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Small Satellite Implementation of a Lunar Relay Satellite

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ABSTRACT

The authors will report on a small lunar relay satellite concept, capable of fitting in a Minotaur V launch vehicle, to provide communication and navigation (C&N) services to users on the moon. Ultimately, the C&N system will provide multiple fixed and mobile users, human and robotic, with service from launch to lunar landing and throughout occupation. In order to provide complete coverage for users on the moon, a constellation of relay satellites will ultimately be required. In the near term, however, the relays will focus on certain important locations, particularly the South Polar region, where water may be located and early landings may occur. These locations are only visible from Earth antennas for 14 days of each 28 day time period, requiring a relay satellite to fill the gaps. One alternative for this satellite is a small, less costly lunar relay capable of launching on the relatively inexpensive Minotaur class of launch vehicles, the Minotaur V in particular. In this paper, this spacecraft concept will be discussed, including the mission context, concept of operations, mission requirements, payload concept, ground system concept and spacecraft concept.

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INTRODUCTION

NASA's Vision for Space Exploration envisions a permanent human presence on the moon beginning no later than 2020. The South Polar region has been chosen as the initial destination for both resource extraction and scientific reasons. However, this region presents a communication challenge. For around 14 days out of a month, the lunar south pole doesn't have a direct Line of Site (LOS) to the Earth. Since wide ranging expeditions on the lunar surface are expected, as well as resource extraction within craters, crater walls and other features on the lunar surface will cause additional LOS blockages. A lunar relay satellite system would provide coverage during the outages and give users in craters or terrain screened from the Earth communication and navigation services.

NASA has conducted various studies to investigate the requirements and develop design concepts for lunar relay satellites (LRS). Most of these are relatively large requiring Delta II class vehicles, or lacking the Delta II, Delta IV and Atlas V launch vehicles. To reduce the launch vehicle requirements and, therefore, lower the cost, our study focused on developing a satellite concept that can fit on a relatively cheap Minotaur V class vehicle.

Although Minotaur V compatibility was the goal of this study, the orbits, coverage, pointing scheme and navigation discussions apply to any lunar relay satellite. These general considerations fed into the payload and satellite implementations, with the additional constraint of low mass to conform to the Minotaur V capabilities.

CONCEPT OF OPERATIONS

The small S-band relay provides Communication and Navigation to surface assets with limited Direct to Earth (DTE) access during 14 days periods or no visibility in craters. Users within 250km of the south pole are supported with the specified capability of the relay discussed in the payload section below, those outside 250km are supported with potentially lesser capability depending on their location and the equipment they carry.

The LRS services provided include forward command at 4 Kbps, return mission data telemetry at 192 Kbps for up to 3 simultaneous users, 3Mbps for 1 user, 1 and 2 Way ranging and Doppler tracking and time dissemination to users in the lunar vicinity including tracking during approach, low orbit, descent/ascent, and surface operations.

The proximity links (lunar links) are S band. The trunk line back to Earth is at Ka 37/40Ghz. One

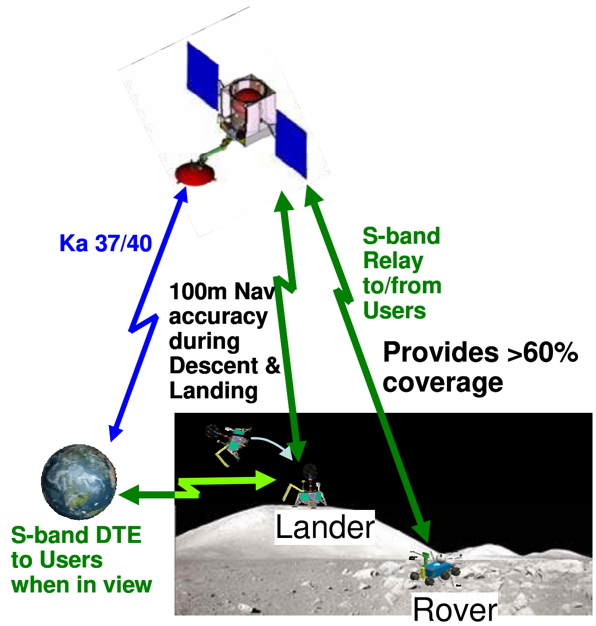

Figure 1: LRS Concept of Operations Diagram

relay satellite can provide 60% coverage from a 12 hour frozen orbit. The orbits will be discussed in more detail later.

LAUNCH VEHICLE

A driving requirement of this satellite is to lower the cost by fitting within the capabilities of a relatively inexpensive Minotaur V launch vehicle[9,10]. The Minotaur V launching from Wallops Island, VA is expected to loft between 400 to 450 kg of usable wet mass to Lunar transfer orbit. In comparison, a small Delta II (7326 -9.5) can loft 605 kg to Trans Lunar Injection (TLI) orbit and a Delta 7920-10 could direct inject 650+ kg to TLI. Launch windows are limited to several short windows a month per lunar cycle.

The Minotaur V is a Minotaur IV with a Star 48 instead of Orion 37 4th stage and an added Star 37M or 37FMV 5th stage. Stages 1 to 3 on both the V and IV are of Peacekeeper heritage. The Minotaur V is projected to be available 24 months after contract award (including 6 months Non Recurring Engineering, normal flow 18 months).

ORBIT AND COVERAGE

Previous lunar relay studies[1] have examined a wide range of orbit options ranging from polar orbits (circular & elliptical) to orbits at the Earth-Moon libration points. In the end, the orbit selected to meet the requirements listed above was an inclined, elliptical, lunar frozen orbit[2,3,4]. Judicious selection of the frozen orbit parameters allows for stable and bounded motion of the orbit semi-major axis, inclination, and most importantly the argument of periapsis. The stability of the argument of periapsis ensures that the apoapsis location can be set to be in the southern hemisphere and it will stay there for the duration of the mission, even in the absence of stationkeeping maneuvers. Furthermore, a southern hemisphere apoapsis maximizes coverage to South Pole users. The orbit selected for the LRS is a 718 x 8090 km altitude orbit (12-hour period) with a lunar inclination of 57° and an argument of periapsis of 90°.

Using the Minotaur-V to launch the lunar relay satellite to the Moon does present some challenges. The main challenge is being able to compensate for the launch vehicle dispersions from the all-solid launch vehicle. A direct injection into a trans-lunar orbit would require a large ΔV (> 230 m/s) correction maneuver 24 hours after separation. Using a phasing loop strategy would help to mitigate the large launch dispersions and reduce the correction ΔV by half. Both strategies would require large maneuvers for lunar orbit insertion and establishing the frozen orbit conditions. In the end, the phasing loop strategy ΔV budget was computed to be 841 m/s while the direct injection ΔV budget was computed to be 965 m/s. These ΔV budgets were computed assuming impulsive maneuvers. In designing the spacecraft bus it became apparent that storing the amount of propellant for a chemical system became prohibitive. For this reason, it was decided to move to a solar electric propulsion (SEP) system. In using the low-thrust SEP system, the relay satellite is launched directly to the moon. Following a lunar swingby, the thrusters are used to target the capture and spiral into the final mission orbit[5].

The coverage analysis for this orbit can be broken up into two segments – the coverage from a South Pole user to the lunar relay and the coverage from the lunar relay back to some network of ground stations at the Earth (e.g. the Deep Space Network). A single lunar relay satellite has 63% visibility (an average 7.6 hours of coverage per 12 hour orbit) to a South Pole user (assuming a 15° minimum elevation). The visibility can be increased to 100% by employing a two-satellite constellation with the satellites phased apart by 180°. The two-satellite visibility also includes overlapping coverage overlap (from both relay satellites) ranging from 1 - 2 hours. Coverage from the lunar relay back to the Earth is dependent on the number of ground stations available. Full coverage (100%) of the lunar relay satellite (minus the time when the relay satellite is occulted by the Moon) is only possible with three, equally spaced ground stations. Reduced coverage is available with use of two (81%) or one (43%) ground stations.

