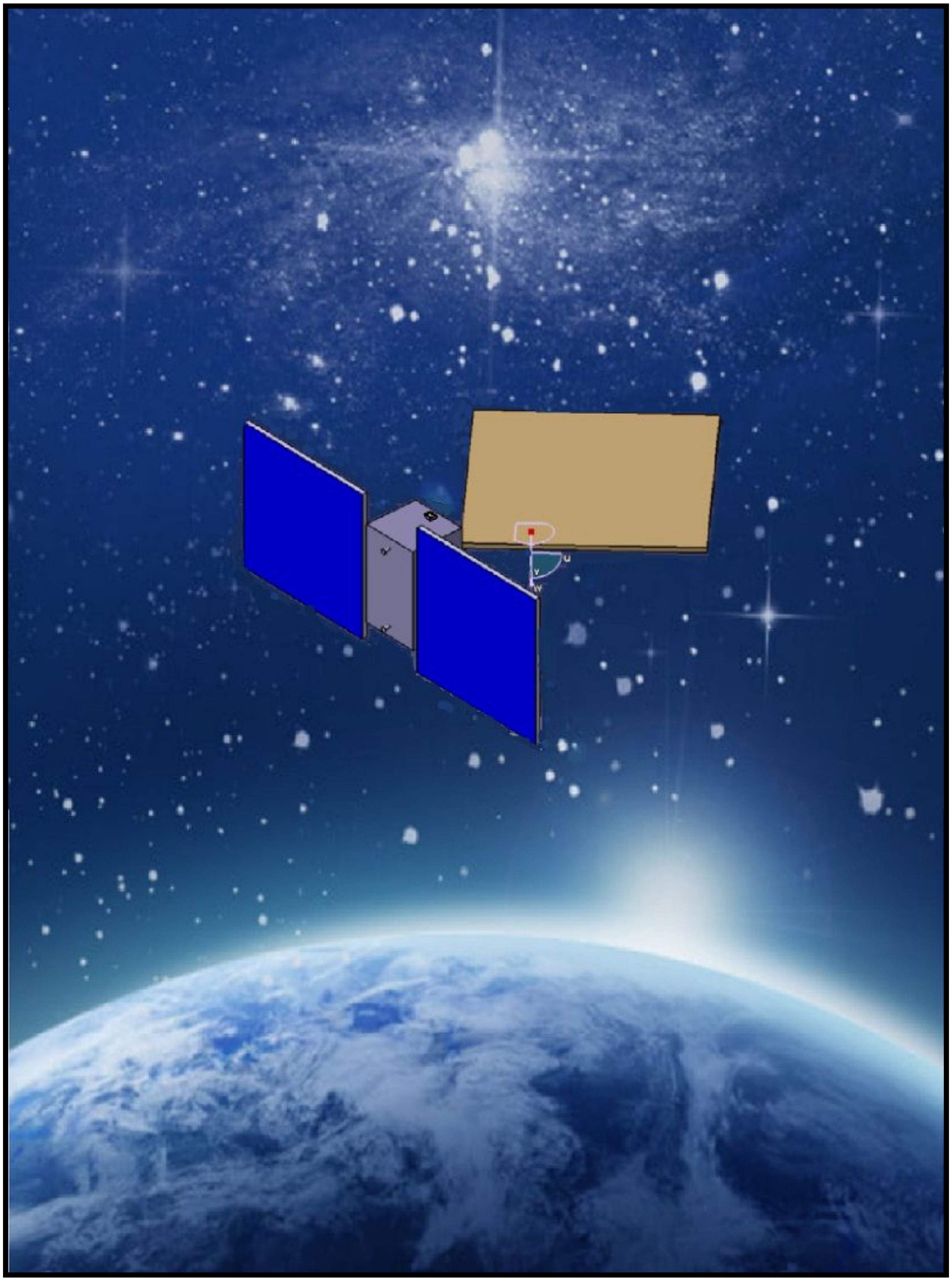
Project GAEA



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B. Fact Sheet

Mission Overview

Project GAEA is a three year mission proposal to NASA to collect soil moisture data, which is extremely powerful in weather forecasting, drought and flood prediction, crop production, and analyzing the water and carbon cycles. Signals of Opportunity will be used to capture the direct and reflected signals of various operational constellations. By capturing measurements of root zone soil moisture (5-100 cm) we will gain a much deeper look at the Earth's profile compared to SMAP's *estimate* of soil moisture in the root zone. The GAEA mission will additionally collect freeze and thaw measurements on land.

Launching in late 2026 the GAEA constellation's primary science objectives will be to:

1. Understand the processes that link the terrestrial water, carbon and energy cycles
2. Estimate the global water and energy fluxes at the land surface
3. Quantify net carbon flux in boreal landscape
4. Enhance weather and climate forecast
5. Develop improved drought monitoring and flood prediction capability

Project GAEA will support the development of twelve spacecraft to Low Earth Orbit via the Minotaur I. Eight of these spacecraft will orbit Earth at an inclination of 70 degrees. The remaining four will orbit at 85 degrees and orbit at 500 km altitude. The two sets of spacecraft will be evenly spaced in true anomaly. The total cost to design, manufacture, and operate GAEA comes to $181 M.

The instruments on board will be able to receive signals of opportunity in the P, L, and VHF frequency bands. For L band, GAEA will track signals from GNSS including Galileo, GPS, Beidou, GLONASS. For P band, GAEA will only track MUOS. For VHF, GAEA will track Orbcomm, Meteor, and NOAA. X and S band transmission will be used to communicate between GAEA and ground stations in the NASA near Earth network (NEN).

Once soil moisture data has been processed it will be sent to various organizations including weather forecasters from several nations and researchers and planners from the U.S. Department of Agriculture, U.S. Geological Survey, U.S. Centers for Disease Control and Prevention, U.N. World Food Programme and other organizations.

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Proposal Structure and Page Limits

D. Science Objectives and Requirements

1. Science Traceability Matrix

The science objectives and scientific measurement requirements are provided by the PI. The instrument function requirements are the same as SNOOPI as GAEA will use the same instrument as SNOOPI. The mission function requirements were derived from the other STM information provided by PI and SNOOPI.

Science Objectives	**Scientific Measurement Req**	**Instrument Func Req**	**Mission Func Req**
Understand processes that link the terrestrial water, energy and carbon Cycles; Estimate global water and energy fluxes at the land surface	Soil Moisture: ~4% volumetric accuracy in top 5 cm for vegetation water content < 5 kg m^-2 Hydrometeorology at 10 km; Hydroclimatology at 40 km;	<u>Surface Soil Moisture:</u> L band reflectivity Resolution - 10 km Soil Depth - 5 cm Incidence Angle <60 deg <u>Root Zone Soil Moisture:</u> P and I Band reflectivity Resolution - 40 km Soil Depth - 1 m Incidence Angle <60 deg	NASA data archiving and distribution Field validation testing P and VHF Band measurements for root zone within 12 hours
Quantify net carbon flux in boreal landscapes; Enhance weather and climate forecast skill;	Freeze/Thaw State: Capture state transitions in integrated vegetation -soil continuum with two-day precision, at the spatial scale of landscape variability (3 km)	<u>Freeze Thaw State:</u> L band reflectivity Resolution - 3 km Incidence Angle < 60 deg	To use Signals of Opportunity to map deeper soil moisture data
Develop improved flood prediction and drought monitoring capability.	Observation over a minimum of three annual cycles		Orbit: < 750 km, circular 70-85° inclination
	Sample diurnal cycle at consistent time of day. Global, 4 day revisit; Boreal, 2 day	Minimum three year mission life	Baseline three years in orbit gathering data Observe between 83.4° and -60° Lat. 95% coverage needed for global coverage

2. Top Level Requirements

The Top Level Requirements (TLR) can be categorized into three sections: Functional requirements, Operational requirements and constraints. TLR are derived from the STM and requirements listed in the NASA Announcement of Opportunity. They serve as the driving factors of development for all the subsystems. We made back of the envelope calculations to get a very rough approximation of altitude, inclination and number of satellites to meet the mission requirements.The upper limit on altitude was established based on the altitude of a P-band transmitter satellite constellation which is essential to be in the FOV of the receiver satellites to meet the coverage requirements. The inclination range was based on the determined observable land region. These numbers served as the initial case for coverage simulation.

Functional Requirements:

Frequency	The spacecrafts shall obtain measurements in I-band and P-band within 12 hours of each other for root zone moisture
Orbit	The spacecrafts shall be placed in a circular orbit at an altitude below 750 km
	The spacecrafts shall be placed between 70° and 85° inclination
	12 satellites with I-band, P-band and L-band receivers shall be used
Instrument	15 kbits/sec - per each spectral point
Observation	Observe land regions between 83.4° and -60° Lat.

Operational Requirement:

Lifetime	The mission shall operate for minimum 3 years

Constraints:

Budget	The mission will be built and flown for at most $190 Million
Development Time	The mission should not take longer than 5 years to achieve operational capability
End of Mission	The spacecraft should deorbit after 25 years of end-of-mission.

E. Science Implementation

1. Specific Instrument

GAEA will use Signals of Opportunity technology to make the measurements and it will use the same receiver and antenna as SNOOPI. Thus, the instrument does not need a development plan as it was developed for SNOOPI, and is TRL 9 rated. The heritage reduces the risk of the entire mission. The specifications of the receiver and antenna for P/I-band are:

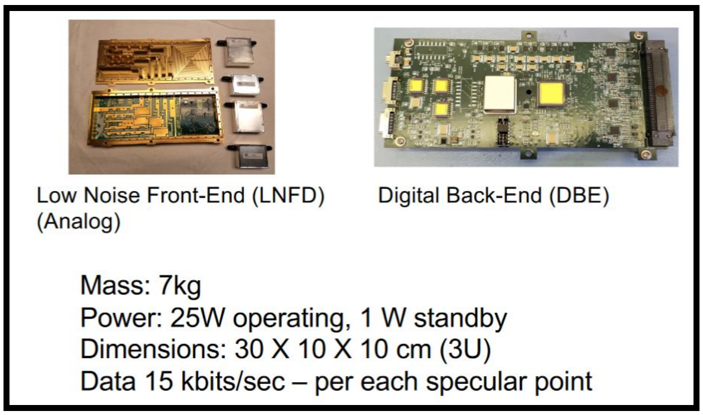

Fig E.1: Receiver Specifications

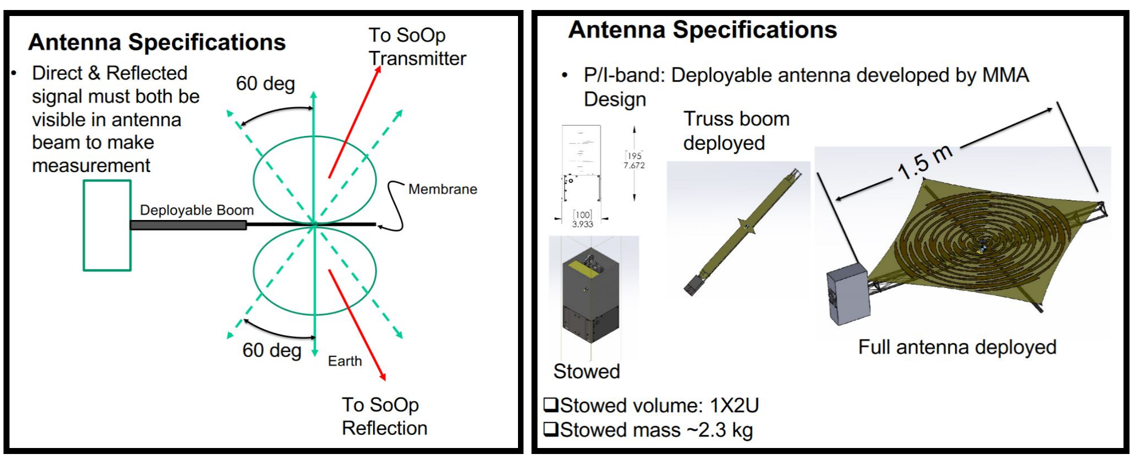

Fig E.2: Antenna Specifications

2. Science Mission Profile and Operations

This mission has several science requirements related to gathering of soil moisture data using Signals of Opportunity. The overall objective is to understand processes that link the terrestrial water, energy, and carbon cycles and to estimate global water and energy fluxes at the surface of Earth. To do this, GAEA will gather surface soil moisture data using L-band radio frequencies upto 5 cm depth and root zone soil moisture data using I-band and P-band satellites upto 100 cm depth. For L-band transmission, GAEA is using GNSS satellites: Beidou, Galileo, GLONASS, and GPS. For I-band transmission, ORBCOMM, NOAA, and METEOR are being used. For P-band transmission, MUOS is satisfactory. The transmitting satellites will all be in a Low-Earth Orbit at an altitude of 500 km in circular orbits. This will be discussed further in **F.1 Mission Design**. Water content less than 4% volumetric accuracy will be gathered. To further accomplish this requirement, hydrometeorology is performed in grids of size 10 km cells for L band and 40 km grid cells for P and VHF bands. GAEA will also gather data to help develop improved flood prediction and drought monitoring capability. To accomplish this requirement, GAEA will continue coverage for a minimum of three annual cycles while achieving a revisit time of 4 days globally, and 2 days within the Boreal region. Finally, GAEA will quantify net carbon flux in the Boreal region and enhance weather and climate forecasting ability. 3 km grid cells were used to capture the freeze/thaw state transitions with L band. Data downlink is performed every other day for data transmission to ground stations. Overall, GAEA is using components of Signals of Opportunity to gather soil moisture data to further contribute to climate, energy, and precipitation research.

3. Data Sufficiency and Data Plans

Data plans including ground station architecture and plans for data downlink will be discussed in the **communications** section. Onboard data storage is sized in order to meet a data downlink of every other day with a contingency of at least being able to store 6 days worth of data onboard. This worst case accounts to roughly 15 GB of mission data stored onboard. Onboard data storage is provided by the onboard computer and communications transceivers each having more than 32 GB of non-volatile onboard storage. See fig I.23 in the appendix for a breakdown of onboard storage and section F.1.3.f for the main computer selection.

F. Mission Implementation

1. Mission Concept Definitions

The GAEA mission will use Signals of Opportunity to make soil moisture measurements using PI band upto 5 cm depth and L band upto 100 cm depth. GAEA consists of 12 satellites, each equipped with L, P and VHF band antennas. 8 satellites will be placed at 70 deg inclination and 4 will be at 85 deg inclination, all at 500 km altitude. The satellites will be launched through Minatour 1 from Kodiak Launch Complex. The satellites will have X and S band transmitters and receivers to uplink and downlink data. They will contain the necessary ADCS components, thermal control and space radiation hardening, so that the mission can last for at least 3 years.

1.1. Mission Design

Since the science requirements placed an emphasis on Boreal regions, thorough coverage of extreme northern regions was crucial. For even coverage of the earth, a circular orbit was the best choice. Additionally, signals of opportunity can only be measured if the transmitter is *above* the receiver, therefore a low-earth orbit constellation was chosen. Using simulation tools, it was determined that the ideal inclination for good global coverage was in the 70-85° range.

Our orbit configuration consists of 12 GAEA receiver satellites at 500km circular orbits. 8 of the satellites are placed at 70° inclination, evenly spaced around the earth by true anomaly. The remaining 4 are at 85° inclination, also evenly spaced by true anomaly (0,90,180,270°). GAEA will be making measurements in 3 primary bands: I-band (100-200 MHz), P-band (250-500 MHz), and L-band (1-2 GHz)[1]. For I-band, the SoOps being observed include the ORBCOMM constellation, as well as METEOR and NOAA satellites. For P-band, MUOS is the primary signal of interest. For L-band, GNSS satellite constellations (incl. Beidou, Galileo, GLONASS, and GPS) give many possible signals, making this the most plentiful band to measure with. Each of our GAEA satellites can make measurements in all 3 bands to maximise the number of specular points.

Our science traceability matrix outlines 3 main coverage requirements. First, L-Band measurements of freeze/thaw state at 3km resolution, with a two day or less revisit period. This requirement is only necessary in Boreal regions. Second, global L-band coverage for surface soil moisture at 10km resolution with a 3-4 day revisit period. Finally, root zone soil moisture must be measured globally by both P and I band sources, within 12 hours, over a 40km resolution. To determine how well we satisfied these requirements with our architecture, 3 grids superimposed over the earth's surface were used. Three grids were strategically located to simulate the entire

[1] *Frequency Letter Bands*, www.microwaves101.com/encyclopedias/frequency-letter-bands.

Earth landmass to decrease the simulation computation. Each grid was 100 by 100 cells, at different resolutions and locations. Grid 1 was a 3km resolution grid placed in the Boreal region of Canada. Grid 2 was a 10km resolution grid placed in the Amazon rainforest. For both of these grids, L-band coverage was the most important thing to examine. P and I band measurements were quantified in Grid 3, a 40km resolution grid in Asia. **A**dditionally, mean revisit time was calculated for the cells that were visited at least twice.

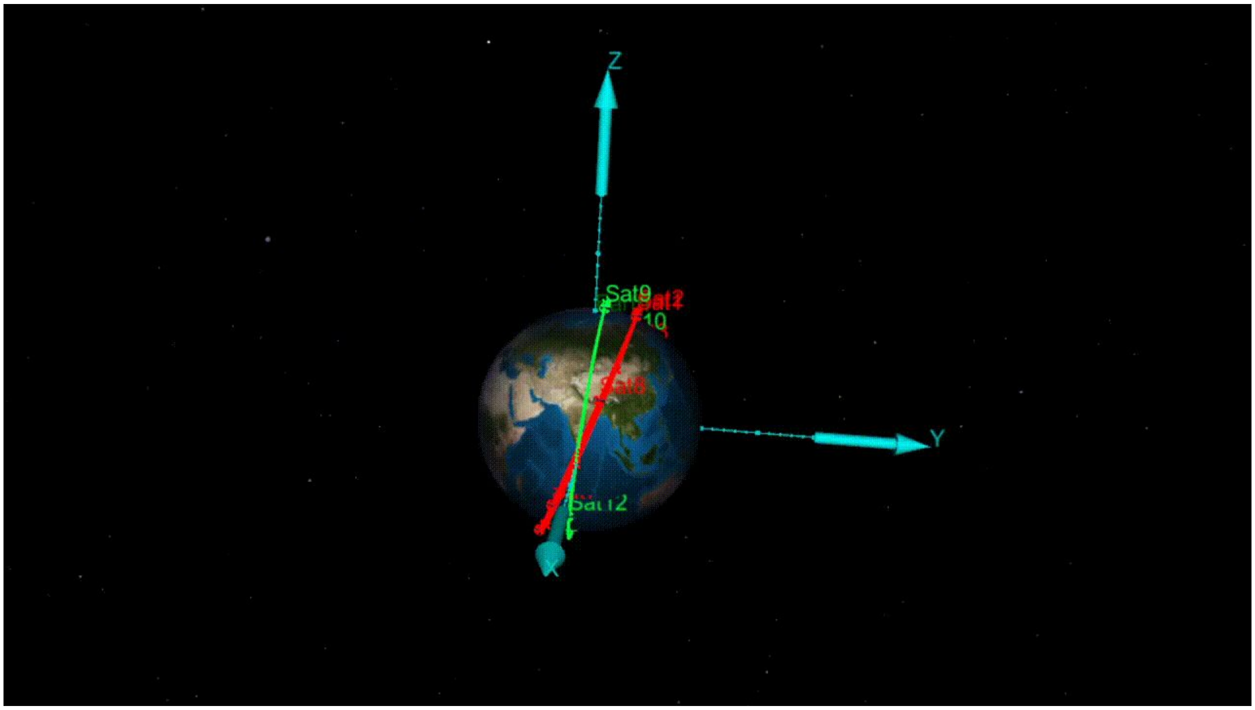

Figure F.1: The four satellites at 85 degree inclination can be seen in green, while the eight satellites at 70 degree inclination can be seen in red

Orbit Decay

The total functional lifetime of the mission is 3 years. Within these 3 years, atmospheric drag will slowly deteriorate the orbit over time. A study was performed to analyze the impacts of orbit decay on our mission. The orbit of a single GAEA satellite was inputted into NASA's General Mission Analysis Tool (GMAT) to gather data regarding atmospheric drag. GMAT uses the MSISE90 drag model to calculate the altitude of a spacecraft over time. In **Figure F.2** the relationship between orbit altitude and time can be observed.

