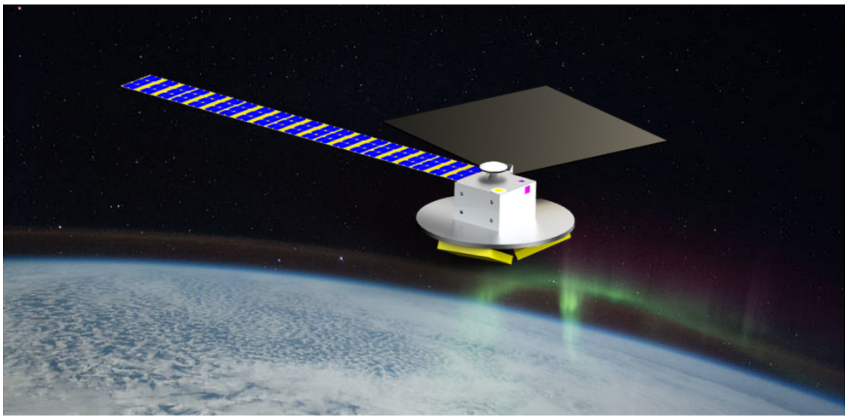
Pre-Phase A NASA Earth Venture Mission (EVM) Proposal for a Remote Sensing Satellite Constellation to Measure Soil and Surface Moisture

Fall 2021 Purdue Senior Spacecraft Design
AAE-450 Team 7



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1 Summary

The MoIST mission will deploy a constellation of twelve 45 kg satellites to Low Earth Orbit. This constellation will utilize transmitted signals from the ORBCOMM, Iridium, GPS, MUOS, and SWARM constellations as 'Signals of Opportunity' to measure surface soil moisture levels, root zone soil moisture, freeze-thaw state, and snow-water equivalent across the globe using P-band up to 1 m depth, L-band up to 5 cm depth, and VHF/I-band up to 1 m depth. The following quad chart describes the mission objectives and requirements, orbit and mission design, key spacecraft design metrics, and a preliminary program schedule.

Moisture in Soil Tracker (MoIST)

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Objective

- Understand processes that link terrestrial water, energy & carbon cycles
- Estimate global water and energy fluxes at the land surface
- Quantify net carbon flux in boreal landscapes
- Enhance weather and climate forecast skill
- Develop improved flood prediction and drought monitoring capability

Working Requirements:

- Global (+/- 50 deg latitude), 3-day revisit
- Boreal (+/- 60-70 deg latitude), 2-day revisit
- Snow Cover (+/- 70 deg latitude), 15-day revisit

Key Spacecraft Characteristics

- Mass: 45.1 kg
- Power: 60.5 W

Instrument Characteristics

- 1 Dual P-Band Antenna
- 3 L-Band Antennas

Schedule Summary

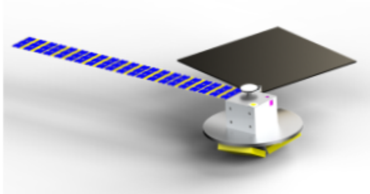
- Launch Year: 2026

Total LCC

- $134.7 (FY 2022 $M)

TRL_{inst} = 6

TRL_{Comp} = 9



Approach

- Launch constellation of receiving satellites with deployable P and L band antennas into 2 orbital planes
- Science instrument receives direct and earth-reflected signals from transmitting satellites to gather moisture data
- Notable transmitting constellations:
    - ORBCOMM, Iridium, GNSS, MUOS, SWARM

	Number of Satellites	Orbit Inclination	Orbit Altitude
Plane 1	7	80°	350 km
Plane 2	5	63.5°	550 km

Key Milestones

Project Initiation	09/21
Blue Team Review	10/21
Red Team Review	11/21
Final Presentation	12/21
Preliminary Design Review	2022
Critical Design Review	2024
Flight Readiness Review	2025
Launch	2026
Data Collection & Processing	2026
Mission Completion	2029

MoIST

Moisture in Soil Tracker

Contents

2 Science Investigation

2.1 Scientific Background

Project MoIST is a mission that plans to gather moisture measurements from around the world using Signals of Opportunity (SoOp). SoOp is a novel method of earth remote sensing that uses established signal transmitting satellites to measure many types of data about earth's surface and atmosphere. This mission is going to utilize the direct and reflected signals of L-band, P-band, and VHF/I-band transmissions to gather surface soil moisture, root zone soil moisture, freeze/thaw states, and snow water equivalent measurements. The result of this data collection could prove very valuable to crop success, drought, and flood prediction in many locations around the world. An upcoming mission with similar goals is SNoOPI, a mission to use P-band transmitting signals to measure root zone soil moisture. This proposal improves on that mission by adding L-band reflectivity measurements, and other scientific requirements.

2.2 Science Requirements

A science traceability matrix is shown in table 1. The baseline/threshold science mission is described in terms of the science requirements in the second column of table 1. This figure shows the relationship between the types of measurements to be taken, how often those measurements must be taken, and the global coverage of those measurements. For example, to meet surface soil moisture requirements we must have at least one measurement in every 10 km by 10 km square over Earth's land mass within 3 days between -50° and +50° latitudes globally. This can be interpreted similarly for each of the science requirements. This shows the connection between each of the science requirements and the instrument's capabilities.

The instrument continually outputs data for all signals in its field of view, and onboard processing will determine if that measurement is not useful towards the science requirements, such as if the measurement is over water, or outside of our latitude requirements. After that, all useful data will be downlinked and analyzed, then distributed to NASA as useful moisture measurements from around the globe. Our orbit and mission simulations prove that we generate more than sufficient data to meet these baseline science requirements. In order to prove we meet each of these science requirements, simulations used a grid imposed on the earth's surface corresponding to the defined resolution for each requirement. Our analysis shows we consistently cover >98% of the Earth's surface within each revisit requirement. More information on our mission simulations is found in section 3.2.

<table>
<tr><th>Science Objectives</th><th>Scientific Measurement Requirements</th><th>Instrument Functional Requirements</th><th>Mission Requirements</th></tr>
<tr><td rowspan="5">Understand processes that link the terrestrial water, energy and carbon cycles;

Estimate global water and energy fluxes at the land surface;

Quantify net carbon flux in boreal landscapes;

Enhance weather and climate forecast skill;

Develop improved flood prediction and drought monitoring capability.</td><td>Surface Soil Moisture (SSM):
5 cm depth
Global (+/- 50° latitude), 3-day revisit;
Boreal (+/- 50-70°latitude), 2-day revisit</td><td>L-Band Reflectivity:
10 km resolution
Incidence angle <60°</td><td rowspan="4">Data archiving and distribution (maximum 14 days from data collection to public release)

Ground station downlink

Constellation of various multi-functional RS satellites

Deployable P/L-Band antenna

Appropriate SoOp sources in L-Band, P-Band, and VHF</td></tr>
<tr><td>Root Zone Soil Moisture (RZSM):
1m depth
Global (+/- 50° latitude), 3-day revisit;
Boreal (+/- 50-70° latitude), 2-day revisit</td><td>P-Band or VHF:
40 km resolution
Incidence angle <60°</td></tr>
<tr><td>Freeze-Thaw State (F/T):
3 day revisit, region subject to freezing (+/- 60° latitude)</td><td>L-Band Reflectivity:
Incidence angle <60°
3 km resolution</td></tr>
<tr><td>Snow-Water Equivalent (SWE) (L- and P-Band Phases):
15 day repeat interval
85% global snow coverage (+/- 70° latitude)</td><td>L-Band Reflectivity:
Repetitive coverage - specular point ground tracks within 100-m
100 km spatial sampling
P-Band Reflectivity:
Within +/- 60° latitude at 100 km spatial sampling</td></tr>
<tr><td>Observation over a minimum of three annual cycles</td><td>Minimum three-year life</td><td>Three year baseline mission</td></tr>
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Table 1: Science Requirements Overview

3 Science Implementation

3.1 Instrumentation

The science instrument is a modified version of that which will fly on SNoOPI, and is capable of taking a measurement once every second for every transmitting signal in its view. The instrument weighs 7kg, uses 25 W while operating and 1 W on standby mode. The dimension of the instrument is 30 x 10 x 10 cm (3U). These instrument specifications and others are shown in figure 1.

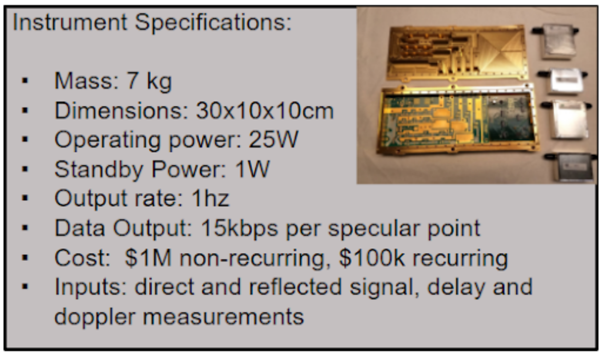


Figure 1: SNoOPI Derived Instrument Summary

In order for the instrument to take a measurement, both the direct signal and earth-reflected signal from a transmitter must be within the satellites' field of view. There is more information on this geometry in figure 2. The instrument's observations are then treated as specular points on the earth's surface, located where the signal reflects off the earth. This corresponds to the resolution requirements by ensuring there is at least one specular point measurement within any size grid resolution on the earth's surface. The instrument generates 15 kbits of data per specular point per second, resulting in 21.1 GB of data generated per day according to our simulations. Data generation is discussed in greater detail in section 3.4.

	Beam width	Mass	Dimensions
P-band Antenna	120°	2.3 kg	1X2U stowed (1.5m deployed)
L-band direct	125°	0.224 kg	3.5 inch diameter
L-band reflected	42°	0.75 kg	300x300x40mm

Table 2: Antenna Specifications Summary

Along with the instrument, there is one P/I-band deployable antenna developed by MMA Design, dimensioned 1X2U when stowed and weighs 2.3 kg. When deployed, the antenna forms a square of diagonal length 1.5 meters. The antenna has a 120° beamwidth for both direct and reflected signals. There are 3 reflected GNSS L-band antennas from European Antennas, dimensioned 300 x 300 x 40 mm and weighing 0.75 kg each. These reflected antennas have a 42° beamwidth. There is one GNSS Antcomm 3G1215A-MTR-5 direct L-Band antenna with diameter 3.5 in and weighing 224 grams. This direct antenna has a 125° beamwidth. These specifications are summarized in table 2.

3.2 Science Mission Profile

The science mission will launch in 2026, be operational for at least 3 years, and have a 25 year deorbit requirement. All final data products must be released to the public within 14 days of collection. Our mission uses 12 satellites split in two orbital planes: one at 350 km and 80° inclination, the other at 550 km and 63.5° inclination. There are 7 satellites in the 350km orbit and 5 satellites in the 550km orbit. This uneven split is the most effective distribution for meeting our coverage and revisit for each requirement. Each plane also offers unique geometries that are conducive to meeting particular science requirements. For example, the 80° inclination orbit offers better high-latitude coverage, while the 63.5° inclination orbit offers better equatorial region coverage.

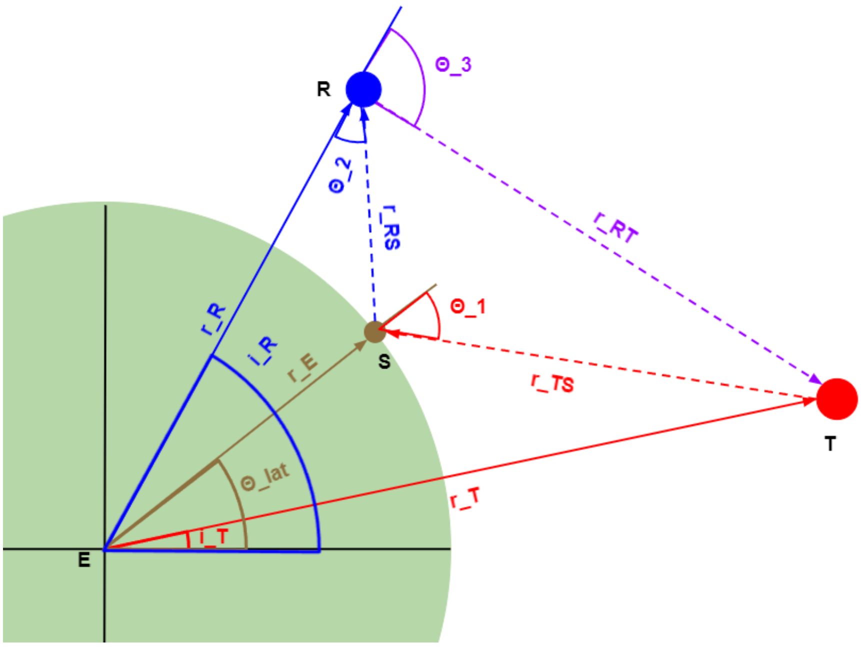


Figure 2: Representation of the Geometry of the MoIST Receiver (R) Detecting the Specular Point (S) whilst Maintaining a Direct Signal with the Transmitter (T).

These orbits were chosen based on a first-order analysis of the edge-case geometric constraints of the problem, primarily the relationship between orbit inclination, altitude, and visibility of high Earth latitudes. A diagram of the geometric constraints of the mission is more clearly shown in figure 2. These three elements are traded off based on the angular constraints on the direct and reflected signal angles relative to the P-band and L-band antenna FOV. This resulted in a small trade-space of orbit altitudes and inclinations that would provide the most visibility of transmitting satellites and upper earth latitudes. The relevant trade-spaces are shown in figures 3-5. These trade spaces demonstrate that a single orbit plane will not optimally satisfy coverage requirements, and so two different orbital planes were selected to optimize coverage and revisit time. The assumptions made in this analysis includes just one reflected L-band antenna, rather than the three we have proposed, that the spacecraft is always nadir-pointing, and using a spherical earth approximation. This analysis was validated after higher-fidelity orbit simulations showed that coverage requirements were met utilizing the selected orbital planes.

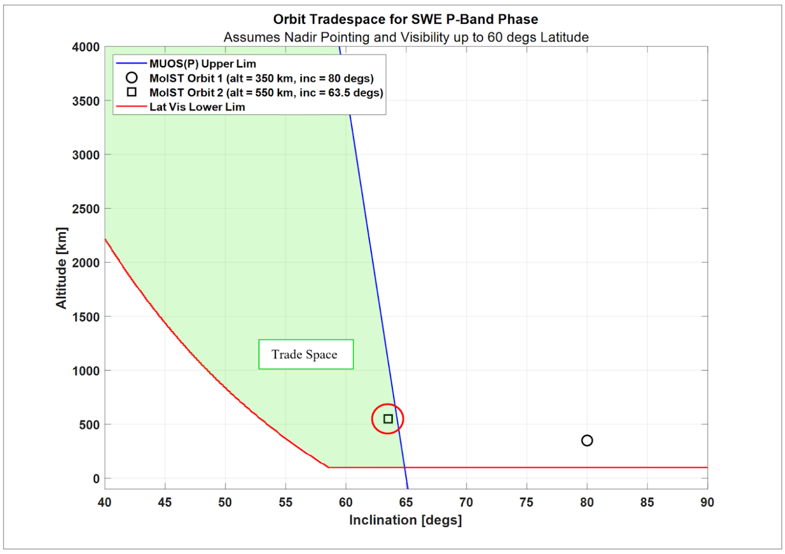


Figure 3: Region for Which High Latitudes of the Earth can be Observed Using MUOS

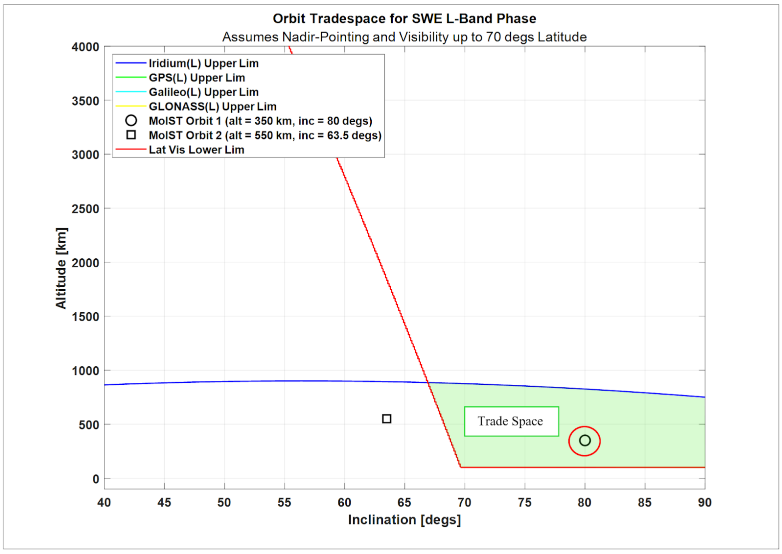


Figure 4: Region for Which High Latitudes of the Earth can be Observed Using L-Band Constellations

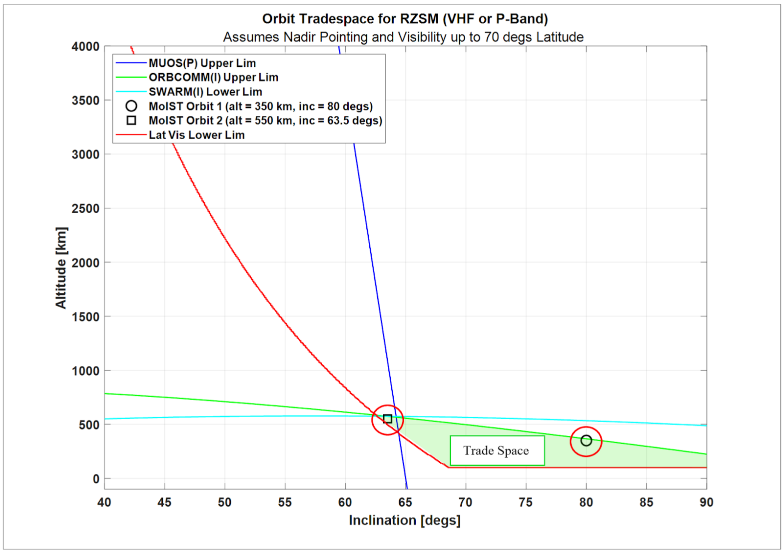


Figure 5: Region for Which High Latitudes of the Earth can be Observed Using P and VHF/I-Band Constellations

A summary of these results is provided in table 3, which clearly demonstrates the need for two orbital planes. We note then that the two planes have been chosen to maximize the amount of transmitter constellations visible at the high latitudes, which should in theory allow us to improve our coverage.

Transmitter Constellations		
Constellation	**Altitude Range [km]**	**Inclination Range [deg]**
ORBCOMM	500 - 150	65 - 90
Iridium	800 - 150	67 - 90
GPS	23,000 - 150	40 - 90
MUOS	17,000 - 500	40 - 65
SWARM	500 - 150	63 - 90
Receiver Orbit Constellations		
Plane 1	350	80
Plane 2	550	63.5
2 Orbit Plane Solution • First Orbit plane covers all L-band requirements and **some** P-band • The second orbit plane allows for **upper latitude** P-band coverage through MUOS		

Table 3: Summary of figures 3-5 Used to Determine MoIST Orbital Planes

Constellation Performance Characterization

Though the first order analysis was useful in determining the available trade space and coverage, the science requirements also include revisit requirements which must be met for mission success. In order to estimate how well the designed constellation meets these requirements, the team developed code which simulates the launched constellation as it gathers science data. The outline of the developed code is shown in figure 6. The first step of the developed code is to pre-propagate the transmitter and MoIST constellations in GMAT (General Mission Analysis Tool). The tool is chosen due to its ability to include all pertinent orbital perturbations and the team members' experience with it. GMAT is used to save latitude and longitude every 15 seconds of all these spacecraft into files. These files are then ingested into a Python script which interpolates these down to 1 second and then uses the Alhazen-Ptolemy problem solution to find the specular point for each set of transmitter, receiver and time. Once we have solved for all available specular points for a given amount of time, the median revisit and other relevant statistics are calculated.

