DUST Final Report

SPACE 582

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1. Introduction

1.1 Mission Objective

The DUST program seeks to create a distributed CubeSat platform that provides a flexible, modular, and cost-effective communications pathway for ongoing and future space missions. The distributed platform is comprised of multiple communications CubeSats connected in a mesh network. In particular, the DUST-Lunar program seeks to support Lunar Gateway by facilitating communication among Lunar ground assets as well as communication between Lunar ground assets and Lunar Gateway. DUST-Lunar relay CubeSats will fill some of the communication gaps from Lunar Gateway to assets on the Moon. Through its valuable assistance to interplanetary communications, the DUST program will showcase the benefits of mesh architectures and inspire a new generation of mesh technology.

1.2 Program Overview

1.2.1 History and Objective

The DUST program began in 2017 as a student multidisciplinary design program sponsored by JPL. The purpose of the program was to develop a mesh CubeSat system around the Earth. The student team finalized some designs, and a prototype is currently in the process of being fabricated. In 2019, JPL expanded the project to include a Lunar mesh system design. A team working on a Pre-Phase A concept study began exploring this option.

The overall program objective for DUST is to investigate swarm mesh networking technology. All work done by the LEO and Lunar teams follows this goal of exploring and expanding upon the possibilities of mesh networking. Each team also has a separate objective based on the goal of the project. The LEO team is tasked with performing ground tests of mesh communications and establishing a demonstration of CubeSats in LEO. The Lunar team is tasked with developing a mission concept for interplanetary mesh networking.

1.2.2 Stakeholders

The overall DUST project is a joint program between JPL and the University of Michigan. Therefore, JPL and Michigan are the two main stakeholders because both parties have time and money directly invested in the program.

1.2.3 Organization

The DUST team is led by three investigators: James Smith, Darren McKague, and Jose Velazco. James Smith and Jose Velazco are members of staff at JPL, and Darren McKague is a professor at Michigan in the Climate and Space Sciences department. The remainder of the organization is broken up into two distinct teams: DUST-LEO and DUST-Lunar. The LEO team is made up of undergraduate and graduate students from Michigan with Taylor Sun leading as the Chief Program Director. This team works on designing and fabricating a CubeSat mesh technology demonstration for low Earth orbit (LEO). The smaller Lunar team is made up of five students from the SPACE 582 class. This team works on a Pre-Phase A concept for mesh communication CubeSats around the Moon.

1.3 Key Technology Question

1.3.1 Definition of Mesh

Mesh networking is a type of network topology in which all devices act as decentralized nodes. Data is freely transferred between nodes, and no single node inherently has more responsibility or value than the others. As opposed to a centralized network where data always routes through a master node, a mesh network will route data through whichever path of nodes is determined to yield the lowest latency. For satellite communication, a mesh network is also known as a distributed satellite system (DSS) and has several theoretical benefits. The first benefit is decentralization of resources. By using a network with smaller, simpler satellites, the system gains the benefits of modularization. The next benefit is spatial distribution which refers to the fact that multiple satellites will ideally have different viewing conditions of the target area. Another benefit is an increased number of satellites. With more satellites tasked to the same function, the division of work will decrease the complexity and processing power required of each satellite. The last benefit is redundancy. Without a centralized node, every node has equal importance and the network can ideally still function in the absence of a node. This minimizes the overall number of single point failures in the system.

2. Mission Definition

2.1 Top Level Requirements

From the DUST-Lunar defined concept, top level requirements were defined for the overall mission. The DUST-Lunar team worked with JPL to define the necessary requirements given in Table 2.1.1. Only a single primary requirement was chosen to ensure the mission was solely focused on providing distributed communications for future Lunar ground assets. Several secondary mission requirements were also listed with a focus on expanding the possible applications for mesh networking. In the future, this network would create a platform for Lunar sensing constellations, serve as a test bed for new technologies, and set a standardized communications framework.

Table 2.1.1. List of top level mission requirements

P-1	Deploy a mesh communication network around the Moon to facilitate communication between lunar ground assets and earth
S-1	Create platform for future distributed Lunar sensing satellite constellation
S-2	Serve as a test bed for new components and technology
S-3	Develop standardized communications framework for future mission applications

2.2 Mission Constraints

As with any mission, constraints are unavoidable and must be carefully planned and accounted for. For DUST-Lunar, mission constraints can be broken up into 6 categories as shown in Table 2.2.1.

Table 2.2.1. List of mission constraints

Constraint		**Description**
C-1	Cost	SMEX - $120 million maximum
C-2	System	Homogeneous system of nodes
C-3	Regulations	FCC, COSPAR, FAA, NASA
C-4	Environment	Lunar space environment (radiation)
C-5	Interfaces	Interoperable through network of available ground and orbital assets
C-6	Structural	Artemis and Lunar Gateway

The first constraint is cost. This mission will be targeting a small explorer class (SMEX) budget. This choice is because SMEX missions have the lowest cost while still providing a very workable amount of money ($120 million). The relatively low cost of SMEX missions will increase the marketability of DUST-Lunar to programs such as NASA. This also means that the cost per satellite as well as number of satellites must be carefully monitored to ensure the mission is kept on budget. The second constraint is the system which means that the CubeSat network must be made up of a homogeneous system of nodes. For ease of operation and practicality sake, all relay nodes must be identical in the network. This constraint also is connected to the mesh concept with regard to ensuring the relay network has no master node. With all nodes designed and fabricated identically, no single node will have reason to dominate the others, so data can be shared freely in network as intended. The third constraint is regulations. Several government organizations and committees have established rules for space operation, and all rules must be obeyed. This includes concerns such as radio transmission frequency and power. The fourth constraint is environment. From the mission definition, the CubeSats will orbit the Moon and, therefore, must be capable of surviving in the Lunar space environment. This includes concerns such as satellite radiation tolerance. The fifth constraint is interfaces. The communications protocols used by the mesh relay network must conform to standards preset by existing hardware on the surface or orbiting the Moon. This ultimately ensures that the mesh network can assist ground assets as intended. The sixth constraint is structure. An Artemis SLS is planned to deliver the CubeSats to Lunar Gateway, and Gateway is planned to launch the CubeSats into orbit around the Moon. The DUST-Lunar CubeSats must then meet the payload and CubeSat structural requirements provided by Artemis and Gateway to match the planned ConOps.

2.3 Axiomatic Design Process

The design process begins with defining functional requirements, and requirements have already been set for DUST-Lunar. From these functional requirements, design parameters are chosen following the independence axiom. The first fundamental design axiom, also known as the independence axiom, states that all functional requirements should be kept independent of each other. A design matrix is used to track independence by mapping design parameters to functional requirements. Dependencies between functional requirements can be read from the matrix. A diagonal design matrix demonstrates an ideal, uncoupled design where each functional requirement maps to only a single design parameter. As shown in Table 2.3.1 and Fig. 2.3.1, DUST-Lunar design parameters were chosen to create an uncoupled design which follows the design axioms.

Table 2.3.1. List of functional requirements and chosen design parameters

Functional Requirements		**Design Parameters**	
FR-1	Implement internode and ground station comms in a mesh	DP-1	Communications Payload
FR-2	Mesh network remains functional in single-node failure	DP-2	No. of CubeSats/nodes
FR-3	Complete coverage of polar regions	DP-3	Orbit Inclination/Altitude
FR-4	Compatible with future additional nodes	DP-4	Software Architecture

$$\begin{bmatrix} \text{FR-1} \\ \text{FR-2} \\ \text{FR-3} \\ \text{FR-4} \end{bmatrix} = \begin{bmatrix} 1 & 0 & 0 & 0 \\ 0 & 1 & 0 & 0 \\ 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 1 \end{bmatrix} \begin{bmatrix} \text{DP-1} \\ \text{DP-2} \\ \text{DP-3} \\ \text{DP-4} \end{bmatrix}$$

Figure 2.3.1 Uncoupled design matrix of functional requirements & design parameters

2.4 Derived Functional Requirements

Functional requirements were derived directly from the top level mission requirements. Overall mission objectives taken from the top level requirements were translated into necessary system functions. Table 2.4.1 lists all of the derived functional requirements along with the top level requirement each was derived from.

Table 2.4.1 List of functional requirements

FR-1	System shall implement in-network and out-of-network communications	P-1
	FR-1.1: A CubeSat shall be in contact with at least 2 other CubeSats in the mesh	
	FR-1.2: A CubeSat shall have the ability to relay data from/to a ground station OR Lunar Gateway	
FR-2	Mesh network communication shall remain functional in the event of single node failure	P-1
FR-3	Mesh network shall ensure complete coverage of the Lunar polar regions	P-1
FR-4	Mesh network shall be operational with additional CubeSats added to mesh network in the future	P-1

FR-1 and FR-2 were noted as critical requirements for the mission. FR-1 is critical because the mission is completely based on communications. A mesh network of satellites without relay capability is no longer a mesh, so communications are perhaps the highest priority in the mission. To distinguish between in-network and out-of-network communications, FR-1 was split into two sub-requirements. FR-1.1 defines in-network communications to be that CubeSats shall be in contact with at least two other CubeSats. DUST-Lunar has set the minimum number of contacting nodes to be two to preserve the mesh. When a node can contact only one other node, then data can only travel along one path to and from that node. This node is no longer of use to the mesh because data sent to this node is stuck at a dead end, so maintaining contact with two nodes is necessary for mesh networking. FR-1.2 defines out-of-network communication to be that CubeSats shall have the ability to relay data between ground stations as well as Lunar Gateway.

The overall purpose of any communications network is to pass data from a starting location to a desired final location. For DUST-Lunar, data is meant to travel between Lunar ground assets and Lunar Gateway, so the mesh must be capable of facilitating communications between these points. FR-2 is also critical because of required redundancy. Mesh communication is the mission's only primary top level requirement, and, as explained with FR-1.1, the mesh is maintained when nodes are in contact with at least two other nodes. Anomalies can cause a CubeSat to fail, and any single failure should not cause the entire network to lose value. To guard against this, the redundancy must be added meaning that a single node failure shall not interrupt the operation of the mesh.

2.5 Concept of Operations

The mission is planned to begin in 2024. Lunar Gateway is essential to this mission with DUST CubeSats relaying data to Gateway. By 2024, Lunar Gateway should already be established and orbiting the Moon, and the Artemis program will begin sending astronauts and equipment to Gateway. This is the ideal time to begin the DUST mission because DUST CubeSats can then be launched with an Artemis SLS. Artemis shuttles will already be traveling to Lunar Gateway, so launching with Artemis is a logical choice. Once the Artemis shuttle arrives at Lunar Gateway, DUST CubeSats can be unloaded and kept on Gateway. The ideal CubeSat launch points can be determined from Gateway's orbit, and the DUST CubeSats will then be launched from Gateway at the designated points. After the CubeSats are launched from Lunar Gateway, the orbital insertion thrusters will set the correct orbit for each of the CubeSats. CubeSats will begin normal relay operation once set in orbit, and will ideally continue operating for a planned lifetime of 3 years. There will be no official deorbit, so CubeSats will operate until part failure or until crashing into the surface of the Moon. A summary of the ConOps is shown in Table 2.5.1.

Table 2.5.1 Mission ConOps

Mission Phase	Description
1	Launch from Earth on Artemis SLS
2	SLS docks with Lunar Gateway
3	CubeSats are released from Gateway around the Moon
4	CubeSats begin Lunar orbital injection
5	CubeSats acquire initial navigation state
6	Start of mission ops, operate until failure

2.6 Work Breakdown Structure

The DUST-Lunar team is comprised of five students from the SPACE 582 class. Each member was assigned a different subsystem to research and conduct trade studies as shown in Table 2.6.1. All team members worked together to draft documentation such as reports and presentations.

Table 2.6.1. Work breakdown structure for DUST-Lunar team

Team Member	Department	Responsibilities
Kelsen Case	Electrical Power Systems	Develop Power Budget, Solar Array Sizing, EPS Systems Integration
Tim Kiyabu	Financial Operations	Cost Estimates, Mission Budget, Trades
Koray Kachar	Attitude Control and Determination System	AD&CS Component Selection, Trade Studies
Michael Burton	Payload & Communications	Link Budgets, Radio & Antenna Selection
Prit Chovatiya	Concept of Operations	Launch Vehicle Research. Orbit Selection, Trade Studies

2.7 Mission Timeline / phases

The DUST-Lunar team defined phases of the mission and development. The key decision authorities include NASA for launch operations and ground station operations and DUST-Lunar for on-orbit operations. Since this program proposal is in support of a NASA Announcement of Opportunity, we refer to the NASA Space Flight Program and Project Management Requirements and heritage NASA programs for guidance in the development of the timeline. Common across many NASA programs are Key Decision Points which are defined as “the event where the Decision Authority determines the readiness of a program of a project to progress to the next phase of the life cycle. Transition to the following phase occurs immediately following KDP approval except for transition from Phase D to E where transition occurs following on Orbit checkout and initial operations”. For this program, the Decision Authority is NASA. Table 2.7.1 outlines the timeline of development and is derived from Landsat 8.

