The Designated Professor Approves the Project Report Titled

System Integration Architecture and Testing Development for a Sun Tracking Heliophysics Smallsat

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Abstract

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This report presents research on applying spacecraft design under real conditions, accounting for constraints and issues faced throughout the mission and spacecraft development. A Heliophysics science recording cubesat was initially attempted, taking into account the constraints required to accomplish the objectives, such as: instrument selection, specific attitude control and determination requirements, and any mission-focused orbital design scenarios. This mission was later divided into various tiers, understanding the need to build the various driving requirements towards a future full science mission. This forced the current design into a technology demonstration mission testing a spin stabilized active attitude control and determination system, using a TechEdSat spacecraft bus as the base platform –with the idea of leaving the first Tier as a foundation for the further advancing missions. The following analysis reviews the details of: LEO mission operations, relevant trade studies, parametric design models with concurrent engineering tools, and proposes the testing conditions to verify all potential flight hardware. It is found that the nano-sat is a viable intermediate platform to advanced next generation Heliophysics and other science missions.

Acknowledgments

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Nomenclature

a	=	Semi-major axis
b	=	base
Bps	=	Bits per second
E	=	modulus of elasticity
f_E	=	Eclipse Fraction
h	=	height
h	=	altitude of the spacecraft
i	=	Inclination Angle
μ	=	Gravitational constant
P	=	Period of the Orbit
P_d	=	Power consumed by the spacecraft in daylight
P_e	=	Power consumed by the spacecraft in eclipse
P_{SA}	=	Power of the Solar Array
R	=	Radius of the earth
T	=	Torque (external)
T_d	=	Length of in daylight
T_e	=	Length of in eclipse
U	=	Unit of cubesat volume
X_d	=	Efficiency of paths in daylight
X_e	=	Efficiency of paths in eclipse
ω	=	angular velocity

I. Introduction

WHILE many apply their effort in developing the newest telecommunications system or the most efficient Solar Panels, few stop to consider the compatibility of the many individual components, analyzing the integration into a complete system. This is not exclusive to Spacecraft or Aircraft. Although study of systems engineering initially mostly used for aerospace applications or highly complex systems, many have adapted this method for product development, from cars to computers. The benefits of modular interfacing between subsystems allows for a high level understanding of interconnectivity between various subsystems and their interdependencies.

As everything else, spacecraft are tending to get smaller and cheaper, relatively speaking. Various form factors, led principally by volume, shape and mass, have been introduced, but one has left a cost friendly form factor that has changed the aerospace industry enough that ITAR laws are being modified to accommodate such systems and partially having them as “open source”.[1] The cubesat is the perfect platform for education, technology development/demonstration and in particular cases, SCIENCE!

A major issue encountered when relating to cubesats is attitude determination and control which at the moment is either very precise and thus very expensive or too rudimentary for the majority of science missions out there. The sweet spot for pointing accuracy is between 0.5 and 5 degrees. Many passive systems based primarily on magnetorquers promise that in ideal conditions could approach, maybe even achieve, 5 degrees or less, but this leaves little or no margin for requirements that are strict on the pointing accuracy. On the other side, companies such as Maryland Aerospace[2] and Blue Canyon Technologies[3] have developed systems that promise accuracy of well below 1 degree, (±0.007° [1 sigma])[3]. This is perfect for many government or privately funded missions with budgets surpassing $3M, but for the self-funded educational cubesats, which try to stay under $100k, it can be difficult to spend 50% or more of their budget on one subsystem. This leaves a gap, a gap that this report will attempt to fill, at least partially.

SJSU has a vast experience in developing and flying cubesats for many of these reasons, yet not science. The Heliophysics department at NASA Ames believe the future of solar mapping can be done with nano-sats and one of the constraint that’s being encountered is the attitude determination and control for precise sun pointing. Using the TechEdSat platform with required modifications to adapt an spin stabilized ADCS system as a technology, demonstration to later fly a hard X-Ray spectrometer, this will allow to explore the sweet spot for ADCS, with low cost as main focus. This report will be the road map for the design, development and, if time allows, flight of the modified TechEdSat.

II. Process Overview

Spacecraft mission design is not a simple task. This study attempts to approach as many aspects as possible, but this task can only be achieved through experimenting the real life process. The following chart will go through an overview of the aspects that were touched in this study.

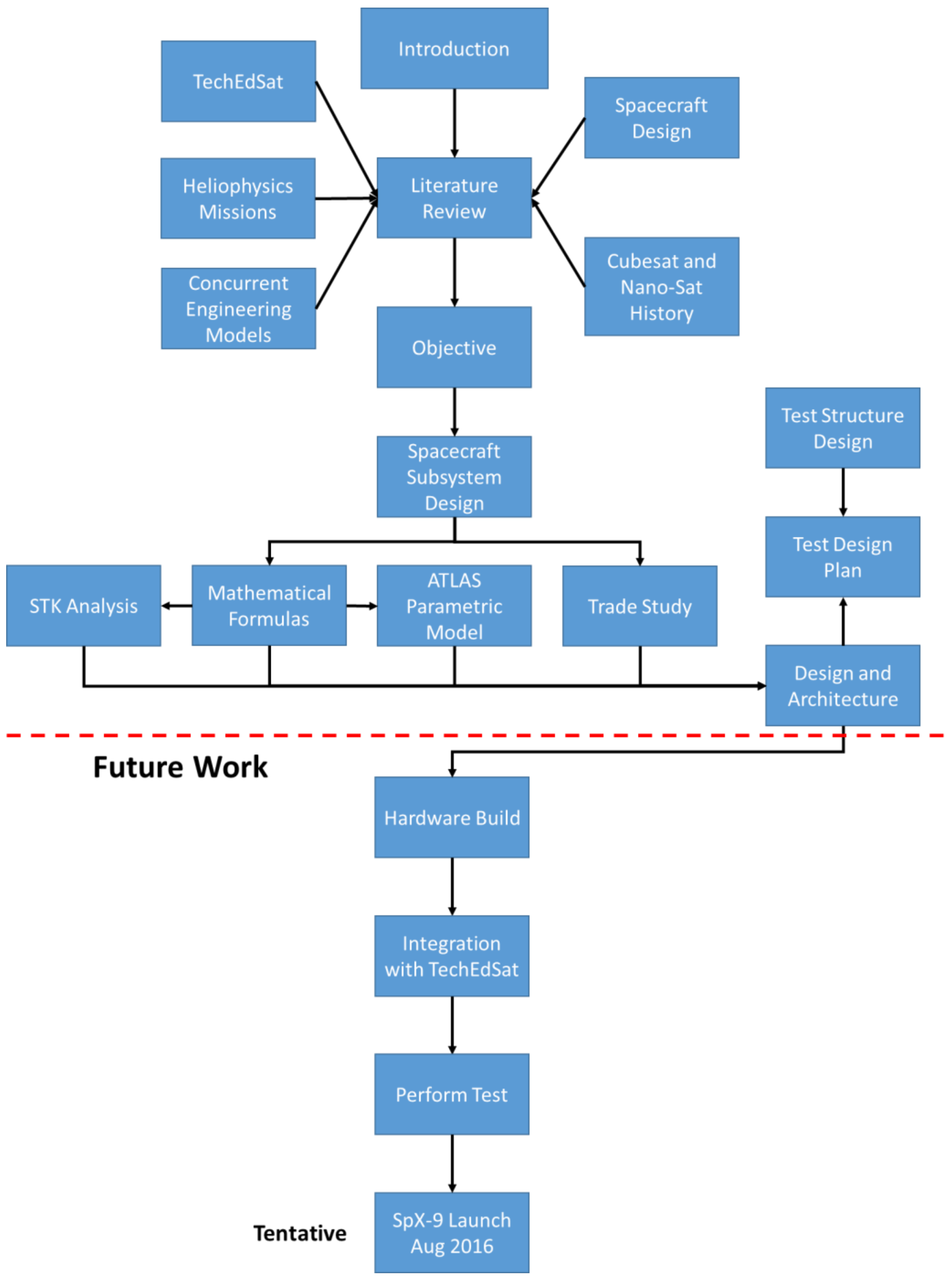

Figure 1: Project Process Overview

III. Literature Review

A. Problem

While many have tried, to date nobody has been able to successfully fly a reliable, accurate Attitude Control and Determination System (ADCS), much less one that's cheap, economically speaking. The specific need has been recently required for Heliophysics purposes. This could base the proof of concept of the next large solar science space mission. While Cubesats, are relatively inexpensive, these tend to increase cost with the level of sophistication required. Thus mission required could very rapidly increase the bill.

The obvious need for a more economically viable ADCS is how this study, including the integration of this system with a completed Spacecraft system. Whether this is an existing spacecraft or not, the integration will play a crucial role in the success of the study. This leaves at least two incognitos that are encountered: the creation of an ADCS system and the integration of it with a spacecraft.

What are the best methods to achieve this? What systems are out there? How will the author distinguish the capabilities of this study understanding the obvious economical and infrastructure restraints? These are some of the questions that this study will try to answer.

B. Space System Integration

Space system integration starts with one or more broad objectives from independent sub-systems, proceeding to define them as a system that will then comply with all of these objectives, at the lowest possible cost. Having broad objectives and constraints are elemental to this process.[4]

Space missions have a large range of applications which's list is too long for this research, from interplanetary exploration to Earth observation. This allows to assume, accurately, that there is no single process that will fully cover all possible contingencies that could be encountered. Space Mission Analysis and Design includes in its first chapter a table that summarizes a practical approach that's evolved over the first 40 years of space exploration.[4]

Typical Flow	Step		Section
	Define Objetives	1. Define broad objectives and constraints	1.3
		2. Estimate quatitative mission needs and	1.4
	Characterize the Mission	3. Define alternative mission concepts	2.1
		4. Define alternative mission architectures	2.2
		5. Identify system drivers for each	2.3
		6. Characterize mission concepts and	2.4
	Evaluate the Mission	7. Identify critical requirements	3.1
		8. Evaluate mission utility	3.3
		9. Define mission concept (baseline)	3.4
	Define Requirements	10. Define sysetm requirements	4.1
		11. Allocate requirements to system elements.	4.2-4.4

Table1: The Space Mission Analysis and Design Process (Table 1-1)[4]

When handling anything related to space, it's known it'll be expensive. This is why cost is becoming the fundamental limitation of more and more space missions.[4]

Analysis and design are iterative, progressively updating requirements and methods of achieving them. Thus, the process from the previous table must be repeated as many times as needed during the concept of the mission. The continued process will generate a more detailed, better-defined space mission concept. Even in the reflight of space missions, the broad objectives analysis is repeated to find if there is an even more cost efficient way of meeting the objectives.[4]

For the past forty years the engineering design of spacecraft has grown from a more infant discovering state, to one with well-defined techniques supported by analysis tools, the evolution of manufacturing methods and technology, and proof of flight level, a.k.a. "space-qualified" hardware.

The next paragraphs will explain some of E. Reeves, of chapter 10 of Space Mission Analysis and Design. With the purpose to further understand the needs of a Spacecraft or Space mission systems engineer, englobing from the design process to some constraints and budgets that the systems engineer must comply by from a higher level to assure a successful architecture integration.

Three elements are what is needed to compose an unmanned spacecraft: a payload, a spacecraft bus, and a booster adapter. Since the author’s priority is to explain and demonstrate the engineering of the payload integration with the spacecraft bus, the booster adapter, which can be known as the launch vehicle and/or propulsion system, will not be further mentioned unless its mentions it is connected with any orbital maneuvers required for mission success. The Payload can be described as the item directly related with the objective of the mission. This can be a specific piece of hardware which is going to prove a technology usually known as "tech demo", or an instrument to obtain some sort of science or service. Missions, commercial and non-commercial, commonly will have more than one payload when the design allows it, thus, taking advantage a spacecraft bus and exploit it as much as possible. The spacecraft bus is what the unit that carries the payload, and will most of the time provide, power, housekeeping data and communications. Many have attempted, and although some have partially succeeded, the idea of having a universal spacecraft bus to fit all payloads. This can get very complicated and is largely defined by the payload's environmental and interface requirements. Having a design standard allows for a more modular approach, such as a cubesat, but a large amount of payloads will still require a spacecraft bus that is basically, designed around it.[4]

C. Mission Requirements

The word requirement has been thrown around more than once within the text, without a specific definition in the context the word is utilized. While most know the definition of the word requirement, '*requirements*' when referred to in the space mission design world can be the best tool or worst enemy a systems engineer. The requirements are the mission needs and objectives written in detail, from a general state (Level 1) to a specifically detailed component state (Level 4 and beyond), all this without designing the item it's referring to. This may sound simple, but it can become complicated, specifically when a writing very detailed lower level requirement. The verification of poorly written or understood requirements, could define mission success or failure. Many missions are cancelled due to not being able to verify all requirements.

Higher level requirements and constraints are set by the concept and architecture of the mission, and by the payload operations. Going from a spacecraft design perspective, the orbital trajectory design will also affect and demand changes or certain constraints from a controls, communications, thermal and other subsystems, to only name a few. This example applies as well, for the needs of the payload objective and will affect it directly in most cases. The idea is to understand how intertwined the subsystems are with each other, although subsystem leads will often fight for "small" parameter changes to their benefit, it could cause catastrophic issues with how that subsystem might interact with the other subsystems.

The payload will usually be the largest driver for spacecraft design. All of its parameters, whether they're volumetric requirements or specific operational requirements, such as thermal constraints or power requirements, will drive the physical parameters of the spacecraft. As mentioned before, when possible the spacecraft is designed around the payload. The good news is that in most situations, most parameters, needs and constraints of the payload are known long before those of the spacecraft. Thus, we can decide on many important design parameters with the understanding of the payload and its operations.[4]

While trying to define requirements for the nano-sat mission, advantages of using an intermediate nano-satellite mission before a full scale mission, would allow early verification of such ambitious requirements. As well as proving an improved risk management for this larger mission to come.

D. Heliophysics Missions

Diving into a Heliophysics mission cannot be done without exploring what's been done before. Many spaceflight missions have dedicated to studying the sun's physical activity, mostly to predict solar weather but also to realize various studies, such as how the Sun's magnetic field is generated and structured.[5] Missions such as Solar Dynamics Observatory (SDO) and Interference Region Imaging Spectrograph (IRIS), while generate large amounts of data, two of the instruments of SDO generate 1.5 TB of data per day alone[6], can cost hundreds of millions of dollars.[7]

Taking as main example SDO, the three major drivers for a mission of this type are: communications, precise pointing and instrumentation requirements. SDO lives in an inclined geosynchronous orbit, that allow for nearly-continuous, high-data rate, contact with a single, dedicated groundstation.[5]

The SDO carry's three instruments, Helioseismic and Magnetic Imager (HMI) and Atmospheric Imaging Assembly (AIA), and Extreme Utraviolet Variability Experiment (EVE). Of these three, HMI and AIA are very sensitive to jitter and while they count with image stabilization systems onboard, and require ≤ 2 arcsec, 3 σ, from the center of the solar disk. This can only be accomplished due to its 4 guide telescopes that count with 0.5-deg field-of –view, which allows to fine point after utilized the spacecraft's two star trackers to get it to the 0.5 deg range.[8] Attitude of this type is not common for other satellites, but due to the sensitivity of the data that is being recollect, and the limitations of the instrument to filter noise, this is the only way to accomplish the mission objectives.

As mentioned before, the amount of data that SDO produces surpasses 1.5 TB per day that is being taken, this will require an advanced communication system, which will allow it to transfer all of this data. Thanks to the high gain antennas on-board, with a data rate of 130 Mbps of continuous, high rate data stream at a Ka-Band frequency of around 26 GHz and having enough power to do it with its 1450 Watts from its 6.6 m^2 of Solar arrays[5], 1.5 TB is achieved.

Technology is rapidly evolving, whether related to size, quality or accuracy. The following image compare sun images taken from three very important Heliophysics mission that have been flow throughout the years. From left to right, Solar and Heliospheric Observatory (SOHO) in 1995, Solar Terrestrial Relations Observatory (STEREO) in 2006[9] and the mission that's being used as example SDO from 2010.

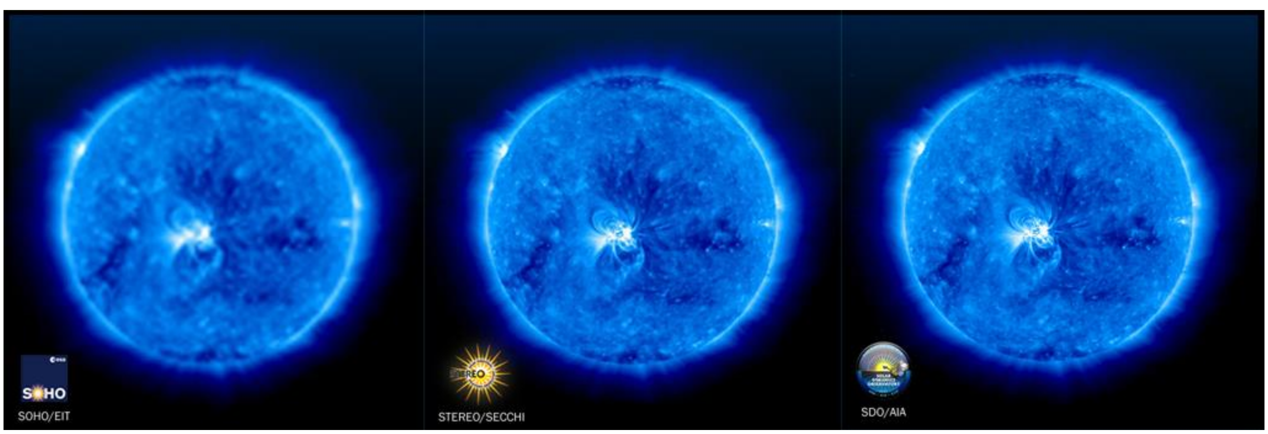

Figure 2: Image Quality comparison SOHO, STEREO, and SDO[39]

Coincidently or not, and due availability of information, it was found that the AIA instrument on board SDO is 211 mm[10], and the UV telescope found on IRIS is 20 cm[11], assimilating the needs of both instruments, it could be initial assumed that a range of 200 to 250 mm of aperture would be required for a mission dedicated to Heliophysics.

It would be worth mentioning that this mission is described to understand what has been done, and what intermediate steps or technology proving mission could be done at lower cost and concluding with a final full scale science mission.

E. Cubesat and Nano-Sat History

In the classrooms of Cal Poly around 1998, Dr. Jordi Puig-Sauri started teaching a satellite design class. He had the idea of creating a small satellite that could be built by students allowing for early hands on experience on actual flight hardware. Around the same time, Stanford also had a small satellite group in their Space Systems Development Laboratory with the effort of creating a picosatellite called OPAL. This group was led by Professor Bob Twiggs, who thought the satellite should be the size of a "beanie baby" box. An alliance commenced between Cal Poly and Stanford to develop the P-Pod deployer, that served to carry and eject several cubesats into space.[12] Little did they know they were changing they would set a standard and the portal to low cost access to space.

The cubesat is based on a unit system. A 1 unit (U) cubesat is approximately 10 cm x 10 cm x 10 cm. This was great for initial launches but most are taking advantage of the platform in larger configurations such as a 3U or 6U, which are also known as Nano-sats, but virtually any combination/configuration is an option when as long as a launcher can be found or design/created.

While science missions were first hard to believe by most of the dominant players in the Aerospace community. NASA Ames was one of the first with its Genesat/PharmaSat missions all in 3U configuration. Obtaining valuable biological science, by warming up E-Coli to growth temperature (~32°C) with a custom accommodated metal/Kapton heater films under a closed-loop control, which would then be "*resuscitate*" the E. Coli by pumping a sugar solution to remove the statis buffer used during load and launch operations.[13] This and many other successful launches have slowly attracted more and more to the platform, thus creating an entire industry revolving around Cubesats. Many new companies have utilized Cubesats as their only platform of use. These companies are not only being successful using them, but they are innovating and taking them to the next level, evolving the technology further and pushing the limitation that were believed to exist.

NASA Ames is currently working on the first Deep-Space cubesat, called BioSentinel which is meant to fly on the Orion EM-1 launch and will drift into a Heliocentric orbit. Although the science being done on the spacecraft does not require specific pointing requirements. The communication and power requirements depend on pointing to assure the solar arrays are facing the sun and the patch antenna can turn and face Earth to transmit data. These are challenges that are currently being faced, and while pointing requirements are not as strict as those mentioned for SDO, combining these small spacecraft platforms with strict data requirements can cause complications or not seem factual for the state of current available technology.

F. Attitude Determination and Control for Cubesats

Attitude Determination and Control systems have been around long before the cubesat and are not exclusive to spacecraft. Aircraft, although not usually named this way, have an ADCS. All control surfaces, instruments to indicate orientation and directions can be considered part of the ADCS. For spacecraft things are a little different though, one difference with aircraft is the lack of air. So although large surface area can affect the attitude of a spacecraft, for precise maneuvering control surfaces will not suffice.

As on aircraft, ADCS are composed of sensors and actuators, but instead of pitot tubes, radars and rudders, spacecraft use inertial measurement units (IMU), star cameras, reaction wheels, thrusters, torque rods and others that will probably are in development. Many of these systems are still in the process of being miniaturized to be capable of fitting in the cubesat platform.

1. Actuators

Thrusters are widely used in larger spacecraft for maneuvering and while many systems are being developed, none count with the flight heritage and could complicate the control algorithm utilized due to their very low thrust.[14,15] The other viable options as actuators are reaction wheels and magnetorquers. The size of these components don't allow or create interest for individual sale of these but many do offer 3-axis stabilization systems. Blue Canyon Technologies has 3-axis system that is compact enough to fit in a 0.5U volume called XACT[3], which promises ±0.007° [1 sigma] of control. This allows for plenty of space for all the other systems required in the satellite. The only constraint is the cost. The author is not at liberty to discuss cost, but it is well out of the reach of the majority of the schools who are looking to build and fly a cubesat. Other options such as the MAI-101, that take up more volume (1U) and still lack some items required for an ADCS system.[2]

2. Sensors

The pointing of the satellite can only be as good as its ability to know its state. IMU's which are basically 3-axis accelerometers are now available the size of a penny.[16] Sun sensors can provide some knowledge, specifically to know where to point the solar panels, can also help to get within a couple of degrees of knowledge.[17] The Blue Canyon Technologies XACT system also counts with a star camera, that promises much more accurate knowledge within ±0.003° (1 –sigma) [3], which is set to fly late 2015 and leaves the technology as plausible, the issue for many is cost.