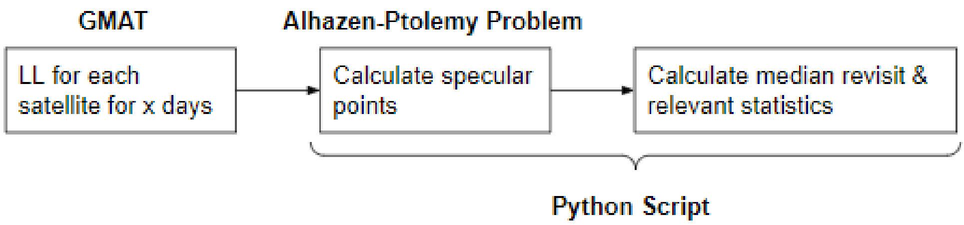


Figure 6: Outline of the Code Developed to Verify the Revisit Requirements

The GMAT portion of the code base first downloads TLEs from Celestrack for all the transmitters considered. These TLE's are all propagated to a single starting epoch to ensure the 319 transmitter satellites from the ORBCOMM, Iridium, GPS, MUOS, and SWARM constellations are all synced up. A python script is utilized to translate this data into an input file for GMAT. For the MoIST receivers, no TLEs are used as we can simply generate the required GMAT input file directly by applying the previously mentioned orbital elements. GMAT is then run using these two separate input files and outputs two separate files (one for transmitters and another for the receivers). Finally, we note that the GMAT upper and lower time step bounds were set to 15 seconds to reduce the run time of the GMAT scripts. The data will be interpolated in the pre-processing python script to ensure accurate results. Generating latitude and longitude information every second for all the required satellites takes up 44 GB of data, which is too large for the Purdue AAE servers. Instead, saving data every 15 seconds was chosen as the ground track is linear at this resolution and the data load is reduced by 15x.

Once these files with the latitude and longitude points are saved, they can be read into a Python script which performs the remainder of the calculations. The first step that this code performs is to ingest the transmitter and receiver latitude and longitude and interpolate it down to 1 second. Linear interpolation is used for this step as the groundtracks are approximately linear for these small time steps. By utilizing an analytical solution to the Alhazen-Ptolemy problem, we are able to find the location of the specular point for all transmitter/receiver pairs at each time step. The Alhazen-Ptolemy problem describes the point of reflection off of a circle between an observer and a source as seen in figure 7. In order to use this, the team assumed that each reflection is perfect and that the Earth can be modelled as a sphere. A full derivation for the solution to the Alhazen-Ptolemy problem can be found in [1]. This type of problem has been used in the past by a team studying the Saturnian moon Titan [2], validating our approach for a 2nd order algorithm.

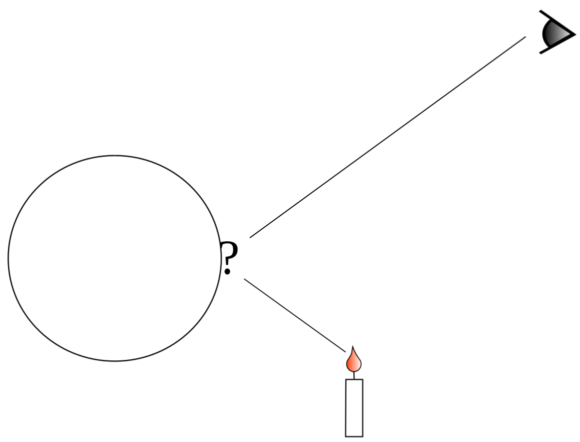


Figure 7: Visualization of a Reflection off a Sphere Used in the Alhazen-Ptolemy Problem

In short, finding the specular point requires the definition of three vectors with the origin set at the center of the Earth. The vector S defines the location of the observer, the vector L defines the location of the source and the vector N defines the location of the specular point. The specular point vector can then be defined as a linear combination of the other two vectors: $N = xS + yL$. All that is left is to find the values of x and y which allow us to fully determine the value of N. To do this, we begin by finding the positive roots of y through the equations below:

$$a = S \cdot S,\ b = S \cdot L,\ c = L \cdot L \text{ [1]}$$
$$r(y) = 4c(ac - b^2)y^4 - 4(ac - b^2)y^3 + (a + 2b + c - 4ac)y^2 + 2(a - b)y + a - 1 = 0 \text{ [1]}$$

Next, we use these positive roots to find the positive values of x through the following equation:

$$x = \frac{-2cy^2+y+1}{2by+1} \text{ [1]}$$

Once these specular points have been found, we must ensure that they can be seen by the receiver. As was previously noted, the antennas the team have available are not omni-directional, and so it is feasible that a reflection exists but it cannot be seen by the MoIST spacecraft. On top of seeing the specular point, the MoIST spacecraft must also be able to see the transmitter from which the signal originally originated. Finally, in order to ensure that the point of reflection is mostly specular (instead of diffuse), an incidence angle requirement is applied. These three angles are shown in figure 2 and defined in the set of equations below. θ_1 is the incidence angle, θ_2 is the angle that the specular point makes with the MoIST spacecraft and nadir, and θ_3 is the angle that the transmitter makes with the receiver. The required maximum angles are determined by the antennas that were selected to complete this mission.

The angles shown are determined through simple trigonometry. Any specular point which does not meet the minimum angle requirement is discarded.

$$cos\theta_1 = \frac{r_{RS} \cdot r_E}{||r_{RS}|| \cdot ||r_E||},\ cos\theta_2 = \frac{r_{RS} \cdot r_R}{||r_{RS}|| \cdot ||r_R||},\ cos\theta_3 = \frac{r_{RT} \cdot r_R}{||r_{RT}|| \cdot ||r_R||}$$

Finally then, we are left with a simple list of specular points and their associated times. On its own though, this does not tell us much, so we must perform some additional calculations to ensure that the constellation meets the revisit requirements. The first step is to map the observed specular points onto the appropriate spatial grid, which is determined by the science requirements. In order to do this, we first generate a grid of the Earth made up of squares which have sides defined by the requirements (e.g. Freeze-Thaw has 3 km x 3 km grid elements). Then, a nearest neighbor approximation is used to determine in which grid element each specular point belongs. Once we have generated these, we can group the specular points into their appropriate grids and calculate revisit from there.

To calculate revisit, the specular points within the grid are sorted in time. Once this is done, it is simple to use Python Pandas to calculate the time difference between subsequent specular points. With this time difference, the median revisit for a single grid element can be calculated using Pandas. Finally, the code calculates two numbers per science requirement. The first is the ninety-ninth percentile of the revisit for the whole Earth. The second is the percentage of grid elements which meet (or exceed) the revisit requirement specified in the science requirements.

This type of calculation is performed for all the science requirements except for the snow-water equivalence science measurement in the L-band frequency range. This requirement states that subsequent science measurements must be made 100 meters away from the original measurement. In order to calculate this science requirement, similar steps are taken to group each specular point measurement into individual grid elements. For each grid element then, the code iterates through each specular point and uses a nearest neighbor approximation in order to determine whether there is at least one point within 100 meters. The haversine formula, which is provided below, is used in order to calculate the distance.

$$d = 2rarcsin(\sqrt{sin^2(\frac{\phi_2 - \phi_1}{2}) + cos(\phi_1)cos(\phi_2)sin^2(\frac{\lambda_2 - \lambda_1}{2})})\ [3]$$

Once at least one specular point has been found which has another point which is 100 meters away (or less) that grid element is counted as revisited within the simulated time span. Therefore, no revisit figure is given for this requirement and instead only a coverage figure is provided.

Finally it is noted that the simulation treats the Earth as if it is fully covered by land, and does not remove points which are over water. The reason for this is two-fold.

The first reason is that this is computationally expensive, which makes it difficult to test outside of the live environment. As testing in the live environment is more difficult than in the test environment, developing this functionality would have been difficult. The second reason is that the problem seems to be longitude independent. This means that the distribution of specular points is not a function of longitude (though it certainly is a function of latitude). Removing points over water would be effective if this were not the case. An additional reason for not developing this functionality is that the chosen constellation works well even when we interpret the Earth as being fully covered by land. Given this, there is no benefit to developing code which can remove specular points over land, so the code was not developed. Note that this is not necessarily the case for the real satellites, but this should not affect the approximation of coverage and revisit.

	# of Satellites	Orbit Inclination	Orbit Altitude
Plane 1	6	80°	350 km
Plane 2	4	63.5°	550 km

Table 4: Current Constellation Definition

Now that the code can determine the relevant statistics, we are ready to run it for our chosen constellation. As previously mentioned, the orbital elements for the two planes were chosen using a first order analysis, but are repeated in table 4. Note that we have 10 satellites total, but they are not distributed evenly between the two planes. We chose this set-up as we have fast revisit requirements, and so we wanted to have more satellites at the lower orbit due to its lower period. Additionally, as shown in figure 4, the lower orbital plane is used to cover the higher latitudes of the specular points in the L-band. As this includes two of our most stringent requirements (Freeze-Thaw and Snow Water Equivalent), we wanted to increase the number of satellites in this orbit to increase the number of specular points. To maximize our chances of meeting the requirements we consider all relevant, currently launched constellations. Only the MUOS constellation transmits in P-band. For the VHF/I-band, that is SWARM and ORBCOMM. For L-Band, that is Iridum, GPS, Galileo, and GLONASS. We then run this scenario for 15 days at a 1 second time step.

When running this set-up all of the science requirements are met with the exception of the Freeze/Thaw state. It is clear that the current constellation design does not meet the revisit requirements for this due to the small grid size of 3 km. We noticed that only about half of the Earth gets covered at a 3 day revisit. In order to solve this issue, three solutions are considered. The first solution is to simply increase the number of satellites in each orbital plane until the revisit is less than 3 days. The next approach is to change the orbit. This is difficult as we did not develop a simple code which could determine the revisit requirements without running the simulation. Another solution presents itself if we consider the actual effects of increasing the number of satellites. Practically speaking, adding more satellites simply increases the number of antennas which are capable of detecting the reflected signals. A similar effect can be attained by increasing the number of antennas per satellite (specifically L-band). When doing this, we must ensure that the antennas are mounted in such a way that the antenna view cones are tangential to each other on the surface of the Earth. For three antennas, they must be 28.9° off nadir and rotated 120° apart.

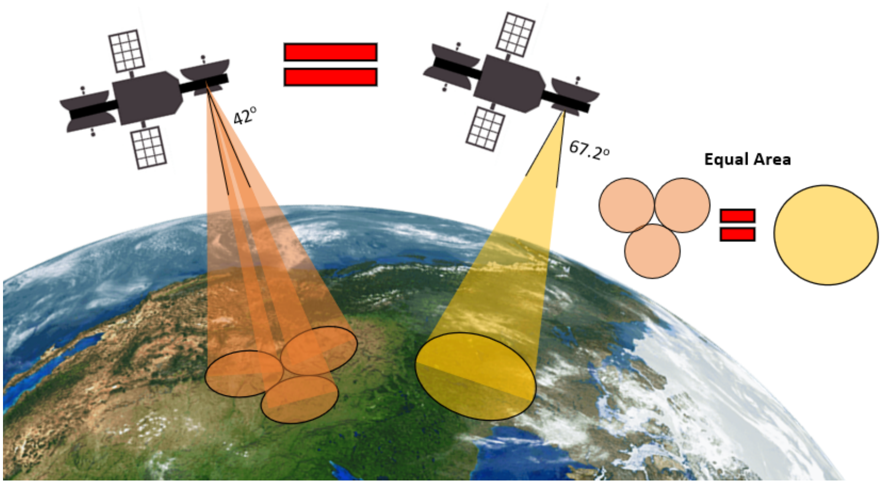


Figure 8: How Adding More Antennas can Help Improve the Revisit of the Freeze-Thaw Science Measurement (red). How These Additional L-band Antennas are Modeled in the Code (yellow)

Unfortunately, actually determining whether a specular point falls within one of the view cones of the antennas on the satellites is significantly more complicated than doing so for a single antenna. Therefore, an approximation was made in order to estimate the size of the antenna view cone on the ground. We did this by summing the area of the field of view of each individual antenna to get an approximate summed field of view. The half angle of the associated cone can then be easily calculated and plugged into the simulation that was developed. Figure 8 demonstrates this, and the updated results of the simulation are shown in table 5.

Requirement	Expected Revisit (days)	Resolution	99%-ile **Median** Revisit (days)	Coverage at Expected **Median** Revisit
SSM Global (+/- 50°) (L-Band)	3	10 km	0.3690	100%
SSM Boreal (+/- 50-70°) (L-Band)	2	10 km	0.4467	>99.99%
RZSM Global (+/- 50°) (P-Band + VHF)	3	40 km	0.0788	100%
RZSM Boreal (+/- 50-70°) (P-Band + VHF)	2	40 km	0.0959	100%
F/T (+/- 60°) (L-Band)	3	3 km	3.1048	98.83%
SWE (+/- 60°) (P-Band)	15	100 km	0.4432	100%
SWE (+/- 70°) (L-Band)	15	100 km	<15	100%

Table 5: Simulation Results for 15 Days, 1 Second Time Step, 2 Orbital Planes, All Transmitters, Full MoIST Constellation and 3 L-band Antennas per Satellite

Finally then we can see that the constellation chosen can meet all the science requirements within 98% coverage. In order to improve the resiliency of our constellation, we will increase the number of satellites per plane by one. In our case then, we can see that we can meet the science requirements even if we were to lose two satellites. More discussion on this is provided in the risk section.

	# of Satellites	Orbit Inclination	Orbit Altitude
Plane 1	6+1	80°	350 km
Plane 2	4+1	63.5°	550 km

Table 6: Constellation Definition with Increased Resiliency

3.3 Data Sufficiency

The data to be collected will sufficiently measure many types of moisture levels around the globe using surface soil moisture, root-zone soil moisture, snow-water equivalent, and freeze-thaw measurements. The data requirements are directly derived from these science requirements and the provided metric that the science instrument generates 15 kbps of data. Using the resolution, revisit time, and global coverage of each requirement, an analysis was done to determine the minimum amount of data to be generated per day to meet the requirements. The results of this show 21.1GB of data generated per day is the theoretical minimum assuming perfectly efficient coverage. Realistically we are going to generate more data than this as coverage is not perfectly efficient and many regions of the earth get covered faster due to transmitting signal geometries. Our simulations show that with no data sorting, using every transmitted signal available, this mission would generate up to 200GB of data per day. However, this would be impossible to downlink every day. To solve this, we're going to simulate the mission ahead of time and tell each satellite when to efficiently take measurements. This is elaborated on in section 4.4.3. This will reduce the data generated daily from 200GB to 21.1GB, and this is the required amount of data to be handled by ground station processing.

3.4 Data Plans

The plan for data retrieval is based on a 14 day requirement for release of all science data to NASA. In order for the science instrument to make a measurement, it needs both the direct and reflected signal, as well as the delay and doppler measurements. These delay and doppler measurements must be transmitted to the satellite to make any measurement. In order to meet requirements effectively, mission operations will simulate the satellite orbits and their measurement locations, and determine which specular points are needed to hit the requirements efficiently. Only those delay and doppler measurements will be sent to each satellite each day to measure data effectively. While mission operations would do their best to determine even coverage for each satellite, it is likely there will not be a perfect spread of data between every satellite. Because of this, the communications subsystem is built with sufficient margins to ensure downlink capabilities for an uneven spread. Detailed information about our spacecraft's command and data handling subsystem is found in section 4.4.3 . The data is being transmitted through Amazon Web Services wideband reserved. Amazon's service is significantly cheaper than using the Near Earth Network and use of their cloud processing would allow extremely quick turnaround from satellite downlink to NASA.

4 Mission Implementation

4.1 General Requirements and Mission Traceability

Requirement	Factors which typically impact the requirement	Requirements
FUNCTIONAL (HOW WELL IT MUST PERFORM)		
Performance	Primary objective, payload size, orbit, pointing, antenna specifications	**Primary Objective:** Measuring soil moisture and snow depth **Science Instrument Payload size:** 3Us, 7 kg, 25 W operating / 1 W standby, 15 kbps/specular point **Pointing accuracy:** 1° **P-band antenna:** Stowed as 2U, 2.3 kg, extends to 1.5 m length, two cones +/60° **L-band antenna:** 300x300x40 mm, 0.75 kg, one cone +/ 21° **GNSS antenna:** +/ 62.5° Each satellite will fit within the volume specified by its respective launch vehicle fairing.
Coverage & Revisit	Orbit, swath latitude width, amount of global snow coverage/regions that are subject to freezing, number of satellites, scheduling	**Surface Soil Moisture:** L-band, +/- 50° latitude 3 day revisit, +/50-70° latitude 2 day revisit, 10 km resolution **Root Zone Soil Moisture:** P-band & VHF, +/- 50° latitude 3 day revisit, +/- 50-70 ° latitude 2 day revisit **Freeze-Thaw State:** L-band, +/- 60° latitude 3 day revisit **Snow Water Equivalent:** L-band, +/- 70° latitude 15 day revisit **Snow Water Equivalent:** P-band, +/- 60° latitude 15 day revisit Orbit planes shall be maintained within 340-350 km altitudes and 80° or within 540-550 km altitudes and 63.5°. There shall be at least 4 satellites in the 550 km orbital plane, and at least 6 satellites in the 350 km orbital plane. Orbits shall be circular.
Interpretation	Cloud cover, image quality, a manual or automated interpretation	N/A

Responsiveness (Timeliness)	Communications architecture, processing delays, operations, type of data downlink (store and dump versus direct downlink), operations	Make mission data available to the public within 14 days of collection.
Accuracy	Orbit, specular point	**Surface Soil Moisture:** 10km spatial sampling (L-band) **Root Zone Soil Moisture:** 40km spatial sampling (P-band & VHF) **Freeze-Thaw State:** 3km spatial sampling (L-band) **Snow Water Equivalent:** 100 km spatial sampling, 100m between subsequent grountracks (L-band) **Snow Water Equivalent:** 100km spatial sampling (P-band)
Secondary Mission		N/A
OPERATIONAL (HOW IT IS TO BE USED)		
Commanding	Who will do commanding tasking from the field, need for real-time schedule changes	Entire constellation will be commanded by designated primary or back-up Amazon Web Services ground stations .
Mission Design Life	Duration of need, level of redundancy, altitude	The mission will have a life of minimum 3 years.
System Availability	Level of redundancy, weather, interference	>98% availability for continuous Earth coverage.
Survivability	Orbit, hardening, electronics	Constellation must be able to survive designed orbit space environment for required mission life. 12 hour recovery period in the event of a tumble.
Data Collection and Distribution	Communications architecture; user needs; ancillary communication channels	Can store enough science data and relay to ground stations within latency requirements. Mission data will be fully available to the public within six months following its collection. Data will be delivered to a NASA assigned data center, e.g., one or more of the Earth Observing System Data and Information System (EOSDIS) Distributed Active Archive Centers (DAACs).

Data Content, Form, and Format	User needs, level and place of processing, antenna bands used, compressed versus uncompressed data, payload	Data shall include location and extent in lat/long for % water content for SSM, RZSM, F/T, SWE.
User Equipment	Mass, size, power; existing equipment or new, user interface	Ground station needs to be taken care of by AWS services. MoIST team will require a basic office space in which to access the AWS interface.
CONSTRAINTS (LIMITATIONS IMPOSED ON SYSTEM)		
Science Hardware	AAE450 Instrument Specifciations	LNFE (receiver front end): - 1 hour time to initialize (cold start). - Operating temperature -10 to 40 °C. - Only can track specular points in view of the antennas view cones. DBE (receiver back end): - Data (output) 15 kbits/s - per each SP - Required inputs: reflected/direct signal, time/frequency reference (1pps/10MHz), Delay/Doppler for each specular point.
Cost	Number of spacecraft, size and complexity, orbit	The PI-Managed Mission Cost (PIMMC), including all mission phases, is capped at the AO Cost Cap of $190M FY 2022 dollars.
Schedule	Technical readiness, program size	Proposals shall plan for a launch readiness date no later than 30 November 2026 (or five years after the contract is in place, whichever is later).
Risk	Primary and Seconday customers, schedule, cost	Probability of success shall be >95%.