Table 2.7.1 Program timeline for DUST-Lunar

Program Timeline for One Satellite		
Date	**Phase**	**Description**
February, 2025	IOP	Initial Operation Phase
December, 2024	Mission Science team meeting	Prior to and during launch, meeting of science and technical leads
December, 2024	Spacecraft launch	
October, 2024	LRR	Launch readiness review
September, 2024	Mission dress rehearsal	
September, 2024	FRR	Flight readiness review
June, 2024	KDP-E	Key decision point following successful system assembly, integration, and test
April, 2024	Transport to launch pad	
January, 2024	Fairing encapsulation	
December, 2023	SMSR	Safety and Mission Success Review

December, 2022 – June, 2023	KDP-D: CDR, PRR	Critical Design Review, Production Readiness Review
December, 2021	KDP-C: PDR	Preliminary Design Review
December, 2020	KDP-B: SRR, MDR, SDR	System Requirements Review, Mission Definition Review, System Definition Review
June, 2020	KDP-A: MCR	Mission Concept Review

The mission architecture is broken up into ground operations, launch operations, and on-orbit command operations. As noted earlier, NASA is the Decision Authority for the development of the vehicle and all critical operations decisions. The completion of the mission is contingent on multiple satellites entering orbit, which elevates risk of failure before mission operations even begin.

3. System Drivers

As mesh networking is heavily dependent on the number of satellites and the inter-nodal distance between them, it is critical to study various constellation configurations to understand the impact of these choices on the sub-system level design and further, the overall system design. Two major areas were studied, namely: constellation orbital parameters and the communication system, in detail as a result of these choices and the trade-offs between them.

3.1 Orbital Design Choices & Trades

One of the most crucial components of the overall system design stemmed from the orbital design since it significantly impacted various other subsystems. Considerations like orbit altitude, inclination, number of satellites in the constellation, and orbital arrangement played crucial roles in meeting our functional requirements and primary mission objectives. Hence, it was important for the DUST team to conduct trades and provide a robust basis for the design choices

3.1.1 Design Parameters

As mentioned above, four key design parameters were determined to be the main drivers in the orbit design for the system.

The DUST-Lunar team places emphasis on future planned manned and unmanned lunar exploration interests of NASA as a part of the Artemis Program and therefore, aims at providing its services to the aforementioned stakeholders. Heightened interest in the southern polar region (80ºS to 90ºS) of the Moon due to its long-daylight periods and presence of ice in the shadowed regions provide very favorable conditions for a lunar outpost, which is a viable starting design point for lunar surface coverage. Assisting assets for the Artemis program like Lunar Gateway are already designed to provide communications to the regions but can only do so for 86% of the area which can be seen from the image below. [3]

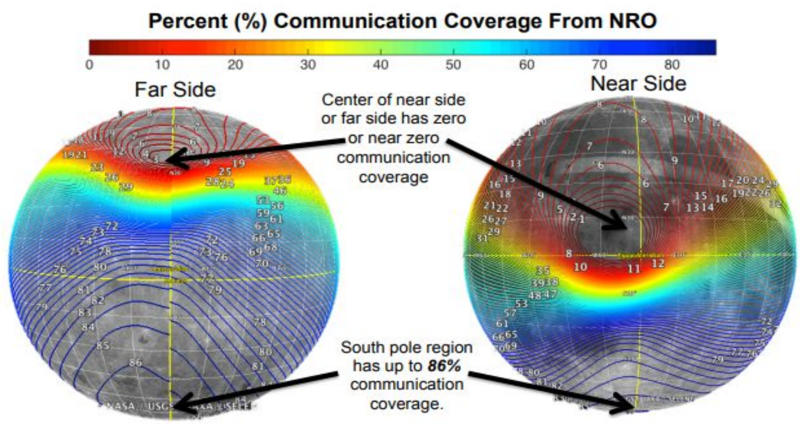

Figure 7. **NRO Communication to Surface Coverage**

Figure 3.1.1.1. Lunar communications coverage of NRO [3]

The near-rectilinear orbit for the Lunar Gateway places it in a highly eccentric orbit with an orbital period of 7 days which includes “blackout periods” for south polar regions when Gateway is eclipsed by the Moon while passing over the lunar north pole. This orbit is also at a significantly farther distance (apocynthion of 70000 km and pericynthion of 3000 km) from the lunar surface which leads to higher latency and round-trip communication distances [4]. The team sees these factors as an opportunity for the DUST architecture to play a vital role in filling these coverage gaps and complimenting Gateway’s role at a considerably lesser cost. As human exploration increases in these regions, it will be critical for faster lunar ground-to-ground and near-constant communications with Earth, which has been an important factor for our design choices for the orbital inclinations and south polar coverage.

Our functional requirement #1 stems from the mesh networking to be operational given that a node/satellite is in communication with at least 2 other nodes at the same time. This requirement directly affects the number of satellites in orbit and constellation design.

Heritage missions from the Apollo-era created a new stream of data and analysis for behavior of satellites in the lunar gravitational field. Greater than expected perturbations were observed for satellites in lunar orbits and were explained to be caused by anomalies in mass concentrations throughout the surface of the Moon [5]. These anomalies can be fatal to a satellites lifetime and can lead to catastrophic outcomes over extended periods of time. This further requires a very demanding attitude determination and control system to account for these perturbations for stationkeeping.

Considering these factors, “frozen orbits” discovered by Elipe and Lara in 2003 [6], were considered as primary solution subset for tackling this issue. It was found that inclinations of 27°, 50°, 76°, and 86° were stable orbits which had the minimum effect from these perturbations and were “frozen orbit” ensuring long lifetimes for the satellite.

Simplistic viable constellation design choices were essential for keeping the system costs low and reducing risk failures. One of the key assumptions for the deployment of the constellation was that the DUST satellite mesh would be launched from Lunar Gateway at high inclination south pole orbits as Gateway would be at the closest approach to the lunar surface. This further restricted our choice to 86° inclination orbit which is the closest frozen orbit to the deployment orbit and also provides multiple revisits over the south poles. This also forced our team to take a conservative decision on limiting the number of orbits to 2 orbits at an 86° inclination and a 90°RAAN increment between those orbits to ensure homogenous coverage and multiple equally spaced satellites overpasses for the polar outposts.

These considerations played a key role in shaping our design and providing meaningful design envelopes around which the team could start shaping its initial minimum viable orbital configuration for meeting the mission requirements.

3.1.2 Constellation Analysis

STK was used as a primary tool to get preliminary estimates on the orbital characteristics of various constellations and compare the solution space for a minimum viable design that met all the requirements. The results were analysed and compared based on the 4 key criteria discussed above. Initially a single satellite coverage analysis for the south polar region (defined as the area between the 80-90 deg lat) and the entire lunar surface was conducted to gain an understanding about the speed of coverage and revisit times. The table below considers a satellite at an altitude of 600 km and an inclination of 86°.

Table 3.1.2.1 Orbital coverage of a single satellite.

1 satellite: @ 600 km altitude, 86 degree inclination; epoch starting from 21 November				
Coverage Definition	Percent Coverage	Time Coverage Achieved	Duration Coverage Achieved	Max Gap Duration for 99% Coverage Area
80-90 deg lat	90.98%	5 Dec 2019	14 days approx	5 days
Global	99.86%	5 Dec 2019	14 days approx	13 days

As it can be noted from the table above, a single satellite provides significant coverage for the south poles as well as the entire lunar surface but has considerably higher revisit times. This can be seen from the right most column which shows the maximum revisit time of 5 days for 99% of the south polar regions and 13 days for 99% of the lunar surface. Upon gaining these insights, a 4 and 6 walker satellite constellation was used to recompute the coverages and revisit times. A walker satellite constellation is an orbital arrangement of satellites equally spaced via even RAAN increments and true anomalies phasing between them. This ensures a simplistic design and deployment of the constellation and also provides even coverage and revisit times. For the reference of the reader the arrangement of the satellites can be seen from the figures below.

Figure 3.1.2.1 shows the 4-satellite constellation with 2 satellites per plane with 2 planes at a RAAN increment of 90 deg and true anomaly phasing of 45 deg which ensures equally spaced satellites in orbit. This arrangement results in even coverage and revisit times for locations on the lunar surface.

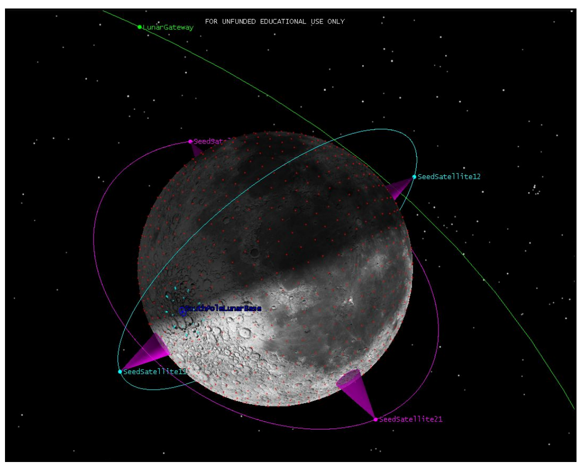

Figure 3.1.2.1 STK Rendering of 4-satellite Constellation

Figure 3.1.2.2 shows the 6-satellite constellation with 3 satellites per plane with 2 planes at a RAAN increment of 90 deg and true anomaly phasing of 30 deg which yet again ensures equally spaced satellites in orbit.

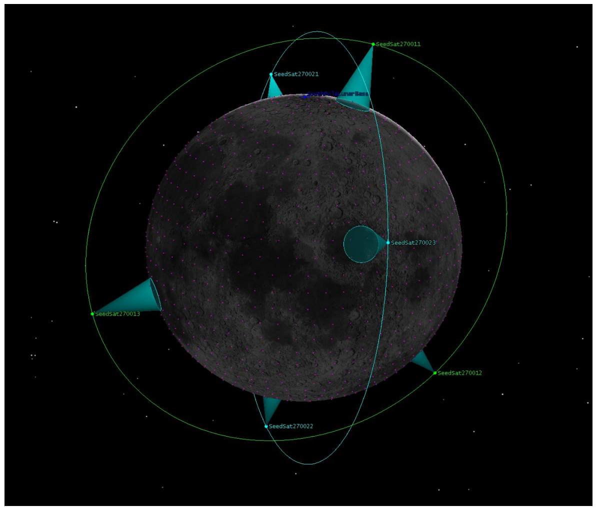

Figure 3.1.2.2 STK Rendering of 6-satellite Constellation

Upon STK simulation analysis, the following results were obtained for the 4 and 6-satellite constellation respectively.

Table 3.1.2.2. Orbital coverage of Four satellites

4-satellite constellation: 2 satellites per plane with 2 planes at a RAAN increment of 90 deg and true anomaly phasing of 45 deg			
Coverage Definition	Percent Coverage	Duration Coverage Achieved	Max Gap Duration for 99% Coverage Area
80-90 deg Lat	90.98%	8 days approx	83 minutes
Global	99.86%	8 days approx	6 days

Table 3.1.2.3. Orbital coverage of Six satellites

6-satellite constellation: 3 satellites per plane with 2 planes at a RAAN increment of 90 deg and true anomaly phasing of 30 deg.			
Coverage Definition	Percent Coverage	Duration Coverage Achieved	Max Gap Duration for 99% Coverage Area
80-90 deg Lat	90.98%	8 days approx	55 minutes
Global	99.86%	8 days approx	5.93 days

From the results, it can be deduced that the constellations offer considerably faster revisit times and speed of coverage. For a 4-satellite constellation the maximum revisit time is reduced to just 83 minutes and further the 6-satellite constellation has a maximum revisit time of just 55 minutes. Although they offer faster coverage, the total coverage is identical to a single satellite which can be attributed to a lower altitude and/or smaller sensor coverage definition.

Although these results certainly assisted the team in understanding the coverage advantages of adding more satellites to the constellation, it did not provide any insights about the inter-node/satellite links and access times for the mesh network to be operational. This was critical to the constellation design as this directly attributed to meeting the Functional Requirement-1 of having a node/satellite be in communication with at least 2 other nodes at the same time to relay the data. In the case of a 4-satellite constellation, there was no inter-nodal access due to two reasons: the altitude of orbit was considerably lower with which the horizon of the moon obstructed the line of sight and the lesser number of satellites neighbouring a node. So the access analysis of a 6-satellite constellation was conducted and the results are posted below. Here, the access of a single node is plotted with the remaining other nodes in the constellation.