G. TechEdSat Heritage

The TechEdSat was develop by San Jose State University and University of Idaho with the guidance of NASA Ames Research Center. The 1st TES (TechEdSat) was jettisoned from the ISS on October 04, 2012.[18] The mission was successful and functional for 7 months before it re-entered the earth's atmosphere. TES-2 was the 1st burst data satellite-to-satellite communication experiment, and was launched on August 21, 2013. This experiment successfully demonstrated the technology over the course of 24 hours. TES-3 was the 1st 3U CubeSat jettisoned from the ISS on November 19, 2013, and utilized an Exo-Brake system that was deployed as planned. TES-4 was jettisoned from the ISS on March 3, 2015 after being dormant on the ISS for almost a year. Building on flight heritage, TES-4 validates passive drag modulation and satellite-to-satellite communication. TES-4 has been flown and re-entered April 6, 2015.

Figure 3: TechEdSat (accompanied by another 2 cubesats) being deployed from ISS.[10]

The TechEdSat bus is a great candidate for an initial validation flight to prove some of the critical driving requirements for a full scale science mission, such as ADCS and Communications.

H. Concurrent Engineering Models

Real-time collaborative, concurrent engineering (CE) design centers for space have only been around for about 15 years. In true CE fashion, each concept design center (CDC) team relies on tools and models that share design requirements and data parameters. The majority of these tools reside in Microsoft Excel workbooks.[19]

In the past years, the use of concurrent engineering models (CEM) has increased for rapid development of conceptual spacecraft. The aerospace industry has been using CEM for many years, and while these tools began as simple spreadsheets, they have evolved into powerful tools that interact and interconnect with other software packages to increase function such as databases and powerful modeling software.[20]

CEM's are ideal for quick turnaround projects or proposals, which can be based on previous heritage and past studies. An important benefit of using past experience and projects is that the more the model is used the better it gets.

Initial Systems engineering tools used to help with the road map of the project development have been implemented to understand the interdependencies within the system. In this case in particular the integration of the TechEdSat platform with an ADCS system. Due to the fact that the ADCS has not been developed, the initial analysis will be done with a MAI-101, which has the most approximate specifications that are required.[2] Future iterations will be done with developed ADCS system.

IV. Objective

The objective of this report is to inform and be used as a guide of how to start with space mission design, without leaving out the objective of designing, testing and, if budget and time permits, fly onboard a nano-sat an attitude control system at low cost that will help stimulate the industry of small satellites. This will also serve as a stepping stone to what could be a larger mission to help bring down the cost of heliophysics' missions proving the stringent control requirements needed. Without excluding the real life properties of working on a space mission, which include budget cuts, project partner disagreement, and changes in the concept of operations.

V. Spacecraft Subsystem Design

SMAD defines Space Mission Engineering as "*the definition of mission parameters and refinement of requirements so as to meet the broad and often poorly defined objectives of a space mission in a timely manner at minimum cost and risk"*.[21] This explanation states no formulas, no equations, nothing that has always been the basis of not only an aerospace engineer, but any branch of mechanical engineering. This leaves the thought of, "Why?"

Differential equations, Fourier series and optimal control theory are essential for any engineer. History has taught that knowing what is wanted before it is made is the best way to design anything. Without a specification or requirements set, the objective gets blurred allowing for inaccuracies, misunderstandings and, many times failure.

Although "*Requirements*" per say, are usually not considered a subsystem, this report will take the approach of assuring, that since the requirement verification are what defines success before the product is delivered. Requirements are going to be taken into account as an essential subsystem of the spacecraft.

The other items to also be discussed in this section is the individual subsystems that compose the spacecraft and in this case the equations used in the model that was run for the system integration. Each section will outline the outstanding subsystems in the project, with some insight, of how cost, schedule, and technological capabilities can modify the direction of the mission. While some of the parametric studies that were done, and still are included in this report and are of great value to the mission, the direction of the mission, at any moment, can change.

A. Initial Mission Requirements

The system requirements are the map to the completion of a spacecraft mission. This is not usually considered a subsystem in a spacecraft mission design report, but to assure the requirements are given the importance they deserve they have been incorporated in to the main report. As in previous sections mentioned, the importance of the mission requirements must be acknowledged, and the following will show the first initial requirements for a first step in towards a full scale science mission.

While this list may seem short, as the design life of the spacecraft continues, so does the requirements into various levels that go from system, to subsystem to part.

The requirements discussed here will only be the high level requirements, which are commonly known as level 1's. Three main objectives have been set by to design the spacecraft, which have been set as the requirements to continue to build towards a full scale science mission.

Subsystem	**Requirement**	**Rational for Full Scale Science Mission**
Attitude Control and Determination	The spacecraft shall have spin-axis control to 1 RPM.	Prove active ADCS and slow enough spin rate to command pointing to the Sun or Earth, for power, communications or science data recording.
Instrument Integration	The ADCS system shall consume less than 2 Watts of power under nominal operating conditions.	Limitation of Power required by the ADCS.
Lifetime	The spacecraft shall have the capability to function and obtain data for at least 3 months of constant operations.	Full scale mission will require, longer lifetime, and must be able to space environment.
Communications	The spacecraft shall downlink at least 10 hours of attitude information (~100 MB total)	Heliophysics data requires high data rates, due to size of data recorded.

Table 2: System Level Requirements for initial Validation Mission

This might seem superficial or unimportant, but these are only the level 1 requirements which will extend to deeper more detailed levels which will branch out until a single verification method can be obtained. The requirement are susceptible to change, but the later this process is done in the lifecycle of a project can have larger impacts on the due date and require subject matter experts for their approval.

Comparing these requirements with what is being done with missions like SDO, and what potential future "stepping stone" missions could provide lower cost missions, that can generate data as valuable as what SDO can generate. Due to limitations of what instrumentation information is available for the platforms proposed, assumptions will be made for payload requirements.

Subsystem	Requirement	SDO	Small Spacecraft (Geo Sync) Tier 3	Nano-sat (Sun Synchronous Orbit) Tier 2	Nano-sat (Low Earth Orbit) Tier 1
Communications	Data per day	at least 1.5 TB	at Least 10 GB	at least 1 GB	10 MB
	Data Rate	130 Mbps	100 Mbps	100 Mbps	10-50 Mbps
Attitude Determination and Control System	Pointing	3-axis Stabilized to 2 arcsec, 3 sigma	3	3-Axis Stabilized to >1 degree	Spin Stabilized
Power	Power Generation	1450 Watts	200 Watts	40 Watts	>2 Watts (For ADCS)
Payload	Instrument Aperture	One is (211 mm)	200 - 250 mm	N/A	N/A
Cost		around $900 M	> $55M (MO)	> $15M (UNEX)	> $1M (TechEdSat)

Table 3: Different Mission Tier Requirement Comparison with SDO[5]

With the initial values and the previous table it can be seen that the road to a mission that offers science similar to one such as SDO. Detailing the previous table:

- Tier 1 – Prove active ADCS system (Spin Stabilized) and higher Rate communications (10-50 Mbps – X-Band), and use X-Ray Spectrometer to attempt take data while in sunlight. While more data could be downlinked thanks to the higher data rate, data that is to be initially recorded does not require large amounts of space.

- Tier 2 – Prove a 3-Axis Active ADCS system, with an optical sensor to help with the precise pointing and take initial sun science data, submitting a University-Class Explorers (UNEX)[22] proposal.
- Tier 3 – Large Scale mission, with partnership for instrument construction, possibly coronagraph, submitting a Mission of Opportunity (MO)[22] proposal.

The approach taken for the following study is for Tier 1. The purpose of denoting future missions is to demonstrate road that is required to be taken for a full scale mission with reduced risk in a still small spacecraft platform.

B. Mission Design

The mission design refers to designing and defining the operational flight parameters of the spacecraft. These parameters include launch, orbit, and duration. Since the interest of this mission is solar observation for space weather, the ideal orbit that comes to mind is a heliocentric orbit, which would mean the spacecraft has left sphere of influence of the Earth and is orbiting the sun. While this orbit is ideal, it may not be viable due to other constrictions imposed, such as cost and design limitations of the spacecraft standard, because of this, other scenarios are also going to be analyzed, taking into account what is desired, and the restrictions. Another viable scenario is a Sun Synchronous Orbit (SSO), these orbits are relatively low altitude orbits but with a very high inclination (usually above 95°), also known as polar orbits. The benefit of a SSO is that it allows for solar for long periods of time, again, great for the needs of the project. The problem that could be encountered is that few launches regularly go to this location and those that do, usually charge more than other Low Earth Orbit (LEO) options. For this reason, the third scenario to be analyzed will be a Low Earth Orbit, probably deploying from ISS, taking advantage of how NASA helps educational programs such as this one.

The three scenarios will be mathematically analyzed and then verified by recreating them with Systems Toolbox Kit (STK) [23]. This will allow for a better understanding of what it takes for each one of the designed scenarios, and it will give foundation to the decision made.

1. Heliocentric Orbit

The analysis done will require various assumptions. Since the idea behind the project is of a cubesat form factor, and no real propulsion has been flown nor tested to a state that can inspire confidence for a mission of this magnitude. The assumption of a “piggy-back” mission will be taken, that is that the spacecraft will launch on a future Heliocentric bound satellite, with a reliable propulsion system and deploy the spacecraft while in heliocentric orbit. This assumption would allow for a no longer needing the complex orbital analysis required to determine desired trajectory and could involve work for a report of this caliber for its self.

This task can however be minimized with a software package that was mentioned before, called Satellite Tool Kit (STK)[23]. This hardware is utilized to do quick yet reliable orbital analysis for Earth and interplanetary missions. The analysis that was utilized with the assumption of it being launched to a Geo Transfer Orbit (GTO) in the US launch the first half of 2018[24]. GTO’s are known for being highly eccentric orbits, in this case *200 km x 35786 km*, that are commonly utilized for trajectory design for interplanetary missions.

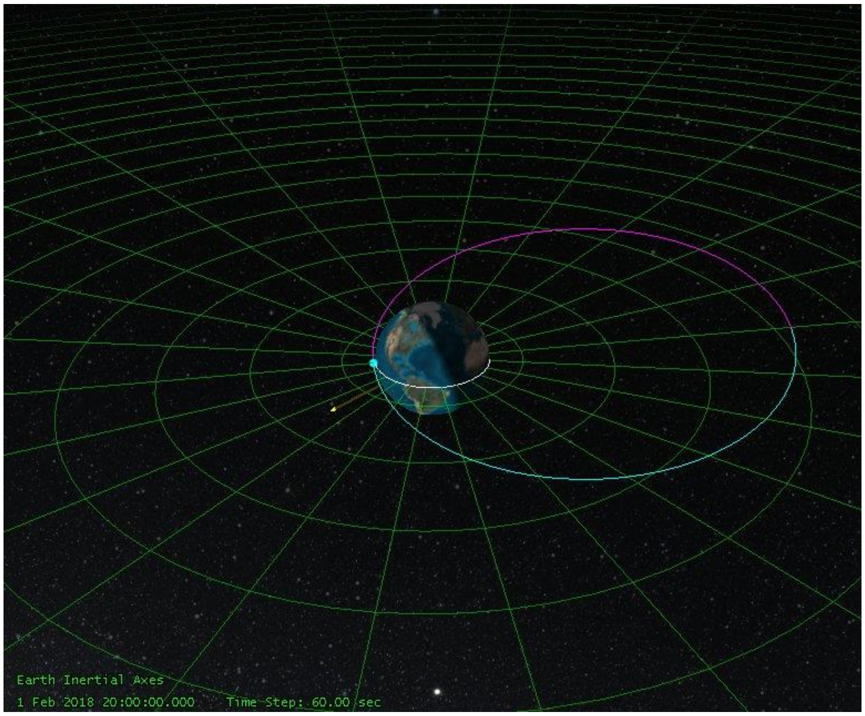

Figure 4: Geo Transfer Orbit

A single hyperbolic escape maneuver will propel the spacecraft beyond the L1 Lagrange point, which is where the Earth's gravitational force is neutralized by the Sun's gravitational force, this will allow to position the spacecraft in an Earth trailing "sweet spot" for a heliocentric orbit, as described by Weston[25].

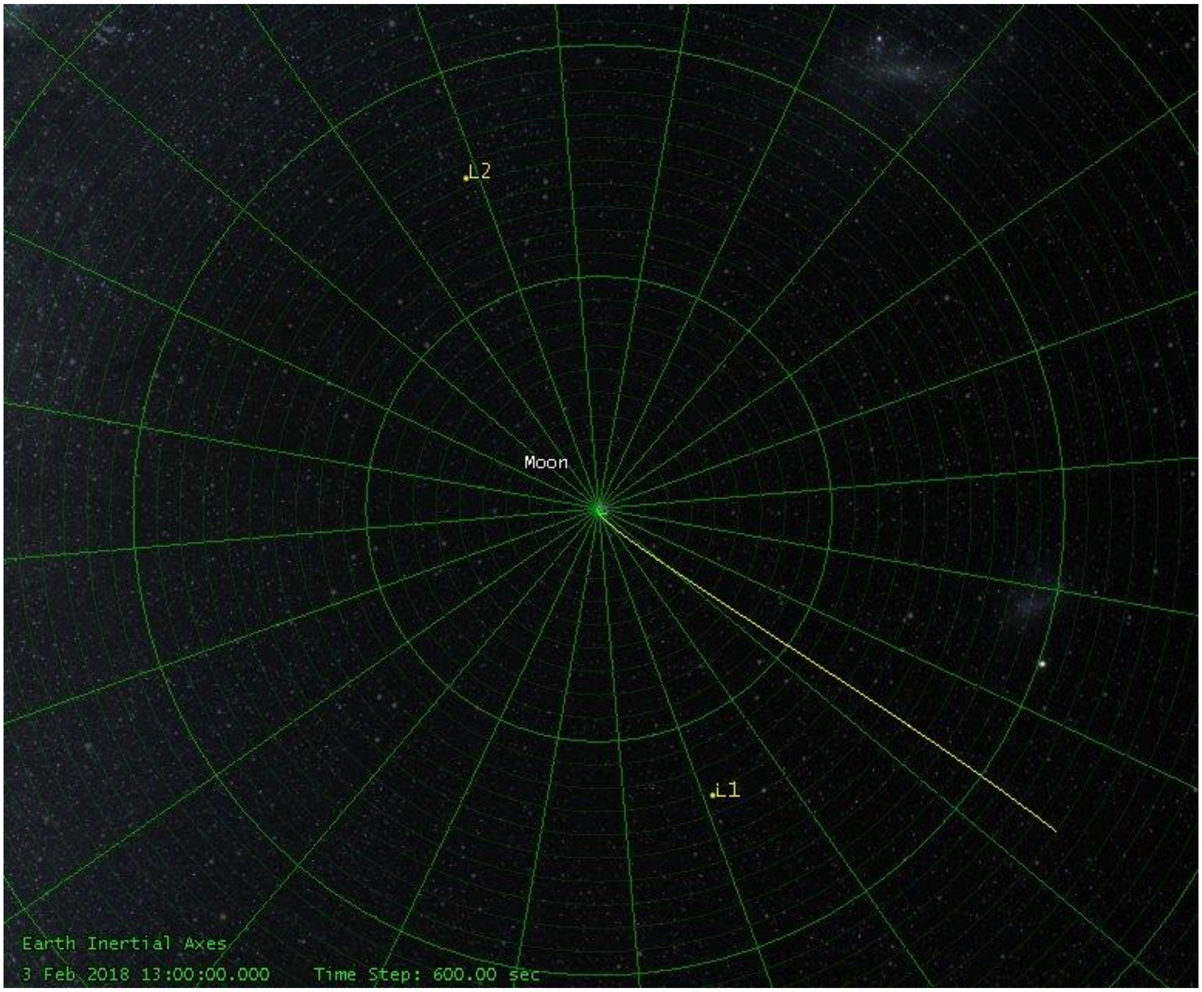

Figure 5: Hyperbolic Escape from a Top View, notice the distance from L1

From a parametric point of view, it can be seen how the trajectory is slung towards the sun.

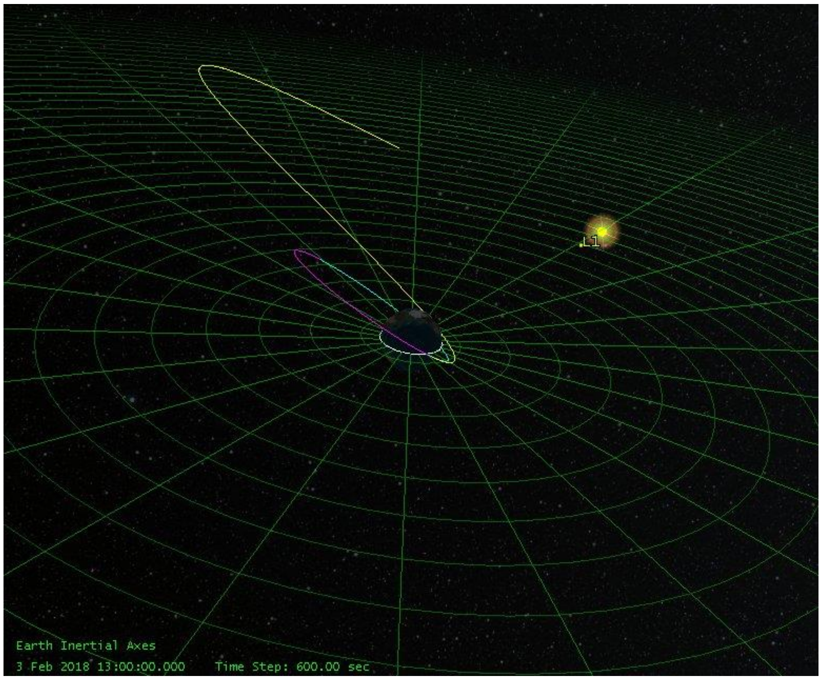

Figure 6: Hyperbolic Escape - Parametric View

At this point due to the velocity of the spacecraft, the Sun's gravitational pull and the fact that the spacecraft is escaping the sphere of influence of the Earth, the spacecraft will end in a heliocentric orbit.

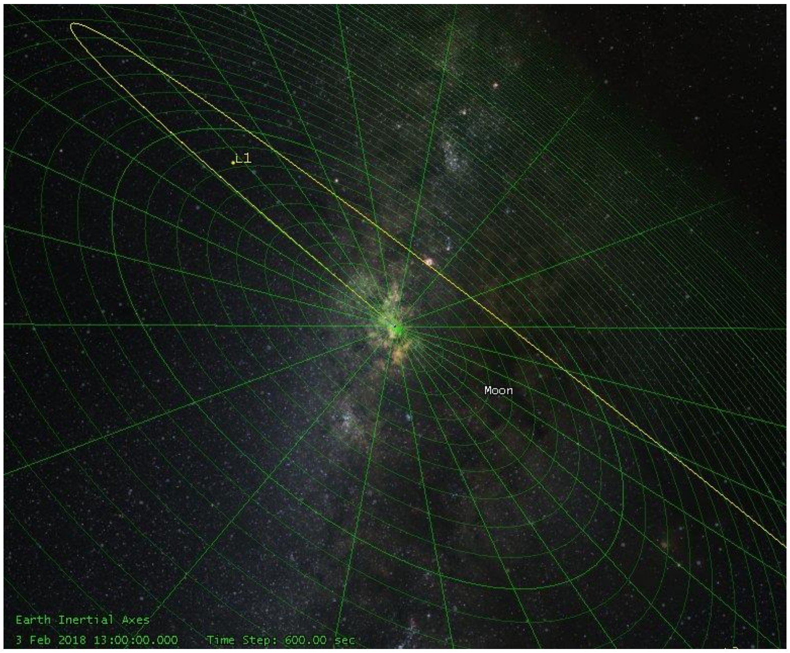

Figure 7: Heliocentric bound Trajectory – Parametric View

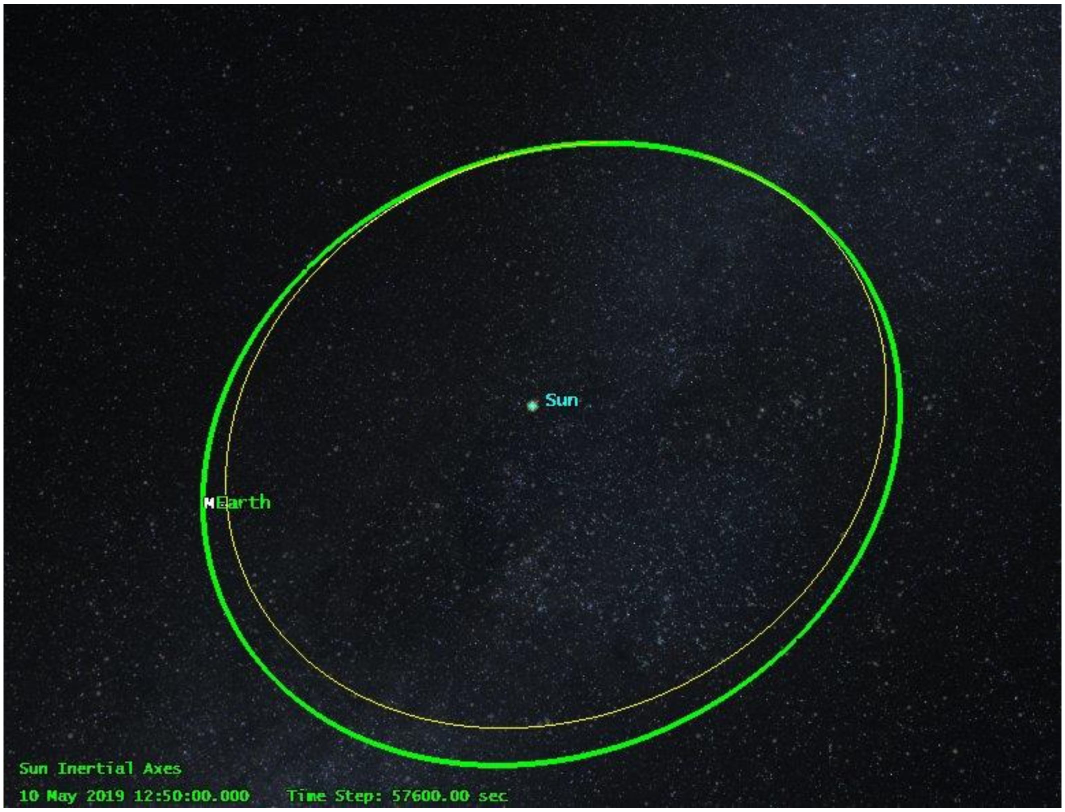

Figure 8: Earth Heliocentric Orbit compared with Spacecraft Heliocentric Orbit

An Earth trailing orbit, will facilitate communications, have more approximations with the Earth and in an Earth Trailing. While STK will give you most of the Keplerian elements needed, a verification of the period will allow for certainty in the simulation. Knowing that the apoapsis (a_{sun}) is 1.465×10^8 km and knowing the gravitational parameter of the Sun ($\mu_\odot$)[26] is $1.32715 \times 10^{11}\ \frac{km^3}{s^2}$, a simple calculation[4] will allow for the verification of the period (P), which is need to find how long it will take to do a full 360° data mapping of the Sun.

$$P = 2\pi\sqrt{a^3/\mu_{Sun}} = 30.114\ x\ 10^6 seconds = 348.544\ days \qquad (1)$$

Comparing this value with what is obtained from STK (*340.05 days*), a slight difference is present due to the lack of using all decimals. Now it's known that if in a scenario where not one, but four spacecraft were sent, a mapping could be obtained even sooner. While this orbit is ideal for science recording, it is farfetched for the budget and scope of the mission. Considering as well, the driving factor of the communications system, this could present a challenge if not equipped with the necessary hardware for deep space communications. Unfortunately, it will be discarded as a viable option for the initial Tier 1, but it could be viable for a full scale mission.

