
Med	19 Lk: 3 Consq; Cost: 3 Sched: 3 Safety: 2 Perform: 4	➢ Given that the data recorder design is based on future technology that may not be available for inclusion in the LRS; there is a possibility that the resulting design will be a larger and/or heavier data recorder. *Context*: The data recorder serves as a buffer between lunar communications and Earth. It mush have enough storage to buffer data when communication is not available. **Mitigation Strategy** : The LRS design must take into consideration that the data recorder may be larger than initially anticipated.
4	Structural	**Potential Structural Failure**
Med	17 Lk: 2 Consq; Cost: 3 Sched: 5 Safety: 1 Perform: 4	➢ Given the chances of foreign object impacts and/or induced vibrations; there is a possibility that the performance may be diminished or the system may be incapacitated. *Context*: Impact damage may cause alignment issues with instrumentation and/or propulsion systems or completely incapacitate the system. Excessive vibrations may reduce instrumentation performance and/or potentially lead to long term structural failure due to fatigue **Mitigation Strategy**: The structure should be designed to NASA standards to withstand a given impact and to have sufficient stiffness and dampening to minimize issues with vibrations. Trajectories are to be planned to minimize the probability of impact with foreign objects

TABLE 17.—CONCLUDED.

No. & Criticality	Team & Attribute Values	Risk Title and Statement
5	Avionics	■ **Atomic Clock at Low TRL** ➢ Given that the atomic clock is at TRL level 6, there is the possibility that it could be larger and use more power than estimated.
Med	15 Lk: 3 Consq; Cost: 3 Sched: 3 Safety: 2 Perform: 2	*Context*: The avionics system is spec'ed out to use an atomic clock which is still under development. **Mitigation Strategy**: Design the avionics system with the largest reasonable growth factor to account the uncertainty in the development of the atomic clock.
6	Communic;	■ **Health Management** ➢ Given that unexpected failures can occur in electrical/mechanical/ structural system, there is a possibility that a critical failure can occur, causing loss of communications and/or navigation capability. .
Med	15 Lk: 3 Consq; Cost: 3 Sched: 3 Safety: 3 Perform: 3	*Context*: Health Technology (i.e., fault detection, identification, and repair) can be designed into the system to provide long term functionality **Mitigation Strategy**: Electrical/Mechanical/Structural systems should be designed with fault tolerance as a requirement

5.2.3 Risk Summary

- Areas of concern for this study include:
  - Overly complex design
  - Design of Avionics
  - Available technology for communications
  - Impact damage to the structures
- These risks, with proper pro-active planning can be mitigated early to avoid becoming problems late in the development life cycle.

Appendix A.—Acronyms and Abbreviations

AF	U.S. Air Force
Al	aluminum
AO	Announcement of Opportunity
BAE	British Aerospace
C&DH	Command and Data Handing
CBE	current best estimate
CEV	Crew Exploration Vehicle
Comm	Communications
COMPASS	COllaborative Modeling and Parametric Assessment of Space Systems
COTS	commercial off the shelf
DMR	Design for Minimum Risk
DSN	Deep Space Network
DTE	direct to Earth
EELV	Evolved Expendable Launch Vehicle
EP	Electric Propulsion
ESPA	EELV Secondary Payload Adaptor
FEA	finite element analysis
FOM	figure of merit
GSFC	NASA Goddard Space Flight Center
GLIDE	GLobal Integrated Design Environment
GN&C	Guidance, Navigation and Control
GRC	NASA Glenn Research Center
hab	habitat
HQ	NASA Headquarters
IMDC	I M Design Center
IP	internet protocol
ISRU	in situ resource utilization
He	helium
JPL	NASA Jet Propulsion Laboratory
KSC	NASA Kennedy Space Center
LCT	Lunar Communications Terminal
LRS	Lunar Relay Station
LSP	Launch Service Program
LSTO	Launch Service Task Order
MAC	Media Access Control
MEL	Master Equipment List
MPU	Makeup Power Unit
NASA	National Aeronautics and Space Administration
Nav	navigation
NIST	National Institute of Standards and Technology