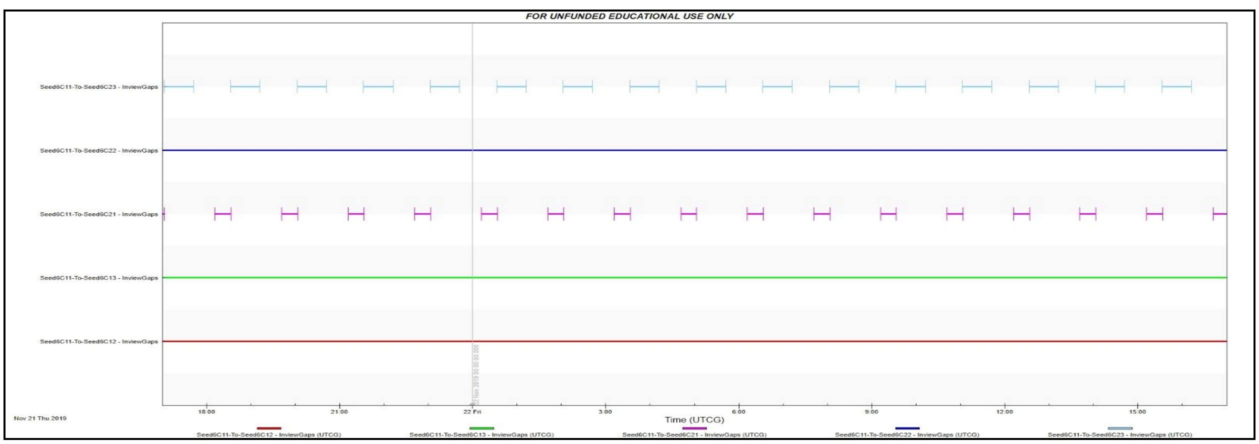

Figure 3.1.2.3 Access gaps for 6-satellite constellation

The solid lines show the access gaps and the empty areas show the accesses. As can be seen clearly, there is no access for 3 satellites and only partial periodic access with the other 2. Adding 2 more satellites to the 4-satellite constellation improves access but still does not meet the functional requirement #1. Upon conducting further analysis, it was found that the horizon effects were still prominent due to a lower altitude and affected the inter-nodal access drastically. A minimum altitude was computed to be about 1800 km for a 6-satellite constellation for the neighbouring nodes to have line of sight access with each other. Even though this gave us a possible design altitude, this exceeded our inter-nodal communication distance design envelope for the link budget. This called for an additional 2 satellites to be added to the constellation design and the results for the access times for an 8-satellite constellation were computed, studied and are shown below.

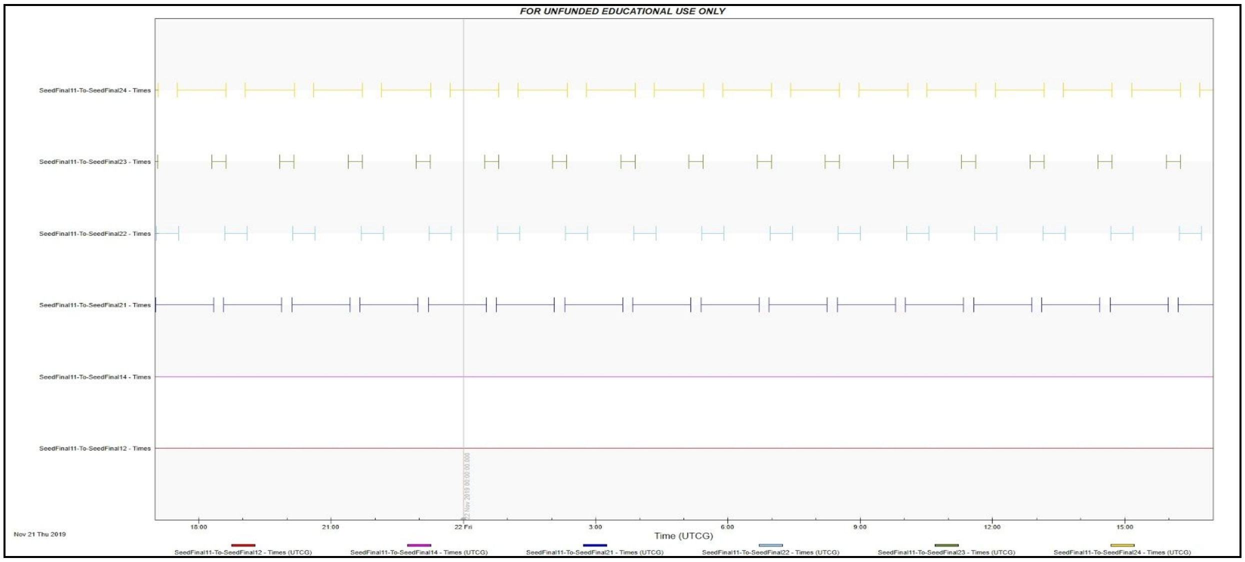

Figure 3.1.2.4 Access times for 8-satellite constellation

Here, the solid lines in the plot shows the access times and the blank areas show the coverage gaps. It can be noted that a satellite in an 8-satellite constellation has continuous coverage for at least two other satellites in orbit and has periodic access of 4 other satellites in orbit, which meets our FR #1 but also provides redundancy in inter-nodal communication links. We can conclude that adding more satellites to the constellation and increasing our orbital altitude overcame the horizon effects and the 8-satellite constellation is a minimum viable design which meets the requirements. The table below summarizes the active inter-nodal connections with the number of satellites in the constellation and their respective revisit times.

Table 3.1.2.4 Constellation Analysis Results Summary

Constellation Type	South Pole Max Revisit Time	Number of Active Mesh Connections
4 Satellites	83 minutes	0
6 Satellites	55 minutes	2 with gaps
8 Satellites	44 minutes	4 at least

3.1.3 Design Decisions

Based on the STK analysis, the functional requirements considerations and simplistic design choices the orbital constellation design can be summarized as follows:

a) Inter-nodal connectivity is maintained at an altitude of 750 km for an 8-satellite constellation with 4 satellites per orbit in two orbits placed at a RAAN increment of 90 degrees and a true anomaly phasing of 30 degrees ensuring evenly spatial arrangement of satellites.
b) The inclination of the orbit will be about 86 degrees to minimize perturbation due to lunar mascons, which further reduces the ADCS requirements for stationkeeping.
c) This constellation will compliment the lunar gateway to ensure that the lunar ground assets have near constant communication coverage and further provide less latency ground-to-ground communications.

A key takeaway which the team chooses to lay emphasis on is that this design is in no way considered to be a final design solution but a minimum viable design which meets the requirements, which should be used further to improvise the constellation design.

4. System Definition

4.1 Operational Requirements

From the constraints and top level requirements, several operational requirements were determined for the mission. These operational requirements specify the operating conditions for the mission and are listed in Table 4.1.1 alongside the constraint or top level requirement that each one was derived from.

Table 4.1.1. List of operational requirements

Operational Requirements		
O1	JPL Shall command DUST system	P1
O2	System shall survive in LLO natural environment	C4
O3	System shall provide means of deorbiting at EOL; System shall provide means of passivating the spacecraft and deorbiting in such a manner as to avoid collision with ground assets at EOL	P1
O4	System shall not have detachable parts or create any space debris during launch or normal mission operations	P1
O5	System shall be operable within regulatory board-allocated band	C3
O6	System shall wait a minimum of 30 minutes after deployment switches are activated to deploy solar panels	P1
O7	System shall wait a minimum of 45 minutes after deployment switches are activated to generate an RF signal	P1
O8	System shall transmit timestamp and relay satellite ID	P1
O9	System shall transmit telemetry data during ground link	P1
O10	System shall withstand 4 to 130 degrees Fahrenheit	C6

4.2 ADCS Requirements

Table 4.2.1 ADCS subsystem requirements

ADCS Requirements		
ADCS-1	Sub-system shall provide pointing accuracy of 0.1 degrees required to complete ground communication link	F1
ADCS-2	Sub-system shall maintain Nadir pointing during nominal operations outside of safe mode	F1
ADCS-3	Sub-system shall provide ground station active pointing (+Z face) during link mode	F1
ADCS-4	Sub-system shall maintain Sun pointing during safe mode	O2
ADCS-5	Sub-system shall supply torque commands required to counteract on-orbit disturbance torques	O2
ADCS-6	Sub-system shall provide internal torque for momentum dumping	ADCS-1
ADCS-7	Sub-system shall provide attitude determination data for CubeSat navigation	O2
ADCS-8	Sub-system propulsion shall pass NASA safety panel	C6
ADCS-9	Sub-system shall not freely spin the spacecraft about the Nadir axis	FR-1.1

4.2.1 System Design

The Attitude Determination & Control Subsystem (AD&Cs) oversees the guidance, control, and navigation determination of the spacecraft and ensures adherence to pre-planned trajectories or adjustments to current trajectories. Design of this subsystem consists of deriving flight conditions from selected orbits, selecting control actuators and sensors, and determining control laws to drive the system to the desired flight state.

The primary inputs into the design of the ADCS subsystem are requirements ADCS-1, ADCS-10, and COM-4, which mandate that the subsystem maintain a lunar nadir-facing orientation while in orbit, maintain pointing attitude with neighboring satellites, and maintain a cross-link distance with neighboring satellites, respectively. Additionally, we consider the Concept of Operations which states that the group of satellites shall need to perform an orbital injection maneuver and acquisition in order to reach the desired station-keeping orbit. Thus, we state that the control modes for the spacecraft are orbital injection, acquisition, and stationkeeping.

The selection of an appropriate control method for the satellite relies on the selection of control actuators and attitude determination sensors as well as the orbit insertion method. Since the spacecraft is performing its own orbital injection into a highly elliptical lunar orbit, smaller stationkeeping control actuators are deemed too weak to complete this task. With these two ADCS objectives in mind, the ADCS subsystem is designed to house a large impulse actuator for orbital injection, smaller actuators for stationkeeping, and a single set of navigation sensors shared between the two control groups. For reference we define stationkeeping as maintaining the lunar nadir-facing and cross-link pointing orientation and attitude state over the course of the lifetime of the mission and not including the injection phase. Stationkeeping in this way is a necessary requirement for maintaining the mesh network connections between satellite nodes.

For orbital injection and initial acquisition, we acknowledge that there exist many state-of-the-art and heritage control actuators for performing high-impulse maneuvers required for this stage of the mission. Given the size of the system, the power requirements, and risk to the system, the mission, and the overall Artemis mission and associated vehicles, we limit the set of possible control actuators for orbital injection to those shown in Table 4.2.1.1. This set was derived considering three system parameters that most factor into the design of the ADCS - power, mass, and volume. An additional consideration was risk presented as a result of low technological readiness outlined by NASA. Electric propulsion systems best satisfy mass and volume constraints, but fall short given their power requirements and lack of flight history. If an electric thruster were to be used, it was determined that every other subsystem would have to divert power to the thrusters to perform injection, which poses technical and mission risk. Cold gas and chemical thruster systems require less power and have been successfully proven out for similar sized spacecraft, however they pose a risk in terms of propellant storage. Chemical thrusters can mitigate some of this risk by using so-called “green” propellants which are safer to humans and more inert than other chemical propellants. Therefore, in support of this initial study, a single thrust-vectored chemical thruster is chosen as a viable propulsion system to perform orbital injection.

