System Integration Architecture for a Sun Tracking and Heliophysiscs Smallsat

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By

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System Integration Architecture for a Sun Tracking and Heliophysiscs Smallsat

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This paper describes the system integration for a Sun tracking small satellite. Based on the TechEdSat architecture, with the modifications accommodate a fit X-Ray spectrometer and the addition of a 3-axis stabilization system. Mission lifetime is meant to be spent on Low Earth Orbit (LEO). Utilizing the TechEdSat as a spacecraft bus with the addition of an in house build ADCS system for controlled pointing capabilities. Initial parametric integration models have been run to assure functionality in operational modes, with a large margin to buffer. Initial testing has commenced with a 1 degree of freedom model to proceed with a 3 degree of freedom system and full spacecraft system integration.

Nomenclature

= Semi-major axis

b = base

E = modulus of elasticity

h = height

h = altitude of the spacecraft

$\mathbf{h}$ = angular momentum stored by rotating object

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H = angular momentum vector

I = Moment of Inertia

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 does not 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 and Blue Canyon Technologies have developed systems that promise accuracy of well below 1 degreee[2,3]. 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 cubesats 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 ADCS system and possibly 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. 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.[0]

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.[0]

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 Proccess (Table 1-1)[0]

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.[0]

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.[0]

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 manufactering 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 futher understand the needs of a Spacecraft or Space mission systems engineer, englobing from the design process to some contraints and budgets that the systems engineer must adbi by from a higher level to assure a successful architecture integration.

Three elements are what is needed to comppose an unmanned spacecraft: a payoad, a spacecraft bus, and a booster adapter. Since the authors pariority is to explain and demonstrate the engineeriong of thhe payload integration with the spacecraft bus, the booster adapter, which can be known as the launchc 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 technolgy usually known as "tech demo", or and instrument to obtain some sort of science or service. Missions, commericial 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 aound it.[0]

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 reffered 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 specificly detailed component state (Level 4 and beyond), all this without designing the item its reffering to. This may sound simple, but it can become complicated, specifically when a writing very detailed lower level requirement. The verification of the requirements will be what defines mission success or failure. Many missions are cancelled when all is set and done due to not being able to verify all requirements. This is why it is not unusual to see several versions of a requirements docuement.

Higher level requirements and contstraints are set by the concept and architecture of the mission, and by the payload operations. Going from a spacecraft design prespective, the orbital trayetory 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.[0]

D. Cubesat 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 Dr. 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.[4]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, 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 big 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.[5]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.

E. 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.[6,7] 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]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

Figure 1: TechEdSat (accompanied by another 2 cubesats) being deployed from ISS.[10]

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.[8] 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.[9] The Blue Canyon Technologies XACT system also counts with a star camera, that promises much more accurate knowledge within ±0.007°, but again the issue is cost.[3]

III. 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.[10] 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 is currently flying and is planned to re-enter atmosphere on April 11 +/- 1, 2015.

IV. 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.[11]

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.[12]

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.

F. Objective

The objective of this report is to inform and be used as a guide of how to start with space mission design, with out leaving out the objective of designing, testing and, if money and time permits, fly onboard a cubesat a attitude control system at low cost that will help impulse 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 helio-physics' missions proving the restringent control requirements needed.

V. Subsystem Description (from SMAD formulas)

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*".[7] 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 each subsystem.

A. Mission Design

The mission design refers to the parameters of flight of the mission. These parameters include orbit, launch and duration. All assumptions that were done to accommodate the model to the needs of the spacecraft. The orbit will match the ISS' at 350 km.[13] The duration of the mission is set to last a year, although requirements expect less, but this will allow for possibility of exceeding the success criteria.

B. Power

The power subsystem can be then divided into 4 parts:[13]

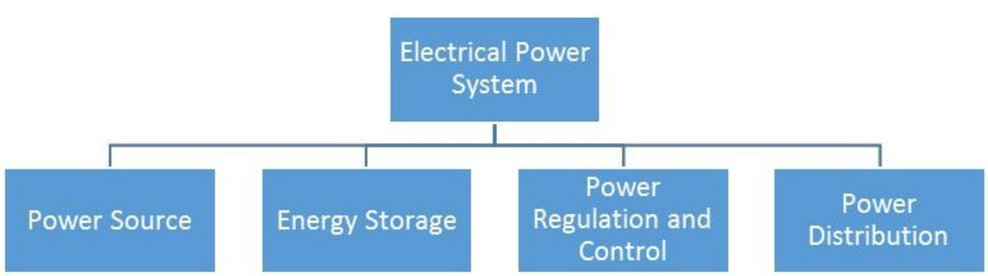


Figure 2: Electrical Power System Decomposition as displayed in "The New SMAD"[13]

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.

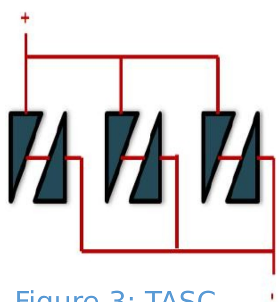


Figure 3: TASC Circuit

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.[14] For each allocated U of solar panel area, 10 pairs of TASC are fit to assure space for circuitry, as explained by Stanton.[15] 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.