2. Sun Synchronous Orbit

An SSO uses the oblateness of the Earth to cause the orbit plane to rotate at the same rate at which the Earth spins around the Sun.[27] Considering the SSO scenario, a couple of assumptions are being made. The next possible launch to SSO is a European launch in the second half of 2016, aimed at 600 km[24]. Assuming a spot can be purchased for this launch and due to lack of information of the launch, it will be necessary to find the inclination angle to maintain a true SSO. Knowing that the altitude is 600 km and the radius of the Earth is *6,371 km* the semi major axis of the orbit will be (a) = *6,971 km*, from Vallado's *Fundamentals of Astrodynamics and Applications*[28] the following equation can be used to find the inclination (i) to maintain a SSO:

$$i = cos^{-1}\left(\frac{-2a^{7/2}\dot{\Omega}_{SunSyn}(1-e^2)^2}{-3R_{\oplus}^2 J_2\sqrt{\mu}}\right) \tag{2}$$

Assuming a eccentricity (e) of 0 and knowing that the second zonal harmonics of the Earth (J_2) is a constant, 10.826269×10^{-4}, the mean motion of the Earth due to (J_2) being ($\dot{\Omega}_{SunSync}$) is $1.991063 \times 10^{-7}\ \frac{rad}{sec}$[28], the Earth's gravitational constant (μ_{Earth}) being $3.986 \times 10^5\ \frac{km^3}{s^2}$ and knowing the radius of the earth ($R_{\oplus}^2$) and the semi-major axis (a), it can be found that the inclination angle (i) is *97.7925°*.

Using STK as a verification simulation of the calculations made, it can be seen that while the orbit is defined as a SSO, it is not receiving constant sunlight.

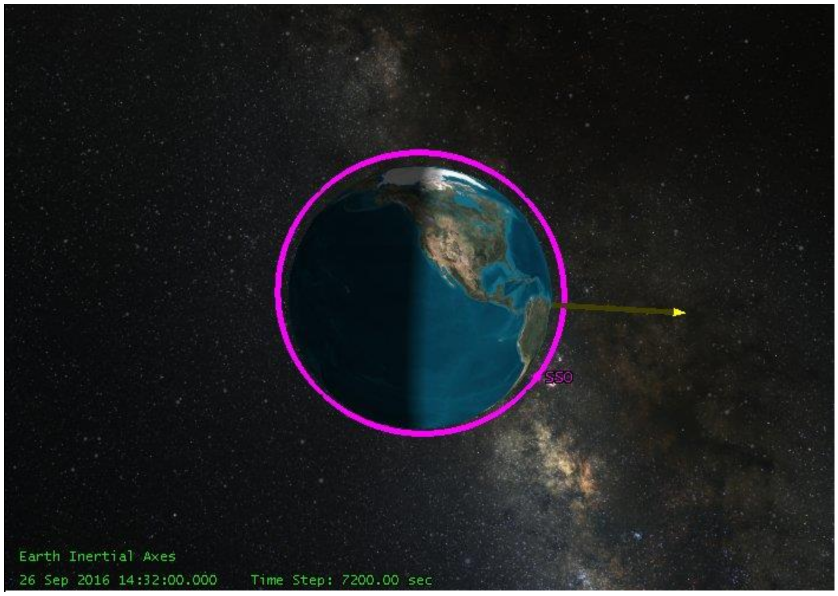

Figure 9: SSO with Sun vector

The following plot shows the lighting times, in one day. Which is not ideal, specifically for the type of science that is trying to be obtained.

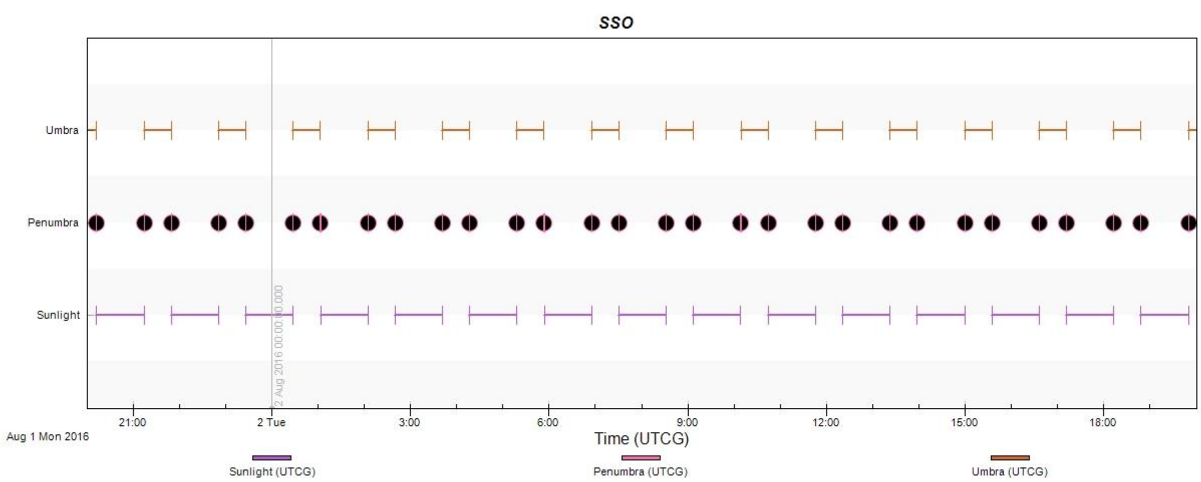

Figure 10: SSO Lighting Times

As mentioned before, one of the benefits of the sun synchronous orbit will bring is constant Sun visibility for a determined period of time, which for this orbit, is not the case. After some investigation, it was found that this is due to the Right Ascension of the Ascending Node (RAAN), which can be affected with the time the spacecraft is launched in orbit. What is required to find the ideal condition is known as a *Terminator Orbit*[29], which places the launch times when the sun rises (*6:00 hours*) or when the surface transitions from light to dark (*18:00 hours*), this will modify the RAAN from 310°, to 220.9°. The following charts will show the differences in the path of the satellite with this small modification.

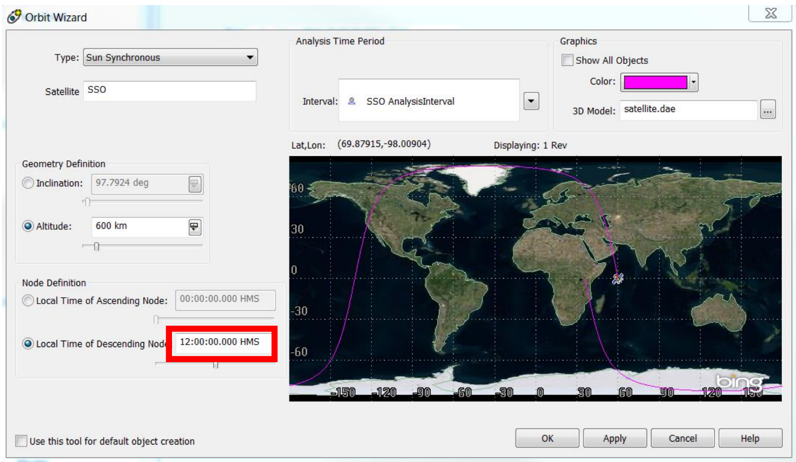

Figure 11: Original Configuration of Local Time of Descending Node (Non-Terminator)

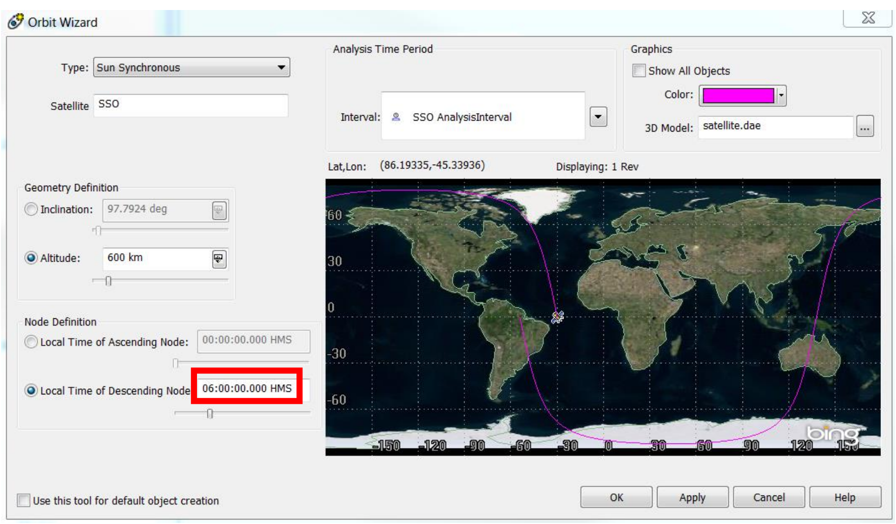

Figure 12: Modified Configuration of Local Time of Descending Node to 6 AM (Terminator)

With the STK visualizations of the *Terminator Orbit*, it clearly shows the benefits of using a 6:00 hours launch time.

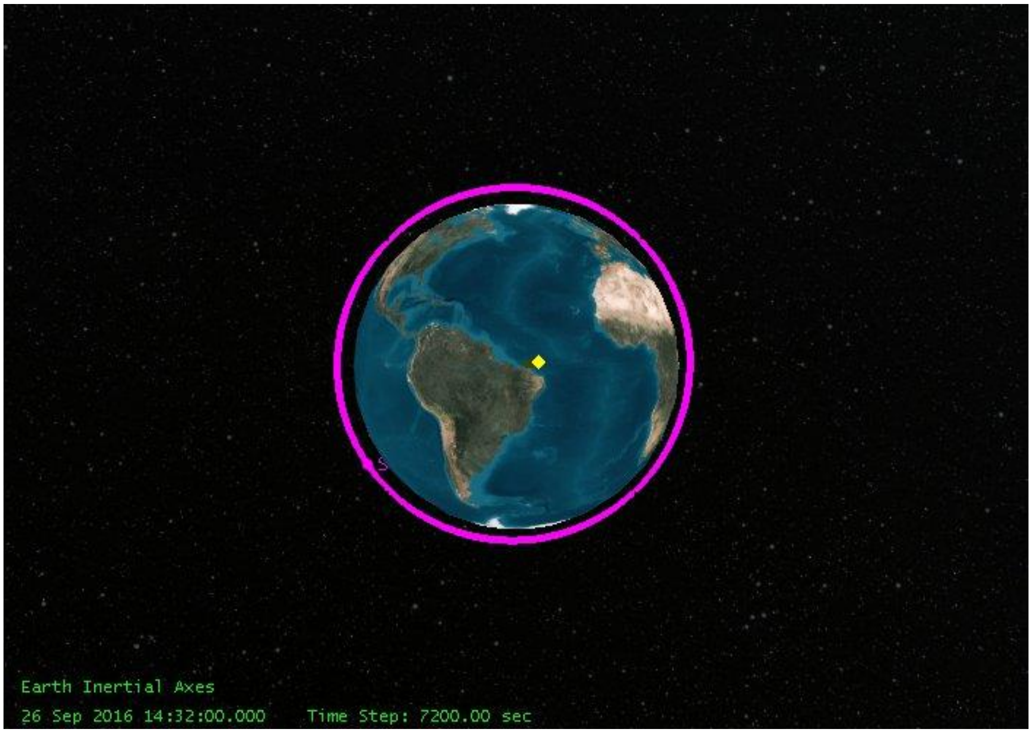

Figure 13: SSO Terminator Orbit Perpendicular to Sun vector (Yellow Point)

Considering launching on the European SSO launch the second half of 2016, set in the analysis to be in August of 2016, the benefits of the Terminator orbit would not start to show until after mid-September, but after this time the spacecraft will be in constant sunlight for until close to the end of February, which would provide 5 months of constant sunlight, ideal for power and Sun observation.

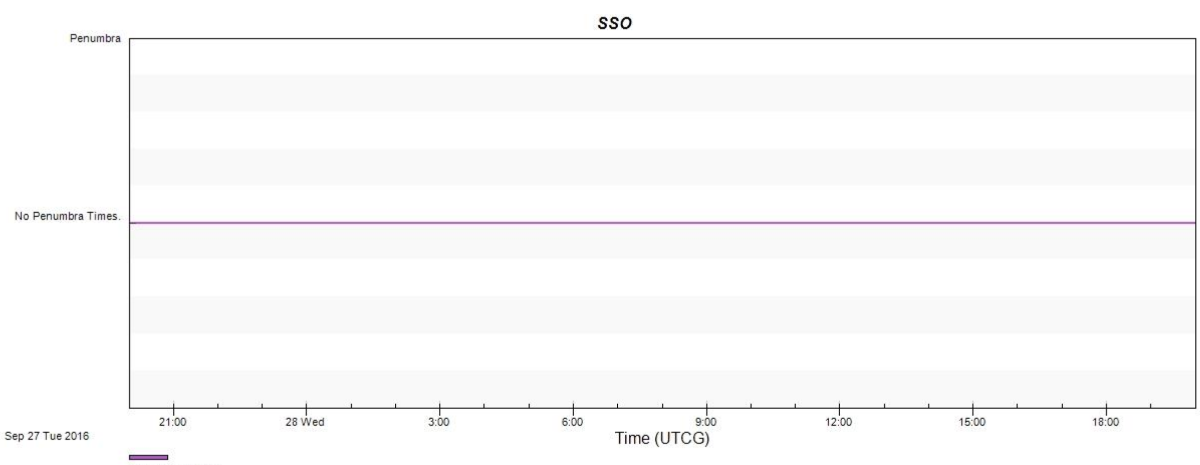

Figure 14: SSO Terminator Orbit Lighting time for Sept 27, 2016 (Constant Sunlight)

It will be assumed that the spacecraft will be deployed at a 97° inclination angle. Using the period (P) equation (1) but with Earth's gravitational parameters instead of the suns, ($\mu_{\oplus}$) being $3.986 \text{ x } 10^5 \; \frac{km^3}{s^2}$and a semi-major axis of *6,971 km* it can be found that the Sun Synchronous Orbit's period will be *5,732.33 seconds,* or *96.538 minutes.* Close enough to the data obtained from STK.

Figure 15: SSO Terminator Orbit Period (from STK)

The results obtained seem strange, the maximum lighting time that was obtained was 178 days of constant sunlight, far better that what could be achieved with a LEO orbit. This would assure positive margin in the power and maximize data recording, but accomplishing a 6:00 hours or 18:00 hours launch, is placing too many constraints on the launch provider. Since the mission is most likely not to be primary payload on the launch, imposing such conditions are not in the realm of possibilities, but could be applied to a future, Tier 2.

3. Low Earth Orbit

For the LEO scenario, other assumptions have also been made. The most common way educational American Cubesats get to LEO is being deployed from the International Space Station, which orbits at around 350 km[21] at an inclination of 51.6°, values that were used to calculate the orbital parameters for this study. However from TechEdSat data has previously been deployed from 400 and 410 km. A higher orbit, will increase lifetime of the mission allowing for slower de-orbiting.

Launch mass for astronaut resupply and ISS payload missions usually do not fill up their capacities, permitting for extra mass allowances on their launches, which is usually offered to universities or other companies to fly Cubesats, which are subject to a waiting list. With this in mind, the analysis that is done for a LEO is much simpler than for the previous two scenarios. Not unlike the SSO, LEO is also basically a circular orbit, or at least it can be assumed that way, meaning that the orbital velocity will be constant. Knowing that the gravitational constant of the Earth (μ_{Earth}) is 3.986 x 10^5 $\frac{km^3}{s^2}$ and that the semi-major axis *a* is *6,721 km.* Also using the previously presented period (P) equation (1) it can be found that the period is *5,483.5 seconds,* or *91.391 minutes.*

Knowing the period of the orbit is not enough for the suggested LEO scenario, unlike the SSO, due to the inclination angle of the orbit, the spacecraft will encounter eclipse times, which will affect observability time to the sun. Using the following equation, the maximum eclipse fraction (f_E)[30] can be calculated. The values needed to calculate are all values that have been previously used, h – altitude of the orbit and R – radius of the Earth (*6,371 km*).

$$(f_E)_{max} = \frac{1}{\pi} cos^{-1}\left[\frac{\sqrt{h^2+2Rh}}{(R+h)}\right] = 0.3968 \tag{3}$$

Now that the eclipse fraction has been calculated, it can be multiplied by the period obtained and it can be found that the eclipse time of the orbit is *36.11 minutes*. Subtracting from the total period time, the duration in sunlight is *55.281 minutes.* Why does this number matter? This will allow not only know how much data can be collected per orbit, it will also allow to set parameter values to the duration entire duration of the mission. Verifying with STK, the data that is obtained for period (P) was *5492.28 seconds* which roughly matches the orbital period (P) that was calculated.

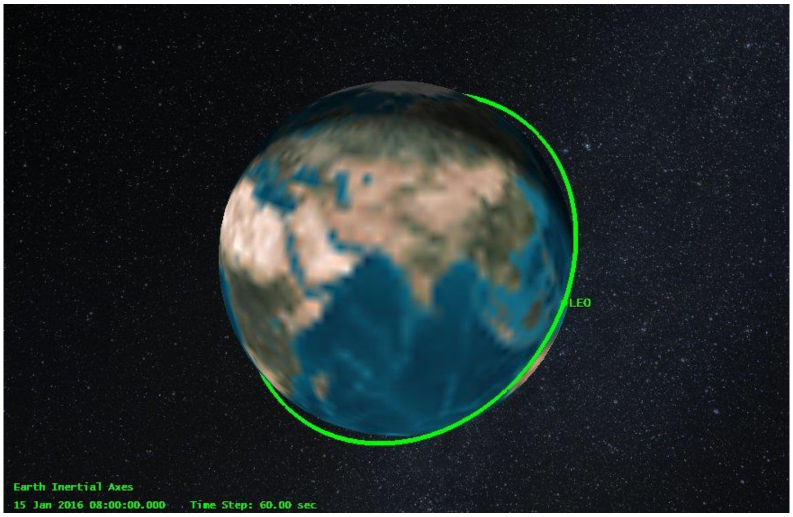

Figure 16: LEO

The lighting times found are also similar to the calculations done. It can be seen in the following chart and with the information from the report that generates the plot, that the mean lighting duration is *3237.54 seconds*, or *53.96 minutes*, which is also very close to the calculations done.

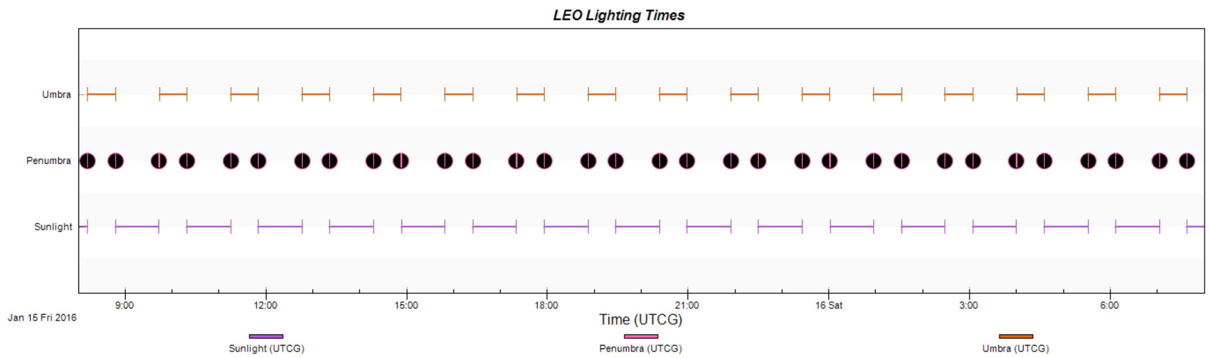

Figure 17: LEO Lighting Times

After all of these analysis, calculation and simulations, it would be thought that the answer is simple, obviously the heliocentric orbit is more than ideal in every single way. While this is very true, from the perspective of an orbital trajectory designer, whom gets bored with simple circular LEO's, and for the scientist who would gladly accepted the science data obtained from an experiment of this scale could generate. In the attempt to comply with the mission within the bounds set on the mission, which means understanding the cost and bus constraints. As mentioned before the analysis was done, other factors are involved when taking decisions such as the one that is being mentioned. While many are attempting to send cubesats into interplanetary space, it has not been done yet, and the economical idea of the cubesat somewhat is lost, due to the time needed to invest in a mission of this magnitude. In the case of this study, proceeding with a heliocentric orbit, would be beyond the costing and team capabilities, and repeating the limiting factors of what's available for the cubesat platform.

The SSO is slightly easier to achieve, but with the seen launch constraints, the possibility of ending up with the perfect launch opportunity and finding space on the launch. While all "nice to haves" it could place the mission at risk, which could possibly lead to failure.