NLS	NASA Launch Services
OMS	Orbital Maneuvering System
PEL	Power Equipment List
PN	pseudo-noise
RCS	Reaction Control System
S/C	spacecraft
SADA	Solar Array Drive Assembly
SEP	Solar Electric Propulsion
SN	signal-to-noise
SPACE	System Power Analysis for Capability Evaluation
SPU	Solar Power Unit
Ti	titanium
TDRSS	Tracking and Data Relay Satellite System
TWTA	Traveling Wave Tube Amplifier

Appendix B.—Rendered Design Drawings

Figure B.1 and Figure B.2 show two views of the stowed configuration of the LRS with arrays folded at the sides, and the Q-band antenna on the top face also showed in the stowed position with major spacecraft dimensions.
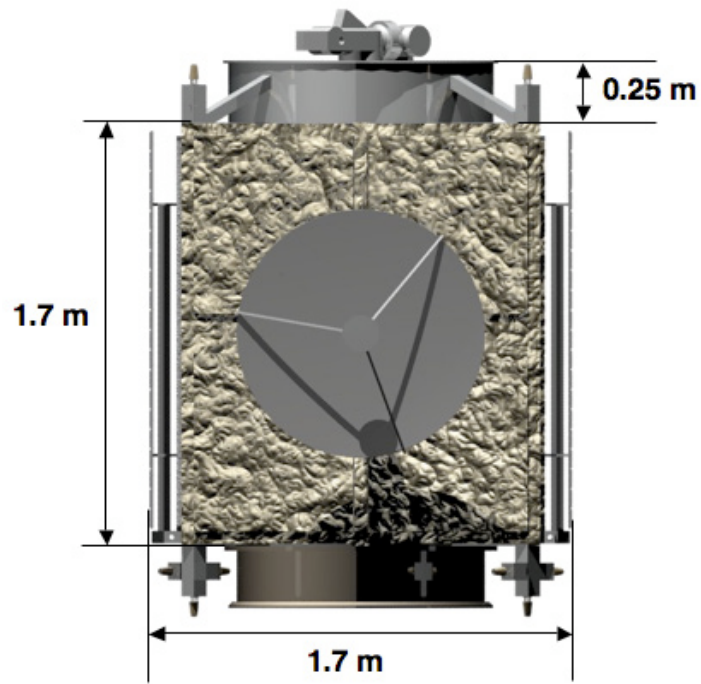
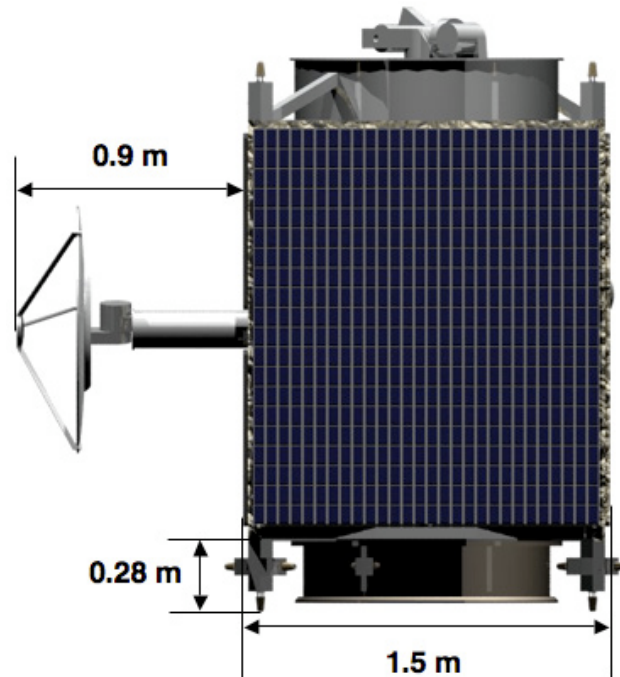Figure B.1.—LRS stowed configuration K/Ka band antenna face.