Regulations	Law and policy	NASA EVM mission rules and regulations detailed in EVM shall be followed. Only these Launch Vehicle Options shall be considered: - EVM-3 investigations must be launched as primary payload on a single expendable launch vehicle (ELV) that NASA will provide as Government Furnished Equipment (GFE) at a cost of $61M, which is to be reflected against the PIMMC. - An EVM-3 Small Satellite or CubeSat investigation may be launched as the primary payload utilizing Venture Class Launch Services (VCLS) that NASA will provide as Government Furnished Equipment (GFE) at a cost of $25M per launch, which is to be reflected against the PIMMC. NASA will provide no more than two launches.
Political	Satellite lifetimes and conjunction analysis (likelihood of collision)	- All MoIST satellites shall deorbit within 25 years and will have a less than 1% chance of collision with debris or other satellites. - Earth-orbiting spacecraft must be passivated at the end of the mission prior to disposal and be deorbited within 25 years of end-of-mission (or 30 years after launch, whichever comes first), or be placed in a disposal orbit above 2000 km but not within 300 km of geosynchronous orbit (GEO).
Environment	Orbit, lifetime, weather	Satellites shall be able to withstand natural space environment defined with our orbit.
Interfaces	Level of data being transmitted	Communications shall relay level 2 data to ground stations.
Development Constraints	Primary and secondary customers, schedule, cost	Within bounds of NASA EVM, No unique operations people at data distribution nodes.

Table 7: Top Level Mission Requirements

4.2 Mission Concept Summary

The MoIST mission will use Signals of Opportunity to measure surface soil moisture levels, root zone soil moisture, freeze-thaw state, and snow-water equivalent using P-band up to 1 m depth, L-band up to 5 cm depth, and VHF/I-band up to 1 m depth. MoIST will have a constellation of 12 satellites, all equipped with L, and P and VHF/I-band antennas. The constellation is split into 2 circular orbit planes, with 7 satellites at 80° inclination and 350 km altitude and 5 satellites at 63.5° inclination and 550 km altitude. Two dedicated launches will be used for MoIST: The 350 km satellites will be launched from Vandenberg Space Force Base, CA on the Northrop Grumman Minotaur 1, and the 550 km satellites will be launched from the Reagan Test Site, Marshall Islands on a Northrop Grumman Pegasus XL. The mission is set to last for 3 years, downlinking over 25GB of science data per day to the Stockholm, Sydney, Oregon, Ohio, and Cape Town Amazon Web Services ground stations. With this constellation, valuable moisture data from around the world will be collected and relayed to NASA for analysis and distribution.

4.3 Launch Services & Compatibility

Launch Selection

Our team has opted to use AO-provided commercial FAA Licensed Launch Services for this mission. LLS has a higher cost ($14 M per launch) [4] than the SpaceX SmallSat Rideshare program ($1.8 M per launch) [5], but poses substantially lower programmatic risks as it allows our spacecraft to be the primary payload and therefore fully determine orbit and launch schedule. As shown in figures 9 and 10, we used the NASA Launch Services Vehicle Performance Analysis Tool [6], and found that the Minotaur 1 (formerly Taurus 3210) is suitable for our 350 km orbit plane, and the Pegasus (XL) is suitable for our 550 km orbit plane.

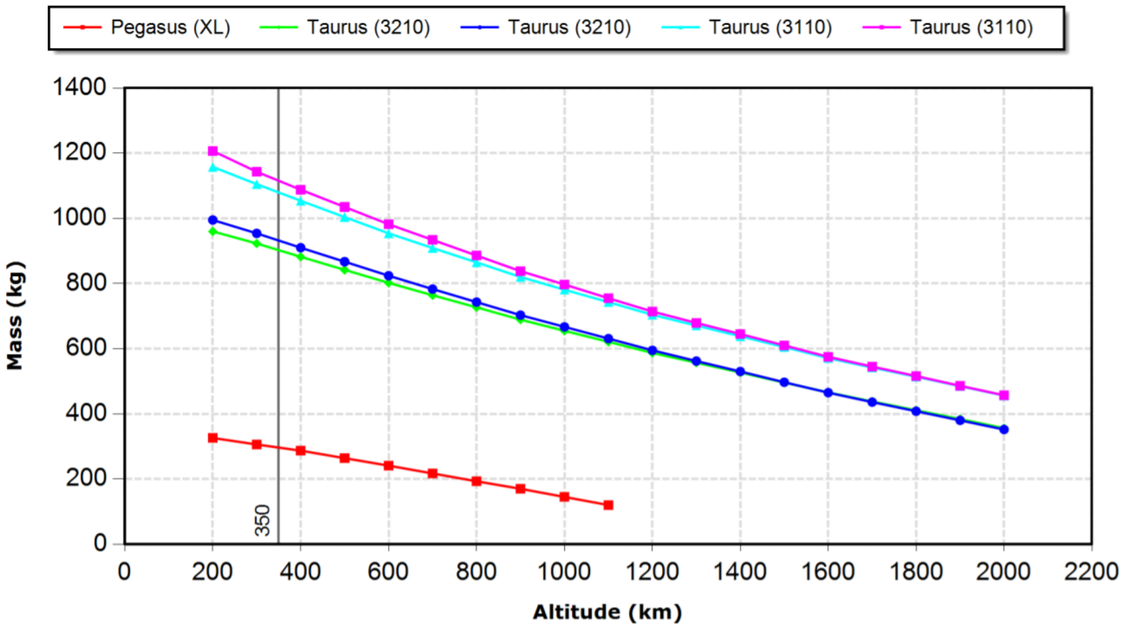


Figure 9: NASA Vehicle Performance Analysis (350 km Orbit Plane)

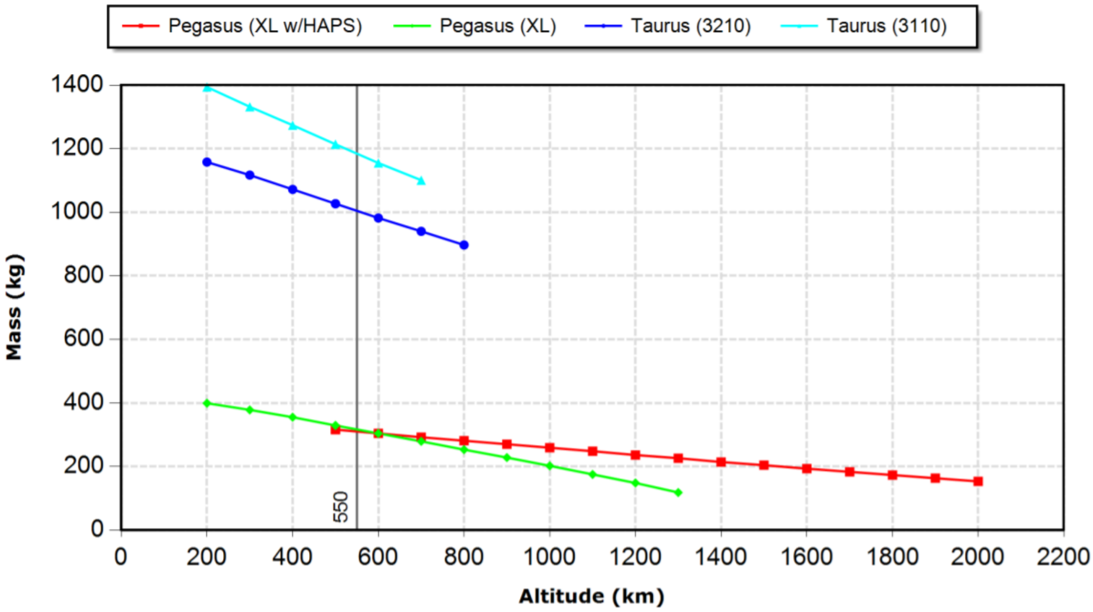


Figure 10: NASA Vehicle Performance Analysis (550 km Orbit Plane)

At our 350 km, 80° inclination orbit, the Minotaur 1 has a payload capability of 795 kg, exceeding the mass of this constellation of 7 satellites (409.95 kg) with a 25% contingency as per the NASA GSFC golden rules [7]. The Performance Analysis tool identified Vandenberg Space Force Base, California as a suitable site for this launch. The Minotaur I Fairing Dynamic Envelope has a diameter of 1.55 m, and a height of 3.84 m. In figure 11, our stowed satellites are shown in a launch configuration inside this fairing envelope. The total stack height for 7 stowed satellites (including 10 cm for the rod/ HDRM system between each satellite) is 2.875 m.

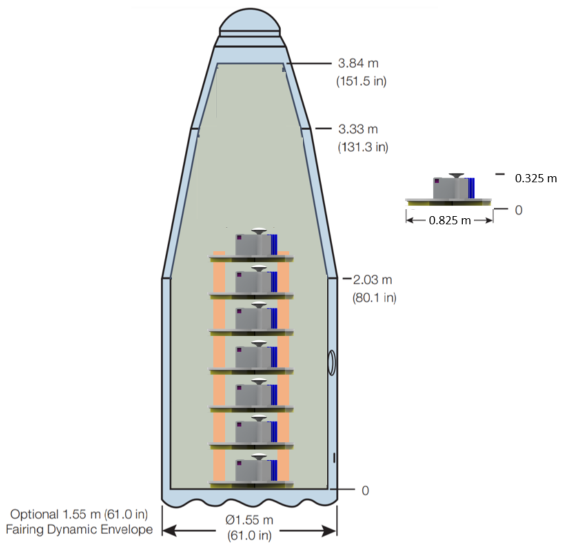


Figure 11: Minotaur Payload Configuration

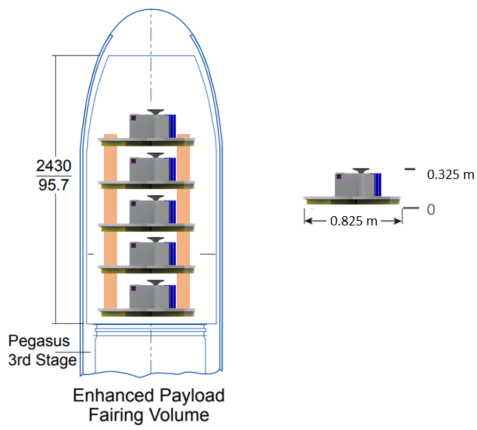


Figure 12: Pegasus XL Payload Configuration

At our 550 km, 63.5° inclination orbit, the Pegasus XL has a payload capability of 315 kg, which exceeds the mass of this constellation of 5 satellites (292.82 kg) with a 25% contingency. Contingencies are discussed in greater detail in section 4.6. The Performance Analysis tool identified the Reagan Test Site, Marshall Islands as a suitable site for this launch. The Pegasus XL Enhanced Payload Fairing Volume has a diameter of 1.3 m, and a height of 2.43 m. In figure 12, our stowed satellites are shown in a launch configuration inside this fairing envelope. The total stack height for 5 stowed satellites (including 10 cm for the rod/HDRM system between each satellite) is 2.025 m.

Vibrational Load Analysis

To calculate if our spacecraft busses would survive the vibrational loads of their respective launch vehicles, we calculated the fundamental frequency of a spacecraft bus and compared that to the minimum required fundamental frequency for the selected launch vehicles. Minotaur has a requirement of >12 Hz, and Pegasus has a >20 Hz requirement. Our chassis, made of Al-6061, has a Modulus of Elasticity of 68.9 GPa. We determined our spacecraft area moment of inertia to be 0.000148 m^4. Using these values as well as our spacecraft mass and length, we calculated our spacecraft chassis fundamental frequency to be 162.3 Hz using the equation below.

$$f_n = \frac{1}{2\pi}\sqrt{\frac{3EI}{ML^3}} \text{ [8]}$$

More analysis will need to be done in later mission phases, such as a structural analysis of the entire stack.

4.4 Flight System Capabilities

4.4.1 Spacecraft Parameters

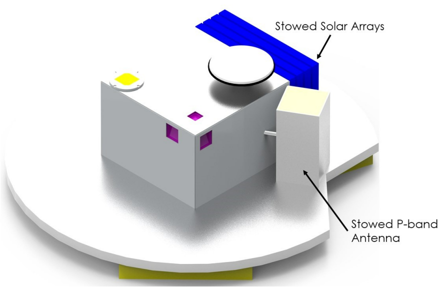


Figure 13: Isometric View of Stowed Spacecraft

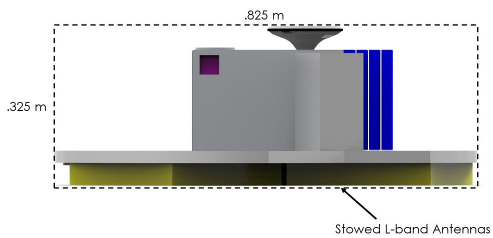


Figure 14: Profile View of Stowed Spacecraft

The stowed view of each spacecraft is shown in figures 13 and 14. For launch, each L-band antenna will be stowed flush with the underside of the aluminum table, the stowed P-band antenna will be contained in a 2-Unit (2U) module, and the solar array will be stowed as shown. By stowing these components, each spacecraft will be stacked and interconnected via 4 cylindrical rods which will be released by the NEA Model 9100 Hold Down & Release Mechanism [9].

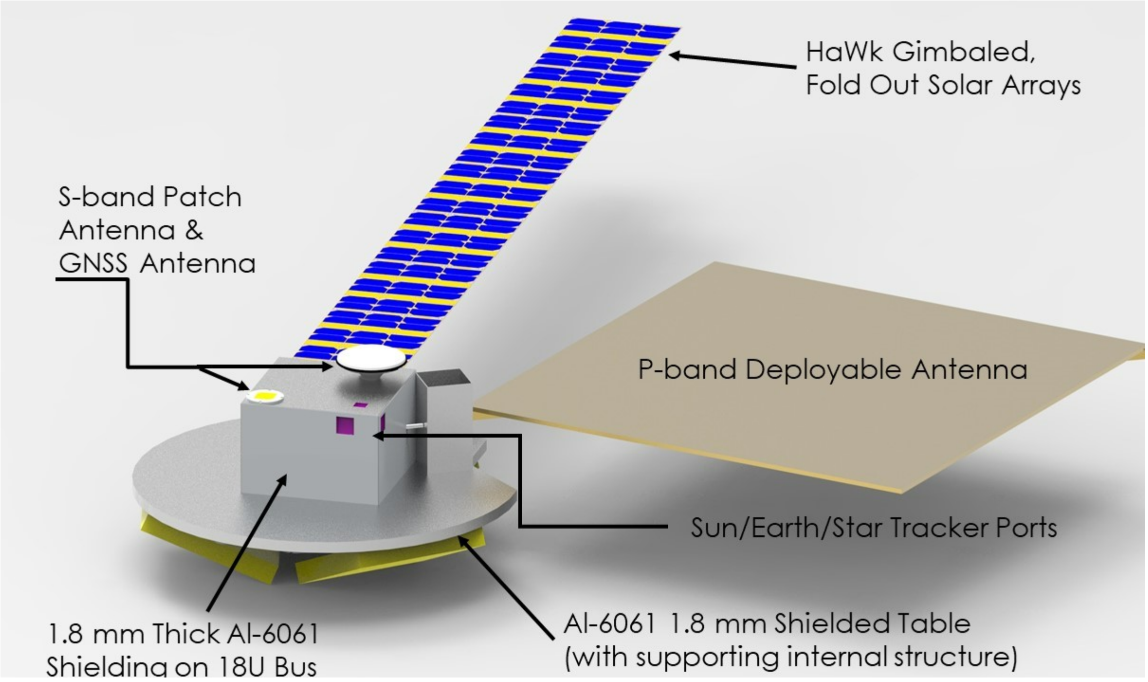


Figure 15: Isometric View of Deployed Spacecraft

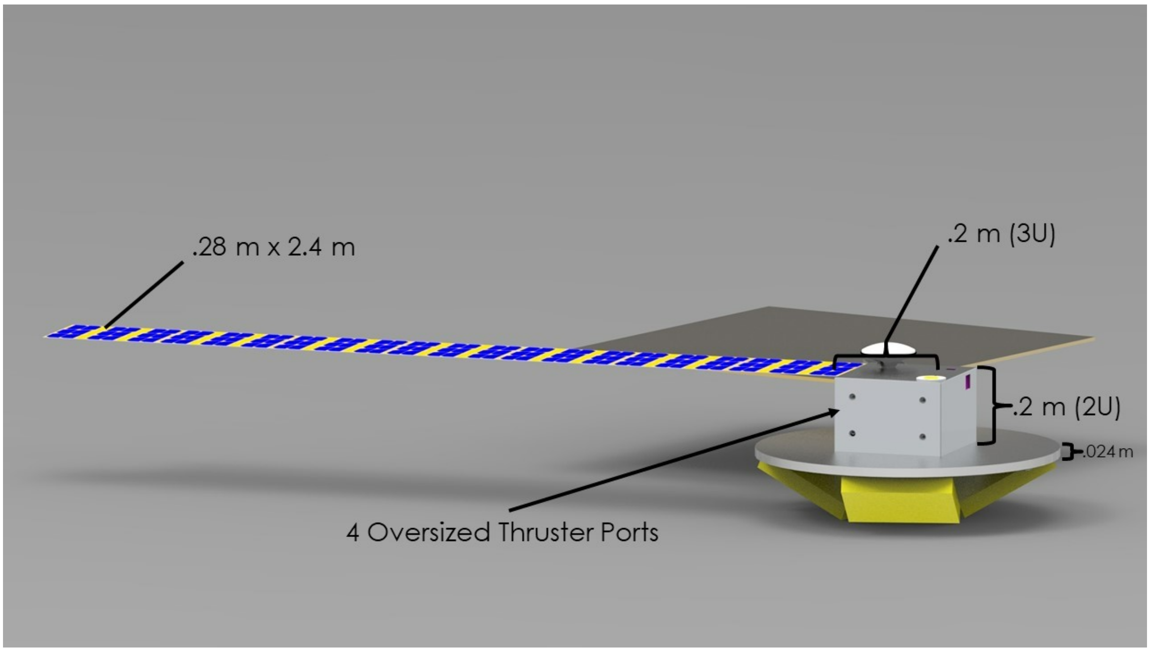


Figure 16: Rear View of Deployed Spacecraft

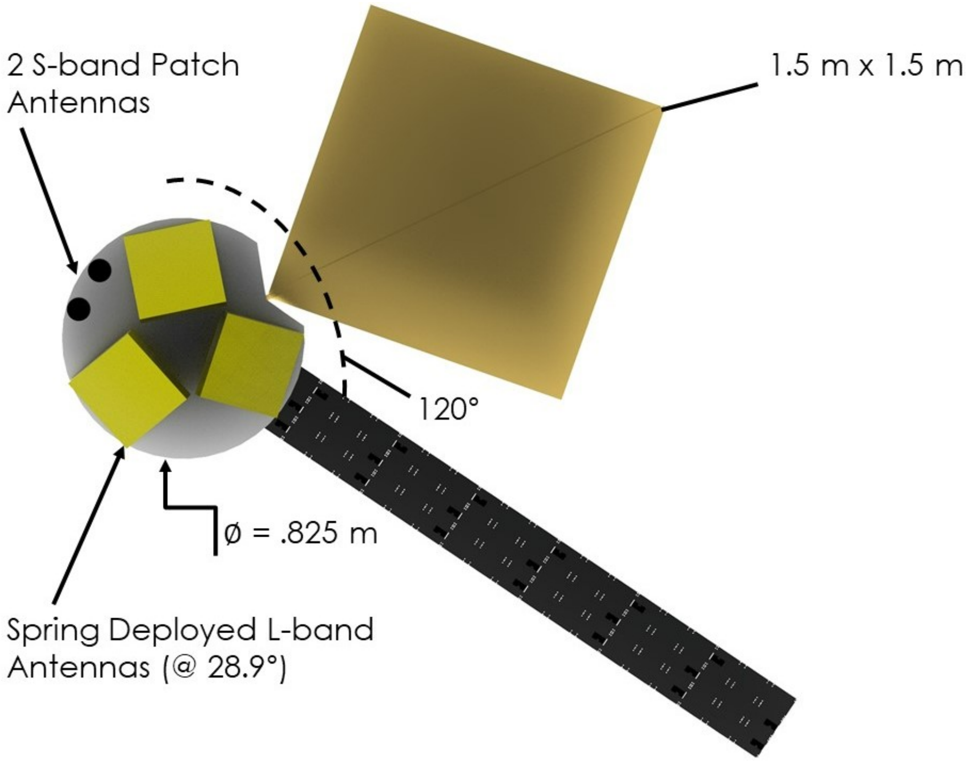


Figure 17: Bottom Profile of Deployed Spacecraft

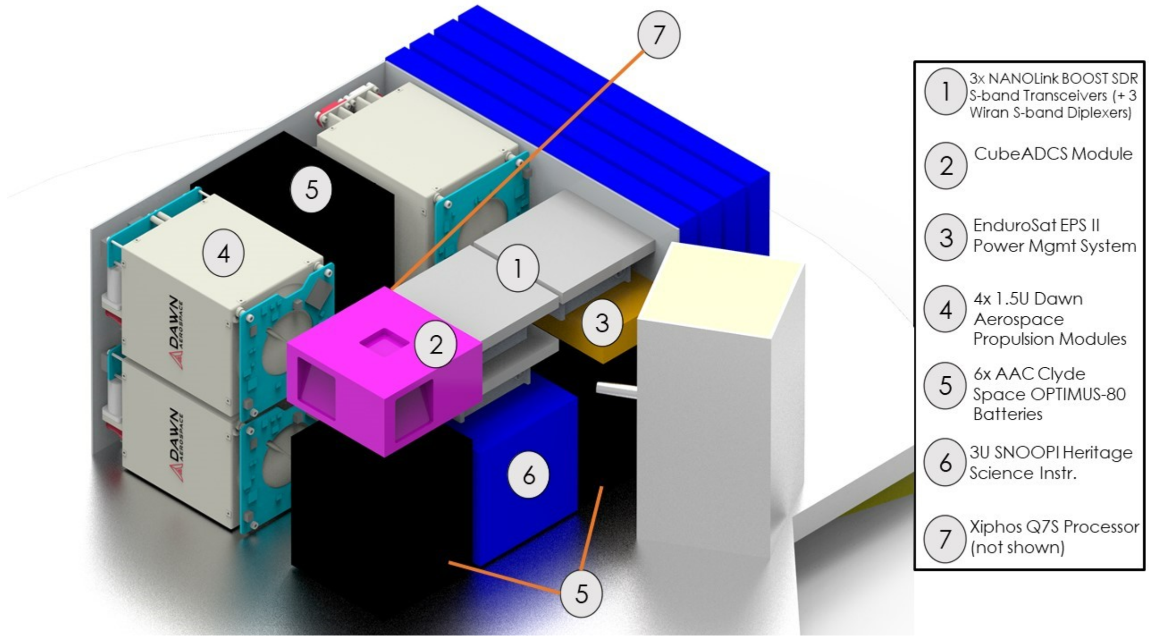


Figure 18: Internal Instrument Configuration

Figures 15-17 show different views of the spacecraft CAD model in its deployed configuration with each mach major external component labeled and configured to prevent interference between each antenna and sensor. The spacecraft bus itself is 18U in size and will be comprised of three separate 6U Cube Sat structures developed by ISISPACE. This 18U structure will contain the core instrumentation and subsystem components while the

Al-6061 1.8 mm shielded table will provide room for spacecraft wire harnessing. Both the table and 18U structure itself are covered in 1.8 mm shielding to ensure mission radiation protection from the external space environment.