While in the relay orbit, the power subsystem must be able to handle solar eclipses due to occultations by both the Moon and the Earth. Lunar eclipses happen frequently (roughly 300 per year) and are usually less than 1 hour per event. The Earth eclipses, while more infrequent (roughly 2-3 per year) can be more severe. Simulations have shown that the Earth eclipse can be up 6.5 hours in length with an average of about 3.3 hours. The frequency of solar eclipses has implications on the satellite design. It must be decided whether to continue operations during these eclipses or to go into a stand-by mode. Continuous operations requires a larger battery which could produce a large mass penalty. An alternative option is to use propellant to perform an orbit phasing maneuver to either eliminate or at least minimize the duration of the eclipse. Further analysis is required on the particulars of such a maneuver. A trade-off of the propellant required for such a maneuver versus increased battery mass will help to shape the strategy employed.

NAVIGATION

The lunar relay system will provide radiometric tracking services to user's in the vicinity of the Moon to aid their navigation during lunar approach, orbiting, descent/ascent, and surface mission phases. The radiometric tracking service provided by the relay is 1-Way or 2-Way range, Doppler, and time correlation (via use of the 1-Way ranging). Additionally, the relay satellite will carry an atomic clock that enables a synchronous 1-Way range service (like GPS) when the lunar network becomes fully populated with multiple orbiters and ground terminals.

The signal design for radiometric tracking works conceptually like GPS, however the system does support the formation of 2-Way observables, in addition to the 1-Way. Conceptually this works as follows:

1. The relay transmits a forward link pn-sequence at a fixed frequency to a user.

2. The user utilizes a demod/remod system that receives and correlates on the arriving pn-sequence and phase locks on the carrier. The user is able to collect 1-Way pseudo-range and carrier phase that are time tagged by the user clock. If they have a USO then the 1-Way data may be sufficient for immediate processing to obtain a navigation solution. For 2-Way data the user coherently transponds the carrier frequency and regenerates the pn-sequence on the return link.

3. The relay receives the signal and correlates on the pn-sequence leading to the measurements of 2-Way range and phase locks to get a 2-Way carrier phase that gets time tagged by the relay's atomic clock.

4. The relay immediately relay's the measurements to the user.

5. Finally, in parallel to the measurement process the relay is continuously broadcasting a low-rate navigation message that contains the relay ephemerides and relay clock models. Collectively, the combination of the navigation message and the observables are enabling for user's to compute dynamic navigation solutions.

Finally, in its final form the signal is designed using CDMA that enables the user to receive multiple links from other lunar network assets that can be used to collect additional 1-Way range and carrier phase measurements simultaneously with the 2-Way data that is being telemetered by the relay. This multiplicity of measurements is enabling for a rudimentary kinematic positioning service on the surface of the Moon.

One possible role of this first relay is validate the radiometric tracking service capability with precursor robotic missions during a robotic missions phases at the Moon including approach (with a relay acquisition distance of up to 80000 km), orbit, and, if it's a lander, descent/landing. Preliminary assessments of navigation during these mission phases with a single orbiter suggests that orbit determination of a low altitude orbiter using a combination of current DSN tracking and relay tracking could range from 10 - 100 m (1-sigma). Detailed navigation performance with this system is a topic of current study.

For the relay to provide a precision navigation service the ephemeris quality of the navigation message must yield at least 10 m (1-sigma) knowledge at the current time and a 100 m (1-sigma) error for predicts of up to a week. To obtain this level of performance the relay employs coupled thrusts for momentum desaturations, and since the orbiter is at such a high altitude lunar gravity errors have a small effect on the overall satellite ephemeris error.

LRS COMMUNICATIONS PAYLOAD

Requirements

The lunar relay satellite communications system is required to provide 8 hours of (near continuous) communications services per 12 hour orbit to surface users within 250 km (radius) of the south pole. The relay also provides communications services to orbiting assets and the Crew Exploration Vehicle on a best effort basis. Surface users are assumed to be landers or robotic rovers. Early lunar exploration missions are expected to have fairly modest set of data rate requirements (summarized below). The perimeter of the 250km radius coverage area corresponds to 82^{o} latitude. Users at latitudes between 82^{o} and the equator, and on the front side of the moon have a direct line of sight to earth at all times, therefore, a larger area of coverage from the satellite is not needed.

The relay return link (user to relay satellite) is required to support one to three users simultaneously. Thus it has a multiple access (MA) and single access (SA) mode. The relay forward link (relay satellite to user) is SA only.

The data rate requirements for the various users are tabulated below.

User	Relay Return Link	Relay Forward Link
Lander/Rover	10 Gb/day	4 kbps
Orbiter	3 Mbps (duration ½ hour)	10 kbps
CEV	192 kbps	72 kbps

Table 1: Relay Link User Requirements

Noting that every 24 hours, the LRS can relay 14.9 hours in real time from a surface user to the ground stations (assuming 10^o elevation mask). With the need of 10 Gb/day, a lander/rover must transfer at least 175 kbps. For simplicity we assume that a lander/rover transmits at 192 kbps, the same rate as the CEV. We further assume that the relay PHY layer is compatible with the CEV which uses Space Network (SN) signaling. Consistent with SN signaling the relay satellite shall support a maximum MA rate of 192 kbps, and a maximum SA rate of 3 Mbps. The CEV data rate requirement is constrained by the CEV EIRP (18.5 dBW) and G/T (-28.1 dB/K) specified in the CEV master link book. The CEV master link book is maintained by NASA's constellation program.

The relay EIRP is required to be high enough to close the lander/rover forward link with 3dB margin to a lander/rover receiving with an omnidirectional antenna.

The satellite trunk line or Direct-to-Earth (DTE) link is required to transfer up to 20 Gb/day. Assuming an 8 hour contact with an earth station every 24 hours, and an earth station elevation angle of 10^o, requires the DTE link to support a minimum data rate of 694 kbps. For real-time communications with an orbiter, it is also desirable for the DTE link to operate up to 3 Mbps.

The link frequencies for relay and DTE links are required to be consistent with the recommendations of the Space Communications

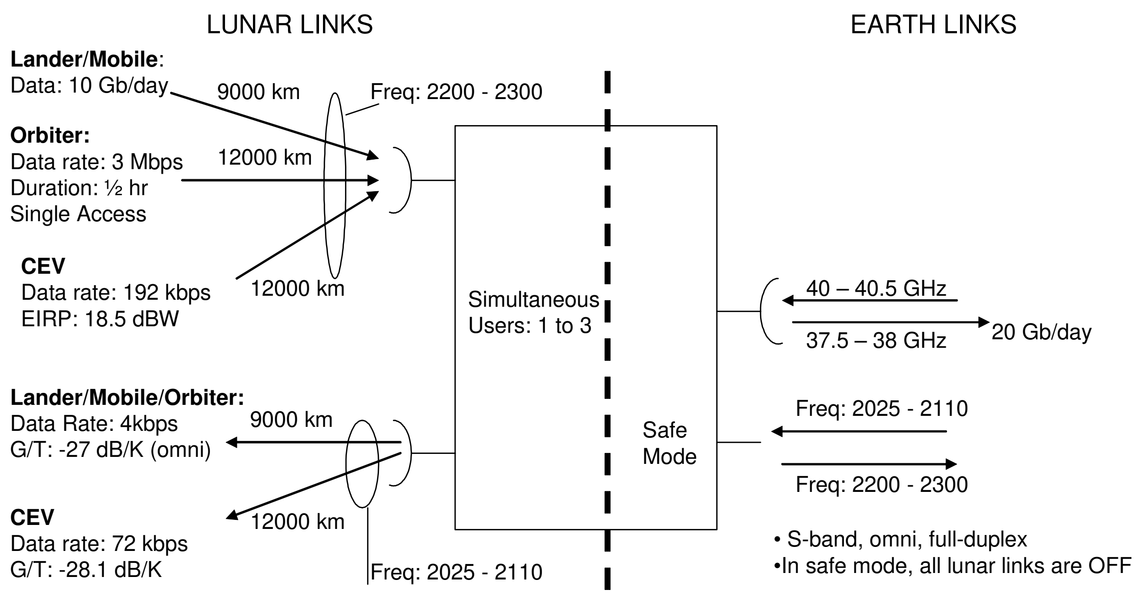

Figure 3: Satellite Frequency Plan, data rates, and slant range to various users

Architecture Working Group[6]. Accordingly, the satellite spectrum plan is required to be as shown in Figure 3. The data rates, and slant ranges for the various links are also shown in Figure 3. Note that the satellite safe mode and relay link operational frequencies are reversed. That is, the relay Tx / safe mode Rx and relay Rx / safe mode Tx are in the same band. To avoid interference between these links, the assumption is that the relay link will be off when in safe mode.