Table 4.2.1.1 Viable High-Impulse Thrusters for Lunar Orbital Injection

	Power	Mass	Volume	Risk	Longevity
Chemical	Low	Low	Low	Propellant storage	Injection phase
Cold Gas	Low	Low	High	Propellant storage	Injection phase
Electric *	High	Low	Low	Lack of flight history	Entire mission lifetime

For stationkeeping, we begin with the pointing and attitude requirements ADCS-1 and ADCS-10 necessitating that the system orient nadir-facing and maintain a pointing accuracy of 0.1° to the ground. Using SMAD Table 11-8, we narrow down the selection of these control actuators to those comprising 3-axis control or momentum-bias, since the desired accuracy is unattainable with passive magnetic or gravity gradient control approaches, given that the lunar environment is void of a magnetic field and the gravity gradient sensors are too coarse in their measurements. Understanding the fact, that the spacecraft will not need to slew and that the orbits are polar, the most appropriate control system is 3-axis control package consisting of three reaction wheels acting in the roll, pitch, and yaw directions, respectively. Table 4.2.1.2 displays a viable set of control actuators for the spacecraft during each of the control modes:

Table 4.2.1.2 Attitude determination and control subsystem control actuators

Actuator	Quantity	Mission Phase	Control Authority
Chemical	1	Orbit Injection	Roll (1), Pitch (1), Yaw (1)
Reaction Wheels	3	Stationkeeping	Roll (1), Pitch (1) Yaw (1) and System Momentum Dumping

We quantify the orbit environment by first estimating the worst-case disturbance torques that the spacecraft could encounter given the orbital parameters and the spacecraft geometry. These disturbances originate from solar-radiation, aerodynamic, magnetic gradient, and gravity gradient of the Earth. The geometry and mass of the vehicle is provided by structural requirements STR-1, STR-2, and STR-7, leading to the calculation of the moments of inertia shown below in Figure 4.2.1.1. (1):

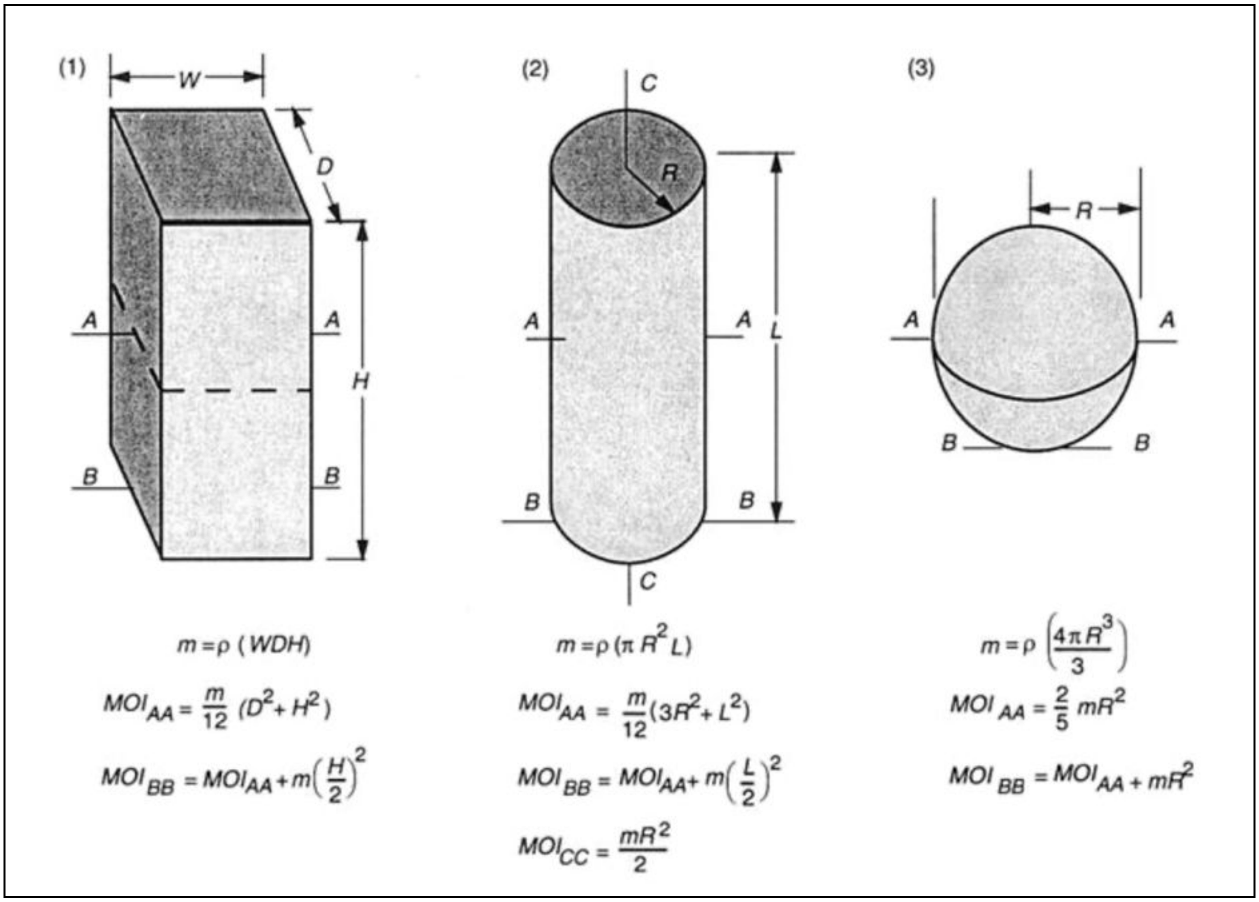

Figure 4.2.1.1 Moment of inertia calculations for basic bodies

Assuming the parameters in Table 4.2.1.3 and recognizing that given the lack of a magnetic field around the moon and presence of a negligible atmosphere, we determine two of four possible disturbance torques around the moon. They are the gravity disturbance and the solar radiation disturbance. Aerodynamic effects and magnetic effects are disregarded. Using equations found in SMAD Table 11-9A and assuming a uniform gravity field as an initial step for eventually performing sizing, we present the values in table 4.2.1.3.

Table 4.2.1.3 Satellite disturbance torque parameters

Parameter	
Spacecraft size	0.12m x 0.24m x 0.36m
I_X	0.006 kg m^2
I_Y	0.012 kg m^2
I_Z	0.0156 kg m^2
Offset of center (solar, aerodynamic) pressure from center of gravity	0.05 m
Solar reflectivity	0.95
Coefficient of drag	n/a
Magnetic dipole	n/a
Atmospheric density	n/a

The gravity gradient torque is found to be 2.95E-9 N-m and the torque due to the solar pressure is 3.84E-8 N-m. Each are shown in Figure 4.2.1.3.

Given that power, mass, and communications budgets are not yet finalized, sizing of each of the control actuators are not performed for this report and are left for a future team.

For attitude determination, we once again recognize that the critical requirement driving the decision for this class of sensors is the pointing requirement for stationkeeping and the requirements orbital injection. In order to minimize mass and volume impact, we opt to have a unified navigation sensor package for both control modes. To support injection, precise sensors are required to perform this high inclination maneuver. Table 11-14 in SMAD shows how inertial measurement systems coupled with communication with the lunar gateway may be enough to generate precise state estimation for the satellite during injection where the communication between lunar gateway shall contain position and navigation information to the spacecraft to aid with the error growth present in inertial sensors.

Two sensors that are appropriate for stationkeeping and nadir pointing orbits are horizon sensors and star sensors. We select a horizon sensor, one for roll and one for pitch, due to cost considerations and specify the scanner/pipper variant since they fit our accuracy needs. For the yaw direction, we choose sun sensors. Each of these stationkeeping sensors shall interact with the inertial measurement unit to compute improved navigation solutions. Table 11-14 SMAD provides a range of weight and power for each of these sensors – sun sensor ranges from 0.1 to 2 kg and 0 to 3 W, horizon sensor 1 to 4 kg and 5 to 10 W.

Since this mission requires the ADCS subsystem to manage two control modes, orbital injection and stationkeeping, and given that these two modes require commanding of two different sets of actuators, there must be two control laws. Each control law shall manage a single control mode, however they shall receive navigation sensor states from a single sensor package. The previous assumption of a uniform gravity gradient would be insufficient to realistically derive control laws for both lunar orbital injection and stationkeeping since the lunar gravity field varies greatly due to many large concentrations of mass. As such, the orbital injection and stationkeeping must account for these perturbations. Furthermore, these derivations are also highly dependent on the final selection of control actuators and sensors which have been omitted from this report and left to future team members. Traditional controllers may potentially be inadequate for performing precise control actions and thus we decided not to conduct a formal controls analysis that would be required for these two modes.

An initial set of AD&C subsystem parameters and components are shown below. For detailed mass and size information, refer to the mass budget:

Table 4.2.1.4 Attitude Determination and Control Subsystem parameters

ADCS Subsystem Parameters	
Gravity gradient disturbance torque, T_g	2.95E-9 N-m
Solar radiation disturbance torque, T_{sp}	3.84E-9 N-m
Magnetic field disturbance torque, T_m	n/a
Aerodynamic disturbance torque, T_a	n/a

Table 4.2.1.5 Attitude Determination and Control Subsystem components

ADCS Viable Subsystem Components				
	Component	Quantity	Mission Phase	Control Authority
Actuators	Chemical	1	Orbital Injection	Roll (1), Pitch (1), Yaw (1)
	Reaction Wheel	3	Stationkeeping	Roll (1), Pitch (1) Yaw (1) and System Momentum Dumping
Sensors	Horizon sensor	2		Roll (1), Pitch (1)
	Star sensor	1		Yaw
	IMU	1		Roll, Pitch, Yaw

4.3 Communications Requirements

Communications system design starts with the users DUST-Lunar will try to contact. Most planned Lunar missions use X or S-band communications, so DUST-Lunar must use either or both of those bands. As seen in Table 4.3.1, for calculation of link budgets, the team uses 5 W input power for each radio, less than 20 W total input power, 100 kbps data rate, and a 6 dB required link margin. A data rate of 100 kbps is required, since this is sufficient for voice communication [12].

Table 4.3.1 COM subsystem requirements

COM Requirements		
COM-1	Sub-system shall operate in the S/X-band range	O5
COM-2	Sub-system shall maintain a 6 dB link margin	F1
COM-4	Sub-system shall relay inter-satellite communications at a separation distance of at most 3500 km	F1
COM-5	Sub-system shall have a data rate of no less than 100 kbps	F1
COM-6	Sub-system shall operate on no more than 20 W in peak operation	F1

COM-7	Sub-system shall be able to identify the source of the received signal	F1
COM-8	Sub-system shall be able to recognize and prioritize priority messages during congestion events	F1
COM-9	Sub-system shall be able to receive and process multiple received messages at the same time	F1
COM-10	Sub-system shall maintain at least one operational radio during safe mode	F1

For component selection, a trade study is done between Endurosat's S-band Commercial antenna, Endurosat's S-band ISM antenna, Anywaves' S-band antenna, AAC-Clyde's S-band antenna, Endurosat's X-band Singular antenna, Endurosat's X-band 2x2 antenna, Endurosat's X-band 4x4 antenna, Anywaves' X-band antenna, AAC-Clyde's PULSAR-XANT antenna, and AAC-Clyde's XANT-PLUS antenna [7], [8], [9]. There are three cases for communication: cross-link, ground communication, and communication with Gateway.

4.3.1 Cross-Link System Design

Cross-link system design is done from a reference range of 3500 km, the range between two adjacent satellites in the same plane the final orbital constellation in section 3.1.3. As Figures 4.3.1.1 and 4.3.1.2 below show, cross-link is not possible with any studied commercially available X or S-band antenna.

				Endurosat X-band Sing	Endurosat X-band 2x2	Endurosat X-band 4x4	Anywaves X-band Ante	AAC-Clyde PULSAR-X	AAC-Clyde XANT-PLU
Item	Symbol	Units	Source	Spacecraft to Spacecraft	Spacecraft to Spacecraft	Spacecraft to Spacecraft	Spacecraft to Spacecraft	Spacecraft to Spacecraft	Spacecraft to Spacecraft
Frequency	f	GHz	Input Parameter	8.00	8.00	8.00	8.00	8.00	8.00
Transmitter Power (DC)	P	Watts	Input Parameter	5.00	5.00	5.00	5.00	5.00	5.00
Transmitter Power Amplifier Efficiency	η_P	--	Input Parameter	0.11	0.11	0.11	0.11	0.11	0.11
Transmitter Power (RF)	P	Watts	$P*\eta_P$	0.55	0.55	0.55	0.55	0.55	0.55
Transmitter Power (RF)	P	dBW	10 log(P)	-2.596	-2.596	-2.596	-2.596	-2.596	-2.596
Transmitter Line Loss	L_l	dB	Input Parameter	-2.000	-2.000	-2.000	-2.000	-2.000	-2.000
Transmit Antenna Efficiency	η_t	--	Input Parameter	0.40	0.40	0.40	0.40	0.40	0.40
Peak Transmit Antenna Gain	G_{pt}	dBi	Eq. (13-18b)	6.00	12.00	16.00	11.50	7.75	11.50
Transmit Antenna Pointing Error	e_t	deg	Input Parameter	0.953	0.953	0.953	0.953	0.953	0.953
Transmit Antenna Pointing Loss	L_{pt}	dB	Eq. (13-21)	-0.120	-0.120	-0.120	-0.120	-0.120	-0.120
Transmit Antenna Gain (net)	G_t	dBi	$G_{pt} + L_{pt}$	5.88	11.88	15.88	11.38	7.63	11.38
Equiv. Isotropic Radiated Power	$EIRP$	dBW	$P + L_l + G_t$	1.28	7.28	11.28	6.78	3.03	6.78
Propagation Path Length	S	km	h/cos(85°)	3.500E+03	3.500E+03	3.500E+03	3.500E+03	3.500E+03	3.500E+03
Space Loss	L_s	dB	Eq. (13-23a)	-181.38	-181.38	-181.38	-181.38	-181.38	-181.38
Propagation & Polarization Loss	L_a	dB	Fig. 13-10	0.0	0.0	0.0	0.0	0.0	0.0
Peak Receive Antenna Gain	G_{rp}	dBi	Eq. (13-18b)	6.00	12.00	16.00	11.50	7.75	11.50
Receive Antenna Pointing Loss	L_{pr}	dB	Eq. (13-21)	-0.120	-0.120	-0.120	-0.120	-0.120	-0.120
Receive Antenna Gain (net)	G_r	dBi	$G_{rp} + L_{pr}$	5.88	11.88	15.88	11.38	7.63	11.38
Data Rate	R	bps	Input Parameter	100000	100000	100000	100000	100000	100000
Effective Data Rate	R	bps	*See cell	222222	222222	222222	222222	222222	222222
E_b/N_o (1)	E_b/N_o	dB	Eq. (13-13)	-17.57	-5.57	2.43	-6.57	-14.07	-6.57
Carrier-to-Noise Density Ratio	C/N_o	dB-Hz	Eq. (13-15a)	35.90	47.90	55.90	46.90	39.40	46.90
Bit Error Rate	BER	--	Input Parameter	1.000E-07	1.000E-07	1.000E-07	1.000E-07	1.000E-07	1.000E-07
Required E_b/N_o (2)	$Req\ E_b/N_o$	dB	Fig. 13-9	5.8	5.8	5.8	5.8	5.8	5.8
Implementation Loss (3)	---	dB	Input Parameter	-1.0	-1.0	-1.0	-1.0	-1.0	-1.0
Margin	---	dB	(1) – (2) + (3) + (4)	-24.366	-12.366	-4.366	-13.366	-20.866	-13.366

