EUROPA MULTIPLE FLYBY CUBESAT:
PRELIMINARY DESIGN REPORT

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1 Motivation and Overall Mission Objectives

Our mission will be a Europa multiple flyby cubesat. Similar to the Europa Clipper, our mission aims to to assess the habitability of Europa, focusing on the ice shell and ocean.

Europa is a moon of Jupiter that has been a subject of interest in terms of its potential to harbor life. Europa is thought to have a large subsurface ocean which could contain organics. The Galileo orbiter discovered that Europa has a weak magnetic moment which requires a layer of electrically conductive material. Scientists hypothesize this is due to a large saltwater ocean. Voyager also discovered reddish-brown material coating fractures on Europa's surface. Spectroscopy measurements have suggested that this is due to sulfur compounds that have been deposited on Europa's surface by the saltwater ocean. Another explanation is that these colored spots are composed of organic compounds that have welled up from cracks in Europa's surface caused by tidal heating.

This subsurface ocean is covered by a thick sheet of ice. However, the Hubble Space Telescope recently collected images of large plumes erupting from the surface of Europa. This significantly improves the ability of missions to collect samples of potential organics in Europa's ocean as sampling would not require drilling below a potentially several mile thick ice sheet.

As a result of the ocean, Europa's surface is constantly changing. Thus, a key mission objective is to identify and characterize scientifically compelling landing sites. The CubeSat mission we propose will address two of Europa Clipper's science objectives: landing site reconnaissance and plume science measurements. Through these, our mission will assess the habitability of Europa's ocean. This cubesat mission is designed to be complementary to the Europa Clipper mission. As such, it will be focusing on the same goals as Europa Clipper but will be conducting more in-depth and higher resolution images of specific landing sites. This mission will also be conducting lower altitude flybys through various plumes on Europa to collect data on organics and specific amino acids.

1.1 Mission Success Criteria

As the Europa Multiple Flyby mission is primarily a science and landing reconnaissance mission, the mission success criteria will be based off these objectives.

1. Successful imaging of Europa's ice-covered ocean surface in order to determine habitability and potential for presence of life. The minimal mission success is imaging the surface of Europa's ocean during one flyby. A partial mission success would include imaging multiple sections of Europa's surface while a full mission success would be imaging the entirety of Europa's surface. These images will be sent back to scientists on Earth who can study them to determine whether this is potential for life in Europa's icy ocean.

2. Identification of a number of possible landing sites. These landing sites should be both close to features of interest, i.e. plumes or Europa's ocean, as well as safe for a lander. These sites of interest will be selected from data from previous and concurrent missions, such as the Europa Thermal EMission Imaging System (E-THEMIS) and Mapping Imaging Spectrometer for Europa (MISE) instruments on the Europa Clipper mission. These sites will be mapped topographically from imaging data collected with the Europa Multiple Flyby Cubesat. One thing to consider is that Europa's surface is rapidly changing due to ice movement caused by the tidal forces of the water in Europa's ocean which is encased beneath an ice layer. As a result, imaging must also be able to determine the rate of change of surface features, to ensure long term viability of landing sites. The minimal mission success criteria is to map one potential landing site. Partial mission success would include topographical mapping of three landing sites. Full mission success would include topographical mapping of six or more landing sites.

3. Collecting compositional data of Europa's atmosphere through plume science. Multiple geysers are present on the surface of Europa, and collecting data from these plumes allows for analysis of potential organics in Europa's ocean without having to drill through the ice cover. A minimal mission success would involve an equatorial flyby allowing the cubesat to take images and samples from the atmosphere above the plume. A full mission success would involve a flyby directly through the plume while an extended mission success would involve multiple flybys through multiple plumes to collect more thorough samples. This mission objective will enhance scientists' ability to determine the habitability of Europa.

1.2 Storyboard

1. 0 Months: Separation from Launch Vehicle

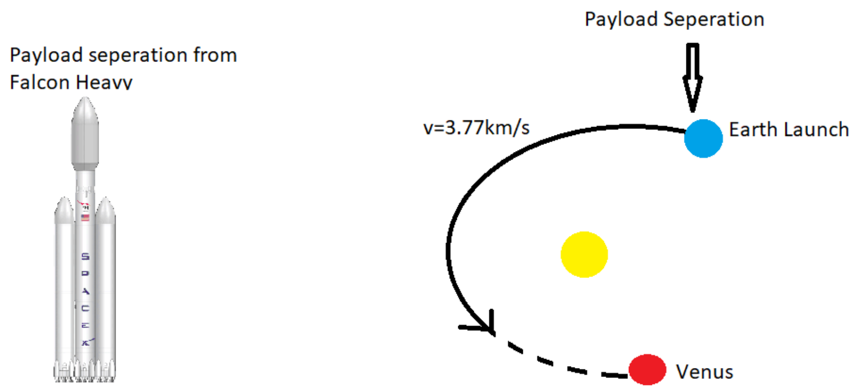

Figure 1: Launch from Earth

The CubeSat will be fixed to the Europa Clipper for the first leg of its mission. The two spacecraft will be transported into space by an EELV class launch vehicle, such as the Falcon Heavy. After the spacecraft separate from the launch vehicle, they will be traveling at approximately 3.77 km/s and will be on course to nearly intercept Venus.

2. 6 Months to 49 Months: VEEGA Interplanetary Trajectory

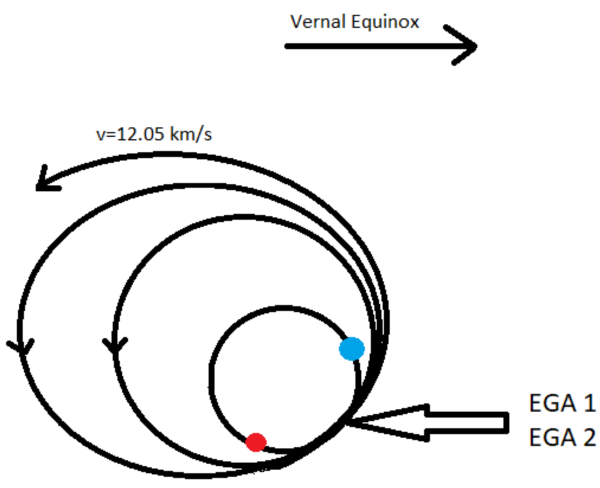

Figure 2: VEEGA Maneuver

This series of maneuvers first involves a Venus flyby and then two subsequent Earth flybys. The purpose is to massively increase the velocity of the spacecraft relative to the sun without using propellant.After the second Earth flyby, the spacecraft are traveling at a velocity of 12.05 km/s, on course to encounter the Jupiter system.

3. 78 Months: Ganymede Gravitational Assist

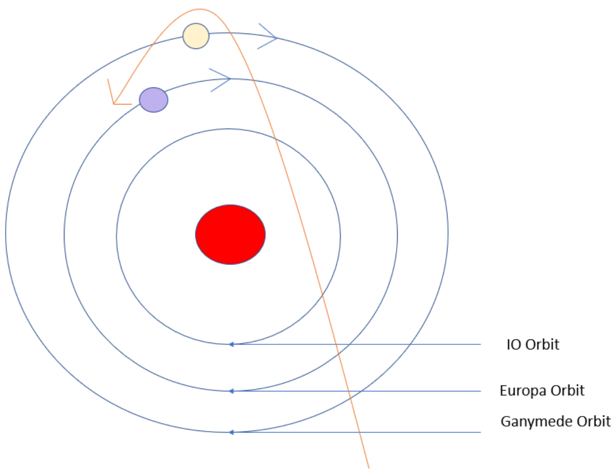

Figure 3: Reducing Orbital Energy Relative to Jupiter

The spacecraft will be nearing the Jupiter system at approximately 7.98 km/s, and a large ΔV would be necessary to perform a Jupiter Orbit insertion from this velocity. To reduce the propellant cost of this maneuver, a gravitational assist in the opposite direction of Ganymede's procession around Jupiter will be used. A flyby 500 km from the surface of Ganymede reduces the velocity of the spacecraft to 7.37 km/s.

