Preliminary Design Review
AIDENNE Mission
Team 29

Table of Contents

1. Introduction and Summary
    1.1. Team Introduction
        1.1.1. Shiza Ahmed
- William Rainey Harper College, Illinois, USA
- Studying Computer Engineering
- UIUC Engineering Pathways Program
- 

        1.1.2. Andrea Chavez
- Texas A&M University, Texas, USA
- Studying Mechanical Engineering

        1.1.4. Caleb Emesiani
- University of Oklahoma, Oklahoma, USA
- Studying Electrical Engineering
- 

1.1.5. Yuri Labuca

- William Rainey Harper College, Illinois, USA
- Mechanical Engineering, Intending on Mastering Aerospace Engineering
- NCAS Mission 1 and 2

1.1.6. Vincent Ma

- University of Illinois at Urbana-Champaign, Illinois, US
- Studying Aerospace Engineering
- Illinois Space Society's NASA Micro-g NeXT Competition

1.1.7. Baylee Peak

- Auburn University, Alabama, USA
- Studying Aviation and Business
- Private Pilot

1.1.8. Bailey Sarna

- University of California - Santa Cruz, California, USA
- Studying Intensive Psychology with a focus on cognition and human factors.
- Current process improvement intern at Johnson Space Center and former SBIR/STTR service design intern at Ames Research Center.

1.1.9. Sumesh Surendran

- Houston Community College, Texas, USA
- Associate degree in Applied Science
- Studying Artificial Intelligence
- Project Manager for NASA MINDS Artemis Mission technical advance.

1.2. Mission Overview

1.2.1. Mission Statement

The AIDENNE (Asteroid Investigation to Defend from Near-Earths) mission seeks to survey the characteristics of a near-Earth object (NEO) named 3361 Orpheus, a potentially hazardous Apollo-class asteroid whose orbit crosses Earth's orbit. On this mission, we will use SAMURAI (Spacecraft for Asteroid Measurement and Understanding Risks Against Inhabitants), an orbiter which will complete a 4.6-million-mile journey before starting its orbital rendezvous around our chosen asteroid, where it will gather data on the asteroid's spectroscopic properties, shape, and gravitational field to determine its composition, mass, and density. The AIDENNE mission will provide valuable insight for further categorization of NEOs, which are currently categorized largely by size and distance from Earth. Understanding the characteristics of NEOs has many important implications for the evolution of planetary defense technology by allowing scientists to develop the appropriate strategies for future kinetic impact or deflection attempts.

1.2.2. Mission Requirements

- Conduct science on the surface of NEO or while orbiting an asteroid with an estimated diameter of 300m
- Begin the mission orbiting the asteroid 1.25km away from its center, at a velocity of 4.56cm/s, for 33.8 hrs., at 0.05AU from Earth
- Design a system that weighs 65 kg or less
- Design the system within the dimensions of 1m x 1m x 1.25m, considering the Unusable Design Space consisting of the predetermined main propellant system
- Determine the communication scheme necessary to relay collected data back to Earth and the required antenna size, power, gain, and frequencies
- Use no more than the allotted $150 million on the mission concept

- Budget personnel time for the for the development of the mission, taking into consideration a scientist's salary of $80,000/year, engineer's salary of $80,000/year, technician's salary of $60,000/year, and manager's salary of $120,000/year
- Determine the elements of the science payload that will achieve the goal of the mission
- Determine a launch date and mission duration, factoring in personnel travel time to launch site and personnel stay at launch site for five days, arriving two days before launch and leaving two days after launch

Req ID	Requirement	Rationale	Parent Req	Child Req	Verification method	Relevant Subsystem
0.1	Shall determine composition, minerals, and elements of the NEO	In line with science goals	Customer	PAY.0X	Demonstration	ALL
SYS.01	System shall deploy its subsystems, including, but not limited to: payload system, power system, and comms system	In order to begin a successful mission, all subsystems must deploy successfully	Customer	PAY.0X, PWR.0X, CDH.0X TCS.0X, ADC.0X	Test	Payload, Power, Command, Navigation, CDH
CDH.01	Comms. (communcations) system shall relay collected data to Earth	For analysis of NEO data	SYS.01	Communications system	Test	CDH, Payload, Command, Navigation
ADC.01	Altitude shall be equal to the orbit diameter of 300m	Predetermined from past missions (Dawn, Lucy), this is to ensure an optimal orbit for study of NEOs (Near-Earth Objects).	Customer	Navigation systems	Demonstration	ADC
PAY.01	Payload shall be deployed and in full working order upon setting of orbit.	To begin the collection of data from NEOs, the payload should deploy and be tested to be in full working order.	0.1	PAY.02, PAY.03, PAY.04	Test	Payload, Power
STR.01	Structure shall be within the dimensions of 1m x 1m x 1.25m	Rocket system constraint	Rocket system	SYS.0X, all subsystems	Inspection	Structural
STR.02	Structure shall weigh no more than 65kg	Rocket system constraint	Rocket system	SYS.0X, all subsystems	Inspection	Structural
PWR.01	Power system shall have a capacity of 300 watts	To accomodate all instruments, systems on satellite bus, ensuring mission success.	PAY.0X, CDH.01, TCS.01	Power system	Analysis/Test	CDH, Payload, Structural, Power
PWR.02	Power system shall recharge a 13.5 Ah battery series.	For non-sunlight periods, the spacecraft needs reserve power to keep in homeostasis at minimum and to send data back to Mission Control ideally.	PAY.0X, CDH.01, TCS.01	Power system	Test	CDH, Payload, Structural, Power
TCS.01	Thermal system shall keep all other subsystems including itself within optimal operating temperatures.	In order to keep the spacecraft system and all subsystems healthy in the harsh environment of space, the thermal subsystem should regulate internal temperatures within optimal operating range. This will ensure an effective, longlasting mission.	0.2, PAY.0X, CDH.01, PWR.01	Thermal system	Analysis/Test	Thermal, Payload, CDH, Power
PAY.02	Gamma Ray and Neutron Detector shall perform elemental analysis on minerals present on the surface of the NEO.	To achieve Science Objective 1	0.1, Customer	GRAND Instrument	Demonstration	Payload

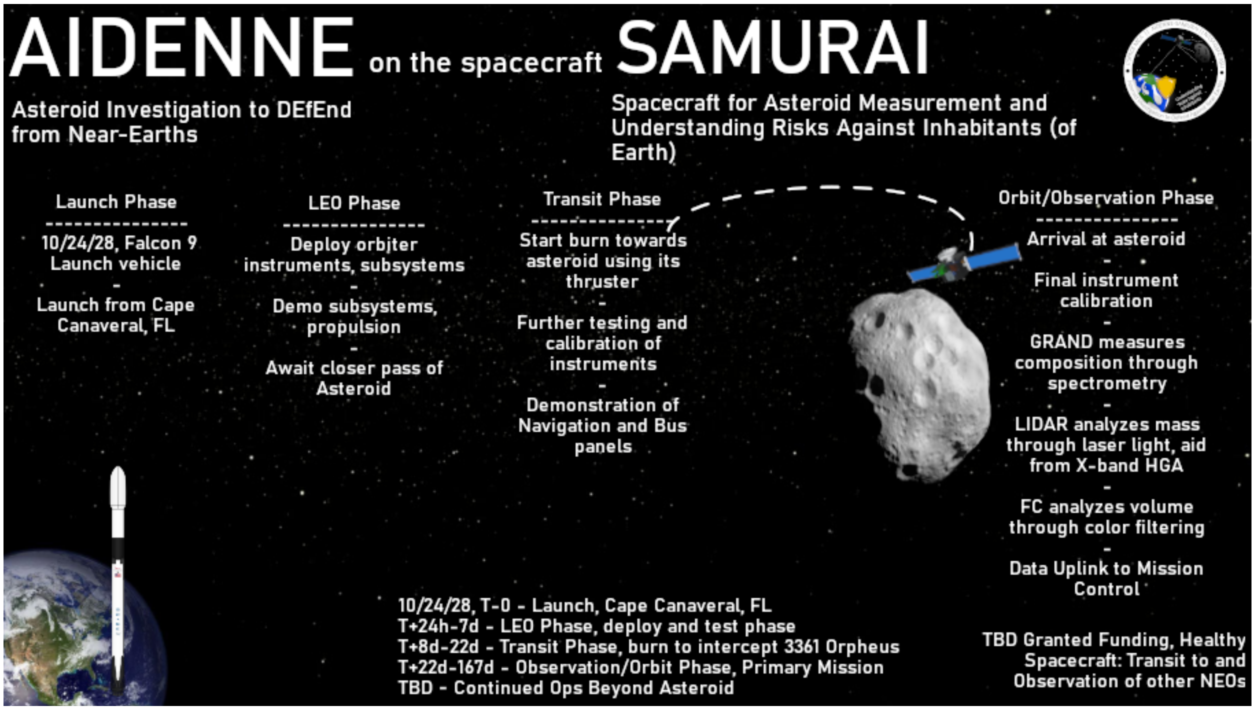
PAY.03	Framing Camera shall image the NEO to help aid with volumetric analysis.	To achieve Science Objective 2, 3	0.1, Customer	Framing Camera Instrument	Demonstration	Payload
PAY.04	Laser Altimeter shall perform volumetric analysis through optical analysis.	To achieve Science Objective 2, 3	0.1, Customer	Laser Altimeter Instrument	Demonstration	Payload
PAY.05	ADC and Comms system shall aid with determining mass.	To achieve Science Objective 3, then 2, the spacecraft telemetry from the ADC subsystem + the Doppler effects on X-band radiowaves can be used to measure the gravitational effect of the asteroid on the spacecraft, helping determine the mass of the asteroid. Then, these measurements can be used to determine density	0.1, Customer	ADC system, Comms system	Demonstration	ADC, CDH
STR.03	2 stage telescopic actuator shall deploy solar arrays	To fully ensure successful deployment of solar arrays.	Power system	2 stage actuator	Test	Mechanical, Power
STR.04	Composite booms shall self-deploy and unroll upon deployment of 2-stage actuator.	For successful deployment of solar arrays.	Power system	Composite booms	Test	Mechanical, Power
STR.05	Y-axis joints that hold the solar arrays shall gimbal towards the Sun in collaboration with coarse sun sensors.	To ensure maximum power is held at all times	Power system	Y-axis joints	Test	Mechanical, Power
STR.06	Composite bus panels shall withstand the space environment	Pioneering in cost-savings also means a tech demo of the composite panels to see if they withstand space.	All	Composite bus panels	Test/Demo	Mechanical, Power
STR.07	Ball-and-socket joint shall gimbal to command from onboard computer.	To ensure highest gain towards Earth or possible usage for geodesic and gravitational analysis.	CDH	Ball-and-socket-joint	Test	Mechanical, CDH
STR.08/ADC.06	3x reaction wheels shall help with keeping spacecraft orientation towards intended orientation on all three coordinate planes.	To ensure a degree of control for a more efficient mission.	Payload, ADC	Reaction wheels	Demonstration	Mechanical, Payload, ADC
PWR.03	Coarse Sun Sensors shall track the Sun.	To ensure the most power generated possible for Solar Arrays.	All systems	Coarse Sun Sensors	Test	ADC, Power
PWR.04	Solar Arrays shall generate the power capacity	To ensure fully working Solar Arrays	All systems	Solar Arrays	Analysis/Demo	Power
PWR.05	Series battery shall have 13.5 Ah capacity	To ensure the most power available during non-sunlight periods	All systems	Series battery	Test	Power
CDH.02	Central computer shall handle all telemetry with some autonomy.	To ensure a harmonious spacecraft ecosystem.	All systems, CDH	Central computer	Test/Analysis	CDH
CDH.03	Both X-band antennas shall communicate their respective gains towards Mission Control	To ensure: a High-gain, high quality data and a Low-gain backup in case of spacecraft communication loss	All systems	X-band antennas	Test/Demo	CDH
TCS.02	MLI shall reduce emissivity and absorptivity to within heat map specifications.	To ensure optimal temperatures for all subsystems	TCS.01	MLI	Test/Analysis	TCS
TCS.03	Thermal sensors shall detect a reduction in temperatures and start the heaters in response.	To ensure a healthy spacecraft during non-sunlight periods	All systems	Thermal sensors, heaters	Test	TCS
TCS.04	Aluminum panel and louvers shall have emissivity and absorptivity values within heat map specifications.	To maintain a homeostatic thermal balance with MLI wrap.	TCS.02, TCS.01	Aluminum radiative panel, passive louvers	Test/Analysis	TCS
ADC.02	Star Sensors shall aid with Navigation Telemetry	To have the best idea of where the spacecraft is located	ADC system	Star sensors	Test/Demo	ADC
ADC.03	IMU shall aid with Navigation Telemetry in terms of inertial data	To have the best idea of how the spacecraft is moving.	ADC system	IMU	Test/Demo	ADC
ADC.04	CSS shall aid with Navigation Telemetry in terms of the Sun's location.	To have the best idea of where the spacecraft is located with regards to the Sun.	ADC system	CSS Pyramids	Test/Demo	ADC
ADC.05	Framing Camera shall aid with picture telemetry whenever necessary in terms of control.	To enhance Mission Control's ability to locate the spacecraft alongside all other ADC systems	ADC system	Framing Camera Instrument	Demonstration	ADC/Payload

*All Mission Requirements

1.2.3. Mission Success Criteria

- Determine elemental makeup of asteroid regolith, spectral type of asteroid (Carbonaceous, Salicaceous, Metallic)
- Measure density through mass divided by volume, using observables from science objective 1 and 3
- Measure gravitational field and dimensional characteristics to calculate mass.

1.2.4. Concept of Operations (ConOps)



The team mission will launch from Kennedy Space Center on October 24, 2028, on the Falcon 9 launch vehicle. This launch date was chosen to account for scientific instrument manufacturing time, which is the longest manufacturing time out of all the components of the spacecraft. After launch, the spacecraft will undergo demonstration in Low-Earth orbit. This will be the LEO Phase, which will take one week while waiting for the asteroid to close its distance to Earth, specifically down to 0.05AU at most. A transfer of orbits will then commence after the asteroid closes its distance T+8 days from Earth. Subsystems will stay deployed. Upon arrival at the asteroid, the spacecraft will adjust its orbit to fit within mission constraints and the instruments will test and calibrate for the final time before starting the observation/science phase of the mission. The orbiter will observe and measure the asteroid over a period of 7 months at minimum to gather as much information about it to send back to Mission Control. After primary mission completion, the health of the spacecraft will be reassessed and any path forward to future asteroid-to-asteroid transits will be brainstormed to have the spacecraft keep transmitting scientific data about asteroids for Earth's planetary defense.