C. Power

The power subsystem can be then divided into 4 parts:[21]

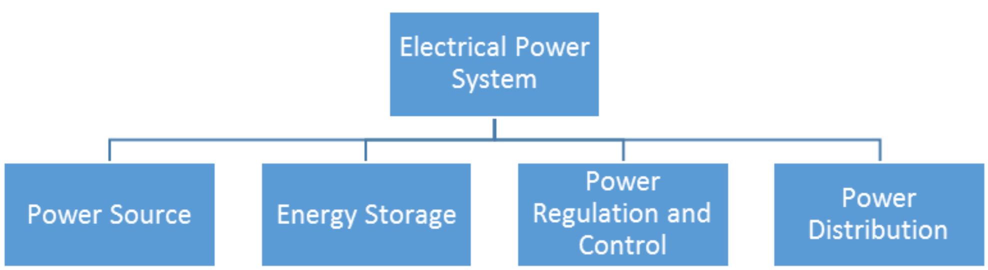

Figure 18: Electrical Power System Decomposition as displayed in "The New SMAD"[21]

Larger spacecraft could have this decomposed even further, but for the study described in this report the *Power Regulation and Control* and *Power Distribution* will be combined. The task will be done with a control board that is specifically designed for the spacecraft. Many instruments or hardware require very specific inputs and outputs. The power control board will have the task of assuring correct voltages and currents are sent as they should throughout the spacecraft.

The two left are the *Power Source* and the *Energy Storage*. In this case in particular it refers to the Solar Panels and Rechargeable Li-Ion batteries. The solar panels that were utilized in this study are the TASC.[31] For each allocated U of solar panel area, 10 pairs of TASC are fit to assure space for circuitry, as explained by Stanton.[32]A large, more efficiently sized solar panels are in the possibilities for the final build, but more margin into the design can help for any possible future issues.

Figure 19: TASC Circuit configuration[32]

To be able to size the solar panels there is a need to find the total power obtained from the Solar Arrays (P_{SA}) using the following from *The New SMAD*:[21]

$$P_{SA} = \frac{\left(\frac{P_e T_e}{X_e} + \frac{P_d T_d}{X_d}\right)}{T_d} \tag{4}$$

As calculated in the Mission Design section, the period of the orbit (P) at 350 km is *91.39 minutes*. Knowing that the period for eclipse (T_e)is the same as the max eclipse fraction (f_E), a value of *36.11 minutes* is obtained, thus the day time period (T_d) is *55.281 minutes.*

The values for (X_e) and (X_d) *represent* the efficiency paths from the solar arrays to the batteries. The values used for these will be the ones provided by *The New SMAD* for peak-power tracking which are X_e = 0.60 and X_d= 0.80.[21] (P_e) and (P_d) represent the values of power consumed and although not usually not meant to be equal, to augment the positive margin of the power system are going to be set equal.

This will allow to know how much power is available, and with the adequate sizing of the batteries the determination of the power budget is closer to being obtained. Continuing the assumption that (P_e) and (P_d) equal, assuming that these values are around *19 Watts,* the power the solar array must generate (P_{SA}) is *44.5 Watts*, with current design, the spacecraft bus will have 0.0394 m^2 of solar array and with the solar illumination intensity value found in *SMAD*[21] of *1,367* $\frac{W}{m^2}$, even equalizing the (P_e) and (P_d) the solar arrays sill generate *53.85 Watts* leaving about a 20% of positive margin in the power system. This with the addition of batteries, will allow the system to go into eclipse without a worry of losing power.

D. Communications

The initial studies done for this report have limited the communication to power and mass. The spacecraft is planned to use the Iridium constellation which orbits at roughly 780 km[33], for command and control. This will allow the spacecraft to have low cost communications and roughly negligible ground operations cost. Calculations done by Guarneros[16], estimate the coverage to be 9.988 hours out of every 24.

The communications subsystem for the project is on the critical path, future TechEdSat flights will demonstrate 10-50 Mbps on Ka-Band, which will help the success of the mission, which is what will be used for science data.

With several assumption an initial, broad data budget could be made. If the it can be assumed that a communication pass could be done twice a week, for at least 3 minutes, and at a data rate of 50 Mbps. Adding 30% of positive margin to stay positive, that would give at least 1.5 GB, which considering the initial Tier 1 requirements would give much more than the excepted (100 MB) and leaving the Iridium Constellation system also as a backup.

E. Control and Data Handling(C&DH)

The C&DH refers to the on-board processing of the spacecraft. This depends highly on the complexity of the instrumentation and/or other subsystems on-board. A high fidelity ADCS system will require a large amount of processing power. Which brings the effort to the conclusion of having a separate processing unit dedicated to the ADCS with interface which is to be determined further in the process.

The main C&DH is composed of an Arduino Mega board that will run the Flight Software, communication system, power system and passive thermal observation system. The Flight Software is estimated to be less than 10k lines of code.

The parametric study being performed is considering mass and power figures of the C&DH, at the moment no further analysis is required.

F. Structure

With so many cubesat that have been flown to date, a structural analysis for a cubesat that will have not loads that differ from every other cubesat built will not require strong analysis. What the mechanical engineer will have to do, is accommodate all of the hardware in the small volume the cubesat occupies. Initial mock up drawing have been done to for the cubesat project, by the mechanical engineer in the project Ricardo Amezquita. This design has simple deployable solar panels to augment the solar facing area, thus increasing power generation. Also can be noted the side hole which will allow the soft X-Ray spectrometer to take science data.

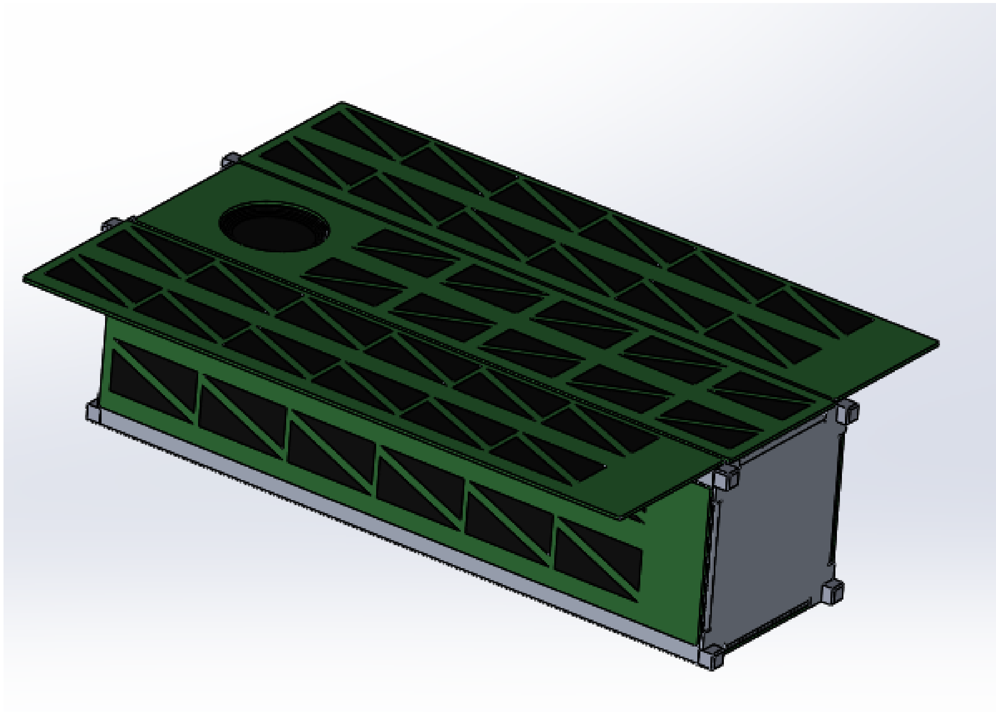

Figure 20: 3U Cubesat with Deployable Solar Arrays per R. Amezquita

Instead of going with the conventional CubeSat Kit 3U Solid Wall Structure made of Aluminum with a mass of 0.439 kgs.[34] The stud will utilized extruded aluminum square tubing, which has the same width and depth of a cubesat and cut to the length of a 3U cubesat. This method has been used several times by the TechEdSat team, cutting cost up to 80%. Which when considering the cost of CubeSat Kit structure, can translate to a lump sum of money.

G. Payload

The payload subsystem will be flow in a set of different tiers that will allow to build towards a full science mission with. Initially the payload planned to be utilized is an Amptek X-123 X-ray detector.[35] Constraints found in within the platform, could leave the instrument to be flown in a future mission Tier.

Limitations for instruments flying on Nano-sats and Cubesats are mostly due volume. While some instruments have begun to be designed in a smaller form factor, optical instruments are forced to avoid these smaller form factors, thus assuring better quality in their science data. Even with all of this, studies of a potential Coronagraph are in the interest of the stake holders for future Tier's. A coronagraph could use the initial tier objectives to help achieve its strict stability control requirements.

H. Attitude Determination and Control System (ADCS)

While initially the ADCS system chosen to do the study with was the MAI-101.[2] The project's controls engineer, Alexander Westfall has designed an initial ADCS control algorithm. Which is not unusual in space mission projects, the initial concept is constantly changed, due to budget cuts, changes in administration, can all cause a drastic change the initial objective in the cubesat.

The initial idea of the ADCS was to utilize a COTS system and generate an algorithm with the system that would allow for precise 3-axis control pointing. Which would allow the amount of control that can be found on any plane, pitch, yaw and roll.

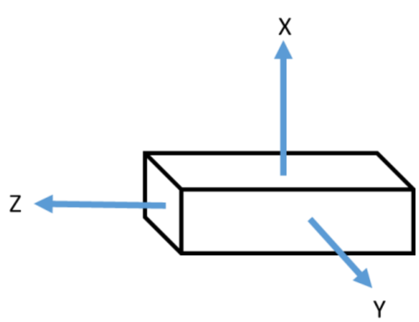

Figure 21: 3U with Axis X, Y, and Z represented

What exactly does this mean? This means that the spacecraft will have the ability to spin in clockwise or counter clockwise around any of the directions of its axis and the combination of the three. In other words, full control over its movements, making it necessary for accurate pointing. Since as the spacecraft orbits around the Earth, it also starts to slew away from its pointing orientation, requiring high frequency sensors and actuators that will allow it to correct and not lose sight of it objective or target. Accurate pointing on a cubesat to this day has not been accomplished, and it is because of not only the volumetric limits of the spacecraft bus itself for all the apparatus needed, but the technological limiting factor that would require to miniaturize as well the instrument to do observations. The factors limit the possibilities of what can be done. With the constraints found throughout the project, there was a need to deviate from this arduous task.

The need to seek for something that could contribute to the space community became priority and was found within the TechEdSat bus needs. While the TechEdSat carries onboard an Exo-Brake that allows for an accelerated de-orbiting and passive control, the need to stabilize the spin of the spacecraft was found. The brought the team, but mostly the control engineer to understand and find out what could be done to solve this still very complicated problem.

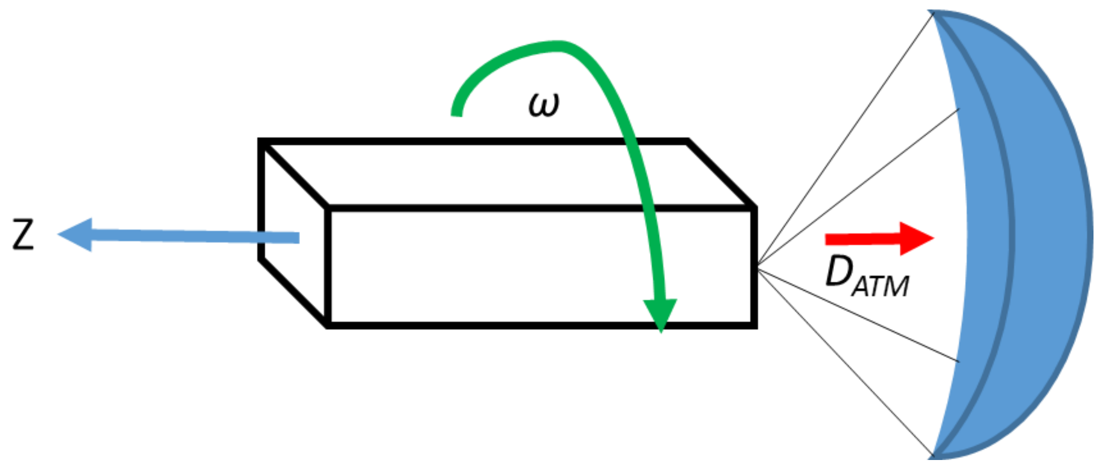

Figure 22: TechEdSat Representation: Spin Rate with Exo-Brake

A single reaction wheel with magnetorquers would be incorporated to the TechEdSat, with the dedicated on-board computer to run the control algorithm.[36]

With a stable spin rate achieved, the possibility to stop the spin for a rudimentary pointing system could be utilized to observe the sun while in sunlight times, testing for the robustness with in the algorithm.

I. Thermal

For most cubesat active thermal control has not been required for most missions. Being in a LEO and ideal placing of the hot components such as the C&DH and ADCS with components that could get cold such as the batteries, would allow for the spacecraft to stay at a survival temperature. Multi-Layer Insulation (MLI) blanket has been the most commonly used passive thermal control that has been utilized for cubesats.

This with the before mentioned placing, the MLI blanket and the multiple flight heritage flights of the TechEdSat bus, should suffice to comply with the thermal analysis needed to this point.

VI. Architecture Selection

The idea of architecture integration starts with the selection of the hardware required to build the system. The best approach to define these individual pieces is a series of trade studies for the most impact producing subsystems and decisions to be made for the system. While many of the decisions have been made a trade study will reaffirm initial estimations on the best decision is, and if that was the path that was taken.

A. Spacecraft Bus Trade Study

The spacecraft bus selection was done after a trade study done considering several important factors that relate to the mission. These factors are technical and non-technical relating mainly to cost and schedule, which are highly important for a short duration, cost friendly oriented space mission.

Spacecraft Bus					
Merits of Interest	Weight	TechEdSat	New Design	PhoneSat	Small Satellite (Non-Cubesat)
Cost	5	9	3	6	1
Supportability	3	9	10	5	2
Flight Heritage	3	9	2	5	8
Mission Scalability	2	7	8	6	8
Risk	4	6	3	5	7
Lifetime	3	5	7	5	7
Totals		152	100	107	100

Table 4: Spacecraft Bus Trade Study

It can be seen that for what the project believes are the most important merits, the TechEdSat bus excels beyond the others. This does not at all diminish the other candidates taken into account. The parametric design that will be shortly reviewed takes into account a TechEdSat and a sort of new design hybrid to appreciate what can be done with the model.

To further discuss this, some details shall be expressed with how the grading process was realized. While the small satellite is probably, in the author's opinion, the best option for a full scale mission to a heliocentric orbit, this goes outside of the concept that the project was initially thought of. A small satellite (around a 100 kg) will be able to select from a variety of flown hardware that fairly exceeds what can be found for the cubesat standard, this of course comes at a price. Most spacecraft of these sizes tend to range from \$20M up to \$100M, or more in certain cases. The Phonesat bus would also work fine, but no partnerships exist with SJSU that could allow some sort of collaboration. This leaves the creation of a new design, which seems to be the trend these days. The problem starts when the designer start to see lack of reliability in the parts that were purchased, or problems with integration when doing validation testing. This with the fact that anything that is purchased with a "Space Rated" sticker could change the price by an order of magnitude. This leave the TechEdSat bus, which has grown and optimized its capability by growing from a 1U to a 3U and with different features.

B. Attitude Determination and Control System Trade Study

The ADCS bus selection was also done after a trade study that was done parallel from the on in Westfall's study[35] with some different parameters that analyze the ADCS as a system and not only actuators. As with the spacecraft trade study, it was done with technical and non-technical merits, so it can be viewed it from a high level system integration perspective.

ADCS Systems					
Merits of Interest	Weight	MAI-101	BCT X-ACT	New Design	Micro Arc Thrusters
Cost	5	6	4	9	7
Reliability	3	6	9	5	3
Flight Heritage	3	2	3	1	0
TRL	3	2	6	2	2
Complexity	4	4	3	5	8
Totals		76	86	89	82

Table 5: ADCS Trade Study

With this trade study, the outcome to create a new design for an ADCS system seems to be the best option for this subsystem. That could seem strange, since for the spacecraft bus the new design was quickly discarded. It should be noted that it is not the same effort to create a new subsystem that what is to create an entire new spacecraft, with no integration heritage. Again, nothing to diminish the other options, the parametric model took into account the parameters of the MAI-101 as a baseline. When considering the best option, while the two fully built systems, MAI-101 and BCT X-ACT, offer outstanding hardware with amazing capabilities, the cost don't allow small funded project to use them, especially when considering the extra units that are required for testing. The thruster, while promising, seem to be far from where the technology can be called "ready for flight". Leaving the in-house built system as the viable option, which for the objectives of this study should suffice.

VII. Initial Models

The model run is being done so with NASA Ames Research Center parametric tool ATLAS.[37] This model was created using the parameters found in the previously mentioned research and analysis done. Several models were generated taking several different factors into consideration which will be evaluated in the following paragraphs. The model allows and encourages modification and "playing" with certain parameters to understand the affects in changing these parameters.

The software relies on two databases and allows input via web browser, Excel and Matlab. The software is designed to allow a single GUI to be used by a standalone user or with a team of experts. Finally, the parts and models used in the design of the software are openly available to all members of the design team.

A. Power Cycles

The analysis was done considering different power modes. What these power modes define is the utilization of the spacecraft hardware during these modes. As seen in *figure 4* these mode will be very similar to will be seen on the concept of operations of

	Name	Duration (min)	Power Draw (W)
0			
1	Initialization	30	1.5
2	De-tumble	15	10.9
3	Sun-Point	10	12.5
4	Science		7.1
5	Eclipse	37	2.6
6	Downlink	15	8.7
7	Fail Mode	93	2.5
8			
9			

Figure 23: Design Power Modes

a spacecraft mission but this is defined in the Power Mode Order, as seen in *figure 5*.

The way the power modes are defined are with the duration that is inputted by the systems engineer. When a duration is left blank it will take the time remaining to finish the orbit. This means that the power modes, when initiated, will start an orbit taking into account the period of the orbit and the eclipse time.[21]

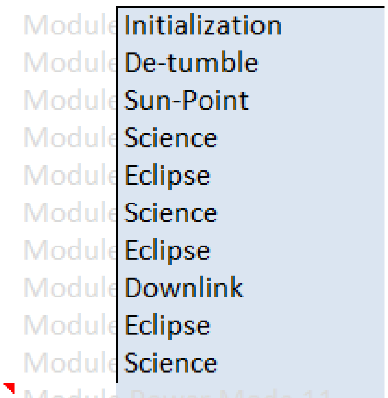

Figure 24: Designed Power Mode Order

The period (P) was previously calculated to be *91.391 minutes*, and with the calculated maximum eclipse fraction (f_E) that was *0.3968* that gives a sunlight time of *55.281 minutes*. With this known, the power modes can be strategically set to fit the concept of operations.

This can be done with simple arithmetic, completing the modes to complete an entire period, taking into account eclipse time and solar time. This will allow for modifications and assure to obtain as much data/science as possible, having the capability of downlinking it and without surpassing the depth of discharge of the battery.

B. Mass and Power Budget Information

The mass and power budgets are not constant when running the parametric model, because in the different power modes the hardware is not running at max power consumption or could be in standby. The three modes displayed are the most important to the author’s criteria in the operations of the spacecraft. Due to the lack of a propulsion system, no mass changes are expected in operations of the spacecraft, this will cause to have a constant mass budget though out the following figures.

1. Downlink

The mode to be discussed is the Downlink mode. This mode is meant to last 15 minutes. The assumption is being made that it will interrupt some of the science operations, but assures that communication isn’t interrupted due to power constraints, if run in eclipse.

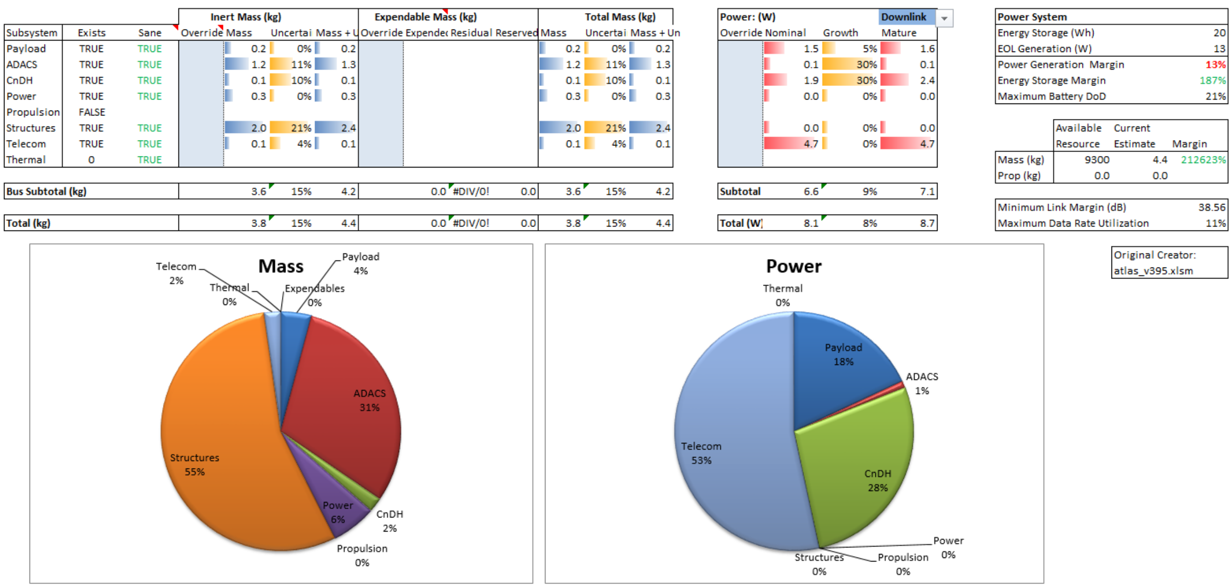

Figure 25: Mass and Power budget of Downlink Mode

It can be noticed that the telecom section take more than 50% of the power while in this phase. Power generation margin is low but still positive and the energy storage margin enough to keep the spacecraft alive for a while if required.

2. *Science*

The second mode to be analyzed is the Science mode, which repeated more than the Downlink. This will be allowed thanks to the on board memory, that will allow for storage of data while the Downlink mode approaches. The Science mode also does not have a defined duration. This is assured so that in combination with the Eclipse mode, the spacecraft can record as much science as possible.