4. 78 Months: Jupiter Orbit Insertion

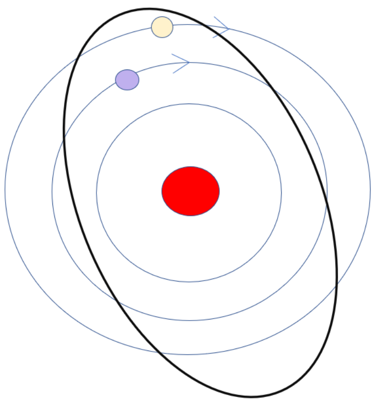

Figure 4: JOI Maneuver

Only 14 hours after the Ganymede flyby, the spacecraft will perform an insertion maneuver into the Jupiter system. Now only requiring a ΔV of 858 m/s, this maneuver would be performed at the periapsis of the previous orbit. This is approximately 706,101 km from the surface of Jupiter, in the outer regions of the radiation belts that are lower intensity.

5. 78 Months to 89 Months: Four Ganymede and Two Callisto Gravitational Assists

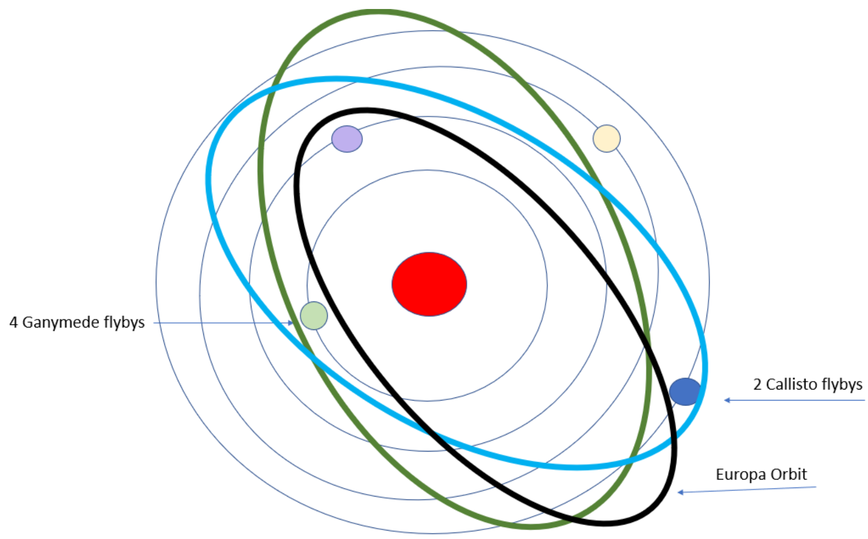

Figure 5: 6 Gravitational Assists in the Jupiter System

A periapsis raise maneuver and six total flybys of Ganymede and Callisto will be used to put the spacecraft into optimal orbit conditions for Europa flybys. The periapsis raise maneuver sets up the first Ganymede flyby in this sequence, and counters perturbations due to the Sun's gravity. The sequence of flybys will change the orbit orientation so that Europa is close to periapsis, while also reducing the spacecraft speed relative to Jupiter so that the orbital velocity relative to Europa is suitable.

6. 89.5 Months: Separation from Europa Clipper

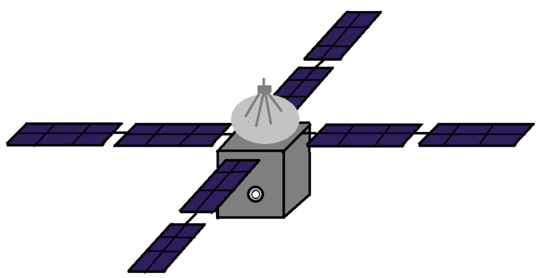

Figure 6: Separation from Clipper and Startup Sequence

As the Europa Clipper performs its second Europa flyby, the CubeSat will dislodge from the Clipper. The on-board computer will boot up and begin diagnostics. The solar panel arrays will extend fully and begin recharging the batteries. The antenna will be extended so that the Cubesat can reestablish contact with the Europa Clipper. All remaining subsystems will then be activated in sequence.

7. 89.5 Months to 90.5 Months: Maneuvers to Reach Imaging Orbit

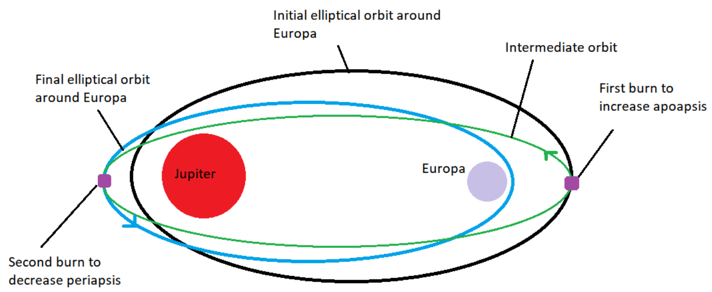

Figure 7: Maneuvers to reach Imaging Orbit

The Clipper's second Europa flyby occurs at the periapsis of its orbit, which is just 225 km higher from Europa's surface than our CubeSat target periapsis. The target apoapsis is 60,193 km farther from Jupiter than the Clipper after the second Europa flyby. One burn at the initial periapsis and a burn at the target apoapsis for a combined ΔV of 57 m/s will be sufficient to reach the CubeSat target orbit.

8. 90.5 Months: Achieve Imaging Orbit

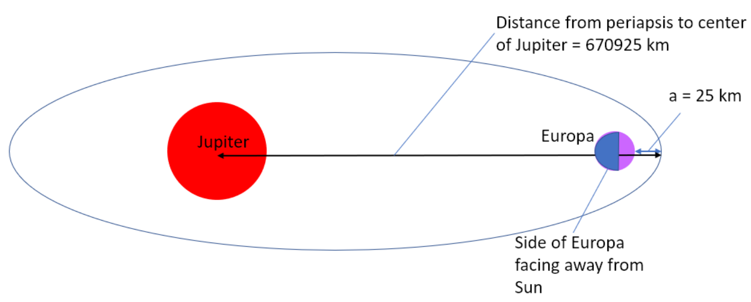

Figure 8: Imaging Orbit Details

An elevation of less than 30 km is required by the size of the CubeSat imager. An orbit with a semi-major axis of 1,690,468 km and an eccentricity of .603 would satisfy this requirement. The periapsis would be 670,925 km from the center of Jupiter, or 25 km from the surface of Europa. The apoapsis would be at 2,709,820 km, outside of Jupiter's radiation belts. The velocity of the CubeSat at periapsis would be 17.4 km/s, or 3.66 km/s relative to Europa. An important aspect of this orbit is that the periapsis will be at the point of Europa's orbit when its anti-Jupiter side is facing towards the sun and completely illuminated. This may require several argument of periapsis maneuvers to maintain.

9. 90.5 Months: Selecting Landing Site Candidates

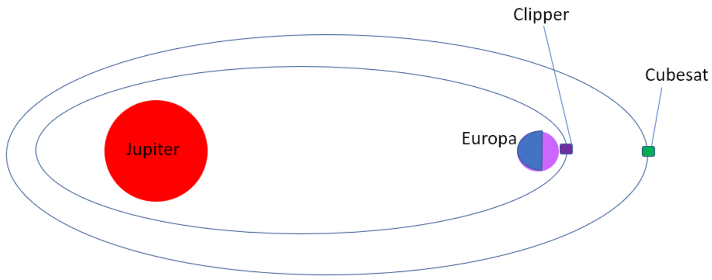

Figure 9: Clipper vs Cubesat Orbit

As the Europa Clipper performs Europa flybys, it will send plume locations from its Thermal Emission Imaging System (E-THEMIS), surface composition information from its Mapping Imaging Spectrometer for Europa (MISE), and visible spectrum images from its Europa Imaging System (EIS) back to the Earth ground station. This information will be used to select sites that require more detailed topographical mapping. The landing site inclinations and positions will be communicated back to the CubeSat via the Clipper.