Regarding specific external components, the solar arrays will foldout and be gimbaled by a MOOGS Type 1 Solar Array Drive in the roll direction (the ADCS module will perform pitch and yaw adjustments as needed). The S-band primary and redundant antennas can be seen in the bottom view of the deployed spacecraft while a third S-band antenna is located on top of the 18U bus to enable omnidirectional communication capability in a worst-case tumbling mode. Lastly, the 3 L-band antennas will be deployed to an angle of 28.9 degrees and are hinge-fastened with a spring and release mechanism to ensure proper deployment once in orbit.

The major internal components can be seen in figure 18. The driving requirement behind this configuration was for symmetry and balance of the spacecraft's center of mass while also ensuring interconnecting wiring has ample space between systems.

4.4.2 Propulsion

The propulsion system was sized using the Delta-V budget for station-keeping and deorbit purposes. Using an orbit decay model, it was determined that the satellites in the 350 km orbit would need propellant mass allocated solely for station-keeping. No propellant is required for deorbiting due to its fast decay rate. On the other hand, the satellites in the 550 km orbit would need propellant allocated to deorbit , but not for station-keeping due to it's slower decay rate. In table 8 the total amount of propellant needed to raise the orbit of one satellite in the 550 km orbit across the duration of the mission is listed as 0.02 kg which is practically negligable. Thus, station-keeping is unnecessary for the 550 km satellites since it maintains its altitude throughout the mission duration. This propellant mass within the 550 km satellites may also be used to deorbit to the 350 km orbit and station-keep from there in case of the failure of one or more satellites in that orbit. The 350 km satellites are necessary for picking up the P-band frequencies from the low-inclination MUOS satellites, so more redundancy is needed to ensure coverage requirements are met for the science objectives involving P-band frequencies.

The methodology for the orbit decay model used to calculate the Delta-V budget is detailed in Deorbit Plan section (Section 7.2.2) of the Appendix. No propellant is needed for the attitude control system since the ADCS module detailed in 4.4.5 consists of reaction wheels and torque rods.

Propellant Mass Calculations per S/C		
Altitude	350 km	550 km
Delta-V	2.87 m/s	2.74 m/s
Total Station-Keeping Prop Mass	2.54 kg	0.02 kg
De-orbit Prop Mass	0 kg	1.82 kg
Total Prop Mass	2.54 kg	1.84 kg

Table 8: Propellant Mass for Station-keeping

The component selected to meet the propulsion needs of the satellite is the TRL-9 Dawn Aerospace CubeSat Propulsion Module [10]. Each module has a dry mass of 1.3 kg, a wet mass capacity of 0.89 kg, and provides approximately 5.3 N of thrust at a specific impulse of 285 s. The propellant burned is a bi-propellant combination of Nitrous Oxide (N2O) and Propylene (C3H6). The power required by the module is 21 W for the first 50 ms of the burn and 0.8 W for the rest of the burn. Benefits of this module are that it is light-weight, is flight-proven, is currently in production, has an excellent specific impulse, is small in size, comes packaged, and burns a green propellant. Each satellite will include 4 of these 1.5U modules to provide a total propellant mass capacity of 3.56 kg and exceed the minimum propellant/Delta-V requirements in Table 8. The excess propellant contained in four Dawn modules allows for contingency of a failed propulsion module while maintaining the ability to deorbit the 550 km satellites and to conduct station keeping for the 350 km satellites.

4.4.3 Command & Data Handling

Ground Station Overpass Analysis

Based on budget requirements as well as data plan considerations, Amazon Web Services Ground Station network was selected for both downlink and uplink. For this proposal the Stockholm, Sydney, Oregon, Ohio, and Cape Town ground stations were baselined for usage as their high and low latitudes provide a consistent number of overpasses per day. However, more ground stations can be utilized around the globe if more overpass time is required or for a failure mode recovery.

Ground station overpass analysis was conducted using GMAT. A single receiver was propagated for 60 days and GMAT was used to report all overpasses and their duration. A python script was used to determine that these ground stations, with a 10-degree minimum elevation angle, provide an average of 15 passes per day and an average of 400 seconds per pass. Adding two one-minute buffers to establish connection and close connection gives an actual data transfer time of 280 seconds per pass. This results in 4200 seconds of total ground station connection time per day for each satellite on average.

For budget estimation purposes, AWS cites a relatively cheap cost of $10/min for reserved wideband coverage [11]. We qualify for reserved coverage as our mission will be able to predict orbital ephemeris well in advance of our mission. Unfortunately AWS requires a <54 MHz bandwidth for cheaper narrowband, and our communications system can only guarantee a <90 MHz bandwidth. However, this does fall into the allowable 500 MHz bandwidth for AWS ground station wideband reserved. Additionally, the selected forward-error-correction method discussed in the Link Budgets section does result in a large symbol rate of 4.5 Msps, but this symbol rate falls within the allowable symbol rate of 5 Msps for AWS S-Band ground stations as seen in the AWS Ground Station overview presentation [11]. Summarizing the AWS ground station usage, 12 satellites over a 3-year mission span will cost a total of $9 M.

Data Rates

As discussed in section 3.4, the data generated per day for the entire constellation is 21.1 GB. For a 12-satellite constellation, this value can be broken down into an average of 1.82 GB of data per day, per satellite. Combining this data generation rate per day with the downlink time per day, the required downlink data rate can be determined to be 3.467 Mbps. Based on historical estimates and past missions including SNoOPI, we will assume a 1.5x data compression rate. However, a 30% margin is added to this data rate based on NASA literature [12] resulting in an effective 1.2x compression rate resulting in a required downlink data rate of 3.004 Mbps.

Complementing this downlink data rate, uplink data rate was calculated based on the TLE's required for calculation of the doppler effect used in the science instruments data collection. Based on a TLE text file from CelesTrak, a TLE file containing 31 satellites data is 6 kB. This translates to 500 kb of data for 319 transmitter satellite TLE's. We have calculated flight software code requirements and on-board processing assuming linear orbit propagation capabilities for the flight computer, so uploading these TLE's at each overpass should be sufficient to maintain accurate ephemeris for all of the transmitters. This data size, combined with the overpass time, results in a very small data uplink rate of 165 bps. It is far more likely that uplink data rate will be constrained by house-keeping data and commanding, so the uplink margin has been sized for 250 kbps from SMAD table 21.2 [13].

The data rates above were all calculated for a nominal operational mode. An additional major consideration for our TT&C architecture was preparing for potential off-nominal scenarios, with a focus on an uncontrolled tumbling failure mode. For the purposes of recovering from such a failure mode, we have allocated for a downlink rate of 500 kbps and an uplink rate of 25 kbps for the purposes of obtaining system health telemetry, debugging, and sending FDIR commands to ideally recover the satellite.

Mode:	Nominal		Tumbling	
	Downlink	Uplink	Downlink	Uplink
Data Rate:	3.004 Mbps	250 kbps	500 kbps	25 kbps

Table 9: Uplink and Downlink Data Rates for Nominal Nadir-Pointing and Contingency Tumbling Failure Modes

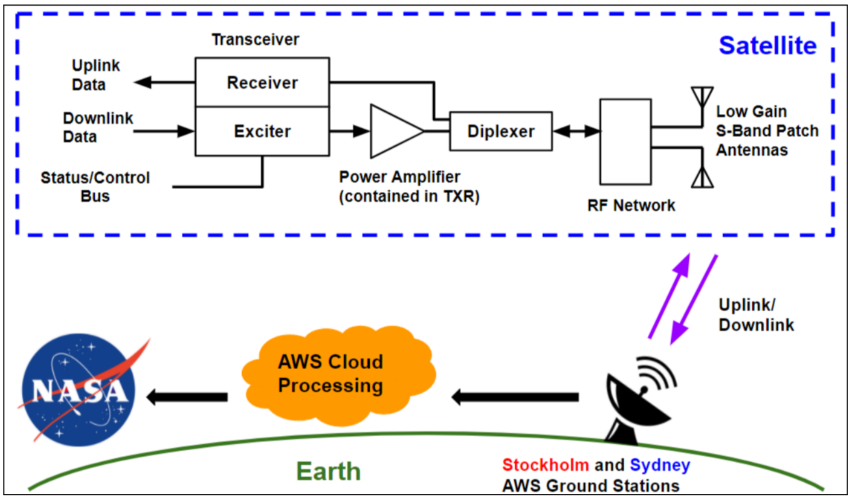


Figure 19: Data Transmission Architecture Flowchart

Communications Hardware Selection

The selection of AWS Ground Stations constrains the selection of our TT&C equipment. AWS supports primarily S-Band communication, and so we selected S-Band components for our communication systems. For the purposes of redundancy and to protect against a tumbling failure mode, we have chosen to include three full S-Band communication systems on each satellite, each including a patch antenna, transceiver, and diplexer. All components selected are TRL-9 flight heritage.

For the antenna, we have selected a wide-FOV low-gain patch antenna from ISIS-PACE. We have selected this antenna for the wide gain pattern as shown in figure 20. This plot illustrates that we can obtain maximum overpass time through a 100° FOV when in nominal nadir-pointing mode. Nominally one nadir-pointing antenna will be active, but a second antenna/transceiver/diplexer set is also positioned nadir-pointing for full redundancy in the event of a failure of the primary set. Additionally, a second full set of antenna/transceiver/diplexer will be positioned opposite-nadir, and with the lower-gain 180° FOV for each antenna, this second antenna gives us a full 360° spherical FOV to maintain constant ground station link in the case of a tumbling failure mode. Additionally, more AWS ground stations can be employed as-needed to increase ground station connection time in an attempt to recover the satellite from a failure mode before losing power.

The selected transceiver is a NANOLink boost-option SDR S-band Transceiver with a power amplifier to provide the maximum power to the antenna to compensate for the low-gain antenna. Finally, a standard WiRan S-band Diplexer will be used. This antenna, transceiver, and diplexer combination will operate in the 2200 - 2290 MHz frequency range (90 MHz bandwidth), standard for S-Band communications and compatible with the AWS ground stations.

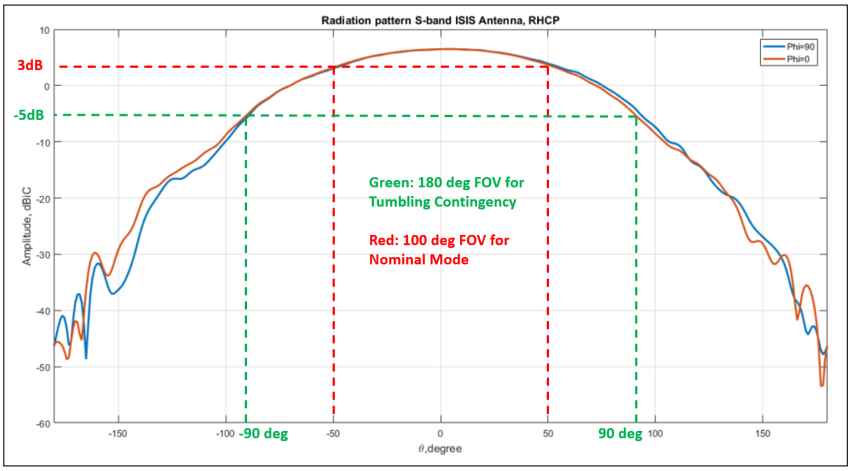


Figure 20: S-Band Patch Antenna H- and E-Plane Gain Patterns

Link Budgets

The information calculated and compiled in the previous sections was fed into the Jan King Link Budget Excel Sheet [14] to determine the link margin. This spreadsheet is standard for NASA missions, particularly CubeSats. SMAD and Jan King default values were used for the majority of the minor detail gains such as line losses, atmospheric losses, noise temperature, etc. AWS provides a Ground Station Figure of Merit (G/T) of 16 dB/K, encapsulating all ground station characteristics for downlink. 2 W of power is provided to the antenna based on the max power input from the component data sheet. A 10 degree minimum elevation angle was used in accordance with the GMAT overpass analysis and SMAD baseline assumptions. For data modulation standard BPSK (binary phase-shift keying) is utilized. Finally, for Forward-Error Correction a Convolutional R = 2/3 and K = 7 coding is utilized to improve the link budget. This FEC method ensures a bit rate error of 1e-6 to get good quality data and reduces the gain-to-noise ratio (Eb/No). Using all of the above inputs, we obtain the following link margins seen in Table 10.

	Nadir Pointing (3dB, 100 deg FOV)		**Tumbling (-5dB, 180 deg FOV)**	
	Downlink Margin	Uplink Margin	Downlink Margin	Uplink Margin
350 km Plane	13.9 dB	12.1 dB	14.8 dB	14.1 dB
550 km Plane	12.1 dB	9.2 dB	11.9 dB	11.2 dB

Table 10: Downlink and Uplink Margins for Nominal Nadir-Pointing and Contingency Tumbling Failure Modes

For the selected components, data rates, ground stations, and assumptions all link margins for both the nominal mode and the tumbling failure scenario have at least a 9.2 dB margin, which is well above the 3 dB link margin recommended by NASA in [12] which provides significant margin for invalid assumptions or inefficiencies in the TT&C subsystem design.

Flight Computer

The flight computer selected for on-board CD&H is a Xiphos Q7S, which is the 7th generation of a line of flight heritage processors, making this a safe TRL-9 choice. This onboard processor will be responsible for handling data collection, storage, and transmission, and subsystem management including ADCS, TT&C, propulsion, thermal, payload, and power components. This processor comes with an ARM dual-core 766 MHz CPU and a real-time clock for accurate state estimation. This processor utilizes only 2 W and weighs only 32 grams [15]. This onboard processor will use two Delkin Industrial SLC 32 GB SD Cards [16] to obtain the maximum possible data storage. These SD cards will serve as redundancies and will hold both flight software as well as science data.

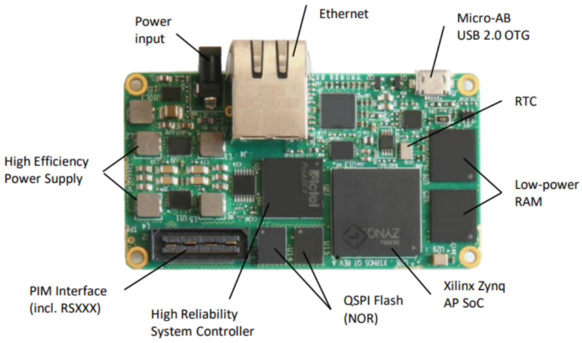


Figure 21: Xiphos Q7S Flight Computer [15]

4.4.4 Power & Solar Array

Sizing of the power systems to include the batteries solar array was performed based on a nominal daytime power load of 58.73 W and eclipse power load of 60.48 W, visible in table 11. The duty cycle for the thermal system was determined by doing the thermal analysis which determined the heater needs in an eclipse. The duty cycle for the TT&C system was determined by doing a groundstation overpass analysis and determining how many minutes per day are spent communicating with them. A 25% contingency is applied as per the NASA GSFC gold rules [7]. Given a long eclipse duration of 35 minutes, the solar arrays were sized with the following equation (Eq. 21-6 in SME-SMAD [13]), where PSA is the required end-of-life power supplied, and Pe, Te, Xe, Pd, Td, and Xd, are the power used, duration, and efficiency values for eclipse and daytime, respectively.

$$P_{SA} = \frac{\frac{P_e T_e}{X_e} + \frac{P_d T_d}{X_d}}{T_d} \text{ [13]}$$

After accounting for the lifetime degradation of the solar cells over the mission duration, we determined a beginning-of-life power generation requirement of 150.60 W from the solar array. To meet this requirement, we selected the MMA Design HaWK solar array. This is a fold-out solar array with flight heritage since 2017 (TRL-9) [17]. For our power requirement, we estimate a required area of 0.58 m^2, which comes with a mass of about 1.51 kg. The array will be gimballed on the roll-axis by the MOOG Type 1

Solar Array Drive Assembly with yaw performed by the ADCS to maximize solar power generation, but we still assumed a 23.5 degree angle to direct sunlight as per SME-SMAD 21.2 [13]. The MOOG drive assembly has a mass of 1.16 kg and a power draw of 8.6 W [18].

Subsystem	Power (W)	Day Duty Cycle	Day Power (W)	Eclipse Duty Cycle	Eclipse Power (W)
Payload	30	1	30	1	30
Structures & Mechanisms	8.6	1	8.6	0	0
TT&C	14	0.05	0.68	0.05	0.68
Processor	2	1	2	1	2
ADCS	4.5	1	4.5	1	4.5
Thermal	35	0	0	0.29	10
Power	1.2	1	1.2	1	1.2
Propulsion	3.2	0	0	0	0
TOTAL:			**46.98**		**48.38**
25% Contingency			**58.73**		**60.48**

Table 11: Power Budget in Nominal Operation (Includes TT&C Through Overpass Analysis)

The minimum total battery capacity was determined with the same power requirements. From Table 21-19 in SME-SMAD [13], where C is capacity, DOD is depth of discharge, N is the number of batteries, and n is efficiency, we have:

$$C = \frac{P_e T_e}{DOD \cdot Nn} \text{ [13]}$$

With an allowed 30% DOD for Li-Ion batteries from SME-SMAD Fig. 21-16 [13], this equation determines a required total capacity of 135.95 W-hr. At this capacity, however, there would be insufficient time to troubleshoot and recover a tumbling satellite in safe mode (about three hours). The battery was thus resized to accommodate for a 12 hour recovery period in the event of a tumble. Based on safe mode power requirements of 32.2 W and 47.7 W in daytime and eclipse, respectively, with full capacity at the beginning of the tumble and no charging during the tumble, we found a necessary battery capacity of 37.9 W-hr per hour of desired recovery window.