1.2.5. Major Milestones Schedule

Phase A	In Phase A (Concept Development), the team will develop a concept for a self-sufficient spacecraft. The journey of this spacecraft will be seven years long. The mission of this spacecraft will be to investigate and assess the composition, density, and mass of targeted NEO. Research will be conducted in this phase to determine which technology is necessary to achieve mission success criteria. The team will also assess generalized risks on how likely they are to occur, as well as

	any consequences. The team will also begin working on the Mission Concepts and Requirements Review.
Phase B	In Phase B (Preliminary Design and Technology Completion), the team will begin to develop a prototype based on the mission requirements and success criteria. This phase includes drafting a potential design of the spacecraft, as well as assessing the budget and roles of the team. The design should be comprehensive, though it can be altered as needed. Additionally, a formal schedule for the mission will be developed, as well as regular check-ins to ensure the team is on the correct path.
Phase C Phase D	In Phase C (Final Design and Fabrication) the team will begin to build the systems and structures needed to operate the spacecraft. Phase D (System Assembly, Integration, and Testing) will prepare the spacecraft for deployment by going through numerous tests to ensure it is durable and ready for launch. The experiments performed in this phase should be determined and planned accordingly during Phase A (Preliminary Design and Technology Completion).
Phase E	In Phase E (Launch, Operations, and Sustainment): The spacecraft will be deployed and sent towards the NEO. Once deployed, the spacecraft will begin experiments to assess the properties needed to satisfy the mission criteria. After completing the experiments and collecting the necessary data, the spacecraft will send the information back to the team on Earth for consulting. This process will continue as needed until the team is satisfied with the results. The products of this phase will be deemed as the results of the mission.

1.3. Vehicle Design Summary

The spacecraft will run on a cubed-shaped bus 0.5m in length, 0.5m in width, and 0.4m in height built in-house. This was ascertained due to the unique payload weight constraint from the customer, which is 65kg, lower than any COTS spacecraft bus that matches the desired extent of SAMURAI. The cube shape of the bus was decided upon because of its simplicity as well as good surface area for instrument mounting. A tally of the resources is provided below. The spacecraft is a mix of bus-oriented design and instrument-oriented design, with enough room for all three instruments lined out above. The spacecraft's systems shall keep the spacecraft healthy for it to do its mission for a design life of 7 months. In total, the spacecraft has dimensions of 794mm x 854.3mm x 652.4mm, and a mass of 38kg, both within the design constraints.

- *GrAND, 180 x 240 x 60 mm* - 5.4 kg, 9W
- *Framing Camera, 340 x 160 x 240 mm* - 4 kg, 17W
- *LiDAR, 200 x 100 x 100 mm* - 4.0 kg, 14.3W Catalog (data.gov)
- *2x Redwire CSS* - 260 g
- *Rocket Lab ST-16RT2 Star Tracker, x2* - 470 g, 2W each
- *Rocket Lab RW0.06 Reaction Wheel, x4* - 904g, 23.4W at 0.12 Nms
- *Redwire ROSA, 110 x 25 cm, x2* - ~1.499 kg each, from 545mg/cm^2 (see panel trade study)
- *Solar Array 2-stage telescopic actuator, (70 x 61 x 20mm first stage, 130 x 41 x 11mm second stage, pi x 15^2 x 10mm y-axis joint, 350 x 10 x ~50 mm holder), x2*- ~1.753 kg, Al material (see material trade study), 7W max guesstimate each for deploy
- *Parabolic Reflector (450mm diameter by 105mm height, 7mm thickness)* - ~3.146 kg, Al material
- *Spacecom HG X-band Antenna* - 20g, 1W
- *Giken QHC-8G04 X-band LGA* - 160g, 20W
- *Reflector + Antenna support, mounted on ball-and-socket-joint (~2539 cm^3)* - ~7kg, Al material, 8W guesstimate
- *4x (0.4 x 0.5 x 0.009525 m) Rockwest Composites CF + Aramid Honeycomb Core Panels* - 1.71kg
- *1x (0.5 x 0.5 x 0.009525 m) Rockwest Composites CF + Aramid Honeycomb Core Panel* - 533g
- *1x (0.5 x 0.5 x 0.009525 m) Rockwest Composites 5052 Aluminum Skin + Aluminum Honeycomb Core Panel* - 1.488kg
- *3x 4s1p VES16 Saft solution, 271.5 x 84.1 x 77 mm* - 2.1kg, 13.5 Amphrs
- *Alen Space TRISKEL OBC-TTC* - 200g, 5.865W maximum
- *1.3m^2 of 20mm MLI* - 1.56kg (https://en.wikipedia.org/wiki/Multi-layer_insulation)

All told: 38kg, 188W, leaving 27kg and 144W for spacewire, additional infrastructure, etc, (INTD DESIGN LIFE 8 MONTHS)
RESOURCES

- Mass available: 65kg
- Power available: 300W from both panels, 13.5Ah from battery pack (see Redwire ROSA technical info)

1.4. Payload and Science Instrumentation Summary

Name	Capabilities
Gamma Ray and Neutron Detector	Spectrometers are designed to measure elemental abundances on surfaces. The instrument uses twenty-one sensors and can measure important atomic constituents of the NEO body’s surface down to a depth of one meter.
Framing Camera	Multispectral CCD imager. Each camera is equipped with an f/7.9 refractive optical system with a focal length of 150mm and can use seven color filters, provided to help study minerals.

Light Detection and Ranging	Sends short pulses of laser light towards an object. The laser light strikes the target and reflects backwards to the detector in the instrument. Provides data on optical properties, composition of gasses in the atmosphere, and other atmospheric measurements.

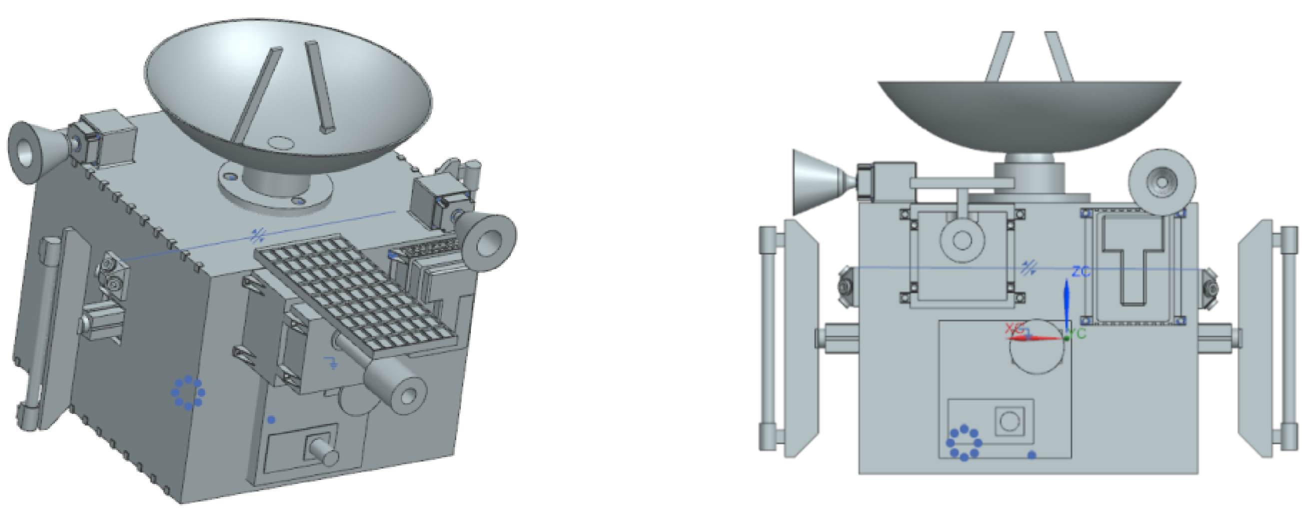
2. Overall Vehicle and System Design

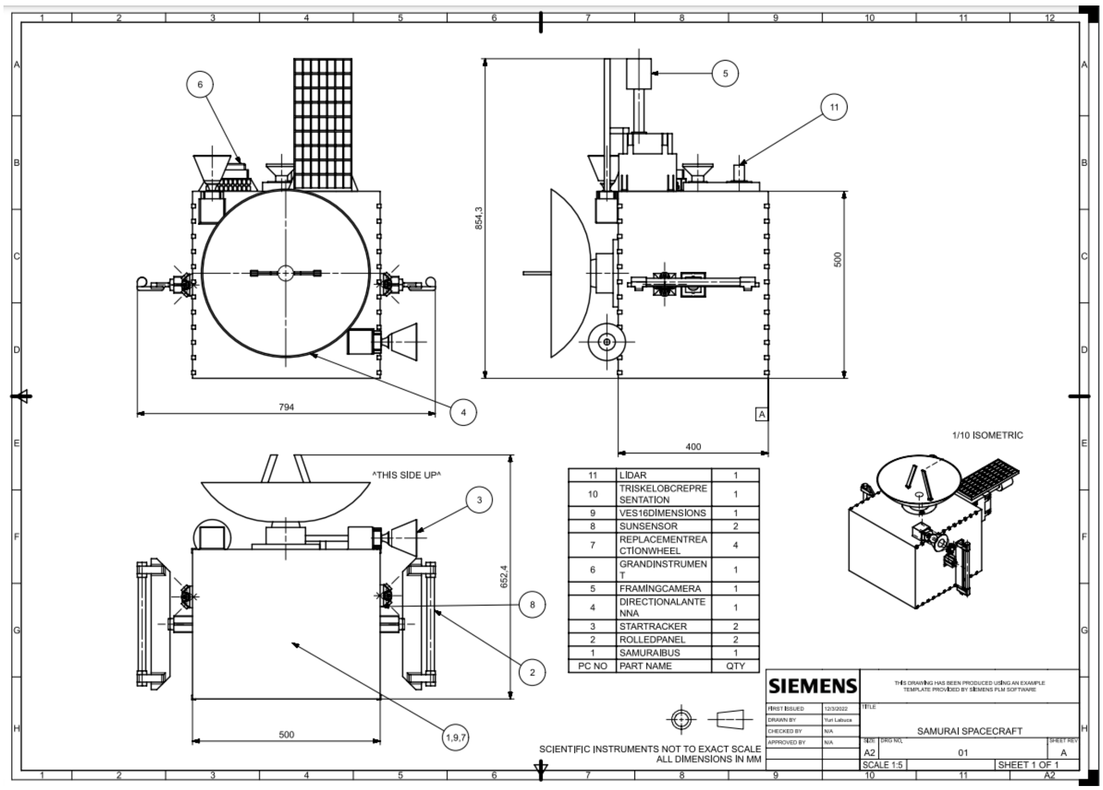
2.1. Selection, Design, and Verification

2.1.1. System Overview

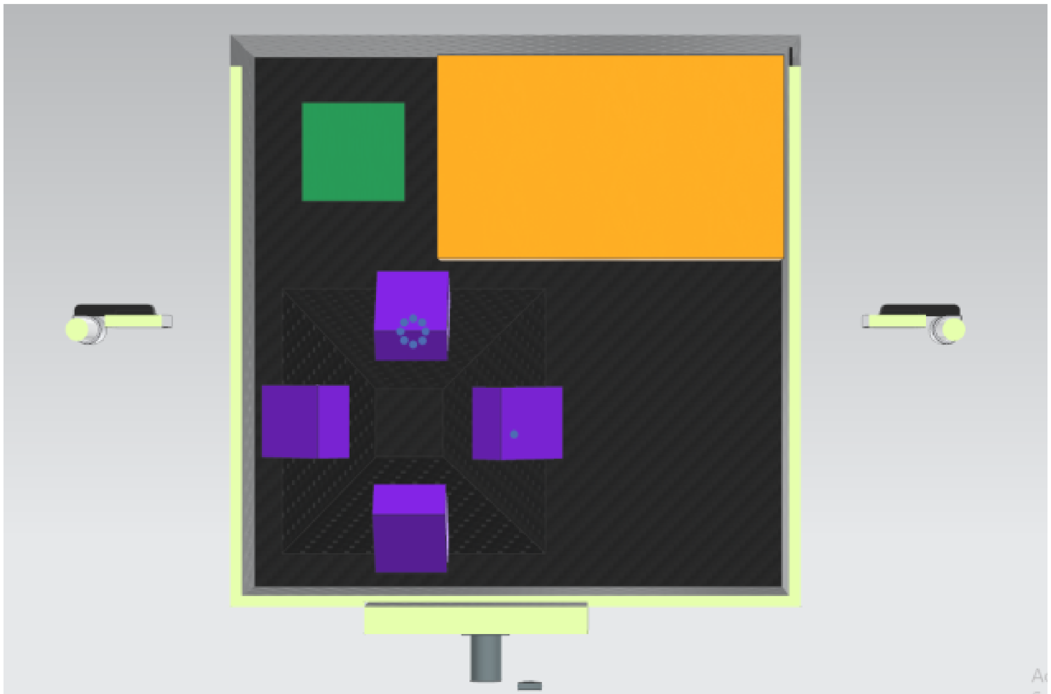
The systems of the spacecraft use a majority mix of COTS components for all of its subsystems. The reason behind using a majority mix of COTS components is to reduce the manufacturing costs of the spacecraft. Due to a small budget, the team does not have the luxury of in-house components. However, there is a great selection of COTS components that have been selected that are verified to operate correctly within their subsystems. The only in-house components, aside from the instruments, are the mechanical components, which are designed specifically for this mission. Requirements for the overall system are listed as SYS.0X in the master requirements listed above.