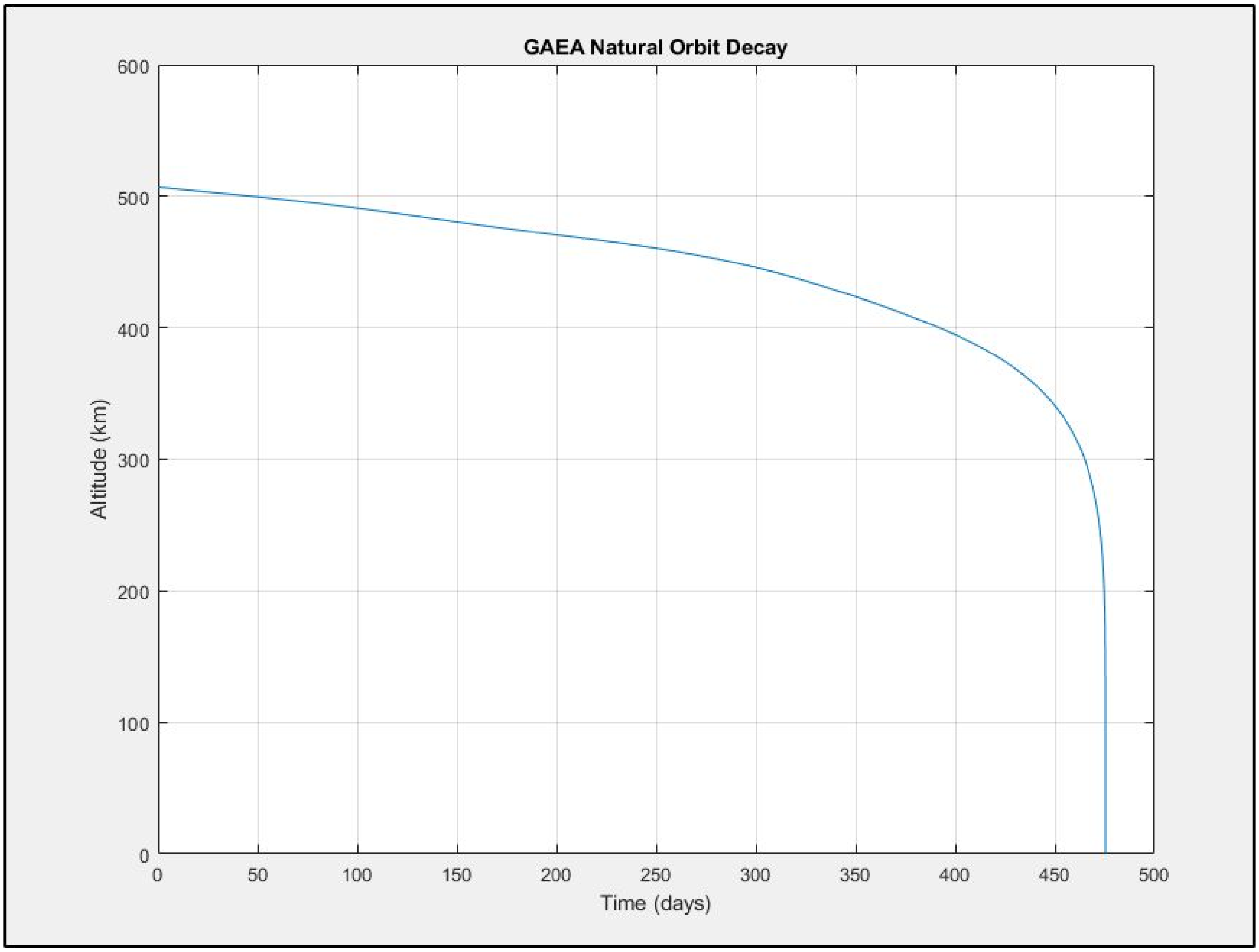

Figure F.2: Plot of the natural decay of the GAEA satellites due to atmospheric decay

It can be observed from **Figure F.2** that the orbit decay is not negligible, since the satellite will crash within 475 days if no additional propulsion is supplied. To remedy this issue, a plan to provide periodic delta-V burns will be implemented. Every 127 days a burn of 32.4m/s will be employed to maintain the satellite at an altitude of 500km for most of the year. An issue with this strategy, however, is that by the end of the 127 day period after any given burn the satellite will have dropped 25km in altitude. Our subsystem teams have acknowledged this periodic slight drop in altitude. After simulating the coverage at 475 km and 500 km, we find a negligible difference in coverage, thus the 25 km drop is acceptable. This delta-V orbit maintenance strategy will be further discussed in **Propulsion**. While this lower altitude technically benefits certain subsystems on the satellite (ORBCOMM in particular), the satellite must be corrected every 127 days to guarantee no severe decrease in altitude nor a crash. For this reason, the GAEA satellites must introduce a burn to maintain their orbit status. Additionally, the velocity change due to the decreased altitude could potentially pose an issue for other subsystems like ADCS. For the discussed reasons, orbit decay is not negligible and must be accounted for via onboard propulsion systems on each satellite. A beneficial effect of the impact of orbit decay is mission end, however. This phenomenon will be used at an advantage to perform a reentry burnup for the end of mission. This will further be discussed in **End of Mission.**

Coverage

To demonstrate that we have global coverage with all of our bands, we can look at the specular points the GAEA constellation sees across the entire globe over a 1-day period for each band. We will see that L-Band is the most densely populated, followed by P-Band then I-Band.

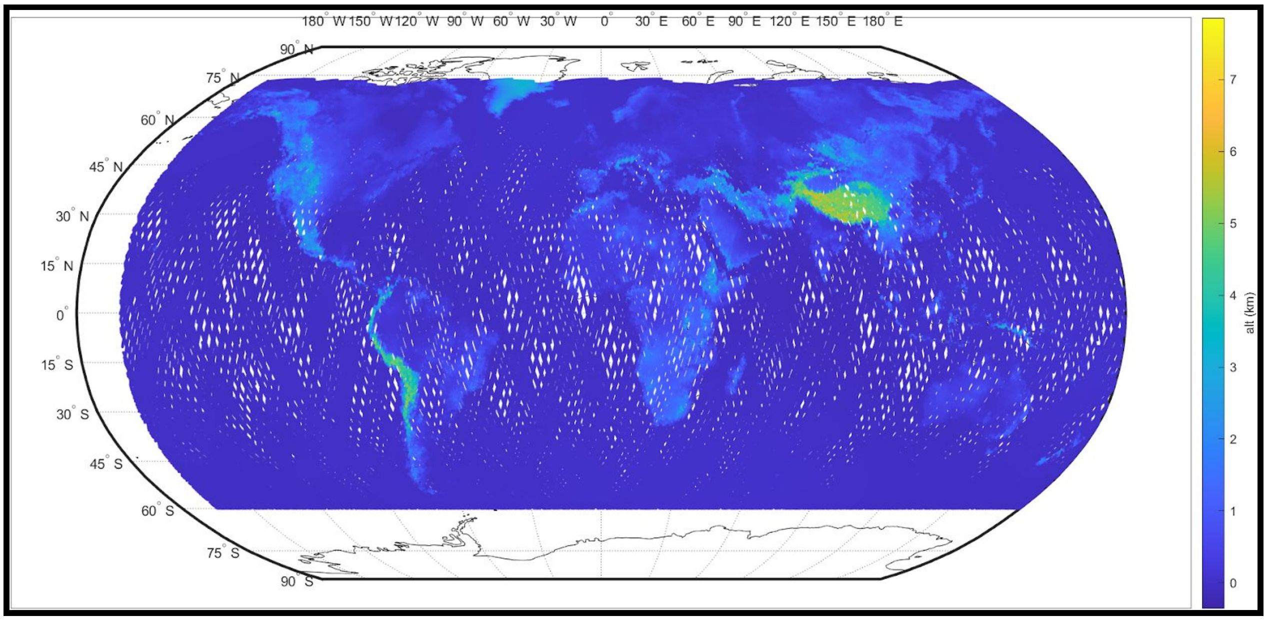

Figure F.4: L-Band coverage is the most dense (due to having the most Tx Satellites) and is clearly global

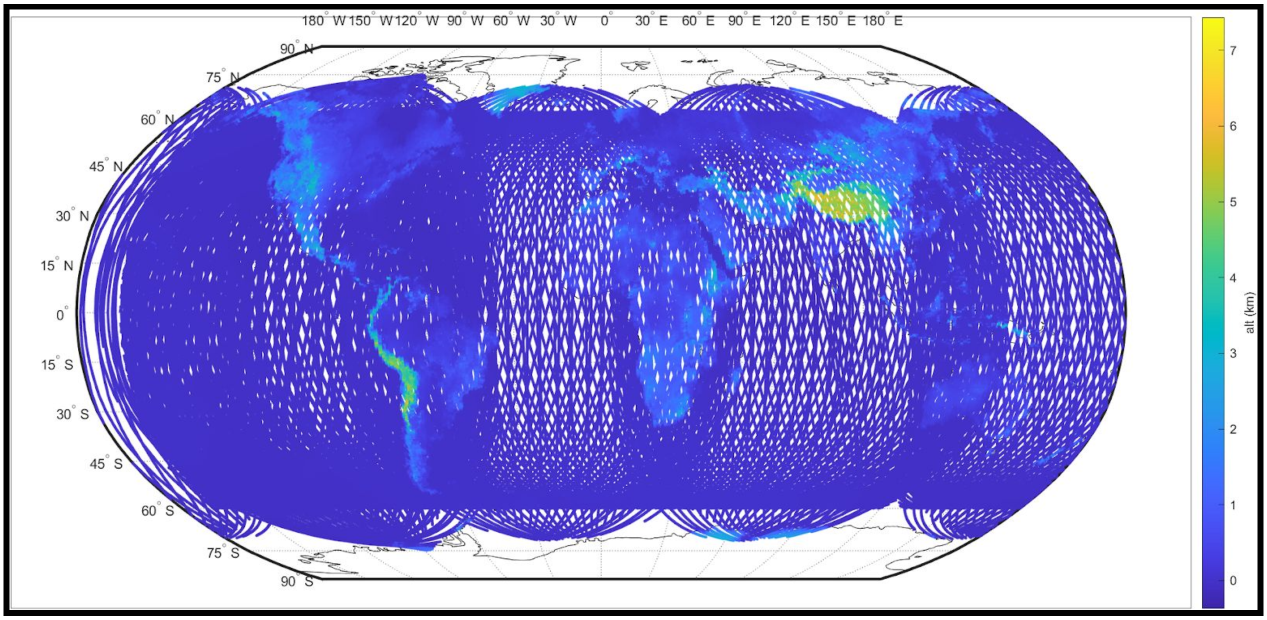

Figure F.5: P-Band Coverage, while only looking at MUOS (5 Satellites), is quite good. However, we notice a region of northern Russia has been unfortunately cut off. Over a period of several days this issue would be fixed and we would see that region. Boreal region coverage is especially dense

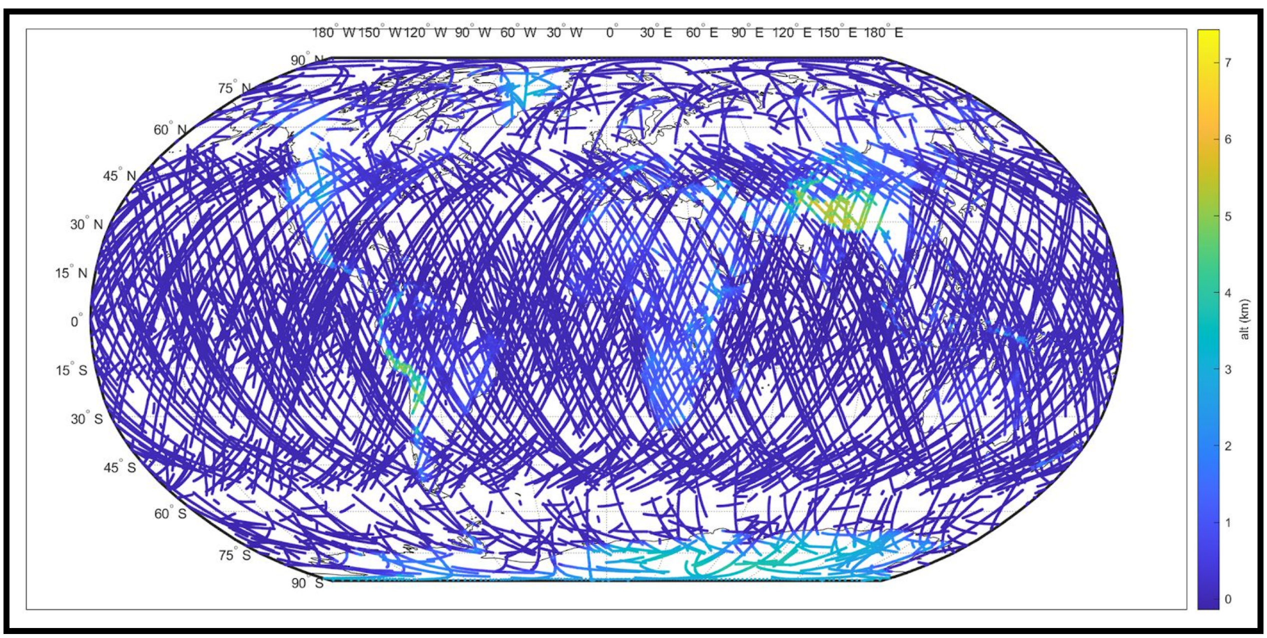

Figure F.6: Our I-Band coverage is the most sparse, but is still global. Luckily, the resolution requirement for this band is the most coarse (40km)

To examine our coverage quantitatively, an analysis was done where 3 grids were superimposed on the earth's surface. These grids are each 100 by 100 cells and represent our 3 resolutions required: 3km, 10km, and 40km. Grid 1 is the 3km grid and was placed in the Boreal region of Canada. For this grid L-Band coverage will be most important. Grid 2 is a 10km resolution grid and was placed in South America. This grid is also relevant to L-Band coverage. Grid 3 is a 40km resolution grid and was placed in India/China. This grid will inform the coverage of our P/I Band measurements, where the resolution requirement was 40km.

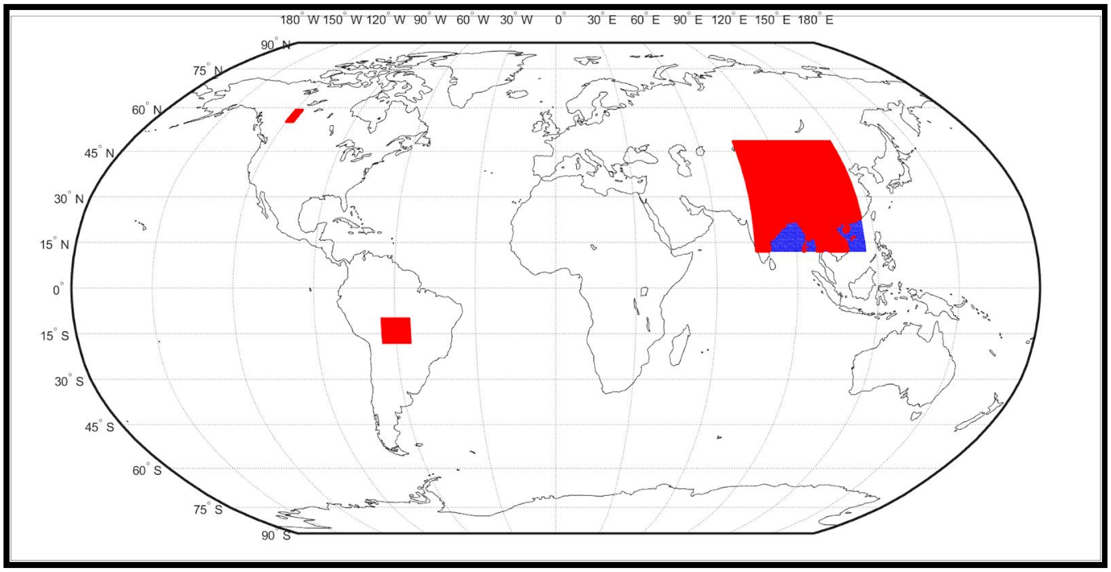

Figure F.3: Global map of grid locations starting from left to right: Grid 1, Grid2, Grid 3

L-Band Coverage

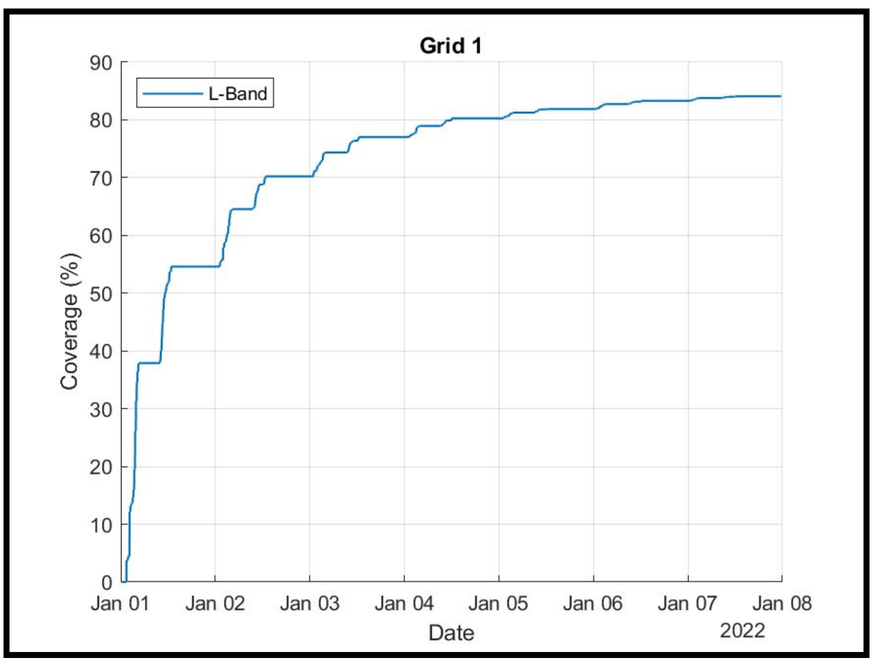

Figure F.7: 70% Boreal Coverage for 2 day revisit

Grid 1 represents our finest resolution (3km) which is desired in our most crucial area, the Boreal region. We see that within 2 days we reach 70% coverage. This is lower than the desired 95-97% to claim a 2-day revisit. However, this is not due to any gaps from the ground-track of the satellites. As can be seen in the following figure, every measurable region of the 300 by 300km grid had specular points.

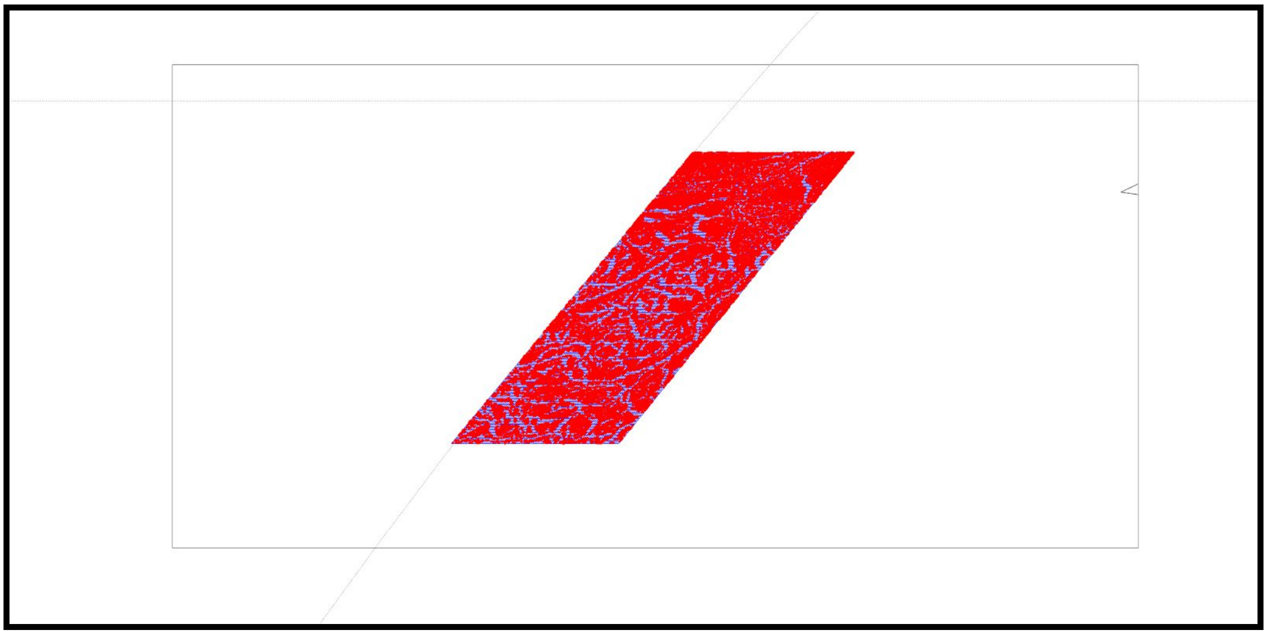

Figure F.8: Red Specular points in Grid 1 (1 week) with Blue Gridlines. Gaps are from rivers/icy areas with no soil to measure.

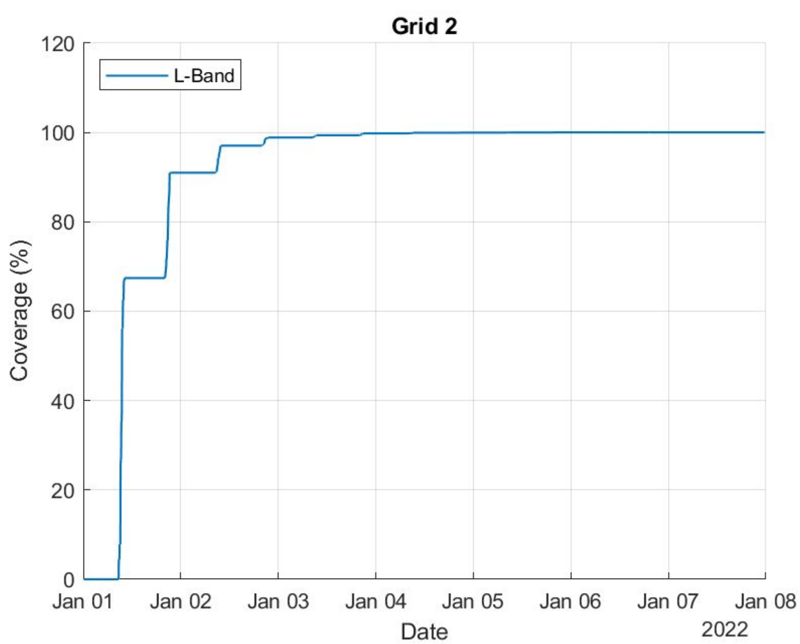

Figure F.9: Grid 2 Lband - 97% Global coverage with 4 day revisit

Grid 2 represents our 10km resolution, which is required globally. As can be seen, we meet our 3-4 day revisit time and have complete coverage, visiting 97% of all grid cells within 4 days.