Figure 4.3.1.1 X-Band Cross-Link Link Margin

Item	Symbol	Units	Source	Endurosat S-band Com Spacecraft to Spacecraft	Endurosat S-band ISM Spacecraft to Spacecraft	Anywaves S-band Ante Spacecraft to Spacecraft	AAC-Clyde S-band Ant Spacecraft to Spacecraft	Inflatable Antenna Spacecraft to Spacecraft
Frequency	f	GHz	Input Parameter	2.50	2.50	2.50	2.50	2.50
Transmitter Power (DC)	P	Watts	Input Parameter	5.00	5.00	5.00	5.00	5.00
Transmitter Power Amplifier Efficiency	η_P	--	Input Parameter	0.11	0.11	0.11	0.11	0.11
Transmitter Power (RF)	P	dBW	10 log(P)	-2.596	-2.596	-2.596	-2.596	-2.596
Transmitter Line Loss	L_l	dB	Input Parameter	-2.000	-2.000	-2.000	-2.000	-2.000
Transmit Antenna Beamwidth	θ_t	deg	Input Parameter	30.480	30.480	30.480	30.480	8.750
Peak Transmit Antenna Gain	G_{pt}	dBi	Eq. (13-18b)	7.00	8.30	6.00	7.00	21.00
Transmit Antenna Pointing Loss	L_{pt}	dB	Eq. (13-21)	-0.120	-0.120	-0.120	-0.120	-0.120
Transmit Antenna Gain (net)	G_t	dBi	$G_{pt} + L_{pt}$	6.88	8.18	5.88	6.88	20.88
Equiv. Isotropic Radiated Power	$EIRP$	dBW	$P + L_l + G_t$	2.28	3.58	1.28	2.28	16.28
Altitude	h	km	Input Parameter	100.00	100.00	100.00	100.00	100.00
Propagation Path Length	S	km	h/cos(85°)	3.500E+03	3.500E+03	3.500E+03	3.500E+03	3.500E+03
Space Loss	L_s	dB	Eq. (13-23a)	-171.28	-171.28	-171.28	-171.28	-171.28
Propagation & Polarization Loss	L_a	dB	Fig. 13-10	0.0	0.0	0.0	0.0	0.0
Receive Antenna Efficiency	η_r	--	Input Parameter	0.40	0.40	0.40	0.40	0.40
Peak Receive Antenna Gain	G_{rp}	dBi	Eq. (13-18b)	7.00	8.30	6.00	7.00	21.00
Receive Antenna Beamwidth	θ_r	deg	Eq. (13-19)	30.480	30.480	30.480	30.480	8.750
Receive Antenna Pointing Loss	L_{pr}	dB	Eq. (13-21)	-0.120	-0.120	-0.120	-0.120	-0.120
Receive Antenna Gain (net)	G_r	dBi	$G_{rp} + L_{pr}$	6.88	8.18	5.88	6.88	20.88
Data Rate	R	bps	Input Parameter	100000	100000	100000	100000	100000
Effective Data Rate	R	bps	*See cell	222222	222222	222222	222222	222222
E_b/N_o (1)	E_b/N_o	dB	Eq. (13-13)	-5.46	-2.86	-7.46	-5.46	22.54
Carrier-to-Noise Density Ratio	C/N_o	dB-Hz	Eq. (13-15a)	48.01	50.61	46.01	48.01	76.01
Bit Error Rate	BER	--	Input Parameter	1.000E-07	1.000E-07	1.000E-07	1.000E-07	1.000E-07
Required E_b/N_o (2)	$Req\ E_b/N_o$	dB	Fig. 13-9	5.8	5.8	5.8	5.8	5.8
Margin	---	dB	(1) – (2) + (3) + (4)	-12.263	-9.663	-14.263	-12.263	15.737

Figure 4.3.1.2 S-Band Cross-Link Link Margin

Cross-link is possible with an inflatable antenna, such as the 1 m antenna modeled in "Inflatable antenna for cubesats: Motivation for development and antenna design" [10]. The antenna in that paper is an S-band, 21 dB gain, 1 m diameter, and 0.5 U storage volume. This provides a maximum range of 10000 km at 5 W input power or a minimum power of 0.53 W at 3500 km range, as shown in figures 4.3.1.3 and 4.3.1.4 below.

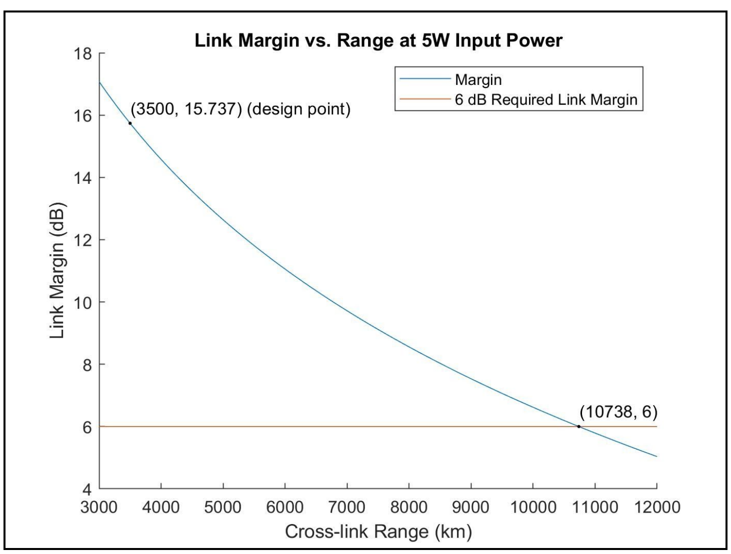

Figure 4.3.1.3 Inflatable Antenna Link Margin vs. Range

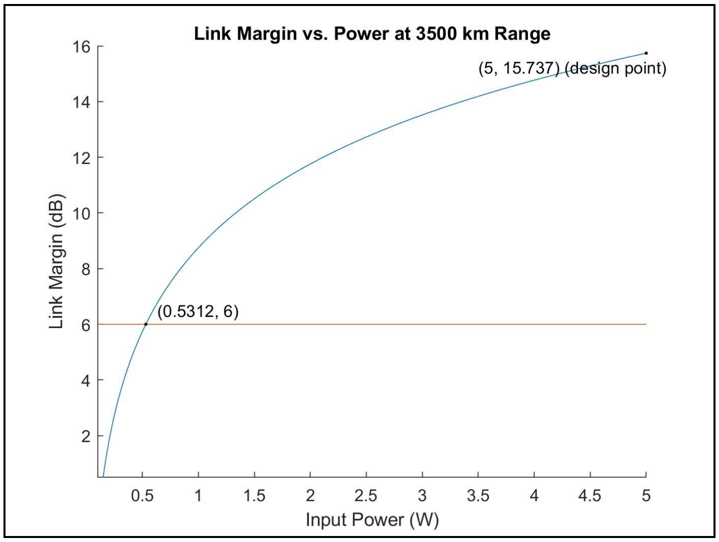

Figure 4.3.1.4 Inflatable Antenna Link Margin vs. Power

There are technical risks with this solution, since the technology is unproven, pointing of the antennas would have to be done mechanically, the antennas would partially block the solar panels, and the antennas would have vibrational and attitude control effects, all of which require further design and analysis to mitigate risks.

4.3.2 Ground Communication

Ground communication is driven by transmission to the ground rather than receiving from the ground, since it is presumed that lunar assets have more power and/or higher gains than DUST-Lunar's antennas. For ground communication, the team assumes a 16 dB receive gain, the gain from Endurosat's X-band 4x4 antenna and a range of 750 km, the altitude of the final orbits from section 3.1.3 [Appendix C]. As figures 4.3.2.1 and 4.3.2.2 below show, the only studied antenna for which transmission to ground is possible is Endurosat's X-band 4x4 antenna, so that is the component selected for this antenna.

Item	Symbol	Units	Source	Endurosat X-band Sing Spacecraft to Ground	Endurosat X-band 2x2 Spacecraft to Ground	Endurosat X-band 4x4 Spacecraft to Ground	Anywaves X-band Ante Spacecraft to Ground	AAC-Clyde PULSAR-X Spacecraft to Ground	AAC-Clyde XANT-PL Spacecraft to Ground
Frequency	f	GHz	Input Parameter	8.00	8.00	8.00	8.00	8.00	8.00
Transmitter Power (DC)	P	Watts	Input Parameter	5.00	5.00	5.00	5.00	5.00	5.00
Transmitter Power Amplifier Efficiency	η_P	--	Input Parameter	0.11	0.11	0.11	0.11	0.11	0.11
Transmitter Power (RF)	P	Watts	$P*\eta_P$	0.55	0.55	0.55	0.55	0.55	0.55
Transmitter Power (RF)	P	dBW	10 log(P)	-2.596	-2.596	-2.596	-2.596	-2.596	-2.596
Transmitter Line Loss	L_l	dB	Input Parameter	-2.000	-2.000	-2.000	-2.000	-2.000	-2.000
Transmit Antenna Efficiency	η_t	--	Input Parameter	0.55	0.55	0.55	0.55	0.55	0.55
Peak Transmit Antenna Gain	G_{pt}	dBi	Eq. (13-18b)	6.00	12.00	16.00	11.50	7.75	11.50
Transmit Antenna Pointing Error	e_t	deg	Input Parameter	0.953	0.953	0.953	0.953	0.953	0.953
Transmit Antenna Pointing Loss	L_{pt}	dB	Eq. (13-21)	-0.120	-0.120	-0.120	-0.120	-0.120	-0.120
Transmit Antenna Gain (net)	G_t	dBi	$G_{pt} + L_{pt}$	5.88	11.88	15.88	11.38	7.63	11.38
Equiv. Isotropic Radiated Power	$EIRP$	dBW	$P + L_l + G_t$	1.28	7.28	11.28	6.78	3.03	6.78
Propagation Path Length	S	km	h/cos(85°)	7.500E+02	7.500E+02	7.500E+02	7.500E+02	7.500E+02	7.500E+02
Space Loss	L_s	dB	Eq. (13-23a)	-168.00	-168.00	-168.00	-168.00	-168.00	-168.00
Propagation & Polarization Loss	L_a	dB	Fig. 13-10	0.0	0.0	0.0	0.0	0.0	0.0
Peak Receive Antenna Gain	G_{rp}	dBi	Eq. (13-18b)	16.00	16.00	16.00	16.00	16.00	16.00
Receive Antenna Pointing Loss	L_{pr}	dB	Eq. (13-21)	-0.120	-0.120	-0.120	-0.120	-0.120	-0.120
Receive Antenna Gain (net)	G_r	dBi	$G_{rp} + L_{pr}$	15.88	15.88	15.88	15.88	15.88	15.88
Data Rate	R	bps	Input Parameter	100000	100000	100000	100000	100000	100000
Effective Data Rate	R	bps	*See cell	222222	222222	222222	222222	222222	222222
E_b/N_o (1)	E_b/N_o	dB	Eq. (13-13)	5.81	11.81	15.81	11.31	7.56	11.31
Carrier-to-Noise Density Ratio	C/N_o	dB-Hz	Eq. (13-15a)	59.28	65.28	69.28	64.78	61.03	64.78
Bit Error Rate	BER	--	Input Parameter	1.000E-07	1.000E-07	1.000E-07	1.000E-07	1.000E-07	1.000E-07
Required E_b/N_o (2)	$Req\ E_b/N_o$	dB	Fig. 13-9	5.8	5.8	5.8	5.8	5.8	5.8
Implementation Loss (3)	---	dB	Input Parameter	-1.0	-1.0	-1.0	-1.0	-1.0	-1.0
Margin	---	dB	(1) – (2) + (3) + (4)	-0.986	5.014	9.014	4.514	0.764	4.514