To meet this goal, we selected six 80 W-hr batteries with a total capacity of 480 W-hr and a mass of 4.02 kg. These are the AAC Clyde Space Optimus-80 batteries, TRL-9 batteries with flight heritage [19]. The inclusion of six batteries provides redundancy, and the increased capacity allows for a depth-of-discharge of only 8.5% per cycle. By having six batteries instead of the two that would have been required without the 12-hr recovery period requirement, we added 2.68 kg to the satellite mass. This small addition did not affect our choice of launch vehicles or limit the mission in any significant way, so it was determined to be worthwhile for mitigating the risk of a satellite tumble.

For power management, we selected the Endurosat EPS II, a TRL-9 system with 250 W peak power output [20], which is well over the requirement of 101.3 W, our satellite's highest power load during propellant ignition. The system weighs 1.28 kg and allows a wide range of input and output voltages.

4.4.5 Attitude Determination & Control

Attitude control requirements were derived from the need for accurate solar panel orientation relative to the sun, accurate burn orientation for station keeping maneuvers, and maintaining consistent nadir-pointing for science measurements. Based on these requirements, it was determined that attitude control to an accuracy of less than one degree pointing error is required. To complement this attitude control accuracy requirement, the attitude determination is required to be less than half a degree accuracy to ensure at least a half-degree margin. These constraints were derived from SME-SMAD Table 19-5 [13].

To determine the necessary size for the ADCS components, the Simulink CubeSat Simulation Blockset [21] was utilized to analyze a 6-DoF dynamics model of the satellites in a nadir-pointing control mode. The 350-km orbital elements were input to obtain the worst-case attitude scenario as the lower altitude experiences higher drag disturbances. The initial conditions were set to zero angular velocity in all directions to simulate deployment from the launch vehicle.

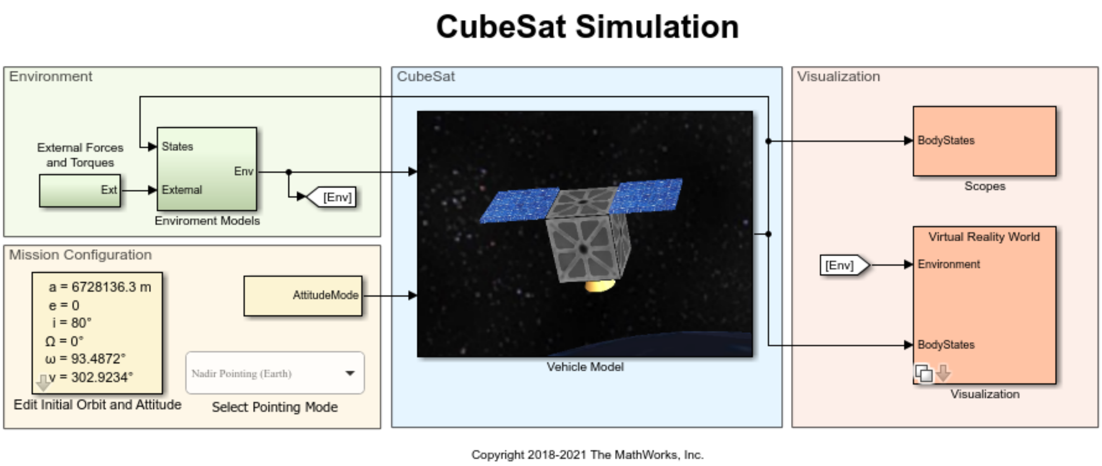


Figure 22: Simulink Model Used to Determine the Requirements for the ADCS System

This dynamics model was used to determine that the maximum torque for this nadir-pointing control mode is 1.1 mNm in the pitch axis to induce the nadir-pointing rotation upon launch vehicle deployment. A combined nadir-pointing and solar panel pointing slew maneuver was not able to be simulated, however the solar panels are gimbaled in the roll axis, and so only slewing around the yaw axis would be required to ensure solar panel pointing whenever possible.

For the nadir-pointing control mode, the simulink model showed very small required torque in the yaw axis, and so it is not expected that a slew maneuver larger than 1.1 mNm would be required in any single direction. Based on this analysis, the CubeADCS integrated module [22] was selected for a TRL-9 solution covering both determination and control requirements. The CubeADCS Gen-1 module is a 1U stack weighing about 1 kg total. For attitude determination the module includes a Sun/Earth sensor, 10x coarse sun sensors, a star tracker, and two magnetometers. Altogether these sensors provide $<0.1^\circ$ attitude determination accuracy with multiple redundancies and extensive margin in both sun and eclipse.

For attitude control, the module comes with three-axis reaction wheels with maximum torque of 2.3 mNm and max momentum storage of 30 mNms. The module also includes 3 torque rods which will be utilized for momentum dumping. Based on the simulink dynamics model, momentum accumulation in the pitch axis requires momentum dumping once every two days, and this same trend is expected for the yaw axis for the solar panel orientation control mode.

Finally, the module includes a fully integrated ADCS computer with pre-configured and custom modes such as nadir-pointing and a nadir-pointing + solar panel pointing. Overall this module costs only $65k which is very cost effective for all the included components.

4.4.6 Thermal Control

To develop a thermal control system, we first tabulated the minimum and maximum operating temperatures of each thermally limited component to derive our operating constraints.

Thermally Sensitive Component	Min. Operating Temperature (°C)	Max Operating Temperature (°C)
AAC Clyde Space Optimus-80 Battery	-10	+50
CubeADCS Module	-10	+60
Xiphos QS7 Processor	-40	+60
Dawn Aerospace CubeSat Propulsion Module	-10	+30
S-Band Patch Antenna	-20	+50
NANOLink SDR S-band Transceiver	-25	+85
WiRan S-band Diplexer	-40	+75

Table 12: Operating Temperatures of Thermally Sensitive Components

As shown in table 12, Our spacecraft is constrained from -10℃ to +30 ℃. Implementing a thermal margin of 5 ℃, we developed our thermal system to maintain spacecraft temperature between -5 ℃ to +25 ℃ in all modes and scenarios (except safe mode, where it is kept above -10 ℃ in order to reduce power consumption.) We implemented both a thermal control coating as well as an active heating solution to meet these requirements.

For coating, we selected Teflon with a 2 mil (.05 mm) aluminum backing, which has an absorptivity of 0.08 and an emissivity of 0.66 [23]. Across our chassis surface area, this coating has a mass of 0.047 kg. We also selected the Kapton Flexible Heater SHK-00101 [24], which is a 76mm diameter disk that provides up to 35 W of heating and weighs 0.002 kg. Assuming all spacecraft power is converted to heat, we were able to use the thermal energy balance equation $Q_{in,solar} + Q_{in,albedo} + Q_{in,internal} = Q_{out,space} + Q_{out,earth}$ [25] to determine the average resulting satellite external temperature of thermal equilibrium for each relevant mode and scenario of our mission, as shown in table 13.

	Thermal EQ Cold Case (550 km Altitude, Eclipse)			Thermal EQ Hot Case (350 km Altitude, Daylight)		
Mode	System Power Usage (W)	Heating Required (W)	Temp	System Power Usage (W)	Heating Required (W)	Temp
Safe	22.7	20	-9	31.3	0	7
Safe - Tumbling	32.2	10	-9.4	32.2	0	7.7
Nominal Operation	37.7	10	-4.7	46.3	0	17.3
Data Transmission	51.7	0	-1.5	60.3	0	25.9
Propulsion Mode	25.9	21	-5	34.5	0	9.3

Table 13: Thermal Equilibrium Temperatures in All System Modes

Total power usage in an eclipse must be greater than 42 W to stay within operating temperatures. If the current mode does not generate that much power, the heater is turned on. In safe mode, heat is only supplied to the minimum operating temperature of -10 ℃, but in nominal operating mode, heat is applied to maintain a spacecraft temp of -5 ℃ to provide a thermal margin above minimum operating temperature. In daylight, active heating is not required in any mode. The hottest case, data transmission mode, yields an equilibrium temperature of 26 ℃. This is 4 ℃ below the maximum operating temperature of the spacecraft, thus no radiator or other cooling method is required for our spacecraft.

4.4.7 Flight Software

To approximate the amount of Software lines of Code (SLOC) used for this mission, COCOMO81 was used to delegate the approximate lines of code dedicated for each computer software component based off historical SLOC estimations for typical spacecraft functions with nominal performance (SME-SMAD, 2011, p. 614 [13]). Applying all of the components useful to our mission adds up to an estimate of 38,700 lines of Code. This includes SLOC estimates for ADCS, payload integration, orbit determination, etc. Using the built in cost estimator of COCOMO81, this translates to a $27.04M cost for developing the software for our mission.

4.5 Additional Mission Elements

Figure 23 provides an overview of all of the subsystems which are needed for the spacecraft to function. Each of these subsystems is covered by a section above.

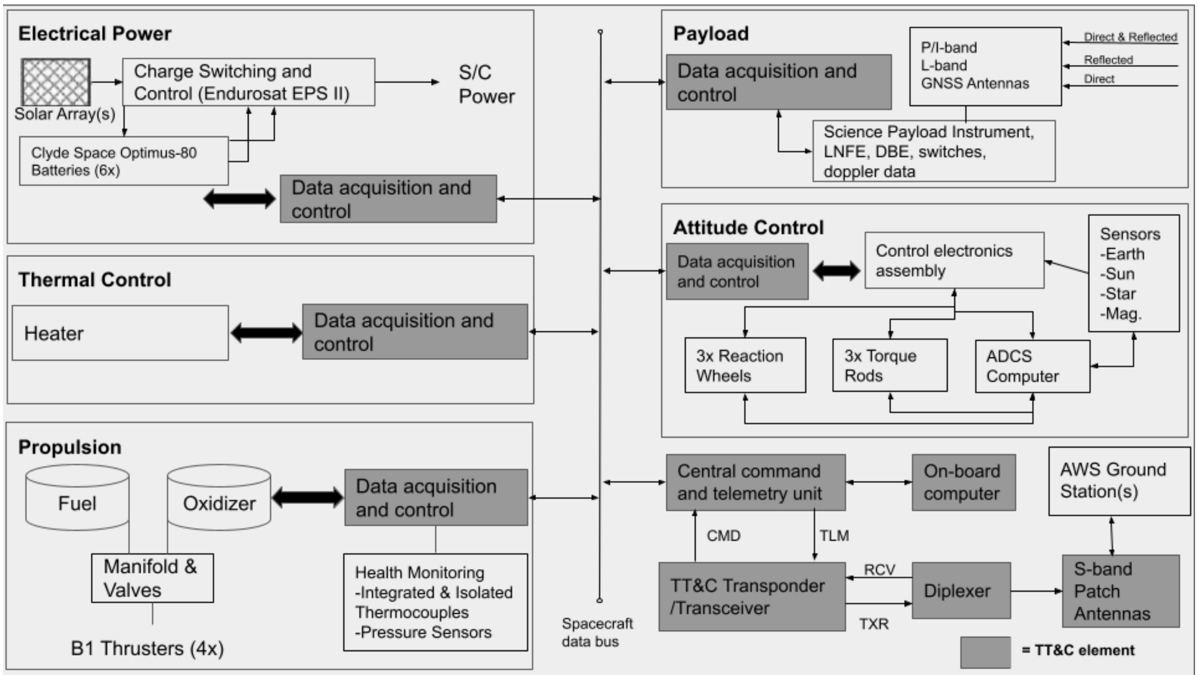


Figure 23: Satellite Block Diagram

4.6 Flight System Contingencies

Mass Contingencies

We implemented contingencies into our spacecraft dry mass, propellant mass, and launch vehicle mass capability. We determined our spacecraft dry mass to be 39.66 kg. This value was the sum of the component masses from each of our subsystems. We added a 25% contingency (as per NASA GSFC gold rules [7]) to our structures and mechanisms subsystem which includes; an antenna mounting table, shielding walls, and empty spacecraft bus. This subsystem has a mass of 7.32 kg, which (with a 25% contingency) yields 1.83 kg to account for bolts, brackets, wiring, etc., and a final spacecraft dry mass of 41.49 kg.

From our propellant mass calculations, we determined that 2.54 kg of propellant is required for our 350 km constellation to account for station-keeping. In our 550 km orbit, 1.82 kg of propellant is required for deorbiting. Each of the propulsion modules we selected have a prop mass capacity of 0.89kg. Each of our spacecrafts are equipped with 4 of these modules, which yields a total propellant mass capacity of 3.56 kg. This yields propellant mass contingency of 28.6% for our 350 km orbit, and 48.3% for our 550 km orbit. This latter contingency is useful in the event of a satellite failure, as it will allow for one of the 550 km satellites to Hohmann transfer down to 350 km if necessary to meet coverage requirements.

As covered in Section 4.3, Pegasus (XL) has a mass capability of 315 kg for our 550 km orbit. Our constellation of 5 satellites in this orbit plane have a wet mass of 225.25 kg, which yields an unused mass contingency of 28.7%. Minotaur has a mass capability of 795 kg for our 350 km orbit. Our constellation of 7 satellites in this orbit plane have a wet mass of 315.34 kg, which yields an unused mass contingency of 60.3%.

Power and Data Storage Contingencies

As covered in Section 4.4.4, we determined that total battery capacity of 135.95 W-hr is required for our spacecraft to allow for a 30% depth of discharge for out lithium-ion batteries. However, we determined that to allow for 12 hours recovery time for a tumbling satellite in safe mode, a battery capacity of 454.8 W-hr would be needed in this extreme case. Using 6 80 W-hr batteries, our total capacity is 480 W-hr, which is a 353% contingency from our nominal operating needs of 135.95 W-hr. For solar panel sizing, a 25% power contingency is used.

As covered in Section 3.4, we determined that each spacecraft generates an average of 1.82 GB of data per day. We have sized our TT&C system to downlink this data daily, as covered in section 4.4.3. We equipped each spacecraft with 2 x 32 GB SD cards (configured in RAID-1 to mirror data between storage devices), which yields a data storage contingency of 1758%.

4.7 Mission Operations

System Modes

In order to ensure that our spacecraft can respond to as many scenarios as possible, we have developed four overarching system modes. These modes are also used in the sizing of our power requirements, as not all subsystems are expected to be on at the same time.

The first mode that we discuss is the nominal operations mode. This mode encompasses most of the mission as it is when the spacecraft will be taking in science measurements. For this mode we must ensure that the solar panels are facing toward the Sun to improve their efficiency. We must also be nadir pointing and will require some heating during the eclipse.

The second mode discussed is the data transmission mode. This mode will be used when the spacecraft is sending the science data to a ground station. No heating is required for this mode as enough is generated by the subsystems to stay above the temperature limits.

The third mode discussed is the propulsion mode. This mode will mostly be used by the lower altitude satellites when they are performing station-keeping maneuvers. The propulsion modules chosen have a peak power input which forces us to turn off the science instruments to avoid overheating. For the most part though, the propulsion modules have a power input which does necessitate some heating. Additionally, we account for the communications module to be on for the burn in case something goes wrong in the middle of a burn and it is needed.

The final mode is the safe mode. This mode is used for any extraneous circumstances and has many sub-modes. In general this mode keeps the spacecraft nadir pointing, communicating with the ground as appropriate, and the solar panels pointed towards the sun. Thermal control is needed for this mode to avoid freezing. A sub-mode which has been previously mentioned is the tumbling safe mode. This sub-mode will never be entered nominally, but exists to protect against a failure resulting in an uncontrolled spacecraft tumble. In this mode the spacecraft stops using the solar panel drive as trying to get the solar panels to point toward the sun may be difficult, and the secondary communications system is turned on. Attitude determination is left on (to ensure we are actually tumbling), but attitude control is turned off. This sub-mode was used to size the batteries to ensure the spacecraft we have designed can survive as long as possible in a tumble. The longer it can survive, the more time the ground team has to determine there is an issue and to find a solution. Additional sub-modes could be generated as the software for the spacecraft is developed to ensure smooth mission operations.

A summary of these system modes and the status of the different subsystems is provided in table 14.

	Currently Operating Subsystems					
System Mode	ADCS	Comms	Science Inst.	Propulsion	Thermal Control	Solar Panel Drive
Safe Mode	ON	ON	OFF	OFF	ON (Eclipse)	ON (Day)
Safe Mode - Tumbling	ON (Determination)	ON (x2)	OFF	OFF	ON (Eclipse)	OFF
Nominal Operation	ON	OFF	ON	OFF	ON (Eclipse)	ON (Day)
Data Transmission	ON	ON	ON	OFF	OFF	ON (Day)
Propulsion Mode	ON	ON	OFF	ON	ON (Eclipse)	ON (Day)

Table 14: Summary of the System Modes

Nominal Satellite Operations

In general, the spacecraft will spend most of its time collecting science data and transmitting it to the ground. When only collecting science data, the spacecraft will be nadir pointing. In the day period of the orbit, it will be spinning about the yaw axis, with the solar panels spinning on their roll axis, to ensure maximum power can be obtained from the solar panels. The science measurements will be taken in by the science instrument derived from SNoOPI, which will then be saved onto the on-board flight computer.

Once a scheduled ground station pass is set to occur, the communications system will be turned on, whilst the instrument continues to take science data. Due to the large beam width of the antennas selected, no pointing is required to communicate with the ground stations so no special tasks are necessary when doing this. The satellites will each send their stored science data and system health information to the ground stations. The ground station will uplink additional doppler measurements (needed for the science instrument) and instructions to the satellite.

In nominal operations for the lower altitude satellites, stationkeeping is a consideration. Stationkeeping for these satellites must be completed roughly every 15 days. To do this, the satellites will perform a Hohmann transfer up to the intended orbital altitude. This process will be mostly automated, with the ground team requiring simple verifications to ensure the system is working well.

Ground Systems and Facilities

This mission will have one dedicated mission operations facility to ensure spacecraft monitoring and maintenance can be performed without delay and with a team of experts for our constellation. Since AWS Ground Stations provide a global infrastructure with a low-latency global fiber network which provides downlink and uplink data access without the need for ground station infrastructure [26], our dedicated operations facility will not have a permanent location. The facilities primary objective will be to house engineers and technicians throughout the design phase and during the missions operations phase as a central hub for design and, following launch, it will act as a mobile mission operations center. The current plan is to have the dedicated operations facility near the launch facility location preceding successful launch of the constellation; following this, the operations center location will be located in the U.S. dynamically throughout the mission lifetime and will move to differing locations depending on the status of the mission and the locations of the employees. A summary of our ground facility details is shown in figure 24.

WBS Element 7.0 Operations	Cost Category	Assumptions, Factors, Calculations
7.1 PMSE (Project Mgmt & Sys Engr)	Labor	15% of other cost (as high as 20%)
7.2 Space Segment SW Maintenance	Labor	38,700 SLOC/16,000 for SWM = 3 FTE's @ $200K
7.3 Ground Segment		Note: values that SLOC is divided by is 'most likely' SLOC maintained per person-year (per SMAD) FTE = full-time equivalent
7.3.1 Mission Operations	Labor	3 engineers @ 200K + 2 technicians @150K
7.3.2. Ground Segment SW Maintenance	Labor	25,000 SLOC/20,000 =1 FTE @ $200K
7.3.3. Ground HW Maintenance	Labor	7% of $1,400K (Hardware Acquisition Cost)
7.3.4 Facilities	Facility Lease	1,000 sqaure meters @ $1.25K per sqaure meter

Figure 24: Operations Center Cost Estimation Summary

Although figure 24 was used primarily for operations cost estimation, it was also useful for estimating how many full-time equivalents will needed for our estimated source lines of code (SLOC) for space and ground systems. Taking our SLOC estimation for space, 3 FTE engineers would be needed for space software (SW) maintenance. For ground SW maintenance, only one engineer is needed since ground hardware will be adapted from an existing software package for maintaining 25,000 SLOC of code based off historical data (SME-SMAD, 2011, pg. 312 [13]). Lastly, we plan to utilize 1,000 square meter properties to serve as our mobile operations facilities which will be rented in different locations as needed.