P/I Band Coverage

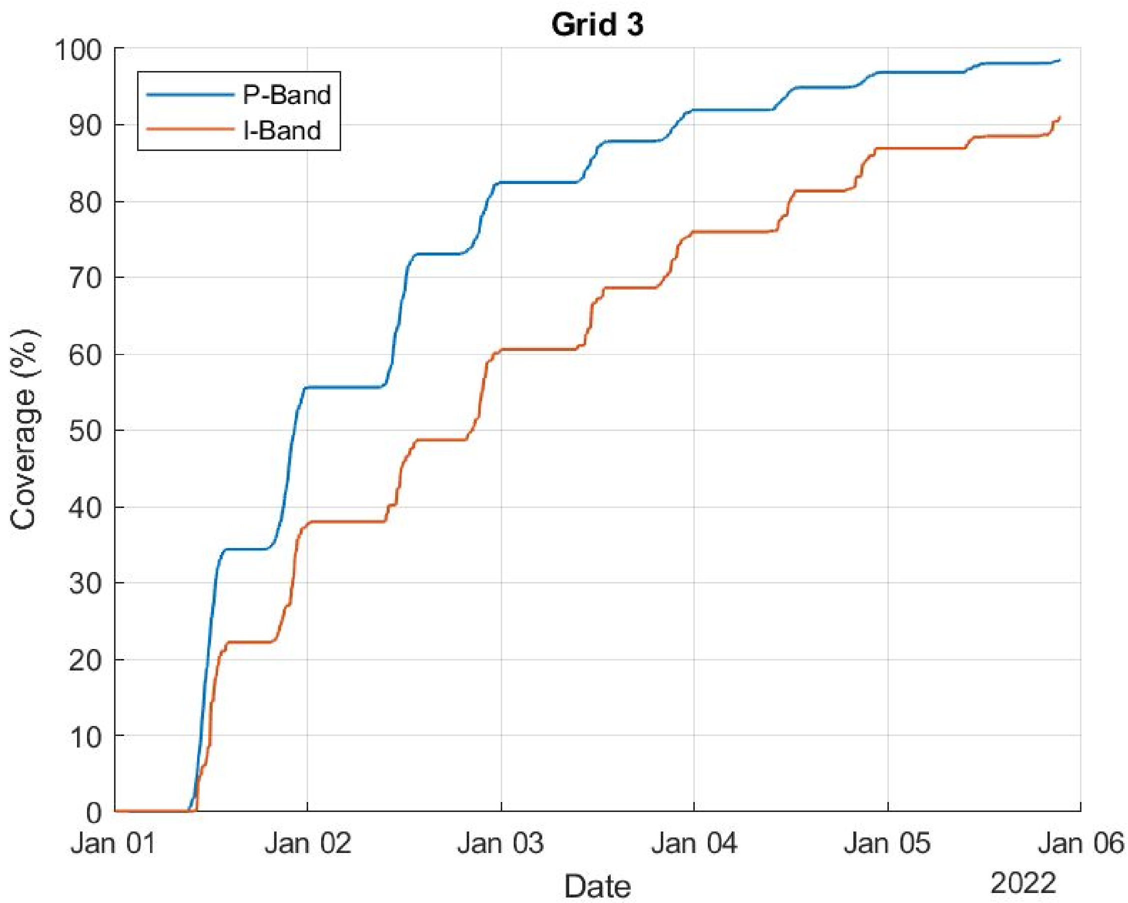

Figure F.10: P/I Band Coverage in Grid 3 (40km resolution)

P-Band Mean Revisit Time: 26hr 45min
P-Band 75% Coverage Time: 1.8 days
I-Band Mean Revisit Time: 28hr 09min
P-Band 75% Coverage Time: 3.5 days

Mean revisit times were calculated by taking time between specular points for all cells that had been measured at least twice. These revisit times are well within our requirements of 3-4 days. While individually the coverage for these bands look good, we must remember that measurements of the same grid cell need to be taken within 12 hours of each other. The following figure shows which cells were measured by P and I band within 12 hours of each other at least once during a 4-day simulation:

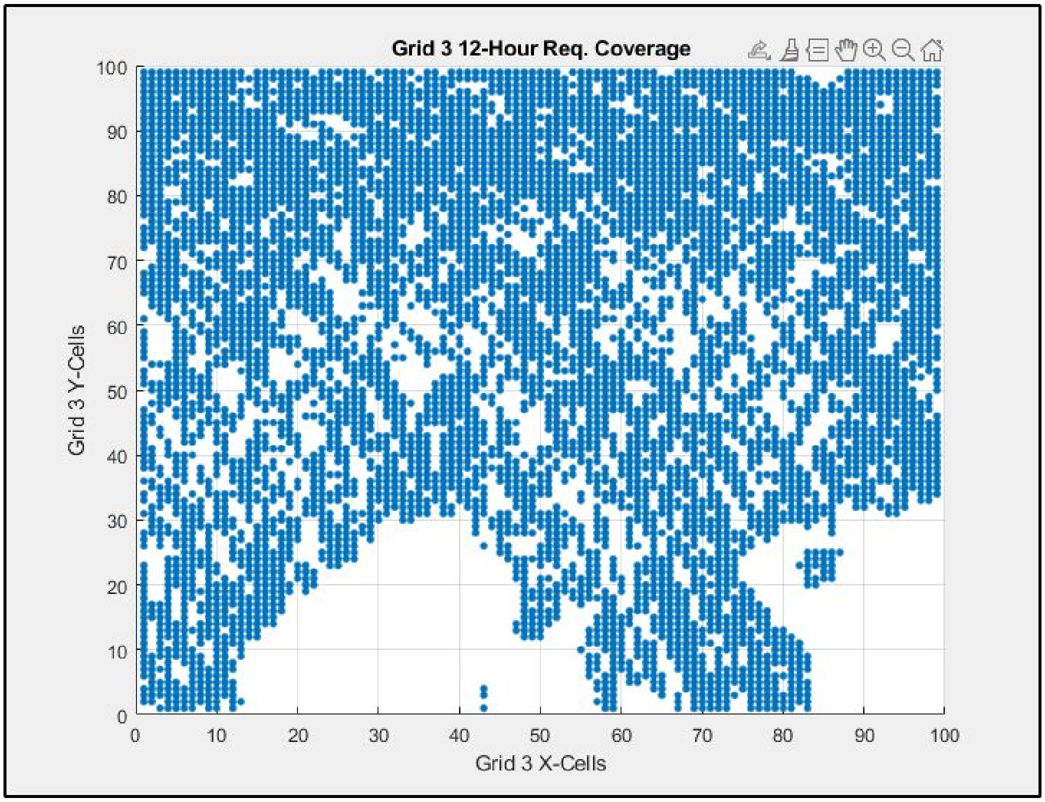

Figure F.11: Grid 3 I/P-Band root zone coverage in 100x100 discretized cells coverage: 76.2%

We get 76.2% coverage with 4 day revisit time from Grid 3. We can assume that globally we would see similar coverage performance, as Grid 3 was placed in an area of especially *sparse* coverage for both P-Band and I-Band. Therefore what we are seeing here reflects worse-case scenario coverage. To conclude, our coverage requirement is met for the 10km resolution, but not entirely met for the 40km (specifically with I-Band) and 3km resolution. While we will reach 100% coverage on all grids eventually, it does not occur quickly enough for us to be able to claim the desired revisit time. To fully meet the requirements, 6 more satellites would have to be added to the GAEA constellation (for a total of 18). Due to the budget constraint we will not be able to meet the coverage requirement. However, we have assessed that based on the incremental increase in coverage per satellite, we will need about 6 more satellites at 70 deg inclination. Thus, if there was no budget constraint then we would have 18 satellites. The 6 additional satellites require $40 M, so the total budget would then be $ 220 M, $30 M over budget with a 30% cost margin.

1.2 Launch Services and Launch Vehicle Capability

For this mission, the launch vehicle Minotaur I has been selected as the optimal candidate. The first step taken to determine the ideal launch vehicle options given by NASA's NPREA was mass requirements and cost analysis. After determining a rough estimate for the propulsion mass it was found that if the total satellite weight with magin stays below forty-eight-point-three kilograms then the weight requirements are met for the cheaper launch vehicle the Minotaur I, see the figure below. This works well because it not only saves money that can later be put towards launch insurance or unforseen costs, it also matches our initial mass estimates.

LSG Options	Max Weight	Obit	Inclination	Cost per Launch	$ per kg	Cost for Mission
Pegasus	443kg	LEO	N/A	$40M	$90,293.45	$52,366,591
Minotaur I	580kg	LEO	28.5 deg	$28.8M	$49,655.17	$28,798,014
Minotaur-C	1458kg	LEO	N/A	$40-50M	$27,434.84	$15,911,111
Minotaur-IV	1735kg	LEO	N/A	$50M	$28,818.44	$16,713,545

Figure F.12: Launch Vehicle Cost Estimates

The Minotaur I launch vehicle meets our additional mission requirements because it is capable of reaching the desired 500 km circular orbit, as well as large enough to carry all 12 of our satellites with flexible payload separation methods. Another key design characteristic that makes the Minatour I an ideal launch vehicle is that it can deliver multiple payloads to different orbits allowing for a simpler satellite design while still being able to achieve the two desired inclinations 70 and 85 degrees. The Minatour I also comes with a thermally controlled fairing volume using Class 100K standard cleanliness with a standard fifty inch fairing, and an optional sixty-one inch fairing. The insertion accuracy is typically 33.4 nautical miles plus or minus a tenth of a degree inclination, both metrics are within three sigma. The Minataur I also has an

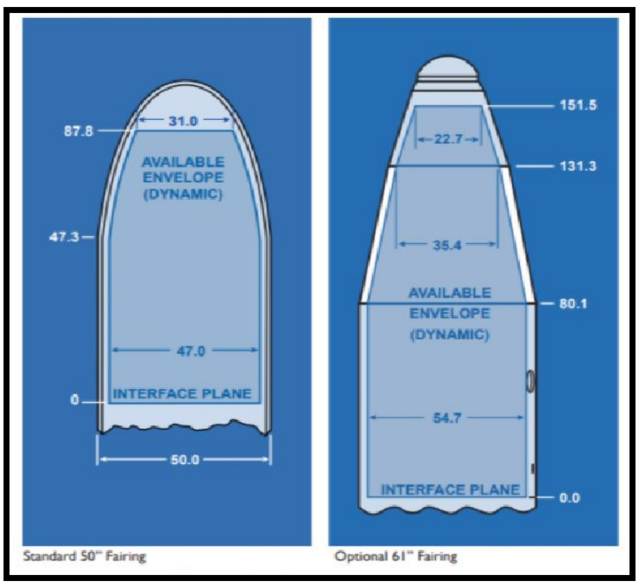

Figure F.13: Launch Vehicle Blueprint

optional soft ride for small satellites for mitigating flight dynamics environments[2]. Due to the nature of the orbital inclination for this project the launch site KLC has been chosen, and from the NASA Routine Payload Checklist launch pad LP-1 is the default for this launch vehicle.

1.3. Flight System Capabilities

a. Mass/Structures and Power

These subsections combine to solve the issue of sizing for our satellite. Our system will weigh 35.04kg, require 54 Watts of Power and will be using Aluminum 6061 for its structural components. Each system relates to the other as increased power increases mass and thus a larger structure will be needed.

The GAEA satellite payload weighs roughly 9.3 kilograms in total. Using historical percentage estimations as the primary method to size the satellite, this puts the total mass at 35.04kg. This is consistent with both mass and power. A 30% margin is utilized for this system, which is consistent with GOLD rules for Pre-Phase A development. The payload uses 25 watts while the system is running and 1 watt while in standby mode. To calculate the total mass of the satellite a percentage table was used, with each component representing a percentage of the total mass which was a function of payload. As we can see in the figure, the main payload makes up most of the onboard bus mass. The structure of the system also uses a good percentage of the onboard mass.

Subsystem (Kg Dry Mass)	LEO w. Prop	LEO w. Prop	LEO w. Prop w/ Margin (30%)
Payload	31.00%	9.30	12.37
Structure and Mechanisms	27.00%	8.10	10.77
Thermal Control	2.00%	0.60	0.80
Power	21.00%	4.60	6.12
Telemetry Tracking and Command	2.00%	0.60	0.80
On-board Processing	5.00%	1.50	2.00
Attitude Determination and Control	6.00%	1.80	2.39
Propulsion	3.00%	7.64	10.16
Other (balance and launch)	3.00%	0.90	1.20
Total (kg)	**100.00%**	**35.04**	**46.60**
Propellant		9.46	12.58

Figure F.14: Mass Budget

The power source for the satellite is provided by photoelectric energy. Gallium Arsenide Ultra Multi Junction cells were used due to their high efficiency and midrange cost per watt. The Solar Panels were calculated using the upper end of the power requirements to determine the area. The area is determined by calculating the beginning of life power generation at various

[2]Orbital Sciences Corporation. “Minotaur I Space Launch Vehicle Fact Sheet” *https://www.nasa.gov/pdf/164059main_Minotaur_I_Fact.pdf* , Orbital, 2006.

inclinations facing the sun. This is also coupled with variables such as eclipse time and degradation over time.

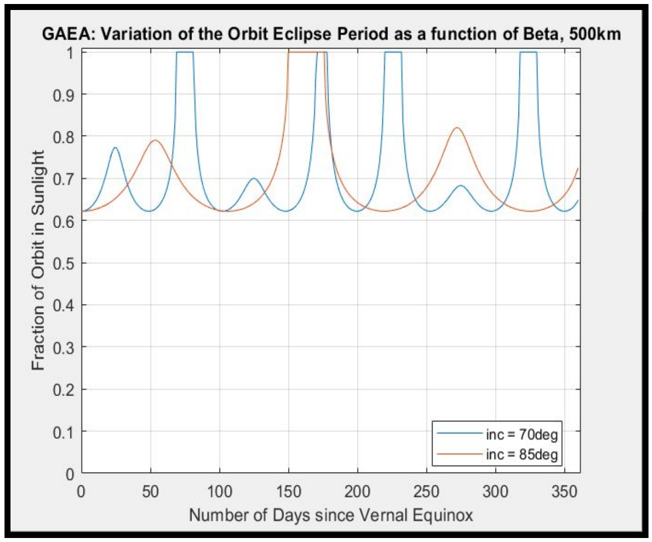

Figure F.15: Percentage of Orbit in Eclipse

Our mission will experience no more than 40% of its orbit time in an eclipse thus this confirms the use of solar energy with smaller batteries. Additionally, the mission is designed to last for three or more years, thus degradation of the panels due to various environmental stresses takes a strong precedence in calculation. This process generates a total solar array size of 0.63 square meters. These relationships are shown in the following equations.

$P_{sa} = (P_e * T_e/X_e + P_d * T_d/X_d)/T_d$ calculates the power output of the panels

$P_{eol} = P_{bol} * L_d$ calculates the end of life power generation

$A_{sa} = P_{sa}/P_{eol}$ uses the needed power and end of life generation to compute area

Subsystem Power (W)	LEO w. Prop	LEO w. Prop	LEO w. Prop w/margin	Margin
Payload	46.00%	25.00	33.25	30%
Structure and Mechanisms	1.00%	0.54	0.72	30%
Thermal Control	10.00%	5.44	7.23	30%
Power	9.00%	4.89	6.51	30%
Telemetry Tracking and Command	12.00%	6.52	8.67	30%
On-board Processing	12.00%	6.52	8.67	30%
Attitude Determination and Control	10.00%	5.44	7.23	30%
Total	**100.00%**	**54.35**	72.29	30%

Figure F.16: Power Budget

On board power storage was the next stage of the design process for this subsystem. Since eclipse time takes a small chunk of the total orbit time, the best type of battery for the

GAEA satellite is lithium ion due to its longevity. To calculate the size of the battery the needed capacitance was solved using this formula.

$$C = Pe * Te / ((DOD) * Nn)$$

Where C is capacitance, Pe is eclipse power, Te is time in eclipse, DOD is depth of discharge and Nn is efficiency times number of batteries. Our system will need two batteries that weigh approximately 0.55kg and they have dimensions of 5.6 x 6.5 x 13cm. Both of these numbers were calculated using the sizes of lithium ion cells and their energy density. The power each battery brings to the table is 68WH. This is good for the system as this can supply the power needed if one battery fails. The satellite spends no more than an hour in eclipse per orbit which shows 68WH to be sufficient for the 70V system.

These batteries have a long life in terms of rechargeability and have a high energy density, which works well for a small satellite system. The system is going to be supported with a Unregulated Direct Energy Transfer. This set up works best with minimal solar input changes which is how GAEA will operate.

The shell of the satellite will be constructed of Aluminum 6061. This was decided primarily off of the launch environment, with a secondary thought of the space environment. Since the Minotaur I is our launch vehicle, the axial forces can reach up to 4.3G and the lateral forces can reach 1.5G. The satellite must also have a fundamental frequency above 25Hz. The housing structure for the satellite is 0.22m long, 0.22m wide, and 0.38m high. To calculate loading beam bending stress formulas were applied to the walls and the top/bottom plates and compared to the loads the Minotaur in each direction. For vibrational loading, the following formula was used. It requires the Modulus of Elasticity of the structure material, the inertia, mass and length of the vehicle.

$$fn = 1/2\pi * \sqrt{3EI/ML^3}$$

Using the values of our system and the modulus of elasticity for Aluminum 6061 our vehicle has a fundamental frequency around 171Hz. Which is well above the threshold of 25Hz. Lastly the radiation environment played a factor in wall thickness. Radiation shielding starts showing diminishing returns around 5mm thickness. Thus this was picked for the wall thickness of the main body. This helps mitigate the risk from high consequence solar radiation such as gamma rays and charged particles.

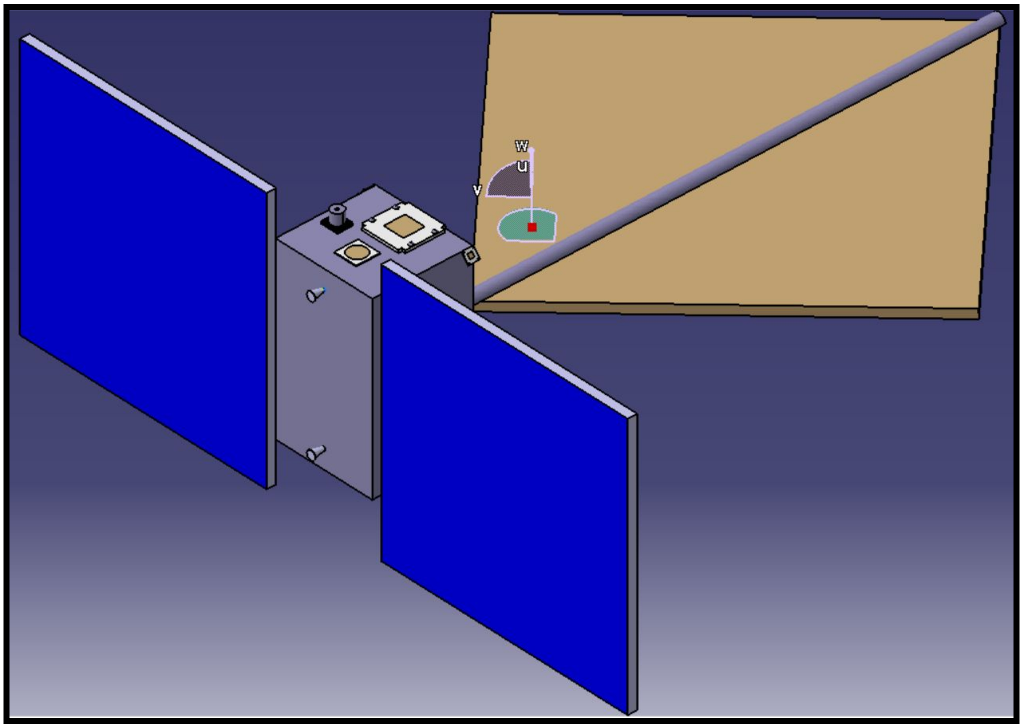

Figure F.17: GAEA Satellite Iso View

Based on the materials and the dimensions of the body and panels, the system will approximately take the above shape. This view is the bottom of the craft which shows Earth sensors and various bandwidth antennas. The solar panels shown are 0.57 meters in both width and height to provide the power generation needed.

The dimensions specified for the body are large enough to contain all the components needed to conduct the mission (See Appendix). All of the receivers, on board computer and propulsion systems fit in the body with additional room. Some of the objects such as batteries were modeled as rectangular prisms to assume each component was on the larger end and that components would all fit with this margin.

b. Propulsion

Each satellite has a deltaV budget of 398.08 m/s, mass of propellant of 5.42 kg, it will use Hydrazine as the propellant and have six MR-111G 4N rocket engines weighing 0.37 kg each.

The first step in the propulsion selection process is the creation of the delta-v budget. There are several factors that go into this, such as accounting for possible errors in orbit placement, stationkeeping, orbit control, atmospheric drag effects, and more. Using historical data as well as hohmann transfer calculations, a comprehensive budget was built for this project as seen in the completed delta-v table attached in the appendix Table I.21.[3]. From the delta-v budget the next step was to list out all of the potential propellant system options and see which system best fits this mission's design requirements.

Some propellant options were ruled out early on, for example electric propulsion systems were too large, too expensive, and provided only low thrust maneuvering. Solid propulsion systems were considered, but due to the large scale and single use nature of them they were deemed a poor fit for this mission. Liquid propellant is ideal because it can meet the orbit control and attitude control requirements while also being capable of orbit insertion. Of the considered liquid propellants hydrazine was deemed the strongest option due to the large variety of small engine options. See Table I.22 for a complete list of liquid propellant options and their corresponding weights. Hydrazine is the only liquid propellant that is within the mass budget for the Minotaur I; if other propellants were decided to be more desirable the mass increase would require a step up to a more expensive launch vehicle. The mass for most of the propulsion systems comes from Space Mission Analysis and Design, chapter 17 page 714[4]. With further research of possible engines for the various types of propellants, it was found that hydrazine offers an array of smaller liquid engines off the shelf. For this mission, it was decided that the strongest candidate was the MR-111G 4N rocket engine assembly by Aerojet Rocketdyne[5]. This engine allows for more flexibility with positioning and quantity due to its small mass, 0.37 kg, and compact design.