10. 90.5 Months: Adjust Inclination

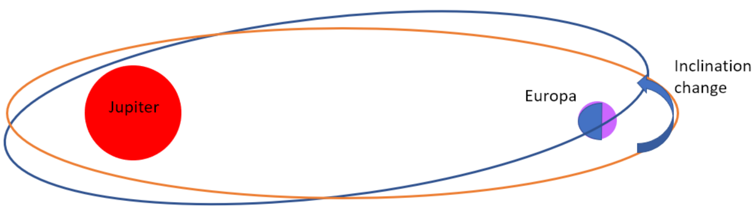

Figure 10: Inclination Change Maneuver

The CubeSat inclination will be adjusted to match one of the landing site candidates for several flybys. A direct out of plane orbit maneuver could be used to match these exact inclinations. There is also a method to convert the gravitational assist effects of passing so close to Europa into an inclination change, known as a Crank-over-the-top (COT) sequence when repeated. This could sometimes be utilized to reduce ΔV, depending on the target inclination.

11. 91 Months: Imaging and Taking Atmospheric Data

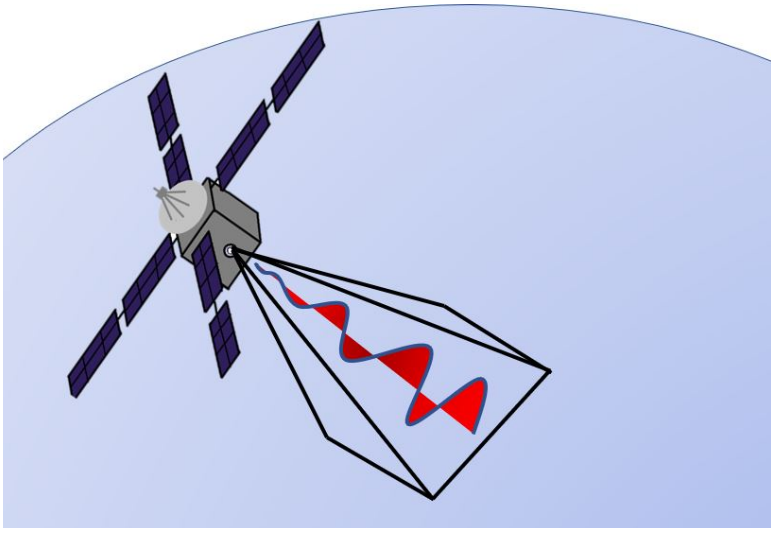

Figure 11: CubeSat Imaging the Surface of Europa

As the CubeSat does a flyby of Europa, it will be taking two types of data. As the CubeSat passes over a likely landing site, it will image the site with its camera. When not over a landing site, the mass spectrometer will be used to sample and determine the composition of Europa's atmosphere. After several flybys at the same inclination, detailed topographical maps of the sites will be developed. These maps and atmospheric data will be transmitted to the Europa Clipper during the rest of the orbit, to be attached to the data package the Clipper is sending back to the Earth ground station. Events 9 through 11 repeat for every 11 Europa flybys or 5.2 months until 3 landing sites are mapped. This satisfies Mission Success Criteria 1 and 2.

12. 106.6 Months: Attempt Plume Science

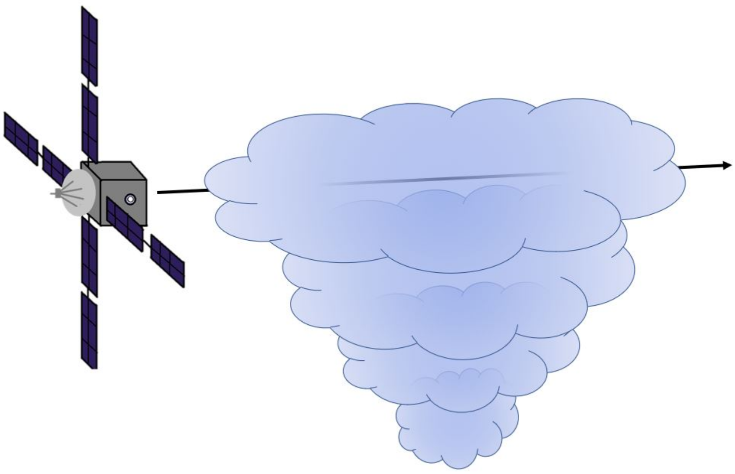

Figure 12: CubeSat Flying Through Plume

The CubeSat primary function during the last ten Europa flybys will shift to attempting to pass through or close by a plume. The mass spectrometer would sample the plume to determine its composition and transmit the results to the Europa Clipper during the rest of the orbit. This event is contingent on the Clipper's Ultraviolet Spectrograph (UVS) determining that the composition of the plumes would not cause severe damage to the CubeSat. This satisfies Mission Success Criteria 3.

13. 111.3 Months: Deorbit into Io

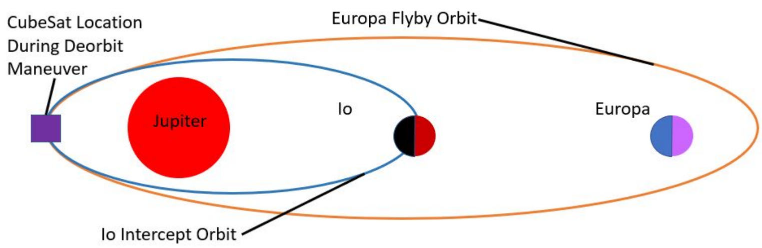

Figure 13: CubeSat Performing Deorbit Maneuver

Due to the Laplace three-body resonance of Europa, Ganymede, and Io, Europa has an orbital period approximately double the orbital period of Io. There is one constant position relative to Jupiter where Europa and Io will always be closest. At end of mission lifetime, the CubeSat will perform a 1.501 km/s ΔV burn at apoapsis to reduce the periapsis by about 250,000 km. The CubeSat will then intercept Io and be destroyed by the impact and Io's high volcanic activity.

2 Mission Requirements

2.1 System-Level Requirements

[SYS 1] The spacecraft shall have a Mission life cycle of 9.25 years, which is separated into 7.5 years of travel and 1.75 years in the target orbit.

[SYS 2] The spacecraft shall be a 12U CubeSat with a maximum mass of 28 kg and a maximum volume of 20 cm x 20 cm x 30 cm.

[SYS 3] The spacecraft shall be able to survive vibration and random shock conditions that may occur during launch.

[SYS 4] The spacecraft shall not have a magnetic field outside its static profile higher than 0.5 Gauss above Earth's magnetic field.

[SYS 5] The spacecraft shall interface correctly with the CubeSat deployment mechanism on the launch vehicle.

[SYS 6] The spacecraft shall be able to survive the space environment conditions during the entire life cycle of the mission.

[SYS 7] The spacecraft shall have sufficient shielding to prevent damage to avionics from the radiation conditions in the space environment during the mission life cycle.

[SYS 8] The system shall maintain its temperatures within the ranges specified by the subsystem requirements.

[SYS 9] The spacecraft materials shall satisfy NASA outgassing requirements.

[SYS 10] The spacecraft shall be able to autonomously image the surface of Europa during a flyby. Derived from Mission-Success Criteria 2.

[SYS 11] The spacecraft shall be able to autonomously perform all necessary orbital maneuvers.

[SYS 12] The spacecraft shall achieve its target orbit using an eccentricity increase maneuver.

[SYS 13] The spacecraft shall achieve a highly elliptical orbit around Jupiter, that makes multiple flybys into the atmosphere of Europa to achieve Mission-Success criteria 3.

[SYS 14] The spacecraft will limit its exposure to the high radiation levels present close to Jupiter to less than a cumulative week.

[SYS 15] The spacecraft will be able to survive Europa atmospheric conditions, including exposure to plumes to accomplish Mission-Success criteria 3.