Figure 4.3.2.1 X-Band Ground Transmission Link Margin

Item	Symbol	Units	Source	Endurosat S-band Com Spacecraft to Ground	Endurosat S-band ISM Spacecraft to Ground	Anywaves S-band Ante Spacecraft to Ground	AAC-Clyde S-band Antenna Spacecraft to Ground
Frequency	f	GHz	Input Parameter	2.50	2.50	2.50	2.50
Transmitter Power (DC)	P	Watts	Input Parameter	5.00	5.00	5.00	5.00
Transmitter Power Amplifier Efficiency	η_P	--	Input Parameter	0.11	0.11	0.11	0.11
Transmitter Power (RF)	P	dBW	10 log(P)	-2.596	-2.596	-2.596	-2.596
Transmitter Line Loss	L_l	dB	Input Parameter	-2.000	-2.000	-2.000	-2.000
Transmit Antenna Beamwidth	θ_t	deg	Input Parameter	30.480	30.480	30.480	30.480
Peak Transmit Antenna Gain	G_{pt}	dBi	Eq. (13-18b)	7.00	8.30	6.00	7.00
Transmit Antenna Pointing Loss	L_{pt}	dB	Eq. (13-21)	-0.120	-0.120	-0.120	-0.120
Transmit Antenna Gain (net)	G_t	dBi	$G_{pt} + L_{pt}$	6.88	8.18	5.88	6.88
Equiv. Isotropic Radiated Power	$EIRP$	dBW	$P + L_l + G_t$	2.28	3.58	1.28	2.28
Altitude	h	km	Input Parameter	100.00	100.00	100.00	100.00
Propagation Path Length	S	km	h/cos(85°)	7.500E+02	7.500E+02	7.500E+02	7.500E+02
Space Loss	L_s	dB	Eq. (13-23a)	-157.90	-157.90	-157.90	-157.90
Propagation & Polarization Loss	L_a	dB	Fig. 13-10	0.0	0.0	0.0	0.0
Receive Antenna Efficiency	η_r	--	Input Parameter	0.55	0.55	0.55	0.55
Peak Receive Antenna Gain	G_{rp}	dBi	Eq. (13-18b)	8.30	8.30	8.30	8.30
Receive Antenna Beamwidth	θ_r	deg	Eq. (13-19)	40.640	40.640	40.640	40.640
Receive Antenna Pointing Loss	L_{pr}	dB	Eq. (13-21)	-0.120	-0.120	-0.120	-0.120
Receive Antenna Gain (net)	G_r	dBi	$G_{rp} + L_{pr}$	8.18	8.18	8.18	8.18
Data Rate	R	bps	Input Parameter	100000	100000	100000	100000
Effective Data Rate	R	bps	*See cell	222222	222222	222222	222222
E_b/N_o (1)	E_b/N_o	dB	Eq. (13-13)	9.22	10.52	8.22	9.22
Carrier-to-Noise Density Ratio	C/N_o	dB-Hz	Eq. (13-15a)	62.69	63.99	61.69	62.69
Bit Error Rate	BER	--	Input Parameter	1.000E-07	1.000E-07	1.000E-07	1.000E-07
Required E_b/N_o (2)	$Req\ E_b/N_o$	dB	Fig. 13-9	5.8	5.8	5.8	5.8
Margin	---	dB	(1) – (2) + (3) + (4)	2.417	3.717	1.417	2.417

Figure 4.3.2.2 S-Band Ground Transmission Link Margin

The Endurosat X-band 4x4 antenna provides 16 dB gain and 18 degree beamwidth as well as a maximum altitude of 1061 km at 5 W input power or a minimum power of 2.4979 W at 750 km altitude, as shown in figures 4.3.2.3 and 4.3.2.4 below [Appendix C].

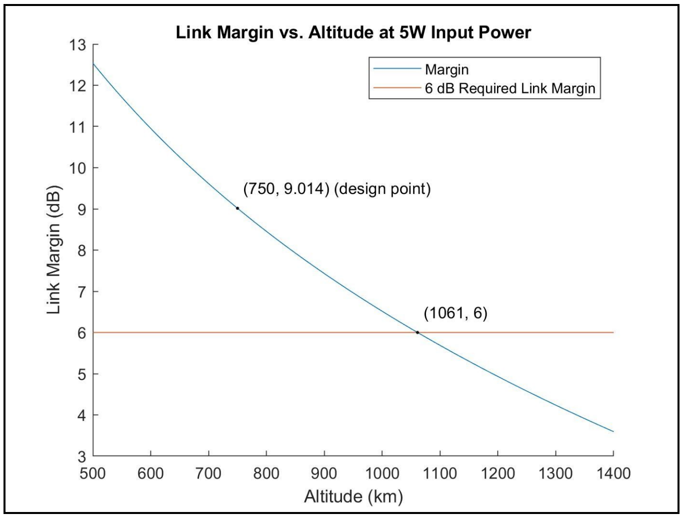

Figure 4.3.2.3 Endurosat X-Band 4x4 Link Margin vs. Altitude

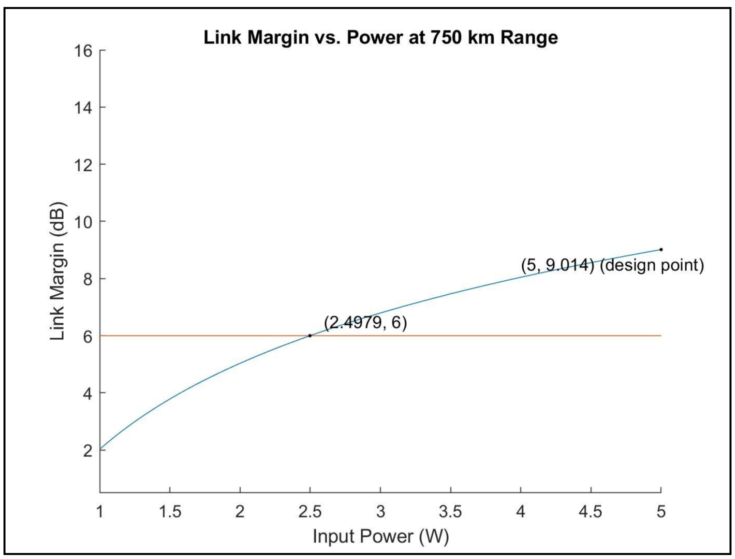

Figure 4.3.2.4 Endurosat X-Band 4x4 Link Margin vs. Power

4.3.3 Communication With Gateway

Communication with Lunar Gateway is driven primarily by transmission to Gateway, since Gateway has higher gain and more power than DUST-Lunar. Furthermore, antenna selection is primarily driven by Gateway's antenna gain, since Gateway's antenna gain is expected to be much larger than DUST-Lunar's antenna gain. However, Gateway's antenna gain is not published in the Gateway System Requirements, so the team estimate Gateway's antenna gain as 68.82 dB for X-band or 65.75 for S-band [11], [Appendix C].

As figures 4.3.3.1 and 4.3.3.2 below show, any medium to high gain commercial antenna will be sufficient to communicate with Lunar Gateway. Therefore, the team select the cross-link 1 m inflatable antenna to communicate with Gateway, since this lowers the communications system volume by not having an antenna for specifically communicating with Gateway. Furthermore, the cross-link antenna has a link margin of about 35 dB, which provides spare margin in case Gateway's communications requirements are descoped.

				Endurosat X-band Sing	Endurosat X-band 2x2	Endurosat X-band 4x4	Anywaves X-band Ante	AAC-Clyde PULSAR-	AAC-Clyde XANT-PLUS
Item	**Symbol**	**Units**	**Source**	**Spacecraft to Gateway**	**Spacecraft to Gateway**	**Spacecraft to Gateway**	**Spacecraft to Gateway**	**Spacecraft to Gateway**	**Spacecraft to Gateway**
Frequency	f	GHz	Input Parameter	8.00	8.00	8.00	8.00	8.00	8.00
Transmitter Power (DC)	P	Watts	Input Parameter	5.00	5.00	5.00	5.00	5.00	5.00
Transmitter Power Amplifier Efficiency	η_P	--	Input Parameter	0.11	0.11	0.11	0.11	0.11	0.11
Transmitter Power (RF)	P	Watts	$P*\eta_P$	0.55	0.55	0.55	0.55	0.55	0.55
Transmitter Power (RF)	P	dBW	$10 \log(P)$	-2.596	-2.596	-2.596	-2.596	-2.596	-2.596
Transmitter Line Loss	L_l	dB	Input Parameter	-2.000	-2.000	-2.000	-2.000	-2.000	-2.000
Transmit Antenna Efficiency	η_t	--	Input Parameter	0.55	0.55	0.55	0.55	0.55	0.55
Peak Transmit Antenna Gain	G_{pt}	dBi	Eq. (13-18b)	6.00	12.00	16.00	11.50	7.75	11.50
Transmit Antenna Pointing Error	e_t	deg	Input Parameter	0.953	0.953	0.953	0.953	0.953	0.953
Transmit Antenna Pointing Loss	L_{pt}	dB	Eq. (13-21)	-0.120	-0.120	-0.120	-0.120	-0.120	-0.120
Transmit Antenna Gain (net)	G_t	dBi	$G_{pt} + L_{pt}$	5.88	11.88	15.88	11.38	7.63	11.38
Equiv. Isotropic Radiated Power	$EIRP$	dBW	$P + L_l + G_t$	1.28	7.28	11.28	6.78	3.03	6.78
Propagation Path Length	S	km	h/cos(85°)	7.000E+04	7.000E+04	7.000E+04	7.000E+04	7.000E+04	7.000E+04
Space Loss	L_s	dB	Eq. (13-23a)	-207.40	-207.40	-207.40	-207.40	-207.40	-207.40
Propagation & Polarization Loss	L_a	dB	Fig. 13-10	0.0	0.0	0.0	0.0	0.0	0.0
Peak Receive Antenna Gain	G_{rp}	dBi	Eq. (13-18b)	69.00	69.00	69.00	69.00	69.00	69.00
Receive Antenna Pointing Loss	L_{pr}	dB	Eq. (13-21)	-0.120	-0.120	-0.120	-0.120	-0.120	-0.120
Receive Antenna Gain (net)	G_r	dBi	$G_{rp} + L_{pr}$	68.88	68.88	68.88	68.88	68.88	68.88
Data Rate	R	bps	Input Parameter	100000	100000	100000	100000	100000	100000
Effective Data Rate	R	bps	*See cell	222222	222222	222222	222222	222222	222222
E_b/N_o (1)	E_b/N_o	dB	Eq. (13-13)	19.41	25.41	29.41	24.91	21.16	24.91
Carrier-to-Noise Density Ratio	C/N_o	dB-Hz	Eq. (13-15a)	72.88	78.88	82.88	78.38	74.63	78.38
Bit Error Rate	BER	--	Input Parameter	1.000E-07	1.000E-07	1.000E-07	1.000E-07	1.000E-07	1.000E-07
Required E_b/N_o (2)	$Req\ E_b/N_o$	dB	Fig. 13-9	5.8	5.8	5.8	5.8	5.8	5.8
Implementation Loss (3)	---	dB	Input Parameter	-1.0	-1.0	-1.0	-1.0	-1.0	-1.0
Margin	---	dB	(1) – (2) + (3) + (4)	12.614	18.614	22.614	18.114	14.364	18.114