Finally, the communications payload mass and DC power are required to be less than 50 kg and 200 W, respectively. The payload mass and power limits are driven by a desire to fit the satellite in a Minotaur V launch vehicle.

Link Design

The following sections derive a specific payload design based on the service requirements above.

Relay Link

A key trade in designing the communications payload is the size of the relay antenna. The relay antenna aperture should be optimized to meet the 250 km (radius) coverage requirement and simultaneously minimize user EIRP. While a smaller aperture increases the area of coverage, it requires a higher user EIRP Conversely, a surface user with insufficient EIRP requires a larger satellite antenna, resulting in an aperture that covers less than 250km. To determine the appropriate relay antenna size, three apertures (1m, 1.5m and 3m) were analyzed in detail.

Because the orbit of the LRS is elliptical and inclined, its area of coverage expands and contracts with time. Figure 4 shows the instantaneous coverage as the satellite points towards the South Pole at a slanted position. When the LRS first sees the South Pole, its coverage cone is small due to its proximity distance. As the LRS rises, so does its area of coverage. To get a temporally meaningful estimate of how many hours of coverage can be provided to a 250 km region around the South Pole, simulations such as the one shown in Figure 5 were run for the three antenna sizes under consideration. The results for a 1 meter antenna (shown in Figure 5) show a color coded time gradient for hours of coverage at all latitudes and longitudes. The figure indicates that 7-9 hours of coverage is achievable well beyond the service area (demarcated by the black circle).

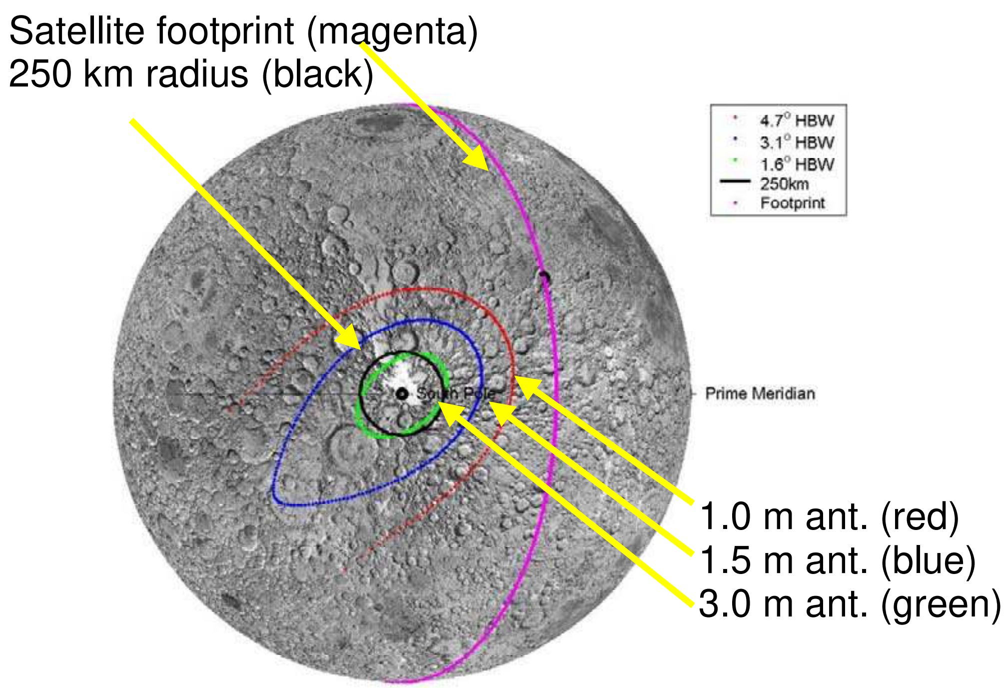

Figure 4: Instantaneous Satellite Coverage for 1m, 1.5m and 3m S-band relay Antenna

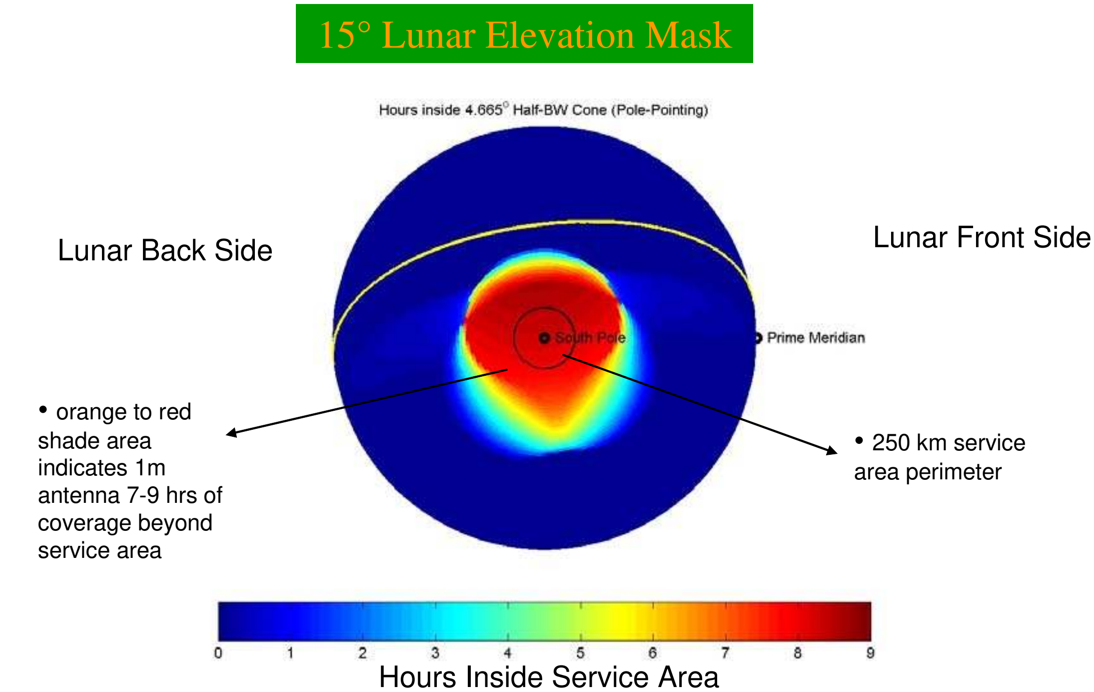

Figure 5: Satellite Coverage Time per Orbit as Function of Latitude and Longitude

Figure 6 shows the available coverage from a different perspective. The ordinate in Figure 6