Operational Communications Plans

This mission will utilize the AWS ground station network, utilizing specifically the Ohio US, Oregon US, Stockholm Sweden, Sydney Australia, and Cape Town South Africa ground stations. This network provides an average of 15 passes per day, with each satellite having a total of over an hour of downlink time per day. This downlink time has been determined to be sufficient to downlink all science data generated to prevent a data backlog on any satellites, and a 30% contingency has been applied to downlink rates to ensure robustness [12].

Based on the calculated value of 1.82GB of data generated per day per satellite, we determined that our antenna and ground station network is sufficient to achieve over a 10 dB link margin for nominal science data downlink. All communications will occur in the S-Band, specifically 2200 MHz to 2290 MHz, a 90 Hz bandwidth is guaranteed by the ISISPACE patch antenna selected.

Finally, we have protected against a contingency tumbling mode by ensuring we can remain powered on for 12 hours in a tumbling safe mode, during which time we have a low-gain, low-data rate transmission mode which keeps a >9.2 dB uplink and downlink margin to ensure debugging and recovery efforts can continue with a 360° FOV via two opposite-pointing patch antennas. In this case, additional AWS ground stations can be used on-demand to gain more ground station communication time.

4.8 Assembly, Integration, & Test

In order to maximize the probability of mission success, every MoIST satellite will undergo necessary testing before being launched to space. Individual components will first be tested to verify the manufacturer's claims as follows:

- Functionality testing
- Structural and deployment testing as appropriate
- Space environment testing (radiation, vacuum, temperature)
- Electromagnetic interference testing for electronics
- Calibration prior to testing and flight

Additional testing will be performed on the assembled spacecrafts:

- Vibration and shock testing
- Vacuum testing
- Radiation testing
- Mass properties (including center of mass) testing

An engineering test unit will also be manufactured for the purpose of qualification testing and to make it easier to troubleshoot spacecraft issues once the constellation has launched. Additional testing may be needed for the lower TRL number components, which includes just the science instruments. It is expected that these will be a higher TRL once the SNoOPI mission launches.

Costs related to integration, assembly and testing (IA&T) were estimated using the Small Satellite Cost Model (SSCM) and known costs for our science payload. The SSCM estimates IA&T by using the estimated cost of the spacecraft bus and components and then applying a predefined CER to output an estimated IA&T cost for integrating the entire spacecraft bus (Table 11-9 in SME-SMAD [13]). Following this methodology, we estimate the cost of the spacecraft IA&T to be around \$1.4 M for the first satellite. In addition to spacecraft IA&T we must also consider the cost of integrating the payload (science instrument and antennas). The recurring \$100 K cost of each payload provided by Purdue Professor James L. Garrison was assumed to include its IA&T with the bus, so this value was applied for the first satellite. By applying the learning curve with a slope of 0.9 to the remaining satellites in our constellation with these IA&T cost in mind, the total IA&T cost for the constellation was found to be approximately \$10.5 M.

4.9 Verification

Verification for the functionality of the MoIST constellation is provided in the relevant section for those elements. For example, verification that the constellation meets the science requirements is discussed in the science implementation section. Verification that subsystems meet their requirements is discussed in a per-subsystem manner.

4.10 Schedule

In allocating months for each phase of the mission, the project timeline had to be approximated for each phase of design from pre-phase A through the conclusion of operations at the end of phase E. The parametric estimates discussed further for project schedule and the associated schedule margin was based off of the AAE-590 Space Flight Operations - Lecture 5 [25]. For our schedule allocations per mission phase, the assumption of inherited technology was utilized considering the majority of the spacecraft will be using high TRL, flight heritage components. Under that assumption, discrete time allocations (in months) for each mission phase (A-D) provided monthly duration estimates per mission phase for a mission utilizing inherited technology (with associated schedule margins).

Regarding schedule allocation, this assumption estimated our project schedule to allocate 3 months for phase A, 3 months for phase B, and 20 months for phases C/D. Schedule margins of 1 month per year were used for both phases A and B while 2 months per year were applied for Phase C until the beginning of Phase D where margins of 2.8 months per year were used considering the greater uncertainty associated with post launch site/instrument integration and testing (IA&T) activities. These estimations resulted in a 33 month (2.75 year) timeline for launch readiness of the entire constellation with a best-case launch readiness date in October, 2024 assuming no schedule overruns beyond the defined margins. The EVM requires that our mission must be ready for launch five years after the contract is in place making our latest planned launch date January, 2026. This early launch readiness date will provide our mission with schedule reserves of 1.25 years (15 months) enabling room for significant schedule overruns due to potential design or launch delays. Finally, our mission operations phase (E) will proceed for three years following the launch and beginning of operational mode for all spacecraft in the constellation. The schedule foldout and proposed major milestones are detailed in figure 25.

	Phase					
Date	A	B	C	D	Reserves	E
January 2022						
April 2022	PMSR					
May 2022						
August 2022		PDR				
Spetember 2022						
February 2023			CDR			
March 2023						
September 2023			ARR			
October 2023						
July 2024				ORR		
August 2024				MRR		
September 2024						
October 2024						
December 2025						
January 2026						
January 2029						

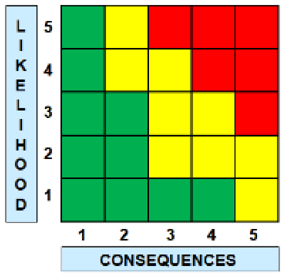
Figure 25: Schedule of Operations

5 Management

5.1 Top 5 Project Risks & De-scopes

The expected top 5 risks to project execution as planned are shown in figure 27 as a risk matrix. The ranking and categorization of these risks was performed in accordance with the fever chart rankings from NASA GPR 7120.4D [27] in figure 26.

Likelihood	Safety Estimated likelihood of Safety event occurrence	Technical Estimated likelihood of not meeting performance requirements	Cost Schedule Estimated likelihood of not meeting cost or schedule commitment
5 Very High	($P_{SE} > 10^{-1}$)	($P_T > 50\%$)	($P_{CS} > 75\%$)
4 High	($10^{-2} < P_{SE} \le 10^{-1}$)	($25\% < P_T \le 50\%$)	($50\% < P_{CS} \le 75\%$)
3 Moderate	($10^{-3} < P_{SE} \le 10^{-2}$)	($15\% < P_T \le 25\%$)	($25\% < P_{CS} \le 50\%$)
2 Low	($10^{-5} < P_{SE} \le 10^{-3}$)	($2\% < P_T \le 15\%$)	($10\% < P_{CS} \le 25\%$)
1 Very Low	($10^{-6} < P_{SE} \le 10^{-5}$)	($0.1\% < P_T \le 2\%$)	($2\% < P_{CS} \le 10\%$)



Consequence Categories					
Risk	1 Very Low	2 Low	3 Moderate	4 High	5 Very High
Safety	Negligible or not impact	Could cause the need for only minor first aid treatment	May cause minor injury or occupational illness or minor property damage	May cause severe injury or occupational illness or major property damage.	May cause death or permanently disabling injury or destruction of property.
Technical	No impact to full mission success criteria	Minor impact to full mission success criteria	Moderate impact to full mission success criteria. Minimum mission success criteria is achievable with margin	Major impact to full mission success criteria. Minimum mission success criteria is achievable	Minimum mission success criteria is not achievable
Schedule	Negligible or no schedule impact	Minor impact to schedule milestones; accommodates within reserves; no impact to critical path	Impact to schedule milestones; accommodates within reserves; moderate impact to critical path	Major impact to schedule milestones; major impact to critical path	Cannot meet schedule and program milestones
Cost	<2% increase over allocated and negligible impact on reserve	Between 2% and 5% increase over allocated and can handle with reserve	Between 5% and 7% increase over allocated and cannot handle with reserve	Between 7% and 10% increase over allocated, and/or exceeds proper reserves	>10% increase over allocated, and/or can't handle with reserves

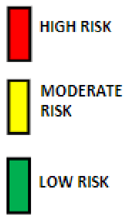


Figure 26: GFSC Risk Matrix Scale

Risk descriptions and mitigations are as follows:

A: Loss of 3+ satellites to subsystem failures. Failure/loss of 3 or more satellites would render the mission success criteria unachievable. The likelihood of this scenario was calculated based on historical data of major subsystem failures in SME-SMAD 21.1.2 [13] applied to a binomial distribution. The most common failure points for a 3 year mission are in TT&C, propulsion, and the solar array. To mitigate this risk, we included redundancies and margins in all 3 of these systems.

B: Launch vehicle failure. A catastrophic failure in a launch to either of the 2 orbital planes would be a major mission failure. With only a fraction of the satellites in orbit, and only in one orbital plane, minimum success requirements are not achievable.

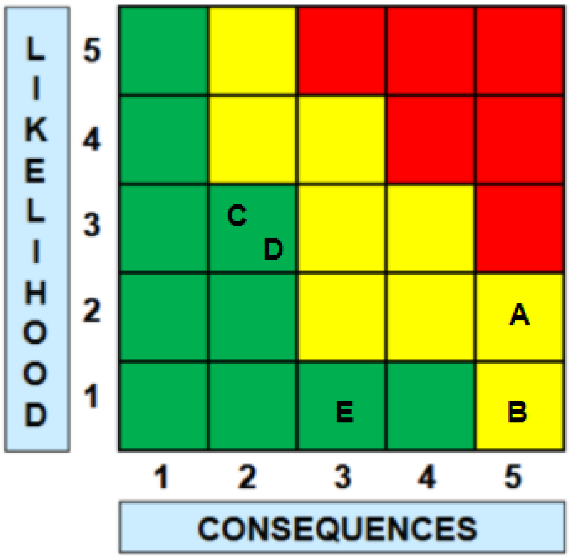


Figure 27: MoIST Top 5 Risk Matrix

C: Loss of single satellite to subsystem failure. Same as risk A above, but the failure is for only a single satellite. This is much more likely to occur based on the binomial distribution, but, given that verification was performed with 2 fewer satellites than will actually be launched, mission impact is minor. Coverage is slightly affected, but the success criteria is still achieved within margin. This risk is effectively mitigated by the inclusion of those 2 additional satellites.

D: System integration/launch delay. Discussed further in section H, difficulty with system integration is the inherent risk in use of COTS components.

E: Loss of satellite in MUOS constellation. As the only P-band transmitter in use by our mission, MUOS is an integral part of meeting our coverage requirements. If even a single MUOS transmitter were to be lost for any reason, our ability to meet mission success requirements would be hindered even more than a single lost MoIST satellite. The likelihood of this scenario is low, however, as MUOS is critical infrastructure. Additionally, less reliable P-band constellations could be considered to improve the catastrophic nature of this risk.

The binomial distribution mentioned in risks A and C was calculated based on the a 7.13% chance of satellite failure within a 3 year mission, interpolated from historical data in SME-SMAD 21.1.2 [13]. The individual subsystem failure probabilities and mitigations are shown in figure 28. The binomial probability of 1 out of 12 satellites failing based on this percentage is 37.92%. The probability of 2 satellites failing is 16.01%. The probability of 3 or more satellites failing, indicating mission failure, is then 4.9%. These estimates are considered to be very conservative, as the data was collected between 1990 and 2008. In conclusion, with the current MoIST constellation we can achieve a greater than 95% probability of mission success, which meets the previously stated requirement.

Subsystem Failure	Historical Failure Rate	Mitigation
Thruster/Fuel	1.34%	Fuel margin, redundant thrusters
Telemetry, Tracking & Command	1.56%	S-band antenna redundancy
Solar Array Deployment/Operation	1.11%	Solar array power margin
All Other Failures	3.12%	Additional batteries, thermal control margin, ADCS margin
Total Failure Rate	7.13%	

Figure 28: Probability of Subsystem Failures for 3 Year Mission

5.2 Management Detail

Multiple management requirements by the EVM AO are not applicable to our proposal. This includes sections about the PI's management and organization approach, the key management team members and roles, and information on all collaborative arrangements. The current management approach and schedule is formed around the structure of the AAE-450 course project team. An in-depth description of project management approach and schedule is outside the scope and intended focus of the course. However, this lack of information does not translate into risk as the proposed cost reserves account for the management and operation of the project in the future with margin.

6 Cost & Cost Estimate Methodology

6.1 Cost of Investigation

The team utilized established cost estimating relationships (CERs) found in SME-SMAD to estimate a total mission lifecycle cost of \$134.7 M. Total costs with the consideration of contingencies and margins are represented in figure 29.

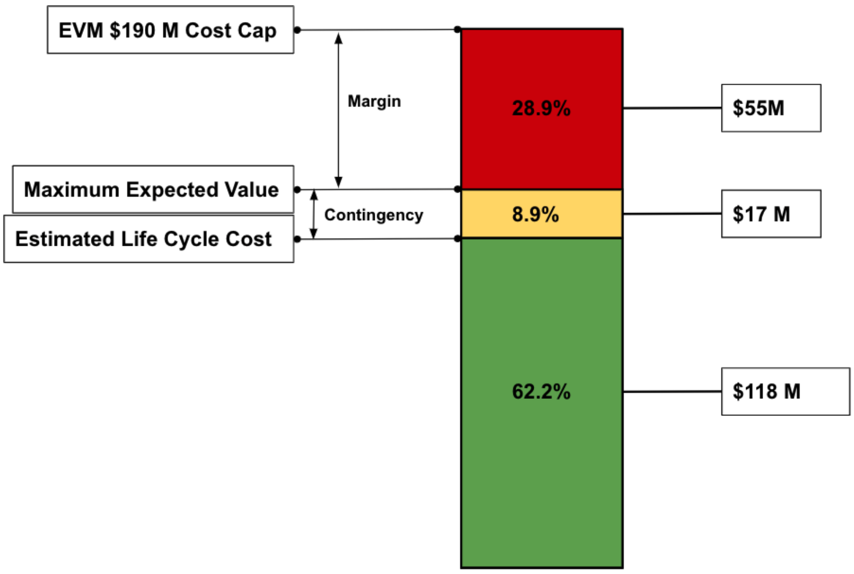


Figure 29: Visual Representation of Total Lifecycle Cost Inclusive of Cost Contingencies with Respect to the EVM-AO Cost Cap.

Establishing an integrated parametric model required the mission to be broken down into relevant work breakdown structure (WBS) elements and was broken down as shown in table 15. An important note is that SME-SMAD utilized a different WBS breakdown than the NASA standard used in cost tables B3a and B3b; for the remainder of this section, references to WBS elements follow the SME-SMAD definition.

1.0 Space Vehicle
1.1 Spacecraft Bus
1.1.1 Structure
1.1.2 Thermal
1.1.3 ADCS
1.1.4 EPS
1.1.5 Propulsion
1.1.6 TT&C
1.1.7 Spacecraft IA&T
1.1.8 Flight Software
1.2 Payload
1.2.1 Communications

1.2.2 Surveillance
1.2.3 Mission Software
1.2.4 Payload IA&T
2.0 Launch Vehicle (Includes WBS 5 Flight Support Operations)
3.0 Ground Command & Control
4.0 Program Level
6.0 Aerospace Ground Support Equipment (AGE)
7.0 Operations

Table 15: Work Breakdown Structure (WBS) for the MoIST Mission

6.2 Basis of Estimate

The cost model designed to estimate the total life cycle cost over the entire MoIST mission lifespan is a combination of the AAE-450 provided costs for the science instruments and payload in addition to the Aerospace Corporation's Small Satellite Cost Model (SSCM), the Constructive Cost Estimation Model (COCOMO81), and an SME-SMAD supplied operations cost model. Each WBS element cost breakdown considers recurring and non-recurring costs spanning the entire life cycle of the three year Earth remote sensing mission.

The SSCM is a historically-based non-recurring and recurring aggregate cost model generated from historical satellites under 100 kg but is applicable to missions using satellites measuring up to 400 kg (p. 299 in SME-SMAD [13]). The model is purposed for estimating costs for rapid development timelines, and thus, tends to overestimate costs for all WBS elements. We were able to verify this by comparing the commercial off the shelf (COTS) component provided cost for each subsystem to the estimated SSCM subsystem cost with the mass of the respective subsystem as an input. To compare the cost for each method of estimation for each subsystem, it is necessary to understand how the SSCM works. To approximate costs for each subsystem within the bus (WBS 1.1.1-1.1.6 from table 15), the SSCM takes the subsystem mass as an input and, through the use of an encoded CER, the program outputs an estimated cost. For services or processes (WBS 1.1.7,4.0, and 6.0 from table 15) costs are estimated based off of the total cost of the spacecraft bus. Since all costs are outputted in FY2010 dollars, NASA-defined inflation rates are then applied to adjust the outputted costs to FY2022 dollars. To directly apply the SSCM to our cost model we chose specific COTS components (detailed in the MEL) for each subsystem that fit our specific mission needs. For each subsytem, we attempted to find a cost straight from the supplier to apply to our model directly. Where the cost was not explicitly defined for the selected components, we used the SSCM to estimate the cost based on the mass of the selected component. In using the SSCM for a subsystem price estimate, we included a 25% mass contingency (in accordance with NASA GSFC Golden Rules [7]) to said subsystem and translated that into a cost contingency that was defined distinctly from our cost margins and base estimate, which is shown in figure 29.

Although it is not considered in the SSCM, another important factor in the team's cost estimate is the developmental cost of the flight software (WBS 1.1.8). For our purposes, we estimated the flight software (FSW) development costs using COCOMO81, which is a model generated from 63 completed software development programs (p. 304 in SME-SMAD [13]). This method utilizes an estimate for required source lines of code (SLOCs) for each spacecraft to generate a cost to design, code, and test the respective software. Since the COCOMO81 estimation tool is provided as a modifiable spreadsheet, the team had to establish what functions our spacecraft required to complete the mission objectives. After defining desired functions, we then broke down those needs into computer software components that have an associated SLOC estimate from historical data (p. 614 in SME-SMAD [13]) which yielded a 38,700 SLOC per spacecraft estimate. Finally, the model estimates the cost per SLOC for unmanned flight to be $550 in FY2010 dollars (p. 323 in SME-SMAD [13]), so we then calculated a flight software development cost for our satellite constellation in FY2022 dollars. A major assumption made for our software development is that we only generate the software once and then distribute it to each satellite in our constellation at no additional cost. This model estimates a FSW development cost of $27 M in FY2022 dollars.

To estimate the total cost of operations (WBS 7.0 from table 15) we utilized an SME-SMAD operations cost model that accounts for "mission operations, ground operations, training, sustaining engineers involved with the mission, the sustainment of the mission control center, spare flight hardware and maintenance, program/project management, and integration and communication services" (p. 311 in SME-SMAD [13]). This model uses a level-of-effort estimating technique that is applicable to low complexity or moderately complex scientific reconnaissance missions. In addition, a full-time-equivalent (FTE) estimation for annual salary and yearly overhead cost is used to calculate the number of engineers required for the space software maintenance for space (using the SLOC estimations for COCOMO81) while 25,000 SLOC were assumed for ground software as "the amount to be adapted from a much larger ground software package intended to manage multiple missions" (p. 312 in SME-SMAD [13]). From these estimations, the number of required FTE engineers (3) and technicians (2) were accounted for while our operations facilities were assumed to be 1,000 square meters at $1,250 per square meter. In summing these individual costs for operations labor, ground/space software maintenance, project management and systems engineering (PMSE), and facilities incorporated in the model a total operations cost of approximately $14.9 M was estimated over the lifespan of our mission. The operations architecture is discussed in greater detail in the mission operations section.

The remaining WBS element costs for launch (2.0 from table 15), flight support operations (5.0 from table 15), and ground command and control (3.0 from table 15) are established through current provided costs by suppliers that COTS components were selected for. For launch, the team decided to utilize two NASA provided dedicated launches at $14 M each as described in the launch services and compatibility section (4.3). The team assumed that this $28 M cost is inclusive of flight support operations that are required with the NASA provided launch vehicles. With respect to ground command and control costs, the team opted to utilize the Amazon Web Services (AWS) ground station network. Although explained in more depth in our downlink/uplink link budget requirements, a \$10 per link minute cost for AWS yielded a \$9 M total mission ground station cost estimate.

With explicit costs defined, we can now break down each cost to be more easily understood. Although our working model has costs broken down to specific subsystem levels, a higher-level summary of the MoIST team cost model can be found in table 16. Tables B3a and B3b, as provided in the cost tables section (6.4), break down the total lifecycle cost into each WBS element in real year (RY) and FY2022 dollars. Table B3a provides uninflated costs in each FY column and the properly inflated total RY cost in the RY columns. Table B3b also provides uninflated costs in their respective years of each mission phase and represents the total costs in uninflated FY2022 dollars.