*CAD Model of System

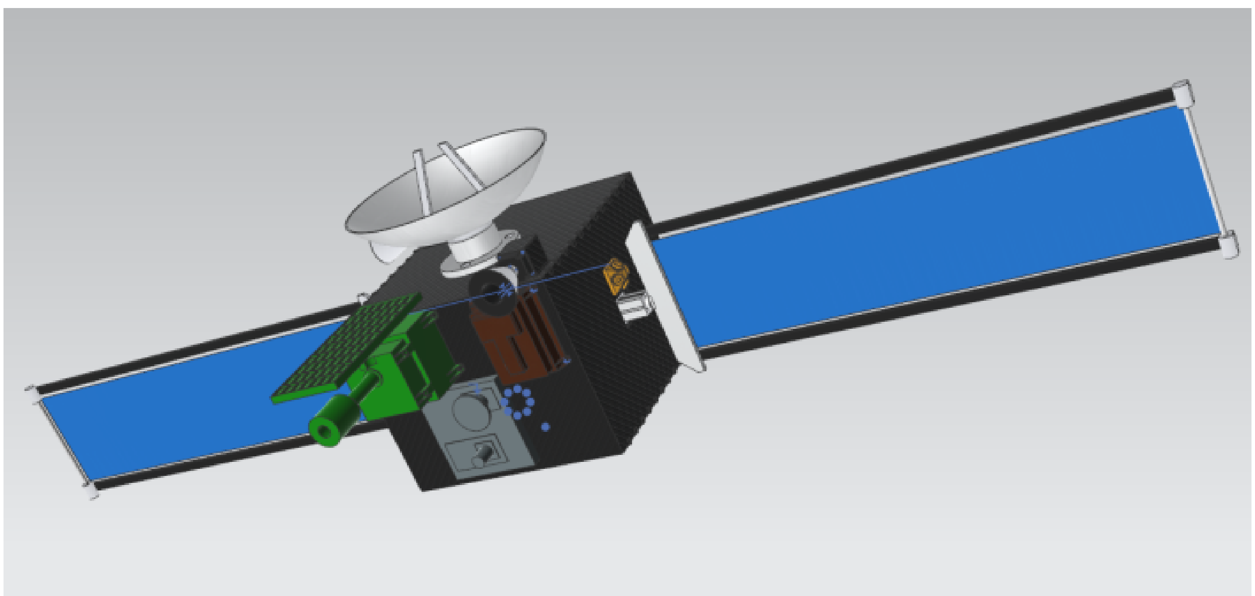




*Engineering Drawings of System



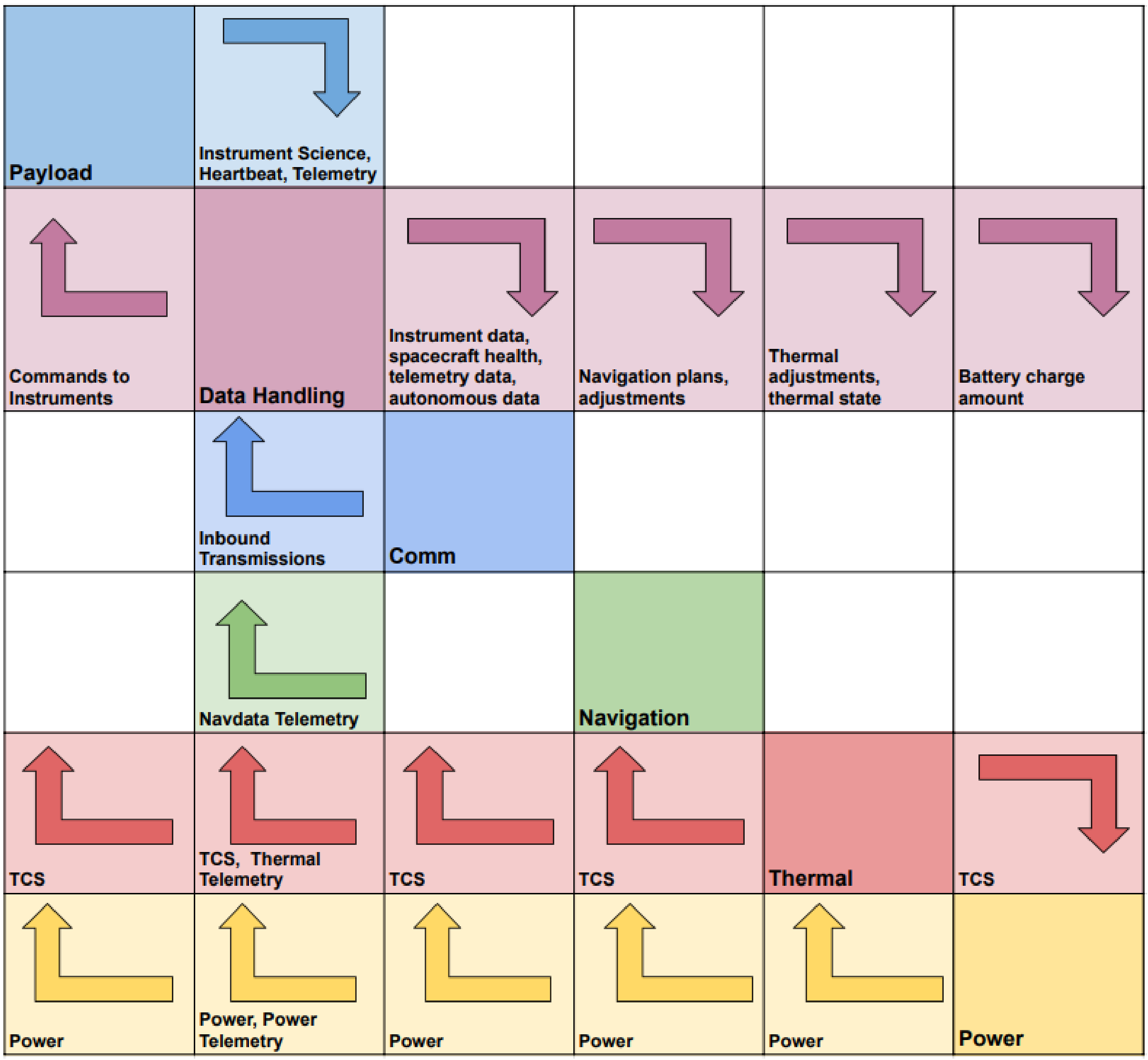
(1)



(2)

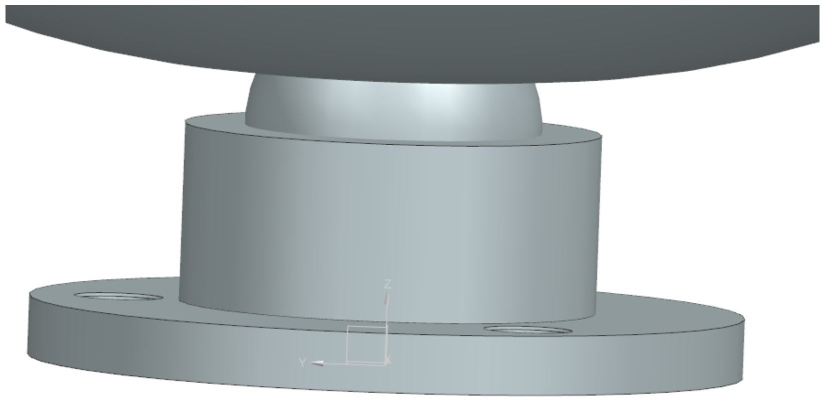
*(1)Inside view, Green is OBC-TTC, Yellow is Battery (Before Downsize), Purple are RWs

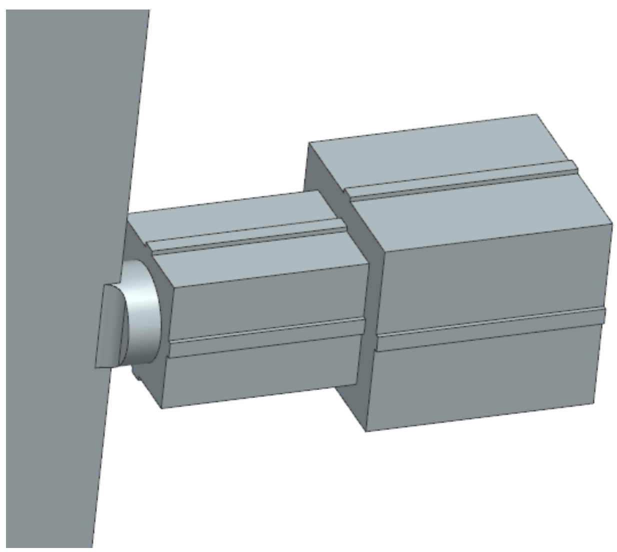
*(2)System deployed, Green is Framing Camera, Brown is GRAND, Gray is LIDAR


Payload
Instrument Science, Heartbeat, Telemetry
Commands to Instruments
Data Handling
Instrument data, spacecraft health, telemetry data, autonomous data
Navigation plans, adjustments
Thermal adjustments, thermal state
Battery charge amount
Inbound Transmissions
Comm
Navdata Telemetry
Navigation
TCS
TCS, Thermal Telemetry
TCS
TCS
Thermal
TCS
Power
Power, Power Telemetry
Power
Power
Power
Power

*N^2 Chart

2.1.2. Mechanical System Overview (BallSocket, Actuator, Rotary Joint Below)

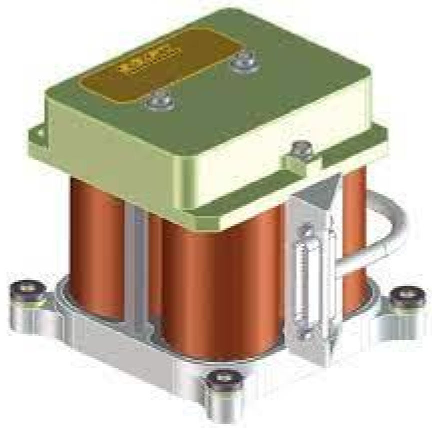




The outside paneling of the spacecraft bus will utilize mostly CFRP Skin + Aramid Honeycomb form of paneling, with 1 panel being Aluminum skin and honeycomb reserved for thermal control, all from Rockwest Composites. This was to ensure a lighter spacecraft due to the nature of the mission constraints. Due to the uncommon choice of the panel, this is a tech demo of the panels. All of these panels will be fastened together by rows of aluminum clamps on the edges and corners of the spacecraft. A trade study showcasing some mechanical properties of the main materials used for the spacecraft bus is shown below. Both solar arrays are mounted on a y-axis rotary joint, which is then mounted on a 2-stage telescopic actuator (dimensions in 2.1.1 Table). This mechanism weighs ~877g each, making for the total seen in the table. The reasoning for choosing this particular mechanism is to keep mechanical simplicity, in-line with the arrays' own mechanical simplicity. It is made out of aluminum, as standard with spacecraft. As there wasn't any time for watt pull analysis, it is assumed based on the mass of the Roll-Out Solar Arrays (ROSAs) that the maximum watt pull will be 7W each. This takes place upon deployment of the system. The ROSAs use strain energy in composite booms to self-deploy, which is different from the standard motor deployment system that requires huge electrical systems. The use of composite booms enables the capabilities of withstanding a dynamic environment, a range of frequencies, and micrometeoroid collisions. A high-gain X-band antenna (HGA) will be mounted on an aluminum parabolic reflector 450mm in diameter, 105mm in height, and 7mm in thickness. This reflector is mounted on a ball-and-socket joint supported by a base, with measurements around 2539 cubic centimeters. The reasoning behind mounting the HGA and reflector on a joint is to help accurately point the HGA towards the Deep Space Network (DSN) for the highest possible gain and to also utilize the system for geodesic and gravitational analysis of the asteroid in the possible failure of the LiDAR instrument. In terms of rotational control, the spacecraft will use 3 Rocket Lab RW0.06 reaction wheels, with 1 for reserve. These wheels are located inside the spacecraft on a pyramidal mount and provide +-20mNm of torque, as claimed by the manufacturer, in the X, Y, and Z planes, which should be sufficient to keep stable rotational control of the spacecraft. Each consumes 23.4W on full load with regenerative braking providing 4.6W back to the system. In all, they also constitute 904g of the spacecraft's mass. It is an active consideration to have an RCS-based rotational control subsystem as a backup in the case where angular momentum desaturation is necessitated. Requirements for this subsystem are marked as STR.0X on the requirements table (1.2.2). With those requirements in place, verification testing in TVAC, EMI, VIBE, and ambient testing will be done on in-house mechanisms, being the actuators, composite panels and the joints. The composite panels are retested in these environments since they are being utilized in a new purpose, even if the manufacturer has done prior testing. These will raise their TRLs accordingly. The reaction wheels have been well-tested and TRL 9, with 41 and counting units in orbit, so verification testing is unnecessary. Validation testing is then done with the same environments upon full assembly of the spacecraft. Branded components are COTS and will be outsourced to other companies. The rest will be built in-house and will take 12-15 months to complete as a rough estimate. The fitting process will be done in validation testing and has been shown through CAD model analysis.

2.1.3. Power System Overview

The spacecraft's power system will utilize Redwire's Roll-Out Solar Arrays, as seen on DART and the ISS. These generate 100-120W per kg, and on rough calculations from the Solar Panel trade study (seen below), they weigh 1.499kg each in a 110 x 25cm configuration. These panels are mounted on the Y-axis rotary joints mentioned in the Mechanical System Overview. The reasoning behind the choice of these arrays is due to their substantial weight reduction, mechanical simplicity, and power efficiency compared to conventional solar panels. As mentioned above, these use mechanical strain energy to deploy, rendering deployment through electrically powered joints redundant, simplifying the mechanical design, and thus reducing strain on the spacecraft's power system and the complexity of the spacecraft as a whole. The joints work in tandem with the Coarse Sun Sensor (CSS) pyramids, also from Redwire. These pyramids are 4 CSS installations, each with a 2pi steradian plus field of view. The CSSs have an accuracy to within +-5 degrees of the Sun's actual position, which is reduced by installed baffles in order to block out stray light. The baffles reduce the accuracy to within +-1 degree of the Sun's actual position, the typical accuracy for these CSSs. Two of these pyramids are mounted just above the telescopic actuators, which allow them to have a full degree view of the sun with respect to the orientation of the arrays in order to consistently communicate to the on-board computer (OBC) where the Sun's position is and keep a consistent track on the sun for the Y-axis joints and, in turn, the solar arrays. It is also an active consideration to mount 1 on the communications deck, which is where the high-gain installation is located, to account for spacecraft rotation with arrays perpendicular to the sun, as the CSSs are unable to have a line-of-sight directly above them. These CSSs do not require any power. All told, the arrays provide 300W of power at their best, more than enough for the 188W pull of the spacecraft. To account for "nighttime" periods, the power system is also equipped with 3 4s1p Saft VES16 battery solutions, in series in length. These constitute 2.1kg of the spacecraft's mass and are 271.5 x 84.1 x 77mm in dimensions. In all, they provide for 13.5 Ah of capacity, which is more than enough for withstanding the nighttime on reduced power. Along with power, the battery solution also provides a heater, thermal sensor, and capability for telemetry connection to the OBC. Requirements for this subsystem are marked PWR.0X. As all of these are well-tested, mission flown components, TRL is at 7-9 and individual component testing (verification) through the aforementioned testing is deemed unnecessary. All of these components are also COTS, with the same assumed rough manufacturing time as above from date of order. However, as with all individual missions, validation testing will still be undertaken with those testing conditions with all of these being fitted and integrated into the spacecraft. Fit has already been analyzed through CAD models. Below are the instruments that consume power and additional info.



*Battery

Instrument	Dimensions	Volume	Mass	Power Consumption
LiDAR	200 mm x 100 mm x 100 mm	2,000,000 cubic millimeters	4kg	14.3W
GRaND	180 mm x 240 mm x 60 mm	2,592,000 cubic millimeters	5.4kg	9W
Framing Camera	340 mm x 160 mm x 240 mm	13,056,000 cubic millimeters	4kg	17W
TRISKEL OBC-TTC	89.30 mm x 12.60 mm x 93.30 mm	104,979.294 cubic millimeters	200g	5.865W
Reaction Wheels (x4)	77 mm x 65 mm x 38 mm	190,190 cubic millimeters	226g each	23.4W each
Combined Mechanical Systems	(see 2.1.1)	(see 2.1.1)	~12kg total	22W combined
Star Trackers (x2)	99 mm diameter x 120 mm	~9,300,000 cubic millimeters	235g each	2W max each
Spacecom HGA	60 mm x 40 mm x 1.8 mm	4,320 cubic millimeters	20g	2W
Gilken LGA	~53 mm x 53 mm x 43 mm	~121,000 cubic millimeters	160g	20W

The total power consumption tallies up to 188W.