[SYS 16] The spacecraft shall be able to image the surface of Europa with high enough quality to determine topographical features and landing site viability, fulfilling Mission-Success criteria 2.

[SYS 17] The spacecraft shall measure the composition Europa's atmosphere, fulfilling Mission-Success criteria 1 and 3.

[SYS 18] The spacecraft shall measure the composition of plumes from Europa, fulfilling Mission-Success criteria 3.

[SYS 19] The spacecraft shall be able to communicate with Earth to send images and scientific data.

[SYS 20] The spacecraft shall be capable of autonomously correcting its attitude to point at the surface of Europa, fulfilling Mission-Success criteria 2

[SYS 21] The spacecraft shall have a NASA approved end of mission plan to deorbit into Jupiter.

[SYS 22] The system shall provide sufficient ΔV to complete the entire mission.

[SYS 23] The electrical power system shall meet the power requirements for all subsystems.

[SYS 24] The spacecraft shall have an Orbital Debris Assessment Report and be in accordance with NPR 8715.6 NASA Procedural Requirements for Limiting Orbital Debris.

[SYS 25] The spacecraft shall have a NASA approved Planetary Protection Plan.

[SYS 26] Spacecraft hazardous materials and propulsion system shall conform to AFSPCMAN 91-710, Volume 3.

[SYS 27] The spacecraft shall not utilize any pyrotechnics.

[SYS 28] Total stored chemical energy on the spacecraft shall not exceed 100 Watt-Hours.

[SYS 29] The spacecraft shall include an RBF pin to cut all power to the satellite when it is placed in the CubeSat deployment mechanism.

Top-Level Subsystem Requirements

2.2 Propulsion

[PROP 1] The propulsion system shall provide sufficient ΔV to complete all orbital maneuvers and reorientations in accordance with [SYS 22].

[PROP 2] The propulsion system shall be able to fire autonomously, fulfilling [SYS 11].

[PROP 3] The propulsion system shall consume the entirety of the propellant load during its life cycle, as defined by [SYS 1].

[PROP 4] The propulsion system shall have at least three inhibits to activation.

[PROP 5] The exhaust plumes shall not interfere with the inputs to the scientific instrumentation.

[PROP 6] The propulsion system, integration, and testing shall conform to AFSPCMAN 91-710, Volume 3, fulfilling [SYS 26].

[PROP 7] The propulsion system shall have a method to maintain its elliptical orbit around Jupiter, allowing it to conduct a fly-by of Europa for a short period of time fulfilling [SYS 14].

[PROP 8] The propulsion system shall be able to perform inclination change maneuvers sent from the ground station to modify its orbit.

2.3 ADCNS

[ADCNS 1] The attitude control system shall be able to maintain an attitude with the imager oriented towards surface of Europa, fulfilling [SYS 20].

[ADCNS 2] The guidance and navigation system shall be able to determine position to an accuracy of 50 meters and velocity to an accuracy of meters per second fulfilling [SYS 16].

[ADCNS 3] The attitude control system shall have a slew rate greater than its mean motion in its elliptical orbit around Jupiter, fulfilling [SYS 20].

[ADCNS 4] The attitude control system shall be able to position the spacecraft with a minimum accuracy of $0.1°$ fulfilling [SYS 20].

[ADCNS 5] The attitude control system shall include a form of momentum dumping allowing it to release excess momentum from its actuators.

[ADCNS 6] The attitude control system shall be able to correct for disturbance torques such as those originating from the strong magnetic field of Jupiter, fulfilling [SYS 14].

[ADCNS 7] The attitude control system shall contain sensors that can determine an attitude accuracy of greater than or equal to 0.1°.

2.4 Power

[PWR 1] Batteries shall be fully deactivated or fully discharged during launch.

[PWR 2] The batteries shall be charged to at least 80% at all times with the exception of during launch, fulfilling [SYS 6].

[PWR 3] Batteries shall be able to provide enough power to the spacecraft through peak power demands, fulfilling [SYS 28].

[PWR 4] The battery power must be viable to operate through situations of low or lacking solar radiation.

2.5 Command and Data Handling

[CDH 1] The spacecraft shall communicate to Earth from Europa, fulfilling [SYS 19].

[CDH 2] The spacecraft shall have three independent RF inhibits, fulfilling [SYS 4].

[CDH 3] The CDH Fault Manager shall detect errors in the spacecraft's subsystems, fulfilling [SYS 6].

[CDH 4] The spacecraft shall autonomously correct errors detected by the Fault Manager, fulfilling [SYS 6].

[CDH 5] The CDH subsystem shall autonomously allocate processing power to appropriate subsystems as required, fulfilling [SYS 23].

[CDH 6] The CDH subsystem shall have enough computing power required by the ADCNS subsystem, fulfilling [SYS 20].

[CDH 7] The spacecraft shall have an onboard clock for computing real-time position and attitude, fulfilling [SYS 20].

[CDH 8] The CDH subsystem shall have enough memory to store each data package before sending it later in the orbit, fulfilling [SYS 19].

[CDH 9] The CDH subsystem shall delete each data package upon confirmation of receipt, fulfilling [SYS 19].

[CDH 10] The CDH subsystem shall interpret and execute commands sent from the Ground Station.

[CDH 11] The CDH subsystem shall be able to autonomously perform all required orbital maneuvers, fulfilling [SYS 12].

2.6 Thermal

[THM 1] The visual cameras shall not exceed their max operating temperature of 40°, fulfilling [SYS 8].

[THM 2] The visual cameras shall not drop below their minimum operating temperature of -20°, fulfilling [SYS 8].

[THM 3] The sample collection instruments shall not exceed their maximum operating temperature of 60°, fulfilling [SYS 8].

[THM 4] The sample collection instruments shall not drop below their minimum operating temperature of -40°, fulfilling [SYS 8].

2.7 Telemetry/Command

[COMM 1] The telemetry system shall have a bit rate high enough to transmit all collected data within the mission lifespan.

[COMM 2] The telemetry system shall transmit data with X or Ka band signals due to the Europa Clipper's available frequencies.

[COMM 3] The command system shall have a bit rate sufficiently high to transmit one orbit of command data per orbit.

[COMM 4] The command system shall transmit data with X or Ka band signals due to the Europa Clipper's available frequencies.

[COMM 5] The command system shall use a signal containing error detection bits.

[COMM 6] The telemetry and command system shall have a bit error rate no greater than 10^{-5}.

[COMM 7] Telemetry and command system shall transmit data via the Deep Space Network, fulfilling [SYS 19].

2.8 Payload

[PL 1] The science payload shall capture images of selected landing sites, fulfilling [SYS 16]

[PL 2] Images shall provide topographic data with greater than 0.5m per pixel resolution using stereo or photoclinometry mapping techniques.

[PL 3] The science payload shall collect composition, and density data of Europa's atmosphere with spectroscopy or sample collection, fulfilling system requirement [SYS 17].

[PL 4] The science payload detect Europa's vapor plume ejections and and collect composition data with spectroscopy or sample collection, fulfilling [SYS 18].

[PL 5] The science payload shall determine the concentration of amino acids in Europa's vapor plume ejections.

2.9 Structures

[STR 1] The spacecraft shall contain only materials that satisfy the low-outgassing criterion defined in CDS Rev. 13, fulfilling [SYS 6].

[STR 2] The mass of the spacecraft shall make the density of the spacecraft less than or equal to water, fulfilling [SYS 2].

[STR 3] All parts of the spacecraft shall remain attached during launch, ejection, and operation, fulfilling [SYS 24].

[STR 4] The spacecraft structure shall be AA 6061 or 7075.

[STR 5] The satellite shall be capable of sustaining 20 Gs of acceleration along each axis.

[STR 6] The first resonant frequency shall be greater than 100 Hz fulfilling [SYS 3].

[STR 7] Structural safety factors shall meet or exceed 2.0 for yield and 2.6 for ultimate stress.