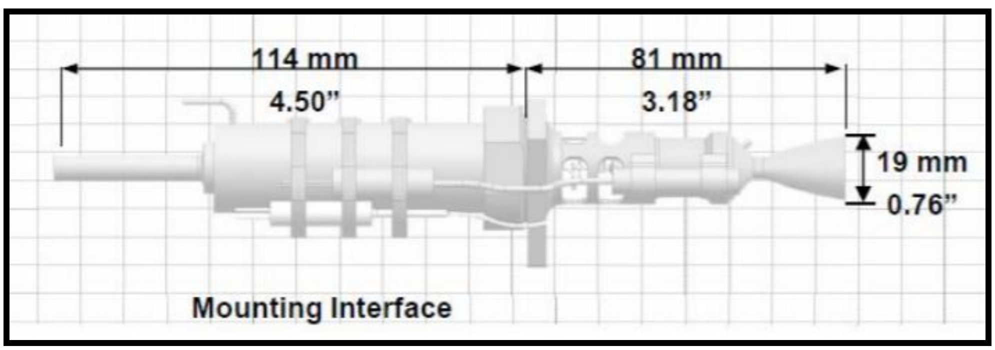

Figure F.18: MR-111G 4N Rocket Engine Assembly

For the scope of this project we can make some rough estimates to get a general idea of the tank sizing for this system. Using the calculated propellant mass and the estimated size of the propulsion system, assuming equal volume oxidizer and propellant tanks, a rough estimate for the sizing of the propellant tanks can be made. This estimate's purpose is to get a general idea of the feasibility of this design, for instance if it was determined the tanks dwarfed the satellite

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3 Chung, Winchell. "MISSION DELTA-V AND FLIGHT TIMES." *Mission Table - Atomic Rockets*, Atomic Rockets, www.projectrho.com/public_html/rocket/appmissiontable.php.

4 Larson, Wiley J., and James R. Wertz. *Space Mission Analysis and Design*. Kluwer Academic Publishers, 1992.

5 Wilson, Fred. "In-Space Propulsion Data Sheets." *Rocket*, Aerojet Rocketdyne, 8 Apr. 2020, www.rocket.com/sites/default/files/documents/In-Space Data Sheets 4.8.20.pdf.

then different propellant options might be pursued. Further analysis is needed to cement these dimensions, such as heat analysis, pressure changes, and in-depth integration testing.

c. ADCS

The attitude determination and control system involves both sensors to determine GAEA's attitude, as well as actuators for control. The control modes required for the GAEA mission include general attitude correction, and maintaining orientation relative to the earth's surface after deployment and during the mission. Satellites need to yaw 180° twice in an orbit to reorient the solar panels within a small fraction of the orbit period. Using the CAD model of the satellite to determine moments of inertia, an analysis can be run to determine the strengths of these external disturbances. Following this we can choose our specific sensor and actuators, then simulate the ADCS to ensure that it will work.

External Disturbances

Solar Radiation: $T_s = \phi / cA_s * (1 + q) * (cp_s - cm) * cos\phi \ = 8.6 * 10^{-6}\ N * m$

Atmospheric Drag: $T_a = 1/2 * \rho * C_d * A_r * V^2 * (cp_a - cm) \ = 4.8 \ * 10^{-7}\ N * m$

Magnetic Field: $T_m = D * (M/R^3\lambda) \ = 5.5 \ * 10^{-5}\ N * m$

Gravity Gradient: $T_g = 3\mu \,/\, 2R^3 * |I_z - I_y * |sin(\theta) \ = 2.0 \ * \ 10^{-6}\ N * m$

Considering these torques and the size of our spacecraft, our chosen reaction wheel is the NanoAvionics 4RW0. The 4th wheel provides redundancy, and 3-axis control will be maintained even if one of the wheels fails. The mass of the system is 665g, and consumes less than 1W of power at steady state, and no more than 12W peak. These numbers are within our weight and power budgets. The maximum torque the wheels can provide is 5.9 mNm, far greater than any of the external disturbances. Maximum momentum storage is 37 mNms. Six thrusters are smartly placed around the satellite for momentum dumping and orbit control.

For orientation sensors, an infrared Earth sensor, magnetometer, and rate gyroscope will be used. The gyroscope provides redundancy in case the horizon sensor or the magnetometer fails. For tracking satellite position and velocity a GPS receiver will be placed on the spacecraft. Redundancy for this will be provided by an accelerometer. See the Master Equipment list at the end of this proposal for more details.

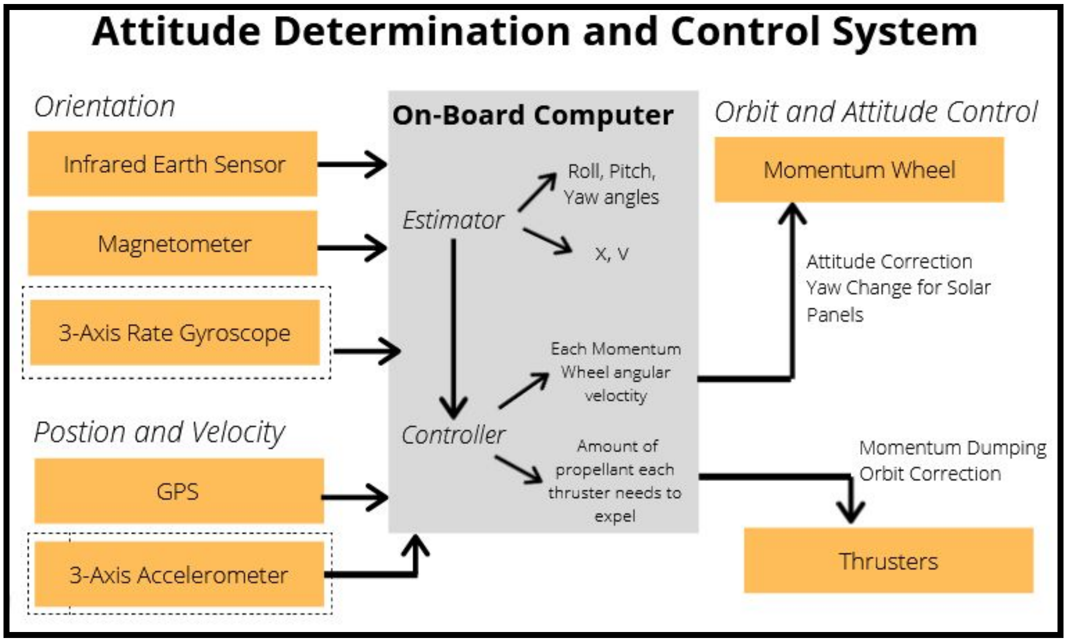


Figure F.19: System operation overview

Simulation

This first simulation represents a tumble recovery. An initial spin of 15°/s was given to all 3 axes. The spacecraft recovered in around 3 hours.

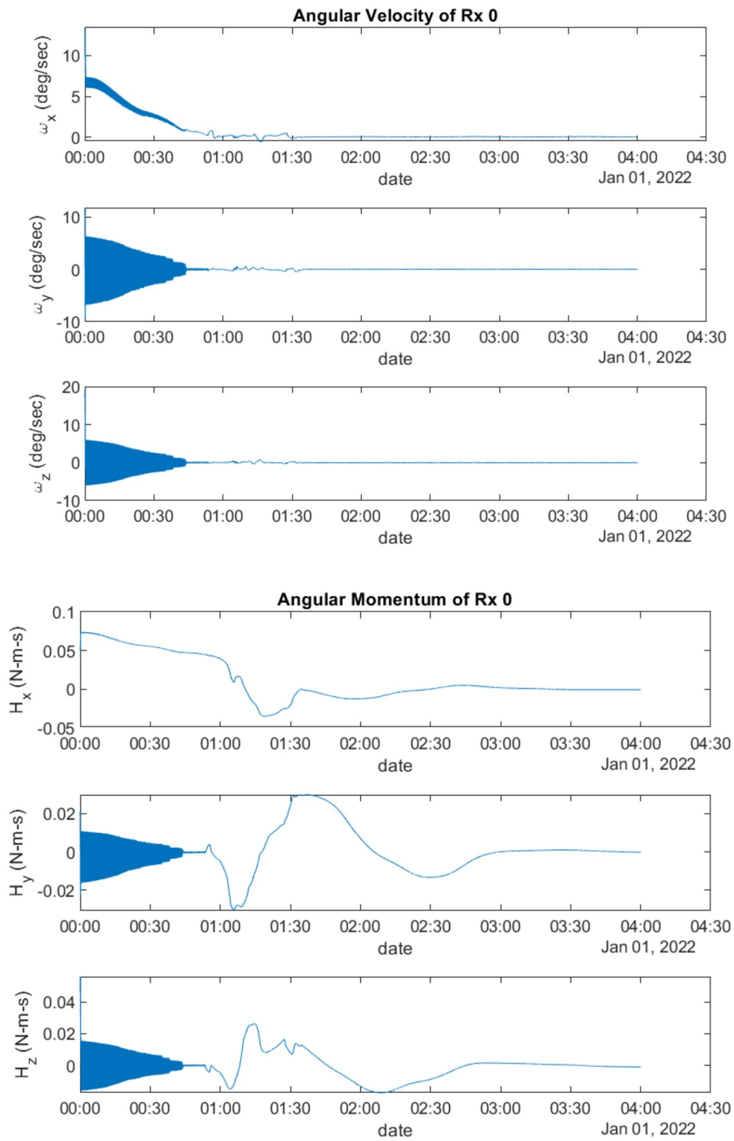


Figure F.20: Angular Velocity and Momentum for Tumble Recover ADCS Simulation

The second simulation was a 180° yaw maneuver. The spacecraft completed the maneuver in approximately 6.5 minutes, or less than 10% of an orbital period. Thus, the selected momentum wheel is sufficient to maintain the attitude of the spacecraft.

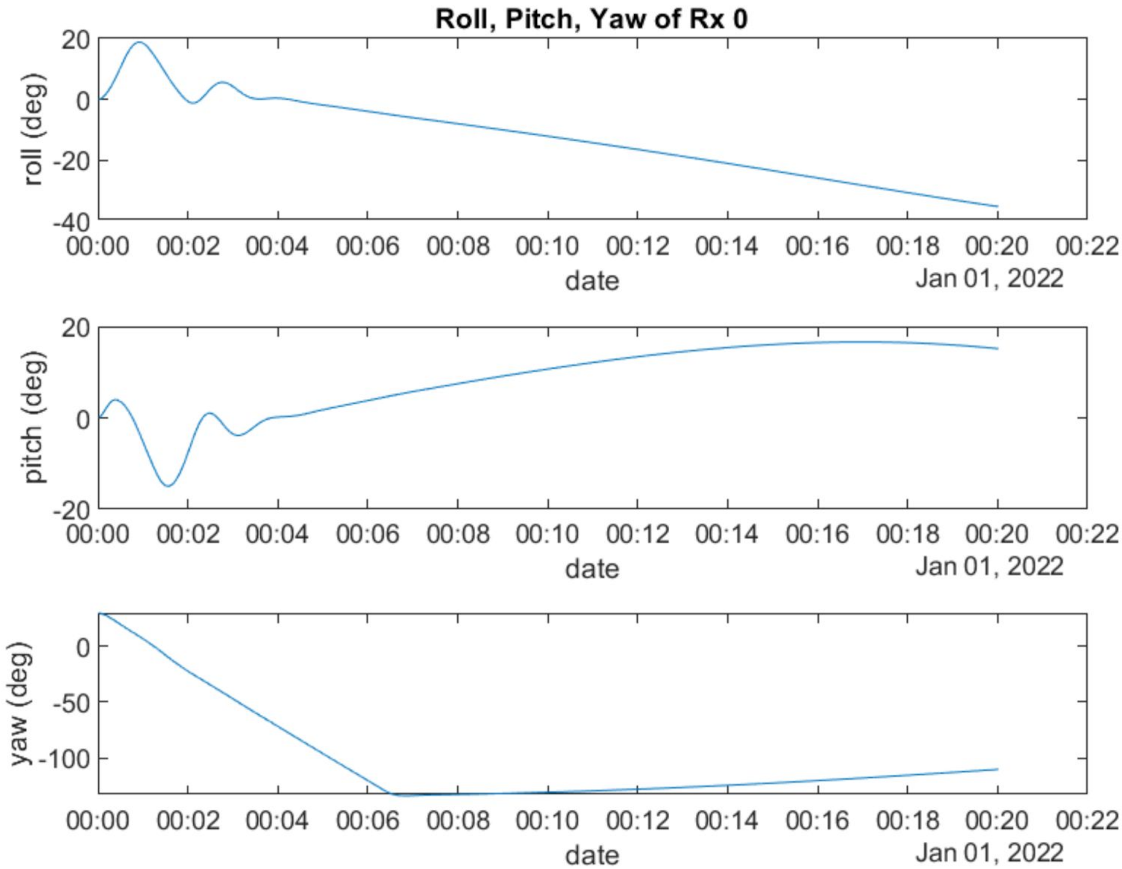


Figure F.21: Roll, Pitch, and Yaw for Yaw Maneuver ADCS Simulation

d. Thermal Control

For designing our thermal control system, we must first know our mission profile. Each satellite will be in circular Earth orbit, with an inclination of either 70° or 85°, with a 3-year life. The altitude is 500km, and our spacecraft can be approximated as an earth-oriented rectangular prism. From our power budget, our internal power dissipation is 70W peak-- for our analysis we will assume this power is being dissipated constantly. Our temperature requirements are most strictly dictated by batteries, which must be kept at 20°±10°C. Due to the high inclination of our orbit, our beta angle will range from 0° to 90°, which translates to 62% to 100% of time in sunlight during our 95 minute orbit period.

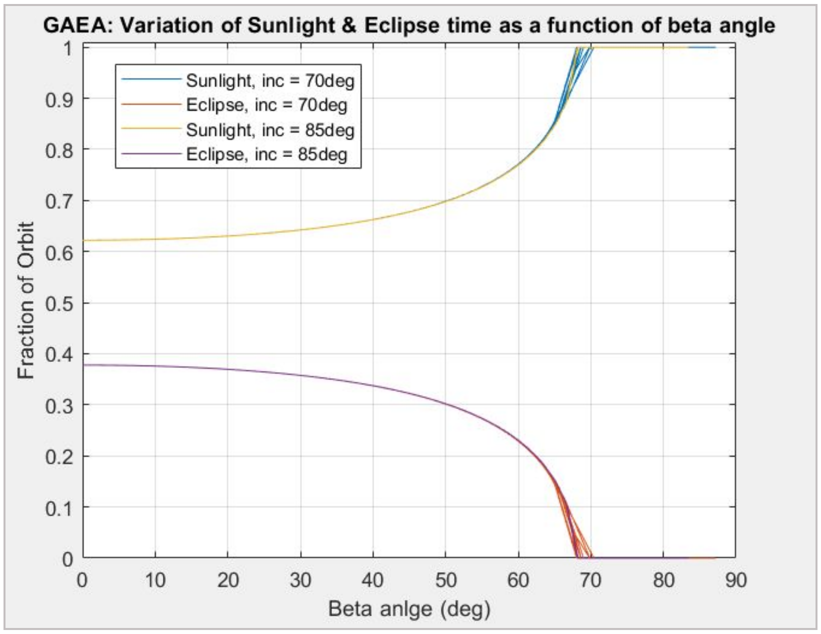


Figure F.22 Eclipse vs. Non Eclipse Time

The dimensions of our spacecraft are approx. 41 by 21 by 22 cm, with our projected area towards the sun A_p being (Total Surf Area - Nadir Surf Area = 0.445 - 0.0462 = 0.399m^2). Our A_{IR}, or nadir area is 0.0462 m^2. Alpha and epsilon, our absorptivity and emissivity, are determined by the surface finish of our spacecraft. By iterating on the equations below until our temperatures are within tolerance, we can determine our surface finishes and other thermal control devices. This initial calculation below will use 2 mm Aluminized Teflon as our surface finish ($\alpha = 0.1$, $\varepsilon = 0.66$)

Absorbed Energy

$\beta = 0°$ (cold)

$Q_{solar} = SA_p(\%\ Sun\ Time)\alpha_{avg} = 1317\ W/m^2 * 0.399m^2 * 0.62 * 0.1 = 32.58\ W$

$Q_{albedo} = Q_{incident-albedo}A_{IR}\alpha_{avg} = 79.1\ W/m^2 * 0.0462m^2 * 0.1 = 0.37\ W$

$Q_{incident-albedo}$ from table 22-11 of SMAD

$Q_{IR} = Q_{incident-IR}A_{IR}\varepsilon_{avg} = 186.8W/m^2 * 0.0462m^2 * 0.92 = 5.70\ W$

$Q_{incident-IR}$ from table 22-11 of SMAD

$Q_{in} = 143.831W$

$\beta = 90°$ (hot)

$Q_{solar} = SA_p(\%\ Sun\ Time)\alpha_{avg} = 1419\ W/m^2 * 0.399m^2 * 1 * 0.1 = 133.56\ W$

$$Q_{albedo} = Q_{incident-albedo} A_{IR} \alpha_{avg} = 21.4\ W/m^2 * 0.0462 m^2 * 0.1 = 0.1\ W$$

$$Q_{IR} = Q_{incident-IR} A_{IR} \varepsilon_{avg} = 224.4\ W/m^2 * 0.0462 m^2 * 0.66 = 6.84\ W$$

$$Q_{in} = 192.976\ W$$

Radiated Energy

$$Q_{out} = \sigma T^4\ \Sigma \varepsilon_n A_n = 5.670373 * 10^{-8} * T^4 * 0.66 * 0.445 m^2 = 2.32145 * 10^{-8}\ T^4\ W$$

Balance to solve for T
Cold average temperature: 11.0°C
Hot average temperature: 26.0°C

This simple worst-case analysis demonstrates that we do not require a radiator on the outside of the spacecraft to regulate temperatures, surface finishes will be sufficient to reflect heat. Despite this, each spacecraft component will have a thermistor to monitor the temperature and perform emergency maneuvers or power consumption changes if necessary.

e. Communications

The following section details the communication systems onboard every GAEA satellite to provide for mission data and telemetry downlink, command uplink and a redundant low rate downlink system.

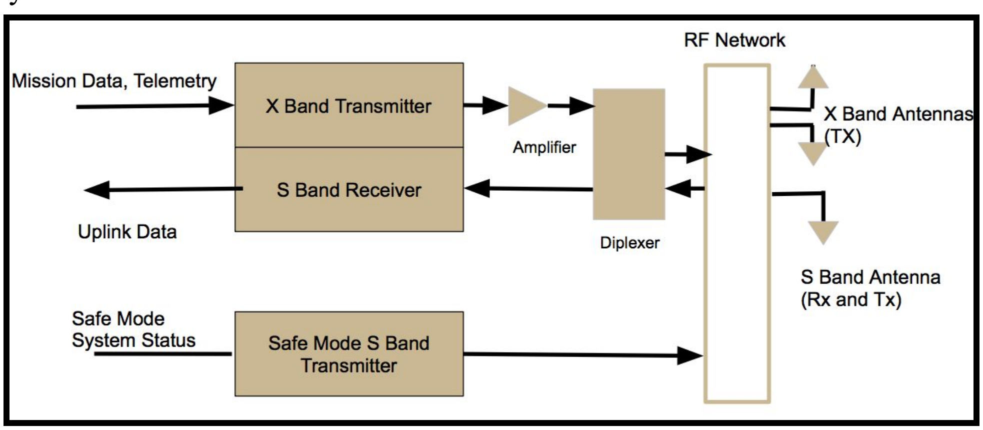


Figure F.23: Communications System Architecture

Mission Data Budget

An accurate estimation as to the amount of generated mission data was deemed a necessary first step prior to the sizing of the communications system. A final usable data rate of 120 Mbps (after encoding) is used in figure I.24 to satisfy requirements of mission data retrieval to the host ground station in Mcmurdo Antarctica. The following factors flowed down from mission critical subsystems to affect the communications system:

- Number of Observed Specular Points
- Cost
- Instrument Data Rate
- Per Pass Downlink Time
- Achievable Downlink Data Rate

Number of specular points per satellite is a sum of each science instrument's observed signals. The 35 specular points herein is a direct flow down from the orbits determination team. Cost is to be considered in the context of cost per pass with our ground station network, $490 as an estimate for using the NASA Near Earth Network (NEN)[6]. Per pass downlink time is also a direct result of the GAEA orbit and is seen in Figure E.3. As a result of these factors, the achievable downlink rate was designed to meet a bi-daily downlink schedule which was determined to be the optimal solution to the above constraints and achievable with COTS equipment.

Figure I.24 shows the complete mission data budget and margin for a bi-daily downlink schedule. Note that the instrument data rate includes a 1.5 factor for other system data coming from the spacecraft. The data generation rate comes from the product of the amount of time spent collecting data over land, the maximum number of specular points viewed at one time and the instrument data rate. The following section will present the design of the X-band communications system to meet the 120 Mbps target.

Downlink

Each project GAEA satellite is equipped with X band downlink capability for all mission data and spacecraft telemetry. This system is composed of 2 X-band patch antennas with a X-band transmitter. The downlink data uses a QPSK modulation with an 4/5 code rate for forward error correction (FEC) to achieve a 6 dB link margin at worst case and bit error rate (BER) of 1E-6. A final usable downlink data rate of 120 Mbps was achieved, which meets the

[6]Schaire, Scott H. "Near Earth Network (NEN) Users' Guide." NASA, NASA, 14 Mar. 2019, explorers.larc.nasa.gov/HPMIDEX/pdf_files/18_[Near_Earth_Network_(NEN)_Users%27_Guide_Revision_4_Redacted_]453-UG-002905.pdf.

requirement of downlinking 39,700 Mbits of data per pass to achieve a downlink schedule of every other day.