**Deployed solar array*

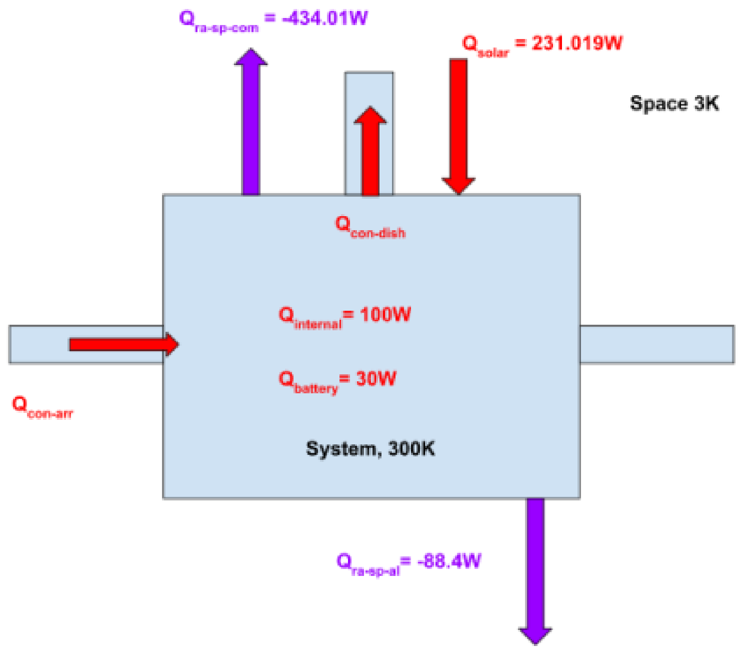
2.1.4. Comms and Data Handling Overview

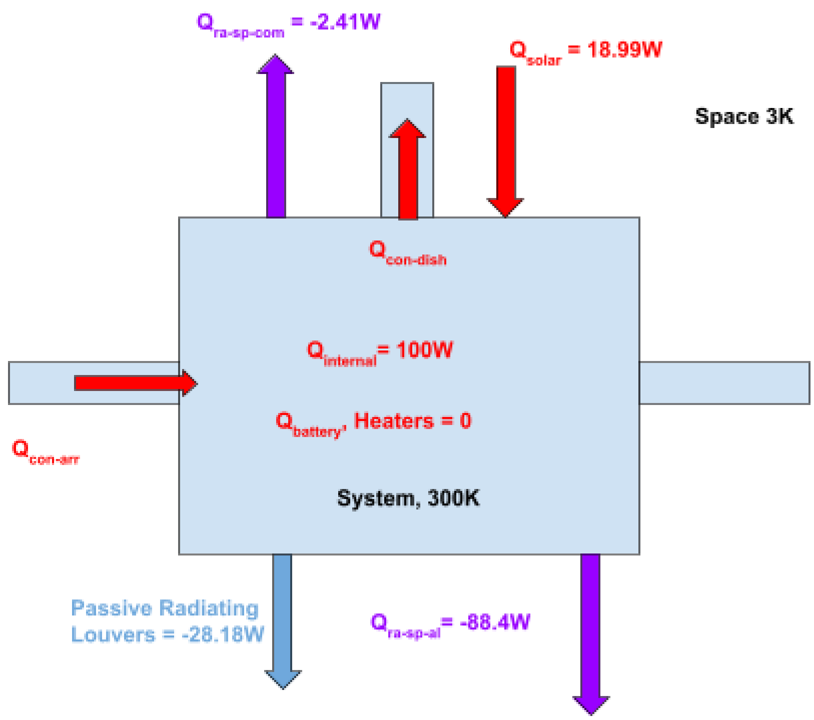
*X-Band Antenna and X-Band Low Gain Antenna (Respectively)

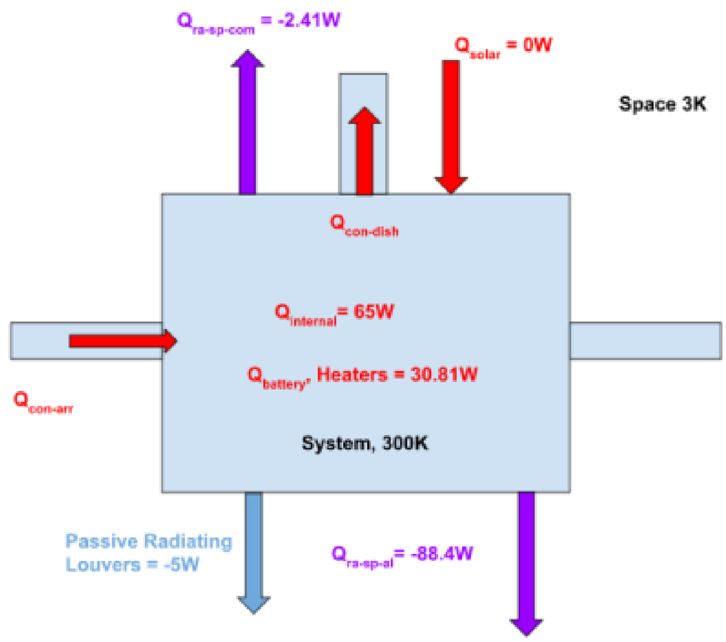
*OBC-TCC

Data handling will be managed by the TRISKEL OBC-TTC, from Alen Space, in which all telemetry and science data will be transmitted to Mission Control through a high-gain X-band antenna. As seen on the block diagram, the OBC-TTC is a consolidated solution, taking part care of a few subsystems, as well as C&DH. The OBC, meaning on-board computer, runs the on-board software for spacecraft management. This architecture also features 1.4 MB of SRAM, on top of the main memory capacity of 256 MB of SDRAM. Contingency memory is also featured in the form of 8MB of MRAM. This OBC also has the ability to take 2 microSD cards for storage, which can be up to 64GB each. Another contingency measure is the Watchdog and Reset Circuit. In the case of a major anomaly, this circuit resets both OBC and TTC (Telemetry, Tracking and Command), which is invaluable for recovery measures. It remains to be seen if this instrument is able to manage power as well, which is the reason behind the inclusion of a power-control unit in the block diagram. This instrument only has space for three payload instruments, hence the limitation to three instruments for the spacecraft. It can interface with CAN, UART, I2C, RS422, and SPI, following the PC104 Space standard, which yields great flexibility of compatibility across all components, as well as modularity in the case of additions. The TTC module has data rates of 1.2-19.2 kbps with a transmission power of 30 dBm. Typical supply current and voltage is 3.3V at 175mA, though maximum stands at 3.45V and 1700mA, hence the 5.865W calculation of the component's watt pull. On top of all that, the OBC-TTC has some autonomous ability upon request by command, which can prove useful in reducing full-time faculty. Two X-band (8 - 12 GHz) antennas, one low-gain and one high gain, were chosen specifically for this mission due to their dual ability to transmit data as well as being used to measure asteroid mass. Giken will provide the low-gain solution, a QHC-8G04 LGA and Spacecom will provide the high-gain solution. The HGA will be mounted on the strut support of the parabolic reflector to yield the highest gain communication towards Earth while the LGA will be mounted on the spacecraft body (not modeled) for spacecraft locating. Requirements for this subsystem are marked CDH.0X in the mission requirements. These are outsourced components, which will take 7-8 months from order. Verification testing will not be undertaken as all of these components are mission-tested. However, validation testing with all components integrated and fitted onto the spacecraft will still be done, with space demonstrations being undertaken where applicable.

2.1.5. Thermal Management System Overview







*Heat Flow Map - Uncorrected Hot, Corrected Hot, Corrected Cold (Respectively)

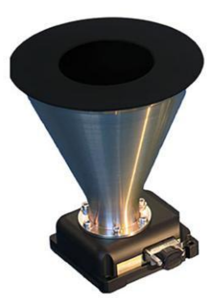

*Louvers and MLI (Respectively)

The thermal management system will be utilizing 20mm MLI fabricated in house. This MLI will be an aluminum sheet and kapton outer layer, and an additional 20 mm below that to fully insulate the spacecraft. This will be wrapped around all of the composite panels of the spacecraft bus. The aluminum panel is left bare due to thermal balance. The reasoning behind choosing MLI as the main form of insulation was due to its absorptivity and emissivity values. The unprotected spacecraft's composite panels have an emissivity and absorptivity rate of 0.9, and their effects on thermal balance within the spacecraft is shown in the unprotected hot case heat map. The Q_{total} of that scenario is -161.39 W leaving the system, highlighting the need for insulation. In addition, an identified risk (may or may not be in the Risk Matrix) of the Aramid honeycomb is its incredible sensitivity to UV light. When exposed, Aramid honeycomb degrades in its mechanical values. The aluminum sheet of the MLI will fully insulate the Aramid honeycomb, reducing the risk of structural failure due to Aramid degradation from UV light. With MLI applied to the five composite panels, this makes up 1.56kg of the total spacecraft mass. As seen in the hot case, MLI reduces both radiation out and radiation in, as seen by the change in the $Q_{ra\text{-}sp\text{-}com}$ and Q_{solar} values to a more manageable number. Because of the reduction in radiation out, Q_{total} amounts to 28.18W, more heat entering the system. The Aluminum panel is left open to radiate, with passive louvers, seen on Rosetta, aiding with its emission. These louvers have spring actuators carefully calibrated to be fully open and fully closed at specific temperatures, which will prove beneficial in the spacecraft cold case. Assuming reduced operation, Q_{total} equals to 30.81W leaving the system. The radiators will emit excess heat, assumed to be a base of 5W alongside the aluminum panel constant, but their closed state is a reduction in their total emission. In the cold case, thermal sensors inside the spacecraft, found on the OBC and the battery assembly, detect the decrease in temperature and signal the battery to start its heaters, as a feature mentioned in the power subsection overview. Additional heaters may be mounted to aid the battery's heaters. Requirements for the subsystem are listed as TCS.0X. Fabrication of the MLI and the thermal louvers should take the same rough estimation of 12-15 months at the most and will be done in house. As standard practice, these will be tested as assembled components as part of the verification phase in their appropriate tests (listed in mechanical subsystem), and once more with it being fully assembled and fitted onto the spacecraft as part of the validation phase.

2.1.6. Guidance, Navigation, and Control (GNC) Overview

*CSS

*Reaction Wheel



*Star Tracker

Guidance and navigation will be handled by the aforementioned OBC-TTC central computer, a pair of Rocket Lab's ST-16RT2 Star Trackers, the aforementioned pair of CSS pyramids, and 4x reaction wheels, and the Framing Camera instrument as a support. The OBC-TTC includes an on-board inertial measurement unit (IMU), which will measure the spacecraft's velocity and altitude. In addition, the star trackers track and measure the speed of any stars in their field-of-view and augments the IMU to get a more accurate measurement of the spacecraft's current position and inertial parameters. These star trackers have the large baffle installed from the manufacturer, which gives them a field-of-view of 7.5 degrees by 10 degrees half-angle, with Sun-Avoidance parameters of 22 degrees sun-to-boresight. Dimensions stand at 99 mm in diameter and 120mm in height, and they stand at 235g each. The CSS pyramids will also aid with guidance in absence of the star trackers' detection of the sun, further augmenting the accuracy and precision of the spacecraft's position and inertial parameters. For transit to the asteroid, if color-filtered images of the spacecraft's heading are deemed necessary, the Framing Camera will supplement guidance in navigation in that regard. Visible light in space is difficult to discern. However, with the instrument's ability to generate images with 7 color filters, it will prove beneficial for supportive spacecraft navigation. As for rotational control of the spacecraft, the 3 reaction wheels will be able to help in that regard within their spec as mentioned in the mechanical subsystem section. Requirements for this subsystem are listed as ADC.0X. All of these components are mission flown and thus require no verification testing in-house. These have been fitted through CAD analysis and will be done like so during the fitting and assembly phase of the spacecraft. As standard, the spacecraft will be tested altogether with this subsystem integrated into it.

2.1.7. Confidence and Maturity of Design

As mentioned before, the team needed to do consistent check-ins to ensure everything is on the right track. This included simulating certain conditions/environments for the instruments, in order to achieve mission success. In order to ensure instruments can operate properly in certain environments, the team will run Thermal Vacuum testing (TVAC), Electromagnetic Interference testing (EMI), Vibration testing (VIBE), and Ambient testing of in-house fabricated components and the spacecraft as a whole to increase TRL levels to 7-9, where possible. TVAC simulates a spacelike environment by removing air and pressure in an enclosed chamber and thermally cycling the spacecraft through hot and cold temperatures to test if the spacecraft's thermals line up with heat map analyses. This is to ensure that the spacecraft stays thermally healthy while conducting its mission. EMI testing then tests the spacecraft's susceptibility to electromagnetic interference from onboard systems or from electrostatic discharge (ESD). A more detailed risk assessment can be found through this form of testing with regards to EMI. Finally, Vibration testing tests the spacecraft's behavior in launch conditions and other vibrational conditions, such as operation of the reaction wheels or the "waving" of the solar arrays, as seen on Artemis I. With all of that being said, test durations will take up to 3 months, and during this phase, all ambient testing of individual components will also be done. The team took the initiative to utilize trade studies to get a better understanding of the power system for the spacecraft. This included trade studies on composite materials and solar panels to ensure our spacecraft is mission-ready. Considering each of the multiple constraints placed on the spacecraft, each system had to be analyzed carefully. In the trade study for composite materials, the team researched thermal expansion/contraction, thermal conductivity/insulation, and density. The team found that material had to be substantially light to account for a considerable weight-to-volume ratio to be able to meet the launch constraint of 65kg. Additionally, the team took heat transfer into account. The team found that the material had to be either thermally conductive or thermally insulative to aid thermal control of the spacecraft. As a result, an unconventional paneling form for the spacecraft bus was chosen, with the majority of the panels being composite panels. It is acknowledged that this form of paneling is practically a tech demo due to its difference from the norm and the application of the panels in strictly space. The team will test these materials thoroughly in assembled simulation in order to get TRL to an acceptable level. In the solar panel trade study, the team researched deployment reliability, power consumption/provided, weight reduction, and the amount of space a panel would take. Roll-Out Solar Arrays (ROSA), Gallium-Arsenide Solar Arrays, and Silicon Solar Arrays were considered. Ultimately, ROSA was chosen.