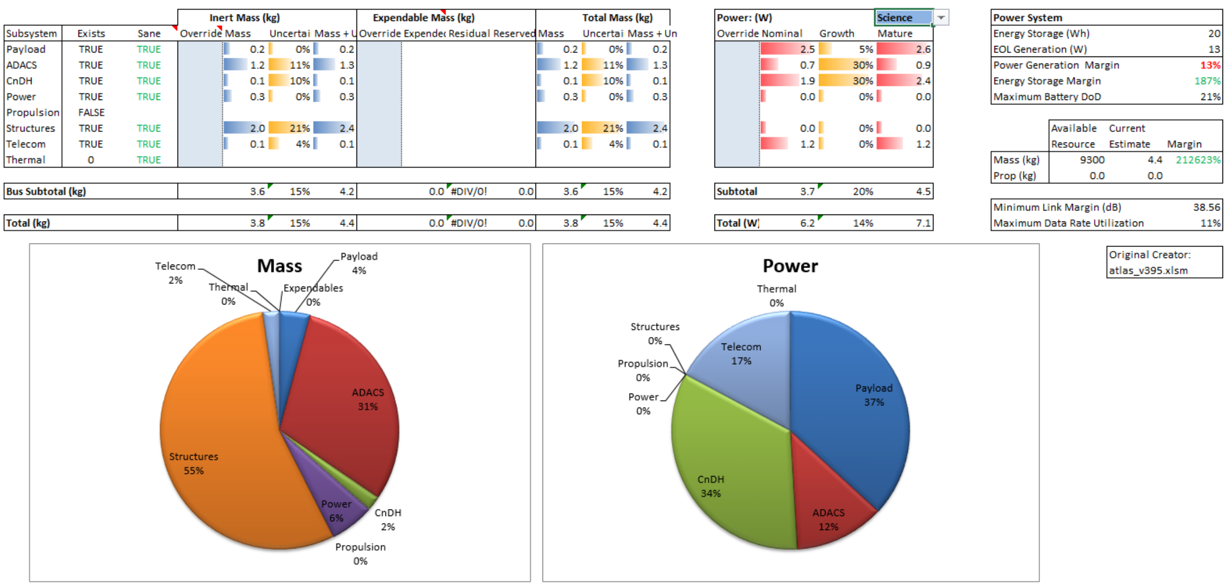

Figure 26: Mass and Power budget during Science Mode

In this case, it can be noticed that telecom (communications) has reduced to less than a quarter of the power and the Payload and C&DH are taking a larger portion of the power.

3. *De-tumble*

The De-tumble mode is a mode that, when everything is going accordingly, should only happen once. This is immediately after being deployed from the launch Vehicle or ISS, which usually require a 30 – 90 minute timer to allow the spacecraft to gain distance between itself and the deployment system. De-tumbling will allow the spacecraft to go from an unstable, uneven state of flight and attitude to a controlled attitude state and preparation for science observation.

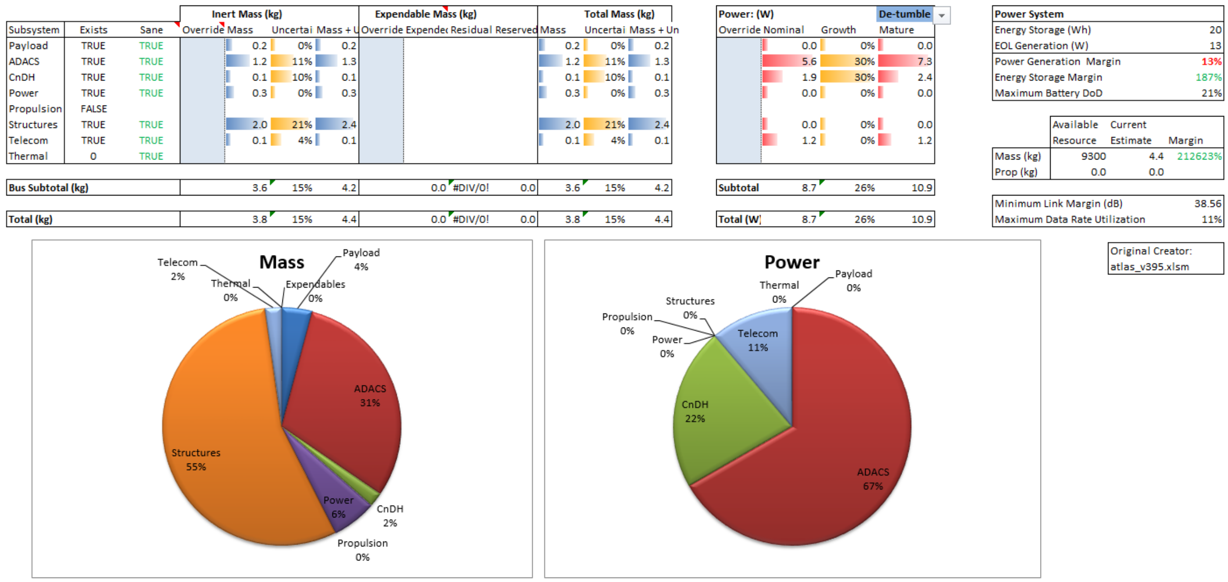

Figure 27: Mass and Power budget during De-tumble Mode

As expected, the subsystem taking the majority of the power is the ADCS. This mode will have the Reaction wheel running at maximum power to correct and have the spacecraft in a stable state. Maneuvers throughout operations are going to be needed but these will be only slight maneuvers to correct the natural slew of the spacecraft due to atmospheric drag and solar pressure.

C. Battery Discharge

The report has mentioned battery discharge several times previously. This refers to the recommended depth of discharge suggested by the manufacturer of a battery. After several cycles, batteries tend to lose operation life or struggle with retaining charge. Assuring that the depth of discharge is not reached will assure maximum life of the battery

The following graph will show, with the summation of the power modes and taking into account eclipse times and solar exposure time for the solar panels, what the battery is going through in a day of operations.

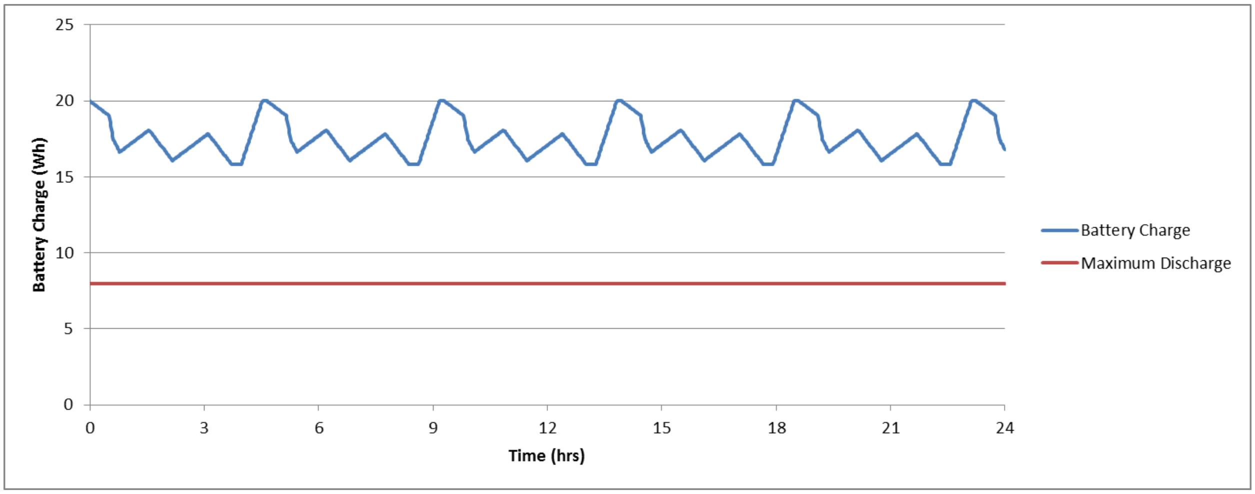

Figure 28: Battery Charge/Discharge during 24 hours of Operations

D. Main Equipment List

The main equipment list is a summary of all of the hardware included in the spacecraft. This will include specifications of the subsystems and what components form part of each subsystem.

Subsystem	**Name**	**Type**	**Model**	**Quantity**	**Unit Mass/kg**	**Unit Power/W (Avg)**	**Unit Power/W (Peak)**	**Unit Power/W (Standby)**
Payload	X-123	Instruments.Sensors	X-123	1	0.18	2.5	4	1.5
				1				
ADACS	MAI-101	Actuators.Wheels	Maryland Aerospace MAI-101	1	0.693	0.3	2.4	0.06
ADACS	IMU	Sensors.IMUs	EPSON M-G350 / S4E5A0A0A1	1	0.007	0.1	0.2	0
ADACS	Sun Sensor	Sensors.Sun	STAR Fine Sun Sensor	1	0.35			
ADACS	Magnetorquers	Actuators.Torquers	NASA ARC Torque Coil for Nanosats (Conceptual Design)	3	0.049	0.09	1	0

				6				
C&DH	ADCS Computer	Computers	Beagle Bone Processor	1	0.03968	0.85	1.75	0.85
C&DH	Main OBC	Computers	Arduino Mega	1	0.036	1	1.5	0.5
				2				
Power	Battery	Batteries	Canon BP 930 (TechEdSat)	1	0.225			
Power	Solar Panels	SolarArrays	Spectrolab TASC	0.039432	1.028			
				1.039432				
Structures	3U Structure	Other	CubeSatKit 3U Solid Wall Structure (p/n: 703-00245)	1	0.439			
Structures			Cabling Mass					
Structures			Secondary Structure					
Structures			Primary Structure					
				1				
Telecom	Iridium	Transponders	Iridium 9602-I	1	0.03	1	7.5	0.225
Telecom	Globalstar	Transponders	Globalstar GSP-1720	1	0.06	3.65	5	1
Telecom	Antenna	Antennae.Local	AstroDev S-band Patch Antenna	1	0.004	0		
				3				

Table 6: Main Equipment List (MEL)

With all of this simulation and analysis allows to believe the path being taken is correct, continue with a lower risk, with everything closing. It’s common that issues happen further in the process, in fact, it’s strange when that doesn’t happen. This initial calculation allows for credibility in what is planned to do.

VIII. Reaction Wheel Design

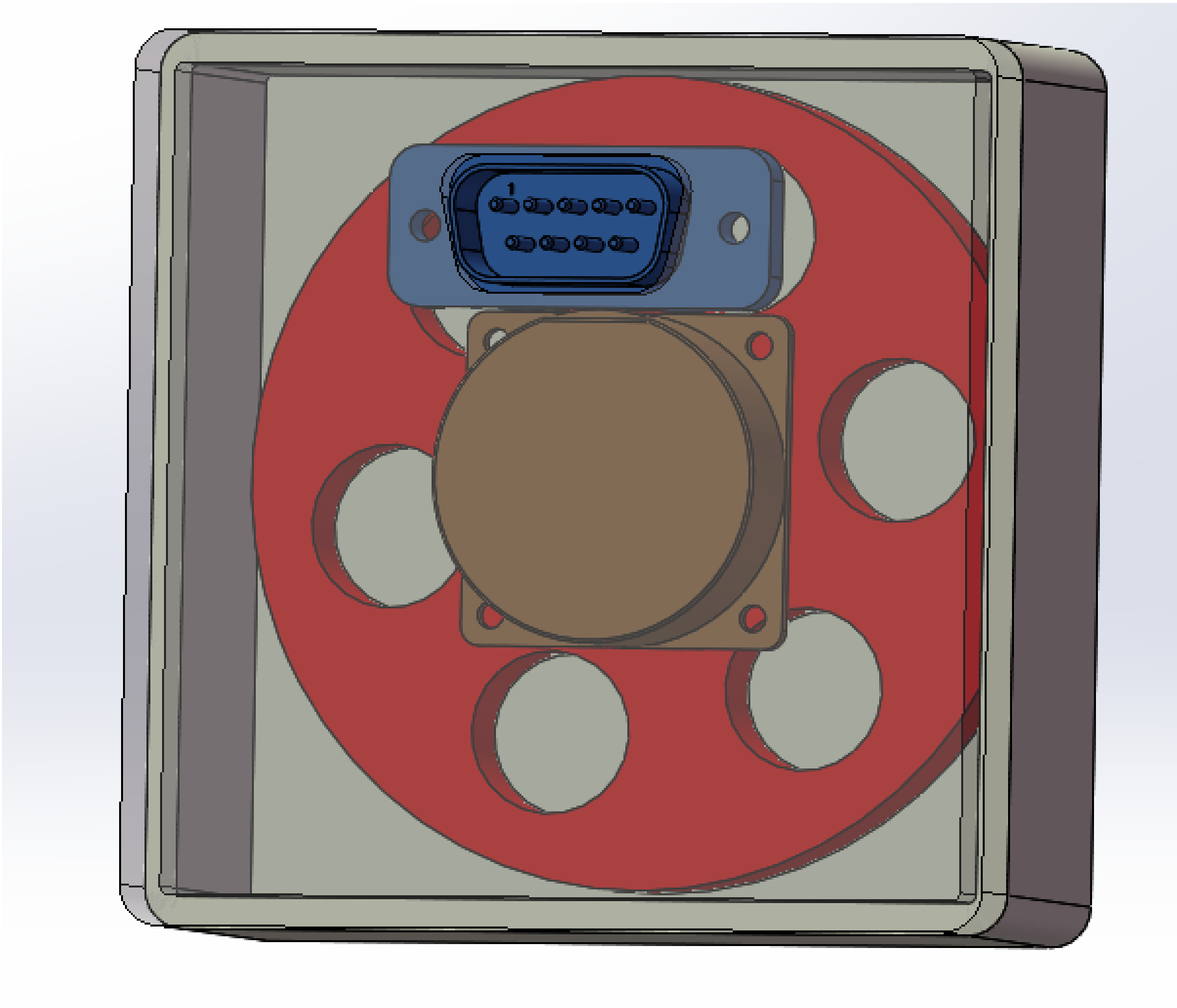

Figure 29: Reaction Wheel Assembly – Motor (Bronze), Wheel (Red), RS-232 Connector (Blue) and Case (Silver)

The previous figure shows the Reaction wheel assembly to be built for testing and eventually flying on TechEdSat. The motor utilized has been tested at Ames for several studies and is made by Faulhaber[38]. While possible that the reaction wheel could cause some disruption in the TechEdSat's power unit, the possibility of adding batter capacity has been discussed with the TechEdSat group, and does not affect their other mission objective.

IX. Integration with TES

The TechEdSat team will provide an Interface Control Document (ICD) for integration of the Spacecraft. Initial integrations conditions are stated in the following table:

Initial Interface Conditions	
Power	The ADCS system shall not consume more than 4 watts of power during operations.
Mechanical	The ADCS shall not occupy more than 0.5 U of space within the TechEdSat Bus.
Software	The ADCS software shall be written in Arduino.
Data/Power Interface	The ADCS system shall interface with the TechEdSat but through a RS-232 connector.

Table 7: Initial Interface Conditions

X. Initial Test Planning

While the initial testing was planned to take place at NASA Ames Research Center, bldg. N269, Spheres Lab, taking advantage of its testing fixture utilized for semi-frictionless 6 DOF movements.

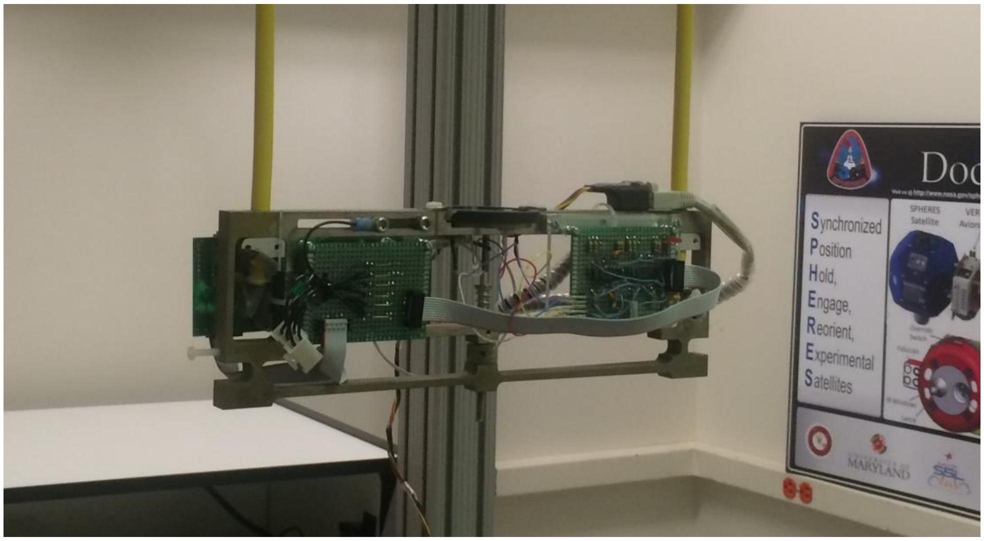

Figure 30: NASA ARC SPHERES Test Fixture

Due to budget and scheduling restraints the NASA ARC SPHERES Test Fixture will not be able to be utilized for the testing, but a simpler test fixture that will comply with the spin requirement was designed and is being built for testing. The cubesat model will be hung from a string to a low friction sting and will be spun up to an undetermined rate and the reaction wheel system will actively control the spin rate.

XI. Test Preparation

Testing can be one of the most exciting, defining and, time consuming process of the spacecraft design, development, and flight process. This will involve 2 types of testing: Verification and Validation Testing.

Verification testing will be the first of the verification and validation process to be done on a product. It will involve assuring that all system requirements are being accomplished as originally stated. In the end this means, was the product realized right? Testing of this type will assure that the end product spins as it should, or doesn't, all this depending on the specifications of the requirement. The type of testing that is to be realized for the final end product to be integrated with the TechEdSat will involve mostly verification testing.[39]

Validation testing is a very important task of the spacecraft development and integration process, which will revolve around the concept of operations of the mission that is being planned. The TechEdSat bus is meant to be designed to launch o various type of launch vehicles, and be deployed from the ISS, but it is not limited to that. This means that the bus must be validated for more than one launch vehicle condition, this is where the validation process and concept of operations meet. The validation testing phase will not involve assuring something can spin as its requirements state, but that it can actually sustain its spin while maintaining the integrity of spacecraft, and continue to function as planned. Validation will assure the spacecraft will indeed have the capability to survive all of the detailed operations it must go through. While validation is an essential portion of the spacecraft development, this will be out of the scope of the analysis and work done, for this is in a further phase of what is planned to be done with the ADCS unit to be developed. The testing that is to be performed (and further described) will be more in the lines of validation testing.

The test to be performed is simple, a 3U sized cubesat structure with the reaction wheel mounted to its center of gravity, while hanging from a designed 8020[40] aluminum structure. Besides the structure, the test will require a couple more pieces of hardware to allow the interfacing between the 3U cubesat structure and the 8020 aluminum structure. The following CAD screenshot is the design that will be utilized for the testing.

Figure 31: 3U Cubesat integrated with Spin Test Unit

The testing will have an onboard Inertial Measurement Unit (IMU)[16] that will allow to measure initial spin inputted into the system. The reaction wheel will detect the spin and specify an output spin rate to slow down the spin to a desired 1 RPM. This is further described in the ADCS algorithm design by Westfall.[36]

The test will take place in the Engineering Evaluation Laboratory (EEL) of NASA Ames Research Center. This laboratory is equipped with a vacuum chamber that can allow for testing, to assure no air pressure will factor in the testing.

Figure 32: Vacuum Chamber at EEL in NASA ARC

The test to be perform is essential to the verification of the requirements and the approval for flight.

XII. Future Work

While this study ends, work is still aplenty. Which sound about on right, anybody whom has worked on a space mission will understand this. The priority of this mission for utilization of NASA ARC's facilities is at the bottom of the list, but it's on it, which I believe is more than what many could say.

Hardware build is planned to be early next year. With this in mind, the author hopes to test the hardware in the month of April 2016.

As mentioned many times throughout this study, science is the final objective of this project. Once proving the functioning and accuracy of the ADCS system in flight. The task to try out a 3-axis stabilized system will commence and what better way to test it than with the payload initially planned to fly on this mission. Thus continuing even if it is with baby steps to have real-time mapping of the sun, and thus predict solar anomalies that could save lives.

XIII. Lessons Learned

One of the most important lessons that should be noted, and the author hopes that not only aerospace systems engineers, but anybody that could be the reader from any field of engineering, is that things hardly ever go as planned. Research, analysis and simulation can tell you one thing, but when it comes down to working with other individuals with different objectives, things tend to deviate for what you could believe is the objective. The walk through of the study shows that only one of the approaches that can be taken for this problem. Others, from a different perspective, could possibly analyze the problem differently or attack it from where they think would be the best option, and that's the beauty of this, there is more than one way to solve it. The systems engineer has to apply its knowledge within all the fields of specific engineering that are found in a spacecraft, and the strengths and weaknesses of his/her team to better solve the problem that they are faced with. Having a contingency plan will help with technical problems that could be encountered, but when the problems encountered are with partnerships ending, funding getting cut, lack of interest of some of the project stakeholders and or a combinations of all of these, this is when your technical skills and desire to innovate, will allow to transform a project to something that is desirable and needed.

Lessons learned could not be completed, if no noted by the importance of knowing when to take advantage of opportunities for work and of collaboration. This will allow to excel in any field and will make sure that what you are trying to accomplish, in fact gets accomplished.

XIV. Conclusion

There is an old saying a very wise person once told me that says, "*Fly as you test, and test as you fly*." This applies directly to what is has been discussed throughout the length of this report. This Tier system that is being proposed is a way of getting to the objective of getting Heliophysics data, while validating all of the driving critical requirements through low cost missions.

It can be said that the nano-sat platform is relevant to prove an initial Tier, on the path to a full scale mission. Active ADCS will not only prove the capability of doing such, but will prove it can be done with a much lower cost. Adding this to the communication validation at 10-50 Mbps, is a huge leap in what only a couple of years ago seems unthinkable. This is the ideology behind the TechEdSat platform, and thus is why the system trade studies performed, -substantiating the choices as a best viable platform.

The Mission specifications for the Tier 1 demonstrator will allow a foundation to be built for the future tiers:

Subsystem	Parameter	Value	Notes
Orbit	LEO Altitude	350 km	
	Inclination Angle	51.6°	
	Period	91.9'	
	Sunlight Time	55.2'	
ADCS	Controlled Roll Rate	1 RPM	Driving Requirement
Power	Solar Panel Required (High Margin)	44.5 W	
	Solar Panel On-board (TASC)	53.85 W	
	Solar Panel Area	0.0394 m^2	
Structure	Extruded Aluminum	0.439 kg	
Communications	Data Rate Science (Ka-Band)	10-50 Mbps	Driving Requirement
	Data Rate Command (Iridium)	1200 bps	

Table 8: Mission Specification for Tier 1 LEO Demonstrator

The validation of these requirements is direct application for the future tiers towards the full scale science mission. While aware of the strict pointing requirements needed for the full scale mission, the process gone through for an active ADCS to the desired 1 RPM rate, will suffice to allow for orientation to Earth and Sun pointing.