Due to the limited downlink time imposed by a LEO orbit and given that the associated costs of the NASA NEN are calculated per pass, independent of link time, the patch antennas are mounted to give the max beamwidth and thus maximum link time per pass. Figure E.3 contains simulated link durations per pass over several NEN sites with a beamwidth of 148°. Downlink times for each of these high latitude NEN sites is 350 seconds. The access times seen in figure F.24 are for Alaska Satellite Facility in Fairbanks, NASA McMurdo ground station in Antarctica, KSAT Svalbard in Norway and NASA Wallops ground station in Virginia. We get approximately the same access time from the 4 ground stations.

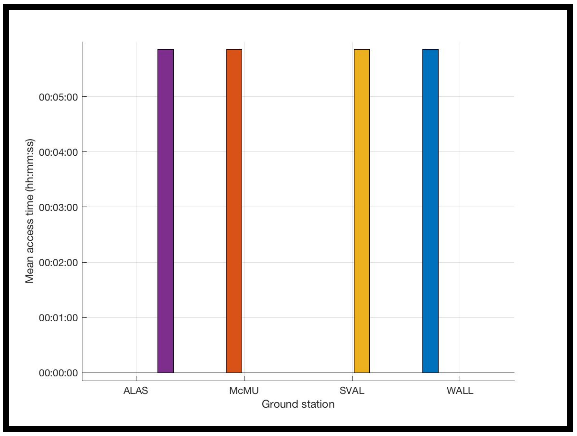


Figure F.24: X band Access Time From SoOp Simulation

X band hardware was downselected from NASAs State of the Art Small Spacecraft Technology[7] primarily based on gain, HPBW and wherein detailed information was available. These specifications can be found in the MEL.

Downlink Link Budget

Using the X band antenna specifications, and those of the MG1 McMurdo ground station in the NEN, the link budget in Figure I.25 for the payload downlink was calculated using a worst case 16° off horizon elevation angle. The required BER as a function of signal to noise ratio was taken from figure 16-17 in SME SMAD. Given the initial full data rate of 150 Mbps, the link closes with a link margin of 6 dB at worst case, meeting the recommended requirement

[7] Frost, Chad, et al. “Small Spacecraft Technology State of the Art.” NASA , NASA, Dec. 2015,www.nasa.gov/sites/default/files/atoms/files/small_spacecraft_technology_state_of_the_art_2015_tagged.pdf.

for a small satellite. The final usable data rate is the amount of unique mission data that can be transmitted.

Uplink Data Budget

Command data will be uplinked to the satellite during the regularly scheduled data downlink times. Interference is avoided due to the separation of systems between the S band uplink antenna system and X-band downlink antenna system. One nadir pointing wideband S band patch antenna will be used for S band uplink and emergency S band downlink should the X band system suffer a failure. The size of the S-band system is driven by the size of instrument commands needing to be sent.

The uplink data budget used estimates as to the size of commands needed to control the science instruments aboard every GAEA satellite. These commands control the data collection of every onboard instrument in .25 seconds time steps. Since the S band radio and antennas are on a separate system from the X band payload downlink, the MG1 facility will be simultaneously uploading these commands during the regularly scheduled bi-daily payload data downlinks. These commands have the by-product of saving computational resources on the spacecraft by leveraging ground based simulation and this is discussed in the **Main Computer** section of F.1.3.f . See figure I.26 for the uplink data budget. An uplink data rate of 6 Mbps was targeted from this data budget.

Specifications of the S-band components are located in the MEL. Like the X-band components, these were from or related to components on the NASA Smallsat State of the Art Report.

Uplink Link Budget:

The S band antenna in the MEL provides almost 150 seconds of uplink time available to communicate all uplink data. The above specifications and MG1 command characteristics were then used to compute the uplink data budget. The final specification for the system is a 6 Mbps data rate with QPSK modulation and no FEC encoding. The path length was computed using a 16 degree off horizon azimuth angle which, given the 70 HPWB is an extreme overestimate but is a mute point considering the link margin of this system is well above the target of 10 dB. This gives a BER well less than 10^{-12}. See figure I.27 in the appendix for the uplink link budget. Note that figure I.28 is the emergency S band downlink link budget with a data rate of 2 Mbps with an uncoded QPSK signal.

Ground Station Network

A trade study was performed to estimate the efficacy of an internally developed and positioned ground station and its associated cost versus the use of a ground station network such as KsatLite or Nasa's NEN. Given the benefits of redundancy, established protocol and worldwide coverage NASA's NEN was chosen as the ground station service provider. With a cost per pass of $490 and multiple high data rate facilities (both owned and operated by NASA itself and the ability to schedule with associate KSAT ground sites). NEN sites also offer a variety of methods for transporting or communicating the mission data to the project station. McMurdo's MG1 ground station was used as an estimate for reliable and consistent 150 Mbps X band downlink availability and S band command uplink. Given the numerous availability of NEN sites, a short term loss of service at McMurdo would not result in a large loss of mission data. Characteristics of the MG1 antenna were used in the link budget calculations and are shown in Figures I.28 and I.29

f. Flight Software/Onboard processing

Flight Software is the brain of a satellite as it processes and executes all the necessary instructions for spacecraft operations. It plays a key role in ADCS calculation and attitude and orbit correction planning, telemetry processing, power management, and thermal control management. The complexity of onboard processing is characterized by the number of source lines of code (SLOC). The estimated source lines of code for GAEA can be found in Fig. I.31 and we get a value of 48,840 lines with 20% margin. We see that the ADCS processing will take up most of the onboard processing. All the values are based on Firesat II source lines of code, and since Firesat II is bigger than GAEA, some of the processing values are already generous. Thus, a tighter margin of 20% was considered compared to a usual subsystem margin of 30%.

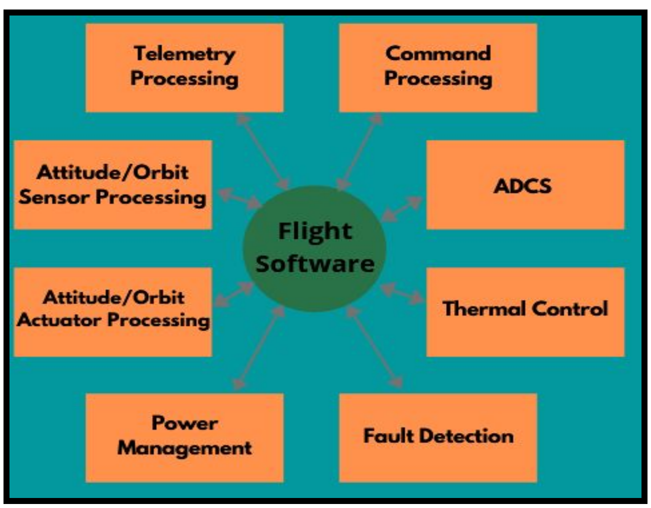


Fig F.25: Flight Software Schematic

Most of the SLOC will be reused from previous missions as all the subsystems are flight tested and have pre existing software. However, the SLOC has to be modified for communication as the communication system of the satellite is unique but since the communication data processing is not taking place on board, a general SLOC for communication can be modified for GAEA's purpose.

The team will prefer to use Agile development method over waterfall and spiral method because Agile method emphasizes iterative evolution and testing without significantly impacting the cost and schedule. Waterfall method has lower reliability due to limited phase by phase testing. Spiral method is typically very costly because it is highly iterative as testing of each function is performed at every phase of the development.

Main Computer

The main computer chosen for this mission is the Q7S by Xiphos Technologies. In addition to having been operating in orbit since 2016, its predecessors have been flight proven since 2002. A downselect of COTS and flight proven hardware was performed by weighting for power draw, clock speed, price, error correction capability, radiation hardening and mass. This decision matrix is shown in the figure below. Multiple options beyond those in the decision matrix were also explored but quickly ruled out due to weight or power concerns.

The primary requirements for the main computer beyond I/O with all other systems, as shown in Figure F.1 is control of the science instruments. Briefly described in section F.1.3.e, in order to the main computer needs to simply follow the instructions communicated from ground in instrument data collection. To save power on board, the satellite will uplink time stamp data, which will help the satellite determine when it needs to turn the payload on and off based on ground calculations of when the receiver will not observe any useful data as all the spectral cells will then be over water. These commands leverage the extensive computational power available on the ground to calculate the transmitter-receiver geometry of the SoOp system. Each instrument is sent these commands in a .25 second time step with no on-orbit calculations necessary. We do not expect any increase in operations cost to compute the uplink data as it will be within the mission operations team's capability. This data simply needs to be transferred to MG1 for uplink.

This allows for a considerable size down of the main computer to the smallsat and cubesat variants that were explored. An additional requirement of the main computer is to interact with the disparate sources of data storage onboard, including on the X band and S band transceivers and allocate the storage appropriate for each downlink. Specifications of the Q7s are shown in the MEL. A final decision for the onboard computer was made from the decision matrix in Figure F.2.

Objective	Weight	DP-OBC-0402 Data Patterns ltd	Q7S Xiphos Technologies	Proton 2X Box Space Micro	OBC ISIS
Power Draw	0.2	4	8	1	6
Clock Speed	0.2	4	7	10	5
Price	0.1	5	5	5	5
Error Correction	0.1	8	10	5	5
Radiation Hardening	0.2	5	8	7	5
Mass	0.2	6	8	3	8
SCORE		5.1	7.7	5.2	5.8
RANK		4	1	3	2

Fig F.26: Onboard Computer Decision Matrix

g. Space Environment

The space environment challenges scientists and engineers have identified in the past have seldom changed. The greatest threats to our satellite and its instrumentation include radiation and debris hazards. These pose the biggest risks because they are highly complex phenomena which does not allow us to concretely simulate and predict these events.

Radiation

Radiation hazards are the largest constant threat to our mission in the space environment. Given the anticipated launch date of November 2026, it is important to understand the behavior of our Sun and the dangers it may bring to our mission. Solar Particle Events (SPEs) are common occurrences of proton emission from the sun either through solar flare activity or coronal mass ejections (CMEs). These solar events have multiple effects on our spacecraft. High-energy protons emitted from the sun can interact with and damage the electromagnetic components of our spacecraft. These events can degrade solar array elements and increase background electromagnetic noise.

Solar Cycle and Solar Particle Events (SPEs)[8]:

[8] “Homepage | NOAA / NWS Space Weather Prediction Center.” *Noaa.Gov*, 2000, www.swpc.noaa.gov/

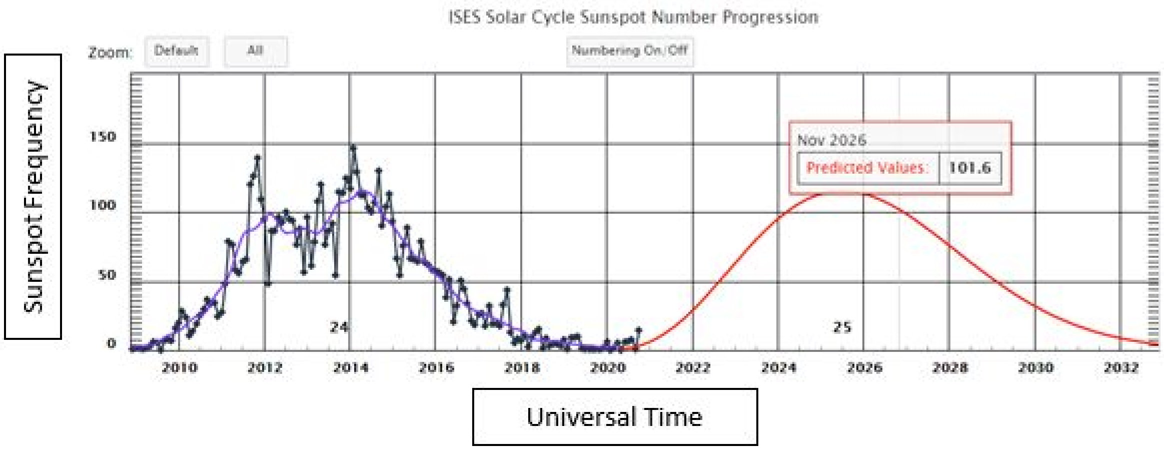


Figure F.27: Sunspot Frequency over time

The Solar Cycle is the periodic reversal of the Sun's powerful magnetic field, which manifests in various levels of sunspot activity over a period of approximately 22 years. On the expected launch date of November 2026 as can be seen by examining the Figure, the Sun is predicted to be near the crest of its solar maximum. This means that launching our satellite increases the inherent threat it receives from SPEs during its mission lifespan, compared to launching during a solar minimum. During a solar minimum, SPE frequency may be a single event in one or two weeks. However, during solar maximum, these events may occur up to once every day. The level of threat can only be mitigated by proactively preparing additional radiation protection measures onboard the satellite in the forms of shielding critical scientific and solar array equipment.

Galactic Cosmic Rays (GCRs):

GCRs pose a large threat to the spacecraft. While the likelihood of large cosmic ray events is unlikely, constant exposure to harmful radiation is expected. The results below are taken from NASA's R-Gentic program for a Sun-synchronous satellite at various levels of altitude from 400 km to 700 km.

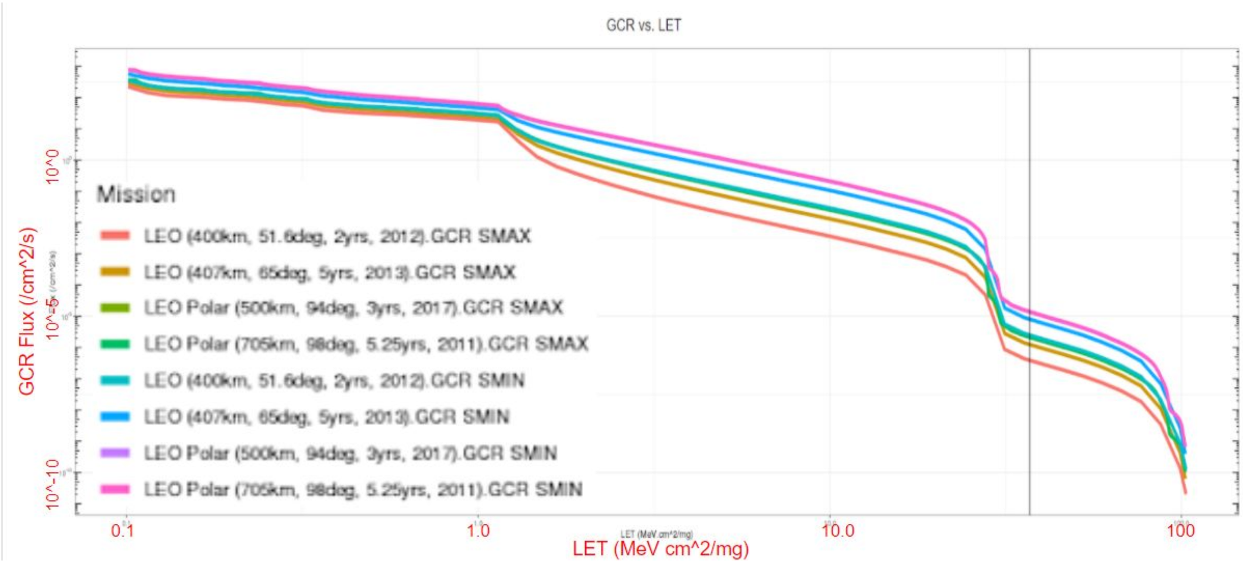


Figure F.28: Galactic Cosmic Rays Flux (/cm^2/s) vs. Linear Energy Transfer (MeV cm^2/mg)

Ionizing Radiation consists of harmful proton emissions from the Sun that interact with electromagnetic systems onboard the satellite. Such interference may affect solar panel arrays, data interpretation and telemetry to ground stations and overall mechanical degradation. Such events are very common, and large amounts of exposure over the mission lifetime poses a serious threat to meeting scientific and mission requirements. The figure below is again taken from NASA's R-Gentic software.

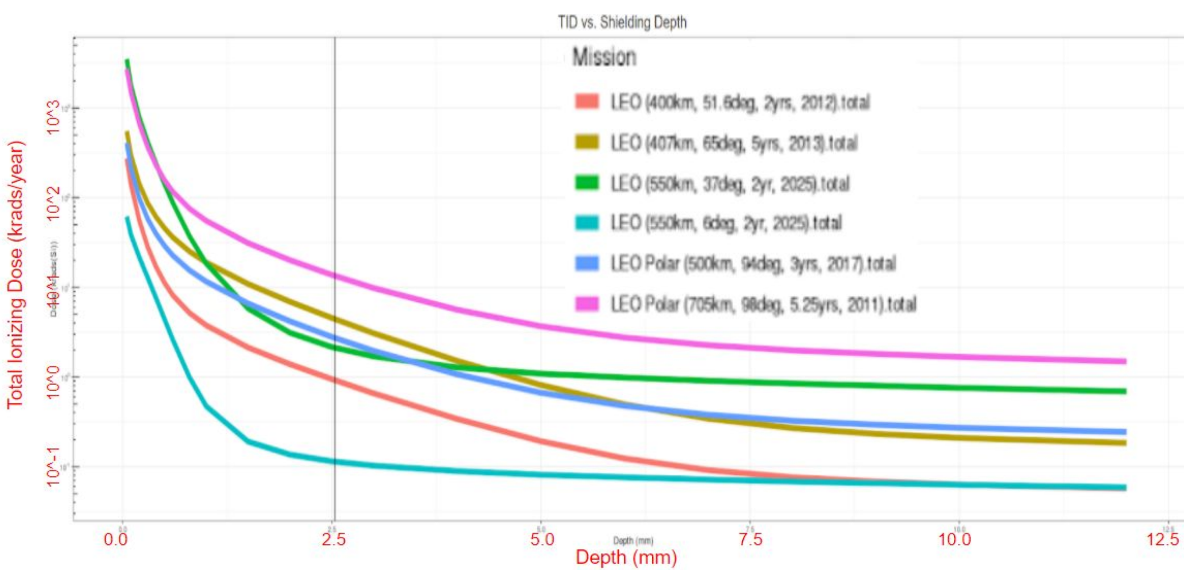


Figure F.29: Total Ionizing Dose (Dose/Year in krads) vs. Shielding Depth (mm)

To mitigate risk associated with radiation, we see from the above graph that it is necessary to have a shielding depth of at least 2.5 mm, in order to mitigate higher consequence risk from our mission.

Space Debris

Space debris poses the next greatest threat to mission success. Taken from the European Space Agency's SPENVIS software, it is important to understand the threat assessment of debris larger than 1 cm in diameter. While the likelihood of events are exponentially unlikely, such an event could result in catastrophic failure. Even so, debris collision events less than 1 cm in diameter occur often every year, which can contribute to gradual wear and tear over the mission lifetime. In order to mitigate debris hazards, GAEA satellites will have 5mm of Aluminum 6061 to provide sufficient shielding.

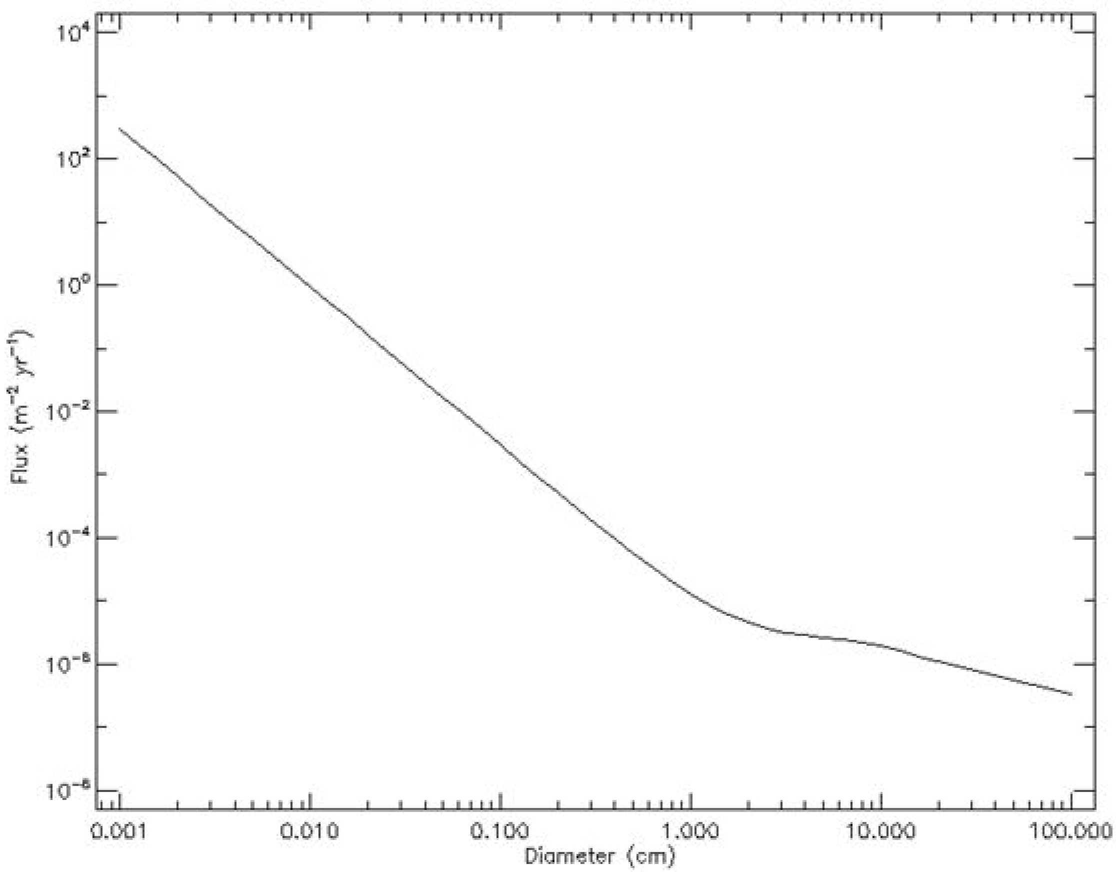


Figure F.30: Debris Flux (square meter per year) vs. Diameter (cm)

2. Flight System Contingencies and Risks

In this section, we will analyze the flight system contingencies. We will also discuss the Technology Readiness Level of our systems, and analyze the risk associated with our operation with a comprehensive risk traceability matrix.

