HEDGE: Hypersonic ReEntry Deployable Glider Experiment

A Technical Report submitted to the Department of Mechanical and Aerospace Engineering

Presented to the Faculty of the School of Engineering and Applied Science

University of Virginia • Charlottesville, Virginia

In Partial Fulfillment of the Requirements for the Degree

Bachelor of Science, School of Engineering

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Spring, 2022

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On my honor as a University Student, I have neither given nor received unauthorized aid on this assignment as defined by the Honor Guidelines for Thesis-Related Assignments

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Technical Report

Introduction

Hypersonic flight occurs at speeds exceeding five times the speed of sound and is an expanding research field in the aerospace industry with military and civil applications. Military applications include hypersonic missiles, both offensive and defensive, and high-speed aircraft. Civil applications include access to space and commercial air travel. A CubeSat is a small satellite flown in low earth orbit that is well suited for undergraduate education. This technical project team will utilize a CubeSat to perform a hypersonic glider flight experiment. These experiments are difficult to replicate in wind tunnels and expensive to achieve on rockets and aircrafts. By using a CubeSat, university students are able to conduct these experiments at a lower cost, and with greater accessibility.

Mission Overview

The purpose of this mission is to assess the feasibility of using CubeSats in hypersonic flight experiments for sustained flight applications. In planning for and designing a CubeSat to satisfy this goal, the Capstone team considered ways to aerodynamically manipulate the traditional CubeSat design to prolong hypersonic flight. The team utilized the Space Mission Engineering process of going from broad mission objectives or a vague concept to an operational mission. As students approached this mission concept, they formulated four objectives which are motivated by a combination of technical and educational considerations.

The design requirements are derived from the objectives of the CubeSat Mission. The objectives are shown below; they focus on the ability of undergraduate students to use CubeSats to conduct hypersonic research in Extremely Low Earth Orbit (150 - 250 km). The fulfillment of these objectives will be used to gauge mission success.

Primary Objectives	
O1	Demonstrate the feasibility of CubeSats as a platform for hypersonic glider flight research.
O2	Demonstrate that undergraduate students can conduct hypersonic glider flight experiments at lower cost and with greater accessibility than traditional programs.
Secondary Objectives	
O3	Provide an opportunity for undergraduates to gain hands-on experience and generate interest in the spaceflight industry.
O4	Collect and transmit sustained flight data.

Table 1: Mission Objectives

The National Aeronautics and Space Administration (NASA) as an independent government facility monitors and protects objects in Space to ensure mission success. Their major directives originate from the intersection of scientific and political interests, federal funding, and the public interest. One of these directives is the Hypersonic Technology Project aimed to research sustaining hypersonic consistency for applications in high-speed propulsion systems, reusable vehicle technologies, and high-temperature material research (Gipson 2021). Through several of their past missions, NASA has dealt with tracking reentry vehicles as well as monitoring smaller debris and spacecraft entering the atmosphere. NASA supports CubeSat projects as a mechanism for low-cost technology research and development to help bridge strategic knowledge gaps between students and industry professionals. Through the CubeSat initiative, NASA assists in attracting and retaining students in STEM disciplines by providing a holistic educational opportunity. The design of the HEDGE CubeSat connects this accessibility to space for students as well as aligning with the hypersonic research goals of NASA. The Department of Defense (DOD) has shown a growing interest in pursuing the development of hypersonic systems. With the high profile advancements in defense systems by competitive nations such as China, the DOD has more political support to research advancements in hypersonic missile production. As a cost-effective method for data collection, this CubeSat can provide hypersonic research to advance defense glider designs. By studying the application of hypersonics within aerodynamic design, the HEDGE project is producing a knowledgeable, innovative workforce for companies such as NASA and the DOD. These objectives align with NASA and DOD mission goals for CubeSat usage.

Mission Architecture & Concept

Based off of the Space Mission Engineering process developed by James Wertz, David Everett and Jeffrey Puschell, the below *Mission Architecture* is a set of elements or components that together form a framework for this hypersonic space mission. The elements of this space mission are detailed below. The following *Mission Concept* is a fundamental statement of how the space mission will work - how data will be collected, how it will be powered, the players and how the mission satisfies the end users.

Mission Architecture

1. Subject
    a. The spacecraft will interact with the Earth's atmosphere in Extreme Low Earth Orbit (ELEO) in a hypersonic environment (Active Subject).
    b. The spacecraft will also track its location, speed, thermal properties and trajectories (Passive Subject).
2. Payload
    a. The payload will include distance, trajectory, velocity and position sensing capabilities, as well as a control board, power unit, and communication system all housed within the rear 1U section of the spacecraft. Depending on future calculations, extra mass may be placed towards the nose of the spacecraft to aid in aerodynamic stability. This extra mass will tentatively be composed of lead.
3. Spacecraft Bus
    a. The spacecraft fuselage fits within an approximate 3U CubeSat volume. From the rear of the spacecraft to the front, the features are as follows:
        i. A 1U section.
        ii. A transition section with length 1.5U.
        iii. A conical section with length 0.5U.
    b. The spacecraft features large, deployable fins hinged on the 1U section where it meets the transition section. The fins start folded over the transition and conical sections of the craft to fill the remaining volume around the front ⅔ of the craft. The fins are hinged to fold back over the rear 1U of the fuselage and will hang

over the dead space behind the fuselage. They are appropriately large to ensure aerodynamic stability. The hinges and fins are composed of material strong and thick enough to withstand the harsh conditions of hypersonic atmospheric reentry.
   c. The spacecraft will fit inside a CubeSat Dispenser (CSD) within the 2nd stage of NG Antares. Lubricated tabs along the bottom of the spacecraft will allow for easy launch from the rocket and limited vibrational loads.
4. Ground Segment
   a. The spacecraft will interact with the Iridium satellite network to track its location and upload its gathered data to a software location which can be accessed by those running the experiment on the ground. The power and control of the spacecraft will be designed to be self contained and self sufficient for the operating lifecycle.
5. Mission Operations
   a. The spacecraft will launch from the NG Antares second stage in extreme low earth orbit (~180-220 km altitude). There will be a minimum 15 minute delay before the fins deploy, in which the power system will activate the sensors and data collection will begin. Extraneous frame pieces will fall away from the spacecraft and will burn up in the atmosphere.
   b. The spacecraft will operate independently during its orbital lifetime in extreme low earth orbit. It will upload gathered sensor data to the Iridium Satellite network which will later be downloaded to the ground and analyzed.
   c. The spacecraft will see a lifetime between 2-7 days following the launch from Antares in which it will collect and transmit data.
   d. Roughly 2-7 days after deployment from NG Antares, the re-entry experiment will begin and will last 15-20 minutes. At the conclusion of the experiment, the spacecraft will burn up completely in the atmosphere as the thermal protection system will be designed to survive for less time than the total possible length of re-entry.
6. Command, Control, and Communications Architecture
   a. The spacecraft will operate independently. It will upload collected data from sensors to the Iridium satellite network; the data will be downloaded to the ground for analysis.
7. Orbit
   a. The vehicle, after launch from the NG Antares second stage, will be in extreme low earth orbit for 2-7 days at hypersonic speeds before burning up completely in the atmosphere.
   b. The fins will deploy shortly after launch. Any extraneous frame pieces will fall away and burn up.
8. Launch Concept
   a. Within the 2nd stage of NG Antares, the spacecraft will reside in a CSD and will launch by sliding off lubricated tabs in the CSD.
   b. After launch, at least 15 minutes will pass before the deployment of the fins and the start of the data collection.

Mission Concept

The end users include the MAE 4700 Spacecraft Engineering students at the University of Virginia, the University of Virginia School of Engineering and Applied Sciences, NASA, Northrop Grumman, and the hypersonic research community.
The HEDGE spacecraft will be constructed such that it will remain intact for the duration of orbit and is aerodynamically stable. It will have sufficient power to last the entirety of the orbital lifetime and will transmit data to the Iridium network for analysis and use by UVA Spacecraft Engineering students and the hypersonic research community. It will deploy from the 2nd stage of NG Antares in the Fall 2024, will last for approximately 2-7 days in extreme low earth orbit and will collect and transmit sensor data. After 2-7 days in extreme low earth orbit, the spacecraft will begin re-entry which will last approximately 15-20 minutes before the vehicle burns up completely in the atmosphere.

Conceptual Design

The Conceptual Design process produces an understanding of the relationships between major system elements. It specifies these relationships, as well as subsystem elements for every subsystem. It delineates the flow-down of high level requirements to each of the subteams. High level requirements are specified by the functional and operational requirements. In this section, these requirements and constraints will be specified and then used to analyze the subsystem operations.
For mission success, the CubeSat must perform within the following functional and operational requirements. In order to achieve and sustain hypersonic flight, the satellite must be able to survive the launch, deploy correctly, and maintain structural integrity in orbit around the Earth; this is dependent on the materials of the CubeSat. These requirements are tabulated below (Table 2).

Functional Requirements	
F1	Hypersonic vehicle fins must automatically deploy
F2	Center of Pressure behind Center of Gravity for aerodynamic stability
F3	Withstand launch and orbit conditions
F4	Sustain M>5 during flight for 15-20 minutes
F5	Second Stage Antares Deployment (compatible with canisterized satellite dispenser)

Table 2: Functional Requirements

Furthermore, operational requirements exist to ensure the CubeSat functions correctly. The satellite must operate for one week, therefore power to the CubeSat must last for one week. At the end of the experiment, the frame and vehicle must be destroyed to limit any space waste per Department of Defense (DOD) and NASA regulations.

Operational Requirements	
OP1	1 week orbital lifetime
OP2	Automated, powered system control and data collection

OP3	Frame and vehicle must burn up in atmosphere

Table 3: Operational Requirements

Limitations to experimentation are imposed by launch provider requirements, tabulated below (Table 4). CubeSats are a subclass of satellites: CubeSat standards to comply with launch provider requirements state that measurements must be at most 3U (10 cm x 10 cm x 30 cm) and approximately 6 kg in weight. Cost of materials must be within the grant budget, and the materials of the CubeSat must be strong enough to maintain the structural integrity of the CubeSat during orbit, but burn up after experimentation (about 1 week of orbital lifetime). Operation constrains the CubeSat to launch on the second stage of the Antares Rocket while complying with FCC (Federal Communications Commission) regulations and Northrop Grumman Launch requirements.

System Constraints		
C1	Comply with CubeSat Regulations for Launch Provider:	1. 3U maximum size 2. Total mass $\lessapprox$ 6 kg
C2	Launch on second stage Antares Rocket:	1. Operational Summer 2023
C3	Cost within amount granted from NASA	(<$100,000)
C4	Material durability and structural integrity of the CubeSat for sustained flight:	1. Survive hypersonic flight 2. Components burn up upon conclusion of reentry experiment
C5	Comply with FCC regulations for space launch	

Table 4: System Constraint

Attitude Determination and Control Systems & Orbits

The Attitude Determination and Control Systems & Orbits (ADACS) subsystem level requirements (Table 5) were chosen in coordination with the system-level mission constraints, functional requirements, and operational requirements. First, the ADACS solution must ensure highly stable flight in and outside the atmosphere. This is the most important requirement, as the hypersonic vehicle must be able to transmit flight data to the Iridium constellation and the experiment would lose validity. A tumbling or unstable vehicle would be harmful to the success of this transmission. Second, the ADACS system must minimize weight to conform to launch provider requirements for on-board deployable spacecraft. Third, the ADACS system should minimize system volume in order to maximize volume for the hypersonic glider experiment. Fourth, the ADACS system should minimize power consumption so as to allow power consumption by other subsystems.