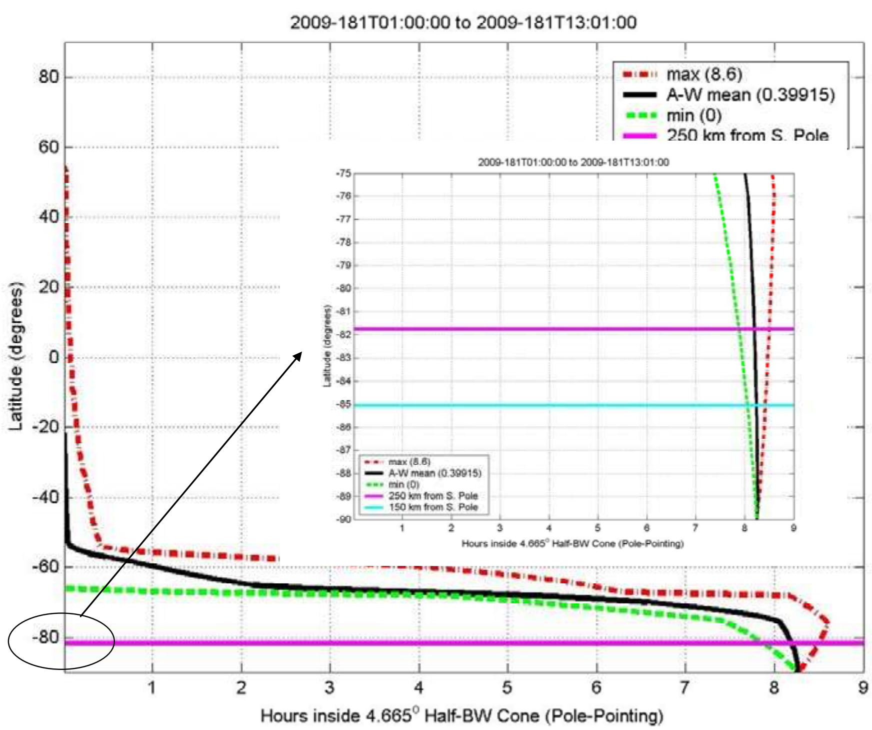

Figure 6: Satellite Coverage Time per Orbit as a Function of Latitude

are the lunar latitudes. The red curve corresponds to maximum available coverage at a given latitude. (Note that the in general, the longitude corresponding to the maximum coverage time may change from one latitude, to the next.) Similarly, the green curve shows the minimum coverage at a given latitude, and the black curve, the average. The interior figure is a zoom of the region between -75 and -90 degrees latitude. It shows that regions up to 150 km away from the South Pole have minimum coverage of at least 8 hours. Stated another way, up to 150 km from the South Pole, 8 hours of coverage per satellite orbit can be had at any longitude. Between 150km and 450 km from the South Pole, the average coverage is 8 hours. Since the average is over longitude, some longitudes get more than 8 hours of coverage, and some get less.

For the satellite under consideration, a 1m relay antenna was chosen even though it provides 8 hours well beyond the 250km coverage requirement. The antenna that comes closest to covering exactly 250km is a 1.5 m antenna. However, going to a 1.5 m aperture requires a larger nadir (bottom) deck on space craft. After taking this and other mechanical complexities into consideration, it was decided that although a 1.5m aperture would reduce user EIRP by approximately 3.5 dB, a 1m aperture is a better choice for the relay link. A 3m aperture provides 8 hours of coverage up to 150km from the south pole, and therefore, doesn't meet the 250km requirement.

Selection of the spacecraft antenna fixes the s/c receiver G/T which in turn allows us to compute the required EIRP for various relay link data rates as indicated in the Table 2.

Lander EIRP (dBW)	Relay Return Data Rate (kbps) (6 dB Margin)
5	8
8	16
11	32
14	64
17	128
20	256
23	512
26	1024
29	2048
32	4096

Table 2: Required User EIRP as a function of relay return link data rates.

Most lunar users will be power limited so their transmit powers are anticipated to be limited to 5 to 10 W. Therefore to achieve the EIRPs shown in the table above, users will have to rely on pointing medium to high gain antennas towards the satellite. To facilitate pointing, the satellite will send its ephemeris data on the relay forward link. This implies users must have the capability to receive pointing instructions through an omnidirectional antenna. The satellite relay link EIRP was chosen to be 32 dBW (obtained by using a 10W SSPA with a 1m S-band patch array), to enable an user with an omni to receive 16 kbps with 6 dB margin at a range of 9000km. Thus the forward link capability is well above the 4 kbs forward link requirement in Table 2.

Forward and return links to orbiters and the CEV are supported on a best effort basis. Link analysis shows that the satellite can support communications rates of 16 kbs and 192 kbps on the CEV forward and return links, respectively.

Hence we meet the return link CEV requirement but not the forward.

Direct-to-Earth Link

Strictly speaking, the maximum data rate on the DTE link needs to be high enough to support 3 users in real time. The highest aggregate rate on the relay side is 2 users at 192 kbps plus an orbiter at 3 Mbps. Therefore, the required DTE rate is 3.4 Mbps. Nevertheless, the DTE link EIRP was chosen to be high enough to support a 10 Mbps data rate. As shown later, the satellite has a data recorder on-board. For several reasons, such as minimizing earth antenna coverage time, it is desirable to be able to play back recorded data at as high a rate as possible. A DTE rate of 10 Mbps is the maximum rate that can be accommodated while still maintaining the 200 W DC power limit.

Using link analysis it can be shown that the required s/c EIRP to transmit 10 Mbps to a 18-m ground station (G/T of 46.2 dB/K) is approximately 48 dBW. To obtain this, the satellite transmit power was chosen to be 30 W and the DTE antenna aperture was chosen to be 0.2m. It is worth noting that the 3-dB beamwidth of 0.2m antenna covers the entire earth. So if operational simplicity is desired, the antenna pointing can be set to the earth's center. Thereby, avoiding tracking a particular ground station from the satellite.

Link Summary

The satellite connectivity is summarized in Figure 7.

Payload Architecture

The Communications payload demodulates and remodulates data received by the satellite. The satellite also has a Solid State Recorder (SSR) to which Relay Link and DTE link data flow independently. Relay data received by the satellite can be stored until it is convenient to transmit it to a Ground Station on Earth, or retransmitted in real time (with a very small processing delay). The same store and forward technique can be used for data received from the ground station if necessary.

Because of the independence of the links, the implementation of each one can be shown separately.

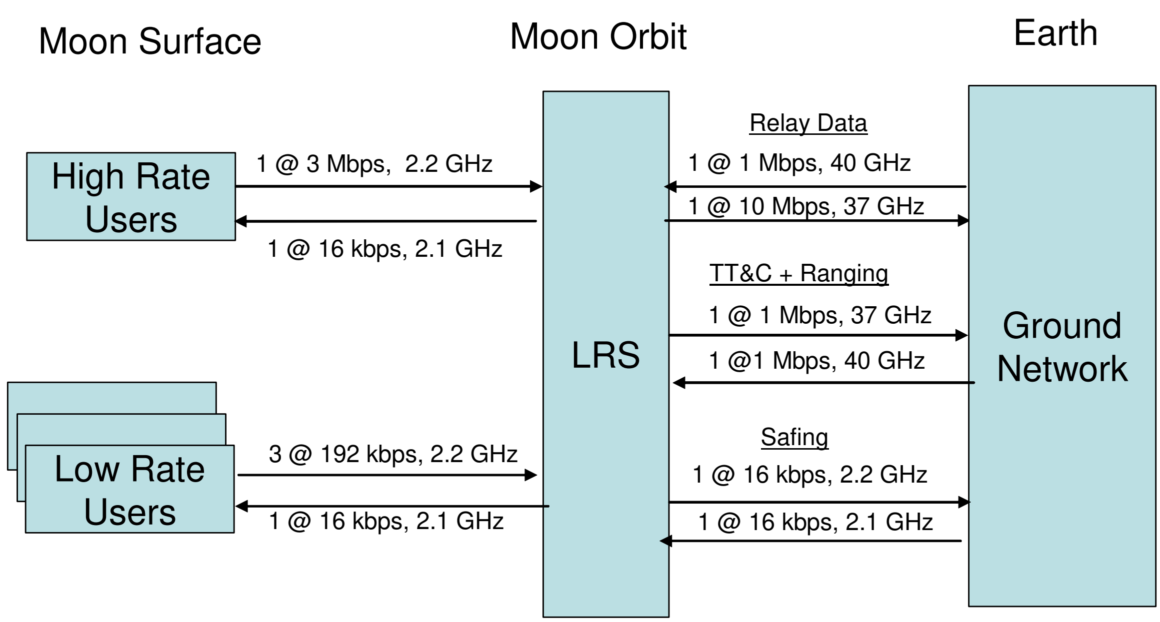

Figure 7: Lunar Relay Satellite Link Summary

Figure 8 shows the implementation of the Relay and Safing Links. It uses dual-redundant, cross-strapped transceivers. The transceivers are based on Electra Lite[7], converted to S band. At low data rates the transceivers receive Space Network signal structure[8] which provides the capability of measuring User range (through the SN PN-code) and Doppler. The transceivers also will have the capability of reversing their transmit and receive frequencies, since the Relay and Safing Links use reversed frequencies.