Parameter	Margin
Dry Mass	30%
Mass of Propellant	30% deltaV + 30% mass + 62 km Altitude
Power	30%
Attitude Control System	30%
Thermal control	30%
SLOC	20%
Overall Cost	30%
Overall Schedule	30%

Figure F.31: Margin of different subsystems

The above margins are based on GOLD rules[9] for Pre-Phase A mission development. These margins give us a substantial buffer to make any necessary changes in the future without compromising the development of other subsystems. This will also enable us to tackle any unforeseen challenges.

[9] United States, NASA, GSFC. *Rules for the Design, Development, Verification, and Operation of Flight Systems*. standards.nasa.gov/standard/gsfc/gsfc-std-1000.

A Technology Readiness Level (TRL) is a method developed by NASA and the ESA to estimate the maturity of acquired technologies for mission operations.

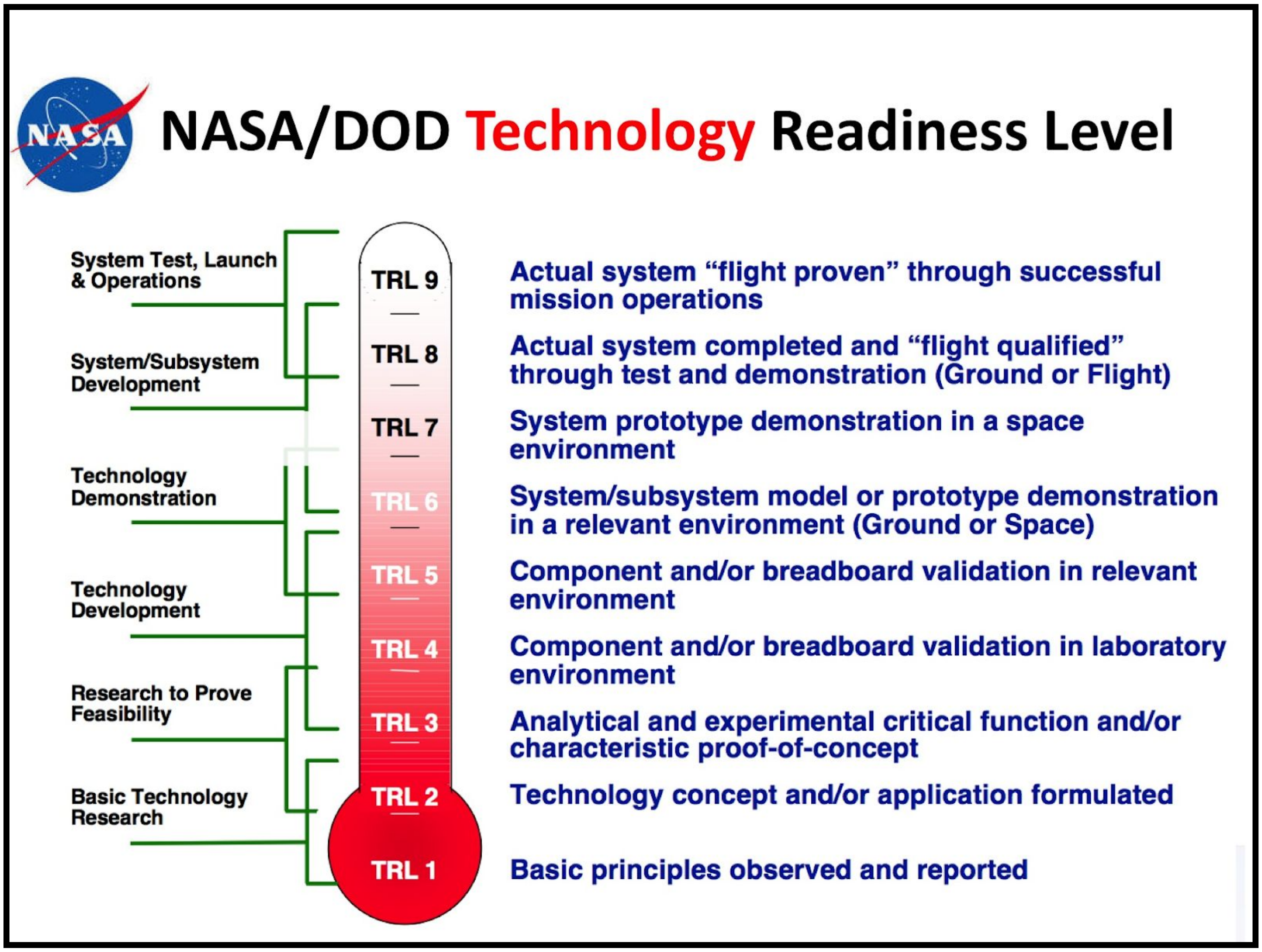


Figure F.32: Technology Readiness Level

Space-qualified components with TRL-9 will be used to minimize risk in this operation. This means that vacuum testing of materials, scientific instruments, solar panels, and thermal equipment is necessary. Radiation, extreme temperature, and micro debris testing is necessary for instrumentation, thermal equipment, and materials, as the satellite will exist in a harsh space environment, described in Section F.1.3.g. Some key component testing needed at TRL-9 include shake table testing of the momentum wheel, as well as static and vibration testing.

It is important to understand that there are many areas of risk in this operation. A detailed risk analysis will be provided in this section. As we can see from the risk traceability matrix above, we will begin by addressing what poses the highest level of risk to our operation. We can see that while there are no mission imperative risks of both high likelihood and high consequence as we carefully designed our mission to mitigate any risk that would have lied in that region. However, there are many other risks that need by simultaneously addressed, which are Actuator/Attitude Failure, Insufficient Delta V, Structural Vibrations, Solar Radio Flux

Levels, Space Debris of 1 cm diameter, and GCR flux of less than 10^{-5} cm/s. To mitigate these higher level risks, we will address them individually. The way to mitigate solar radio flux levels is certainly difficult, as predicting them is very difficult. However, 5 mm shielding will help prevent radiation hazards from penetrating our scientific instruments, solar panels, and thermals. While mitigating space debris of large diameter is difficult, the shielding is also designed to withstand micro impacts of space debris of less than one centimeter. Actuator control, attitude control, structural vibrations and insufficient delta V are all mitigated by using TRL-9 equipment, sufficient margins and redundant systems. In addition, Extensive testing is conducted to minimize these failure points.

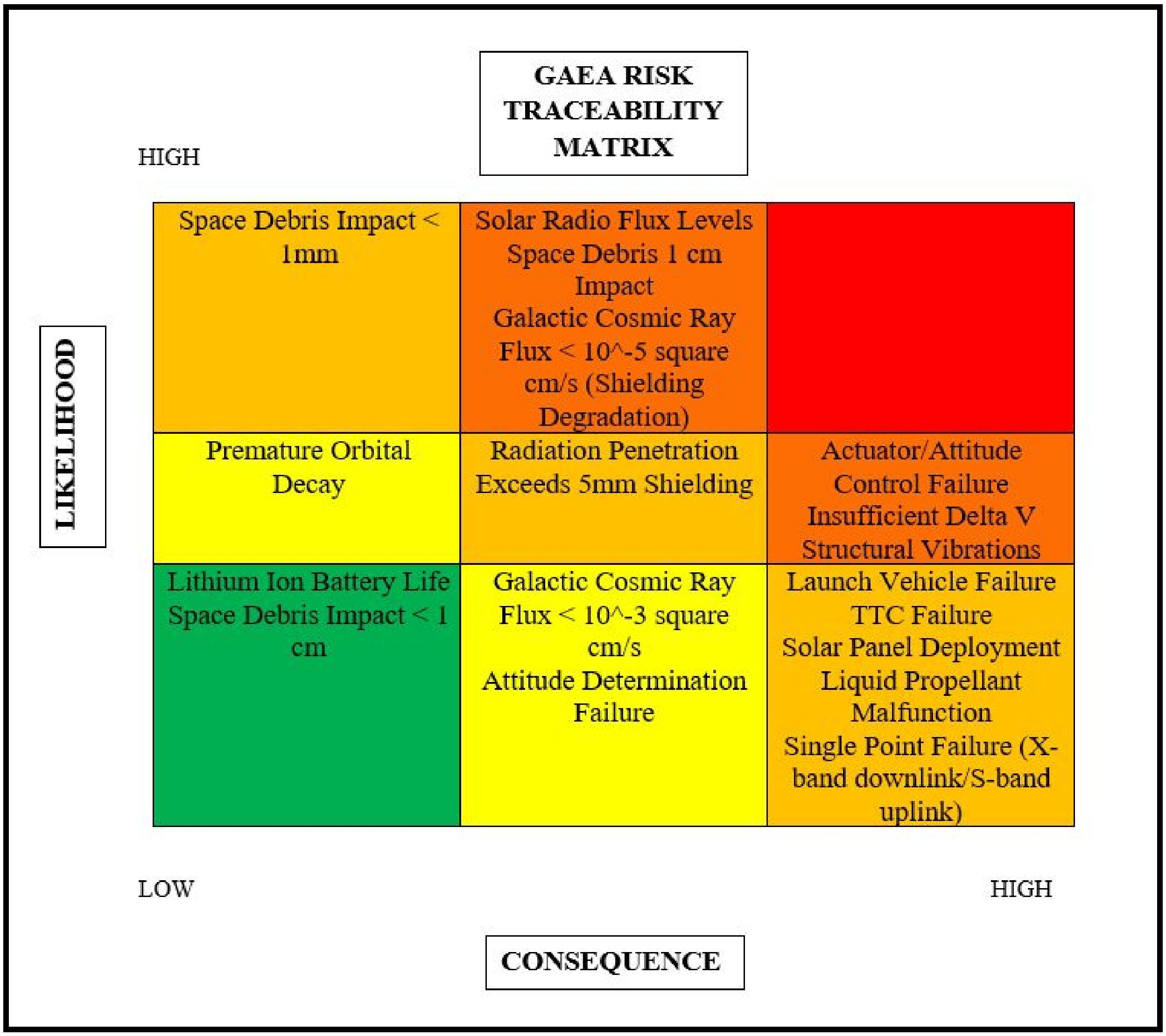


Figure F.33: Risk Traceability Matrix

3. Mission Operations

The GAEA mission will entail a 3 year Mission Operations phase. Using a trade study on the calculations for a delta-V budget, the GAEA satellites will be launched into orbit using a liquid propellant Dual Mode (N2O4/N2H4). This is accomplished using the Minotaur 1 as the launch vehicle as determined by another study described in **F.1.2 Launch Services and Launch Vehicle Capacity**. Since GAEA is using a previously tested and launched spacecraft, there is no need for facilities related to designing, building, and testing the launch vehicle. Derived from the inclinations required for this mission, the launch site Kodiak Launch Complex (KLC) now named Pacific Spaceport Complex (PSC) was chosen. Using the NASA Routine Payload Checklist, the default launchpad for this spacecraft is LP-1. LP-1 at KLC being the default launch location for the Minotaur 1 spacecraft satisfies the requirements that Minotaur 1 needs to properly launch the GAEA satellites into their orbit. Once Minotaur 1 reaches 500 km altitude at the respective inclinations, the satellites are jettisoned into their proper orbits one-by-one. Thus ends the launch phase, and begins the data gather and downlink phase.

The first 8 satellites are spread out evenly by true anomaly (0, 45, 90, 135, 180, 225, 270, 315) at 70 degrees inclination, while the remaining 4 satellites are also spread out evenly by true anomaly (0, 90, 180, 270) at 85 degrees inclination. Adjustments will be needed to maintain the satellites in their respective orbits for the entire 3-year mission lifetime and are accounted for in the delta-V budget. L-band transmitters will gather data for surface soil moisture while I-band and P-band transmitters will gather data for the root zone soil moisture. The satellites will downlink data every other day to NASA's Near Earth Network (NEN) ground stations. Due to the numerous ground stations in this network, temporary loss of signal to a single ground station poses no issue and can be scheduled for another temporary site. This is further discussed in **F.1.3.e Communication/TTC.** A COTS X-band antenna is used to transmit science data to the ground stations (NEN). S-band communications are used for the command signal to the satellite as well as an emergency low data-rate downlink. The main project site in the NEN is McMurdo Ground station in Antarctica.

To gather data, the GAEA satellites have an array of RF transmitting antennas to gather soil moisture data. For surface soil moisture data, L-band is used via Beidou, Galileo, GLONASS, and GPS. For root zone soil moisture data, I-band is used via ORBCOMM, NOAA, and METEOR and P-band is used via MUOS. To ensure root zone soil moisture readings, I-band and P-band measurements are assured to take measurements within 12 hours of each other. Satellites will take measurements at all times to ensure maximum coverage. These communications systems in addition to the mission operations of GAEA can be observed below.

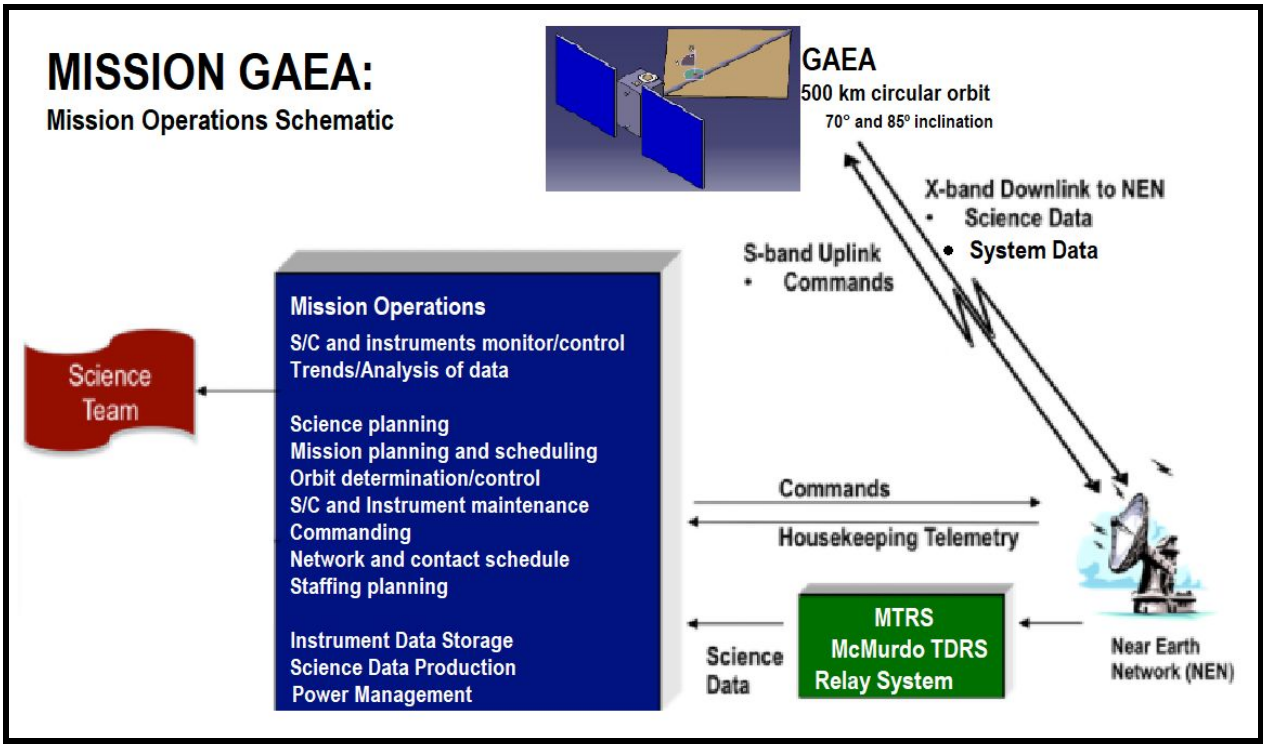


Figure F.34: Visualization of mission operations communication organization between GAEA, NEN, mission operations, and the client

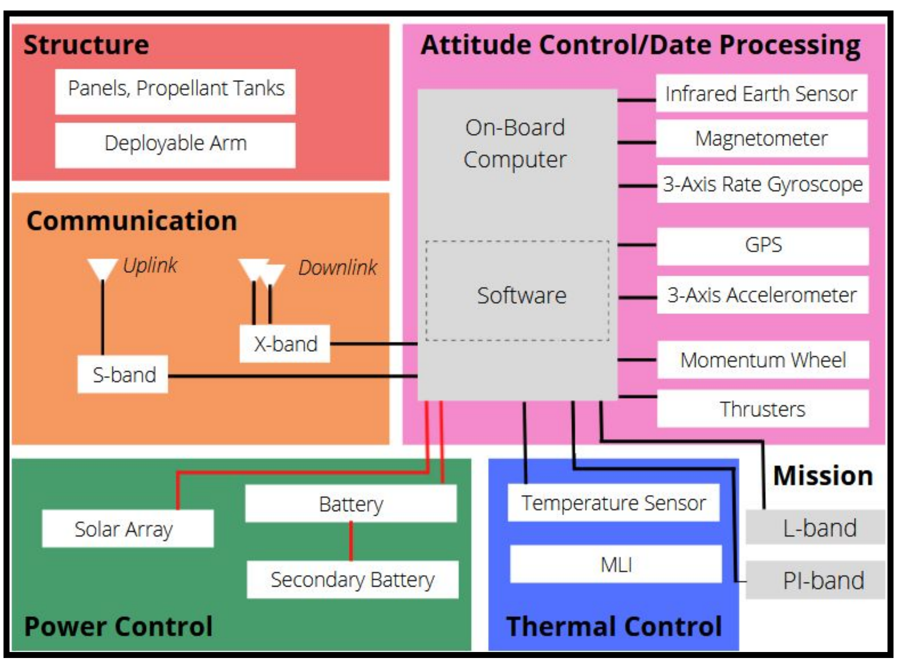


Fig F.35: Visualization of power and data flow on board, red lines are power flow, black lines are data and power flow

4. Development Approach

System Engineer will be responsible for interface management.

Software development is one of the few areas that, if planned correctly, can greatly increase the reliability of a satellite and decrease cost and schedule. We plan to use Agile development method as it easily adapts to dynamically changing environments, which is most suitable for a highly iterative process of satellite development. It allows easy distribution of load, and a good estimate of cost and schedule prediction. It emphasizes continuous testing without excessive costly testing like in the spiral development method.

All the relevant mechanical and electronic components to be used for the software development purpose will be space qualified and rated TRL 9 to minimize the risk of failure. Enough margins in memory and SLOC are considered to account for unexpected additions to flight software. The team will closely follow NASA's guideline: "NPR 8705.6, Safety and Mission Assurance (SMA) Audits, Reviews, and Assessments"[10] to increase reliability and to formulate a strong fault management plan.

The mantra for development will be to keep the software simple and structured to increase reliability and decrease cost of evaluation.

Few methods that we plan to use to minimize the risk of failure of Flight Software are:

1. Follow IEEE software-based tools and standards to structure the development, to improve development management and technical control

2. Add a special circuitry referred to as Watchdog Timer to restart computer when hung due to any reason

3. Include Error Detection and Correction (EDAC) circuitry to counteract single event upsets.

The key to successful development is smart testing, so we plan to have a comprehensive Software Test Plan (STP) to maximize software reliability.

[10] United States, NASA, GSFC. *NPR 8705.6, Safety and Mission Assurance (SMA) Audits, Reviews, and Assessments*. nodis3.gsfc.nasa.gov/displayDir.cfm?t=NPR&c=8705&s=6C.

5. Assembly, Integration, Test and Verification

Assembly, Integration, Test and Verification (AITV) is one of the very few tools that we have to manage any risks and to check if the satellite as a whole will be able to meet all the mission requirements as it is nearly impossible to make any corrections to a satellite once it is on orbit. We hope to carry three types of test: Functional Test, Integration Test and Environmental Test. This will help us to find any potential fixable source of errors for each sub-system, when multiple subsystems will be integrated and tested in a simulated space environment. Since 12 satellites need to be built, a strong AITV process is required to minimize the recurrence of any error. Hence, an Engineering Test Unit (ETU) will be developed to streamline the development and testing process. The important tests include:

1. Functional Tests:

    a. Static and Dynamic Load Test: To test the structural integrity of the satellite, so that it can withstand the expected structural loads during launch and deployment.
    b. Antenna Performance Characterization Test: To test if each antenna component works as expected
    c. Vibration Test: To test if the satellite will be able to withstand the expected vibrations during launch, deployment and solar panel correction maneuver.

2. Integration Tests:

    a. Flat-sat: To streamline assembly and integration planning
    b. ADCS with flight software: To test performance of ADCS subsystem with the necessary software
    c. Uplink and Downlink performance of antenna placed at a considerable height within the link range with a ground station: To test performance of X-band and S-band receiver and transmitter with the necessary software

3. Space Environment Tests:

    a. Vacuum Test: To validate required performance of all the sub-systems in vacuum
    b. Thermal Test: To test performance of thermal control subsystem and if all the other subsystems are able to perform as expected in the expected space thermal environment
    c. Helmholtz Cage: To test if the satellite is able to optimally perform under expected magnetic field environment.

All these tests can be performed in the various facilities of NASA Goddard Space Flight Center. The test facilities include the High-Capacity Centrifuge, Space Environment Simulator, Shaker Table and High Bay Clean Room. Since these facilities are easily available and frequently used, testing will not change the planned cost and schedule of the mission.

Developing an accurate bottom's up cost estimate for AITV is very difficult due to the classified nature of cost data pertaining to testing and testing operations. Luckily, the Small Satellite Cost Model (SSCM) regresses classified historical data from AITV operations costs of past space satellite projects. When inputting the required parameters into the SSCM, implementing the total lot cost equation for 12 satellites, and adding a cost margin of 30% as per the GOLD rules, the cost of AITV comes out to be $3,934,000. To better understand how this number came to be, and the costing of the GAEA project in general, see **section H**.