[STR 8] All spacecraft components shall be designed to a fatigue life factor of safety of 10 with respect to number of cycles.

[STR 9] The spacecraft structure shall accommodate venting during ascent.

3 Narrative

3.1 Orbit Selection

The CubeSat dimensions and the requirement on imaging resolution (.5 meters) set the maximum altitude of the mission's fly-bys of Europa. The CubeSat specification for 12U CubeSats sets the dimensions of the satellite to 30cm x 20cm x 20cm. This sets the maximum focal length the Short Range Reconnaissance Imager (SRRI), which is the visible light imager for landing site reconnaissance. To achieve a ground spatial resolution of 0.5 m or less, the maximum flyby altitude is 30km. To achieve the desired off-nadir resolution, a 25km orbit was selected.

A 25 km orbit is well below the height of the plumes on Europa. It is estimated that Europa's plumes rise about 200 km. above the surface and are mostly gaseous. As a result, for the mass spectrometer to collect plume samples, a direct fly-by through the plumes will be the most advantageous. While this is a more risky option than flying adjacent to a plume for data collection, it better achieves the mission success criteria of conducting plume science. This is also one of the reasons the imaging portion of the mission will occur before plume science. Additionally, the plumes are mostly gaseous and will likely not significantly damage the spacecraft.

3.2 Science Payload

The selection of and sizing of the science instruments was driven by the CubeSat size requirements, and the ground resolution requirement. An iterative sizing process was used to determine the required aperture diameter to achieve an off nadir resolution of .5 meters for the SRRI instrument. This was achieved with a 4.28cm diameter aperture, and a sensor that is 4.28cm x 4.28cm, with a pixel sensor size of 5μm x 5μm. The properties of this system were assumed to be similar to the Long Range Reconnaissance Imager (LoRRI) [11] on the New Horizons spacecraft. The weight and power draw of the SRRI were assumed to scale linearly with the aperture area of LoRRI. This produced an expected power draw of 0.246W and a mass of 0.343kg.

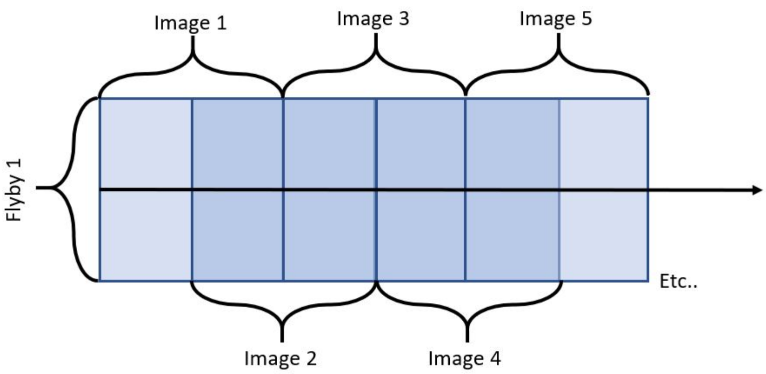

Figure 14: Imaging Pattern for Landing Site reconnaissance

Due to weight restrictions on the CubeSat, it was determined that a single camera would be used to map landing site topography. This necessitates imaging each landing site from different angles to generate a 3D topographical map.

Three landing sites will be mapped on the surface of Europa. For each landing site, an area of 117.8 km by 39.3 km will be mapped with less than .5m resolution. This area will be mapped in 11 flybys, each flyby with a swath width of 3.57 km and a length of 117.8 km. Data will be collected in 66 images, each image being 8562 x 8562 8 bit pixels. This set of images makes up one data packet, which will be fully transmitted during the orbit. Each image will overlap 50% of the previous image, creating 32.5 composite images that each contain enough data to create a 3D topography map of the surface. This process is repeated on every flyby to fully map the landing area. This landing area swath was picked to be 25 times the landing footprint of the Curiosity Mars Rover [14].

To determine the composition of Europa's exosphere and plumes, an ultraviolet spectrograph or a mass spectrometer is required. While the ultraviolet spectrograph allows composition to be determined without passing through the medium, spectrographs have an order of magnitude fewer spectral bins for an equivalent mass and power draw compared to a spectrometer. It was determined that a mass spectrometer was a better option due to its higher spectral resolution. The Compact Ion and Neutral Mass Spectrometer (CINMS) [12] was selected. This spectrometer can identify molecules with a mass up to 40 amu. However, since there are many organic molecules that could indicate the presence of life, such as HCO_3-, in the 40-65 amu range, a Mass spectrometer that can identify molecules up to 65 amu will be required for this mission. It was assumed that this mass spectrometer would have a 30% power and mass increase from the (CINMS). The spectrometer for this mission will be Extended Compact Ion and Neutral Mass Spectrometer (ECINMS) instrument, and will have a mass of 0.75kg, and have a power draw of 2.08 Watts.

3.3 Telemetry/Command

Typical CubeSat's of this size use UHF-band transmission, an initial power and antenna sizing were based on these frequency ranges and expected bit rates. Transmitting directly to the DSN on Earth requires both an unrealistic antenna size and excessive power requirements. This leaves using Clipper as a relay as the only reasonable method for communication with the ground station.

A maximum antenna diameter of 25 cm was based on the initial sizing of the CubeSat. The antenna will be stowed during launch and will deploy after separation from Clipper. Using an average downlink frequency of 250 MHz and the maximum expected distance from Clipper of 44,000 km, the CubeSat would need to provide approximately 350 W of power during transmission, which is unfeasible for this spacecraft. Additionally, the maximum bit rate for UHF-band transmission isn't high enough to reliably transmit all the science data between data gathering periods.

The higher power X-band transmission frequency range is uncommon on CubeSats, but has been attempted and shown effective. At a higher transmission frequency of 7.5 GHz, and a higher bit rate of 500KBPS, the X-band antenna of 20 cm and .25 kg requires at most 3.6 W of power to transmit. Per orbit, there will be 38.7 GB of data to transmit to the Clipper.

3.4 Command and Data Handling

Based on reasonable approximations for the number of instructions per common functions, the spacecraft will have approximately 48,700 instructions, about 50% of which are for ADCS.

The ADCS instructions will be run at 500 Hz, and the other functions' instructions will be run at 50 Hz. These relatively high data processing requirements follow from the higher degree of autonomy required for a CubeSat orbiting Jupiter, and for the high pointing accuracy required by the science payload.

A Digital Signal Processor (DSP) is well equipped to meet the data processing requirements. At 2 clock cycles per second for ADCS instructions and 5 clock cycles per second for other general instructions, the system requires 50 MIPS, and a small DSP can provide 100 MIPS, giving 100% margin of instructions per second if needed.

DSP's can also have up to 2GB of RAM, as well as ROM and Flash Memory storage, which is sufficient to store the approximate 9.7GB of data gathered per fly-by.

3.5 ADCNS

Due to the 0.5 m resolution of the imager in the science payload, the attitude determination, control, and navigation system must be able to determine attitude to an accuracy of 0.001°. With an imaging altitude of 25 km, an ADCNS system with this level of accuracy allows the spacecraft to differentiate between pixels. Thus, only two types of attitude control systems provide this level of accuracy - reaction wheels and control moment gyroscopes. CMGs are useful for applications requiring high rates of slew which are not necessary for the cubesat.

To determine position, a star tracker must be used as it can provide the most accurate position data (an accuracy less than 0.01°) of all ADCNS sensors. An NST-1 Nano Star tracker will be mounted in the CubeSat.

Due to the imaging pattern of the spacecraft, the CubeSat must be able to move over 1.8 km between every image. Accounting for the speed of the spacecraft around Europa (at periapsis), this designates an additional maximum necessary slew rate of 0.215° per second. As a result, a maximum torque of 0.53 mNm is required. Torquing along-track is necessary near Europa to satisfy our pointing requirements. The spacecraft will be using a set of four MAI-400 miniature reaction wheels. Three will be used for normal operation and the fourth will be redundant. These satisfy the accuracy and slew rate requirements. Additionally, the CubeSat will be using magnetic torquers for momentum dumping. These will be especially useful given Jupiter's strong magnetic field. The momentum dumping will only occur near Europa where the magnetic field is strong enough. The CubeSat will be using an ISIS magnetorquer board suitable to detumble spacecrafts that are up to 12U in size.