2.2. Recovery/Redundancy System

With smaller-class missions, redundancies are further reduced to account for dwindling mission resources. Cubesats are high risk, high reward, for example. The team has ensured consolidated ways to possess any backup plans. As such, these backup plans for any disruptive anomalies of the spacecraft are supplemented by the OBC-TTC's ability to self-restart through watchdog and reset circuits in the case of such anomalies. When the spacecraft goes into safe-mode in any case of adverse spacecraft conditions, the LGA will stay open to communications, with Mission Control utilizing its low-gain wideband ability to help locate the spacecraft for further commands. In the case of the IMU or one of the star trackers failing, commands to rebalance GNC will be sent. Such an example would be star tracker mode, where the spacecraft discerns its inertial parameters through only star tracking, as seen on Mars Reconnaissance Orbiter and MAVEN. In the case of a reaction wheel failure or premature wear, the 4th reaction wheel serves as a backup, which will be commissioned to do control on the behalf of the failed reaction wheel. Damage to the parabolic reflector will be mitigated through a refined gimbal to compensate for reduced X-band reflection. With all of that being said, however, that is as much as the team is able to do in terms of redundancies. It is acknowledged that the solar arrays and the payload have little to offer in the form of redundancies, as these constitute some of the more resource-heavy loads of the mission. A potential redundancy plan for the power system would be cycling of watt-heavy instruments, using which instrument is most necessary at the time being instead of all three in tandem. This will be a reduction in science output but will not critically compromise the mission. In the case of LIDAR failure, the X-band HGA will take its place in mass analysis through doppler geodesy. However, these are reduced science alternatives, and there are no alternatives to failures of the Framing Camera and GRAND due to their uniqueness. As such, these instruments will be precisely built and stringently tested in order to ensure their longevity for the mission. This explains the long fabrication time, in excess of 30 months from CDR for the GRAND instrument as an example.

2.3. Payload Integration

The science instruments have been test-fitted on the CAD model seen above. An engineering drawing with labeling of the parts is also featured. These were fitted before budgetary cuts and reductions in science instrumentation footprint. There are no mechanical systems necessary for the pointing of the main scientific instruments towards the asteroid. However, the reaction wheels will be responsible for pointing the instrumentation deck towards the asteroid, of which the interface can be streamlined into the OBC-TTC. The OBC-TTC only has ports for 3 payloads, hence the limitation in the payload.

3. Science Instrumentation

- Light Detection and Ranging (LiDAR)
- Dawn Framing Camera
- Gamma Ray and Neutron Detector (GRaND)

3.1. Selection, Design, and Verification

This mission seeks to determine the composition, density, and mass of a Near Earth Object (NEO). These objectives will be met with the use of Light Detection and Ranging (LiDAR), Dawn Framing Camera, and Gamma Ray and Neutron Detector (GRaND) instruments. The instruments are successfully operated in an actual mission. Software and hardware are integrated to the mission for NEO. TRL-9 satisfies the instruments as it is operated in a similar operational environment.

3.1.1. System Overview

In this mission, LiDAR measures gravitational field and dimension characteristics. This means detecting the structure of NEO while maintaining 1.25Km at a velocity of 4.56cm/s. Additionally, LiDAR calculates the mass of the NEO. The Dawn Framing Camera verifies the composition and shape of the NEO after readings from other observables. Observables include asteroid origin and evolution, as well as volume of the asteroid. By having the mass of the NEO from LiDAR, and the volume from the Framing Camera, the team can calculate the density of the NEO. Lastly, GRaND can detect the element makeup of the asteroid regolith. Observes include the emission spectra of carbon, silicon, and metals.

3.1.2. Instrumentation Evolution

The scientific objective of this mission is to build a spacecraft that can investigate numerous properties of a Near Earth Object (NEO) such as composition, density, and mass under various constraints. To further our understanding of the chosen NEO, the team will need to consider elemental analysis, asteroid origin/evolution, optical imaging, and dimensional characteristics. Initially, the team decided to achieve the following objectives with the use of:

a. Gamma Ray and Neutron Detector (GRaND):
    i. Instrument that uses 21 sensors that can measure important atomic constituents of the NEO body's surface to a depth of one meter.
b. Framing Camera:
    i. Each camera is equipped with an f/7.9 refractive optical system with a focal length of 150mm and can use 7 color filters, provided to help study minerals. Determines the asteroid origin and evolution, as well as shape and volume of asteroid.
c. Light Detection and Ranging (LiDAR):
    i. Sends short pulses of laser light toward an object. The laser light strikes the target and reflects backwards to the detector in the instrument. Provides data on optical properties, composition of gasses in the atmosphere, and other atmospheric measurements.
d. Visible and Infrared Mapping Spectrometer (VIMS)
    i. Multispectral imager covering the spectral range from .30 to 1.05 micrometers. Collects both light that is visible to humans and infrared light of slightly longer wavelengths.

The team decided that four instruments would make the mission surpass the constraints of the spacecraft and alter the route to mission success. The Visible and Infrared Mapping Spectrometer (VIMS) was removed from the systems due to the mass of the instrumentation. The instruments stated alongside this instrument provide similar data, much more efficiently. Additionally, the payload can avoid being overwhelmed by removing this instrument. When reviewing the power constraints, having more than three instruments would not be efficient. The team chose instruments in a way that each instrument could be linked to the other. One instrument could attain an observable, and then the other could verify the data. This can be shown through the connection of GrAND and the Framing Camera, as GrAND determines the composition and elemental makeup of the NEO, and the Framing Camera verifies the composition and shape of the asteroid after readings from other observables while simultaneously obtaining its own observables.

3.1.3. Subsystem Overview

Instrument	Dimensions	Volume	Mass	Power Consumption
LiDAR	200 mm x 100 mm x 100 mm	2,000,000 cubic millimeters	4kg	14.3W
GRaND	180 mm x 240 mm x 60 mm	2,592,000 cubic millimeters	5.4kg	9W
Framing Camera	340 mm x 160 mm x 240 mm	13,056,000 cubic millimeters	4kg	17W

Above shows the designated dimensions, volume, mass, and power consumption of each instrument.

3.1.4. Procurement Plans

In the AIDENNE Mission, the team uses in-house fabrication for the spacecraft bus and all of the science instrumentation. In the mechanical systems, only the actuator, y-axis rotary joint, and the ball and socket joint for the reflector installation. Most of the spacecraft infrastructure uses Commercial-Off-The-Shelf (COTS) instrumentation. Manufacturing of systems components will begin in May 2023 and end in May 2024. This makes our assembly of systems period to be May 2024 to December 2024.

3.1.5. Verification and Validation Plan

As the mission proceeds, the team must do consistent check-ins to ensure everything is on the right track. They shall check that each component is manufactured correctly to avoid future consequences. This includes testing each system individually to ensure functionality, and then proceeding to test all systems in unity to ensure that the spacecraft is mission ready. By simulating certain conditions/environments for the spacecraft, the team will be able to achieve the highest mission success. One of the tests used will include vibrational testing, in which the spacecraft will be put onto a moving platform that is driven by a vibration. Vibrational testing will ensure that the spacecraft is exposed to similar conditions/environment as it will be on the mission, as well as verify that the system will perform as it is expected to, even under harsh conditions. The spacecraft will undergo shock testing as well. Shock tests are used to measure impacts of sudden movements caused by a collision. By doing this, the team can verify the system's performance by simulating conditions that the spacecraft may face on the mission, inevitably increasing the confidence in mission success. After conducting all tests, the team shall verify that the systems are up to mission standards. If standards are not being met, the team will proceed with the steps necessary to ensure that change is being implemented. As a final test, all systems will be deployed and tested in LEO to field test and check in after the strains placed on the spacecraft upon launching to orbit.

3.2. Science Value

3.2.1. Science Payload Objectives

- Use Light Detection and Ranging (LiDAR) to map the size, shape, and surface characteristics of the asteroid, which will be incorporated into a calculation to determine its mass. Additionally, shifts in X-band radio data will be used to measure the asteroid's gravitational field, which will also be used in the mass calculation.
- Use Gamma Ray and Neutron Detector (GRaND) to determine elemental makeup of the asteroid. Allows the categorization of the asteroid by spectral type. Gives a basis for determining future planetary defense strategies based on the asteroid's physical characteristics.
- Use Framing Camera to verify composition and shape of an asteroid after readings from other observables. Additionally, it can contribute to finding asteroid origin/evolution. Density can be determined through the calculation of mass divided by volume.

3.2.2. Science Traceability Matrix

Science Goals	Science Objectives	Science Measurement Requirements		Instrument Performance Requirements		Predicted Instrument Performance	Instrument	Mission Requirements
		Physical Parameters	Observables					
After the success of the Double Asteroid Redirection Test (DART) mission, we will need a better way to prepare our newly developed planetary defenses accordingly to different Near-Earth Object (NEO) classes. Can we send spacecraft to these NEOs to measure certain aspects of such NEOs so that we are better prepared?	Obj. #1: Determine the composition of a Near Earth Object (NEO)	Determine elemental makeup of asteroid regolith, spectral type of asteroid (Carbonaceous, Silicaceous, Metallic)	Emission spectra of Carbon, Silicon, and Metals	Wavelength Range	100nm - 700nm	Carbon - 192.9nm, 247.7 nm; Silicon - 251.5nm, 288.08nm, 390.38nm; Metals - 400 to 700 nm	Gamma Ray and Neutron Detector	Vehicle must maintain a distance of 1.25 km from the center of the asteroid.
				Data Rate	< 0.665 kbps	< 0.665 kbps		
				Spectral Resolution	0.61keV - 1.22 keV	0 meV - 2.5meV		
				Intrinsic Efficiency	0.104 keV @ 1332 keV	0.104 keV @ 1332 keV		Instrument must perform elemental analysis on all minerals present on the surface of the NEO.
	Obj. #2: Determine the density of a Near Earth Object (NEO)	Verify composition and shape of the asteroid after readings from other observables. Calculate via mass and volume	Asteroid origin and evolution, as well as shape and volume of asteroid.	Integration Time	1.2 seconds	1.2 seconds	Dawn Framing Camera	
				Field Curvature	< 10 µm	< 10 µm		
				Instantaneous Field of View (IFOV)	93.7 µrad px^-1	93.7 µrad px^-1		The total power consumption of all instruments shall be 545W from both panels, as well as 18Ah from battery pack.
				Spectral Range	400nm - 1050nm	350 nm - 1050nm		
	Obj. #3: Determine the mass of a Near Earth Object (NEO)	Measure gravitational field and dimensional characteristics. (Consider composition when determining mass.)	Shifts in Doppler tracking data; optical imaging of NEO's size, shape, and center location; surface feature mapping; elemental composition (from Obj. #1); remote sensing; telemetry from altitude control subsystem	Wavelength Ranges	300nm - 1500nm	1064 nm (near infrared), 532 nm (visible green), 355 nm (ultraviolet)	Laser Altimeter (Light Detection and Ranging)	
				Precision and Accuracy	.05m - 10m	0.05m - descent and landing; 0.2m - mid-altitude mapping; 10m - initial survey		System must be 65kg or less.
				Range Rate Due To Topography	1 m/s	1 m/s		
				Scan Angle	> 10 degrees	30 degrees		

3.2.3. Payload Success Criteria

The success criteria are as follows:

- Determine elemental makeup of asteroid regolith, spectral type of asteroid (Carbonaceous, Salicaceous, Metallic)
- Measure density through mass divided by volume, using observables from science objective 1 and 3
- Measure gravitational field and dimensional characteristics to calculate mass

To achieve success criteria, the mission must do the following: determine the elemental makeup of asteroid regolith, verify the composition and shape of the asteroid readings from other observables, and measure the gravitational field and dimensional characteristics. Observables for the AIDENNE mission would include the emission spectra of carbon/silicon/metals, asteroid origin and evolution, shape, and volume of the asteroid, as well as optical imaging of the NEO's size, shape, and center location. Additionally, each instrument used on the AIDENNE mission has a specific function that goes directly with the objectives. Some instruments such as the Dawn Framing Camera and GRaND go hand in hand, due to the relationship between both science objectives. This method was the most efficient way to achieve the mission success criteria.

3.2.4. Experimental Logic and Evolution, Approach, and Method of Investigation

This scientific plan was created to collect information on the surface of the NEO. Upon the arrival of the spacecraft on targeted NEO, the spacecraft will observe and measure the asteroid over the course of half a year (at a minimum) to gather information. The Gamma Ray and Neutron Detector (GRaND) is an instrument that is used to determine elemental composition. It measures gamma rays and neutrons that either bounce off or are emitted by a body. The instrument can use 21 sensors to measure important atomic constituents of the NEO body's surface down to a depth of one meter. GRaND will detect carbon, silicon, and other metals wavelengths to determine the elemental makeup. The Dawn Framing Camera, which uses a refractive optical system to help study minerals, will verify the readings from GRaND. LIDaR, will perform its own readings by gathering optical imaging of the NEO's size, shape, and center location.

3.2.5. Testing and Calibration Measurements

The instrumentation systems will go through tests during Phase D: System Assembly, Integration and Test, launch in similar environmental conditions to ensure the operations of the systems are at a satisfactory level. The calibration for the payload will undergo certain testing environments to make sure optimum performance and failure in the system. Tests such as TVAC, EMI, VIBE, and Ambient in the facility ensure failure due to the harsh environment in space. As per requirement constraint, the mission takes 25 to 32 months in manufacturing and testing process. In this period, instruments go through multiple testing and remodeling for the optimum performance in the mission.

3.2.6. Precision and Accuracy of Instrumentation

Name	Accuracy

Gamma Ray and Neutron Detector	Results can be achieved with a scan time of 60 seconds and with a further data processing time of less than 60 seconds, for gamma sources of ~300kBq and neutron sources of 1,000,000 neutrons per second (total) in proximity.
Framing Camera	The calibration of each filter is accurate to within a few percent for point sources. It has a resolution of about 140 meters per pixel.
Light Detection and Ranging	Able to achieve range accuracy of .5mm to 10mm relative to the sensor and a mapping accuracy of up to 1 cm horizontal (x,y) and 2cm vertical (z).

3.2.7. Expected Data & Analysis

Mission concept is specific to find the composition, mass, and density of the NEO. We are equipped with three instruments such as GrAND, FC and LIDAR which can meet the mission requirement. Three of these instruments are proven in quality and success in several missions.

GrAND simulates galactic cosmic rays as high as 100 MeV can create 500 keV on the surface of the asteroid. The reflected neutrons from the surface of the asteroid can be measured with fast neutron flux such as Fe, Ti, Gd, Sm. Hydrogen can detect <1 eV neutral flux and 1 eV to 500 keV can detect presence of H2O. Gamma-Ray from stable elements can detect the presence of Si, Fe, S, Na, Ca, Al, Mg and can deduct radioactive elements such as K,Th and U. Gamma-Ray originates from GrAND can penetrate 10s of cm in-depth which is not capable in similar instruments.

FC is equipped as the eye of the mission to see the things and process in real-time to accomplish the mission. FC processes data through onboard image processing. This optical image processing can determine the orientation, mass and gravitational field and importantly mapping the surface to identify the size and volume. The surface mapping can identify craters and morphology, tectonic activity such as volcanism and faulting. Multispectral image processing can identify elements and mineralogical composition of targeted NEO.

LiDAR is an efficient instrument in determining the mass and density of the NEO dynamics. The technology uses fiber laser return of zero pseudo-noise with single photon sensitive detection. The specified values can identify the shape and trajectory. A single laser pulse can detect pixelated 3-D images of the surface with details. LiDAR can also detect the spacecraft's altitude and velocity from NEO.