A1[F1]	Highly stable flight inside and outside the atmosphere

A2[C2]	Minimize weight to ensure successful Antares Integration
A3[C1]	Minimum system volume to maximize hypersonic glider volume
A4[OP2]	Minimize power consumption of ADACS system

Table 5: Attitude Determination & Control Systems Requirements

ADACS systems can be either active or passive. Active systems utilize on-board sensors, processors, and actuators to determine and control spacecraft attitude. These systems can be fairly complex and expensive. Passive control systems utilize existing environmental forces, such as gravity, magnetism, and aerodynamics to control the spacecraft attitude without requiring onboard power, sensors, or actuators.

In ELEO, aerodynamic, gravity gradient, and magnetic torques are the primary sources of torque on small spacecraft. At extreme low earth orbit and below, however, aerodynamic torques are much larger in magnitude than any other torque (Rawashdeh & Lumpp, 2013). Moreover, a 1U CubeSat will experience insignificant solar radiation and gravity gradient torques due to its symmetry and low surface area. Then, for the purposes of this mission, understanding the aerodynamic torques is paramount to selecting an ADACS system to accomplish the subsystem requirements, and by proxy, the mission requirements.

Multiple ADACS systems were evaluated for their fulfillment of the subsystem level requirements. First, active systems were evaluated. Common active systems include magnetorquers and reaction wheels. Magnetorquers create a magnetic force via electromagnetics to interact with the Earth's magnetic field to alter the spacecraft attitude. They are relatively lightweight, very precise, and expensive. Reaction wheels utilize the law of conservation of energy to store energy and control the spacecraft attitude. Next, passive systems were evaluated. Passive systems utilize the primary sources of torque on spacecraft (aerodynamic, gravity gradient, and magnetic) to achieve spacecraft attitude control. Utilizing aerodynamics to passively stabilize the spacecraft requires there to be significant aerodynamic force, which is the case at altitudes 500km and below (Rawashdeh & Lumpp, 2013). Gravity gradient stabilization can be achieved by creating a significantly asymmetrical spacecraft, usually achieved by deploying a boom from the side of the spacecraft. Passive magnetic stabilization is achieved by using permanent magnets to properly align the spacecraft with the Earth's magnetic field.

To choose the ADACS system best fulfilling the subsystem level requirements, a decision matrix was created using pugh analysis (Table 6). Metrics were weighted according to how important they were to meeting system constraints and fulfilling subsystem, operational, and functional requirements. For example, power was weighed the most (ADACS subsystem requirement A4), along with cost (System Constraint C3).

	Power	**Stability**	**Simplicity**	**Cost**	**Mass**	**Volume**	**Responsiveness**	**Total**
Category weight	**3**	**2**	**1**	**3**	**2**	**2**	**2**	

Permanent Magnets	+1	+1	0	0	+1	+1	0	+9
Magnetorquer	-1	+1	-1	-1	-1	-1	+1	-7
Reaction Sphere	-1	+1	-1	-1	-1	-1	+1	-7
Reaction Wheels	-1	+1	-1	-1	-1	-1	+1	-7
No ADACS on Bus	+1	0	+1	+1	+1	+1	-1	+9

Table 6: ADACS Candidate Component Trade Off

The decision matrix (Table 6) resulted in the choice between permanent magnets or no ADACS on the CubeSat bus. Both of these ADAC Systems had a final score of "+9" making them the best options when balancing the objectives and constraints of the project. The decision matrix clearly ruled out the active system candidates. Analyzing the potential of the remaining systems (permanent magnets and no ADAC systems), it was decided that no ADACS would be preferred for the hypersonic glider. While the analysis resulted in the same decision matrix score, permanent magnets added unneeded complexity to the attitude control system.

There is precedent for not including an ADACS system: the *2016 Virginia CubeSat Constellation Experiment* did not use an active ADACS system because their satellites did not require directionality (Costulis, 2022). The primary objective of the Constellation Experiment was to measure orbital decay and atmospheric drag while providing undergraduate students with project experience (Costulis, et. al., 2022), which is very similar to the primary objective of this project. While this hypersonic mission does consider pointing and responsiveness a priority, adding permanent magnets to the CubeSat design without fully being able to calculate the side effects makes permanent magnets an undesirable option for ADACS. It was decided that any effects by permanent magnets to orient the CubeSat would be unnecessary when used in tandem with aerodynamic stabilization techniques. No ADACS on the CubeSat bus is the best option: it does not conflict with the power supply needed for other subsystems; it does not take up limited space; and it is a simple, weightless, no-cost option. Moreover, passive magnets would have the effect of damping oscillations of the spacecraft as it orbits the earth. However, data collection is necessary only when in atmospheric flight, at which point the spacecraft will be both statically and dynamically stabilized by virtue of the external structure of the vehicle.

Communications

Subsystem Level Requirements

CM1[OP3]	Be able to transmit data to a satellite constellation
CM2[OP3]	Automated data collection

CM3[OP3, OP5]	In re-entry, collect 4 measurements a second and send data every minute
CM4[OP3, OP5]	In orbit, collect and send 6 measurements every hour
CM5[OP5]	Transceiver and antenna compatible with satellite modem
CM6[C5]	Compliant with FCC and federal regulations (need to apply for radio license to operate radio in space)

Table 7: Communications Subsystem Level Requirements

Subsystem level requirements for the communication subsystem were determined based on the team's system operational requirements and system constraints. These requirements are necessary to the success of the communication subsystem and thus the overall success of the mission. Requirements one through four pertain to data transmission during the mission and specify how often automated data will be transmitted during orbit and reentry. The last two requirements consist of selecting a radio and antenna compatible with the satellite modem and meeting FCC and federal regulations. The table above depicts the communications subsystem requirements as described including supporting system operational requirements and constraints considered when finalizing each subsystem requirement.

Satellite Network

Four candidate major communications networks were under consideration: Iridium, IsatData Pro, GlobalStar, and the MC3 Network. Iridium, IsatData Pro, and GlobalStar are all satellite constellations while the MC3 Network is a radio-based ground station network. A trade study was performed to compare their frequency band, coverage, transmitting and receiving capabilities, price range, compatibility with the mission and data rate. Each of these categories were given a weight from one to three based on their importance to the mission. Each communications network was given a score between one and four for each category. The scores for each category were multiplied by the category's weight and added together for each communications network (See Appendix C).

The MC3 Network ranked lowest due to its low coverage and compatibility with the mission. Since the reentry position of the test article will not be known, relying on ground stations scattered mainly through the United States is not ideal. GlobalStar ranked third due to its compatible radios' low data rates and lack of ability to receive data. IsatData Pro ranked second due to its better price range and receiving capabilities but low data rate. Iridium ranked first due to its high coverage, compatibility with the mission, and transmitting and receiving capabilities (NASA, 2021; Satphonestore, n.d. a-d; Minelli et al., 2019; CubeSatShop, 2019; CubeSatShop, 2021; Globalstar, n.d.).

Radio

Four Iridium transceivers were considered: NAL 9602-LP, Iridium Core 9523, Iridium 9603, and Iridium 9602. They were compared based on their weight, dimensions, data rate, power draw, price, and compatibility with CubeSats. Each category was given a weight and the total scores were added up for each radio (See Appendix D).

NAL 9602-LP scored the lowest due to its large weight and size and low compatibility with CubeSats. Iridium Core 9523 scored third due to its high weight, size, power draw and price. Iridium 9602 scored second due to its high compatibility with CubeSats and low price, yet high power draw and weight. Iridium 9603 scored the highest due to its low weight, size, and price, and high compatibility with CubeSats. Iridium 9603 has a data rate of 17 bytes per second. Assuming measurements are single precision (4 bytes each), the radio can send 4 measurements a second meeting requirement CM3 (Riot et al., 2021; Satphonestore, n.d. a-c; Satellite Phone Store, n.d.).

Antenna

Three antennas were compared: Nooelec Iridium Patch Antenna, NAL Research flat mount antenna, and Taoglas Iridium patch antenna. The three candidate antennas were all patch antennas due to needing a compact design to circumvent the possibility of burnup during reentry. Each antenna was selected to be compatible with Iridum. The categories compared were weight, dimensions, gain, price, operating temperature, and compatibility with CubeSats. Each category was given a weight and the total scores were added up for each radio (See Appendix E).

The NAL Research flat mount antenna scored lowest based on low ratings in its weight, dimensions, and compatibility with CubeSats since it has no CubeSat flight heritage (SYN7391, 2021). The second lowest scoring was the Nooelec Iridium patch antenna due to low ratings in its weight, dimensions, gain, and price (1620 MHz, n.d.). The highest scoring and final selected antenna was the Taoglas Iridium patch antenna scoring high in all categories except gain (Home, n.d.).

Concept of Operations

The team constructed a diagram showing the concept of operations for the communications subsystem process to help facilitate the understanding of its part in the overall mission as seen in Figure 1. The spacecraft bus is shown as a gray rectangle. Its deployment into the hypersonic glider structure in LEO is also shown. This glider is then shown to transfer information to the Iridium satellite constellation as depicted by the red line connections to the satellite constellation. This data will then be sent to one of Iridium's ground stations, depicted by the gray circular dots, following the final data transfer to the University of Virginia. A subsystem elemental flow chart as seen in Figure 2 was also created to depict the internal process of data between communication components.

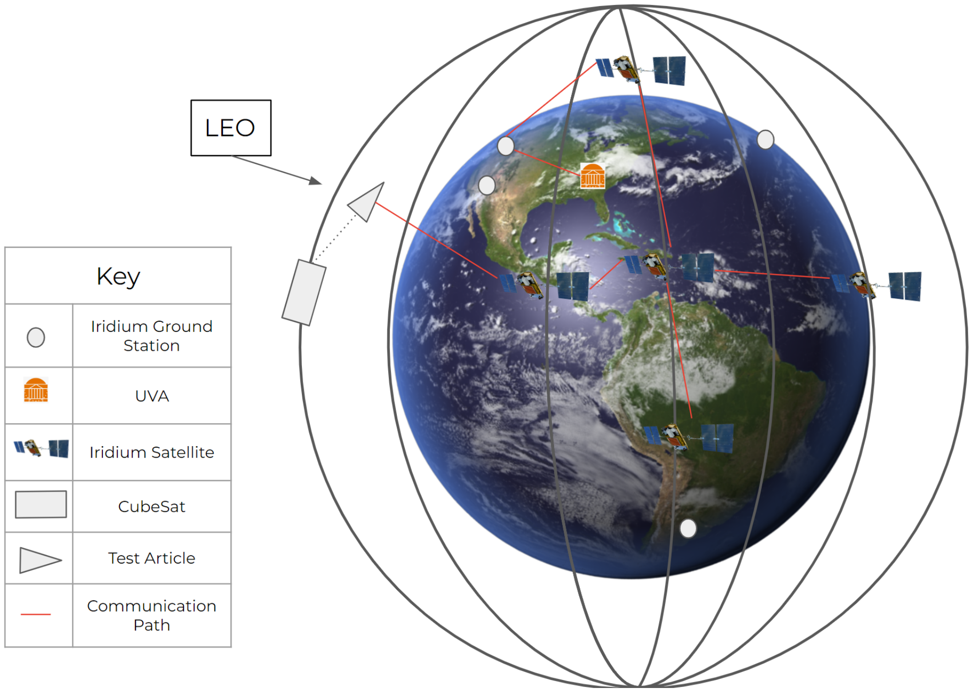


Figure 1: Communications Concept of Operations

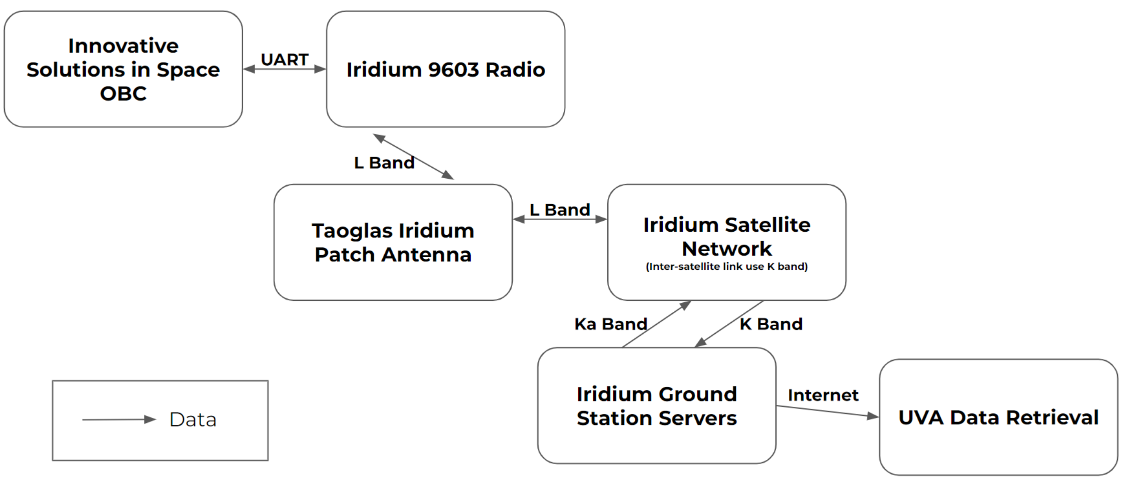


Figure 2: Communications Subsystem Flow Chart (ICAO, 2006)