In allocating costs for each mission phase in tables B3a and B3b, the total mission cost had to be approximated for each phase of the design. Theses budget allocation percentages were based off of AAE-590 Space Flight Operations - Lecture 5 [25]. For our cost estimation by mission phase, we assumed that all technology is inherited due to the high TRL and flight heritage of each selected component. Under that assumption, discrete percentages for each mission phase (A-E) allowed for our entire life cycle cost estimate to be allocated between mission phases. Following launch, our operations, aerospace ground equipment, and program level costs, were then allocated equally among the three year lifetime of the mission.

WBS Element	**Total Constellation Cost (FY 2022 \$K)**	**Source(s)**
1.0 Spacecraft (with contingencies)	78,569	SSCM COTS supplier prices COCOMO81 AAE450 Instrument Specs.
2.0 Launch Vehicle 5.0 Flight Support Operations	28,000	NASA EVM-AO Flight Support Operations
3.0 Ground Command & Control	9,000	Amazon Web Services Ground Station Network
4.0 Program Level	2,228	SSCM
6.0 Aerospace Ground Support Equipment	642	SSCM
7.0 Operations	16,259	Operations Cost Estimation Model
Total	**134,698**	

Table 16: High Level Breakdown of the MoIST Team Cost Model

6.3 Cost Risk

With nearly 30% of the AO cost cap being reserved for margins (with 30% cost margin being the suggested amount by NASA GSFC [7]), the most significant risks to exceeding budget come from the scheduling risks of system integration delay and launch delay. Given that our satellite uses largely COTS components, a smooth process of system integration is not guaranteed. To mitigate this risk and avoid setbacks, we chose components with high modularity and flexibility to integrate with other parts. We also prioritized high TRL (6+) components with flight heritage, ensuring successful historical integration with other missions. Based on historical data, the chance of launch delay is high for small satellite missions [28]. To mitigate this risk, we elected to use a dedicated launch vehicle on which our satellites are the primary payload. This reduces the budget impact of a potential launch delay and provides more flexibility of scheduling.

6.4 Cost Tables

As described previously, we established our original cost model with the WBS elements described in table 15. For this proposal, we will map our cost model onto the desired EVM cost profile spreadsheets in real year and fiscal year dollars. To represent our cost model in the desired form, there must be an understanding of how our WBS elements translate to those in the cost profile. Although our model and the desired cost profiles spreadsheet include the same WBS elements and considerations, the breakdowns are not identical. Figure 30 visualizes a more specific breakdown than our cost model encompassed; for example, WBS element 4 includes systems engineering costs but are not explicitly defined separately from the program level costs.

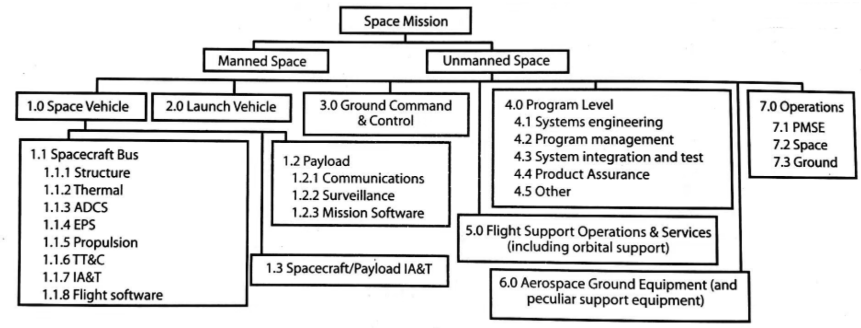


Figure 30: In-Depth Breakdown of Each MoIST WBS Element for Mapping to Cost Tables

With the method for mapping our cost model onto the desired template explained, we can now generate tables B3a and B3b to examine the total cost of our mission lifecycle in the correct format. After considering the various WBS elements, the final expected cost with contingencies is about \$160 M in RY dollars and \$154 M in FY2022 dollars.An important note for the results shown in the cost tables is that no learning curve was applied to the spacecraft component costs to show the raw costs of each part throughout our constellation. With this consideration made clear, slight differences in the MoIST cost model and the translated model onto the cost tables are understood.

WBS#	WBS Element	Phase A FY2022	Phase B FY2022	Phase C/D FY2022	Phase C/D FY2023	Phase C/D FY2024	Phase C/D Total	RY$K A-D Total	Phase E FY2026	Phase E FY2027	Phase E FY2028	Phase E Total	RY$K A-E Total
01	Project Management												
02	Systems Engineering												
03	Safety & Mission Assurance												
04	Science / Technology	17.4	69.5	799.4	295.4	556.1	1650.9	1776.2	163.4	163.4	163.4	490.2	2336.4
05	Payload(s)	22.0	88.0	1012.0	374.0	704.0	2090.0	2248.6					2248.6
06	Spacecraft	838.1	3355.6	38586.4	14260.4	26842.4	79689.2	85737.0					85737.0
	Structure	174.0	696.0	8010.0	2960.4	5571.6	16542.0						
	Thermal	91.3	364.8	4201.2	1552.8	2922.0	8676.0						
	ADCS	60.0	240.0	2760.0	1020.0	1920.0	5700.0						
	EPS	34.8	141.6	1622.4	600.0	1129.2	3351.6						
	Propulsion	61.2	244.8	2809.2	1038.0	1954.8	5802.0						
	TT&C												
	CD&H	146.4	586.8	6745.2	2492.4	4692.0	13929.6						
	Flight Software	270.4	1081.6	12438.4	4596.8	8652.8	25688.0						
07	Mission Operations								5419.7	5419.7	5419.7	16259.1	18580.2
	PMSE								706.9	706.9	706.9	2120.7	
	Space Segment SW Maintenance								668.8	668.8	668.8	2006.4	
	Mission Operations Labor								1935.4	1935.4	1935.4	5806.2	
	Ground Segment SW Maintenance								245.1	245.1	245.1	735.3	
	Ground HW Maintenance								135.5	135.5	135.5	406.5	
	Facility Lease								1728.0	1728.0	1728.0	5184.0	
08	Launch Vehicle / Services								28000.0			28000.0	28000.0
09	Ground System(s)								3214.0	3214.0	3214.0	9642.0	11018.5
	Aerospace Ground Equipment								214.0	214.0	214.0	642.0	
	Ground Command and Control (AWS)								3000.0	3000.0	3000.0	9000.0	
10	Systems Integration & Testing	113.6	454.3	5224.7	1930.9	3634.6	10790.2	11609.2					11609.2
	Budget Reserves	444.6	1778.4	20451.6	7558.2	14227.2	42237.0	45442.7	4180.0	4180.0	4180.0	12540.0	59772.9
	Total Cost Without Reserves	**991.1**	**3967.4**	**45622.5**	**16860.7**	**31737.1**	**94220.3**	**101371.0**	**36797.1**	**8797.1**	**8797.1**	**54391.3**	**159529.9**

Table B3a: FY Costs and Totals in Real Year Dollars (RY$K)

WBS#	WBS Element	Phase A FY2022	Phase B FY2022	Phase C/D FY2022	Phase C/D FY2023	Phase C/D FY2024	Phase C/D Total	Phase E FY2026	Phase E FY2027	Phase E FY2028	Phase E Total	FY2022$ Total
01	Project Management											
02	Systems Engineering											
03	Safety & Mission Assurance											
04	Science / Technology	17.4	69.5	799.4	295.4	556.1	1650.9	163.4	163.4	163.4	490.2	2228.0
05	Payload(s)	22.0	88.0	1012.0	374.0	704.0	2090.0					2200.0
06	Spacecraft	838.1	3355.6	38586.4	14260.4	26842.4	79689.2					83882.9
	Structure	174.0	696.0	8010.0	2960.4	5571.6	16542.0					
	Thermal	91.3	364.8	4201.2	1552.8	2922.0	8676.0					
	ADCS	60.0	240.0	2760.0	1020.0	1920.0	5700.0					
	EPS	34.8	141.6	1622.4	600.0	1129.2	3351.6					
	Propulsion	61.2	244.8	2809.2	1038.0	1954.8	5802.0					
	TT&C											
	CD&H	146.4	586.8	6745.2	2492.4	4692.0	13929.6					
	Flight Software	270.4	1081.6	12438.4	4596.8	8652.8	25688.0					
07	Mission Operations							5419.7	5419.7	5419.7	16259.1	16259.1
	PMSE							706.9	706.9	706.9	2120.7	
	Space Segment SW Maintenance							668.8	668.8	668.8	2006.4	
	Mission Operations Labor							1935.4	1935.4	1935.4	5806.2	
	Ground Segment SW Maintenance							245.1	245.1	245.1	735.3	
	Ground HW Maintenance							135.5	135.5	135.5	406.5	
	Facility Lease							1728.0	1728.0	1728.0	5184.0	
08	Launch Vehicle / Services							28000.0			28000.0	28000.0
09	Ground System(s)							3214.0	3214.0	3214.0	9642.0	9642.0
	Aerospace Ground Equipment							214.0	214.0	214.0	642.0	
	Ground Command and Control (AWS)							3000.0	3000.0	3000.0	9000.0	
10	Systems Integration & Testing	113.6	454.3	5224.7	1930.9	3634.6	10790.2					11358.1
	Budget Reserves	444.6	1778.4	20451.6	7558.2	14227.2	42237.0	4180.0	4180.0	4180.0	12540.0	57000.0
	Total Cost Without Reserves	**991.1**	**3967.4**	**45622.5**	**16860.7**	**31737.1**	**94220.3**	**36797.1**	**8797.1**	**8797.1**	**54391.3**	**153570.1**

Table B3b: FY Costs and Totals in Fiscal Year 2022 Dollars (FY2022$)

6.5 Approved Forward Pricing Statement

Any cost associated with the proposed MoIST mission is represented in Fiscal Year 2022 dollars as requested in the EVM-AO.

7 Appendix

7.1 Table of Proposal Participants

Name	**Main Project Role**	**Supporting Project Roles**
Ryan Alcorn	Project Manager	Risk Management, Cost Analysis, Thermal Analysis, Launch Selection
Michael Berthin	Cost Analysis	Structures, Mission Design, Power Systems
Benjamin Durkee	Space Environment	
Joshua Fitch	FSW / C&DH / Telemetry	Mission Design, Attitude Dynamics & Control
Pol Francesch Huc	Mission Design	Propulsion, Power Systems, Processor System, Systems Engineering
Cody Mancini	Systems Engineering	Mission Design
Kenneth Pritchard	Risk Management	Power Systems, Launch Selection
Simon Sasin	Structures	Mission Design, Cost Analysis, C&DH
Shruti Sundaresh	Propulsion	Mission Design

7.2 Discussion of Limiting the Generation of Orbital Debris and End of Mission Spacecraft Disposal Requirements

Throughout the course of our mission, the only potential debris generation necessary for mission success will be the booster system, fairing deployment, and the Hold Down/Release Mechanisms (HDRM) [9].

For the 350 km orbital plane, our satellites will fly aboard a Northrop Grumman Minotaur 1, which is an expendable rocket and could therefore become threatening orbital debris if left on the orbital plane. However, the launch system is capable of more Delta V than is necessary for our payload, and therefore the upper stage assembly can certainly be deorbited after deployment of the satellites.

For the 550 km orbital plane, our satellites will fly aboard a Northrop Grumman Pegasus XL, which is deployed from reusable first stage aircraft, but the second and third stages are expendable. We will have to confirm with Northrop Grumman that the mission plan includes enough Delta V in the orbital insertion (3rd) stage booster to deorbit it after payload deployment.

In terms of fairing deployment, on both launch systems the fairings are deployed during the ascent stage at a point where the atmospheric exposure is negligible for the safety of the payload and the aerodynamics of the launch system, but the trajectory is still shallow enough that the fairing debris will deorbit rapidly and should not pose a threat to any spacecraft.

For the HDRMs, we selected the EBAD NEA Model 9100 [9]. This HDRM only has two parts: a base release mechanism and a release rod. Each of these parts are fixed to the objects they are securing together, so when the mechanism actuates there is no debris generated.Therefore, following these standards with our launch procedures, we will not be generating any orbital debris.

7.2.1 Satellite De-orbit Plan

Additionally, our 350 km satellites require stationkeeping burns, and without those burns they deorbit within 90 days. Our 550 km satellites will naturally deorbit in 22 years, but we will equip them with more than enough propellent to actively de-orbit them after the mission lifetime has elapsed.

This way, given the survival of all of our satellites, we will not leave any debris for an extended period after our mission lifetime.

7.2.2 Orbit Decay Model

What we want:

- Station-keeping period (days between each orbit raising maneuver).
- Optimal altitude to raise orbit from (minimize propellant mass).

- Total propellant mass needed per satellite across the 3-year mission.

Assumptions:

- 180-550 km starting altitude.
- Re-entry occurs surely and quickly at 180 km.
- Circular orbits.
- Decay due to atmospheric drag only.
- Solar Radio Flux 10 cm - this is generally used in average form, and the average preferred is that of the last 90 days prior to the specified date.
- Assuming attitude control is achieved, this negates tumbling and any variance in cross-sectional area.
- Orbit raising maneuver is a Hohmann transfer.

Inputs:

- Inert Mass = 50 kg.
- Surface area = 1.0 m^2
- Altitude = 350 km or 550 km.
- Isp = 285 s (from propulsion component selection).
- Solar Radio flux (space environment) = 70 (historical average).

Methodology

The decay model used was taken from The Australian Space Weather Agency [29]. The inputs and assumptions of the model are listed above. The model is iterative and calculates the time it takes for an object to decay from a starting altitude. Using this data we could then calculate the delta V needed to return the satellite to its original orbit from each decayed orbit, the propellant mass needed for this maneuver, how many times this maneuver would need to occur within the mission duration, and the total amount of propellant needed to station-keep for the mission per satellite. From this model it was clear that the satellites in the 350 km orbit decayed significantly faster than the satellites in the 550 km orbit.

The propellant mass needed to deorbit uses the same assumptions of the deorbit model, namely the assumption that deorbit occurs surely and swiftly after the orbit decays to 180 km. Calculating the Delta-V needed to perform a Hohmann transfer to this altitude from a higher altitude subsequently allowed for the calculation of the amount of propellant needed for the maneuver. The faster decay of the 350 km satellites (decays to 180 km within 90 days) meant that propellant mass allocation for deorbiting was not necessary. On the other hand, the slower decay of the 550 km satellites (decays to 180 km in 22 years) required for there to be propellant mass dedicated for immediate deorbit or orbit lowering if needed. Results of the calculations described are summarized below in table 17.

Propellant Mass Calculations (Orbit Decay Model)		
Altitude	350 km	550 km
Days to decay to New Altitude	15.5	1552.6
New Altitude	339.96 km	539.99 km
dV to raise	2.87 m/s	2.74 m/s
Prop Mass to raise	0.036 kg	0.034 kg
Number of times to raise over mission	70.65	0.71
Total Prop Mass Needed (to raise)	2.54 kg	0.024 kg
dV to deorbit	0 m/s	~ 100 m/s
Total Prop Mass Needed (to deorbit)	0 kg	1.82 kg
Total Prop Mass need per satellite across mission	2.54 kg	1.84 kg

Table 17: Full propellant mass calculations for both orbital planes using the aforementioned orbit decay model

7.3 Space Environment Analysis

Debris Analysis

Using SPENVIS's [30] orbital debris calculator, we determined mass flux data for both orbital planes. We then multiplied that mass flux by the mission duration and the worst case frontal surface area of the craft when fully deployed to derive a series of lambdas for a Poisson distribution.

We then plugged those lambdas into the Poisson distribution formula and substituted 0 for x to find the probability that there would be no collisions for each possible debris diameter. We then subtracted this probability from 1. This gave us a distribution of the probability of at least 1 collision as a function of debris diameter, which is plotted in figures 31 and 32 for both orbital planes.

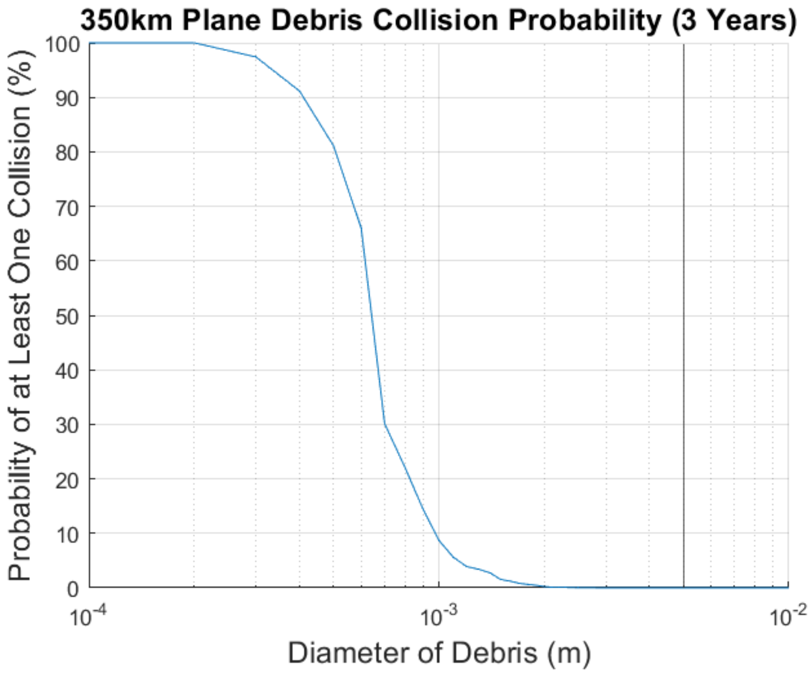


Figure 31: Probability of collision with debris at the 350 km orbit. Highlights debris with a diameter of 5mm

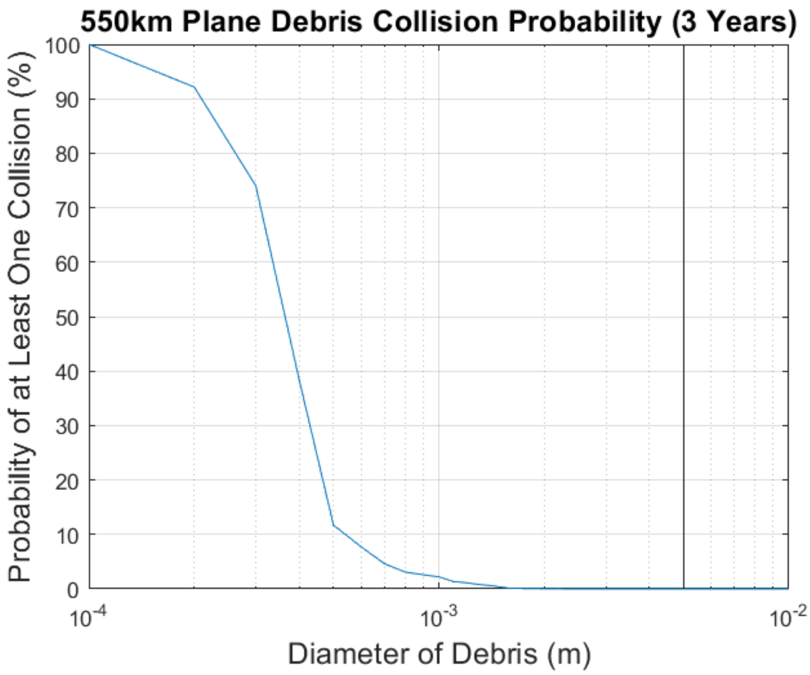


Figure 32: Probability of Collision with Debris at the 550 km Orbit (Highlights Debris with a Diameter of 5mm)

From a paper entitled "Removing Orbital Debris with Less Risk" [31] published by the American Institute of Aeronautics and Astronautics, we found that debris with 5mm diameter or greater constitutes debris that may render the satellite inoperable. Therefore, we considered this our cutoff for catastrophic debris collisions.

From this we summed the probability of collision for all debris with a diameter greater than or equal to 5mm, giving us the probability that a single satellite will experience a catastrophic collision over the course of its mission lifetime, for each orbital plane. We then multiplied these by the quantity of satellites in the corresponding plane, resulting in an overall probability of 0.5302% for us to experience a single catastrophic collision over the course of our mission.