Figure 4.3.3.1 X-Band Gateway Transmission Link Margin

				Endurosat S-band Com	Endurosat S-band ISM	Anywaves S-band Ante	AAC-Clyde S-band Ant
Item	**Symbol**	**Units**	**Source**	**Spacecraft to Gateway**	**Spacecraft to Gateway**	**Spacecraft to Gateway**	**Spacecraft to Gateway**
Frequency	f	GHz	Input Parameter	2.50	2.50	2.50	2.50
Transmitter Power (DC)	P	Watts	Input Parameter	5.00	5.00	5.00	5.00
Transmitter Power Amplifier Efficiency	η_P	--	Input Parameter	0.11	0.11	0.11	0.11
Transmitter Power (RF)	P	dBW	10 log(P)	-2.596	-2.596	-2.596	-2.596
Transmitter Line Loss	L_l	dB	Input Parameter	-2.000	-2.000	-2.000	-2.000
Transmit Antenna Beamwidth	θ_t	deg	Input Parameter	30.480	30.480	30.480	30.480
Peak Transmit Antenna Gain	G_{pt}	dBi	Eq. (13-18b)	7.00	8.30	6.00	7.00
Transmit Antenna Pointing Loss	L_{pt}	dB	Eq. (13-21)	-0.120	-0.120	-0.120	-0.120
Transmit Antenna Gain (net)	G_t	dBi	$G_{pt} + L_{pt}$	6.88	8.18	5.88	6.88
Equiv. Isotropic Radiated Power	$EIRP$	dBW	$P + L_l + G_t$	2.28	3.58	1.28	2.28
Altitude	h	km	Input Parameter	100.00	100.00	100.00	100.00
Propagation Path Length	S	km	h/cos(85°)	7.000E+04	7.000E+04	7.000E+04	7.000E+04
Space Loss	L_s	dB	Eq. (13-23a)	-197.30	-197.30	-197.30	-197.30
Propagation & Polarization Loss	L_a	dB	Fig. 13-10	0.0	0.0	0.0	0.0
Receive Antenna Efficiency	η_r	--	Input Parameter	0.50	0.50	0.50	0.50
Peak Receive Antenna Gain	G_{rp}	dBi	Eq. (13-18b)	66.00	66.00	66.00	66.00
Receive Antenna Beamwidth	θ_r	deg	Eq. (13-19)	84.000	84.000	84.000	84.000
Receive Antenna Pointing Loss	L_{pr}	dB	Eq. (13-21)	-0.120	-0.120	-0.120	-0.120
Receive Antenna Gain (net)	G_r	dBi	$G_{rp} + L_{pr}$	65.88	65.88	65.88	65.88
Data Rate	R	bps	Input Parameter	100000	100000	100000	100000
Effective Data Rate	R	bps	*See cell	222222	222222	222222	222222
E_b/N_o (1)	E_b/N_o	dB	Eq. (13-13)	27.52	28.82	26.52	27.52
Carrier-to-Noise Density Ratio	C/N_o	dB-Hz	Eq. (13-15a)	80.98	82.28	79.98	80.98
Bit Error Rate	BER	--	Input Parameter	1.000E-07	1.000E-07	1.000E-07	1.000E-07
Required E_b/N_o (2)	$Req\ E_b/N_o$	dB	Fig. 13-9	5.8	5.8	5.8	5.8
Margin	---	dB	(1) – (2) + (3) + (4)	20.717	22.017	19.717	20.717

Figure 4.3.3.2 S-Band Gateway Transmission Link Margin

4.4 C&DH Requirements

The C&DH subsystem is responsible for monitoring the overall health and operation of the CubeSat. Communications will not be handled by the C&DH because communications are the main function of the CubeSat and will be controlled by the payload. The C&DH subsystem requirements were mostly taken directly from already defined DUST-LEO requirements. The finalized requirements are shown in Table 4.4.1. The most notable requirement is CDH-6. The CubeSats were chosen to store 24 hours worth of telemetry data. With some added margin, a total of 11 MB are needed in the subsystem for stored telemetry data.

Table 4.4.1. C&DH subsystem requirements

C&DH Requirements		
CDH-1	Sub-system shall log telemetry data	F1
CDH-2	Sub-system shall monitor voltage, current, power consumption, temperature parameters of other subsystems	F1
CDH-3	Sub-system shall process telemetry and trajectory propagation data	F1
CDH-4	Sub-system shall be capable of processing real-time operations	F1
CDH-5	Sub-system shall be able to process data at a speed of at least 1 Mbps	F1
CDH-6	Sub-system shall be able to store at least 11 MB of telemetry data	P1
CDH-7	Sub-system shall be able to indicate when CubeSat is in safe mode	O2
CDH-8	Sub-system shall be able to exit safe mode when nominal operations can be established	O2
CDH-9	Sub-system shall be able to initiate testing sequence	O2
CDH-10	Sub-system shall be able to recover from a radiation-induced memory-corruption event	O2
CDH-11	Sub-system shall be able to store up to 24 hours of health reports	O2

4.4.1 System Design

After researching possible C&DH parts, the ISIS On Board Computer was identified. This flight computer has a processing speed of 400 MHz as well as 64 MB of RAM, and both of these specs exceed the requirements [Appendix B]. The ISIS computer also has flight heritage since 2014, so there will be little risk associated with this component.

4.5 Structure Requirements

The CubeSat structure requirements, shown in Table 4.5.1, have mostly been set by satellite standards or launch constraints. STR-1 and STR-7, which constrain the CubeSat size and mass, were set by Artemis and Lunar Gateway. The ConOps relies on the fact that an Artemis SLS will transport DUST-Lunar CubeSats and the satellites will be launched by Lunar Gateway, so Artemis and Gateway requirements must be met.

Table 4.5.1 CubeSat structure requirements

STR Requirements		
STR-1	Sub-system shall confine within a 6U form factor	C6
STR-2	CoG shall be X:+/- 1.5, Y:+0-+5, Z: +6-+9.5 (mm)	C6
STR-3	Sub-system shall maintain radiation accumulation under 100 rad over CubeSat lifetime for electronics	O2
STR-4	Total Mass Loss (TML) shall be < 1.0 %	P1
STR-5	Collected Volatile Condensable Material (CVCM) shall be < 0.1%	C3
STR-6	Sub-system shall be designed to withstand temperature changes of orbit	O2
STR-7	Sub-system shall weigh less than 12 kg	C6

4.5.1 System Design

For the overall structure of the CubeSat, a pre-built 6U frame will most likely be used. Several pre-built frames have been researched, and overall, it seems to be very easy and relatively cheap to simply buy a frame. All subsystems will then be integrated into the frame for the overall CubeSat design.

4.6 EPS Requirements

Table 4.6.1 EPS System Requirements

EPS Requirements		
EPS-1	Subsystem shall be powered off prior to deployment	C6
EPS-2	Sub-system shall generate at least 15 W of power for continuous operations during safe mode	P1
EPS-3	Sub-system shall generate at least 30 W of power for continuous operations during normal mode	P1
EPS-4	Sub-system shall operate within the batteries' thermal range during nominal operations	O2
EPS-5	Sub-system shall provide regulated power output buses	F1
EPS-6	Sub-system shall include battery circuit protection to avoid cell unbalance	C6
EPS-7	Sub-system design shall not permit the ground charge circuit to energize the satellite systems, including flight computer	O2
EPS-8	Solar panels shall have sufficient backwiring to cancel out magnetic dipoles generated by the solar cells	ADCS-5
EPS-9	Sub-system shall protect the spacecraft against failures within EPS	O2
EPS-10	Sub-system shall suppress transient bus voltages and protect against bus faults	O2
EPS-11	Sub-system shall have adequate battery capacity for the pre deployment phase	O2
EPS-12	Sub-system shall use Lithium-ion 18650 type batteries	C6

Featured in Table 4.6.1 are the design requirements of the EPS system as well as their traceability throughout the complete list of requirements which can be found in Appendix A. Power generation requirements have been backed out based on knowledge that was available at the time of the final presentation. Since then Space Dynamic Lab has returned further information in regards to the power requirements of the Iris V2 Radio. They have confirmed that the radio operates on a stepped type power system. Therefore, during operation of the radios a power draw of over 60W would be required based on the Iris Radio specs that can be found in Appendix C.

4.6.1 System Design

Table 4.6.1.1 Solar Array and Battery Sizing

Parameter	**Outdated Power Estimates**	**Updated Power Estimates**
# Sats	8	8
Altitude	750 Km	750 Km
Max Eclipse	2724.256	2724.256
Min Eclipse	0	0
Power Consumption	30	80
Lifetime	3	3
Max Cell Area	0.194797019	0.519458716
Min Cell Area	0.136029712	0.362745899
Cr	84.08197531	224.2186008

Table 4.6.1.2 details the results of the previously estimated power requirements vs the new power requirements based on the information received from Space Dynamics Laboratory. The power requirements were determined operating on the assumption that the general satellite power draw would not exceed 10 W. This value is obtained from power requirements for the 3U technology demonstration satellite which had an estimated power draw of 5 W for all systems excluding communications. This value was doubled as a coarse estimate of power requirements for the 6U Lunar satellite. For communications, the Iris antenna is an all in one package that requires only an antenna to begin communications. Therefore, using power specifications for the Iris V2 radio, available in appendix B, which indicate a 35W draw during transmission and receiving which can be assumed to be in operation continuously a power draw of 80W is found. For the previous estimate, the power draw was determined to be 5W for each antenna in order to close the link budget to communicate with other satellites in the constellation. In the previous estimation a 50% margin was placed on the estimated value to account for any variations that may mature with the design.

The current power requirements will need to be further refined as they eliminate nearly a third of the weight allotted to the satellites as determined by the structural requirements. Estimates based off of current state of the art hardware suggest that the weight of batteries of sufficient capacity will come to 1.5 Kg if using 154 Whr/Kg performance provided by GOM space and the mass of the solar arrays will come roughly 2.2 Kg if using 155W/Kg performance provided by MMA[1],[2]. Further work will be necessary to appropriately assess power generation and storage needs as well as volume and mass appropriations for the system.

4.7 Thermal Considerations

As the focus of the semester was spent primarily developing a mission concept and budgets, thermal control was not a primary focus. Based on the orbital parameters that have been developed, heat dissipation is the primary concern. The frozen orbit selected results in continuous sunlight for periods of over 2 months. This is highly beneficial to the mission as the antennas have large power requirements but also results in the need to dissipate the excess energy so that the spacecraft does not overheat. Temperatures may be too low in some instances during the longer eclipsed orbits where the satellite will be in darkness for over two hours. Thermal control will be more thoroughly explored in future iterations of this design.

4.8 Cost

The mission is planned around a small explorer (SMEX) budget of $120 million as explained in the Mission Constraints section (2.2). This will include the money needed to develop, build, and launch these CubeSats. The tentative cost of fabricating and launching the CubeSats is shown in Table 4.8.1.

Table 4.8.1 Mission cost budget

Component	Quantity	Cost per Unit ($)	Contingencies (%)	Total Cost ($)
ADCS				
CubeADCS	1	$50,000	20.00%	$60,000
COM				
IRIS Transponder	2	$960,000	5.00%	$2,016,000
Phased Array Antenna	3	$10,000	25.00%	$37,500
COM Board + Electronics	1	$1,000	20.00%	$1,200
EPS				
MMA Solar Array	1	$150,000	25.00%	$187,500
Battery Pack	1	$5,000	20.00%	$6,000
QB50 EPS	1	$300	20.00%	$360
STR				
Frame	1	$8,000	20.00%	$9,600
Hardware	1	$1,500	25.00%	$1,875
CDH				
Flight Computer	1	$6,835	5.00%	$7,177
Sub-Total Cost of Components (Single Satellite):				**$2,327,212**
Margin	25%			$581,803
Mass per Satellite (kg)				12.00
Cost per kg of Lunar cargo				$25,000
Single Satellite Total:				**$3,209,015**
Number of Satellites				8.00
Mission Total:				***$25,672,118***

Cost estimates were generated from several different methods. Some subsystem costs were taken directly from obtained quotes, some were estimated based on the DUST-LEO budget, and others were very roughly estimated due to a limited amount of pricing information found online. The added percentage contingency margins reflect the general uncertainty associated with each subsystem component. As the degree of uncertainty increased for a part, the percentage margin also increased to yield a conservative estimate on the satellite cost. As for the launch cost, an estimated $25,000 per kg was used based on the overall cost of an SLS shuttle. Even with very rough pricing, the overall cost of $25.7 million is well below the total budget of $120 million, so money does not appear to be a major concern at this time. Component pricing must still be monitored, however, to ensure the budget is ultimately met. Labor costs are also not estimated in the current cost budget, but labor required to design and build the satellite will also need to be accounted for.

4.9 Risk Analysis

Each subsystem design comes with an inherent risk. The level of risk varies with each component or subsystem, but all risks must still be addressed and accounted for. We define two principal forms of risk as technical, risk associated with component, subsystem, or system failure, and mission, risk associated with failure to accomplish mission objectives.

The Attitude Determination and Controls subsystem presents significant mission and technical risk for the vehicle to carry out its tasks. The foremost source of risk is storage and maintenance of the propellant for the primary thrusters for the duration of launch and transfer to lunar gateway. If chemical propellant were to escape the spacecraft and related launch systems and lunar gateway would be in immediate danger. Failure of the primary thrusters would immediately prevent accurate orbital injection and would eliminate a node in the mesh. During the lifetime of the mission, this risk would still exist, however it can be mitigated by using “green” propellant. Failure of attitude sensors or stationkeeping actuators would present significant mission risk by potentially creating a scenario where a satellite node drifts beyond cross-link range and pointing requirements.