The conducting of the AITV segment of the project occurs during Phase D. This phase has the Operational Readiness Review, a Key Decision Point (KDP) at the halfway mark, six months, into Phase D. The majority of the tests (5 out 9 tests) are to be scheduled to take place before this KDP, with each test taking no longer than 2 weeks each. In addition, the tests are to be conducted from highest risk to lowest risk, in order to preserve time if a system fails.

6. Schedule

GAEA's project schedule is to follow a schedule similar to that of SMAP's. The table below gives the planned schedule with a 30% margin as recommended for a Pre Phase A project[11].

Phase	***Date Interval***	***Key Decision Point***
Pre-Phase A (Proposal Writing)	Current - 12/11/2020	Proposal Submitted
Pre-Phase A	6/1/2021 - 8/1/2021	MCR
Phase A	8/1/2021 - 10/1/2021	SSR
Phase B	10/1/2021 - 1/1/2022	PDR
Phase C	1/1/2022 - 6/1/2022	CDR
Phase D	6/1/2022 - 6/1/2023	FRR
Phase E	8/1/2023 - 5/1/2026	Mission Ops
Phase F	5/1/2026	Decommission
Phase F (W/ 30% Margin)	5/1/2027	Decommission

Figure F.36: Mission Timeline

[11] "NASA Cost Estimating Handbook (CEH)." *NASA*, NASA, 19 May 2016, www.nasa.gov/offices/ocfo/nasa-cost-estimating-handbook-ceh.

To ensure a more accurate estimate rather than just following a rough outline on SMAP's schedule a schedule duration estimation was run. This duration estimation is a product of two equations from SME-SMAD:

$$Effort\ (MM)\ =\ 3.312 * (KSLOC)^{1.2} * \pi * EAF$$

$$Duration\ (in\ months)\ =\ 4.376(MM)^{0.32}$$

The variables KSLOC and EAF correspond to Thousands of Source Lines of Code and Effort Adjustment Factor, respectively. The latter is a value of 1.00, which corresponds to a nominal level of effort. The result of these two equations estimates a project development estimate of 40.94 months, or 3.41 years without margin. This is equivalent to the schedule of SMAP ± 1 month. Incorporating a schedule margin of 30%, a number which corresponds to the suggested schedule margin for a Pre-Phase A project, the schedule gets pushed back one year. Still with a 30% margin, the spacecraft are to launch before the required launch readiness date of November 30th, 2026 and less than five years prior to proposal acceptance. This schedule is fully displayed in the Gantt Chart located in the appendix.

G. <u>Management</u>

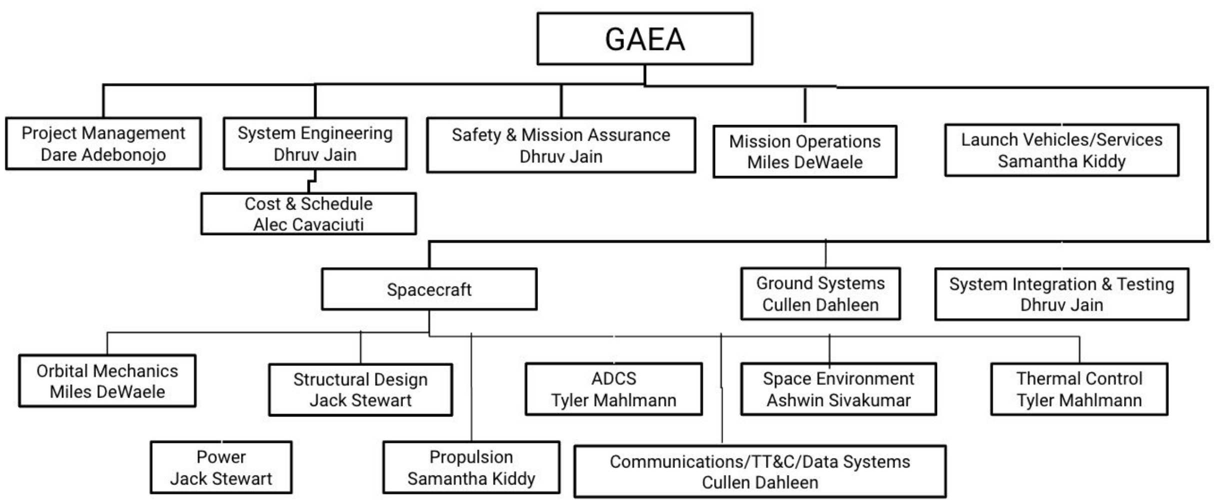


Figure: G.1 GAEA Project Structure

The top risks considered to the Project manager are:

1. Galactic Cosmic Ray Flux < 10^-5 square cm per second
2. Space Debris >= 1 cm Impact

Galactic Cosmic Rays are an area for notice. This radiation can damage electrical and mechanical components and may interfere with our ability to make reliable measurements. Our mitigation strategy for GCR will be implementing adequate shielding on various components especially parts sensitive to GCR (i.e. solar panels). Space Debris is a risk with low probability of collisions with particles greater than 1 cm in diameter. Shielding fragile parts will also help mitigate the risk of system failure due to these collisions.

H. Cost and Cost Estimating Methodology

Parametric cost models are tools to assist cost and scheduling engineers in the process of cost estimating one spacecraft, or a constellation of multiple spacecraft. They are based upon regressed historical data of the actual cost of past spacecraft that have been developed. These cost models do not only estimate the cost of the physical spacecraft, but also the other mission components that are included in the total cost of a spacecraft mission. Each of these components are WBS (Work Breakdown Structure) elements. These WBS elements that are required to develop an accurate cost model, according to SME-SMAD are as followed in **Table H.1**.

SME-SMAD WBS Element
1.0 Spacecraft
1.1 Spacecraft Bus
1.1.1 Structure
1.1.2 Thermal Control
1.1.3 Attitude Determination and Control System (ADCS)
1.1.4 Electrical Power System (EPS)
1.1.5 Propulsion
1.1.6a Telemetry, Tracking and Command (TT&C)
1.1.6b Command and Data Handling (CD&H)
1.1.7 Integration, Assembly & Test (IA&T)
1.1.8 Flight Software
1.2 Payload
1.2.1 Communications
1.2.2. Surveillance
1.2.3 Spacecraft Integration & Test

2.0 Launch Vehicle
3.0 Ground Command & Control
4.0 Program Level
5.0 Flight Support Operations
6.0 Aerospace Ground Equipment
7.0 Operations

Fig H.1 WBS Elements

Determining which parametric cost model to use is based upon a number of factors, with the spacecraft mass as the main driver of the decision. The GAEA constellation consists of 12 spacecraft each with a mass of 46.60kg. The only model available from the parametric cost models in SME-SMAD, of which allowed for spacecraft masses of lower than 500 kilograms, was the Small Spacecraft Cost Model (SSCM), developed by The Aerospace Corporation. The WBS elements that are included in the SSCM are as follows in **Fig H.2**.

SME-SMAD WBS Element
1.0 Spacecraft
1.1 Spacecraft Bus
1.1.1 Structure
1.1.2 Thermal Control
1.1.3 Attitude Determination and Control System (ADCS)
1.1.4 Electrical Power System (EPS)
1.1.5 Propulsion
1.1.6a Telemetry, Tracking and Command (TT&C)
1.1.6b Command and Data Handling (CD&H)
1.2 Payload
1.3 Integration, Assembly, and Test
4.0 Program Level (Project Team)
5.0 Launch and Orbital Operations Support (LOOS)
6.0 Aerospace Ground Equipment (AGE)

Figure H.2 WBS Elements SSCM

In order to determine the cost of each WBS element using the SSCM, the cost driver(s) for each element must be identified. For the SSCM, the cost driver for determining the price of the spacecraft subsystems is the respective subsystem mass. These masses are then entered into the parametric cost model equations developed from historical regressions. The other WBS elements are then determined by the spacecraft bus's total mass, which is a result of summing up each subsystem's individual mass. So in the end, the result outputted by the SSCM is a sum of the majority of the WBS elements needed to accurately determine the cost of a spacecraft mission, by simply inputting the spacecraft subsystem's masses. These subsystems include a 30% mass margin. This inclusion of a 30% mass margin is mandatory as per the GOLD (Goddard Open Learning Design) rules and is described in section **F.2.**

Non-recurring cost for development plus one qualification unit							
SME-SMAD WBS Element	Cost Driver(s)				Estimatined Cost in FY2010 [$K]	Absolute Standard Error of the Estimate (SEE) FY2010 $	Estimated Cost in $K for Fiscal Year: 2022
	Cost Driver	Input a value between the Lower Limit and Upper Limit in the Value					
		Lower	Value*	Upper			
Spacecraft							
1.1 Spacecraft Bus (alternate CER when no component information is available)	Spacecraft Bus Dry Weight (kg)	20	30	400	$3,651	$3,696	$4,639
1.1.1 Structure	Structure Weight (kg)	5	10.77	100	$901	$1,097	$1,145
1.1.2 Thermal Control	Thermal Weight (kg)	5	0.8	12	$339	$119	$430
1.1.3 Attitude Determination and Control System (ADCS)	ADCS Dry Weight (kg)	1	2.39	25	$1,917	$1,113	$2,435
1.1.4 Electrical Power System (EPS)	EPS Weight (kg)	7	6.12	70	$3,247	$910	$4,125
1.1.5 Propulsion (Reaction Control)	Spacecraft Bus Dry Weight (kg)	20	10.16	400	$145	$310	$184
1.1.6a Telemetry, Tracking, and Command (TT&C)	TT&C Weight (kg)	3	0.8	30	$527	$629	$670
1.1.6b Command and Data Handling (CD&H)	Command and Data Handling Weight (kg)	3	2	30	$849	$854	$1,079
Payload							
1.2 Payload	Spacecraft Bus Total Cost ($K)	2,600	4,639	69,000	$1,856	Not Given	$2,357
Spacecraft Integration, Assembly, and Test							
1.3 Integration, Assemby, and Test (IA&T)	Spacecraft Bus Total Cost ($K)	2,600	4,639	69,000	$645	Not Given	$819
Program Level							
4.0 Program Level	Spacecraft Bus Total Cost ($K)	2,600	4,639	69,000	$1,062	Not Given	$1,350
Flight Support							
5.0 Launch and Orbital Operations Support (LOOS)	Spacecraft Bus Total Cost ($K)	2,600	4,639	69,000	$283	Not Given	$359
Aerospace Ground Equipment (AGE)							
6.0 Aerospace Ground Equipment (AGE)	Spacecraft Bus Total Cost ($K)	2,600	4,639	69,000	$306	Not Given	$389
Total							
All Subsystems					$15,390		$19,551

Figure H.3 (SSCM)

The total lot cost equation is used to estimate the entire cost of the satellite calculation. Implementing this equation requires the use of a "learning curve" or metric used to determine the price drop off for every satellite manufactured after the one before it. The learning curve used for this cost model is 0.72 based upon historical data[12].

[12] Brown, Nicholas F, and Timothy P AndersonT. "Learning Rate Sensitivity Model." *Nasa*, The Aerospace Corporation, 14 Aug. 2018, www.nasa.gov/sites/default/files/atoms/files/27_learning_rate_sensitivity_model-2018_nasa_cost_symposium.pdf.

$$Total\ Lot\ Cost\ =\ First\ Unit\ Produced\ \times \#\ of\ Satellites^{\,(1+ln(LearningCurve)/ln(2))}$$

It is important to note that there are differences between the list of required WBS elements to develop an accurate cost model, and those included into the SSCM. However, the WBS element from **Figure H.1**, 1.1.7 (IA&T) is incorporated into the WBS element 1.3 in **Figure H.2**. In addition, the level three payload WBS elements from **Figure H.1** are included in the 1.2 Payload and 1.3 Integration, Assembly, and Test elements of **Figure H.2**. Thus, the WBS elements that are missing from the SSCM estimation and are to be summed using a bottom's up approach. **Figure H.4** lists these specific WBS elements.

SME-SMAD WBS Element
1.1.8 Flight Software
2.0 Launch Vehicle
3.0 Ground Command & Control
7.0 Operations

Fig H.4

These 4 WBS elements are determined using a bottom's up approach, a cost estimation method which sums up WBS elements purely based upon the spacecraft's actual determined WBS element cost. The following equations and results from implementing those equations are given in **Fig H.4**.

The WBS element of F.1.3.f flight software is estimated using the average dollar amount of one Software Line of Code (SLOC) and multiplying it by the number of SLOC in the Flight Software (~48,870)

The WBS element of 2.0 launch vehicle is a flat cost of one Minotaur 1 launch with the desired mission trajectory.

The WBS element of 3.0 is estimated by multiplying the dollar amount of one pass to the ground station by the number of total passes throughout the mission. This number of passes is 1 pass per satellite every 2 days, for the entire mission duration. This comes out to be 6,570 passes. In addition WBS 3.0 includes the facility cost which is estimated by the typical size of a satellite communication center multiplied by the corresponding price per m^2 of the facility (SME-SMAD).

The WBS element 7.0 is a summation of all operations costs that go into a highly complex mission. Using 6 full time engineers N_{ENG} at FTE_{ENG} (Full-Time-Equivalent) of \$200K, 48000

SLOC, a facility flat estimate of $1,250K/yr, and a H/W Acquisition of $1,400K, an accurate bottom's up estimate is achieved. This summation accounts for the PMSE (Project Management and Systems Engineering), Space Segment Software Maintenance, and Ground Segment costs for the GAEA mission.

WBS Element	***Equation***	***Cost ($M)***
1.1.8 Flight Software	$400*SLOC	19.200
2.0 Launch Vehicle	N/A (Flat Rate, *Minotaur 1*)	28.800
3.0 Ground Command and Control	($490)*(Number of Passes + Number of Uplinks) + Facility Cost	6.779
7.0 Operations	$(N_{ENG} \times FTE_{ENG} + N_{SLOC}/28,800 \times FTE_{ENG} + 7\%\ of\ H/W\ Acquisition + Facility\ Lease + N_{SLOC}/16,000 \times FTE_{ENG} + PMSE\ (15\%\ all\ costs))$ * 3 years	12.009

Fig H.5

In order to develop a complete total cost for the GAEA mission, the parametric cost estimation must be combined with the bottom's up approach. Table H6 sums up both approaches and gives a total mission cost with margin. According to the NASA Cost Estimating Handbook, a responsible Pre-Phase A cost margin is 30% as per the GOLD rules. When this margin is applied the mission budget is $9.088M below the cost cap of $190M USD FY22.

Cost Element	***Cost ($M)***
Total Lot Cost (Parametric Approach, *SSCM*)	72.258
1.1.8 Flight Software	19.200
2.0 Launch Vehicle	28.800
3.0 Ground Command and Control	6.779
7.0 Operations	12.009
Total Cost (w/o Margin)	139.055
Total Cost (w/ 30% Margin)	180.912

Fig H.6

I. Proposal Appendices

1. End of Mission

After the three years of service for the GAEA satellites, they will stop performing burns/maneuvers to stay in their circular orbit at 475 km. After this three year mission timeline, GAEA will deorbit with a natural atmospheric reentry burnup to meet NASA's end of mission requirement. Due to slight atmospheric drag and orbit decay discussed in **Orbit Decay**, the satellites will burn up in the atmosphere of Earth. Considering the small volume and mass of the satellites as well as the very low melting point of aluminum there is no risk for the GAEA satellites to damage anything on Earth's surface. In accordance with the **Orbit Decay** section, it will take 348 days for the satellites to achieve reentry in Earth's atmosphere starting at 475 km altitude. Due to the low altitude of the GAEA orbit, there is no additional delta-V requirement for orbits of this altitude.

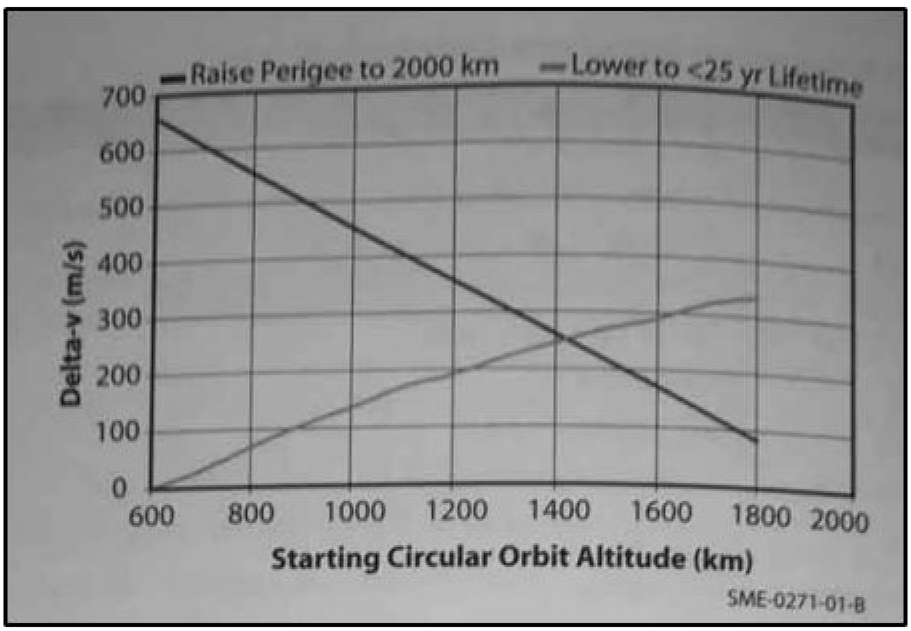


Figure I.1: Delta-V requirements to initiate atmospheric burnup and corresponding orbit altitude

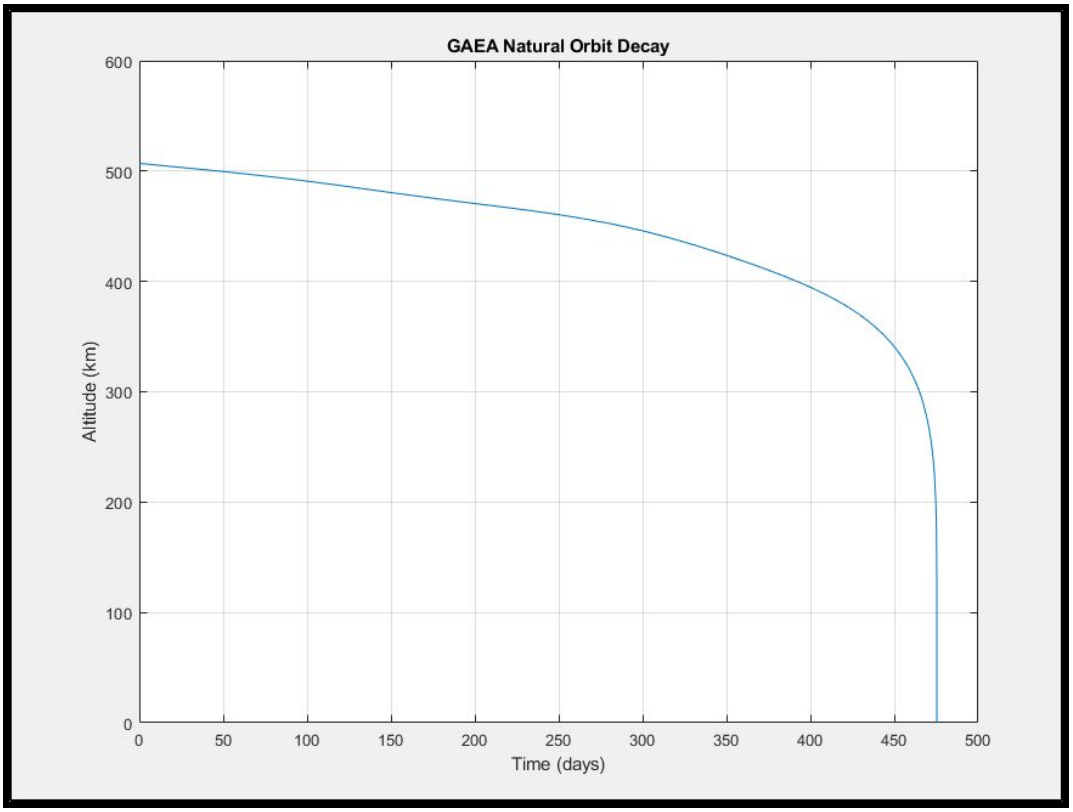


Figure I.2: Orbit decay visualized in a plot of altitude of a single GAEA satellite over time due to atmospheric drag and natural decay

As seen in **Figure I.1**, 600km is below the lowest altitude displayed. It can be extrapolated that 500 km will not need any delta-V to initiate reentry. Additionally, **Figure I.2** demonstrates the timeline for re-entry once delta-V burns are no longer being implemented. Since, we will be at 475 km altitude at the end of mission, so we expect to deorbit in 348 days. Due to all of the discussed factors, an uncontrolled reentry burnup of the GAEA satellites is the optimal end of mission operation.