The Relay Antenna is a 1 meter square passive patch array, while the safing link is supported by two strategically placed LGA's.

Figure 9 shows the implementation of the Ka band DTE trunk line Links. It uses dual-redundant, cross-strapped transceivers. The transceivers are based on the Small Deep Space Transponder (SDST) converted to Ka band. The antenna is a 0.2 meter reflector.

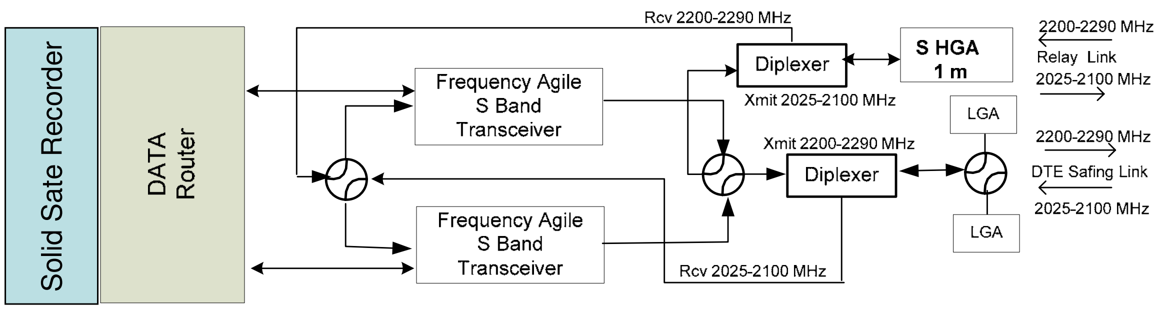

Figure 8: Payload Architecture for Relay Link and DTE Safe Link

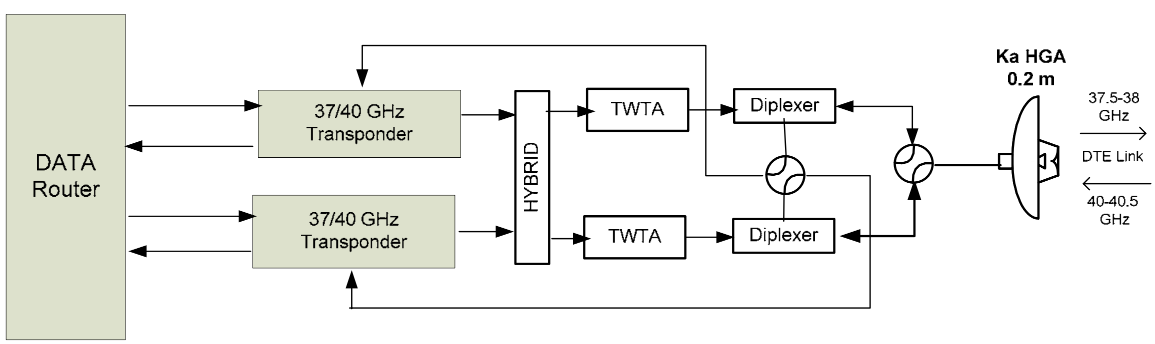

Figure 9: Payload Architecture for DTE Link

The total estimated weight and power for the payload shown in Figures 8 and 9 are 44 kg and 196 W. The proposed payload meets the subsystem weight (50 kg) and power (200 W) constraints necessary to meet the overall satellite mass and power requirements for a Minotaur V launch vehicle.

SPACECRAFT BUS

Spacecraft Overview

The spacecraft will be launched using a Minotaur V launch vehicle from Wallops or Kennedy. Selected redundancy is used to lower the mass. Cruise to the moon will take a few months using an Electric Propulsion System (EPS), also to reduce mass. The spacecraft is designed for a three year lifetime and shall operate in a lunar radiation environment of 50Krads. It will be placed into a frozen orbit with apoapse over the lunar south pole, although the apoapse location can also be over the north pole. It must be able to operate through the relatively short (~1 hour) lunar eclipses and survive the longer duration (up to 6 hour) Earth eclipses.

Antenna Placement and Pointing Strategy

The lunar relay orbit has been described in the earlier. Let us discuss next the antenna placement and the pointing strategy so that we can minimize the number of antenna, maximize the coverage and facilitate mission operations. We start with the description of the LRS orientation and the placement of its antenna and solar panels. While strolling along its trajectory, the LRS will constantly pitch to keep its nadir deck facing towards the Moon. Its lunar antenna is mounted on the nadir deck and it can be gimbaled 90 degrees from the nadir direction. With its dual-axial gimbal capability, the lunar antenna can cover everything underneath it. The two single-axial gimbal solar panels are attached to the LRS side wings. LRS will yaw continuously and laterally towards the Sun. Once aligned with the Sun, the solar panels will be swinging either up or down to expose the solar cells directly to the Sun. The direct to Earth (DTE) antenna is placed in the front deck of the LRS. Its dual-axial gimbal antenna allows it to cover practically the entire front hemisphere.

LRS needs to perform the 180-degree ballerina yaw twice per orbit. Each time, the solar panels need to readjust themselves to render to the Sun. A dynamic profile of the Earth position off boresight of the front deck in degrees during an edge-on period is displayed in Figure 10. Notice that when the off-boresight angles of Earth are beyond 90°, Earth is in the rear deck. During this edge-on period, several ballerina yaws are needed. To reduce the operation complexity hence risk, the number of yaws can be limited to one every two weeks. Thus, there will be some communication loss, which could happen when the LRS is relaying information to Earth and could last up to several hours. To minimize such communication loss, the DTE antenna can be designed mechanically to cover beyond the horizon of the DTE antenna's deck. A recent design at GRC shows that the antenna can be

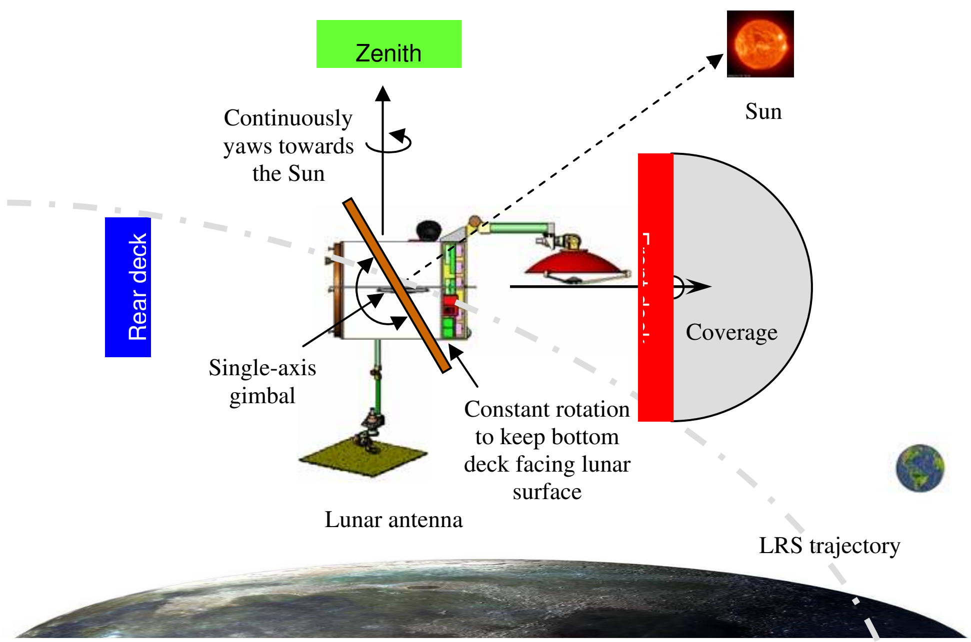

Figure 10: Lunar Relay Satellite Orientation and Pointing Strategy

Each lunar cycle, Earth splits its time between the front and the rear decks and most of the time, Earth remains on one deck the whole satellite orbit (12hrs). Only during the transitional (edge-on) periods, when the orbiting plane is edge on with Earth (every ~2 weeks) or with the Sun (every ~6 lunar cycles), Earth disappears and reappears from the front deck each orbit. To maintain communication connectivity with Earth using a single antenna during such events, the

extended at least 20 by degrees. It should be pointed out from Figure 11 that, with a 20-degree extension, Earth is in the rear deck up to the end of the fifth day. By yawing once on the sixth day, no communication loss will result. Larger extensions can be designed to ensure overall minimal communication loss; however, a loss in pointing accuracy could result as the antenna is pushed further out from the LRS's center of mass. In short, our design, orientation configuration, and pointing strategy will allow the LRS to cover well the Moon, the Earth and the Sun with minimal required mechanism and communication loss.