4. Mission Risk Management

4.1. Safety and Hazard Overview

Proper safety and risk control is critical for mission success, both in pre-flight and orbital conditions. Risks to the spacecraft that could compromise mission success have been identified through analysis of prior missions and research on common component failure points. Mission risk criticality is assessed based on likelihood of occurrence and anticipated level of consequence, as defined in Table 4.1. A risk matrix chart (Table 4.2) is used to quantify the severity of risks. Risks are tracked on a weekly basis to determine the appropriate mitigation approach.

4.1.1. Risk Analysis

L = Likelihood (1-5)	LxC Trend	Approach	Criticallity
1 = not likely	↓ - Decreasing (improving)	A - accept	HIGH
5 = extremely likely	↑ - increasing (worsening)	M- mitigate	MED
C = Consequence (1-5)	→ - unchanged	W - watch	LOW
1 = low consequence	**NEW** - added this month	R - research	
5 = high consequence			

Table 4.1 Risk Assessment Chart

LIKELIHOOD	1	2	3	4	5
5					
4					
3			1,2		
2	7			5,9	4
1				3	6,8,10,11
	CONSEQUENCES				

Table 4.2 Risk Matrix Chart

ID	Summary	L	C	Trend	Approach	Risk Statement	Status
1	Solar Panel Deployment Failure	3	3	↓	R/M	Observed on the Lucy mission, a failure of one or both in deployment can complicate the mission as there would be insufficient power for instruments.	YLW/MED Active (once red)
2	Inertia Measurement Unit (IMU) Failure	3	3	↓	R/M	IMUs, which measure altitude, velocity, and position, can and have failed on past orbiter missions (MAVEN, MRO), and could be debilitating for this mission if we want to model it off of the Lucy mission.	YLW/MED Active (once red)
3	Depletion of Propellant	1	4	↓	R/A	The Dawn mission, which studied Ceres, ended due to the depletion of its hydrazine propellant. A depletion of our own propellant can complicate science value due to being unable to transfer from asteroid to asteroid.	GRN/LOW Active
4	Orbiter Damaged by NEO Encounter	2	5	↓	R/M	If we are designing the mission based on Lucy, the risk of orbiter damage due to NEO encounters will be especially high. NEOs also include space debris, which are prevalent in Low-Earth Orbit (LEO).	YLW/MED Active (once red)
5	Single-State Event Avionics Failure	2	4	→	R	Cosmic radiation particles can energize electronic components, resulting in parasitic drain and sudden electrostatic discharge.	YLW/MED Active
6	Unit Conversion Error	1	5	↓	M/W	Observed on Mars Climate Orbiter, engineers failed to make a simple conversion from English units to metric, resulting in incorrect data.	YLW/MED Active (once red)
7	Lapses in Tracking Data	2	1	→	R	The ground team uses X-band radio to track the orbiter. They can be blocked by the asteroid or positioning of the orbiter, which can compromise mass estimates.	GRN/LOW Active (once red)
8	Memory Failure (Single/Multi-Bit Upset)	1	5	→	R/W	Observed on Hubble, the advent of memory failure could prove mission-ending as commands, science, and communications depend on spacecraft memory to work nominally. Cosmic radiation particles can energize and ionize electronic components, resulting in memory state changes.	YLW/MED Active
9	Battery Wear	2	4	→	R/M/A	Observed on older solar powered spacecraft (MRO, MER-1), the advent of battery wear will come with time, but risk will increase if we depend on only one battery while the spacecraft enters a non-sunlight period.	YLW/MED Active
10	Structural Failure	1	5	→	M/W	Anything dealing with the casing of the spacecraft and its structural integrity during the launch and orbit stages.	YLW/MED Active
11	Communication Failure	1	5	→	M/W	Observed on Ariane 5, communication broke down for more than nine minutes, which resulted in two satellites being placed into orbit with incorrect orbital inclinations 21 degrees versus the planned 3 degrees.	YLW/MED Active

Table 4.3 Spacecraft Risk Summary Table

ID	Summary	Status	Risk Statement	Mitigation Plan
1	Solar Panel Deployment Failure	YLW/MED Active	Observed on the Lucy mission, a failure of one or both in deployment can complicate the mission as there would be insufficient power for instruments.	Running the primary and backup motors for that specific array panel. Solar panels, if not designed like Lucy's, can also be partially deployed.
2	Inertia Measurement Unit (IMU) Failure	YLW/MED Active	IMUs, which measure altitude, velocity, and position, can and have failed on past orbiter missions (MAVEN, MRO), and could be especially debilitating for this mission if we want to model it off of the Lucy mission.	Incorporate backup IMUs into this mission. The mission will be readapted in the case of backup IMUs also failing. (all-stellar mode).
3	Depletion of Propellant	GRN/LOW Active	The Dawn mission, which studied Ceres, ended due to the depletion of its hydrazine propellant. A depletion of our own propellant can also spell mission end, though the likelihood will increase with mission timeline.	Mission will be readapted or concluded based on scientific value of a stable orbit around one asteroid. Unlikely to occur during beginning-middle stages which lowers the consequences of decreased scientific value.
4	Orbiter Damaged by NEO Encounter	YLW/MED Active	If we are designing the mission based on Lucy, the risk of orbiter damage due to NEO encounters will be especially high. NEOs also include space debris, which are prevalent in Low-Earth Orbit (LEO).	NEO Surveyor-Ground Telescope "Eyes" System, infrared cameras on spacecraft
5	Single-State Event Avionics Failure	YLW/MED Active	Cosmic radiation particles can energize electronic components, resulting in parasitic drain and sudden electrostatic discharge.	Implement protective circuitry components into avionics system design.
6	Unit Conversion Error	GRN/LOW Active	Engineers failed to make a simple conversion from English units to metric, resulting in incorrect data.	Mission will be inspected properly and consistently to ensure no errors are made. Mission will be readapted in the case procedure has inaccurate data.
8	Memory Failure (Single/Multi-Bit Upset)	YLW/MED Active	Observed on Hubble, the advent of memory failure could prove mission-ending as commands, science, and communications depend on spacecraft memory to work nominally. Cosmic radiation particles can energize and ionize electronic components, resulting in memory state changes.	Design capacitive hardening of RAM into spacecraft circuitry. Implement error detection and correction codes to execute protective measures when an error occurs and prevent flow-down effects on other electronic components.
9	Battery Wear	YLW/MED Active	Observed on older solar powered spacecraft (MRO, MER-1), the advent of battery wear will come with time, but it will be complicated if we are to depend on only one battery if the spacecraft enters a non-sunlight period.	Inevitable, but the advent of new, longer lasting batteries (TBR) lowers the anticipated likelihood of wear expected from past missions. Non-sunlight periods are also shorter for asteroid orbits due to smaller/more frequent orbit cycles. The mission will be

Table 4.4 Spacecraft Risk Mitigation Plan

4.1.2. Failure Mode and Effect Analysis (FMEA)

Function	Failure Mode	Effects	Sev	Cause	Occ	Prevention	Det	RPN	Actions
Solar Panels	Failure of one more solar panels to deploy.	Insufficient power provided to instruments.	60	Deployment design flaw.	60	Run primary and backup motors.	50	180000	Panels can be used partially deployed.
Inertia Measurement Unit	Failure to measure altitude.	Inability to find and establish position.	60	Interference with unit or insufficient power supply.	20	Implement backup units.	10	12000	The mission will be readapted.
	Failure to measure velocity.	Inability to determine point on course and how to reach desired point.	60	Interference with unit or insufficient power supply.	20	Implement backup units.	10	12000	The mission will be readapted.
	Failure to measure position.	Inability to reach desired point.	60	Interference with unit or insufficient power supply.	20	Implement backup units.	10	12000	The mission will be readapted.
Avionics	Single-state error in memory storage.	Temporary operational system malfunctions.	30	Cosmic radiation particle collides with memory storage cell and corrupts stored information.	30	Capacitive hardening of RAM, Error-Correcting Code (ECC) memory.	90	81000	Switch off power supply to memory circuit for a pre-programmed time using watchdog timer circuitry.
	Multi-bit error in memory storage.	Possible permanent alterations or shutdown of system functionality	80	Accumulation of single-state errors due to lack of mitigation and/or increased burst of cosmic radiation.	20	Capacitive hardening of RAM, increased silicon application on semiconductors, Advanced error-correcting code (ECC) memory.	70	112000	Switch off power supply to memory circuit for a pre-programmed time & data scrubbing
	Drainage of current.	Loss of equipment functionality.	80	Electrostatic discharge caused by overcharging of electronic components by cosmic radiation particles.	40	Overcurrent protective circuitry external to the memory circuit.	10	32000	Detection of excessive current in the board. The board is switched off and the current is later re-established.
	System failure in flight.	Loss of communication, inoperative equipment, inability to compute and deliver data.	100	Cosmic radiation particles energizing and ionizing electric components.	40	Protection and inspection of the system.	10	40000	Assessment and rehabilitation.
	Innoperative communication devices.	Incorrect orbital position.	100	Environmental interference.	20	Static discharging equipment.	20	40000	Take load off the system and allow it to recover.
	Inoperative memory.	Loss of commands, science, and communication.	100	Cosmic radiation particles energizing and ionizing electric components.	20	Protection and inspection of the system.	30	60000	Assessment and rehabilitation.
Data Collection	Incorrect input.	Inaccurate data.	100	Failure of engineers to make unit conversions.	40	Consistent inspection.	40	160000	Mission readaptation.
Vehicle Launch	Inability to lift.	Mission delay.	100	Faulty ignition system.	40	Consistent inspection.	90	360000	Suspend the mission and repair system.
Structure	Structural damage.	Loss of equipment, inoperative equipment, inability of vehicle to perform required tasks.	100	Design flaw or improper loading.	10	Strong structural integrity.	20	20000	Mission readaptation.

Table 4.5 Spacecraft Failure Mode and Effects Analysis

4.1.3. Personnel Hazards and Mitigations

The Safety Officer for this mission is Bailey Sarna. The safety officer is responsible for identifying and documenting occupational hazards to ground personnel, while also promoting the importance of safety culture to ensure compliance to established safety and health procedures throughout the mission. The safety officer is also responsible for facilitating the development of spacecraft risk assessment charts alongside the science and engineering teams, monitoring ongoing risks, and updating mitigation plans, as necessary.

During fabrication, assembly, and testing of the spacecraft, ground personnel will encounter various occupational hazards that could result in damage to facilities, serious personal injury, and/or death without proper precautions and procedures. Severity of anticipated occupational hazards (Table 4.6) has been assessed (To Be Finalized later) and categorized based on likelihood of exposure and their consequences before mitigation, as well as their likelihood of occurring after mitigation procedures have been implemented.

L = Likelihood of Exposure (1-5)	**Severity**
1 = Rare	**HIGH**
5 = Common	**MED**
C = Consequence (1-5)	**LOW**
1 = Low consequence	
5 = High consequence	

Table 4.6 Personnel Hazard Assessment

ID	Environment	Hazard	Effects	L	C	Severity Before Mitigation	Mitigation Procedures	Likelihood After Mitigation
P1	Manufacturing, assembly, testing	Ergonomic hazards (force, repetition, duration, fixed postures)	Strain & personal injury, musculoskeletal disorders (MSDs)	4	3	MEDIUM	Frequent rest breaks, proper lifting techniques and mandated team lift for heavy objects, purchase ergonomic-friendly equipment	LOW
P2	Manufacturing, assembly, testing	Exposure (inhalation, skin) to hazardous chemicals	Chemical burns, respiratory damage	4	5	HIGH	MSDS, wearing proper PPE, monitor exposure limits, store in separate location, eyewash and emergency shower stations, chemical handling training	MEDIUM
P3	Manufacturing, assembly	Welding, brazing, solder (heat, lead, electricity, fumes, radiation, grinding, fire)	Burns, shock, respiratory damage, lead poisoning, vision loss, asphyxiation, cuts	3	5	HIGH	Wearing proper PPE, ventilated work spaces, restrict food and drink from workspace, require certification	MEDIUM
P4	Manufacturing, assembly, testing	Batteries (electricity, chemicals, venting, rupture/explosion, temperature)	Burns, shock, explosions, respiratory damage, vision loss	5	5	HIGH	Wearing proper PPE, proper handling procedures, SDS, eyewash station, ventilated work spaces, battery safety training, fire protection equipment	MEDIUM
P5	Manufacturing, assembly, testing	Combustible gases and liquids	Fires, explosions, burns	2	5	MEDIUM	Proper storage equipment, proper handling procedures,	LOW
P6	Manufacturing, assembly, testing	Slip/trip hazards (cords, equipment, stairs, elevation changes)	Sprains, strains, cuts, personal injury	3	2	LOW	Proper workspace cleanliness, retroreflective caution tape, signage	LOW
P7	Manufacturing, assembly, testing	Excessive noise	Hearing damage	3	4	MEDIUM	Wearing proper PPE, cautionary signage, annual hearing tests	LOW
P8	Manufacuring, assembly	Exposure to heavy machinery and fabrication equipment	Personal injury, death	3	5	HIGH	Wear proper PPE, detailed use procedures, lockout-tagout program, require certification and/or lab-specific training	MEDIUM
P9	Testing	Thermal chamber (temperature, electricity, fumes)	Burns, shock, respiratory damage, personal injury	1	3	LOW	Perform pre-test inspection, detailed use procedures, lockout-tagout program, require certification and/or lab-specific training, wear proper PPE, waste management	LOW
P10	Manufacturing, assembly, testing	Exposure to electricity	Shock, burn, personal injury	4	2	MEDIUM	Wear proper PPE, restrict food and drink from lab, restrict jewelry in lab, ground all work benches, require certification and/or lab-specific training, lockout-tagout procedures, routine inspection of work equipment	LOW
P11	Manufacturing, assembly, testing	Exposure to high voltage electricity	Severe personal injury, death	2	5	MEDIUM	Wear proper PPE, restrict food and drink from lab, restrict jewelry in lab, ground all work benches, require certification and/or lab-specific training, lockout-tagout procedures, routine inspection of work equipment	LOW
P12	Testing	Exposure to RF radiation	Pacemaker interference	1	3	LOW	Cautionary signage, restrict jewelry in test area,	LOW