3.6 Thermal

From power budget estimations, the spacecraft's maximum power is 8.15 W (occurs during the imaging sequence, with all three reaction wheels drawing their potential max power), and the minimum power is 4.55 W (closest to Clipper, not gathering science data, reaction wheels drawing their steady state power). Because the CubeSat is small and so far from the

Sun, the biggest factor in the single-node heat balance is the internal heat produced by the electrical components. The minimum and maximum temperature requirements are derived from the batteries (min operating temperature = -5°C), and the SRRI camera (max operating temperature = 40°C).

The CubeSat surface will be mainly aluminum. Aluminum has the lowest emissivity, which allows the spacecraft to "trap in" the internal heat. Additionally, aluminum provides useful radiation shielding to protect the electrical components until end of life. While utilizing only passive thermal control would be simpler than active control, analysis shows that there simply isn't a combination of absorptivity and emissivity that uses mainly aluminum which can meet both min and max temperature requirements. Thus, active control will be utilized to vary electrical heaters based on the readings from thermocouples mounted in the spacecraft (the spacecraft has a max equilibrium temperature of 36°C, which is within the maximum temperature requirements).

3.7 Power

Data, ADCNS, and science systems contribute to the power budget. The power budget is described in Table 15.

Item	Power (W)
Imager	0.246
Communications	0.5-3.6
Mass Spectrometer	2.08
Onboard computer	1
ADCS	2.55-5
Star tracker sensor	1
Total Maximum Power at Apoapsis	8.15 ± 0.35
Total Minimum Power at Periapsis	4.55 ± 0.35

Figure 15: Power Budget Breakdown

Minimum power is 4.55 Watts and associated with on board computation, steady state ADCNS, and constant star tracker power consumption. Minimum power occurs when Clipper is closest to the CubeSat at periapsis, or when the spacecraft is over Europa. At periapsis, power required fluctuates as the spacecraft alternates between usage of the camera and mass spectrometer. While using the camera, slewing the spacecraft is necessary so moving images can be obtained. This will require the maximum ADCNS power budget of 5 Watts. At periapsis, the communication system will not be used and will therefore not require power consumption. Although the spacecraft will not require imaging or mass spectrometry when it leaves the Europa vicinity it will require increasing power for communications systems. As the spacecraft gets further from Clipper, it will reach the peak power requirement at apoapsis. Maximum power required at a given time is 8.15 Watts. This power is associated with on board computation, star tracker power, steady state ADCNS and maximum communication requirement. Thus the spacecraft solar panel will be sized with the following specifications: a surface area of 1.27 sq m and a mass of 3.1 kg. The solar panels will be GaAs Triple-junction solar cells sourced from AZUR Space. The thin and customizable design will allow for proper fitting to our CubeSat structure.[16] The battery will be sized to have a mass of 0.67 kg. These results were acquired by assuming 13.7 days of illumination, and 2.5 hours of eclipse. Eclipse time is approximated as a safety margin in the rare event the solar radiation is non existent for a period of time. Batteries will also be required to power the initial deployment of the spacecraft.

Figure 16 shows the power requirement over time.

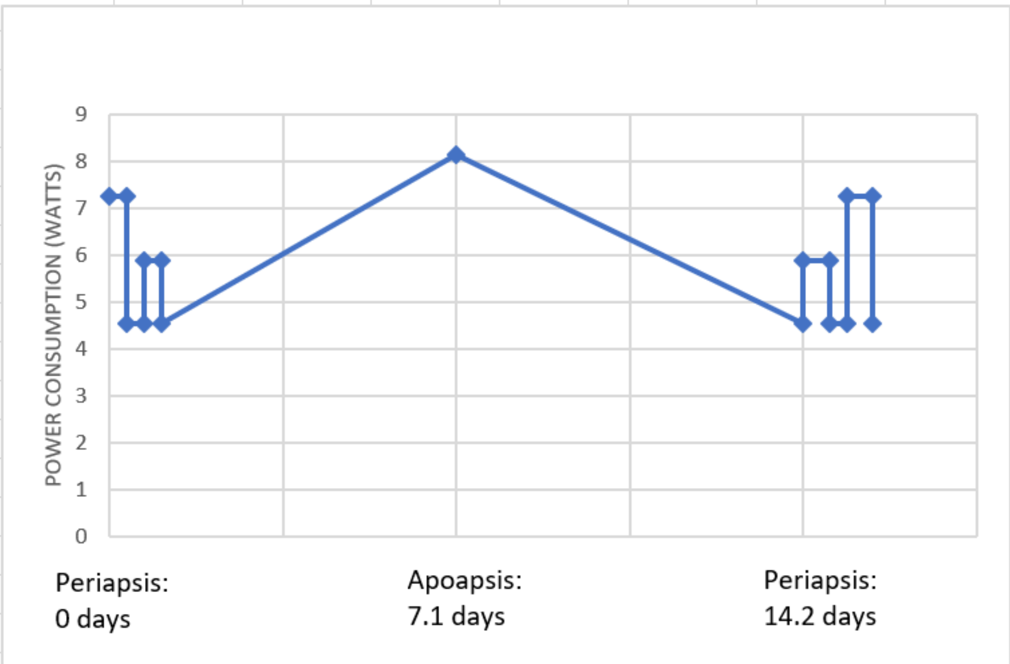

Figure 16: Power Budget

3.8 Propulsion

The propulsion system of the CubeSat has to perform an eccentricity change maneuver, inclination change maneuvers, orbit corrections, and a deorbit maneuver. The eccentricity change maneuver is accomplished using two burns. When the CubeSat separates from the Europa Clipper at the periapsis of their orbit, it performs a 15 m/s ΔV burn to move the apoapsis farther from Jupiter. A second burn of 42 m/s ΔV is performed at the new apoapsis to bring the periapsis of the orbit closer to Europa, 25 km above the surface. The largest inclination change is estimated to be 3° from the ecliptic of Jupiter. Each of these maneuvers would require a ΔV of 226 m/s, and this maneuver may be performed up to 3 times. The orbit correction over the approximately 2.275 year long mission life after separation from the Clipper requires an additional 102 m/s ΔV. The deorbit maneuver into Io was chosen over a deorbit maneuver into Jupiter due to ΔV savings. The chosen maneuver requires a 1.501 km/s ΔV burn at apoapsis to bring the periapsis of the orbit to the mean radius of Io's orbit. The total ΔV budget for the propulsion system of the CubeSat is the 2.338 km/s.

For the maneuvers to be performed, the possible propulsion systems were cold gas, liquid, and electric. With a very low I_{sp}, cold gas was quickly ruled out as requiring too much fuel relative to the mass of the CubeSat. Electric had a very high I_{sp}, but a low thrust that is more preferable for interplanetary maneuvers. That left monopropellant and bipropellant liquid propulsion systems. For a ΔV budget greater than 1 km/s, bipropellant is preferred over monopropellant due to a slightly higher I_{sp}. From the representative bipropellant systems in SMAD, the 10 N Bipropellant Thruster[4], the 5lb Cb[8], and the LEROS LTT[8] were the only options that would fit on the CubeSat. The third option had the lowest I_{sp} by far. The other two options had very close I_{sp}, but analysis of the overall difference in mass showed that the lighter .65 kg 10 N Bipropellant Thruster[4] was the best option. The length of this engine is .159 m, so it can be mounted in the center of the largest wall of the CubeSat. Placing it at the center of mass will prevent additional torques to the spacecraft. With its I_{sp} of 291 s, the propellant mass was calculated to be 13.81 kg of NTO. The size of the propellant tank was then determined to have a volume of approximately 9.59 liters and a mass of approximately 2 kg. The power requirement of the propellant system is much less than 1 watt.