Figure B.2.—LRS stowed configuration, solar array panel side.

Figure B.3 and Figure B.4 show the LRS in deployed configuration with both the two solar arrays and the Q band antenna on the top face of the satellite deployed.
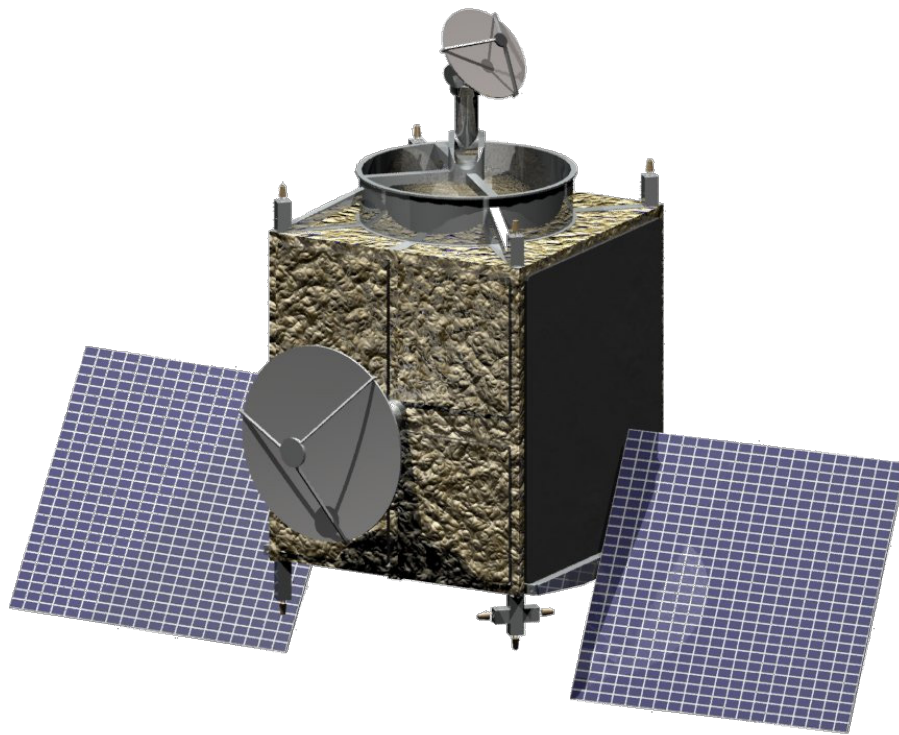
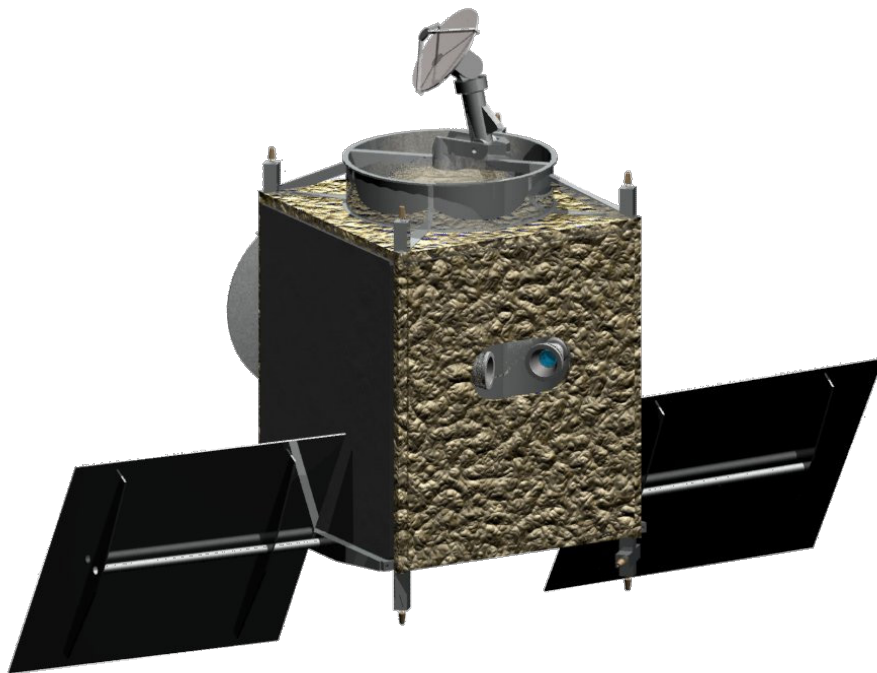Figure B.3.—LRS deployed configuration, K/Ka band antenna face.