Licensing Process

An experimental radio frequency license must be obtained from the FCC in order to operate the radio communications system in space. Since the FCC requires a minimum of 90 days after receiving an application to issue a license, the team will start this process as soon as possible. As stated in NASA's CubeSat Launch Initiative document, "Early, but incomplete license submissions do not confer any benefits" (NASA, 2017). A fully completed application is needed to receive a license and the FCC will reach out with requests for any missing information. Since these queries for more information further delay the time to receive a license, the application will be submitted once all the necessary information is available.

In a research paper analyzing the capability of using Iridium to communicate with a CubeSat in Low Earth Orbit, Riot et al. lay out the process they went through to obtain an experimental radio frequency license. They state that they reached out to their Iridium reseller, from whom they purchased their transceiver and service plan, and let them know they wanted to operate their radio in space. In turn, their reseller reached out to Iridium directly, who started the FCC licensing process for them. Once they provided Iridium with the needed information, Iridium completed the application and sent it to the FCC for approval (Riot et al., 2021).

Our team also reached out directly to Iridium and they replied with the same steps that Riot et al. took. Thus, our team's plan is to reach out to the Iridium reseller used to purchase the transceiver and service plan to start the licensing process. We expect this process to begin in late August to early September of 2022, giving the FCC well over six months to review the application and issue a license.

In the event that Iridium will not carry out the licensing process, the team will work with the FCC directly. The team has spoken to the FCC directly about the application process. The mission will require four pieces of information in order to receive a license: a Special Temporary Authority (STA) form, satellite orbital debris mitigation compliance document, International Telecommunication Union (ITU) Cost Recovery letter, and SpaceCap notice. The STA form is required for missions lasting less than six months. It is filled out on the FCC's Experimental Licensing System (ELS) website (FCC OET, n.d.). The orbital debris mitigation compliance is documented in an Orbital Debris Assessment Report (ODAR). It assures that the CubeSat will not be a hazard to other satellites, will deorbit in a reasonable amount of time, and completely burn up upon reentry (NASA, 2017). The ODAR will be prepared consistent with NASA standards in order to meet the FCC's requirements. NASA's Orbital Debris Program Office (ODPO) website provides their Debris Assessment Software, handbooks, and standards for debris assessment which will be used to construct the report (FCC, 2013). The ITU Cost Recovery letter indicates we are aware that we are responsible for processing fees charged by the ITU for satellite network filings (See Appendix A for letter template). SpaceCap is a software used to capture data about transmitting stations in space needed for license approval. It can be downloaded from the ITU's website and once it is run, it creates a data file needed in the license application. In addition to the SpaceCap software file, an ITU SpaceCap cover letter is needed (see Appendix B for letter template) (FCC OET, n.d.).

Software & Avionics

Subsystem Level Requirements

The software and avionics subsystem level requirements are listed below in Table 8. They were developed to comply with the system level function and operational requirements as well as the system level constraints. Most of the software and avionics requirements focus on accomplishing operational requirement OP2 which refers to automated system control and data collection. Choosing components with a proven flight heritage is a priority to ensure survival for the duration of the data collection period. Additionally, processing power is a key consideration to ensure valuable data is properly and efficiently collected.

SA1[OP2, F3]	Radiation hardened/tolerant electronics with flight heritage (operate in extreme low earth orbit)
SA2[OP2, C4]	Must be able to operate in a reentry environment (under load, high vibration)
SA3[OP2]	Single flight computer to control data processing and tasking
SA4[OP2]	Ram speed and SSD must be fast enough to process collected data (have necessary processing power to read/store collected data)

SA5[OP2]	Be able to process 1000 bytes of data in a minute

Table 8: Software & Avionics Subsystem Level Requirements

On-Board Computer Selection

The on-board computer (OBC) was selected based on seven key characteristics: mass, dimensions, interfaces, power draw, processor clock rate, price, and proven flight heritage (see Appendix F). From the technical interchange meeting, the selection of OBCs was reduced to two candidates: the EnduroSat and Innovative Solutions in Space (ISIS). Performing a side by side comparison of the selected OBCs revealed a smaller volume favoring the ISIS OBC with 107,136mm^2 total volume. The ISIS model also had a slight edge in mass coming in at 100g versus 130g for the EnduroSat. In regards to processing power, the ISIS model had a newer processor with 400 MHz of raw processing power versus 216 MHz for the EnduroSat. Available interfaces built into the EnduroSat exceeded those in ISIS in terms of quantity and practicality. Lastly, the price reflected was a significant concern. The EnduroSat OBC required SDK licensing to integrate an operating system, adding an additional $6,100 to the base price of $4,300. Optimal price is reflected in the $5,600 ISIS base price as there were no additional requirements essential to operability.

Weighted significance on a scale of 1-3 was applied to the seven key characteristics. From there, the two models up for final selection were subject to scores ranging from -1 to +1. A score of -1 represents a worse than baseline, 0 represents an "at baseline" (no disadvantages/advantages), and +1 represents a better than baseline rating. The baseline rating for the weights originate from a predetermined baseline model discussed in the technical interchange meeting. Weighted totals for the ISIS OBC and EnduroSat were 10 and 2 respectively. The ISIS model scored higher than baseline ratings for all categories except for power draw and price–scoring at baseline ratings for these areas (see Appendix F).

Thermocouples Selection

Thermocouple candidates were selected based on processor temperature range, termination type, cable insulation, sensor application, thermocouple type, and price. Candidate selections were the high temperature Inconel (candidate 1) and the bolt-on with washer (candidate 2). Processor temperature ranges were 0 to 980°C and 0 to 482°C for candidate 1 and 2 respectively. Candidate 1 features a standard connector type and an Inconel overbraid whereas candidate 2 features a stripped lead connector type with fiberglass insulation. The only considerable differences were the price for each thermocouple; candidate 1 came in at $65/unit and candidate 2 was $13/unit. Final thermocouple selection utilized the same weight rubric used in selecting the OBC. Weighted results were close (see Appendix G), coming down to the processor temperature range, termination type, and price. Results favored the bolt-on with washer thermocouple to the Inconel thermocouple–falling short by 2 points. The selected thermocouple will be paired with the MAX6675 voltage converter to ensure compatibility with the ISIS OBC.

Pressure Transducers Selection

Two main candidates were researched for selection of the pressure transducers: the Kulite XCE-80, and the OMEGA PX409 Series standard Pressure Transducers. For each model, six main characteristics were considered: measurement pressure range, digital communication, operation temperature, power requirement, weight, and price. Additionally, to comply with the mission's objective, it was decided an external pressure transducer is essential to examine the change in pressure during hypersonic flight in the external environment; both of these models are external pressure transducers. Examining the six main characteristics, the Kulite XCE-80 possessed the best digital communication, operational temperature, power requirement, and weight. Therefore, the Kulite XCE-80 was chosen for being the most optimal and efficient model for our mission's objective. The weighted results may be found in Appendix H.

In terms of technical characteristics, the Kulite XCE-80 posses dimensions of 6.4x2x2 mm and 0.41x25.4 for the pressure reference tube, a measurement pressure range of 0.35 to 70 bars, an operation temperature of -67 °F to 525 °F, a power requirement of 10 to 12 VDC/AC, a weight of 4 grams, and a cost ranging from $300 to 500.

In terms of structural integrity within the CubeSat, five transducers will be placed in different locations: one for each face of the CubeSat's front section, and one on the back face of the structure. The front side transducers will aid in determining the positioning of the spacecraft, such as if it is in a straight position or if it is tumbling. Both the sides and back transducers will aid in determining the change in pressure between the front and the back. For the signal conversion, the pressure transducer will be connected to a custom printed circuit board (PCB); this way, the pressure transducer's voltage output will be read and converted into a digital signal, which will be sent to the motherboard.

Flight Software Recommendations

The two possible flight softwares are NASA's Flight System (cFS) Framework and Kubos' cloud-based mission control system. Access to Github files, an online platform used for collaboration on software projects, for both softwares are available. Notable features of NASA's cFS include configurable parameters, free downloads, formalized software reuse, and it was developed by the Goddard Space Flight Center. Alternatively, the Kubos flight software is cloud-based which allows for automatic updates and remote access. While not free, depending on the chosen subscription plan the Kubos flight software also has access to an expanding ground station network. Between the two software options, NASA's cFS is the most cost effective but as the mission requirements evolve both are viable options.

Subsystem Flow Chart

The overall subsystem flow chart for the avionics and software components can be seen in Figure 1. The respective connection ports to the OBC are shown on the arrows connecting each component to it. The GOMSpace battery and the EPS board will be connected through the I2C interface. Both the thermocouples and the pressure transducers will be connected to the battery power. The thermocouples will connect to the sensor amplifier using the SPI ports on the OBC, up to a maximum of 6 thermocouples. The pressure transducers will connect to the digital convertor through wire leads. Both signal converters will be connected to the OBC, to transmit the data. The Patch Antenna has an indirect connection to the OBC through the Iridium 9603 transceiver, connected using the UART interface.

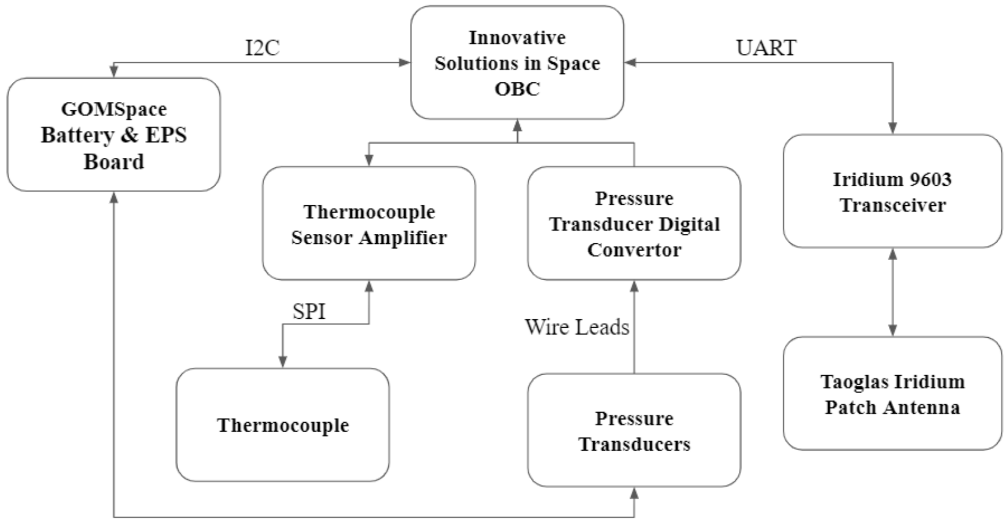


Figure 3: Software & Avionics Components Flow Chart

Power, Thermal & Environment

Subsystem Requirements

The Power/Thermal subsystem level requirements are listed below in Table 9. They were developed to comply with the system level functional and operational requirements as well as the system level constraints.