3.9 CubeSat Size

The spacecraft will be a 12U CubeSat with a maximum mass of 28 kg. This became necessary as our spacecraft could not be kept below the 14 kg maximum mass of 6U CubeSats. This is most likely due to the large amount of propellant required to perform all of the planned orbital maneuvers. The mass budget of all of our spacecraft components are shown in Figure 17.

Item	Mass (kg)
Magnetorquers	0.392 ± .005
Camera	0.373 ± .005
Mass Spectrometer	0.75 ± 0.05
Antenna	0.45 ± 0.05
Engine	0.65 ± 0.3
Propellant	13.81 ± 0.05
Tank	2 ± 0.25
Structures	2 ± 0.25
Batteries	0.67 ± 0.2
Solar Panels	3.1 ± 0.2
Reaction wheels	0.36 ± 0.05
Star tracker	0.15 ± 0.02
Total	24.705 ± 1.43

Figure 17: Mass Budget

4 Cost Analysis

Subsystem	Mass (kg)	Cost ($K)
Satellite		
Spacecraft Bus	2.5	1123.1942
Structure	0.5	415.95685
Thermal Control	0.01	335.00057
ADCNS	0.902	1859.5192
Elecrical Power Supply	3.77	2662.3756
Propulsion	18.31	171.20637
Telemetry and Command	0.45	504.88556
Total Spacecraft Cost		7160.7852
Additional Costs		
Payload		2864.3141
Integration, Assembly, and Test		995.34914
Program Level		1639.8198
Launch and Orbital Operations Support		436.8079
Ground Support Equipment		472.61182
Total Cost		17644.985

Table 1: Cost Estimate of Satellite

This table was developed based on the Small Satellite Cost Model (Table 11-11) in Space Mission Analysis and Design as seen in figure 18. This model roughly estimates the cost of all subsystems based on their masses. There are some factors that may make our satellite more expensive, for example, the fact that we will be using specialized ADCNS instruments to fulfill our mission's high accuracy pointing requirements. The subsystem costs include the cost of development, the actual cost of each subsystem component, and integration and testing, as well as other extra costs.

Table 11-11. SSCM Earth Orbiting Total Non-recurring Cost (development plus one protoflight unit) CERs in FY2010 Thousands of Dollars. Useful for spacecraft weighing less than 500 kg. It is presumed that these CERs include the cost of contractor program management, systems engineering, product assurance, and I&T. See Tables 11-35 and 11-38 for application of these CERs. 301

SME-SMAD WBS Element	CER Y = total non-recurring cost of development plus one protoflight flight unit in FY10 $K	Cost Driver(s)	Cost Driver Input Range	Standard Error of Estimate (absolute) FY10 $
1.1 Spacecraft				
1.1 Spacecraft Bus (alternate CER when no component information is available)	$Y = 1{,}064 + 35.5\,X^{1.261}$	X = Spacecraft Bus Dry Weight (kg)	20–400 kg	3,696
1.1.1 Structure	$Y = 407 + 19.3\,X \times \ln(X)$	X = Structure Weight (kg)	5–100 kg	1,097
1.1.2 Thermal Control	$Y = 335 + 5.7\,X^2$	X = Thermal Control Weight (kg)	5–12 kg	119
1.1.3 Attitude Determination & Control System (ADCS)	$Y = 1{,}850 + 11.7\,X^2$	X = ADCS Dry Weight (kg)	1–25 kg	1,113
1.1.4 Electrical Power Supply (EPS)	$Y = 1{,}261 + 539\,X^{0.72}$	X = EPS Weight (kg)	7–70 kg	910
1.1.5 Propulsion (Reaction Control)	$Y = 89 + 3.0\,X^{1.261}$	X = Spacecraft Bus Dry Weight (kg)	20–400 kg	310
1.1.6a Telemetry, Tracking, & Command (TT&C)	$Y = 486 + 55.5\,X^{1.35}$	X = TT&C Weight (kg)	3–30 kg	629
1.1.6b Command & Data Handling (CD&H)	$Y = 658 + 75\,X^{1.35}$	X = Command & Data Handling Weight (kg)	3–30 kg	854
1.2 Payload				
1.2 Payload	$Y = 0.4\,X$	X = Spacecraft Bus Total Cost ($K)	2,600–69,000 ($K)	
1.3 Spacecraft Integration, Assembly, and Test				
1.3 Integration, Assembly, & Test (IA&T)	$Y = 0.139\,X$	X = Spacecraft Bus Total Cost ($K)	2,600–69,000 ($K)	
4.0 Program Level				
4.0 Program Level	$Y = 0.229\,X$	X = Spacecraft Bus Total Cost ($K)	2,600–69,000 ($K)	
5.0 Flight Support				
5.0 Launch & Orbital Operations Support (LOOS)	$Y = 0.061\,X$	X = Spacecraft Bus Total Cost ($K)	2,600–69,000 ($K)	
6.0 Aerospace Ground Equipment				
6.0 Ground Support Equipment (GSE)	$Y = 0.066\,X$	X = Spacecraft Bus Total Cost ($K)	2,600–69,000 ($K)	

Figure 18: Small Satellite Cost Model

From the table, it can be seen that the electrical power supply and the ADCNS system are two of the biggest costs. While the Electrical Power Supply makes up more of the total mass, the ADCNS system is the more technically complex system which will cost more.

The spacecraft bus will be another one of our more costly components. This is due to the fact that we needed to use a 12U satellite bus in order to carry enough propellant to deorbit. Because most CubeSats use up to 6U buses, our spacecraft will likely require more development time as well.

The second category of Additional Costs is calculated based off of the total spacecraft cost. Our specific payload, however, may be more expensive than listed. Our mission requires a compact ion and neutral mass spectrometer that can collect particles large enough for plume science. This spectrometer has not actually been developed yet, and investing money in both its development as well as buying the actual item may cost more money than is listed on the table.

The program level cost for such a mission, however, will most likely be lower than for other CubeSats as it is being developed in conjunction with the development of the Clipper. As a result, many of the discussions and costs related to mission planning will be covered by the Clipper program, especially as the cost estimation includes the mission trajectory to reach Jupiter. This portion of the mission has already been planned out for Clipper.

Overall, the estimated cost of this mission is about 18 million dollars. This is reasonable given the fact that this is a larger CubeSat than is usually used, and is also a specialized science mission into the outer solar system. This project will also be given priority as the goal is to have the CubeSat ready for launch by Clipper's launch date.

5 Risk Assessment

5.1 ADCNS Risks

Loss of attitude control due to either a hardware failure in the reaction wheels or in the avionics would render the spacecraft incapable of pointing. Multiple on-board computers can increase robustness, and extra attitude control hardware provides redundancy in case of a failure. The spacecraft will nominally utilize three reaction wheels with one dormant spare. This configuration reduces overall consumption as opposed to running the four reaction wheels together.

Finally, the above point also applies to Launch and Orbital Operations support. Launch support for this mission will be the same as launch support for Clipper's mission, as our satellite will be launched aboard the same vehicle and will be attached to Clipper until separation which occurs during Clipper's first Europa flyby.

5.2 Mechanical Risks

The propellant tank and engine must be tested to operating temperatures and pressures. Both the solar panels and parabolic antenna must be deployed after separation from the launch vehicle, and launch vibrations could cause damage that may prevent deployment. Vibration testing can validate the effectiveness of these deployment mechanisms, and use of COTS parts can reduce risk of failure.

5.3 Thermal Risks

Initial thermal analysis used a single-node heat balance to estimate equilibrium temperature. While passive thermal control meets the thermal subsystem requirements, any unanticipated thermal event could endanger certain temperature-sensitive components. To combat this, thermocouples will be placed on the most temperature-sensitive components, with the CDH Fault Manager periodically reading the temperatures and taking corrective actions should the temperatures cross specific thresholds.