Figure B.4.—LRS deployed configuration.

Appendix C.—Study Participants

Lunar Relay Satellite (LRS) Design Session			
Subsystem	**Name**	**Center**	**Email**
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1. REPORT DATE (DD-MM-YYYY) 01-04-2012	2. REPORT TYPE Technical Memorandum	3. DATES COVERED (From - To)

4. TITLE AND SUBTITLE COMPASS Final Report: Lunar Relay Satellite (LRS)	5a. CONTRACT NUMBER
	5b. GRANT NUMBER
	5c. PROGRAM ELEMENT NUMBER
6. AUTHOR(S) Oleson, Steven, R.; McGuire, Melissa, L.	5d. PROJECT NUMBER
	5e. TASK NUMBER
	5f. WORK UNIT NUMBER WBS 986249.01.11.20.22.03

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES) National Aeronautics and Space Administration John H. Glenn Research Center at Lewis Field Cleveland, Ohio 44135-3191	8. PERFORMING ORGANIZATION REPORT NUMBER E-17877

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES) National Aeronautics and Space Administration Washington, DC 20546-0001	10. SPONSORING/MONITOR'S ACRONYM(S) NASA
	11. SPONSORING/MONITORING REPORT NUMBER NASA/TM-2012-217140

12. DISTRIBUTION/AVAILABILITY STATEMENT
Unclassified-Unlimited Subject Categories: 17, 18, and 19 Available electronically at http://www.sti.nasa.gov This publication is available from the NASA Center for AeroSpace Information, 443-757-5802

13. SUPPLEMENTARY NOTES

14. ABSTRACT

The Lunar Relay Satellite (LRS) COllaborative Modeling and Parametric Assessment of Space Systems (COMPASS) session was tasked to design a satellite to orbit in an elliptical lunar polar orbit to provide relay communications between lunar South Pole assets and the Earth. The design included a complete master equipment list, power requirement list, configuration design, and brief risk assessment and cost analysis. The LRS is a half-TDRSS sized box spacecraft, which provides communications and navigation relay between lunar outposts (via Lunar Communications Terminals (LCT)) or Sortie parties (with user radios) and large ground antennas on Earth. The LRS consists of a spacecraft containing all the communications and avionics equipment designed by NASA Jet Propulsion Laboratory's (JPL) Team X to perform the relay between lunar-based assets and the Earth. The satellite design is a standard box truss spacecraft design with a thermal control system, 1.7 m solar arrays for 1 kWe power, a 1 m diameter Ka/S band dish which provides relay communications with the LCT, and a Q-band dish for communications to/from the Earth based assets. While JPL's Team X and Goddard Space Flight Center's (GSFC) I M Design Center (IMDC) have completed two other LRS designs, this NASA Glenn Research Center (GRC) COMPASS LRS design sits between them in terms of physical size and capabilities.

15. SUBJECT TERMS
Lunar communication; Parabolic reflectors; Lunar satellites; Relay satellites; Aerospace systems; Satellite orbits; Elliptical orbits; Lunar orbits

16. SECURITY CLASSIFICATION OF:			17. LIMITATION OF ABSTRACT	18. NUMBER OF PAGES	19a. NAME OF RESPONSIBLE PERSON STI Help Desk (email:help@sti.nasa.gov)
a. REPORT U	b. ABSTRACT U	c. THIS PAGE U	UU	46	19b. TELEPHONE NUMBER (include area code) 443-757-5802