P1[F2]	Add redundant switches to outside of CubeSat for activation of test article at deployment
P2[OP1]	Sufficient voltage/current to supply all electronic subsystems according to a power schedule: peak power during atmosphere reentry data transmission, minimal power pre glider deployment periodic data transmission
P3[OP1]	Battery life lasts duration of launch, mission (~ 7 days) without recharging and maintains charge throughout pre-launch standby time (~ 4 months)
P4[OP2]	Ensure no materials used in the construction outgas/deteriorate under space conditions
P5[OP2]	Ensure thermal shielding materials survive fluctuating high/low temperatures
P6[OP2]	Do not exceed thermal tolerances of electronics, materials, and structures during orbital or reentry temperature phases
P7[OP2, C5, C6]	Ensure no large debris survives reentry and is instead broken up by aerodynamic and thermal stresses after hypersonic phase
P8[C1, C2]	Ensure that equipment (power source, shielding, wiring) conforms to CubeSat size/weight standards
P9[C3]	Ensure that subsystem equipment does not exceed budget limitations

Table 9: Power, Thermal & Environment Subsystem Level Requirements

Power System

It was decided that it was not necessary to include photovoltaic panels in the final design, due to the short duration of the mission. A total of four EPS candidates were considered: The Clydespace Starbuck Nano, the ISIS iEPS power system, the GOMSpace P31U with the BPX battery, and a consumer-grade EPS sourced from drone parts. These EPS candidates were evaluated based on cost, reliability, capacity, bus compatibility, mass, and geometry. Each category was given a weight and the total scores were added up for each EPS candidate (Appendix I).

The GoMSpace EPS ended up scoring the highest, mainly because it surpassed the ClydeSpace in maximum power capacity options (75 Wh vs 50 Wh). The ISIS candidate suffered due to a low power capacity and high mass relative to the other candidates, while the consumer-grade EPS faced serious concerns regarding reliability. The final EPS selection comes in at a quoted price of $15,700. A power flow chart of the spacecraft is shown below in Figure 4.

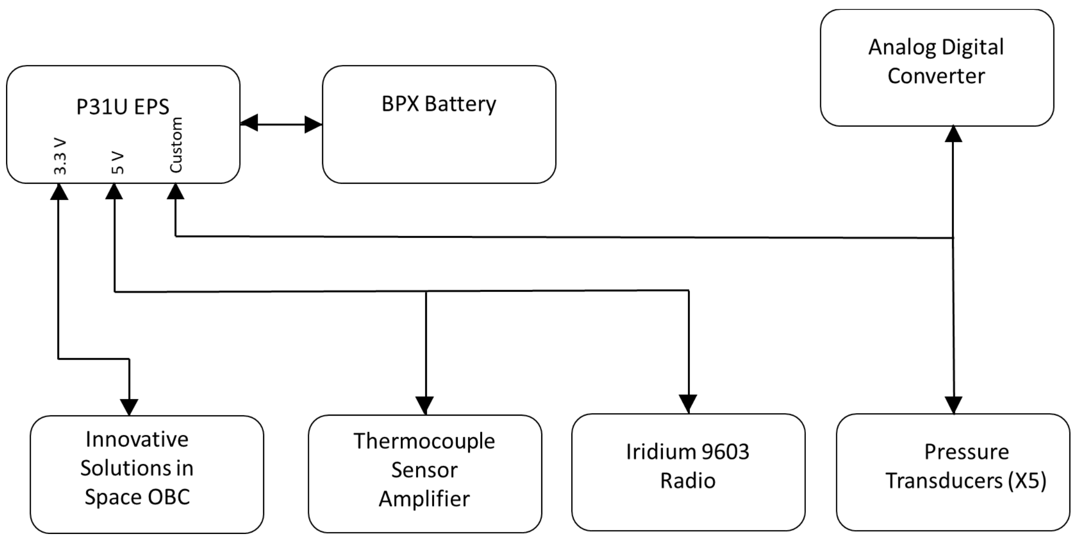


Figure 4. Power Flow Chart

In order to calculate the estimated power budget (Table 10), it was first necessary to determine the power available. This was done by dividing the 75 Wh capacity of the chosen EPS by the maximum mission duration in hours, resulting in an estimated power available of 0.45 W. Next, the average power consumption of each component was calculated by first approximating the percentage of the mission time a component will be in either a low, high, or sleep state. These duty cycle percentages were multiplied by their respective power draws, and summed to get an average power consumption for each component. Finally, the power utilization margins were calculated, and it was found that the spacecraft exceeded the 10% design convention. The resulting information was put into a table, which can be found below. Based on data from the manufacturer, battery capacity will only decline by 2.14 Wh over a period of 4 months, well within the margin.

		Power Consumed (W)			**Duty Cycle (%)**			**Power Required (W)**	**% Power Utilized**
Subsystem	**Component**	**Sleep**	**Low**	**High**	**Sleep**	**Low**	**High**		
Communications	Radio	0	0.18	0.7	70	25	5	0.08	17.8
Software/Avionics	Computer	0	0	0.4	70	0	30	0.12	26.7
Power/Thermal	Battery & EPS	0	0	0.16	0	0	100	0.16	35.6

Instrumentation	Pressure Transducers	0	0	0.25	95	0	5	0.0125	2.8
	Thermocouple Amplifier	0	0	0.25	95	0	5	0.0125	2.8
	A/D Converter	0	0	0.25	95	0	5	0.0125	2.8
Margin								0.0525	11.7
Total								0.45	100

Table 10: Power Budget

Thermal Protection System

The design of a thermal protection system within the requirements of this project was challenging in that the protection was required to survive some, but not all, of a reentry environment, along with surviving an outer space environment. The final design incorporated a passive thermal control system, with high temperature material at the nose of the glider and ablative materials coating the rest of the glider (See Figure 5). The high temperature material is placed to maintain the aerodynamic shape of the glider for as long as possible, as well as protecting the glider's leading edge as it experiences the highest temperature due to aerodynamic heating loads. In locations at which shape is of less importance, ablative materials provide sufficient thermal protection and help ensure prevention of complete reentry survival.

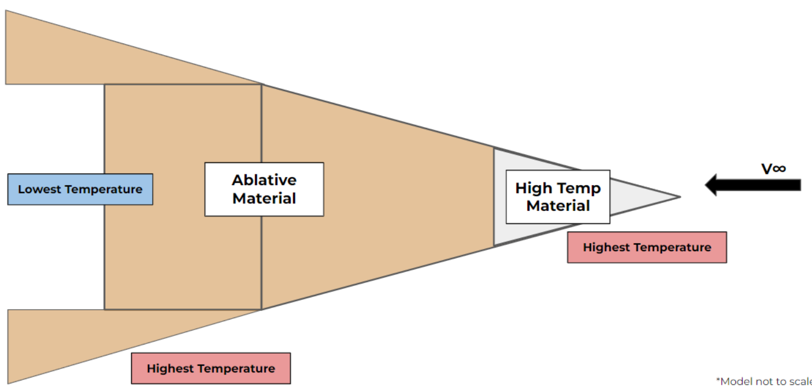


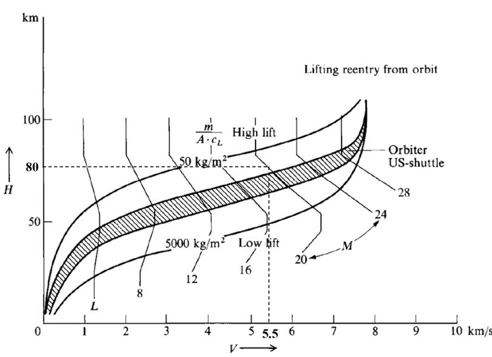
Figure 5: Thermal Protection System Overall Layout

Selection of a high temperature material began with three candidates: Silicon Carbide, Inconel Alloy, and Niobium Alloy, based on a conversation with NASA's Dr. David Glass, who is experienced in designing thermal protection systems for aerospace applications. Further analysis was conducted on all three materials using ANSYS Granta Materials Database. Initially, Silicon Carbide was ruled out as a candidate based on its maximum service temperature due to the possibility of it surviving reentry, failing a design requirement despite possessing superior density qualities (See Appendix C, Figure 1).

Inconel and Niobium Alloys were compared based on categories including price, density, maximum service temperature, and thermal conductivity (See Appendix C, Figures 2 & 3). Although Inconel Alloys are superior in both cost and density, Niobium Alloys possess advantageous thermal qualities including a lower maximum service temperature to ensure burnup, and higher thermal conductivity to protect the test article's interior electronics. When ablating, Inconel Alloys also leave behind a protective oxide layer as they ablate, offering further thermal protection, while Niobium cleanly sublimates as it ablates. Finally, Niobium Alloy possesses substantial flight heritage, having been tested on the Space Shuttle's outer skin as a primary passive thermal protection system, along with a plethora of other modern aerospace applications. Ultimately, Niobium Alloy was selected from the candidate materials.

Ablative material candidates consisted of Teflon and cork phenolic, at the recommendations of Dr. Glass and the Qarman CubeSat-based reentry experiment, respectively. This comparison was somewhat difficult due to an overall lack of information on cork phenolic, preventing an analysis similar to that of the high temperature materials. Ultimately, Teflon was selected due to its well-documented ablative properties, ease of procurement, and manufacturability. Unlike cork phenolic, Teflon also ablates more cleanly and does not leave behind a secondary protective layer, ensuring easier simulation of ablation and highest likelihood of burnup during reentry.

Knowing the materials of the TPS, its dimensions could be determined using aerothermal conditions of reentry. The modeling of atmospheric reentry trajectory is a complex task, and creating a model to predict the trajectory of the test article was determined to be outside the scope of this experiment. Given that reentry models created in the past by the Air Force and NASA are difficult to access it was decided that for a conceptual design, using published figures of typical reentry for lifting bodies from LEO would be sufficient to estimate hypersonic flight conditions (Jameson 2006; Sanson, 2019).



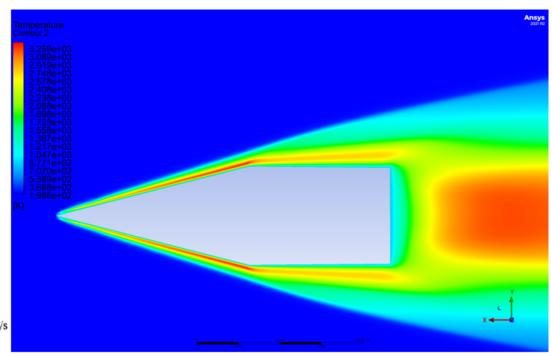


Figure 6: Velocity vs. Altitude for LEO Reentry (Sanson, 2019); Figure 7: Static Temperature Contour at 80km

Using the known aerodynamic parameters of the test article and Figure 6, the experiment was taken to begin at 80km and the associated velocity in the figure was coupled with US Standard Atmosphere conditions to perform a steady-state, 2D, axisymmetric Fluent simulation of reentry with simplified geometry. Figure 7 above shows the resultant temperatures of the test article resulting from the simulation. Actual heating conditions will exceed those at 80km as the object continues its descent, and so the simulation instills confidence that it will survive long enough to perform the experiment but certainly burn up during reentry. For validation in addition to numerical convergence (Appendix K), standard oblique shock relations were compared to the simulated airflow and found to resemble the simulation closely.