The technical risks for the communications subsystem come from the inflatable antenna, since the technology is unproven, pointing of the inflatable antennas would have to be done mechanically, the inflatable antennas would partially block the solar panels, and the inflatable antennas would have vibrational and attitude control effects, all of which require further design and analysis to mitigate risks.

The C&DH subsystem has very little risk. The subsystem will not handle any communications processing because the payload will control all communications. This reduces the responsibility of the C&DH subsystem and the necessary processing power. The identified ISIS computer exceeds all the necessary specs and has flight heritage, so the overall C&DH subsystem will have little risk.

The EPS system carries a couple risks. The hardware selection of the communications system will greatly drive power requirements which will then size solar arrays and batteries. This is highlighted in particular in Table 4.6.1.1 where the alteration from previously assumed values to those with the correct specifications offered by Space Dynamics Laboratory.

The tentative CubeSat cost from parts and launch comes in far under budget, but the cost of labor was not estimated. There is still little risk associated with the cost, though, because less than 25% of the overall budget is already accounted for. This leaves a massive margin for labor cost.

5. Conclusion

At the current time, a mission outline and hardware outline have been developed for a Lunar meshed network. It faces several key challenges: power generation to storage optimization, antenna technology for communication with Lunar Gateway, and thermal control during constant sunlight orbits. These technical challenges will need to be resolved in order to make the DUST-Lunar mission possible.

References

1. Yost, B. (2019, November 6). 03. Power. Retrieved December 13, 2019, from https://sst-soa.arc.nasa.gov/03-power.
2. HaWK Specifications for Comparison. (n.d.). Retrieved December 13, 2019, from https://mmadesignllc.com/specs-table/.
3. R. Whitley and R. Martinez, “Options for staging orbits in cislunar space,” *IEEE Aerosp. Conf. Proc.*, vol. 2016-June, 2016.
4. Staff, S. X. (2019, July 19). Angelic halo orbit chosen for humankind's first lunar outpost. Retrieved December 13, 2019, from https://phys.org/news/2019-07-angelic-halo-orbit-chosen-humankind.html.
5. P. M. Muller and W. L. Sjogren, “Mascons: Lunar mass concentrations,” 1968.
6. A. Elipe and M. Lara, “Frozen orbits about the Moon,” *J. Guid. Control. Dyn.*, vol. 26, no. 2, pp. 238–243, 2003.
7. Cubesat Antennas. Retrieved December 13, 2019, from https://www.endurosat.com/products/#cubesat-antennas
8. SMALLSATS. Retrieved December 13, 2019, from http://www.anywaves.eu/products/
9. COMMUNICATIONS. Retrieved December 13, 2019, from https://www.aac-clyde.space/satellite-bits/communications
10. Babuscia, A., Corbin, B., Knapp, M., Jensem-Clem, R., Loo, M., and Seager, S. (2013). Inflatable antenna for cubesats: motivation for development and antenna design. Acta Astronautica, Volume 91, pages 322-332. Retrieved December 13, 2019, from https://www.sciencedirect.com/science/article/pii/S0094576513001951
11. (2019). Gateway System Requirements. Retrieved December 13, 2019, from https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20190029153.pdf
12. Voice Over IP - Per Call Bandwidth Consumption. Retrieved December 13, 2019, from https://www.cisco.com/c/en/us/support/docs/voice/voice-quality/7934-bwidth-consume.html

APPENDIX A: Requirements and Constraints

Table A.1 Complete Requirements Listing

Top Level Requirements		
P-1	Deploy a mesh communication network around the Moon to facilitate communication between lunar ground assets and earth	
S-1	Create platform for future distributed Lunar sensing satellite constellation	
S-2	Serve as a test bed for new components and technology	
S-3	Develop standardized communications framework for future mission applications	
Functional Requirements		
FR-1	System shall implement in-network and out-of-network communications*	P-1
	FR-1.1: A CubeSat shall be in contact with at least 2 other CubeSats in the mesh	
	FR-1.2: A CubeSat shall have the ability to relay data from/to a ground station OR Lunar Gateway	
FR-2	Mesh network communication shall remain functional in the event of single node failure*	P-1
FR-3	Mesh network shall ensure complete coverage of the Lunar polar regions	P-1
FR-4	Mesh network shall be operational with additional CubeSats added to mesh network in the future	P-1
Operational Requirements		
O1	JPL Shall command DUST system	P1
O2	System shall survive in LLO natural environment	C4
O3	System shall provide means of deorbiting at EOL; System shall provide means of passivating the spacecraft and deorbiting in such a manner as to avoid collision with ground assets at EOL	P1
O4	System shall not have detachable parts or create any space debris during launch or normal mission operations	P1
O5	System shall be operable within regulatory board-allocated band	C3
O6	System shall wait a minimum of 30 minutes after deployment switches are activated to deploy solar panels	P1
O7	System shall wait a minimum of 45 minutes after deployment switches are activated to generate an RF signal	P1
O8	System shall transmit timestamp and relay satellite ID	P1
O9	System shall transmit telemetry data during ground link	P1

O10	System shall withstand 4 to 130 degrees Fahrenheit	C6
C&DH Requirements		
CDH-1	Sub-system shall log telemetry data	F1
CDH-2	Sub-system shall monitor voltage, current, power consumption, temperature parameters of other subsystems	F1
CDH-3	Sub-system shall process telemetry and trajectory propagation data	F1
CDH-4	Sub-system shall be capable of processing real-time operations	F1
CDH-5	Sub-system shall be able to process data at a speed of at least 1 Mbps	F1
CDH-6	Sub-system shall be able to store at least 11 MB of telemetry data	P1
CDH-7	Sub-system shall be able to indicate when CubeSat is in safe mode	O2
CDH-8	Sub-system shall be able to exit safe mode when nominal operations can be established	O2
CDH-9	Sub-system shall be able to initiate testing sequence	O2
CDH-10	Sub-system shall be able to recover from a radiation-induced memory-corruption event	O2
CDH-11	Sub-system shall be able to store up to 24 hours of health reports	O2
STR Requirements		
STR-1	Sub-system shall confine within a 6U form factor	C6
STR-2	CoG shall be X:+/- 1.5, Y:+0-+5, Z: +6-+9.5 (mm)	C6
STR-3	Sub-system shall maintain radiation accumulation under 100 rad over CubeSat lifetime for electronics	O2
STR-4	Total Mass Loss (TML) shall be < 1.0 %	P1
STR-5	Collected Volatile Condensable Material (CVCM) shall be < 0.1%	C3
STR-6	Sub-system shall be designed to withstand temperature changes of orbit	O2
STR-7	Sub-system shall weigh less than 12 kg	C6
EPS Requirements		
EPS-1	Subsystem shall be powered off prior to deployment	C6
EPS-2	Sub-system shall generate at least 15 W of power for continuous operations during safe mode	P1
EPS-3	Sub-system shall generate at least 30 W of power for continuous operations during normal mode	P1
EPS-4	Sub-system shall operate within the batteries' thermal range during nominal operations	O2

EPS-5	Sub-system shall provide regulated power output buses	F1
EPS-6	Sub-system shall include battery circuit protection to avoid cell unbalance	C6
EPS-7	Sub-system design shall not permit the ground charge circuit to energize the satellite systems, including flight computer	O2
EPS-8	Solar panels shall have sufficient backwiring to cancel out magnetic dipoles generated by the solar cells	ADCS-5
EPS-9	Sub-system shall protect the spacecraft against failures within EPS	O2
EPS-10	Sub-system shall suppress transient bus voltages and protect against bus faults	O2
EPS-11	Sub-system shall have adequate battery capacity for the pre deployment phase	O2
EPS-12	Sub-system shall use Lithium-ion 18650 type batteries	C6
COM Requirements		
COM-1	Sub-system shall operate in the S-band range	O5
COM-2	Sub-system shall maintain a 6 dB link margin	F1
COM-4	Sub-system shall relay inter-satellite communications at a separation distance of at most 1200 km	F1
COM-5	Sub-system shall have a data rate of no less than 100 kbps	F1
COM-6	Sub-system shall operate on no more than 20 W in peak operation	F1
COM-7	Sub-system shall be able to identify the source of the received signal	F1
COM-8	Sub-system shall be able to recognize and prioritize priority messages during congestion events	F1
COM-9	Sub-system shall be able to receive and process multiple received messages at the same time	F1
COM-10	Sub-system shall maintain at least one operational radio during safe mode	F1
ADCS Requirements		
ADCS-1	Sub-system shall provide pointing accuracy of 0.1 degrees required to complete ground communication link	F1
ADCS-2	Sub-system shall maintain Nadir pointing during nominal operations outside of safe mode	F1
ADCS-3	Sub-system shall provide ground station active pointing (+Z face) during link	F1

	mode	
ADCS-3	Sub-system shall maintain Sun pointing during safe mode	O2
ADCS-5	Sub-system shall supply torque commands required to counteract on-orbit disturbance torques	O2
ADCS-6	Sub-system shall provide internal torque for momentum dumping	ADCS-1
ADCS-8	Sub-system shall provide attitude determination data for CubeSat navigation	O2
ADCS-9	Sub-system propulsion shall pass NASA safety panel	C6
ADCS-10	Sub-system shall not freely spin the spacecraft about the Nadir axis	FR-1.1

Table A.2 List of mission constraints

Constraint		**Description**
C-1	Cost	SMEX - $120 million maximum
C-2	System	Homogeneous system of nodes
C-3	Regulations	FCC, COSPAR, FAA, NASA
C-4	Environment	Lunar space environment (radiation)
C-5	Interfaces	Interoperable through network of available ground and orbital assets
C-6	Structural	Artemis and Lunar Gateway

APPENDIX B: Estimation of Gateway Antenna Gain

The team estimates Gateway's antenna gain from its system requirements, specifically its data rate requirement to be able to receive 1.62 Terabits per day, which is a minimum of 18.75 kbps, from the Lunar surface and its apogee altitude of 700000 km [4], [11]. Furthermore, the team assumes that Lunar assets use a 16 dB gain antenna, from section 4.3.2, at 10 W power, twice the power DUST-Lunar provides. As the figure B.1 below shows, these assumptions yield gains of 68.82 dB for X-band or 65.75 for S-band for a typical 3 dB link margin.

				X-band	S-band
Item	**Symbol**	**Units**	**Source**	**Spacecraft to Ground**	**Spacecraft to Ground**
Frequency	f	GHz	Input Parameter	8.20	2.50
Transmitter Power (DC)	P	Watts	Input Parameter	10.00	10.00
Transmitter Power Amplifier Efficiency	η_P	--	Input Parameter	0.13	0.13
Transmitter Power (RF)	P	Watts	$P*\eta_P$	1.33	1.33
Transmitter Power (RF)	P	dBW	$10\log(P)$	1.249	1.249
Transmitter Line Loss	L_l	dB	Input Parameter	-2.000	-2.000
Peak Transmit Antenna Gain	G_{pt}	dBi	Eq. (13-18b)	16.00	8.30
Transmit Antenna Pointing Loss	L_{pt}	dB	Eq. (13-21)	-0.120	-0.120
Transmit Antenna Gain (net)	G_t	dBi	$G_{pt} + L_{pt}$	15.88	8.18
Equiv. Isotropic Radiated Power	$EIRP$	dBW	$P + L_l + G_t$	15.13	7.43
Propagation Path Length	S	km	Input Parameter	7.000E+04	7.000E+04
Space Loss	L_s	dB	Eq. (13-23a)	-207.62	-197.30
Peak Receive Antenna Gain	G_{rp}	dBi	Eq. (13-18b)	68.82	65.79
Receive Antenna Pointing Loss	L_{pr}	dB	Eq. (13-21)	-0.457	-0.043
Receive Antenna Gain (net)	G_r	dBi	$G_{rp} + L_{pr}$	68.37	65.75
Data Rate	R	bps	Input Parameter	18750000	18750000
Effective Data Rate	R	bps	*See cell	41666667	41666667
E_b/N_o (1)	E_b/N_o	dB	Eq. (13-13)	9.80	9.80
Carrier-to-Noise Density Ratio	C/N_o	dB-Hz	Eq. (13-15a)	86.00	86.00
Bit Error Rate	BER	--	Input Parameter	1.000E-07	1.000E-07
Required E_b/N_o (2)	$Req\ E_b/N_o$	dB	Fig. 13-9	5.8	5.8
Implementation Loss (3)	---	dB	Input Parameter	-1.0	-1.0
Margin	---	dB	(1) – (2) + (3) + (4)	3.000	3.000

Figure B.1 Gateway Antenna Gain

APPENDIX C: Component specs