Table 4.7 Personnel Hazards and Mitigation Procedures

5. Activity Plan

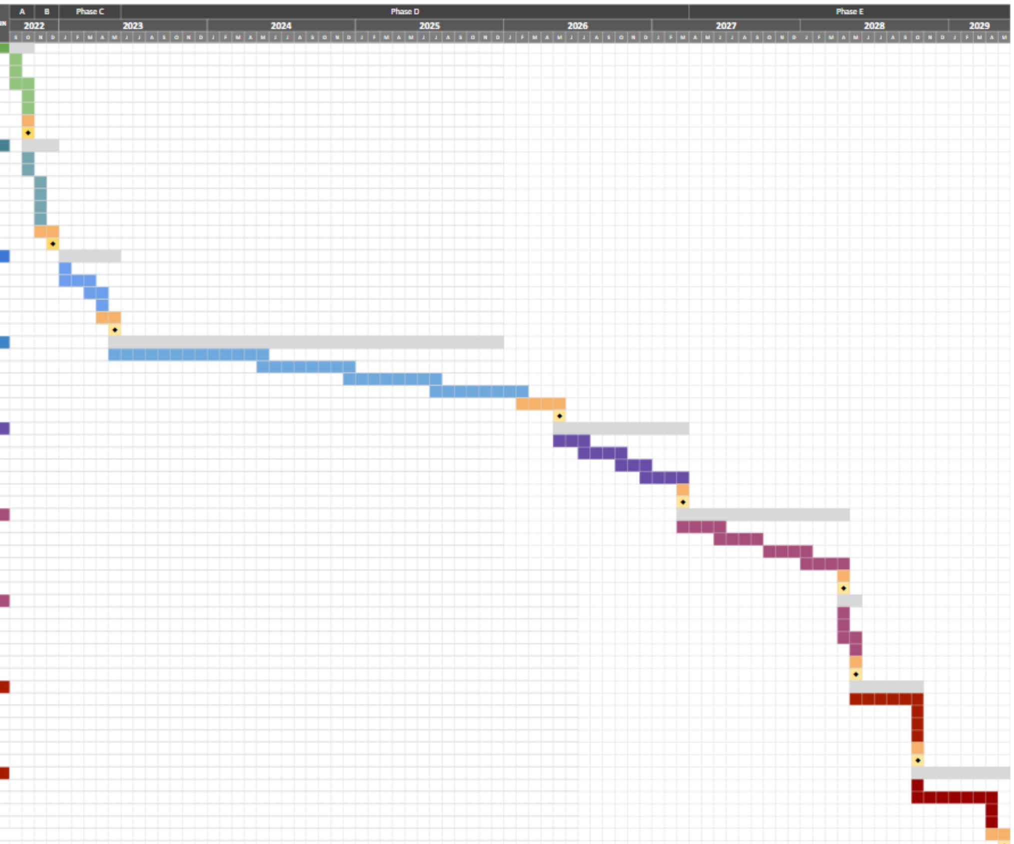
5.1. Schedule

Mission AIDENNE, Spacecraft SAMURAI

Team Number: # 29

Project Team Members: Shiza Ahmed, Andrea Chavez, Miguel Cunanan, Caleb Emesiani, Yuri Labuca, Vincent Ma, Baylee Peak, Bailey Sarna, Sumesh Surendran

ID#	TASK	ASSIGNED TO	PROGRESS	START	END	DAYS	MARGIN
1	**Concept and Technology Development**		**100%**	**9/13/22**	**10/11/22**	**29**	**1**
1.1	Develop preliminary science goals and objectives	Entire team	Complete	9/13/22	9/20/22	8	
1.2	Define science measure requirements	Entire team	Complete	9/20/22	9/27/22	8	
1.3	Define instrumentation performance requirements	Entire team	Complete	9/27/22	10/4/22	8	
1.4	Define instrumentation and mission requirements	Entire team	Complete	10/4/22	10/11/22	8	
1.4	Draft MCRR	Entire team	Complete	10/11/22	10/11/22	1	
1.5	Schedule Margin			10/11/22	10/11/22	1	
1.6	Mission Concept and Requirements Review Completed		Complete	10/11/22	10/11/22	1	
2	**Preliminary Design and Technology Completion**		**100%**	**10/11/22**	**12/13/22**	**64**	**7**
2.1	Identify and analyze mission risks	Entire team, safety officer	Complete	10/11/22	10/18/22	8	
2.2	Generate preliminary schedule, budget, outreach plans	Project manager, business, outreach	Complete	10/18/22	10/25/22	8	
2.3	Define payload design and science instrumentation plans	Engineering, science	Complete	11/1/22	11/15/22	15	
2.4	Define overall vehicle and systems design	Engineering, science	Complete	11/15/22	11/29/22	15	
2.5	Draft 1 PDR	Entire team	Complete	11/15/22	11/29/22	15	
2.6	PDR Final Draft	Entire team	Complete	11/15/22	11/29/22	15	
2.7	Schedule Margin			11/29/22	12/5/22	7	
2.8	◆ Preliminary Design Review (PDR) Presentation		Complete	12/13/22	12/13/22	1	
3	**Final Design and Fabrication**		**0%**	**1/9/23**	**5/5/23**	**117**	**8**
3.1	Finalization of Vehicle Design Model	Engineering Team	Not complete	1/9/23	1/16/23	8	
3.2	Fabrication of Vehicle Design Model	Engineering Team	Not complete	1/17/23	3/17/23	60	
3.3	Testing of Model According to Requirements	Science Team	Not complete	3/18/23	4/18/23	32	
3.4	Finalization of Vehicle Design	Engineering Team	Not complete	4/19/23	4/26/23	8	
3.5	Schedule Margin			4/27/23	5/4/23	8	
3.6	◆ Critical Design Review (CDR) Presentation		Not complete	5/5/23	5/5/23	1	
4	**System Manufacturing and Assembly**		**0%**	**5/6/23**	**5/6/26**	**1097**	**85**
4.1	Manufacturing of Systems Components	Project Manager	Not complete	5/6/23	5/6/24	367	
4.2	Assembly of Systems	Engineering Team	Not complete	5/7/24	12/7/24	215	
4.3	Testing of Systems According to Requirements	Science Team	Not complete	12/8/24	7/8/25	213	
4.4	Assembly of Vehicle	Engineering Team	Not complete	7/9/25	2/9/26	216	
4.5	Schedule Margin			2/10/26	5/5/26	85	
4.6	◆ Final Assembly Procedure of System Completed		Not complete	5/6/26	5/6/26	1	
5	**Systems Integration**		**0%**	**5/7/26**	**3/28/27**	**326**	**14**
5.1	Integrate Mechanical Subsystems with Respective Electrical	Mechanical and Electrical Engineers	Not complete	5/7/26	7/7/26	62	
5.2	Testing of Integral Subsystems	Science Team	Not complete	7/8/26	10/8/26	93	
5.3	Integrate Subsystems	Engineering Team	Not complete	10/9/26	12/9/26	62	
5.4	Testing of Systems Integration	Science Team	Not complete	12/10/26	3/13/27	94	
5.5	Schedule Margin			3/14/27	3/27/27	14	
5.6	System Integration Review (SIR)		Not complete	3/28/27	3/28/27	1	
6	**Testing**		**0%**	**3/29/27**	**4/17/28**	**386**	**14**
6.1	Testing of Subsytem Deployment	Science Team	Not complete	3/29/27	6/29/27	93	
6.2	Testing of Communications and Data Collection	Science Team	Not complete	6/30/27	9/30/27	93	
6.3	Testing of Payload Deployment and Functionality	Science Team	Not complete	10/1/27	1/1/28	93	
6.4	Testing of Power and Thermal Systems	Science Team	Not complete	1/2/28	4/2/28	92	
6.5	Schedule Margin			4/3/28	4/16/28	14	
6.6	◆ Test Readiness Review (more to come)		Not complete	4/17/28	4/17/28	1	
7	**System Acceptance Review**		**0%**	**4/16/28**	**5/21/28**	**36**	**7**
6.1	All Systems Deployable and Fully Functioning	Science and Engineering Teams	Not complete	4/16/28	4/22/28	7	
6.2	Communications System Relay Data Collection	Science and Engineering Teams	Not complete	4/23/28	4/29/28	7	
6.3	Power System Has a Capacity of 300W and Can Recharge Two Batteries	Science and Engineering Teams	Not complete	4/30/28	5/6/28	7	
6.4	Thermal System Maintains Optimal Operating Temperatures of Systems	Science and Engineering Teams	Not complete	5/7/28	5/13/28	7	
6.5	Schedule Margin			5/14/28	5/20/28	7	
6.6	◆ System Acceptance Review (SAR)		Not complete	5/21/28	5/21/28	1	
8	**Launch**		**0%**	**5/22/28**	**10/24/28**	**156**	**2**
7.1	SAMURAI Out for Delivery	Project Management	Not complete	5/22/28	10/22/28	154	
7.2	Project Management Arrives at Launch Site	Project Management	Not complete	10/22/28	10/22/28	1	
7.3	SAMURAI Arrives at Launch Site	Project Management	Not complete	10/22/28	10/22/28	1	
7.4	Technicians, Engineers, and Scientists Arrive at Launch Site	Technicians, Scientist, and Engineers	Not complete	10/22/28	10/22/28	1	
7.5	Schedule Margin			10/22/28	10/23/28	2	
7.6	◆ Launch Day		Not complete	10/24/28	10/24/28	1	
9	**Arrival and Research Operations**		**0%**	**11/19/28**	**5/6/29**	**169**	**7**
7.1	Arrival	Whole Team	Not complete	11/19/28	11/19/28	1	
7.2	Data Collection	Whole Team	Not complete	11/20/28	4/20/29	152	
7.3	Primary Mission Completion	Whole Team	Not complete	4/21/29	4/21/29	1	
7.4	Reassessment and Brainstorming	Whole Team	Not complete	4/22/29	4/28/29	7	
7.5	Schedule Margin			4/29/29	5/5/29	7	
7.6	◆ End of Orbital Research		Not complete	5/6/29	5/6/29	1	



5.2. Budget

Additional Information

# People on Team	FY 1	FY 2	FY 3	FY 4	FY 5	FY 6
Science Personnel:	3	4	4	4	5	5
Engineering Personnel:	5	8	8	8	8	5
Technicians:	3	10	10	10	8	0
Administration Personnel:	5	5	5	5	5	3
Management Personnel:	2	2	2	3	3	2

NASA L'SPACE Mission Concept Academy Budget - *AIDENNE Mission*

Year	Yr 1 Total	Yr 2 Total	Yr 3 Total	Yr 4 Total	Yr 5 Total	Yr 6 Total	Cumulative Total
PERSONNEL							
Science Personnel	$ 120,000.00	$ 320,000.00	$ 320,000.00	$ 320,000.00	$ 400,000.00	$ 400,000.00	$ 1,880,000.00
Engineering Personnel	$ 200,000.00	$ 640,000.00	$ 640,000.00	$ 640,000.00	$ 640,000.00	$ 400,000.00	$ 3,160,000.00
Technicians:	$ 90,000.00	$ 600,000.00	$ 600,000.00	$ 600,000.00	$ 480,000.00	$ -	$ 2,370,000.00
Administration Personnel	$ 200,000.00	$ 400,000.00	$ 400,000.00	$ 400,000.00	$ 400,000.00	$ 240,000.00	$ 2,040,000.00
Project Management:	$ 150,000.00	$ 300,000.00	$ 300,000.00	$ 450,000.00	$ 450,000.00	$ 300,000.00	$ 1,950,000.00
Total Salaries	$ 760,000.00	$ 2,260,000.00	$ 2,260,000.00	$ 2,410,000.00	$ 2,370,000.00	$ 1,340,000.00	$ 11,400,000.00
Total ERE	$ 212,116.00	$ 630,766.00	$ 630,766.00	$ 672,631.00	$ 661,467.00	$ 373,994.00	$ 3,181,740.00
TOTAL PERSONNEL	$ 972,116.00	$ 2,890,766.00	$ 2,890,766.00	$ 3,082,631.00	$ 3,031,467.00	$ 1,713,994.00	$ 14,581,740.00
TRAVEL							
Total Flights Cost	$ -	$ -	$ -	$ -	$ 15,600.00	$ -	$ 15,600.00
Total Hotel Cost	$ -	$ -	$ -	$ -	$ 14,976.00	$ -	$ 14,976.00
Total Transportation Cost	$ -	$ -	$ -	$ -	$ 1,162.00	$ -	$ 1,162.00
Total Per Diem Cost	$ -	$ -	$ -	$ -	$ 11,180.00	$ -	$ 11,180.00
Total Travel Costs	$ -	$ -	$ -	$ -	$ 42,918.00	$ -	$ 42,918.00
OUTREACH							
Total Outreach Materials	$ 21,455.50	$ 21,455.50	$ 21,455.50	$ 21,455.50	$ -	$ -	$ 85,822.00
Total Outreach Venue Costs	$ 3,600.00	$ 3,600.00	$ 3,600.00	$ 3,600.00	$ -	$ -	$ 14,400.00
Total Outreach Costs	$ 25,055.50	$ 25,055.50	$ 25,055.50	$ 25,055.50	$ -	$ -	$ 100,222.00
DIRECT COSTS							
|> Science Instrumentation	$ 19,700,000.00	$ -	$ -	$ -	$ -	$ -	$ 19,700,000.00
|> Other Payload Costs	$ -	$ -	$ -	$ -	$ -	$ -	$ -
Total Payload Costs	$ 19,700,000.00	$ -	$ -	$ -			$ 19,700,000.00
|> Mechanical Subsystem	$ 4,400,000.00	$ -	$ -	$ -	$ -	$ -	$ 4,400,000.00
|> Power Subsystem	$ 5,380,000.00	$ -	$ -	$ -	$ -	$ -	$ 5,380,000.00
|> Thermal Control Subsystem	$ 1,400,000.00	$ -	$ -	$ -	$ -	$ -	$ 1,400,000.00
|> Comms/Data Handling Subsystem	$ 1,000,000.00	$ -	$ -	$ -	$ -	$ -	$ 1,000,000.00
Total Vehicle Costs	$ 12,180,000.00	$ -	$ -	$ -			$ 12,180,000.00
|> Manufacturing Facility Cost	$ 9,500,000.00	$ -	-	$ -	$ -	$ -	$ 9,500,000.00
|> Test Facility Cost	$ -	$ 6,000,000.00	$ 6,000,000.00	$ 12,000,000.00	$ -	$ -	$ 24,000,000.00
Total Facilities Costs	$ 9,500,000.00	$ 6,000,000.00	$ 6,000,000.00	$ 12,000,000.00	$ -	$ -	$ 33,500,000.00
Manufacturing Margin	$ 20,690,000.00	$ 3,000,000.00	$ 3,000,000.00	$ 6,000,000.00	$ -	$ -	$ 32,690,000.00
Total Direct Costs	$ 62,070,000.00	$ 9,000,000.00	$ 9,000,000.00	$ 18,000,000.00	$ -	$ -	$ 98,070,000.00
Total MTDC	$ 15,940,000.00	$ -	$ -	$ -	$ -	$ -	$ 15,940,000.00
FINAL COST CALCULATIONS							
Total F&A	$ 1,594,000.00	$ -	$ -	$ -	$ -	$ -	$ 1,594,000.00
Total Projected Cost	$ 64,661,171.50	$ 11,915,821.50	$ 11,915,821.50	$ 21,107,686.50	$ 3,074,385.00	$ 1,713,994.00	$ 114,388,880.00
Total Cost Margin	$ 19,398,351.45	$ 3,574,746.45	$ 3,574,746.45	$ 6,332,305.95	$ 922,315.50	$ 514,198.20	$ 34,316,664.00
Total Project Cost	$ 84,059,522.95	$ 15,490,567.95	$ 15,490,567.95	$ 27,439,992.45	$ 3,996,700.50	$ 2,228,192.20	$ 148,705,544.00