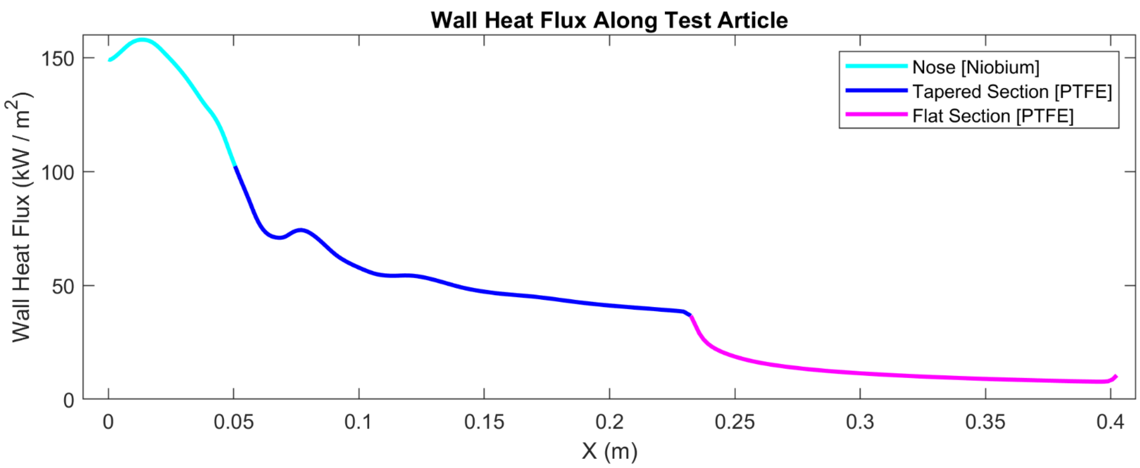


Figure 8: Wall Heat Flux vs. X-Position on Test Article

Finally, using published experimental ablation rates of Teflon and the average heat flux over the two Teflon sections depicted in Figure 8 of 36.17kW/m², the required mass of Teflon in order to allow vehicle survival for 10 minutes at 80km was calculated as 1.39kg (Galfetti, 2003). To preserve aerodynamics and manufacturability the Teflon was assumed to be applied evenly across the ablative surface area of the test article, yielding a constant thickness of 4.5mm (Appendix L).

Structures & Integration

Requirements and Constraints

Functional and operational requirements and constraints for the structures and integration subsystem can be seen in Table 11. These requirements and constraints stem from and are designed to meet the mission objectives as a whole.

S1[OP2]	CubeSat must withstand aerodynamic and launch forces
S2[OP2]	Hypersonic test vehicle must withstand hypersonic environment/flight for sufficient time (~15 min)
S3[OP2]	Frame must deploy and burnup in atmosphere, hypersonic vehicle must burnup in atmosphere after conclusion of flight
S4[C1]	House all communication equipment inside test vehicle
S5[C1]	House all deployment mechanisms within max 3U CubeSat (10x10x30 cm)
S6[C2]	Entire system must weigh less than 2 kg per U
S7[C3]	CubeSat and test vehicle system must be easily assembled without specialized tools
S8[C5]	CubeSat must be compatible with CubeSat Canisterized Satellite Dispenser (CSD)

Table 11: Structures & Integration Subsystem Level Requirements

Design

The proposed system consists of a hypersonic test vehicle to be deployed from a Canisterized Satellite Dispenser (CSD). To best house and integrate subsystem components while maintaining aerodynamically stable flight, the test vehicle will have a 1U rear section, a 1.5U transition section ending with a 0.5U conical nose. The rear, 1U section is large enough to house all the power, communications, and control systems of standard CubeSat component size. Large fins will be attached to the vehicle at the beginning of this transition on all flat sides of the 1U. While

housed within the CSD, these fins will be hinged forward to occupy the entire 3U space around the nose, stabilizing the test vehicle during launch and acting as its frame. As compatible with the CSD, preloaded tabs will be added along the length of the bottom fins as part of their stabilization plates. Other than the bottom fin having additional width through the tabs, the system will be symmetrical along its z-axis. The hinges and brackets utilized to mount these fins will be sufficiently sized to maintain structural stability throughout launch and deployment. To minimize protruding volumes and ensure smoothness of flow around the vehicle, the hinges will additionally be countersunk. Images of the complete test vehicle enclosed during launch and once fins are deployed back can be seen in Figure 9. Upon deployment, the vehicle will remain in its ‘closed’ position for a minimum of 15 minutes. After the quiet period, the fins will fold back over the 1U section and over the dead space behind to aerodynamically stabilize the vehicle in its orbit (see Figure 9). When deployed, the fins may be locked in the open position via magnets or latches. The entire vehicle, after 2-7 days in extreme low earth orbit conducting its aerodynamic experiment in reentry, will burn up completely in the atmosphere.
The component integration has been approximated using rough dimensions of subsystem components and arranging them within the fuselage. See Figure 10 below. As designed for each internal component, four rods will be inserted through their corners to allow them to rigidly stack in place within the fuselage.

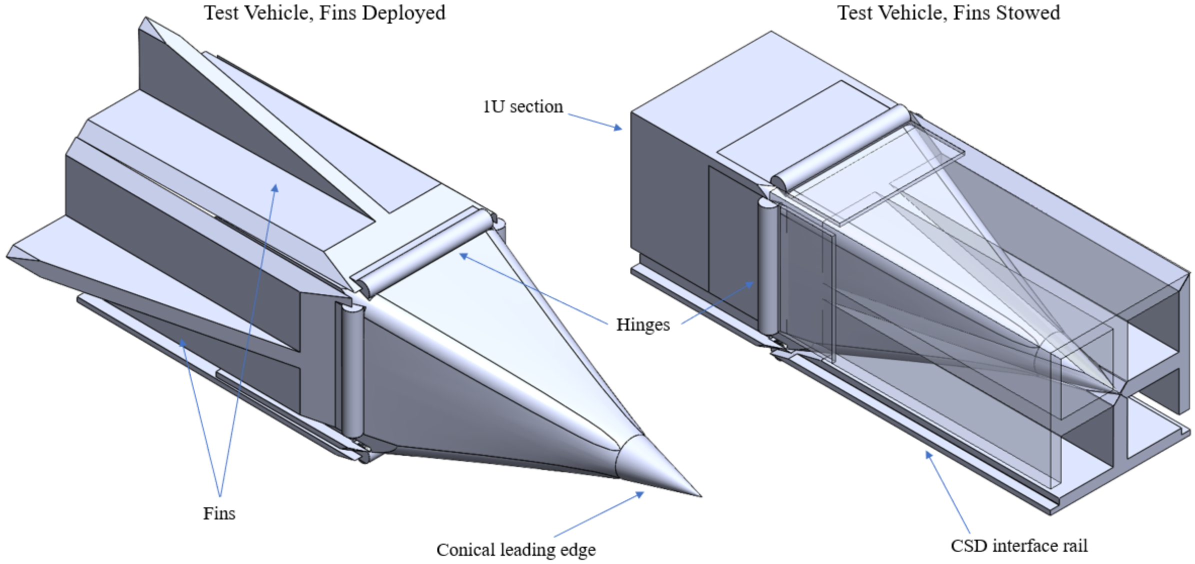


Figure 9: 3D Model of CubeSat with fins stow deployed and stowed

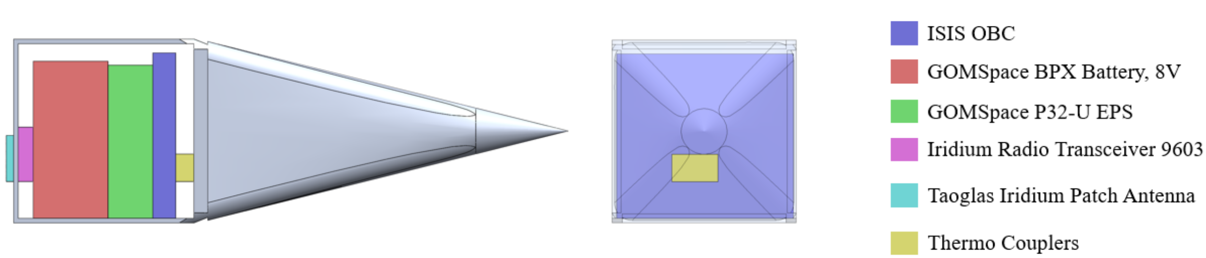


Figure 10: Model of internal assembly

Justification

The full 1U rear section of the fuselage was implemented to house the other subsystem components, namely the on board computer which has a 1U cross sectional area. The rest of the fuselage includes a transition section leading to a conical nose - elements present to decrease drag - lowering added heat and increasing aerodynamic stability. The large fins are present both to fully constrain the front 2U (transition and cone) within the CSD when hinged forward and to bring the vehicle's center of pressure behind the center of gravity, thereby increasing the stability of the spacecraft. The hinges are large and wide to cope with the high levels of heat present during hypersonic atmospheric reentry, and the fins' flat, perpendicular plates are present to give the fins a platform to rest on the rear 1U section. Finally, the hinges are countersunk into the 1U section to decrease drag and turbulent flow around the fins and the rear of the fuselage.

Mass Budget

The contribution from the structures components to the overall mission mass budgets can be seen in Table 12. The mass contributions from the test vehicle fuselage, nose cone, aft fins, and connecting hardware total to an estimated 5952.1 g. Barrowman equations were used to estimate the centers of gravity and pressure to ensure aerodynamic stability in flight, which is presumed if the center of gravity is 30% forward of the center of pressure. The center of gravity was calculated at 23.5 cm from the nose, and the center of pressure was calculated at 28 cm. Resulting in a static margin of 4.5 cm.

Below in Table 12, the mass contributions from each subsystem, including the structures components are listed. There is a contingency mass of 1000 g listed to account for errors in manufacturing, approximation or future changes in components or materials. The predicted total mass without the system contingency is approximately 4952.1 g or 82.53% of the preliminary maximum total mass, and with that system contingency, the total is 5952.1 g or 99.20% of the maximum.

Component	Estimated Mass (g)	Estimated Mass Percentage of Total (%)
Radio	11.4	0.23
Antenna	10.0	0.20
On Board Computer	100	2.02
Pressure Transducers & PCB	25	0.51
Thermocouples & Signal Converter	20	0.41
Thermal Protection System - Teflon	1420	28.8
Thermal Protection System - Niobium	28.9	0.59
Battery & EPS	600	12.2
Test Vehicle	1730	35.1
Total Allocated	3933.9	79.7
System Contingency	1000	20.2
Preliminary Total System Mass	4933.9	100.0
Theoretical Maximum System Mass	6000	121.6

Table 12: Mass Budget

Manufacturing

The aft fins will be milled from 6061 Aluminum for precision and structural strength and will be coated with a thermal protection coating to account for their receiving high total energy during reentry. The hinges will be made of Inconel 718 to account for their receiving high total energy and high force during fin deployment. The front 2U of the fuselage will be milled out of solid 6061 Aluminum for structural strength and precision and mechanically fastened to a commercially available 1U rear structure. The front 2U of the spacecraft will also have a thermal protection coating.