- Mass: 5kg

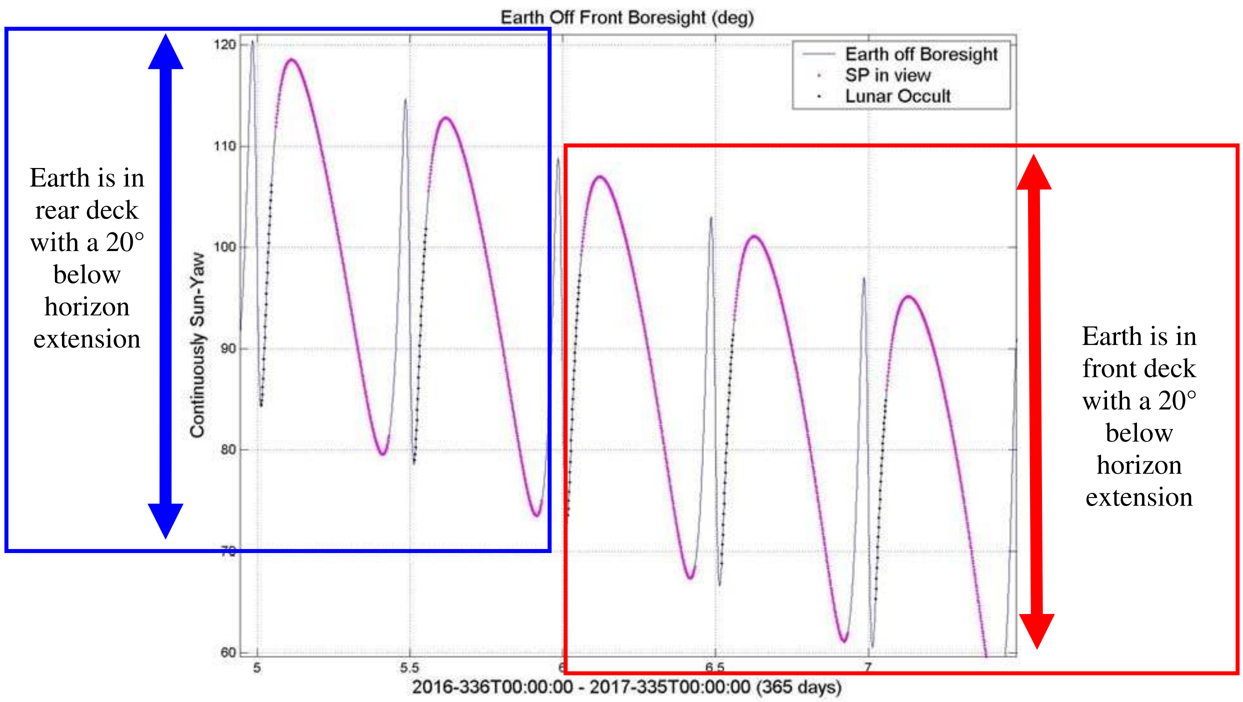

Figure 11: A zoom-in view of the Earth off front boresight angles during an edge-on period

Attitude Control System

A preliminary cost assessment of ACS slew maneuvers was performed. Specifically, the goal is to provide the cost (in terms of fuel use and time) required to slew and point for data transmissions, and then to slew back to a nominal attitude. The analysis consists of two parts: slew-time estimation and momentum management analysis.

Estimation of Slew-Time Capability

Here we assume that the (4-wheel) reaction wheel assembly (RWA) meets the following minimum requirements:

- Maximum angular momentum: 6 N-m-n at 6000RPM
- Maximum toque capability: 0.075 N-m
- Peak power: 80W

These requirements can be met by space qualified commercially available RWAs. For margin purposes we will neglect the increase in torque capability due to the 4th wheel. Also we assume that the slew torque limit is set at 0.05 N-m.

The IMU is chosen has a mass of less than 1kg and a dynamic range of 1000deg/sec. As a result, the gyro rate limits will far exceed those expected from slew rates resulting from attitude profiling.

The mass of the spacecraft (see Table 5) for the purposes of this analysis is 435.8 kg The principal moments of inertia about the x (roll), y (pitch), and z (yaw) axes are given by

- Ixx = 186 $kg - m^2$
- Iyy = 239 $kg - m^2$
- Izz = 305 $kg - m^2$

In order to produce a lower bound on the slew time, a minimum time ("bang-bang") slew profile about each axis is assumed. Note that smooth slew profiles will result in longer times. Also, we have neglected cross-axis coupling of torques in this analysis.

The slew time estimates are given in the tables below for slews of 90 degrees and 180 degrees about each principal axis.

	Inertia ($kg-m^2$)	Max Torque (Nm)	Max Rate (deg/s)	Slew Time (s)
X-Axis (Roll)	186	0.05	1.177	153
Y-Axis (Pitch)	239	0.05	1.039	173
Z-Axis (Yaw)	305	0.05	0.919	196

Table 3: 90 Degree Slew

	Inertia ($kg-m^2$)	Max Torque (Nm)	Max Rate (deg/s)	Slew Time (s)
X-Axis (Roll)	186	0.05	1.665	216
Y-Axis (Pitch)	239	0.05	1.469	245
Z-Axis (Yaw)	305	0.05	1.300	277

Table 4: 180 Degree Slew

The maximum rate limit is set at 2 deg/sec per axis (typical ACS bound) and this bound is not violated for the minimum time slew profile

<u>Disturbance Torques and Momentum Unloading Analysis</u>

The primary disturbance torques acting on the orbiter are gravity gradient and solar radiation pressure. The worst case gravity gradient torque acting on the spacecraft was estimated as 3e-6 N-m. The worst case solar radiation torque (assuming a cross section spacecraft area of 5 m^2 and a center-of-mass to center-of-pressure offset of 0.25m) was estimated as 5e-5 N-m. The frequency of momentum unloading given this disturbance environment is approximately once per week. Over the lifetime of the mission approximately 5 kg of fuel is required for momentum unloading.

Mechanical Configuration

The structure uses aluminum alloy honeycomb for the decks and panels. Aluminum alloy is used for the thrust tube. The use of an Electric Propulsion System (EPS), discussed below, allows the structure to be much smaller since the large central tank of a chemical propulsion system is not required. The additional mass for the solar array & battery combined with the electrical propulsion system components is less than the mass of the chemical propellant systems and propellant. This is summarized in the table below.

Table 5: LRS Mass Summary[a]

	Electrical Propulsion	Chemical Propulsion
Subsystem	Mass with Contingency	Mass with Contingency
Payload	44	44
Structure	45.44	45.44
Avionics	17.68	17.68
ACS	30.78	30.78
Propulsion	99.74	44.21
Power	98.48	72.38
Fuel	53	181.3
Total	389.1	435.8

[a] The mass estimate is based on studies conducted at Goddard Space Flight Center's Integrated Mission Design Center

The mass, with contingency of 30% in most places, less in some heritage components, fits within the Minotaur V capabilities.

The volume within the Minotaur V fairing is not a limitation to the mechanical design. As Figure 12 shows[9,10], the spacecraft fits easily.

The overall mechanical configuration, along with the placement and orientation of the antennas and solar arrays in the deployed configuration, can be seen in Figures 13 and 14.