2. Master Equipment List

Reaction Wheels
Name: 4RW0
Manufacturer: NanoAvionics

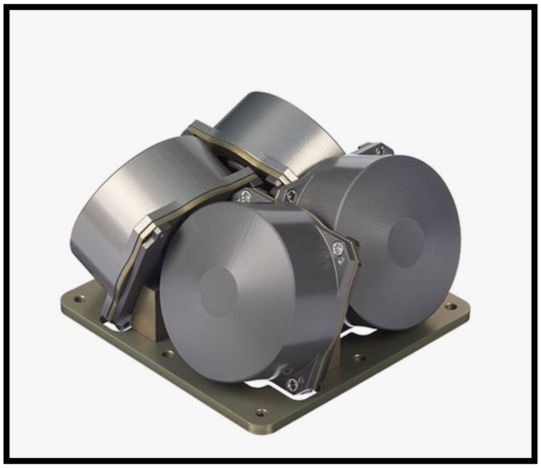


Figure I.3: Reaction Wheel

Parameter	Reaction Wheels
Maximum Torque Around X/Y Axis	5.9 mNm
Max Momentum Storage Around X/Y Axis	37 mNms
Power Consumption (Steady state, 1000 RPM each)	600 mW
Weight	665 g
Size	82.3 x 82.3 x 51.2 mm

Figure I.4: Reaction Wheel Specifications

More Specifications: https://satsearch.co/products/nanoavionics-reaction-wheels

Earth Sensor
Name: CUBEIR Horizon Sensor
Manufacturer: CubeSpace

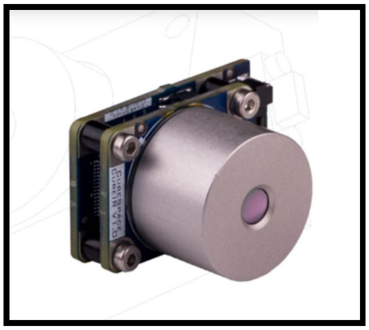


Figure I.5: Earth Sensor

Specifications: https://satsearch.co/products/cubespace-cubeir

Magnetometer
Name: Magnetometer
Manufacturer: O.C.E. Technology

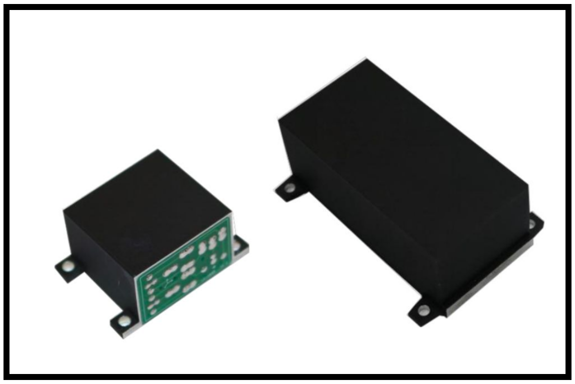


Figure I.6: Magnetometer
Specifications: https://satsearch.co/products/oce-technology-magnetometer

IMU (Includes Accelerometer/Gyroscope)
Name: MEMS IMU/Gyro
Manufacturer: Tamagawa Seiki Co.

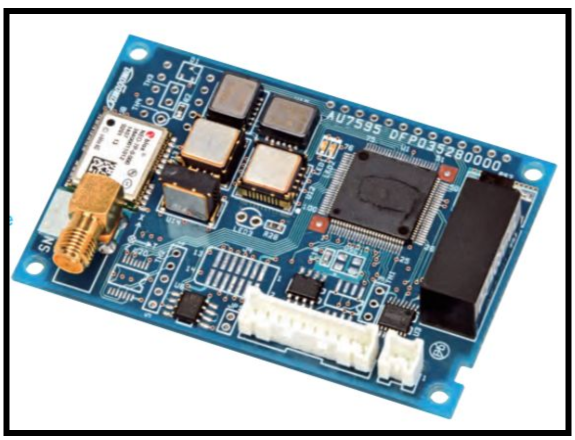


Figure I.7: Accelerometer
Specifications: https://satsearch.co/products/tamagawa-seiki-mems-imu-gyro

Temperature Sensor
Name: Si435 Silicon Diode Temperature Sensor
Manufacturer: Scientific Instruments

Specifications:
https://satsearch.co/products/scientific-instruments-si435-silicon-diode-temperature-sensor

Parameter	X Band Antenna
Frequency	2.0-2.11 GHz (receive) 2.2-2.3 GHz (transmit)
Peak Gain	7 dBi
Half Power Beamwidth	74 deg
Power	4 W
Polarization	RHCP
Size	2.2 g
Weight	24X24X5 mm
Name	X- Band Patch Antenna
Manufacturer	Endurosat

Figure I.8: X-band Antenna Specifications

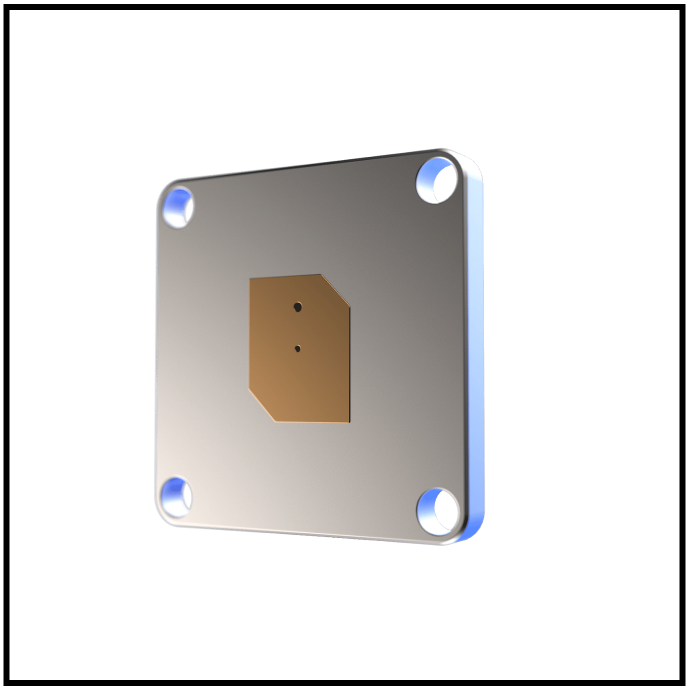


Figure I.9: X-band Antenna

Parameter	X-Band Transmitter
Frequency	8.025-8.4 GHz
RF Power	27-33 dBm
Data Rate	150 Mbits
Power	1W at 32 dBm
Modulation	QPSK, 8-PSK, 16-PSK,32-APSK
Storage	32 GB
Mass	275
Manufacturer	Endurosat

Figure I.10: X-band Transmitter Specifications

Figure I.11: X-band Transmitter

Parameter	S-Band Antenna
Frequency	2025-2110 MHz and 2200-2290 Mhz
Peak Gain	5 dBi
Half Power Beamwidth	70 deg
Power	4 W
Polarization	LHCP/RHCP
Weight	250 g
Name	S
Manufacturer	Endurosat

Figure I.12: S-Band Antenna Specifications

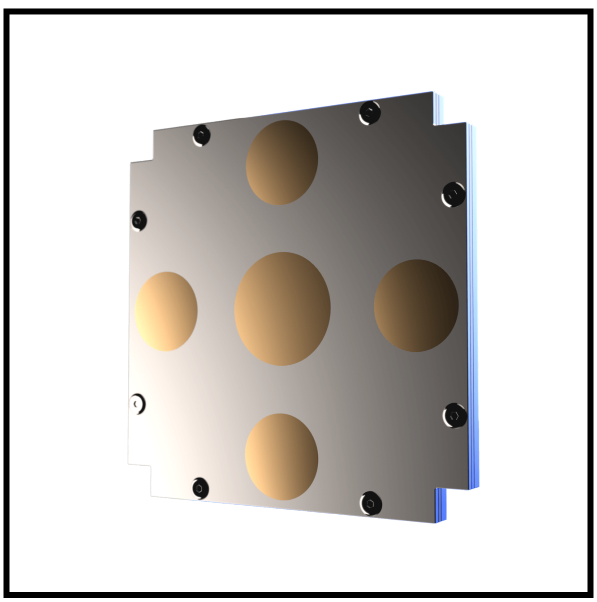


Figure I.13: S-Band Antenna

Parameter	S-Band Receiver
Frequency	2025-2110 MHz
Noise Figure	2.1 dB
Data Rate	4 Mbits
Power	2.5 W
Modulation	BPSK, QPSK
Storage	32 GB
Mass	180
Manufacturer	Endurosat

Figure I.14: S band Receiver Specifications

Figure I.15: S-Band Receiver

Parameter	S-Band Transmitter
Frequency	2200-2290 or 2400-2450 MHz
Noise Figure	2.1 dB
Data Rate	20 Mbps
Power	.5-2 W
Modulation	BPSK, QPSK, 8-SPK, 16-PSK
Storage	32 GB
Mass	250 g
Manufacturer	Endurosat

Figure I.16: S-Band Transmitter Specifications

Figure I.17: S-Band Transmitter

Parameter	Main Computer
Power	< 1W
Clock Speed	ARM Dual Core @ 766 Mhz
Error correction	Triple Mode Redundancy, EDAC protected RAM, Upset and Multi Current Monitoring, Watchdog Timer, FPGA but-stream scrubbing
Radiation Hardening	TID> 25 krad
Mass	32g

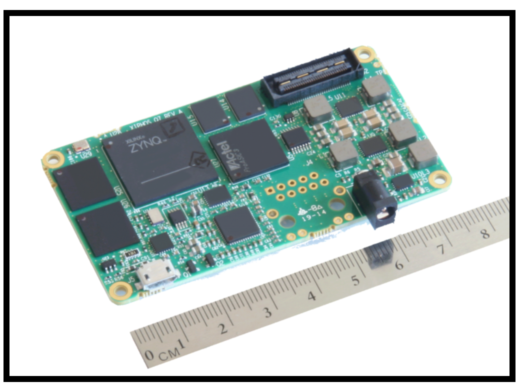


Figure I.18: Xiphos Q7S Onboard Computer

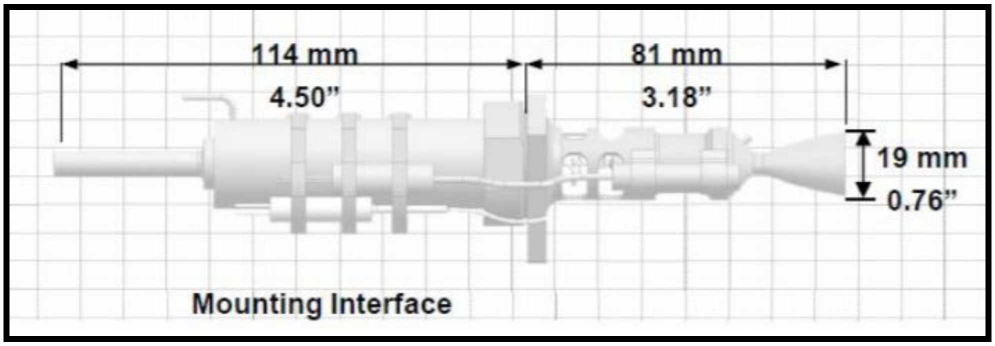
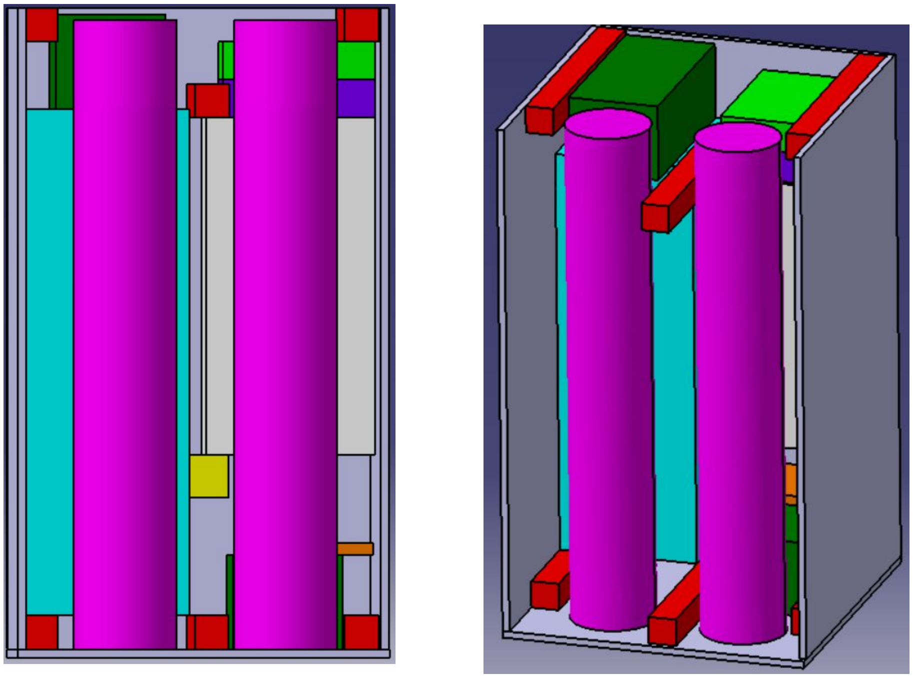


Figure I.19: MR-111G 4N Rocket Engine Assembly

Parameter	Engine
Mass	0.37 kg
Propellant	Hydrazine

3. Appendix: Tables and Figures



Component	Color
Computer	
Xband Reciever	
Sband Receiver	
P/I Band Case	
P/I Receiver	
Propellant Tanks	
Battery	
Momentum Wheel	
Flow Control Devices	

Figure I.20: Internal Structure and Layout

Propellant Delta V Budget	**Delta V**	**Units**
Launch Vehicle Correction	34.58	m/s once
Orbit Control		
Drag Compensation	75.83	m/s per year
East-West Stationkeeping	3	m/s per year
North-South Stationkeeping	35	m/s per year
Spin up and Despin	10	m/s total
Attitude Control		
3-axis Control	2	m/s per year
Momentum Wheel Unloading	2	m/s per year
Total Delta V Required A	**398.08**	**m/s for one satellite**

Table I.21: Delta-V Calculation

Altitude and Orbit Control	ISP (s)	Mass of Propellant (kg)	Mass of one engine (kg)	Mass of Propellant with six engines (kg)
Cold Gas	75	19.67	7.08	62.15
LO2/LH2	451	2.58	138.25	832.08
Dual Mode (N204/N2H4)	339	3.48	4.50	30.48
RP1	300	3.97	168.00	1011.97
Solid Motors	300	3.97	4.68	32.07
N2O4/MMH	340	3.47	4.08	27.95
Hybrid (O2 and rubber)	225	5.42	4.87	34.66
Hydrazine	**225**	**5.42**	0.37	7.64

Table I.22: Various Propellant Options and Respective Masses

Onboard Storage		
1 Data Downlink	39690	Mb
Required Max of 2 Missed Downlinks	119070	Mb (equivalent to 3 Downlink Periods)
S band Receiver	32	GB
X band Transmitter	32	GB
S Band Transmitter	32	GB
Main Computer	64	GB
Margin	1160930	

Figure I.23: Onboard Data Storage Storage

Data generation rate		
	35	#specular points
	25200	Time of Collection (s)
	15	Instrument Data rate (kbits/s/specular-point)
	22.5	Instrument Data Rate * 1.5
Data Generated:	19845000	kb
	19845	Mb
2 days of data	39690	Mb
Average Downlink Time	351	Seconds
X band McMurdo @120 Mbps	42120	Mb
Margin	2430	Mb

Figure I.24: Downlink Data Budget

X Band Downlink - Mission Data	**Equation**	**Value**	**Notes:**
Parameter	Sym	GAEA	
RF Frequency (GHz)	f_c	8.1	X band Down
Distance to Ground Station (km)	d	1359	Elevation Angle = 16 deg
Information Bit Rate (Mbps)	R	150	Before error correction
Phase Modulation Index (rad pk)	β	QPSK	
Transmit Power (dBm)	P_t	36	4 Watt Max Transmit Power
Transmit Passive loss (dB)	L_p	-4.5	Typical Value
Transmit Antenna Gain (dBic)	G_t	6	6 dBic Max Antenna Gain
EIRP (dBm)	$P_t G_t L_p$	37.5	
Path Loss (dB)	$\left(\frac{4\pi d}{\lambda}\right)^2$	-173.28409	
Atmospheric Loss (dB)		-1	Typical loss for X band (no rain)
Ground Antenna Gain (dBic)	G_r	56	NASA MG1 McMurdo
Total Received Power (dBm)	P_r	-80.7840895	
Data-to-total-Power (dB)	$sin^2(\beta)$	0	
system Noise Density		-174.6	
Receiver G/T (dB-K)		32	NASA MG1 McMurdo
Carrier to Noise	$EIRP + \frac{G}{T} - B_N - L_{comb}$	98.0549979	
Received Eb/No (dB-Hz)	$\frac{P_r sin^2(\beta)}{N_o R}$	12.0549979	
Required Eb/No (dB)		4	
Receiver System Loss (dB)		-2	
Link Margin (dB)		6.0549979	Required > 6
Code Rate	ρ	0.8	(4/5)
Effective Usable Data Rate (Mbps)		120	

Figure I.25: Downlink Link Budget

S Band Uplink		
Chars/line	200	
Time step	0.25	Seconds
Time Between Passes	172800	Seconds (two days)
Number of lines	691200	
Chars	138240000	
Bits after encoding (6bits/char)	829440000	bits
	829.44	Mbits
Uplink Time	150	Seconds
Uplink Data Rate	6	Mbps
Margin	70.56	Mb

Figure I.26: Uplink Data Budget

S Band Command Uplink	**Equation**	**Value**	**Notes:**
Parameter	Sym	GAEA	Notes:
RF Frequency (GHz)	f_c	2	S band Up
Distance to Space (km)	d	1359	Elevation Angle = 16 deg
Information Bit Rate (Mbps)	R	6	Before error correction
Phase Modulation Index (rad pk)	β	QPSK	
Transmit Power (dBm)	P_t	44	Nasa MG1 Mcmurdo very Conservative estimate
Transmit Passive loss (dB)	L_p	-4.5	Typical Value
Transmit Antenna Gain (dBic)	G_t	53	44 dBic Max Antenna Gain
EIRP (dBm)	$P_t G_t L_p$	92.5	Matches MG1 value of 63 dBW
Path Loss (dB)	$\left(\frac{4\pi d}{\lambda}\right)^2$	-161.13499	
Atmospheric Loss (dB)		-1	Typical loss for S band (no rain)
Space Antenna Gain (dBic)	G_r	5	S Band Antenna Selected
Total Received Power (dBm)	P_r	-64.634989	
Data-to-total-Power (dB)	$sin^2(\beta)$	0	
system Noise Density		-174.6	
Received Eb/No (dB-Hz)	$\frac{P_r sin^2(\beta)}{N_o R}$	42.1834984	
Required Eb/No (dB)		11	
Receiver System Loss (dB)		1	
Link Margin (dB)		30.1834984	Requirment > 10 dB

Figure I.27 S Band Uplink Link Budget

S Band Downlink - Backup Comms	Equation	Value	Notes:
Parameter	Sym	GAEA	Notes:
RF Frequency (GHz)	f_c	2.1	S band down
Distance to Space (km)	d	1359	Elevation Angle = 16 deg
Information Bit Rate (Mbps)	R	2	Before error correction
Phase Modulation Index (rad pk)	β	QPSK	
Transmit Power (dBm)	P_t	36	4 Watt Max Transmit Power
Transmit Passive loss (dB)	L_p	-4.5	Typical Value
Transmit Antenna Gain (dBic)	G_t	5	5 dBic Max Antenna Gain
EIRP (dBm)	$P_t G_t L_p$	36.5	
Path Loss (dB)	$\left(\frac{4\pi d}{\lambda}\right)^2$	-161.558775	
Atmospheric Loss (dB)		-1	Typical loss for S band (no rain)
Ground Antenna Gain (dBic)	G_r	45	NASA MG1 McMurdo
Total Received Power (dBm)	P_r	-81.058775	
Data-to-total-Power (dB)	$sin^2(\beta)$	0	
system Noise Density		-174.6	
Received Eb/No (dB-Hz)	$\frac{P_r sin^2(\beta)}{N_o R}$	30.53092501	
Required Eb/No (dB)		11	
Receiver System Loss (dB)		1	
Link Margin (dB)		18.53092501	

Figure I.28: S Band Emergency Downlink Link Budget

Characteristic	Value
Frequency	2025 – 2120 MHz
Maximum EIRP	≥ 63 dBW
Polarization	RHC or LHC
Antenna Beamwidth	1.05 deg
Antenna Gain	44 dBi
Command Modulation	FSK no bit blanking FSK+half-bit blanking. Duty cycle: 50/50 BPSK, BPSK+AM,PCM/PSK/PM/FM/QPSK Direct PCM/PM
Modulation Index	FM: 1 kHz – 5.0 MHz deviation PM 0.01 – 2.50 Radians
Carrier Data Rate	1 Kbps – 1 Mbps
Subcarrier Frequency	5 KHz – 2 MHz
Subcarrier Modulation	FSK and BPSK
Subcarrier Data Rate	100 bps – 250 Kbps
Data Format	NRZ-L, M, or S; or Biφ-L, M, or S

Figure I.29: MG1 S band Command Characteristics

Characteristic	Value
Frequency	7700 – 8500 MHz
G/T include radome	32.0 dB/K (clear sky & 10° elevation angle)
Polarization	RHC or LHC
Antenna Beamwidth	0.26 deg
Antenna Gain	56 dBi
Modulation Type	BPSK, QPSK, OQPSK, UQPSK, 8PSK, GMSK, (16,32,64 APSK)
Demodulator Data Rate – Mono Ch	500 Kbps – 240 Mbps (BPSK) per ch 1 Mbps – 350 Mbps (QPSK, OQPSK) per ch 1 Mbps – 2X240 Mbps (UQPSK) per ch 1 Mbps – 350 Mbps (8PSK) per ch
Data Format	NRZ-L, M, S, Biϕ-L, M, S, DNRZ
Decoding	Derandomization, Viterbi and/or Reed-Solomon (Para 1.3 ref s.)

Figure I.30: MG1 X Band Telemetry Characteristics

Flight Software

Computer Software Component	**Source Lines of Code**	
Executive	1000	
Communication		
Command Processing	1000	
Telemetry Processing	1000	
Attitude/Orbit Sensor Processing		
Rate Gyro	500	
Earth Sensor	1200	
Magnetometer	200	
GPS (output with position/velocity)	800	
Accelerometer	500	
Attitude Determination and Control		
Integration	2000	
Kalman Filter	8000	
Error Determination	800	
Orbit Propagation (linear models)	3000	
Attitude Actuator Processing		
Thruster Control	1200	
Momentum Wheel	3000	
Fault Detection (Monitor, Identification)	6000	
Utilities (Mathematics and Time)	8500	
Power Management	1200	
Thermal Control	800	Margin (20%)
Total	40700	**48840**

Figure I.31 Source Lines Of Code

Gantt Chart

Figure I.32: Gantt Chart

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