Concerns

Central structural concerns lie with the hinges and the fins, as they will be receiving a high level of total energy from the harsh conditions of hypersonic atmospheric reentry. The design of the spacecraft presents concerns about bringing the center of pressure rear of the center of mass for the purpose of aerodynamic stability. The fins may have to be hinged at a higher angle (i.e. not resting against the 1U rear section) to bring the center of gravity forward. If it is determined that the fins need to be hinged at a greater angle, the center of pressure will be brought towards the nose of the vehicle, further complicating the optimization of aerodynamic stability in the design. Brackets would be required if the fins are at an angle and the strength and thermal resistance of those brackets will need to be high enough to withstand the environment.

Future Work

Future work will be conducted to bring this conceptual design to fruition. Computational models will be created to simulate loads during launch and throughout deployment and flight to ensure aerodynamic stability and structural integrity. Confirmation of the overall system's aerodynamic characteristics will also be completed at this time. Physical test rigs will also be utilized to confirm these models. From these results, additional refinements and adaptations will be made to the structures and integration design to ensure requirements are met in the most efficient manner possible. Manufacturing details will also continue to be finalized. For example, the entire fuselage may be milled out of a single Aluminum piece rather than having a commercially available 1U rear section attached to a front 2U milled piece to decrease costs at future students' discretion.

Team Management

For execution of the technical project, the Capstone team is composed of a combination of undergraduate aerospace and mechanical engineering majors. For this project, each team consists of 15 members. Within the group, subsystem teams were formed to allow students to focus on specific aspects of the CubeSat design. These subsystem teams were Program Management, Attitude Determinations and Control Systems & Orbits (ADACS), Communications, Software & Avionics, Power, Thermal & Environment, and Structures & Integrations. Program management consists of one capstone leader. The rest of the teams consist of two to three team members. Students contributed to the overall conceptual design of the system with specialized attention to their subsystem requirements. The 2021-2022 team assignments were designated as seen in Table 13 below.

Functional Teams	**Members**
Project Management	Emma Jensen
Communications	Parker Johnson
	Samantha Castro
Software and Avionics	Jashianette Fournier Jaiman
	Cristina Rodriguez
	Ryan Jansen
Power, Thermal, and Environment	Michael Fogarty
	Adam Obedin
	Josh Willoughby
Attitude Determination and Control Systems (ADACS) and Orbits	Jonathan Cummins
	Eva Paleo
	Brendan Angelotti
Structures and Integration	Margaret Che
	Nicholas Lu
	Desmond DeVille

Table 13: Team Member Roles

Capstone has recurring scheduled meetings Mondays and Wednesdays from 2 to 3:15. Along with this scheduled class time, individuals contributed to team research and project development through the school year. To begin, students considered the Space Mission Engineering process to proceed with project schedule design. This process allowed students to determine the mission concept and architecture as well as define subsystem level requirements. This culminated in an end of the semester presentation where subsystems further developed the CubeSat initial design. Students also submitted a written prospectus to accompany the presentation. For the Spring 2022 semester, students worked toward a Conceptual Design Review (CoDR). The CoDR was formatted into a presentation given to the Capstone advisor and industry professionals. For this, the presentation incorporated the Mission Definition Review (MDR), and the Mission Concept Review (MCR) and System Requirements Review (SRR). As students considered specific design choices and candidate components, the Capstone group held meetings with industry officials such as the Virginia Space officials who work with the Antares Launch. The CoDR presented specific candidates chosen for subsystem teams. Alongside the Conceptual Design Review, students wrote a technical paper of the Capstone’s progress for submission as the project advances to the Preliminary Design Review phase in the 2022-2023 year.

Schedule

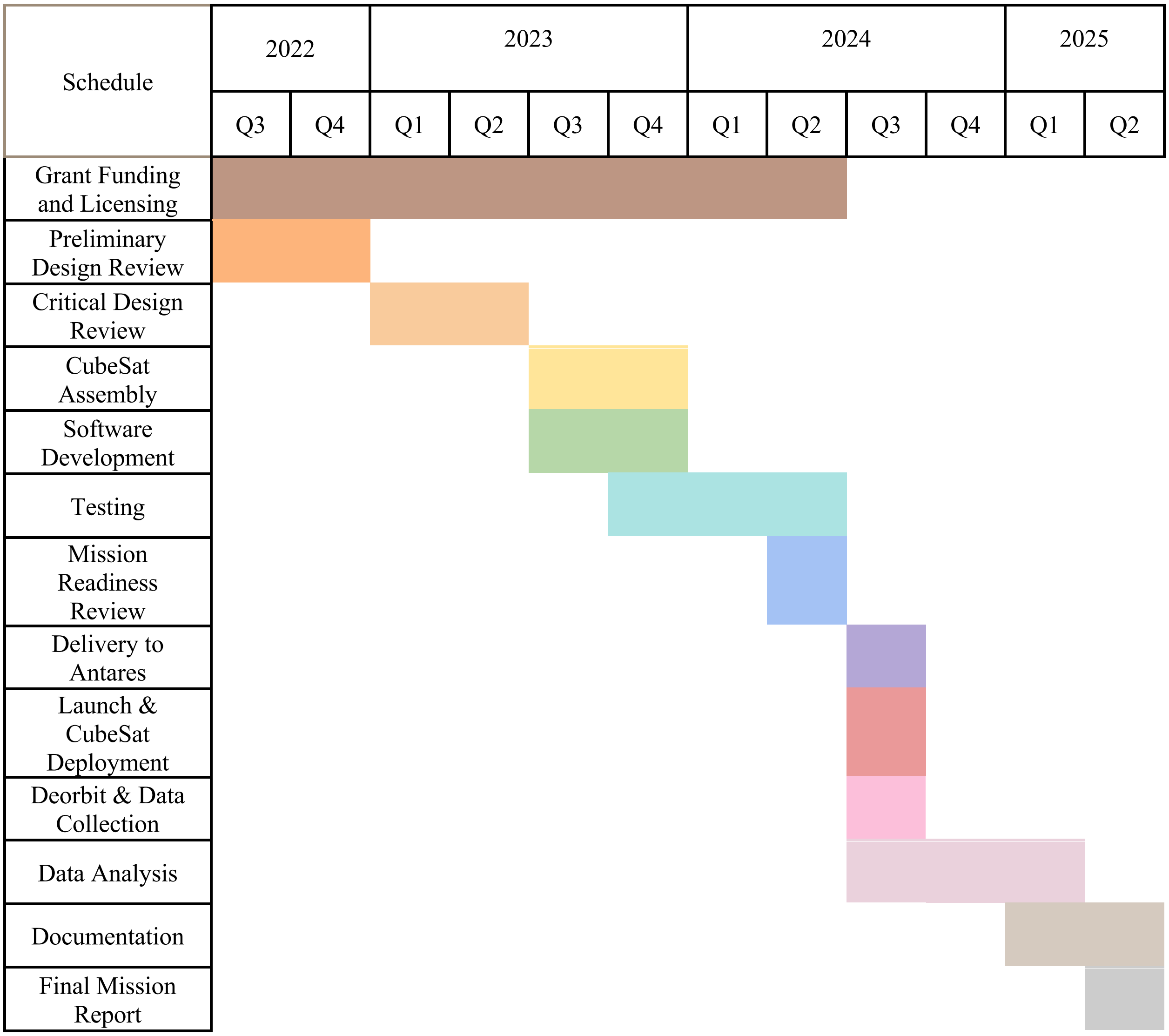


Figure 11: Schedule for Continuation of Project

After approval for the continuation of the project, a new class of 30 fourth year undergraduates in aerospace and mechanical will enter the course in the 2022-2023 school year. They will follow the format of the current class as it advances through the schedule shown in Figure 11, with a scheduled meeting time to ensure availability as well as additional meetings and research time added as needed. Functional teams will be adjusted to accommodate the changing needs of the team, including the formation of a Fabrication & Testing team. Additionally, in the Spring semester a selection of third year engineering undergrads will be able to join the class as an independent study to keep mission continuity as the project enters the launch phase for Summer of 2023.

The next cohort of undergrads will be entering the Preliminary Design Review Stage in Fall 2022. Initially, students will continue to apply for FCC licensing and begin applying for grants to fund the CubeSat project. After the Critical Design Review (CDR) is completed, students will enter the Production and Deployment phase. Using the lab and machining equipment available to engineering undergraduates, students will complete CubeSat assembly in Fall 2023. The CubeSat will undergo safety and environmental testing to ensure it passes the Safety Data Review by MARS, NASA & Northrop Grumman. These will be done to determine if the CubeSat complies with functional and operational requirements. A final Mission Readiness Review will be completed with partnered organizations. Finally in Fall 2024, the CubeSat will remain at the launch site for approximately one month until launch off the Second Stage of the Antares. Orbit will be 2 to 7 days before reentry. Main data collection is taken during the ~20 minutes of reentry and transmitted to the Iridium satellite.

Cost

The preliminary cost for the HEDGE project based on subsystem design requirements is $67,722 (Table 14). This total cost will account for resources to build two CubeSats. The doubled quantity of resources will account for potential broken or malfunctioned parts that may appear in the manufacturing process. Additionally, if two complete CubeSats are able to be built, one will be designated for CubeSat testing.

Team	Components	Quantity	Estimated Cost per Unit	Estimated Cost
ADACS	None	-	$0	$0
Communications	Iridium 9603 Transceiver	2	$199.00	$398.00
	Taoglas Iridium Patch Antenna	2	$8.21	$16.42
	Service Plan Activation Fee	2	$40.00	$80.00
	Data Cost (Ground Testing)	2	$203.94	$407.88
	Data Cost (Orbit and Reentry)	2	$45.48	$90.96
Software & Avionics	Innovative Solutions in Space (ISIS) On-Board Computer	2	$7,145	$14,290
	Bolt-On Thermocouples with SS Washer Housing	10	$13	$130
	Thermocouple Sensors	10	$10	$100
	Kulite XCE-80 Pressure Transducers	10	$500	$5,000
Power, Thermal, & Environment	EPS (including board and battery)	2	$15,700	$31,400
	Thermal Protection Systems (Teflon and Niobium Alloy) and Application	2	$1,000	$2,000
Structures & Integration	Material, Construction tools	2	$1,904.32	$3,808.64
	Miscellaneous Materials and Supplies	1	$10,000	$10,000
Total Cost				$67,722

Table 14: Estimated Cost Budget for CubeSat

Attitude Determination and Control Systems & Orbits

As seen in the system trade off in Table 6, the CubeSat design will contain no ADACS system and instead rely on the aerodynamic structural design to orient the CubeSat during orbit. For this reason, there is no cost associated with this subsystem.

Communications

The communications subsystem components cost an estimated \$496.63. The Iridium 9603 Transceiver costs around \$199.00, and the Taoglas Iridium Patch Antenna costs around \$8.21 (Satphonestore, n.d.c; Home, n.d.). There is a \$40 activation fee for the service plan as well as a \$0.05 mailbox check fee. The mailbox check will need to be performed once per hour during mission operations yielding a \$1.20 cost per day. The SBD 30 service plan costs \$33.99 per month and provides 30 KB of data. Every KB of data extra costs \$1.09. Ground testing will last around six months and a maximum of 30 KB of data will be sent each month totaling \$203.94. The CubeSat will be in orbit for about 7 days and six data measurements (24 Bytes) will be sent every hour totaling 4.032 KB of data. The reentry phase will last around 30 minutes and four measurements (16 Bytes) will be collected every second totaling 28.8 KB of data. This means that during the month of the experiment, 2.832 KB of data will go over the monthly limit yielding an extra \$3.09 (Satphonestore, n.d.c).