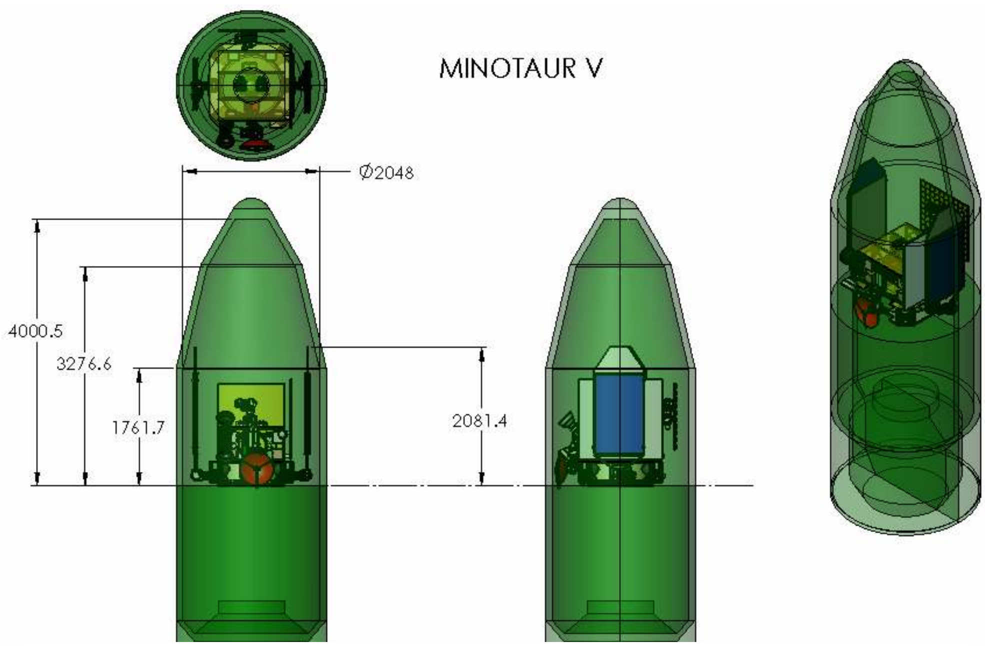

Figure 12: LRS in Minotaur V Fairing

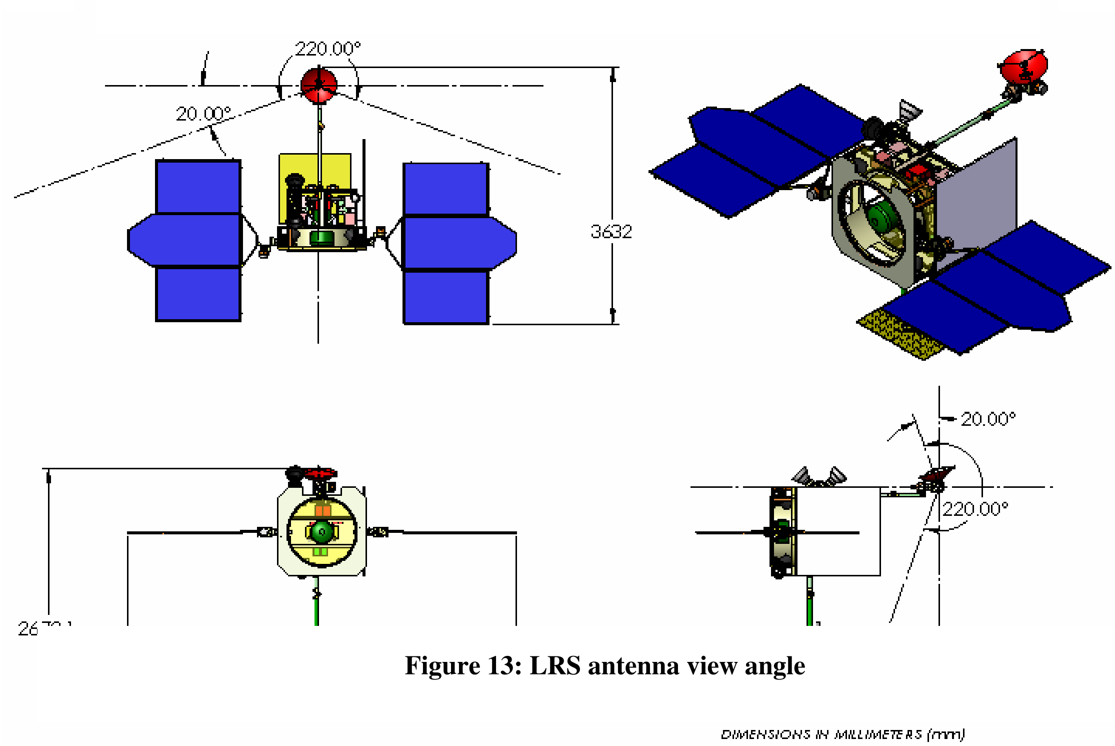

Figure 13: LRS antenna view angle

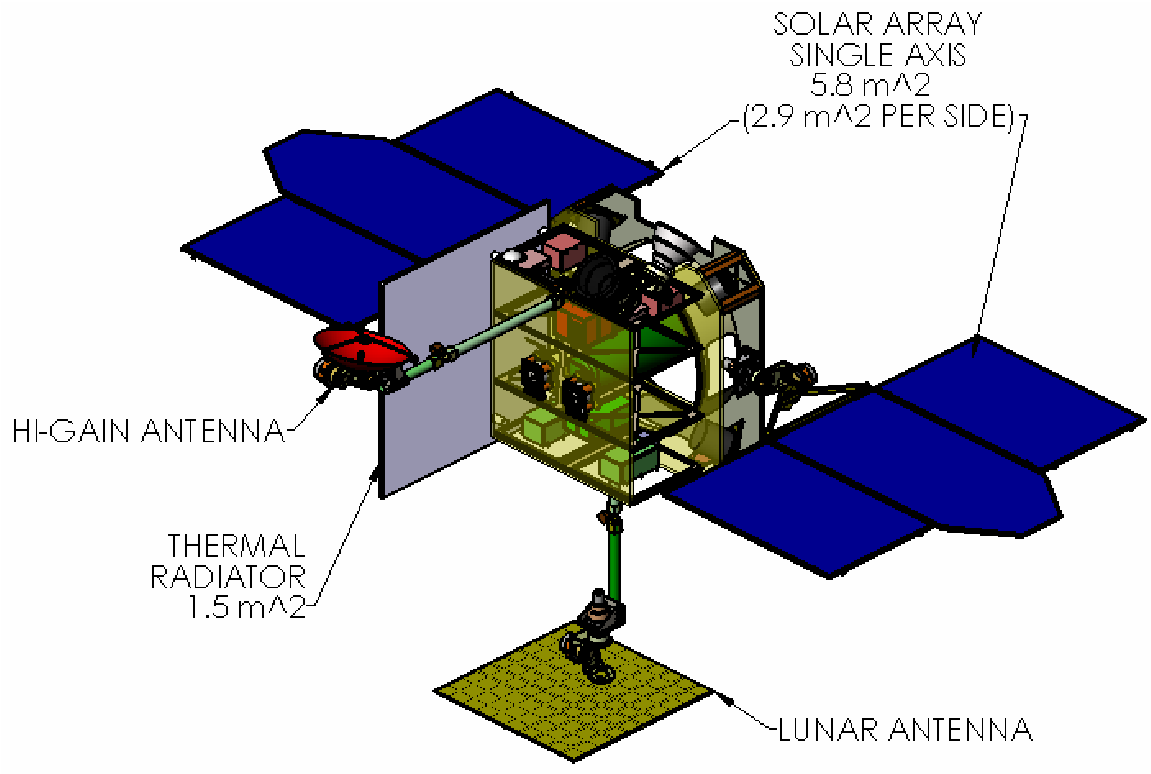

Figure 14: LRS Spacecraft

The Earth antenna is on a two axis gimbaling mechanism that can point throughout a hemisphere plus 20 degrees. The solar arrays are on one axis gimbals. The major components are labeled on the following figure.