Orbital Plane	Probability of Catastrophic Debris Collision
350 km (Single Satellite)	0.0461%
550 km (Single Satellite)	0.0414%
All Satellites (Total)	0.5302%

Table 18: Summary of Figures 31 and 32

Radiation Analysis

Using R-GENTIC's [32] Total Ionizing Dose (TID) data for historical missions with trajectories similar to ours, we based our radiation shielding thickness on the results of these missions. As illustrated in figures 33 and 34, the worst case scenario for our satellites is a TID of 1 krad/year on the 550 km plane, with a minimum shielding depth of approximately 1.5 mm. In our current design, we opted to add a 20% margin to this, resulting in a shielding thickness of 1.8 mm.

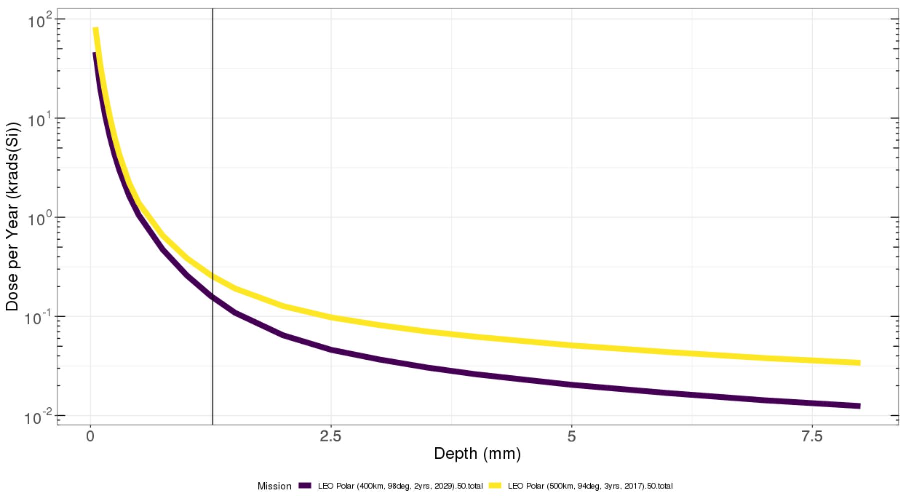


Figure 33: Radiation Dose per Year of Satellites in Similar Orbits as the 350 km Plane. Highlights the Recommended Minimum Radiation Shielding Depth

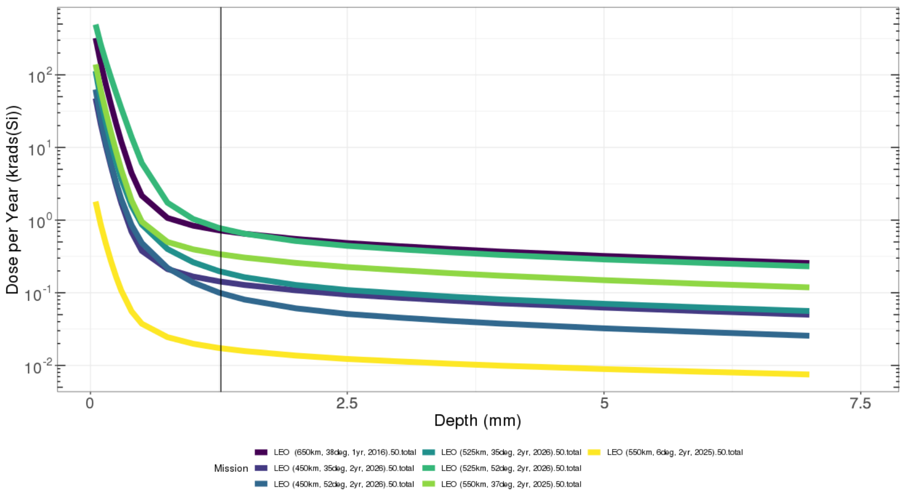


Figure 34: Radiation Dose per Year of Satellites in Similar Orbits as the 550 km Plane (Highlights the Recommended Minimum Radiation Shielding Depth)

8 Master Equipment List

Payload		# OF UNITS	FLIGHT HARDWARE MASSES	FLIGHT HARDWARE POWER
Subsystem/Component	Unit Mass, Current Best Estimate (CBE)	Flight Units	Total Mass, kg CBE	Total Power, W CBE
Science Instrument (Receiver)	7.00	1	7.00	25.00
P/I-Band Antenna	2.30	1	2.30	1.00
L-Band Antenna	2.25	3	6.75	3.00
GNSS Antenna (3G1215A-MTR-5)	0.224	1	0.224	1.00
Total Mass/Power			**16.274**	**30.00**

Figure 35: Equipment List for Payload

Structures and Mechanisms		# OF UNITS	FLIGHT HARDWARE MASSES	FLIGHT HARDWARE POWER
Subsystem/Component	Unit Mass, Current Best Estimate (CBE)	Flight Units	Total Mass, kg CBE	Total Power, W CBE
Empty S/C Bus	1.12	1	1.12	0.00
Shielding Walls	0.44	5	2.20	0.00
Antenna Mounting Table	4.00	1	4.00	0.00
MOOGS Type 1 Solar Array Drive	1.16	1	1.16	8.60
HDRM NEA Model 9100	0.07	7	0.49	0.00
Connectors, Brackets, etc			1.83	0.00
Total Mass/Power			**10.80**	**8.60**

Figure 36: Equipment List for Structures and Mechanisms

Thermal Control		# OF UNITS	FLIGHT HARDWARE MASSES	FLIGHT HARDWARE POWER
Subsystem/Component	Unit Mass, Current Best Estimate (CBE)	Flight Units	Total Mass, kg CBE	Total Power, W CBE
Aluminized Teflon 2mil Surface Finish	0.047	1	0.047	0.00
SHK Kapton Flexible Heater	0.00228	1	0.00228	25.00
Total Mass/Power			**0.04928**	**25.00**

Figure 37: Equipment List for Thermal Control

Power		# OF UNITS	FLIGHT HARDWARE MASSES	FLIGHT HARDWARE POWER
Subsystem/Component	Unit Mass, Current Best Estimate (CBE)	Flight Units	Total Mass, kg CBE	Total Power, W CBE
Solar Array	1.51	1	1.51	0.00
AAC Clyde Space OPTIMUS-80 Battery	0.67	6	4.02	1.20
EnduroSat EOS II Power Management System	1.28	1	1.28	0.00
Total Mass/Power			**6.81**	**1.20**

Figure 38: Equipment List for Power Systems

TT&C		# OF UNITS	FLIGHT HARDWARE MASSES	FLIGHT HARDWARE POWER
Subsystem/Component	Unit Mass, Current Best Estimate (CBE)	Flight Units	Total Mass, kg CBE	Total Power, W CBE
S-Band Patch Antenna	0.05	3	0.15	2.00
NANOLink BOOST SDR S-Band Transceiver	0.17	3	0.51	9.00
S-Band Diplexer	0.22	3	0.66	3.00
Total Mass/Power			**1.32**	**14.00**

Figure 39: Equipment List for TT&C

On-Board Processing		# OF UNITS	FLIGHT HARDWARE MASSES	FLIGHT HARDWARE POWER
Subsystem/Component	Unit Mass, Current Best Estimate (CBE)	Flight Units	Total Mass, kg CBE	Total Power, W CBE
Xiphos Q7S Processor	0.032	1	0.032	2.00
Delkin Industrial SD Card	0.002	2	0.004	0.00
Total Mass/Power			**0.036**	**2.00**

Figure 40: Equipment List for On-Board Processing

ADCS		# OF UNITS	FLIGHT HARDWARE MASSES	FLIGHT HARDWARE POWER
Subsystem/Component	Unit Mass, Current Best Estimate (CBE)	Flight Units	Total Mass, kg CBE	Total Power, W CBE
CubeADCS Module	1.00	1	1.00	4.50
Total Mass/Power			**1.00**	**4.50**

Figure 41: Equipment List for ADCS

Propulsion		# OF UNITS	FLIGHT HARDWARE MASSES	FLIGHT HARDWARE POWER
Subsystem/Component	Unit Mass, Current Best Estimate (CBE)	Flight Units	Total Mass, kg CBE	Total Power, W CBE
Dawn Aerospace 1.5U Propulsion Module	1.30	4	5.20	3.20
Total Mass/Power			**5.20**	**3.20**

Figure 42: Equipment List for Propulsion Systems

9 Heritage

SigNals of Opportunity P-Band Investigation (SNoOPI) is a scientific investigation mission to demonstrate the measurement of the complex reflection coefficient over various land surface conditions for use in future CubeSat constellations. The primary investigator on the project is James Garrison, a professor in the School of Aeronautics and Astronautics. The mission plans to use P-band transmitters to gather soil moisture around the globe. SNoOPI is the pathfinder mission for MoIST, demonstrating the technology used in signals of opportunity, so that MoIST can add the capability of using L-Band for MoIST's mission objectives. The science instrument is also derived from SNoOPI's instrument, modified for taking both P and L-band science measurements.

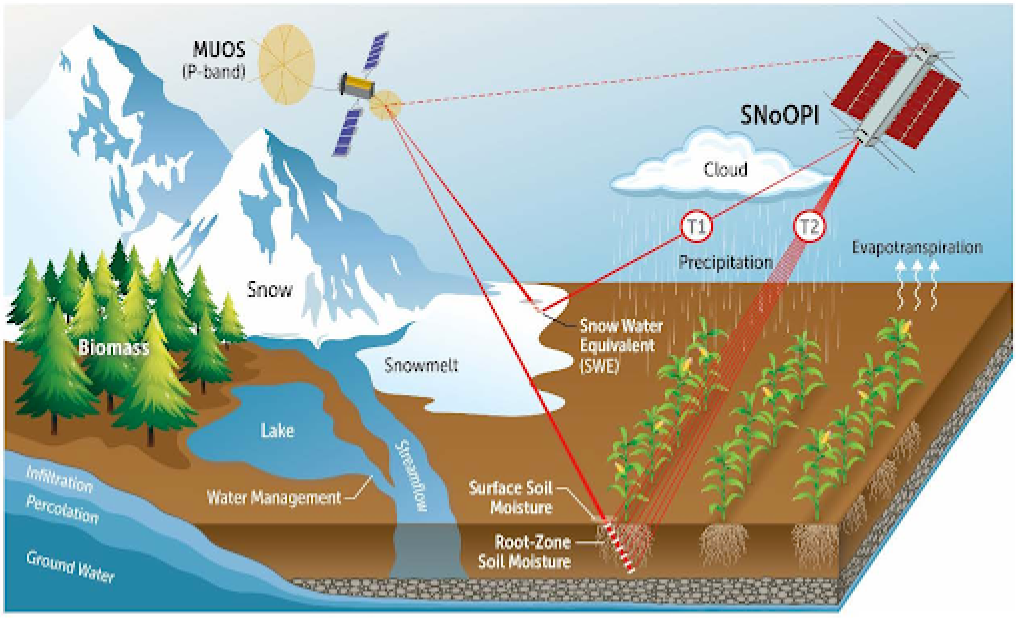


Figure 43: Highlight of SNoOPI's Mission Objectives

Table 19 below shows flight heritage information for all COTS components selected for our spacecraft.

Subsystem	Component	Heritage Examples
Payload	Science Instrument (Receiver)	SNoOPI (TRL-6) [33]
	P/I-Band Antenna	SNoOPI (TRL-6) [33]
	L-Band Antenna	TRL-9 Flight heritage [33]
	GNSS Antenna	TRL-9 Flight heritage [33]
Structures and Mechanisms	ISISPACE 6-Unit CubeSat Structure	Flight heritage since 2016 [34]
	Shielding Walls	9 historic missions with comparable architectures to MoIST (as analyzed by R-GENTIC) TRL-9 [32]
	MOOGS Type 1 Solar Array Drive	Mighty Sat, Wild Geese, P925, STP/SIV [18]
	HDRM NEA Model 9100	Flight heritage since 2000 [9]
Thermal Control	Aluminated Teflon 2mil Surface Finish	Hubble Space Telescope [35]
	SHK Kapton Flexible Heater	Compass-1, MASAT-1, OUTFI-1 [36]
Power	MMA Design HaWK Solar Array	Flight heritage since 2017, MarCO [17]
	AAC Clyde Space OPTIMUS-80 Battery	Flight heritage for 10+ years, over 100 missions [19]
	EnduroSat EPS II Power Management System	Flight-proven Electric Power System [20]
TT&C	S-Band Patch Antenna	TRL-9 Flight Heritage since 2021 [37]
	S-Band Diplexer	TRL-9 Flight Heritage [38]
On-Board Processing	Xiphos Q7S Processor	Operating in orbit since 2016, used on ISS, certified for manned space flight [15]
	Delkin Industrial SD Card	TRL-9 Flight heritage [16]
ADCS	CubeADCS Module	80+ ADCS modules flown in orbit, 140+ delivered to date, first successful flight in 2014 [39]
Propulsion	Dawn Aerospace 1.5U Propulsion Module	3 missions: Hiber Four CubeSat, Hiber Three CubeSat, D-Orbit's ION SVC Lucas [10]

Table 19: Heritage of COTS Components

References

[1] D. Eberly, "Computing a point of reflection on a sphere," Feb 2008. [Online]. Available: https://www.geometrictools.com/Documentation/SphereReflections.pdf

[2] W. J. Miller, J. W. Barnes, and S. M. MacKenzie, "Solving the alhazen–ptolemy problem: Determining specular points on spherical surfaces for radiative transfer of titan's seas," *The Planetary Science Journal*, vol. 2, no. 2, p. 63, mar 2021. [Online]. Available: https://doi.org/10.3847/psj/abe4dd

[3] G. Van Brummelen, *Heavenly mathematics the forgotten art of spherical trigonometry*, course book ed. Princeton: Princeton University Press, 2013.

[4] NASA, "Commercial faa licensed lsp information summary," 2021. [Online]. Available: https://explorers.larc.nasa.gov/2021APMIDEX/pdf_files/CommercialFAALicensed_LSP_InformationSummary%202021-04-30.pdf

[5] SpaceX, "Spacex smallsat rideshare program," 2021. [Online]. Available: https://www.spacex.com/rideshare/

[6] NASA, "Nasa launch vehicle performance website," 2021. [Online]. Available: https://elvperf.ksc.nasa.gov/Pages/Query.aspx

[7] N. GSFC, "Rules for the design, development, verification, and operation of flight systems," 2016. [Online]. Available: https://standards.nasa.gov/standard/gsfc/gsfc-std-1000

[8] T. P. Classroom, "Fundamental frequency and harmonics," 2013. [Online]. Available: https://www.physicsclassroom.com/class/sound/Lesson-4/Fundamental-Frequency-and-Harmonics

[9] EBAD, "Nea model 9100 hold down & release mechanism," 2021. [Online]. Available: https://www.ebad.com/wp-content/uploads/2020/06/EBAD-2020-NEA-Model-9100-Hold-Down-Release-Mechanism-RND4.pdf

[10] D. Aerospace, "Dawn aerospace cubesat propulsion modules," Oct 2021. [Online]. Available: https://static1.squarespace.com/static/5e82736a5e6bb91e8af13ea7/t/618487cb952c2571c74c0e8e/1636075477416/DA+CubeSat+Propulsion+Module+Specifications.pdf

[11] A. W. Services, "Aws ground station overview," February 2020. [Online]. Available: https://esto.nasa.gov/wp-content/uploads/2020/03/AWS-FCD_GroundStation_Feb2020-NOS.pdf

[12] K. T. Nock, K. M. Aaron, and D. McKnight, "Link analysis of a telecommunication system on earth, in geostationary orbit, and at the moon: Atmospheric attenuation and noise temperature effects," February 2007. [Online]. Available: https://ipnpr.jpl.nasa.gov/progress_report/42-168/168E.pdf

[13] J. R. Wertz, D. F. Everett, and J. J. Puschell, *Space Mission Engineering: The New SMAD.* Hawthorne, CA: Microcosm Press, 2011.

[14] J. King, "Jan king link budget excel spreadsheet," 2013. [Online]. Available: http://www.amsatuk.me.uk/iaru/spreadsheet.htm

[15] Xiphos, "Q7s specification - datasheet," 2021. [Online]. Available: http://xiphos.com/wp-content/uploads/2015/06/XTI-2001-2020-e-Q7S-Spec-Sheet.pdf

[16] Delkin, "Industrial sd memory cards," 2021. [Online]. Available: https://www.delkin.com/products/industrial-sd/

[17] M. Design, "Hawk specifications for comparison." [Online]. Available: https://mmadesignllc.com/specs-table/

[18] Moog, "Type 1 solar array drive assembly (sada)," 2019. [Online]. Available: https://www.moog.com/content/dam/moog/literature/Space_Defense/spaceliterature/spacecraft_mechanisms/moog-type-1-solar-array-drive-assembly-datasheet.pdf

[19] A. C. Space, "Optimus datasheet," 2021. [Online]. Available: https://www.aac-clyde.space/wp-content/uploads/2021/10/AAC_DataSheet_Optimus.pdf

[20] Endurosat, "Eps power module ii," 2021. [Online]. Available: https://www.endurosat.com/cubesat-store/cubesat-power-modules/eps-power-module-ii/

[21] Mathworks, "Simulink cubesat simulation blockset," 2021. [Online]. Available: https://www.mathworks.com/help/aeroblks/model-and-simulate-cubesats.html

[22] CubeSpace, "Cubeadcs 3-axis integrated module," 2021. [Online]. Available: https://www.cubespace.co.za/products/integrated-adcs/3-axis/

[23] J. Henninger, "Solar absorptance and thermal emmitence of some common spacecraft thermal-control coatings," 1984. [Online]. Available: https://ntrs.nasa.gov/citations/19840015630

[24] Kapton, "Flexible heater design guide," 2020. [Online]. Available: https://www.tempco.com/Tempco/Resources/09-Flexible-Resources/KaptonCatalogPages.pdf

[25] J. Mansell, "Aae 590: Spaceflight operations - lectures," 2021.

[26] A. W. Services, "Aws ground station," 2021. [Online]. Available: https://aws.amazon.com/ground-station/

[27] G. S. F. Center, "Gpr 7120.4d: Risk management," Aug 2012.

[28] BryceTech, "Smallsat launch delays," 2020. [Online]. Available: https://brycetech.com/reports/report-documents/Bryce_Metrics_Smallsat_Launch_Delays_2020.pdf

[29] J. Kennewell, "Satellite orbital decay calculations," 1999. [Online]. Available: https://www.sws.bom.gov.au/Category/Educational/Space%20Weather/Space%20Weather%20Effects/SatelliteOrbitalDecayCalculations.pdf

[30] ESA, "The space environment information system (spenvis)," 2021. [Online]. Available: https://www.spenvis.oma.be/

[31] K. T. Nock, K. M. Aaron, and D. McKnight, "Removing orbital debris with less risk," *Journal of Spacecraft and Rockets*, vol. 50, no. 2, p. 15, mar and apr 2013. [Online]. Available: https://arc.aiaa.org/doi/pdf/10.2514/1.A32286

[32] NASA, "R-gentic," 2021. [Online]. Available: https://vanguard.isde.vanderbilt.edu/RGentic/

[33] J. Garrison, "Aae-450: Senior spacecraft design - lecture slides," 2021.

[34] ISISPACE, "Isispace 6 unit cubesat structure," 2021. [Online]. Available: https://www.isispace.nl/product/6-unit-cubesat-structure/

[35] K. . B. B. . T. J. Dever, Joyce & de Groh, "Effects of radiation and thermal cycling on teflon ® fep. high performance polymers," mar 1999. [Online]. Available: https://www.researchgate.net/publication/241649116_Effects_of_Radiation_and_Thermal_Cycling_on_Teflon_R_FEP

[36] NASA, "Smallsat institute - thermal control," 2021. [Online]. Available: https://www.nasa.gov/smallsat-institute/sst-soa/thermal-control

[37] ISISPACE, "Isispace s-band patch antenna," 2021. [Online]. Available: https://www.isispace.nl/product/s-band-patch-antenna/

[38] W. Poland, "Wiran s-band diplexer," 2021. [Online]. Available: https://www.wiran.pl/en/s-band-diplexer

[39] CubeSpace, "Cubespace company about page," 2020. [Online]. Available: https://www.cubespace.co.za/about/