********** Do not change percentages in the boxes below unless mission concept instructions specify otherwise.

F&A %	10%	10%	10%	10%	10%	10%
Manufacturing Margin	50%	50%	50%	50%	50%	50%
Total Cost Margin	30%	30%	30%	30%	30%	30%
ERE - Staff	28%	28%	28%	28%	28%	28%

COTS (or outsourcing) COSTS, ASSUMING $1m for every 1kg unless otherwise found, roughly:

Redwire CSS: $260k
RL Star Tracker: $280k (from company)
RL Reaction Wheels: $140k (from company)
Redwire Solar Panels: ~$3,000,000
4x RC CF+Aramid (0.4 x 0.5 x 0.009525): $700 (cost per m^3 derived from company pricing https://www.rockwestcomposites.com/shop/2437-3302-50100)
1x RC CF+Aramid (0.5 x 0.5 x 0.009525): $220 (cost per m^3 derived from company pricing https://www.rockwestcomposites.com/shop/2437-3302-50100)
1x RC Al (0.5 x 0.5 x 0.009525): $100 (cost per m^3 derived from company pricing https://www.rockwestcomposites.com/shop/74016-4896)
3x 4s1p VES16: $2,100,000
Alen Space OBC-TTC: $200k
Spacecom HGA X-band: $20k
Giken LGA: ~$300k

All others, MCCET, assume built in house

MCCET COSTS:
Thermals: 642(1.56^.62) = ~$1.4m w/o wraps, $2m with
Mechanical: 219(11.899^0.41)(19^0.52) = ~$4.4m w/o wraps, $6m with
Software: Assume $500K for compatibility testing, modding4,

Total subsystem cost: $18,701,020 ($19m) without wraps, $24,801,020 with ($25m)

Scientific Instrument MCCET Costs:

GRAND (Particles) - 223(5.4^0.31)(9^0.44)(7^0.52) = $4.3m, $6m with wraps
LIDAR (Optical) - 392(4^0.45)(14.3^0.49)(7^0.32) = $7.9m, $11m with wraps
FC (Optical) - 392(4^0.45)(13^0.49)(7^0.32) = $7.5m, $10m with wraps

$27m with wraps

SI Test costs:
G = $2m per test
L = $3m per test
F = $3m per test

Once through for testing of assembled components, twice to account for spacecraft testing

$6m for test of in-house components
$16m for test of SIs

COTS Approx testing 25% of COTS cost
CSS: $65k
Star Tracker: $70k
Reaction Wheels: $35k
Solar Arrays: $750k
Bus panels: $255
Batteries: $525k
Central Computer: $50k
HGA: $5k
LGA: $75k
Software: $500k

COTS Approx Testing cost: $2.1m

Total testing cost: $24m

All branded components are COTS and have their specs pulled from technical info websites or datasheets found online

Narrative: Above shows the budget and the reasoning behind direct costs of the spacecraft. The difference between unwrapped and wrapped costs were implemented into manufacturing facility costs. It was assumed that any COTS components were $1m for every 1kg, unless found directly, which is a rough estimate that was done only due to the time-consuming process of quoting manufacturing costs. COTS testing was assumed to be 25% of the original cost, as all COTS components were tested as part of the outsourced manufacturer plan by themselves or have had TRL experience already. Therefore, the only necessary stringent testing was on the complete system. All in-house component costs were calculated through the MCCET tool, resulting in $24m of testing cost, spread over the proceeding years as components arrive and are tested. All personnel start out in small numbers during FY1 dueto the majority of COTS equipment being ordered around this time. FY2, technicians, engineers, and science personnel are increased due to the arrival and testing of some components and analyses. These stay constant until FY5. FY4, management will be increased by 1 due to the number of components present, to be tested and assembled. FY5 is the launch year, technicians get a reduction due to finalization testing and scientists are increased due to the beginning of the science phase. FY6, only engineers remain to look over spacecraft health. Because of the spacecraft's partly autonomous ability, these engineers were reduced.

5.3. Outreach Summary

The outreach officer for this mission is Shiza Ahmed. The responsibilities for this role include planning the outreach of the mission by fundraising through public events, as well as promoting the mission/goals to the community across the country.

To increase public awareness and appreciation for STEM (especially for younger generations) and promote the mission, AIDENNE, the team will reach out to nearby colleges and universities who are interested in hosting educational events for K-12 students. The most effective way to encourage a child's interest in STEM related topics is to show how they interact with certain concepts on a day-to-day basis. The team will also utilize social media to promote our mission on platforms such as Instagram, Facebook, and TikTok. These platforms will allow the team to reach a wider audience.

SCHOOL EVENTS: As in-person events are what bring community outreach, providing opportunities for students in the area will provide them with practical experience. The events hosted by nearby schools will include an intelligible and interactive presentation for the students that could introduce the team's mission. Then, the team would be able to move towards hands-on experience, such as building 3D paper models, or even 3D printing. Having subject matter experts present with the team would benefit students even more. With in-person events, the team would be able to outreach to underprivileged students, for those who do not have access to Wi-Fi or social media. Events would be open to all students, whether they are presently attending or not.

SOCIAL MEDIA: By making a social media presence for the AIDENNE mission, the team will be able to announce to community members the progress of the mission. With the outreach on social media platforms such as Instagram, Twitter, and TikTok, the team could promote online presentations for the mission as well, for those who did not have the opportunity to come in-person. With the outreach on social media, as well as the online educational events, the team would be able to ensure that AIDENNE is actively engaging people, even outside of the community.

5.4. Program Management Approach

Team roles were delegated in a democratic process where team members submitted a statement of interest for prospective roles. Assignment for roles that received multiple statements of interest were determined through an election process or deferment from the role. Work on the PDR was done in weekly collaborative meetings, where each team contributed to different sections of the PDR whilst seeking input and feedback from the rest of the team.

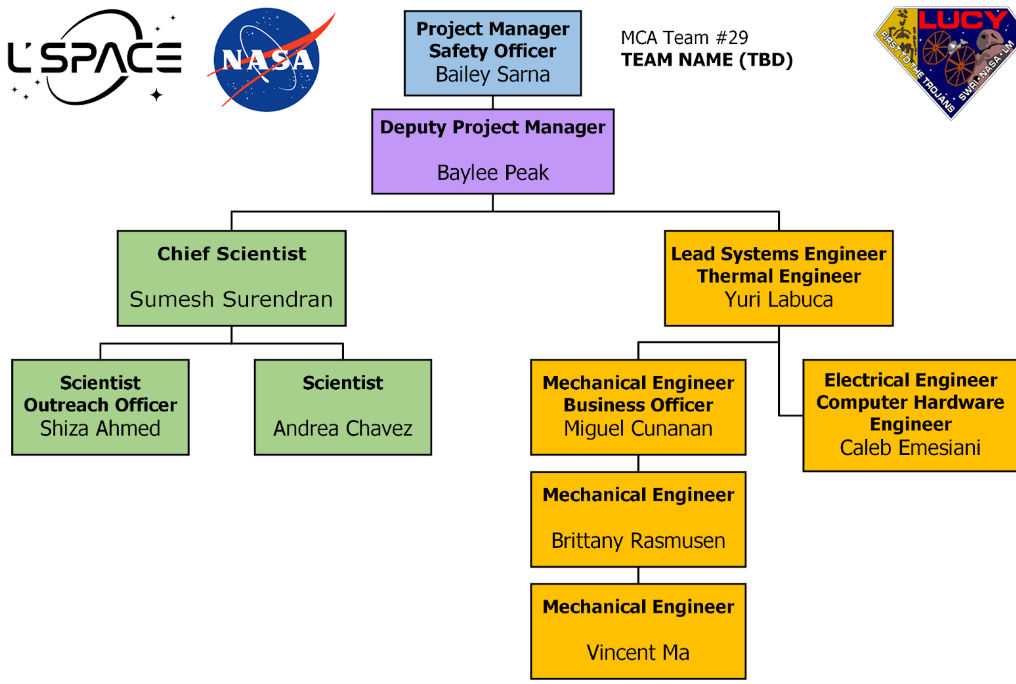


Figure 5.1 Team 29 Organizational Chart

6. Conclusion

This preliminary design review is written to guide our mission, Asteroid Investigation to Defend from Near Earths (AIDENNE) on the spacecraft, Spacecraft for Asteroid Measurement and Understanding Risks Against Inhabitants (SAMURAI). By completing this document, the team has completed the research necessary to advance in the mission and has become familiar with the characteristics of the designated Near-Earth Object.

As the mission progressed, the team reviewed the mission leading L'SPACE, and incorrect oversights were made sure to be improved upon. If there were more time allotted, the team would work on defining the fidelity of our spacecraft even more. In the CDR phase, this would be the same path taken by the team.

References:

[1] Coradini, A., De Sanctis, M. C., Reininger, F., Bonsignori, R., Racca, G., & Chicarro, A. (1996). LUVIMS lunar UV and IR mapping spectrometer. *Planetary and Space Science*, *44*(10), 1099–1119. https://doi.org/10.1016/S0032-0633(96)00004-9

[2] Dawn Spacecraft & Mission Overview – Spacecraft & Satellites. (n.d.). Retrieved from https://spaceflight101.com/spacecraft/dawn-spacecraft-mission-overview/

[3] FC Instrument | Technology. (n.d.). Retrieved from NASA Solar System Exploration website: https://solarsystem.nasa.gov/missions/dawn/technology/fc-instrument/

[4] NASA - Scientific Instruments. (2011). Retrieved from Nasa.gov website: https://www.nasa.gov/mission_pages/dawn/spacecraft/instruments.html

[5] PDS/PPI Home Page. (n.d.). Retrieved December 6, 2022, from pds-ppi.igpp.ucla.edu website: https://pds-ppi.igpp.ucla.edu/mission/Cassini-Huygens/CO/VIMS

[6] PDS: Instrument Information. (n.d.-a). Retrieved December 6, 2022, from pds.nasa.gov website: https://pds.nasa.gov/ds-view/pds/viewInstrumentProfile.jsp?INSTRUMENT_ID=FC2&INSTRUMENT_HOST_ID=DAWN

[7] PDS: Instrument Information. (n.d.-b). Retrieved December 6, 2022, from pds.nasa.gov website: https://pds.nasa.gov/ds-view/pds/viewInstrumentProfile.jsp?INSTRUMENT_ID=GRAND&INSTRUMENT_HOST_ID=DAWN#:~:text=Location%20%3D%3D%3D%3D%3D%3D

[8] PDS: Instrument Information. (n.d.-c). Retrieved December 6, 2022, from pds.nasa.gov website:

https://pds.nasa.gov/ds-view/pds/viewInstrumentProfile.jsp?INSTRUMENT_ID=VIMS&INSTRUMENT_HOST_ID=CO

[9] Sierks, H., Keller, H. U., Jaumann, R., Michalik, H., Behnke, T., Bubenhagen, F., … Richards, M. L. (2011). The Dawn Framing Camera. *Space Science Reviews*, *163*, 263–327. https://doi.org/10.1007/s11214-011-9745-4

[10] Visible and Infrared Mapping Spectrometer (VIMS) | Cassini Orbiter. (n.d.). Retrieved from NASA Solar System Exploration website: https://solarsystem.nasa.gov/missions/cassini/mission/spacecraft/cassini-orbiter/visible-and-infrared-mapping-spectrometer/

[11] Multi-layer insulation. (2022, November 24). Retrieved December 6, 2022, from Wikipedia website: https://en.wikipedia.org/wiki/Multi-layer_insulation

[12] Overview of materials for Epoxy/Carbon Fiber Composite. (n.d.). Retrieved from www.matweb.com website: https://www.matweb.com/search/datasheet_print.aspx?matguid=39e40851fc164b6c9bda29d798bf3726

[13] Kevler Armed Fiber Technical Guide. (n.d.). Retrieved from https://www.dupont.com/content/dam/dupont/amer/us/en/safety/public/documents/en/Kevlar_Technical_Guide_0319.pdf

[14] MatWeb - The Online Materials Information Resource. (n.d.). Retrieved from www.matweb.com website: https://www.matweb.com/search/datasheet.aspx?MatGUID=77b5205f0dcc43bb8cbe6fee7d36cbb5&ckck=1

[15] MatWeb - The Online Materials Information Resource. (n.d.-b). Retrieved

December 6, 2022, from www.matweb.com website:

https://www.matweb.com/search/DataSheet.aspx?MatGUID=b3430ccca1334449b0d59cde9f977b57

[16] Mohammed Bekhti, Messaoud Bensaada. Solar Arrays and Battery Power Sources Conceptual Design for Low Earth Orbit Microsatellites. Engineering and Applied Sciences. Special Issue: Research on Emerging Technologies in Design and Manufacturing.

Vol. 5, No. 3, 2020, pp. 66-70. doi: 10.11648/j.eas.20200503.13

[17] Metaferia, W., Schulte, K. L., Simon, J., Johnston, S., & Ptak, A. J. (2019).

Gallium arsenide solar cells grown at rates exceeding 300 µm h−1 by hydride vapor phase epitaxy. *Nature Communications*, *10*(1), 3361.

https://doi.org/10.1038/s41467-019-11341-3

[18] 2001 Mars Odyssey (MGM). (2019). Retrieved

from Skyrocket.de website: https://space.skyrocket.de/doc_sdat/mars_odyssey_2001.htm

[19] *For the Solar System diagram and Dawn rendering on the cover*. (n.d.). Retrieved

from https://descanso.jpl.nasa.gov/DPSummary/090924dawn-FinalCorrex--update5G.pdf

[20] Astronauts complete deployment of 1st ISS roll-out solar array. (2021, June 21). Retrieved

December 6, 2022, from SpaceFlight Insider website:

https://www.spaceflightinsider.com/missions/iss/astronauts-complete-deployment-of-1st-iss-roll-out-solar-array/

[21] Garcia, M. (2021, January 11). New Solar Arrays to Power NASA's International Space Station Research. Retrieved from NASA website:

https://www.nasa.gov/feature/new-solar-arrays-to-power-nasa-s-international-space-station-research

[22] Hall, L. (2022, March 30). Impact Story: Roll-Out Solar Arrays. Retrieved from NASA website:

https://www.nasa.gov/directorates/spacetech/Rolling_Out_a_Path_to_Future_Space_Travel

[23] What is the Size of a Solar Panel & Weight of a Solar Panel. (2019, November 8). Retrieved from Lets Save Electricity website:

https://letsavelectricity.com/what-is-the-size-of-a-solar-panel-weight-of-a-solar-panel/

[24] IQ Spacecom X-Band Antenna Sheet. (n.d.). Retrieved from

https://catalog.orbitaltransports.com/content/brands/IQ%20spacecom/SmallSat_Catalog_IQ%20spacecom%20X%20Band%20Antenna%20Datasheet%2004.2021.pdf

[25] X-*Band Low Gain Antenna*. (n.d.). Retrieved from

https://www.antenna-giken.co.jp/global/business/satellite/pdf/X-Band%20Low%20Gain%20Antenna.pdf

[26] Reaction Wheel Data Sheet. (n.d.). Retrieved from

https://www.rocketlabusa.com/assets/Uploads/RL-RW-0.06-Data-Sheet.pdf