Power System

The power system is designed to be a conventional photovoltaic array with a battery module to provide power during the solar eclipse periods. The bus voltage is specified to be 28Vdc

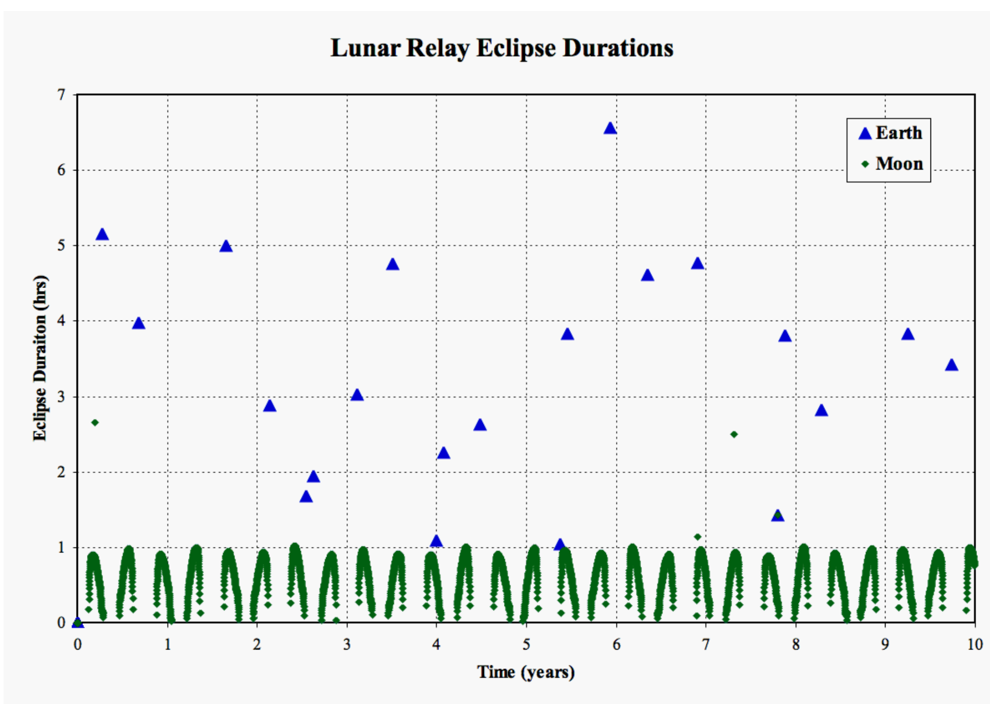

Figure 15: Lunar Relay Eclipse Durations

and it is assumed to be an unregulated bus to reduce losses and overall system mass. The estimated required life of the power system is three years. While this may have some effect on the photovoltaic arrays mainly due to radiation losses, the battery life is primarily governed by number of cycles. For a longer seven year orbit and mission, an estimate of approximately 3000 short-duration (< 1 hr) cycles due to lunar occultation and 21 long-duration (< 6.5 hr including penumbral periods) cycles due to earth occultation. Figure 15 shows this over a 10-year period within which the LRS could launch.

The photovoltaic cell is assumed to be a 4x6 cm triple junction GaAs cell with a cell efficiency of 28%. The cells are connected such that approximately 32 Vdc is produced from the array to account for losses downstream. At the panel level, including a honeycomb core, composite face sheets, and covers, the mass per area is calculated at 2.75 kg/m2. The required structure with each array wing is assumed to also be a function of array area and is specified at 0.275 kg/m2. The array gimbal drive assembly consists of a 6.5 kg unit with an additional 1 kg for electronics and 1 kg for the drive motor. A dual-axis drive assembly is specified to ease pointing flexibility and accuracy due to the antenna requirements. An offpoint angle of 15 degrees is added on as a small margin to also ease pointing operations. No shadowing is assumed due to the other margins and assumptions. The arrays are sized by a non-eclipse peak power load requirement. Before LOI, there may be some significant power requirements due to propulsion needs as well as spacecraft testing. However, the battery system will be available as well, and full view of the sun is expected. An important trade for this array system is whether to assume one or two wings (where the two wing approach would have the same total area). The one wing approach saves mass due to less structure and one less SADA, but two wings provide balance and some fault tolerance. Two wings also reduce available spacecraft area for antennae or other external components.

The battery cell is assumed to be a lithium ion type cell with a specific energy at the cell level of 170 Whr/kg. At the module level, combined with efficiency losses, degradation, packaging, and structure, the specific energy is near 70 Whr/kg. The maximum depth-of-discharge is specified to be 80% which is acceptable due to the relatively low number of cycles expected. In addition, this depth-of-discharge is rarely seen due to the low frequency of long eclipse periods. The battery is primarily sized for the energy required for the longest eclipse. Considerable effort was made to reduce the power load during the eclipse period. The resulting approach is that a temporary safe-hold power level is assumed during the 6.5 hr eclipse where only the absolute required power loads are allowed without any communication requirements. This reduces the battery mass significantly and is sensible because this event is seen once throughout the ten-year period shown above. The next longest eclipse, seen in Figure 15 is about 5 hr, nearly 25% shorter. For each shorter eclipse, more energy will be available, enabling more complete operation of the spacecraft.

Electric Propulsion System

The propulsion system is made up of an electric propulsion component for primary propulsion and a small hydrazine system for tip-off, wheel dumping and small maneuvers.

Several electric propulsion systems are available or in development in the 500 W to 1500 W power range that is needed to perform the LRS-1 delivery mission. They include those based on the flight qualified Russian SPT-70 and SPT-100 Hall thrusters as well as higher performing units currently under development by both NASA Glenn Research Center and the Busek Company. The NASA thruster development effort, through the High Voltage Hall Accelerator (HIVHAC) task, has developed a thruster capable of specific impulses up to 2800 seconds while Busek has available a 600 W thruster that is derivative of their 200 W thruster successfully demonstrated on the Air Force TACSAT 2 spacecraft.

Both Russian thrusters, the SPT-70 and SPT-100, have suitable performance for this mission. At 660 W and 1500 W input powers respectively, the SPT-70 and SPT-100 have been successfully flown by the Russians since the 1980's. Additionally the SPT-100 was recently put into service by Loral on their 1300 series of geostationary spacecraft. Loral has developed a power processor and feed system for vacuum rated spacecraft to operate the SPT-100. These systems could also be adapted to the SPT-70. For the purposes of this study the 1450 second Isp, 40 mN thrust, SPT-70 was baselined using the Loral ancillary systems. Assumptions for the SPT-70 systems are shown in Table 6. (reference News from Moscow No. 26/2000)

Table 6

EP System

Throughput Limit 35.0 kg

SPT-70	unit mass	qty	total mass
Thruster	2.0 kg	2	4.0 kg
PPU	7.0 kg	2	14.0 kg
DCIU	0.0 kg	0	0.0 kg
Gimbal	1.0 kg	14	14.0 kg
Gimbal Drive	0.5 kg	14	7.0 kg
Feed System Fixed	1.8 kg	1	1.8 kg
Feed System per thruster	1.0 kg	14	14.0 kg
		total:	54.8 kg

Xenon Tank	7.0 kg
Xenon Prop	40.0 kg

Some higher thrust maneuvers are required by the spacecraft which cannot be provided by the electric propulsion system. Several options exist to provide these maneuvers including a cold gas Xenon system (by common use of the electric propulsion xenon tank) or an off-the-shelf hydrazine system. While the cold gas xenon system would save money by eliminating an additional propellant system, further knowledge of the required maneuvers were needed to determine if this approach was viable so a small hydrazine system was baselined. The system assumptions are shown in Table 7.

Table 7

ACS System

N2H4 Prop 13 kg n2h4 cap

Monoprop syst	#	Mass (each)	Total
blow-down tar	1	2.5 kg	2.5 kg
PT	4	0.3 kg	1.1 kg
latch valve	3	0.8 kg	2.3 kg
Fill and Drain	2	0.2 kg	0.5 kg
filter	1	0.3 kg	0.3 kg
1 lb thrusters	8	0.3 kg	2.6 kg
Tubing	1	2.0 kg	2.0 kg
			11.3 kg

CONCLUSION

The return to the moon will require communication and navigation solutions that meet both the technical requirements and the cost constraints of the Constellation Program. The addition of small lunar relay satellite concepts like this one to the available options provides technically capable but less costly alternatives to help meet the technical, schedule and cost objectives of the overall program.

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