5.4 Radiation Risks

Jupiter missions experience a high radiation environment. In particular, the inner moons Io and Europa orbit within the Jupiter radiation belts. To combat this risk, the spacecraft will be effectively shielded with aluminum and will be placed in an orbit around Jupiter (with a Europa periapsis) rather than a circular orbit around Europa. Utilizing the same thickness that Clipper is using for a radiation design factor of 2, the aluminum vault thickness will be 1 centimeter. This is a conservative estimate as our mission lifetime is shorter than that of Clipper.

5.5 Communications Risks

Communications loss could be due to avionics, electrical, or link budget failures. Redundant on-board computers can increase robustness, and the 20 cm parabolic antenna is gimbaled and can be pointed directly at Clipper to maintain communications and maximize the antenna gain at all times.

5.6 Electrical Power Risks

The solar panels are only slightly over-sized, and if one or multiple are damaged during launch or fail to deploy, the spacecraft will unable to function at full power. Structural considerations can mitigate risks associated with launch vibrations or deployment. The spacecraft can still be operational without all solar panels, and can still satisfy mission requirements while taking fewer images.

5.7 Plume Risks

The exact composition of the plumes is currently unknown and caution must be taken before flying directly through one. Harmful chemicals could damage sensitive components, while larger ice particles could impact and damage the spacecraft at high velocities. To address this, the CubeSat will start by doing the imaging sequence of its mission, allowing Clipper to do initial plume composition data collection. This will determine if it is safe to fly the CubeSat through a plume. The first plume fly-throughs will occur at high altitudes where the plumes have the lowest density. At lower altitudes, if it is unsafe to fly through the plumes, the CubeSat will fly close enough to the plumes such that it can still collect useful composition data.

5.8 Outgassing Risks

All CubeSat materials must be carefully chosen to minimize outgassing effects in the near vacuum space environment. Outgassed particles could condense onto our solar panels or camera lens. Both could be catastrophic to our mission. The condensed particles on the CubeSat surfaces can also increase the absorptivity of the spacecraft without changing the emissivity. This can cause an increase in the equilibrium temperature of the spacecraft over time.

5.9 Mission Technology Risks

One of the biggest issues with such a mission is that it is both time-bound and requires technology that has not been developed yet. This can cause delays in the determination of crucial items such as power budgets and ΔV budgets.

Because teams are still developing both of these items, moreover, this technology will not actually have been proven. While this will halt the time line of the CubeSat mission, it will either delay Clipper's launch date but more likely could be defunded.

6 Long-Lead Items

6.1 Mass Spectrometer

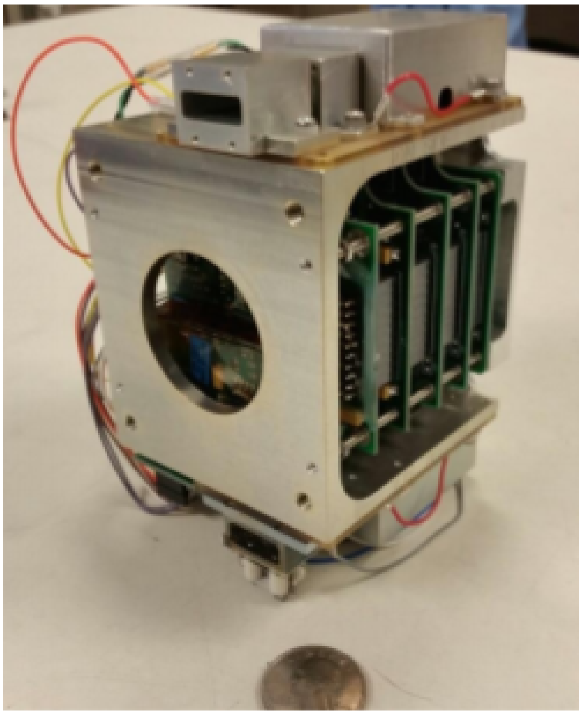

Figure 19: Compact Ion and Neutral Mass Spectrometer in Development

This mission requires a mass spectrometer that can collect samples from Europa's atmosphere and plumes. Goddard Spaceflight Center has developed the type of mass spectrometer we would like to use on this mission except that it can only collect data for particles with an atomic mass of 40 or less. Particles that may indicate life such as C_3H_4 are much heavier.

This spectrometer will take a while to build and should be developed at Goddard Spaceflight Center, as the team that developed the first compact ion and neutral mass spectrometer will know best how to modify and expand the original design for our mission.

6.2 Short Range Reconnaissance Imager

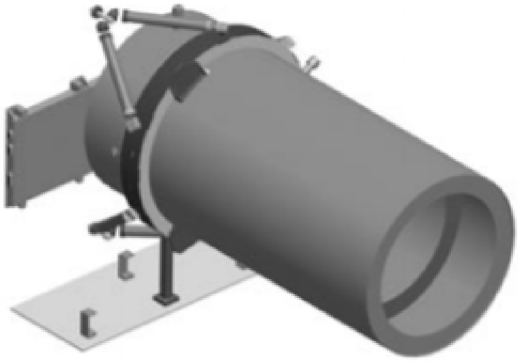

Figure 20: Short Range Reconnaissance Imager in Development

The Short Range Reconnaissance Imager is a 4.28cm diameter aperture, with a sensor that is 4.28cm x 4.28cm, and a pixel sensor size of 5μm x 5μm. The properties of our system are similar to the Long Range Reconnaissance Imager on the New Horizons spacecraft. The Long Range Reconnaissance Imager is a panchromatic high-magnification imager, consisting of a telescope with an 20.8-centimeter aperture that focuses visible light. Images will surpass Hubble-quality resolution, providing never-before seen details each day. It has a wide range of operational temperatures. The long range imager has all the qualities desired in our imager however, will need to be scaled down and custom made to fit the size and weight requirements of our spacecraft. It will need it to be small, light weight and consume low power. Johns Hopkins University Applied Physics Laboratory has worked on the LoRRI, and our team will work with them to continue to develop the SRRI with specifications for our mission.

6.3 Star Tracker

Figure 21: NST-1 Nano Star Tracker

While the star tracker is an off-the-shelf component, due to the high degree of accuracy and radiation hardened components, it will have a long lead time. This star tracker has a maximum slew rate greater than 2° per second and a maximum power draw of 1 Watt.

6.4 12U Bus

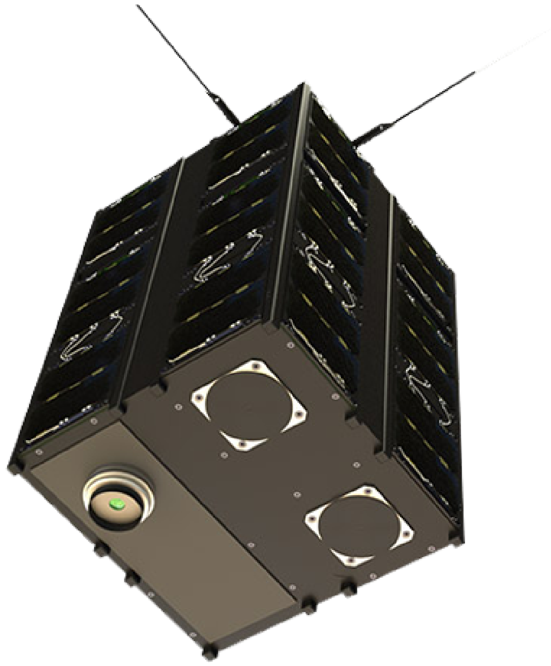

Figure 22: 12U CubeSat Bus

The 12-Unit CubeSat Structure from Innovative Solutions in Space (ISIS) would be a good choice for our spacecraft bus. The primary and secondary structures weigh approximately 2 kg total, which is exactly the amount we allotted in our mass budget. This bus design is modular with many mounting configurations and options. This allows us to be very flexible with our design and placement of components, but the custom nature of the bus likely means that we will have to allot a long lead time for it.

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