Software & Avionics

The software and avionics components cost an estimated \$8,711. The Innovative Solutions in Space OBC and its daughter board costs around \$7145. The selected thermocouple and signal converters will cost \$66. Lastly, the pressure transducers will cost an estimated \$1500.

Power, Thermal, & Environment

The EPS components including the GOMSpace P31U and BPX cost an estimated \$15,700. The materials for the thermal protection system, including Teflon and Niobium alloy components are estimated to cost approximately \$500 based on required amount and cost per unit of mass (ANSYS Granta Materials Database). Labor and machining costs for application of the protection system are also estimated to be about \$500.

Structures and Integration

The structures and integration materials and components including manufacturing fees are estimated to cost \$1896.05 (Appendix M). The 1U commercially available CubeSat from PumpkinSpace is estimated to cost \$1215. The raw 6061 Aluminum to CNC mill the front 2U transition and cone sections of the fuselage, from Grainger, costs \$166.25 and manufacturing can be done in house. The 6061Aluminum used to mill the fins is estimated to cost \$514.80 and the Inconel used for the hinges is estimated to cost \$8.27 (Pumpkin, n.d.; Grainger, n.d.; Scrapregister, n.d.). Additionally, a miscellaneous fund of \$10,000 is reserved for various tools, screws, and extra needed resources not addressed in the preliminary cost.

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Appendices

Appendix A

ITU Cost Recovery Letter Template (FCC OET, n.d.)

[**DATE**]

Secretary
Office of the Secretary
Federal Communications Commission
445 Twelfth Street, S.W.
Washington, D.C. 20554

Subject: ITU Cost Recovery Fees for the [**Satellite Name**] network
Reference: FCC experimental application file number [________]

Dear FCC Secretary,

The [**APPLICANT]**, proposed operator of the subject network, is aware that as a result of actions taken at the International Telecommunication Union's 1998 Plenipotentiary Conference, and modified by the ITU Council in 2001, 2002 and 2004, processing fees will now be charged by the ITU for satellite network filings. As a consequence, Commission applicants are responsible for any and all fees charged by the ITU. The applicant hereby states that it is aware of this requirement and accepts responsibility to pay any Cost Recovery fees associated with these applications. Invoices for such fees should be sent to the point of contact specified below:

[POC name
APPLICANT
Address
E-Mail:
Phone:]

Sincerely,

/s/

Appendix B

ITU SpaceCap Cover Letter Template (FCC OET, n.d.)

FEDERAL COMMUNICATIONS COMMISSION
INTERNATIONAL BUREAU
WASHINGTON, D.C. 20554

fax: +1 202 418 1208; TWX: 710 822 0160
e-mail: IBMAIL@FCC.GOV

IN REPLY REFER TO:

800C2/SEB15109

Atila Matas, Head of SPR
Radiocommunication Bureau
International Telecommunication Union
Place des Nations
CH-1211 Geneva 20

Subject: Advance Publication Information for the BisonSat Satellite network.

Dear Sir:

In accordance with the provisions of Article 9.1 and Appendix 4 of the Radio Regulations, the Administration of the United States is submitting a request for the Advance Publication Information of the BisonSat satellite network.

Enclosed is a diskette containing an electronic copy using Space Capture V.7 of the information for the BisonSat satellite network.

In addition we would like to provide the cost recovery billing information under Agency code 528 in the Preface to the International Frequency List (IFL) for U.S.A. commercial operators. Cost-recovery invoices associated with this filing should be sent to the following point of contact:

POC name
Salish Kootenai College
Address
E-Mail:
Phone:
Fax:

Technical questions regarding this submission should be directed to:

POC name
Licensee/Applicant Organization
Address
E-Mail:
Phone:
Fax:

We request that a courtesy copy of all correspondence be sent to our Administration also.

Please acknowledge receipt of this information.

Thank you for your kind consideration in this matter.

Sincerely,

Chief, Strategic Analysis and Negotiations Division
International Bureau

Enclosure

If replying by fax, reply to 1 202 418 1208 (preferred) or 1 202 418 0398 (alternate)

Appendix C

Communications Subsystem Candidate Major Communications Networks

	Iridium	IsatData Pro Orbcomm/ Inmarsat	GlobalStar	MC3 Network
Type	Satellite Constellation	Satellite Constellation	Satellite Constellation	Ground Station Network
Frequency Band	L	L	L	VHF,UHF,S
Coverage	100% Global Coverage	90% Coverage (extreme polar regions excluded)	80% Coverage (extreme polar regions and some mid ocean regions excluded)	10% Coverage (only 8 locations across U.S. territory)
TX/RX	TX/RX	TX/RX	TX	TX/RX
Price Range	$404.90* - $1295.90*	≅$744.95*	$2376.56 - $7469.19	$7350.01
Compatibility	High	Medium	High	Low
Data rate	≅17-22 Bytes / second	≅7 Bytes / second	8 Bytes / second 600 Kbytes / day Max	125,000 Bytes / second

Table 15: Candidate Major Communications Network Data Table

	Weight	Iridium	IsatData Pro Orbcomm/Inmarsat	GlobalStar	MC3 Network
Frequency Band	**1**	2	2	2	3*
Coverage	**3**	4	3.5	3	1
TX/RX	**3**	4	4	2	4
Price Range	**1**	3.5	3.5	2	1
Compatibility	**2**	4	3	4	1
Data rate	**3**	2	1	1	4
	Weighted Results	43.5	37	30	33

Table 16: Candidate Major Communications Network Trade Study

Appendix D
Communications Subsystem Radios

	NAL 9602-LP	Iridium Core 9523 L-Band Transceiver	Iridium 9603 Transceiver	Iridium 9602
Weight	136 g	32 g	11.4 g	30 g
Dimensions	69x55x24 mm	70.44x36.04x14.6 mm	31.5x29.6x8.1 mm	41x45x13 mm
Data Rate	~17 Bytes/s	~22.2 Bytes/s	~17 Bytes/s	~17 Bytes/s
Power Draw	Idle 0.000325 W	Idle 0.322 W Transmit 1.38 W Receive 0.506 W	Idle 0.17 W Transmit 0.725 W Receive 0.195 W	Idle 0.175 W Transmit 0.7 W Receive 0.2 W
Price	$983.00	$1,250.00	$199.00	$213
Compatibility	Low-Medium	High	High	High

Table 17: Radios Data Table

	Weight	NAL 9602-LP	Iridium Core 9523 L-Band Transceiver	Iridium 9603 Transceiver	Iridium 9602
Weight	**3**	1	2.5	4	2.5
Dimensions	**3**	2	2	4	3
Data Rate	**3**	3	4	3	3
Power Draw	**2**	4	1.5	2	2
Price	**1**	2.5	2	4	4
Compatibility	**2**	2	4	4	4
	Weighted Results	32.5	38.5	49	41.5

Table 18: Radios Trade Study

Appendix E

Communications Subsystem Antennas

	Nooelec Iridium Patch Antenna	NAL Research Flat Mount Antenna	Taoglas Iridium Patch Antenna
Weight	25 g	31.18 g	10 g
Dimensions	82 mm x 80 mm x 15 mm	39.12 mm x 39.12 mm x 9.39 mm	25.1 mm x 25.1 mm x 4 mm
Gain	~3.1 dBi	4.9 dBi	2 dBi
Price	$29.95	$288.00	$8.21
Operating Temperature	-40℃ to +85℃	-40℃ to +85℃	-40℃ to +85℃
Compatibility	Medium	Medium - Low	High

Table 19: Antennas Data Table

	Weight	Nooelec Iridium Patch Antenna	NAL Research Flat Mount Antenna	Taoglas Iridium Patch Antenna
Weight	**3**	2	1	4
Dimensions	**3**	1	2.5	4
Gain	**1**	2.5	4	1
Price	**1**	2	1	4
Operating Temperature	**2**	3	3	3
Compatibility	**2**	3	1	4
	Weighted Results	25.5	23.5	43

Table 20: Antennas Trade Study

Appendix F
OBC Candidate Selection

Flight Computer	EnduroSat On-Board Computer	Innovative Solutions in Space (ISIS) OBC
Flight Heritage	Yes	Yes
Dimensions (mm)	93.9 x 89 x 23.1	96 x 90 x 12.4
Mass (g)	130	100
Power Draw (W)	*Unspecified	0.4 typical
Processor & Clock Rate	ARM Cortex M7, up to 216 MHz	ARM9 processor, up to 400 MHz 32-bit
Memory/Storage	512 kB RAM, 2 MB program memory, 2x MicroSD slots	64MB SDRAM, 1MB NOR Flash, 2x2 GB high reliability SD Cards for fail safe data storage
Interfaces	4x RS-485, 2x RS-422, 3x UART, 2x I2C, SPI, USB	SPI , I2C, 2x UART (RS-232+RS-232/RS-485/RS-422), LEDS and UART, USB, Image Sensor
Price	$4300 w/ existing SDK license $10400 w/o existing SDK license	$5600 w/o EM Daughter Board $7145 w/ EM Daughter Board

Table 21: OBC Candidate Comparison

	Weighted Importance	EnduroSat On-Board Computer	Innovative Solutions in Space OBC
Mass	3	0	1
Dimensions	2	0	1
Interfaces	1	1	1
Power Draw	2	0	0
Processor & Clock Rate	1	1	1
Price	2	-1	0
Flight Heritage	2	1	1
Weighted Results		2	10

Table 22: OBC Weighted Selection

Appendix G
Thermocouple Candidate Selection

Thermocouple	High Temperature Inconel Overbraided Silica Fiber Insulated Thermocouples	Bolt-On Thermocouple with SS Washer Housing
Process Temperature Range	0 to 980°C (32 to 1800°F)	0 to 482°C (32 to 900°F)
Termination Type	Standard Connector	Stripped Leads
Cable Insulation	Nextel with Inconel overbraid	Fiberglass
Sensor Application	Bolt-on	Bolt-on
Thermocouple Type	Type-K	Type-K
Price	$65	$13

Table 23: Thermocouple Candidate Comparison

	Weighted Importance	High Temperature Inconel Overbraided Silica Fiber Insulated Thermocouples	Bolt-On Thermocouple with SS Washer Housing
Process Temperature Range	**3**	1	0
Termination Type	**2**	-1	1
Cable Insulation	**1**	1	0
Sensor Application	**2**	1	1
Thermocouple Type	**2**	1	1
Price	**1**	-1	1
Weighted Results		5	7

Table 24: Thermocouple Weighted Selection

Appendix H

Pressure Transducer Candidate Selection

Pressure Transducer	Omega PX409 Series Standard Pressure Transducers	Kulite XCE-80
Measurement Pressure Range	Low Pressure: 10 inH2O Standard Ranges: 5 to 5000 psi Metric Ranges: 25 mbar to 345 bars	0.35 to 70 Bar 5 to 1000 Psi
Digital Communication	Cable, Mini-DIN, Twist-Lock, M12 Connector	4 Leads 36 AWG 36" Long
Operation Temperature	-45 to 121 °C (-49 to 250 °F) or -45 to 115 °C (-49 to 240 °F)	-67°F to 525°F
Power Requirement (x5)	10 to 30 VDC, 10 mA	10 to 12 VDC/AC
Weight (g)	115	4
Price	$741	$300-$500

Table 25: Pressure Transducer Candidate Comparison

	Weighted Importance	Omega PX409 Series Standard Pressure Transducers	Kulite XCE-80
Measurement Pressure Range	**3**	1	1
Digital Communication	**2**	-1	1
Operation Temperature	**1**	0	1
Power Requirement	**2**	0	1
Weight	**2**	-1	1
Price	**2**	-1	0
Weighted Results		-3	10