It was found that the nan-satellite platform can't meet all of the Heliophysics top requirements –as this must be accomplished in the larger small-sat form-factor (and is a non-LEO). However, key attributes may be developed and tested at the nano-satellite scale that will greatly contribute to lowering both the technical risk and cost uncertainties of evolving to the next generation capability.

While the proposed future scale mission is using SDO as objective, this stepped process allows to deviate to missions that are not exclusive to Heliophysics. A coronagraph experiment for astrophysical science would require similar requirements to be met, and many others could spin off to its specific field. The heritage obtained while flying these types' missions can virtually eliminate risk. Allowing for more science data to be recorded and help us understand a bit more of how our solar system and/or universe work.

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Appendix

A. Amptek Mini X-Ray

AMPTEK

Miniature X- Ray Source **Mini–X**

Mini–X is a self-contained, miniature X-ray tube system, which includes the X-ray tube, high voltage power supply and USB controller. Designed for X-ray fluorescence analysis applications - XRF.

Features

- 50 kV / 80 µA
- Ag or Au target
- USB controlled
- Stable output
- Fast
- Low power
- Small

Mini-X is the first of its kind; a self-contained, packaged, miniature X-ray tube system, which includes the X-ray tube, the power supply, the control electronics and the USB communication to the computer. It is designed to replace radioisotopes in X-ray fluorescence analysis applications.

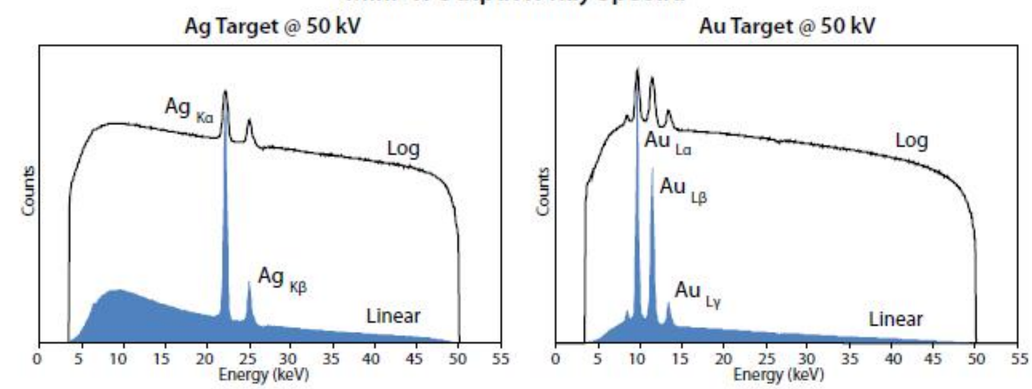
Mini-X has been designed to simplify the XRF process by providing a grounded anode, variable current and voltage controlled via USB and ease of operation. It features a 50 kV/80 µA power supply, a gold (Au) or silver (Ag) transmission target, and a beryllium end window. It is designed for continuous operation in industrial environments.

To further simplify the use of Mini-X an AC adaptor is provided to supply the 12 VDC needed to power the system. The only connections needed to operate the tube are a USB cable and AC adaptor. A flashing red LED and a beeper warns the user when x-rays are present.

Applications

- X-Ray Fluorescence (XRF) analysis
- Portable systems
- OEM
- Process Control
- Research
- Teaching

Mini–X Output X-Ray Spectra



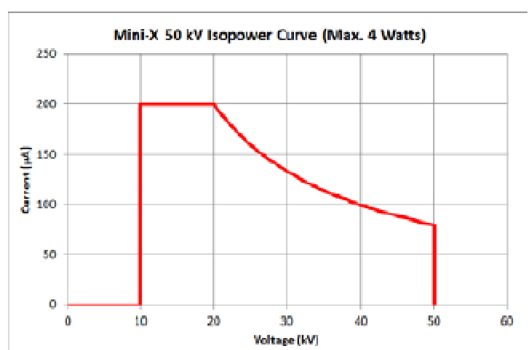
The Mini–X is based on the Newton Scientific Inc. miniature X-ray source.

AMPTEK INC. 14 DeAngelo Drive, Bedford, MA 01730-2204 U.S.A.
Tel: +1 (781) 275-2242 *Fax:* +1 (781) 275-3470 *e-mail:* sales@amptek.com **www.amptek.com**

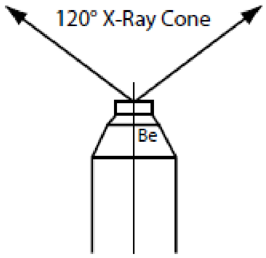
Mini–X Specifications

Target Material	Silver (Ag)	Gold (Au)
Target Thickness	0.75 µm (±0.1 µm)	1 µm (±0.1 µm)
Tube Voltage	10 to 50 kV	10 to 50 kV
Tube Current	5 µA min. / 200 µA max.	5 µA min. / 200 µA max.
Approximate Dose Rate	10 Sv/h @ 30 cm on axis, 50 kV and 80 µA	13 Sv/h @ 30 cm on axis, 50 kV and 80 µA
Approximate Flux	10^6 counts per second/mm^2 on the axis at a distance of 30 cm (50 keV/1 µA)	1.3×10^6 counts per second/mm^2 on the axis at a distance of 30 cm (50 keV/1 µA)
Continuous Power	4 W max. @ 100% duty cycle	4 W max. @ 100% duty cycle
Window Material	Beryllium (Be); window at ground	Beryllium (Be); window at ground
Window Thickness	127 µm	127 µm
Focal Spot Size	Approximately 2 mm	Approximately 2 mm
Output Cone Angle	120°	120°
Cooling	Air cooled	Air cooled
High Voltage Stability	< 0.03% RSD	< 0.03% RSD
Leakage Radiation	<5 µSv/h (0.5 mrem/h) at 5 cm with safety plug installed	<5 µSv/h (0.5 mrem/h) at 5 cm with safety plug installed
Power Consumption	9 W at 50 kV and 80 µA	9 W at 50 kV and 80 µA
Input Voltage	12 VDC (AC adapter included), connector	12 VDC (AC adapter included), connector
Control	USB, mini-USB connector (cable included)	USB, mini-USB connector (cable included
Setting Time	Typical < 1 second	Typical < 1 second
Weight	360 g	360 g
Humidity	30 to 90% (non condensing)	30 to 90% (non condensing)
Operating Temperature	-10 °C to +50 °C	-10 °C to +50 °C
Storage Temperature	-25 °C to +60 °C	-25 °C to +60 °C
Safety Controls and Indicators	1) External hardware interlock 2) Flashing LED 3) Beeper	1) External hardware interlock 2) Flashing LED 3) Beeper
Software	Mini-X Control Software controls voltage and current Mini-X API for custom programming applications	Mini-X Control Software controls voltage and current Mini-X API for custom programming applications
Warranty	One year or 2000 hours, whichever comes first	One year or 2000 hours, whichever comes first

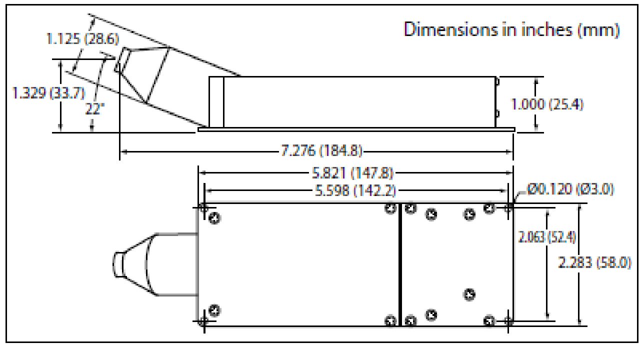
USB Software Interface. Allows the user to set the voltage and current as well as monitor both parameters.

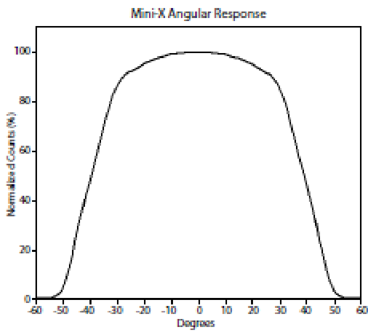


Mini–X Mechanical Dimensions

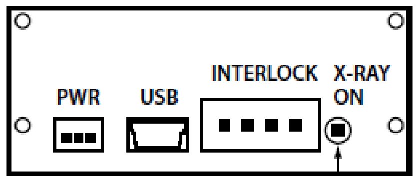


Mini-X 120° cone.
NOTE: When using the 2 mm collimator, the X-ray cone is 5°.





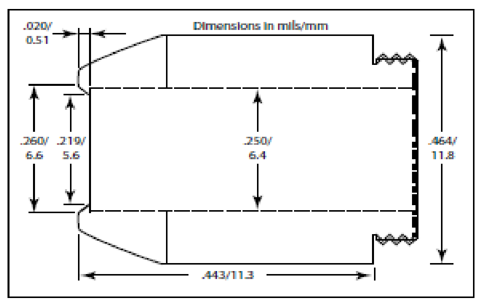
Back panel connectors



Flashing LED/Beeper
X-Rays ON

Collimator and Safety Plug

The Mini-X is provided with a collimator to facilitate its use in XRF applications. It consists of a brass collimator with an aluminum (Al) insert and a cover that screws into the Mini-X. The collimator has a 2 mm diameter hole. The brass safety plug when installed, reduces the flux from an operating tube to less than 2.5 mrem/h at 5 cm away in accordance with Requirements 5.2.2.2.2 of the NBS Handbook for Radiation Safety for X-Ray Diffraction and Fluorescence Analysis Equipment.



Collimator cover mechanical dimensions (mils/mm).

Filters

There are many reasons to use filters on the x-ray tube. They can help eliminate low energy photons to create a clean background and they can filter the characteristic lines of the tube's target. Keep in mind that when any filter is used it reduces the flux coming out of the tube. An Al filter reduces the flux much less than a Mo or Ag filter. The higher the Z of the filter or the thicker the filter, the less flux will be available. It is therefore necessary to raise the current of the x-ray tube to compensate.

Please see http://www.amptek.com/minix.html for output spectra with various filters.

Filters Provided		
Material	**Thickness (µm/mils)**	**# Provided**
Al	1016 / 40	5
Al	254 / 10	5
Cu	25.4 / 1	3
Mo	25.4 / 1	2
Ag	25.4 / 1	1
W	25.4 / 1	1

The Mini–X shown with the Amptek XR-100CR X-Ray Detector and PX5 Digital Pulse Processor.

The Mini–X mounted on MP1 with X-123SDD.

The Mini–X and X-123SDD shown with vacuum couplings.

Mini-X-OEM X-Ray Tube for XRF

The Mini-X-OEM X-ray tube is not the same as the Mini-X.

The Mini-X has a USB interface to control the voltage and current through PC software. The Mini-X is an end-user, packaged device.

The Mini-X-OEM is controlled by user supplied analog voltages

Radiation Precautions

The Mini–X is intended to generate x-ray radiation during normal operation. The Mini–X has been designed to focus radiation in the designated output direction, however radiation in other directions is possible and should be addressed with shielding and/or monitoring in the final application.

Radiation Levels external to the X-ray tube housing with the brass safety plug ON do not exceed 25 µS/h (2.5 mrem/h) measured 5 cm from the surface of the housing in accordance with Requirements 5.2.2.2.2 of the National Bureau of Standards (NBS) Handbook for Radiation Safety for X-Ray Diffraction and Fluorescence Analysis Equipment.

Examples of Shielding (that comply with the above standard)

1 mm (0.040 inch) of Pb will result in radiation levels of 0.5 mrem/h.

6.35 mm (0.250 inch) of Fe will result in radiation levels of 0.5 mrem/h.

3.18 mm (0.125 inch) of Brass will result in radiation levels of 2.5 mrem/h.

The inside of the housing can also be lined with 3.18 mm (0.125 inch) of aluminum (Al) in order to absorb the XRF from the shielding material.

Caution

The Mini-X is only one component of an X-ray instrument. It is the responsibility of the user, the OEM customer, or experimenter to provide a fail safe metal enclosure to prevent escaping radiation while using this product. The final product (turn-key system) must comply with local government regulations to protect personnel from exposure to radiation. Amptek Inc., bears no responsibility for the incorrect use of this product.

Caution

This device produces X-Rays when energized. To be operated only by qualified personnel.

AMPTEK INC. 14 DeAngelo Drive, Bedford, MA 01730-2204 U.S.A.
Tel: +1 (781) 275-2242 *Fax:* +1 (781) 275-3470 *e-mail:* sales@amptek.com **www.amptek.com**

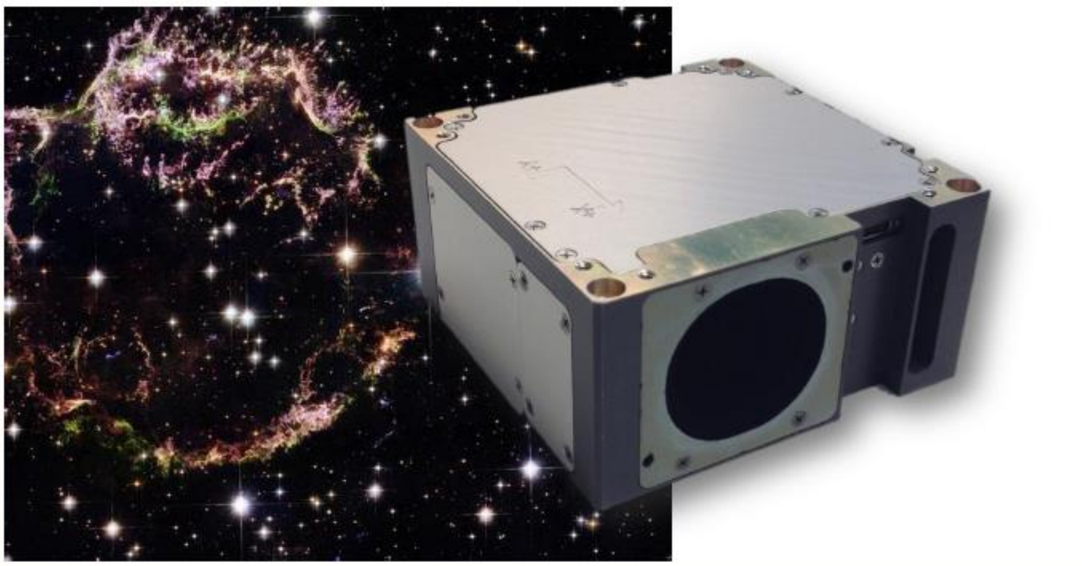
B. BCT XACT

BCT

XACT™

High Performance Attitude Control for CubeSats

Precise 3-axis stellar attitude determination in a micro-package

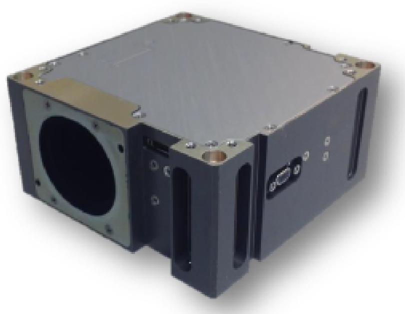


Key Features

- 3-axis stellar attitude determination with integrated stray light baffle
- 0.5U Micro-package
- Multiple pointing reference frames: Inertial, LVLH, Earth-Fixed, Solar
- Low jitter 3-axis reaction wheel control (also sold as single wheels)
- User friendly software for simulation, integration, and customization
- Self-calibrating reaction wheels, with advanced digital controls, provide unparalleled torque precision.

Blue Canyon Technologies
2425 55th St, Suite 150, BLDG A Boulder, CO 80301
720.458.0703 www.bluecanyontech.com

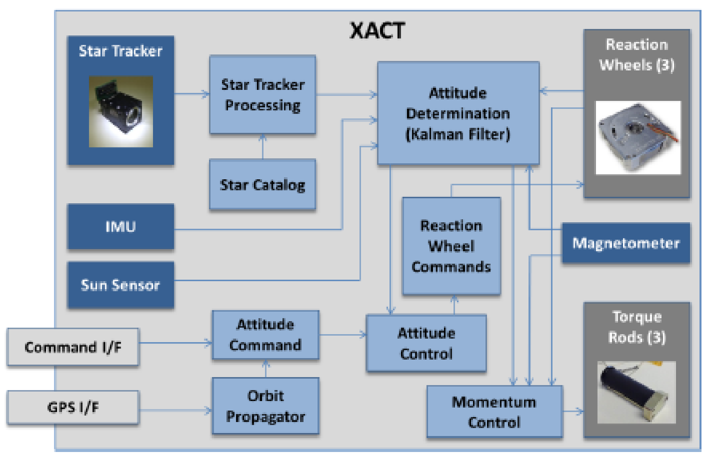
XACT is a reliable CubeSat attitude control system compatible with a variety of configurations and missions. The highly integrated XACT architecture leverages a powerful processing core with BCT's micro-Star Tracker and micro- Reaction Wheel assemblies to enable a new generation of highly capable, miniaturized spacecraft. XACT features 3-axis Stellar Attitude Determination in a micro-package. Built-in flexible commanding allows for multiple pointing reference frames: Inertial, LVLH, Earth-Fixed, Solar. Precise 3-axis control is provided by low jitter reaction wheels, torque rods and integrated control algorithms. Software is available to support simulation, system integration, and customization of the ADCS functionality.



XACT Capability

Specification	Performance
Spacecraft Pointing Accuracy	± 0.003 deg (1-sigma) for 2 axes ± 0.007 deg (1-sigma) for 3rd axis
Spacecraft Lifetime	3 Years (LEO)
XACT Mass	0.85 kg
XACT Volume	10 x 10 x 5 cm (0.5U)
XACT Electronics Voltage	5V
XACT Reaction Wheel Voltage	12V
Data Interface	RS-422 (can support I2C and SPI)
Slew Rate (8kg, 3U CubeSat)	≥10 deg/sec

Operational Case	Power (W)
XACT (low power standby mode)	0.85
XACT (5 Hz operation)	1.05
XACT + 3 TR (ON STATE)	1.80
XACT + 3 RW (@ 600 rpm)	1.19
XACT + 3 RW (@ 600 rpm) + 3 TR (ON STATE)	1.94
XACT + 3 RW (@ 1500 rpm)	2.20
XACT + 2RW (@600 rpm) + 1 RW (max speed @6000 rpm)	2.83



Blue Canyon Technologies
2425 55th St, Suite 150, BLDG A
Boulder, CO 80301
720.458.0703
www.bluecanyontech.com

For details please visit bluecanyontech.com

C. MAI-101

Maryland Aerospace, Inc.
CubeSat ADACS Experts

2145 Priest Bridge Court Suite 15
Crofton, MD 21114-2463
Phone: (410) 451-2505
Fax: (410) 451-2507
www.maiaero.com

MAI-101™
Miniature 3-Axis Reaction Wheel

Budgetary Quotation
MAI-QD-2015-04-01-MAI-101

Prepared for
Eddie Uribe
San Jose State University

April 6, 2015

Page 1 of 6

MAI-QD-2015-04-01-MAI-101
This quotation is valid for 30 days

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MAI-101

Miniature 3-Axis Reaction Wheel

The MAI-101 is a 3-axis Miniature Reaction Wheel (MRW) in a hermetically sealed 3" x 3" x 2.7" enclosure. It is suitable for up to about 20lbs nanosatellite applications requiring precise 3-axis pointing such as is required for Earth imaging or celestial pointing. In the same family of products are the MAI-100 complete ADACS system and the MAI-200/201 for larger spacecraft.

Torque and momentum storage have been sized for a micro or nano-spacecraft in a high agility application. For example, the MAI-101 can slew an Earth imaging CubeSat size spacecraft to rapidly retarget between successive aim points, each located ± 30° off track.

Specifications

Performance Item	Unit	Specification
ADACS		No
Dimensions	in	3 x 3 x 3.75
Mass	g	640
Momentum Storage@1000rpm	mNms	1.1
Max Torque	mNm	0.635
Operating Voltage	V	12
Power Consumption		
Idle	W	0.48 (0.04A @ 12V)
Steady State/Typical	W	3.84 (0.32A @ 12V)
Peak	W	4.56 (0.38A @ 12V)
Command/TLM interface		RS232
Operating Temperature	deg C	-40 to 80
Launch Environment Vibration Spec	g rms	12
Radiation	krad	30

MAI-QD-2015-04-01-MAI-101
This quotation is valid for 30 days

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MAI-101

ITEM	DESCRIPTION	PRICE
	Miniature 3-Axis Reaction Wheel	
1	**MAI-101 Flight Model:** 3-Axis Reaction Wheels, Reaction Wheel Drivers. Academic Discount ($14,900.00-$2,235.00=$12,665.00)	**$12,665.00**
2	**Freight**	**$75.00**
	Total	**$12,740.00**

NOTES

1. **This offer is effective for 30 days from date of quote. All prices in $US.**
2. The MAI-101is ITAR controlled, an export license is required. Price does not include charges and fees for procuring the license from the U.S. State Department.
3. Maryland Aerospace, Inc. welcomes the opportunity to quote integration & test of the MAI-101in the customer's spacecraft.
4. Maryland Aerospace, Inc. normally ships the MAI-101 within 90 days of receipt of a written purchase order and deposit.
5. Shipping Terms: Ex-Works, Crofton, Maryland.

ACADEMIC DISCOUNTS

A 15% discount for accredited academic institutions is shown in line (1) one above.

VOLUME DISCOUNTS

Apply to total number of units purchased with a single calendar year.

MAI-101

1	5	10	20	30	40	50
$14,900	$14,000	$13,300	$12,700	$13,300	$12,000	$11,800

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FIELD SERVICE/FIELD TRAINING RATES

Maryland Aerospace, Inc. offers its field service to maintain and troubleshoot systems, train customer personnel in the field, and to supervise initial operations at the following per diem rates:

- $720.00 per eight (8) hour day, plus travel and living expenses for a field service technician.
- $1,080.00 per eight (8) hour day, plus travel and living expenses for an engineer.

Additional time worked in excess of eight (8) hours per day shall be paid at the rate of $90.00 per hour for a field service technician and $135.00 per hour for engineers.

CONDITIONS OF SALE

EQUIPMENT, WARRANTIES, PRICE, TERMS AND DELIVERY ARE QUOTED BASED UPON THE FOLLOWING CONDITIONS ONLY.

SHIPMENT

Maryland Aerospace, Inc. normally ships its products within 90 days of receipt of a written Purchase Order and deposit. Delivery time must be verified at the time of order placement as it is subject to change depending upon MAI's workload. Delivery is also contingent of export licensing.

PAYMENT TERMS

All prices quoted in this proposal are in U.S. dollars. The schedule of payment is to be as follows:

- NET 30 The invoice will be mailed at time of shipment.

The prices quoted herein are exclusive of VAT or any other taxes that may be applicable.

Payment for service personnel at per diem rates will be invoiced every thirty (30) days and is due thirty (30) days thereafter. MAI may require a pre-payment of estimated travel and living expenses for extended service and training engagements.

Payments from buyers in countries other than the U.S. must be by wire transfer or single irrevocable letter of credit confirmed by a U.S. bank selected by MAI, available by payment at the counters of confirming U.S. bank with drafts drawn on the confirming U.S. bank and payable at sight of drafts and documents.

All banking fees, U.S. and foreign, including communication charges for amendments to the letter of credit shall be charged to the buyer's account if the letter of credit does not comply with the above instructions, or if the confirming U.S. bank rejects the letter of credit as not being a workable document. Otherwise, only banking fees or commissions outside of the U.S. will be to the buyer's account. Banking fees or commissions inside the U.S. will be to MAI's account.

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WARRANTY

Seller warrants only that the goods shall generally meet the specifications described in this Agreement. This warranty is exclusive, and is in lieu of all other warranties, whether written, oral or implied, including the warranty of merchantability and the warranty of fitness for a particular purpose. The Buyer hereby disclaims all other warranties. In effectiveness or other unintended consequences may result because of such factors as manner of use or other influencing factors in the use of the Product, which are beyond the control of the Seller. Buyer shall assume all such risks, and Buyer agrees to hold the Seller harmless for any claims relating to such factors. Buyer releases, indemnifies and agrees to defend and hold the Seller harmless from any and all claims of whatever kind arising from any use or application of the Product, but Buyer's obligation to indemnify, defend and hold Seller harmless shall not include a claim for breach of the limited express warranty set forth herein. Buyer's obligation to indemnify, defend, and hold Seller harmless shall include, but is not be limited to, reasonable attorney's fees and costs incurred by the Seller. Uses of the Product contrary to the Directions for Use, or under abnormal conditions or under conditions not reasonably foreseeable to the Seller are beyond the control of the Seller and the Buyer assumes the risk of any such use. Buyer acknowledges that it has been informed of the risks of use, and hereby waive any right to assert any claim against the Seller, whether for breach of contract, tort, negligence, or otherwise.

IT IS UNDERSTOOD AND AGREED THE SELLER'S LIABILITY WHETHER IN CONTRACT, IN TORT, UNDER ANY WARRANTY, IN NEGLIGENCE OR OTHERWISE SHALL NOT EXCEED THE RETURN OF THE AMOUNT OF THE PURCHASE PRICE PAID BY THE PURCHASER AND UNDER NO CIRCUMSTANCES SHALL SELLER BE LIABLE FOR SPECIAL, INDIRECT, INCIDENTAL OR CONSEQUENTIAL DAMAGES. THE PRICE STATED FOR THE EQUIPMENT IS A CONSIDERATION IN LIMITING SELLER'S LIABILITY. NO ACTION, REGARDLESS OF FORM, ARISING OUT OF THE TRANSACTIONS OF THIS AGREEMENT MAY BE BROUGHT BY PURCHASER MORE THAN ONE YEAR AFTER THE CAUSE OF ACTION HAS ACCRUED.

SALES AGREEMENT

A Sales Agreement must be signed at time of order placement confirming delivery date as well as the client's acceptance of the above Terms and Conditions.

DRAWINGS

No detail or shop working drawings of the equipment will be furnished aside from mechanical and electrical interfaces. All drawings and specifications furnished by MAI are to remain MAI's property and no copies are to be made for third parties without the prior written consent of MAI.

CHANGES

In case MAI agrees to any changes, at the request of the Purchaser, in any of the specifications, resulting in delays in design or manufacture or in increased cost, the time of shipment shall be extended to compensate for the delay and the Purchaser shall pay MAI for any additional expense occasioned by such changes.

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LAW

This Agreement shall be governed by the internal laws of the State of Maryland, USA, and any claims arising hereunder shall be prosecuted in the United State District Court having jurisdiction of the causes of action arising in the District in which MAI is located.

ENTIRETY

These Sales Terms and Conditions are an essential element of the quotation, which together shall constitute the entire understanding between Seller and Purchaser with respect to this sale. There are no other, written or oral, warranties or representations to Purchaser, or agreements between the parties, except those provided herein. In case of conflict between the Purchaser's purchase order and these Conditions of Sale and this quotation, these Conditions of Sale shall prevail.

D. EPSON M-V340-xx (IMU)

M-V340-xx

EPSON
EXCEED YOUR VISION

IMU (Inertial Measurement Unit)

GENERAL DESCRIPTION

The M-V340-xx is a small form factor inertial measurement unit (IMU) with 6 degrees of freedom: triaxial angular rates and linear accelerations, and provides high-stability and high-precision measurement capabilities with the use of high-precision compensation technology. A variety of calibration parameters are stored in a memory of the IMU, and are automatically reflected in the measurement data being sent to the application after the power of the IMU is turned on. With a general-purpose SPI/UART supported for host communication, the M-V340-xx reduces technical barriers for users to introduce inertial measurement and minimizes design resources to implement inertial movement analysis and control applications.

The features of the IMU such as high stability, high precision, and small size make it easy to create and differentiate applications in various fields of industrial systems.

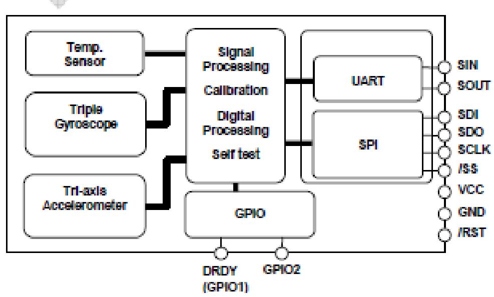
FEATURES

- Small Size, Lightweight : 10x12x4mm, 1 grams
- Low-Noise, High-stability
  - Gyro Bias Instability : 7 deg/hr
  - Angular Random Walk : 0.2 deg/$\sqrt{hr}$
- Initial Bias Error : ±0.5 deg/s (1σ)
- 6 Degrees Of Freedom
  - Triple Gyroscopes : ±450 deg/s,
  - Tri-Axis Accelerometer : ±6 G
- 16bit data resolution
- Digital Serial Interface : SPI / UART
- Calibrated Stability (Bias, Scale Factor, Axial alignment)
- Data output rate : to 1k Sps
- Calibration temperature range : −20°C to +70°C
- Operating temperature range : −40°C to +85°C
- Single Voltage Supply : 3.3 V
- Low Power Consumption : 18mA (Typ.)

APPLICATIONS

- Unmanned systems
- Motion analysis and control
- Navigation systems
- Vibration control and stabilization
- Pointing and tracking systems

FUNCTIONAL BLOCK DIAGRAM



1

■ SENSOR SECTION SPECIFICATION

T_A=25°C, VCC=3.3V, angular rate=0 deg/s, ≤±1G, unless otherwise noted.

Parameter	Test Conditions / Comments	Min.	Typ.	Max.	Unit
GYRO SENSOR					
Sensitivity					
Dynamic Range	—	±450	—	—	deg/s
Sensitivity	—		0.015		(deg/s)/LSB
Temperature Coefficient	1 σ, −20°C ≤ T_A ≤ +70°C	—	10	—	ppm/°C
Nonlinearity	Best fit straight line <±350dps	—	0.1	—	% of FS
	>±350dps	—	0.5	—	% of FS
Misalignment	1 σ, Axis-to-axis, Δ = 90° ideal	—	±0.1	—	deg
Bias					
Initial Error	1 σ	—	±0.5	—	deg/s
Temperature Coefficient (Linear approximation)	1 σ, −20°C ≤ T_A ≤ +70°C		0.001		(deg/s)/°C
In-Run Bias Stability	1 σ	—	7	—	deg/hr
Angular Random Walk	1 σ	—	0.2	—	deg/ $\sqrt{hr}$
Linear Acceleration Effect			<0.01		(deg/s)/G
Noise					
Noise Density	1 σ, f = 10 to 20 Hz, no filtering	—	0.004	—	(deg/s)/ $\sqrt{Hz}$, rms
Frequency Property					
3 dB Bandwidth	—	—	180	—	Hz
ACCELEROMETERS					
Sensitivity					
Dynamic Range	—	±6	—	—	G
Sensitivity	—		0.18		mG/LSB
Temperature Coefficient	1σ, −20°C ≤ T_A ≤ +70°C	—	100	—	ppm/°C
Nonlinearity	≤ 1G , Best fit straight line	—	TBD	—	% of FS
Misalignment	1 σ, Axis-to-axis, Δ = 90° ideal	—	TBD	—	deg
Bias					
Initial Error	1 σ	—	±8	—	mG
Temperature Coefficient (Linear approximation)	1 σ, −20°C ≤ T_A ≤ +70°C		0.02		mG/°C
In-Run Bias Stability	1 σ	—	<0.1	—	mG
Velocity Random Walk	1 σ	—	<0.1	—	(m/sec)/ $\sqrt{hr}$
Noise					
Noise Density	1 σ, f = 10 to 20 Hz, no filtering	—	TBD	—	mG/ $\sqrt{Hz}$, rms
Frequency Property					
3 dB Bandwidth	—	—	180	—	Hz
TEMPERATURE SENSOR					
Scale Factor *1	Output = 1469 @ +25°C	—	-0.0053964	—	°C/LSB

*1) This is a reference value used for internal temperature compensation. We provide no guarantee that the value gives an absolute value of the internal temperature.

Note) The values in the specifications are based on the data calibrated at the factory. The values may change according to the way the product is used.

Note) The Typ values in the specifications are average values or 1σ values.

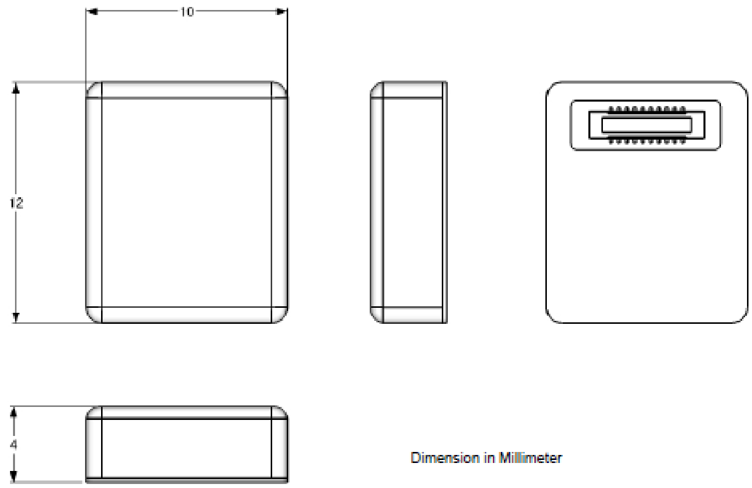
Note) Unless otherwise noted, the Max / Min values in the specifications are design values or Max / Min values at the factory tests.

2

■ RECOMMENDED OPERATING CONDITION

Parameter	Condition	Min	Typ	Max	Unit
VCC to GND		3.15	3.3	3.45	V
Digital Input Voltage to GND		GND		VCC	V
Digital Output Voltage to GND		−0.3		VCC +0.3	V
Calibration temperature range	Performance parameters are applicable	−20		70	°C
Operating Temperature Range		−40		85	°C

■ OUTLINE DIMENSIONS



Dimension in Millimeter

SEIKO EPSON CORPORATION

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Phone: +81-42-587-8291 FAX: +81-42-587-5117

Document code: 412572600
First issue August, 2013 in Japan
Rev.20130717

E. ATLAS MODEL DOWNLINK MODE

			Inert Mass (kg)				Expendable Mass (kg)				Total Mass (kg)		
Subsystem	Exists	Sane	Override	Mass	Uncertai	Mass + U	Override	Expendec	Residual	Reserved	Mass	Uncertai	Mass + Un
Payload	TRUE	TRUE		0.2	0%	0.2					0.2	0%	0.2
ADACS	TRUE	TRUE		1.2	11%	1.3					1.2	11%	1.3
CnDH	TRUE	TRUE		0.1	10%	0.1					0.1	10%	0.1
Power	TRUE	TRUE		0.3	0%	0.3					0.3	0%	0.3
Propulsion	FALSE												
Structures	TRUE	TRUE		2.0	21%	2.4					2.0	21%	2.4
Telecom	TRUE	TRUE		0.1	4%	0.1					0.1	4%	0.1
Thermal	0	TRUE											
Bus Subtotal (kg)				3.6	15%	4.2		0.0	#DIV/0!	0.0	3.6	15%	4.2
Total (kg)				3.8	15%	4.4		0.0	#DIV/0!	0.0	3.8	15%	4.4

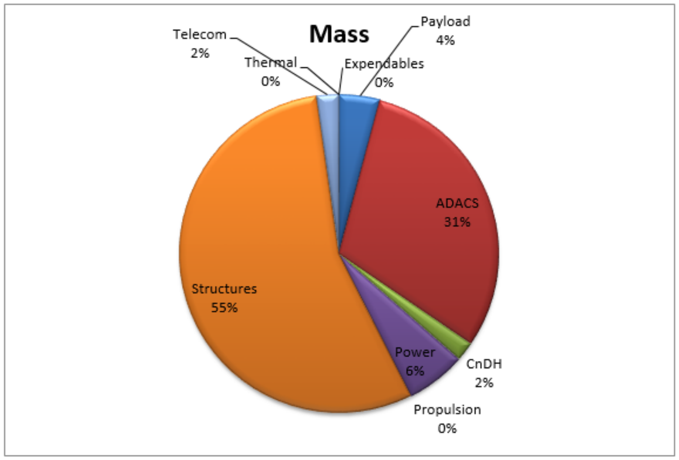
Power: (W)			Downlink
Override	Nominal	Growth	Mature
	1.5	5%	1.6
	0.1	30%	0.1
	1.9	30%	2.4
	0.0	0%	0.0
	0.0	0%	0.0
	4.7	0%	4.7
Subtotal	6.6	9%	7.1
Total (W)	8.1	8%	8.7

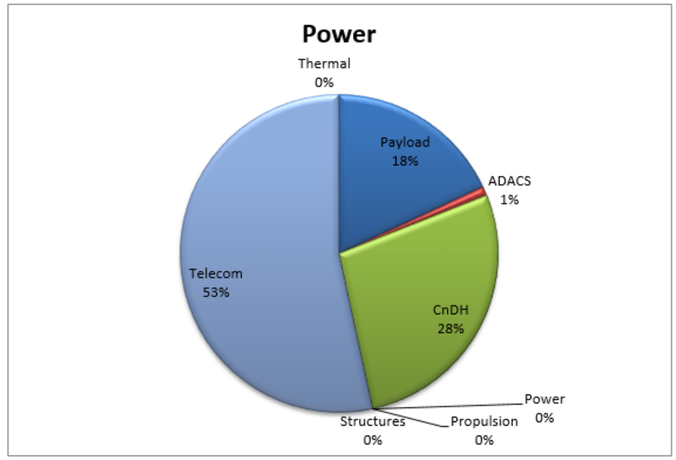
Power System	
Energy Storage (Wh)	20
EOL Generation (W)	13
Power Generation Margin	13%
Energy Storage Margin	187%
Maximum Battery DoD	21%

	Available Resource	Current Estimate	Margin
Mass (kg)	9300	4.4	212623%
Prop (kg)	0.0	0.0	

Minimum Link Margin (dB)	38.56
Maximum Data Rate Utilization	11%

Original Creator:
atlas_v395.xlsm





F. ATLAS MODEL SCIENCE MODE

Subsystem	Exists	Sane	Inert Mass (kg): Override	Mass	Uncertai	Mass + U	Expendable Mass (kg): Override	Expende	Residual	Reserved	Total Mass (kg): Mass	Uncertai	Mass + Un
Payload	TRUE	TRUE		0.2	0%	0.2					0.2	0%	0.2
ADACS	TRUE	TRUE		1.2	11%	1.3					1.2	11%	1.3
CnDH	TRUE	TRUE		0.1	10%	0.1					0.1	10%	0.1
Power	TRUE	TRUE		0.3	0%	0.3					0.3	0%	0.3
Propulsion	FALSE												
Structures	TRUE	TRUE		2.0	21%	2.4					2.0	21%	2.4
Telecom	TRUE	TRUE		0.1	4%	0.1					0.1	4%	0.1
Thermal	0	TRUE											
Bus Subtotal (kg)				3.6	15%	4.2		0.0	#DIV/0!	0.0	3.6	15%	4.2
Total (kg)				3.8	15%	4.4		0.0	#DIV/0!	0.0	3.8	15%	4.4

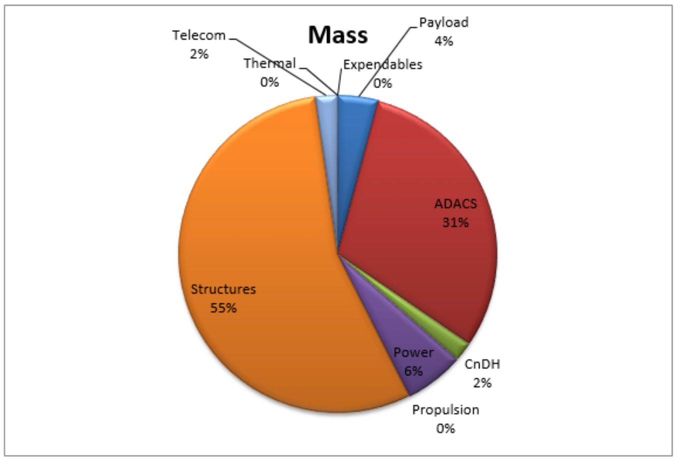
Power: (W) Override	Nominal	Growth	Mature (Science)
	2.5	5%	2.6
	0.7	30%	0.9
	1.9	30%	2.4
	0.0	0%	0.0
	0.0	0%	0.0
	1.2	0%	1.2
Subtotal	3.7	20%	4.5
Total (W)	6.2	14%	7.1

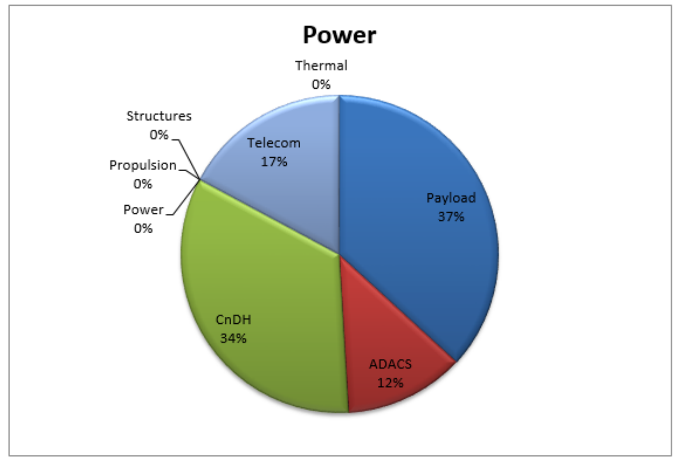
Power System	
Energy Storage (Wh)	20
EOL Generation (W)	13
Power Generation Margin	13%
Energy Storage Margin	187%
Maximum Battery DoD	21%

	Available Resource	Current Estimate	Margin
Mass (kg)	9300	4.4	212623%
Prop (kg)	0.0	0.0	

Minimum Link Margin (dB)	38.56
Maximum Data Rate Utilization	11%

Original Creator:
atlas_v395.xlsm





G. ATLAS MODEL DE-TUMBLE MODE

			Inert Mass (kg)				Expendable Mass (kg)				Total Mass (kg)		
Subsystem	Exists	Sane	Override	Mass	Uncertai	Mass + U	Override	Expende	Residual	Reserved	Mass	Uncertai	Mass + Un
Payload	TRUE	TRUE		0.2	0%	0.2					0.2	0%	0.2
ADACS	TRUE	TRUE		1.2	11%	1.3					1.2	11%	1.3
CnDH	TRUE	TRUE		0.1	10%	0.1					0.1	10%	0.1
Power	TRUE	TRUE		0.3	0%	0.3					0.3	0%	0.3
Propulsion	FALSE												
Structures	TRUE	TRUE		2.0	21%	2.4					2.0	21%	2.4
Telecom	TRUE	TRUE		0.1	4%	0.1					0.1	4%	0.1
Thermal	0	TRUE											
Bus Subtotal (kg)				3.6	15%	4.2		0.0	#DIV/0!	0.0	3.6	15%	4.2
Total (kg)				3.8	15%	4.4		0.0	#DIV/0!	0.0	3.8	15%	4.4

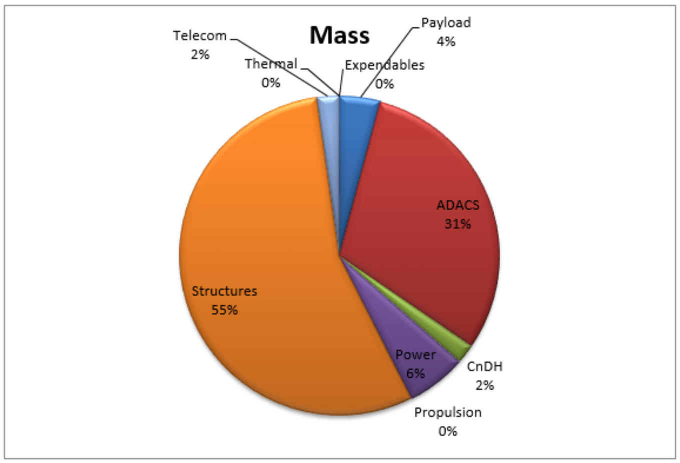
Power: (W)			De-tumble
Override	Nominal	Growth	Mature
	0.0	0%	0.0
	5.6	30%	7.3
	1.9	30%	2.4
	0.0	0%	0.0
	0.0	0%	0.0
	1.2	0%	1.2
Subtotal	8.7	26%	10.9
Total (W)	8.7	26%	10.9