Table 26: Pressure Transducer Weighted Selection

Appendix I

EPS Candidate Selection

	Cost		Reliability		Power Capacity		Bus Compatibility		Mass		Geometry		Total
Category Weight	2	W eig	5	W eig	5	W eig	1	W eig	2	W eig	3	W eig	
ClydeSpac e	0	0	0	0	0	0	0	0	0	0	0	0	0
ISIS	+1	+2	0	0	-1	-5	0	0	+1	+2	0	0	-1
GOMSpac e	0	0	0	0	+1	+5	0	0	-1	-2	0	0	+3
Consumer Grade	+1	+2	-1	-5	-1	-5	-1	-1	+1	+2	+1	+3	-4

Table 27. EPS Candidate Selection

Appendix J

Thermal Protection Systems Materials Comparison Graphs

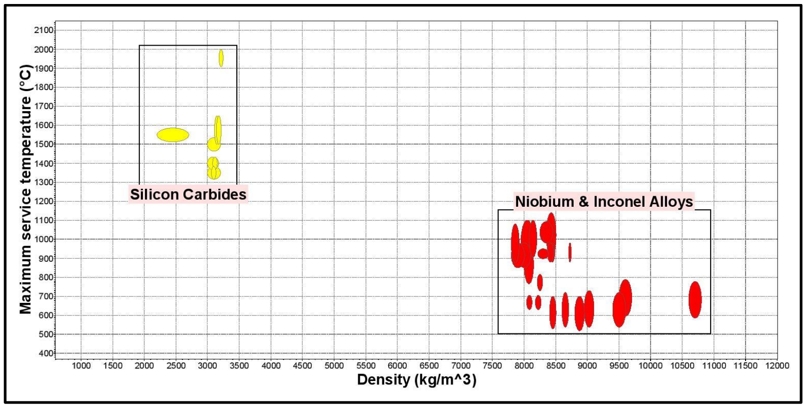


Figure 11: Maximum Service Temperature and Density Comparison Between Candidates

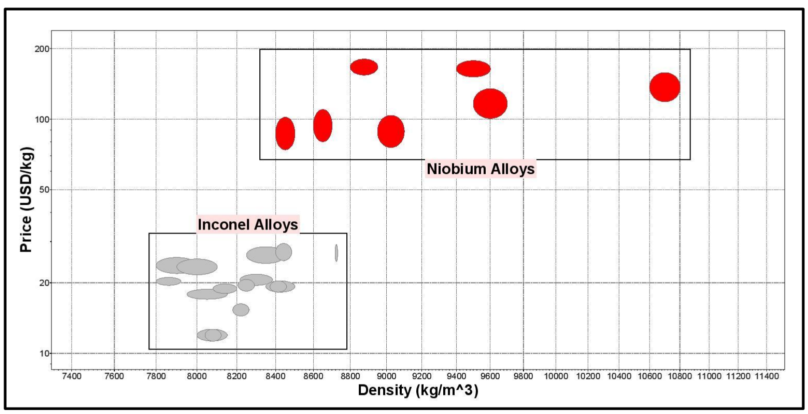


Figure 12: Inconel and Niobium Alloy Price and Density Comparison

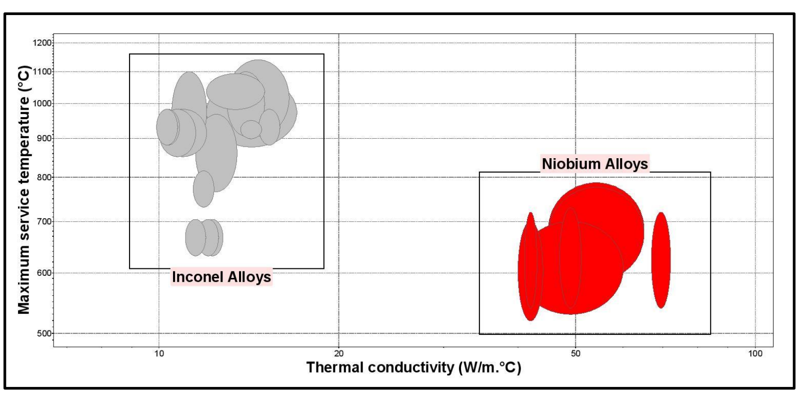


Figure 13: Inconel and Niobium Alloy Maximum Service Temperature and Thermal Conductivity Comparison

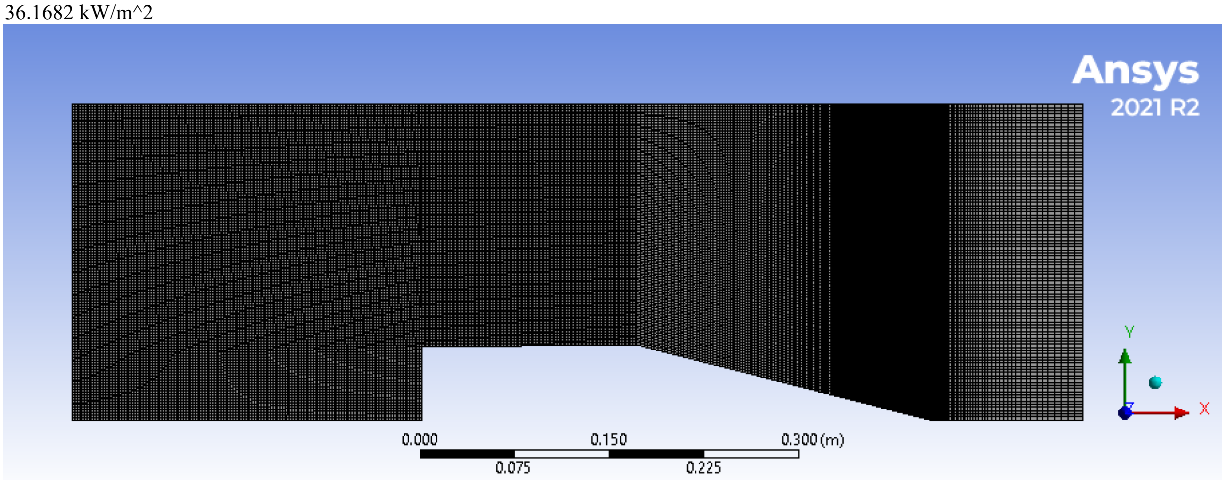
Appendix K
Fluent Meshing and Results

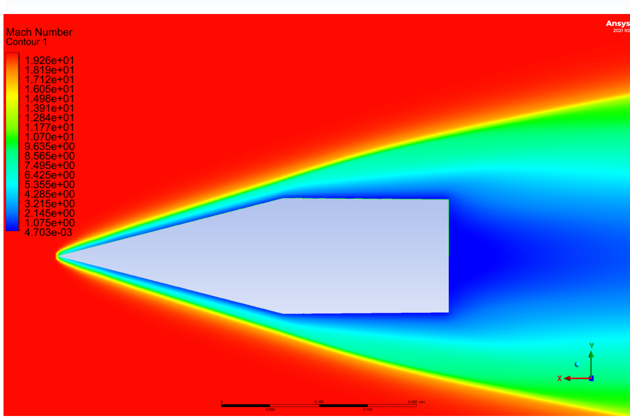
BOUNDARY CONDITIONS

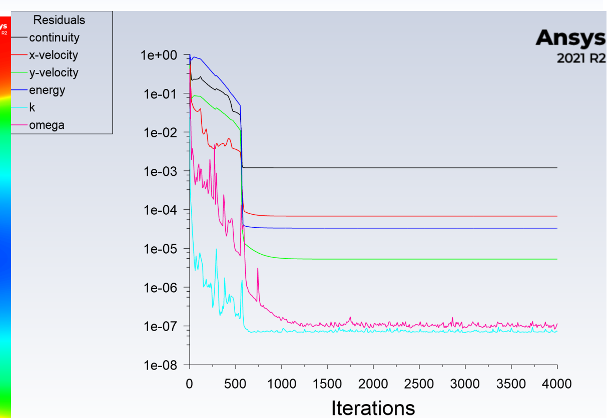
Farfield: 196.65K, 0.88Pa, 20.0 Mach
Standard atmospheric conditions at 80km, Mach for corresponding velocity of typical reentry by a lifting body as per Sanson
Axisymmetric
Wall: 811K
Wall temp taken as a boundary condition so Fluent can predict heat flux itself; used low-end temperature for PTFE ablation

Resultant Avg. Heat Flux on Ablative Surfaces:
avg_flux_ptfe =

36.1682 kW/m^2







Appendix L

Teflon Thickness Calculation (Matlab Code)

```
clc
clear

flux = xlsread("Wall Heat Flux.xlsx", "B33:B333");
flux = flux .* -1 ./ 1000;
% Read in x, which starts with x = 0 at aft edge by default
% Transform x s.t. the leading edge is where x = 0 by subtracting length
x = (xlsread("Wall Heat Flux.xlsx", "A33:A333") - 0.402392477) * -1;
x_nose = x(209:size(x));
flux_nose = flux(209:size(x));
x_tapered = x(101:209);
flux_tapered = flux(101:209);
x_flat = x(1:101);
flux_flat = flux(1:101);

plot(x_nose, flux_nose, 'LineWidth', 2, 'Color', 'cyan');
hold on
plot(x_tapered, flux_tapered, 'LineWidth', 2, 'Color', 'blue');
plot(x_flat, flux_flat, 'LineWidth', 2, 'Color', 'magenta');
xlabel('X (m)');
ylabel('Wall Heat Flux (kW / m^2)');
title('Wall Heat Flux Along Test Article');
xlim([-0.01 0.41])
% Get the average heat flux, neglecting the first 5cm (leading edge)
avg_flux_ptfe = mean(flux(1:209))
legend('Nose [Niobium]','Tapered Section [PTFE]', 'Flat Section [PTFE]')
hold off


Q_capacity = 1.61558; % kJ / g, average ablation rate of teflon (Galfetti, 2003)
time = 10; % minutes, the desired length of the experiment
a = 103524 * 1000^-2;
% m^2, area of ablative portion of theoretical axisymmetric simulated model
% (flat (conical) and tapered (cylindrical) sections)
m_tot = avg_flux_ptfe * a * (time * 60) / Q_capacity;
% g, mass of teflon to dissipate heating via ablation for 'time' minutes


% Calculate thickness assuming that the teflon was just being applied to
% a flat plate since it's a fairly thin layer; neglect that in reality for
% this conical shape the outer surface area grows slightly as teflon is
% applied and r of the cone increases

rho = 2200 * 1000; % g / m^3, density of teflon
a_ablative = 139923.68 * 1000^-2;
% m^2, actual ablative surface area (from solidworks model)
t = m_tot / rho / a_ablative * 1000; % mm, thickness of teflon ablative layer
```

Appendix M

Structures and Integration Mass Cost Breakdown

Component	Material	Estimated Mass (g)	Estimated Cost ($)	Source
Test Vehicle Fuselage - 1U Rear Structure	Al 5052-H32	135.7	$1215	Pumpkin Space 1U SolidWall Chassis Walls
Test Vehicle Fuselage - 2U Conical Front & CSD Deployment Tabs	Al 6061	1620	$166.25	GRAINGER APPROVED Aluminum, Flat Bar Stock, Thickness (Decimal) 4.0 in, Width and Length 5 in x 12 in - 1ZDE4|1ZDE4
Aft Fins	Al 6061	1685	$128.70 x4 = $514.80	GRAINGER APPROVED Aluminum, Flat Bar Stock, Thickness (Decimal) 3.0 in, Width and Length 4 in x 12 in - 2HGL9|61F3X4-12
Hinges	Inconel 718	983.04	$8.27	Inconel 718 Prices in West Coast | 2022 March 18UNITED STATESNickel & Alloy Scrap Price

Table 28. Structures and Integration Mass and Cost Contributions