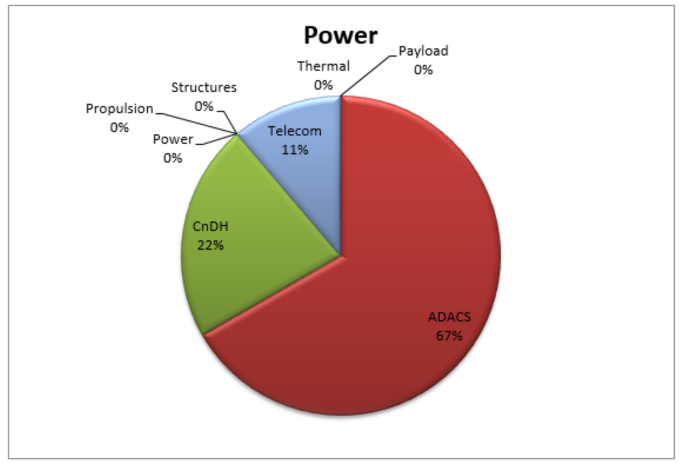
Power System	
Energy Storage (Wh)	20
EOL Generation (W)	13
Power Generation Margin	13%
Energy Storage Margin	187%
Maximum Battery DoD	21%

	Available Resource	Current Estimate	Margin
Mass (kg)	9300	4.4	212623%
Prop (kg)	0.0	0.0	

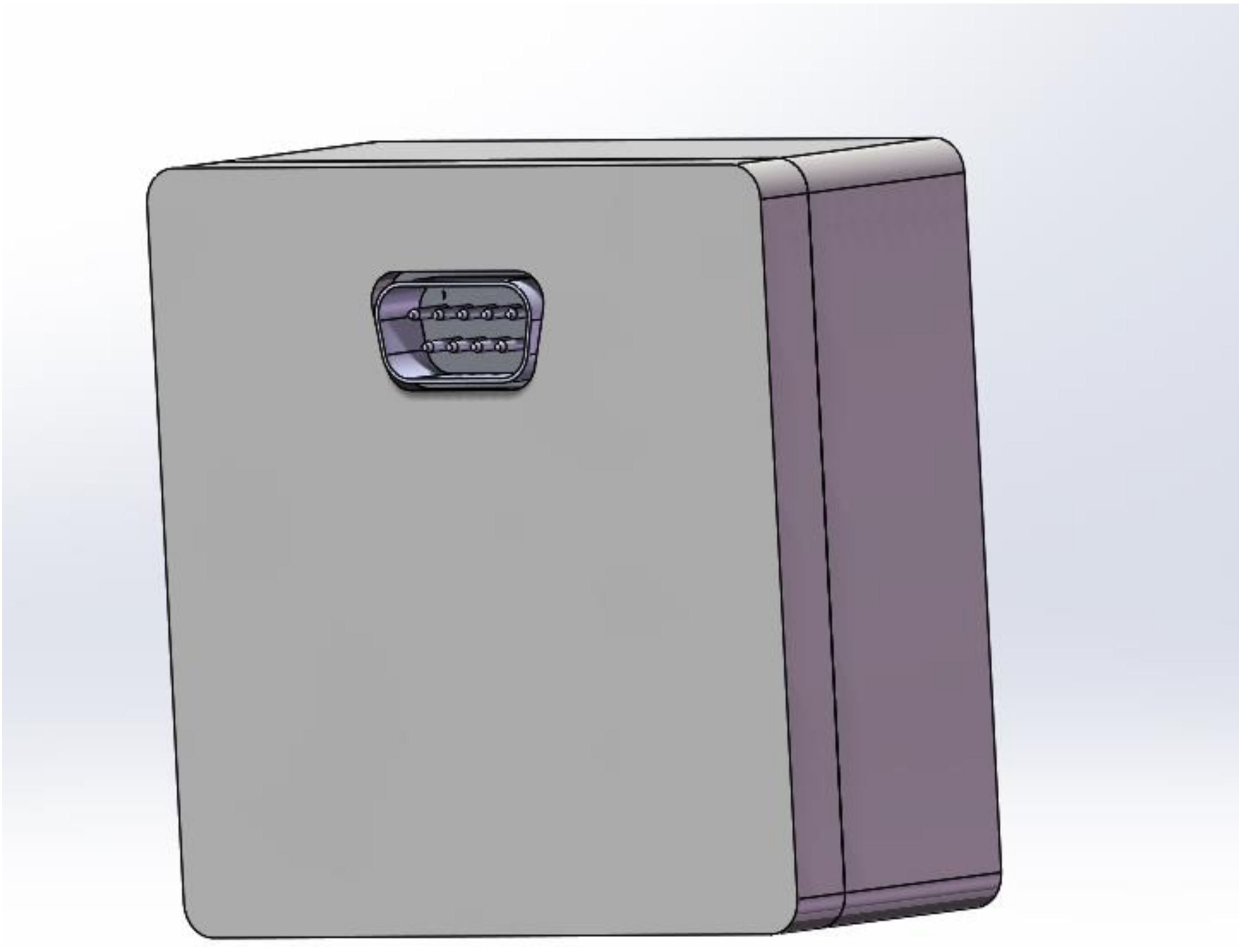
Minimum Link Margin (dB)	38.56
Maximum Data Rate Utilization	11%

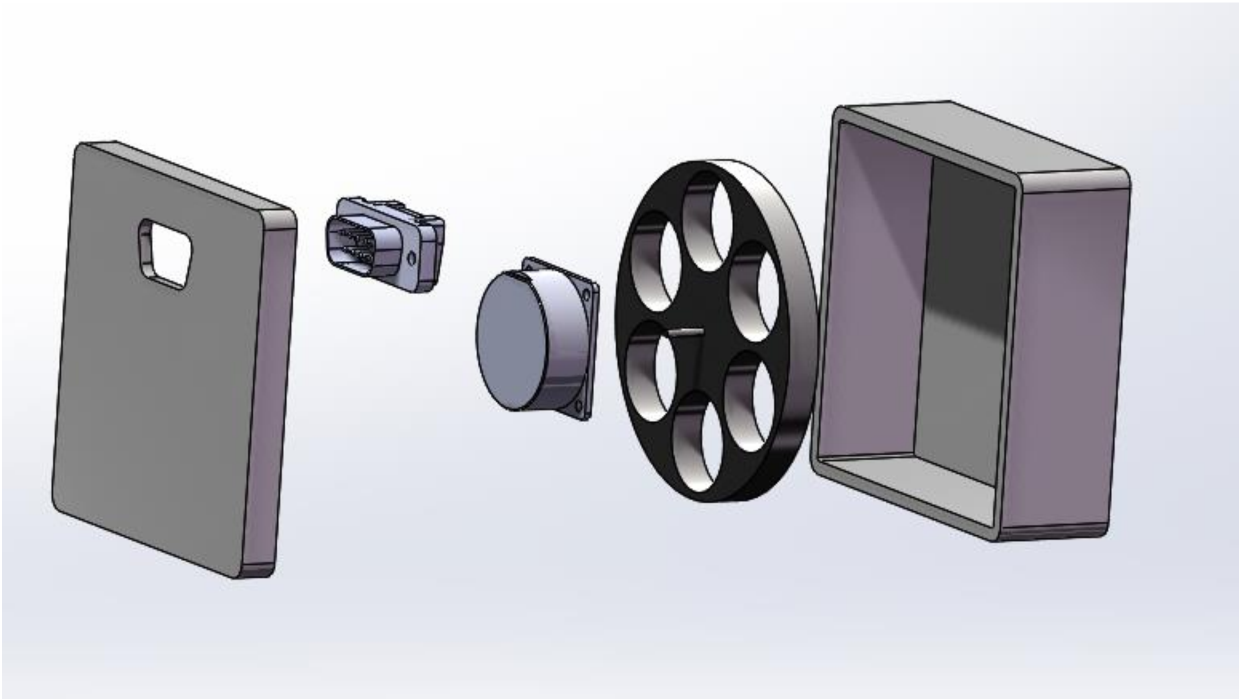
Original Creator:
atlas_v395.xlsm



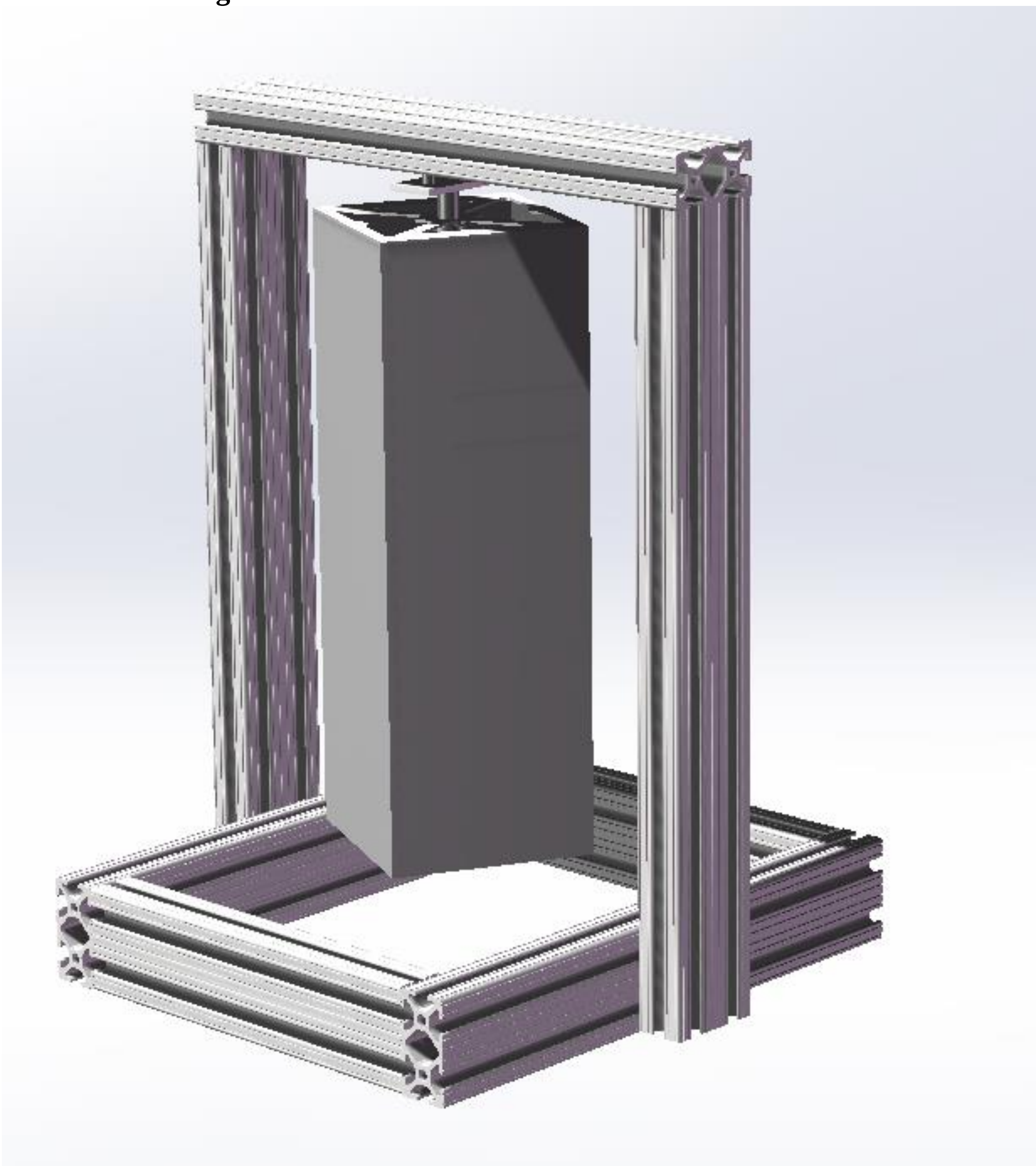


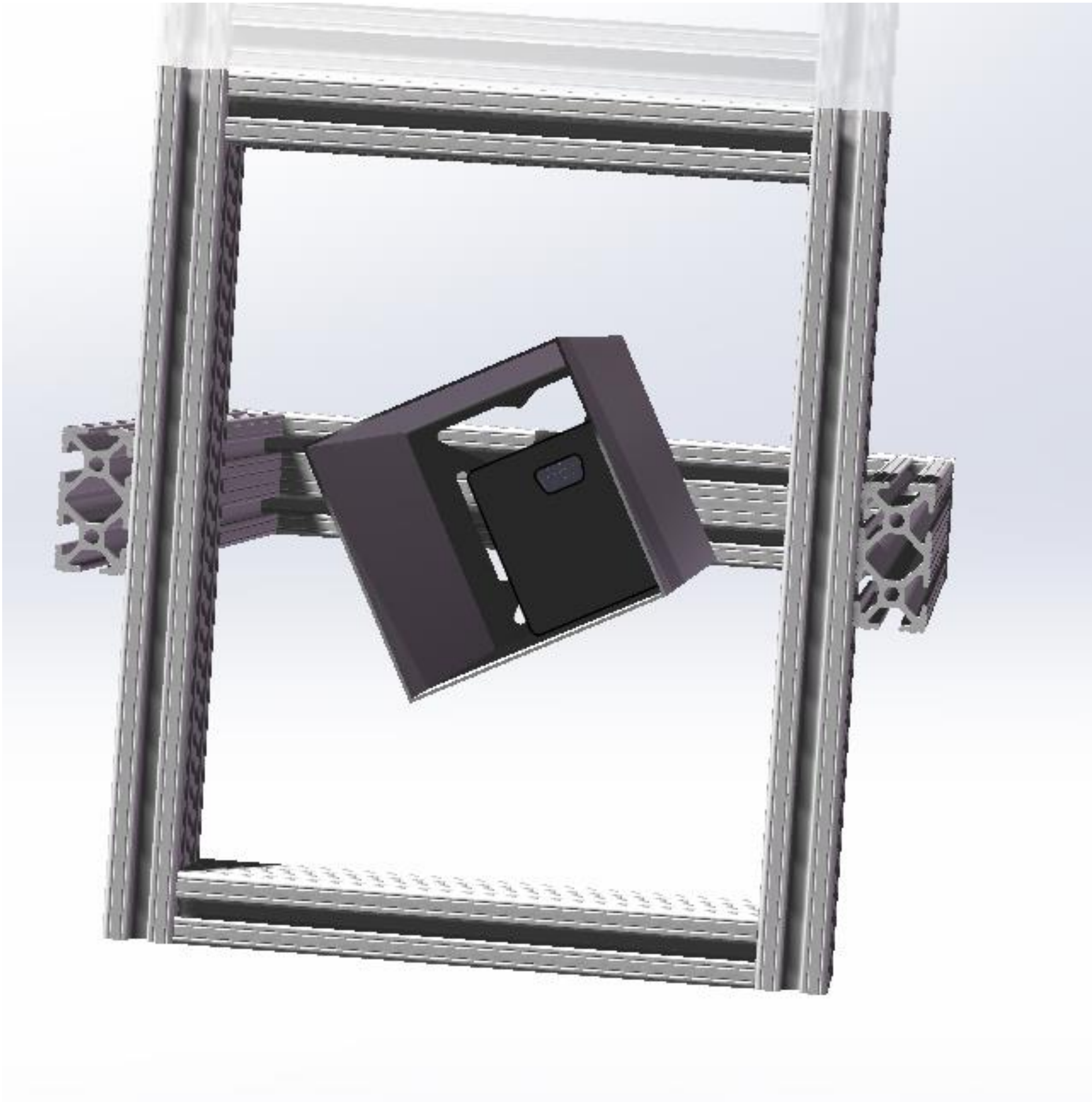
H. Reaction Wheel Design

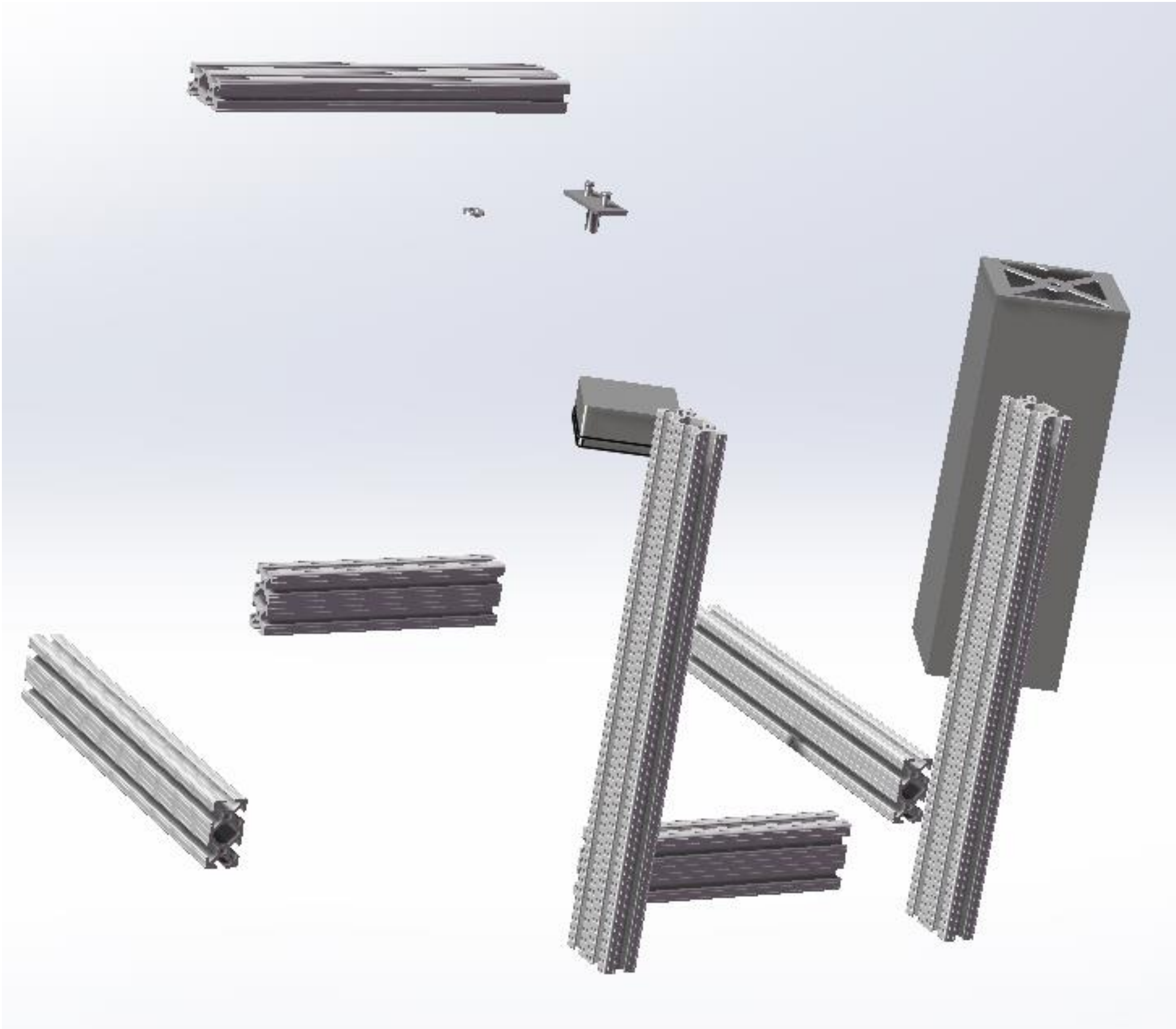




I. Test Fixture Design







J. LEO Segment Summary

Astrogator MCS Segment Summary Astrogator MCS Initial State
Summary
MCS Segment Type: InitialState
Name: Initial State
User Comment: Initializes state data

Satellite State at End of Segment:

UTC Gregorian Date: 15 Jan 2016 08:00:00.000 UTC Julian Date: 2457402.83333333
Julian Ephemeris Date: 2457402.83411093
Time past epoch: 4.8816e+006 sec (Epoch in UTC Gregorian Date: 19 Nov 2015 20:00:00.000)

State Vector in Coordinate System: Earth Inertial

Parameter Set Type: Cartesian
X: 6728.1369999999988000 km Vx: 0.0000000000000000 km/sec
Y: 0.0000000000000000 km Vy: 4.7809743355394767 km/sec
Z: 0.0000000000000000 km Vz: 6.0320883779480789 km/sec

Parameter Set Type: Keplerian
sma: 6728.1369999999997000 km RAAN: 0 deg
ecc: 0.0000000000000000 w: 0 deg
inc: 51.60000000000001 deg TA: 0 deg

Parameter Set Type: Spherical
Right Asc: 0 deg Horiz. FPA: 0 deg
Decl: 0 deg Azimuth: 38.39999999999999 deg
|R|: 6728.1369999999988000 km |V|: 7.6969997918970634 km/sec

Other Elliptic Orbit Parameters :
Ecc. Anom: 0 deg Mean Anom: 0 deg
Long Peri: 0 deg Arg. Lat: 0 deg
True Long: 0 deg Vert FPA: 90 deg
Ang. Mom: 51786.46908885492 km^2/sec p: 6728.1369999999997000 km
C3: -59.24380579646342 km^2/sec^2 Energy: -29.62190289823171 km^2/sec^2
Vel. RA: 90 deg Vel. Decl: 51.60000000000001 deg
Rad. Peri: 6728.1369999999997000 km Vel. Peri: 7.6969997918970634 km/sec
Rad. Apo: 6728.1369999999997000 km Vel. Apo: 7.696999791897063 km/sec
Mean Mot.: 0.0655464659814113 deg/sec
Period: 5492.286954144783 sec Period: 91.53811590241304 min

Period: 1.525635265040217 hr Period: 0.06356813604334238 day
Time Past Periapsis: 0 sec
Time Past Ascending Node: -5492.286954144783 sec
Beta Angle (Orbit plane to Sun): 25.4628538687258 deg
Mean Sidereal Greenwich Hour Angle: 234.218625553279 deg

Geodetic Parameters:
Latitude: 0.08977774842126207 deg
Longitude: 125.986871152752 deg
Altitude: 350.0000520836169900 km
Geocentric Parameters:
Latitude: 0.08920800754402504 deg
Longitude: 125.986871152752 deg

Spacecraft Configuration:
Drag Area: 0.5 m^2
SRP Area: 20 m^2
Dry Mass: 10 kg
Fuel Mass: 0 kg
Total Mass: 10 kg
Area/Mass Ratio: 5e-008 km^2/kg
Tank Pressure: 5000 Pa
Fuel Density: 1000 kg/m^3
Cr: 1.000000
Cd: 2.200000
Rad Press Area: 20 m^2
Rad Press Coeff: 1.000000

User-selected results:
Orbit Period = 5492.2869541447826000 sec