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*:[13]

$$= \quad +$$ (1)

The period of the orbit (P) at 350 km is 92 minutes. Knowing that the period for eclipse is the same as the max eclipse fraction (f_E), a value of 37 minutes is obtained. A simple subtraction will allow to obtain the daytimeperiod .

$$= P -$$ (2)

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 $X_d = 0.80$.[13] P_e and P_d represent the values of power consumed and for this reference are meant to be 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.

C. Communications

The initial studies done for this report have limited the communication to power and mass. The spacecraft is planned to use the GlobalStar constellation which orbits at roughly 1400 km with a 52° inclination angle.[16]This will allow the spacecraft to have low cost communications and roughly negligible ground operations cost.

Projecting that the spacecraft will only downlink health data and small data files which require very little on board memory and although the data rate can seem low at 1200 bps. Initial estimations for the link budget are still in work but from the information obtained from previous TechEdSats and the low quantity of data generated by the instrument it is not considered an item to be on the critical path of the project.

D. 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.

E. Structure

Having the majority of the spacecraft bus subsystems defined, a more detailed design can be made to obtain a bottom up mass budget. *SMAD*mentions that by this time there should be good understanding and documentation of the mission and spacecraft system level requirements.[13]

Utilizing the CubeSat Kit 3U Solid Wall Structure made of Aluminum with a mass of 0.439 kgs.[17]Taking into account mass factors of 5% for cabling, higher fidelity can be obtained from initial mass budget.

With the assumptions taken, which is a rectangular extracted Aluminum 6061 tube of approximate 30 cm x 10 cm x 10 cm, for the 3U structure. To be able to find the bending stiffness of theAluminum has a modulus of elasticity E=68.2 GPa. For the initial study and reference, the moment of inertia will be calculated with:[18]

$$= \frac{h^3}{12} \quad (3)$$

F. ADCS

Mentioned before, the ADCS system for initial analysis is being considered to be a MAI-101.[2]This will allow to incorporate its mass and power values with certain criteria within the different phases of the spacecraft.

The attitude of a spacecraft according to the fundamental equations of motions, specifically for rotational dynamics. Expressed in vector form:[13]

$$= - \times \quad (4)$$

The translational momentum will remain constant unless an external force acts to modify it, this is calculated as mass times velocity.The angular momentum in a body will continue unless disturbed by a torque, this can be calculated with the moment of inertia times the bodies angular velocity. With the previous equation it can be noted that the to change the magnitude of the angular momentum a **T** must be applied, for the term × will only generate a change in the direction of . This allows to come to the conclusion that in absence of external torques, if part of the body starts to rotate in one direction, the rest will start to rotate in the opposite direction, for conservation of momentum. This will allow to relate to with the following equation:[13]

$$= + \quad (5)$$

Here, **I** is the moment of inertia and **h** is the angular momentum of any spinning object, in the case of this study the reaction wheels. This can be rewritten as a matrix equation:[13]

$$+ + = - \times \quad (6)$$

Moving things around:[13]

$$= - - - \times \quad (7)$$

This allows for a comprehension of the different methods that can cause an attitude change.[13]

With the help of these equations and knowing some initial data of the hardware to be utilized, a closer approximated parametric model can be obtained. Taking into account all background references.

G. Thermal

The spacecraft shall have 3-axis accuracy pointing of ±0.5°

The spacecraft shall be able to accommodate the X-123, providing TBD connection for data and provide 2.5W of power within 4V and 6V.

The spacecraft shall have the capability to function and obtain data for at least 3 months of constant operations.

The initial parametric model obtained does not have active thermal equipment.

H. Payload

The payload to be utilized is an Amptek X-123 X-ray detector. The information obtained will allow us to incorporate it into our parametric model with its specifications.[20]

I. 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.

Three main objectives have been set by to design the spacecraft.

Pointing Accuracy	
Instrument Integration	
Lifetime	

Table2: System Level Requirements

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.

VI. Initial Models

The model run is being done so with NASA Ames Research Center parametric tool ATLAS.[22] This model was created using the parameters found in the previously mentioned research and analysis done. Several

Figure 4: Design Power Modes

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

	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			

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.[13]

$$= 2 \qquad 3 \tag{8}$$

In the previous equation the period (P) of the orbit, which is how long it takes for an object, in this case our spacecraft, to orbit another object, in this case earth. The other variables utilized are the semi-major axis (a) and the gravitational constant (μ) which for earth is 3.986004×10^{14} m^3/s^2.[13] To know the eclipse time of the orbit there is a bit more to do than for the period. The maximum eclipse fraction (f_E) needs to be found with the following equation:[21]

$$(f\,) \qquad = \frac{1}{} {}^{-1} \qquad \frac{h^2+2\,h}{+h} \tag{9}$$

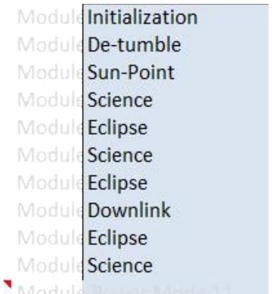


Figure 5: Power Mode Order

Now that the period and eclipse fraction is known the power modes can be strategically set to fit the concept of operations. This can be done with simple arithmetic. 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.

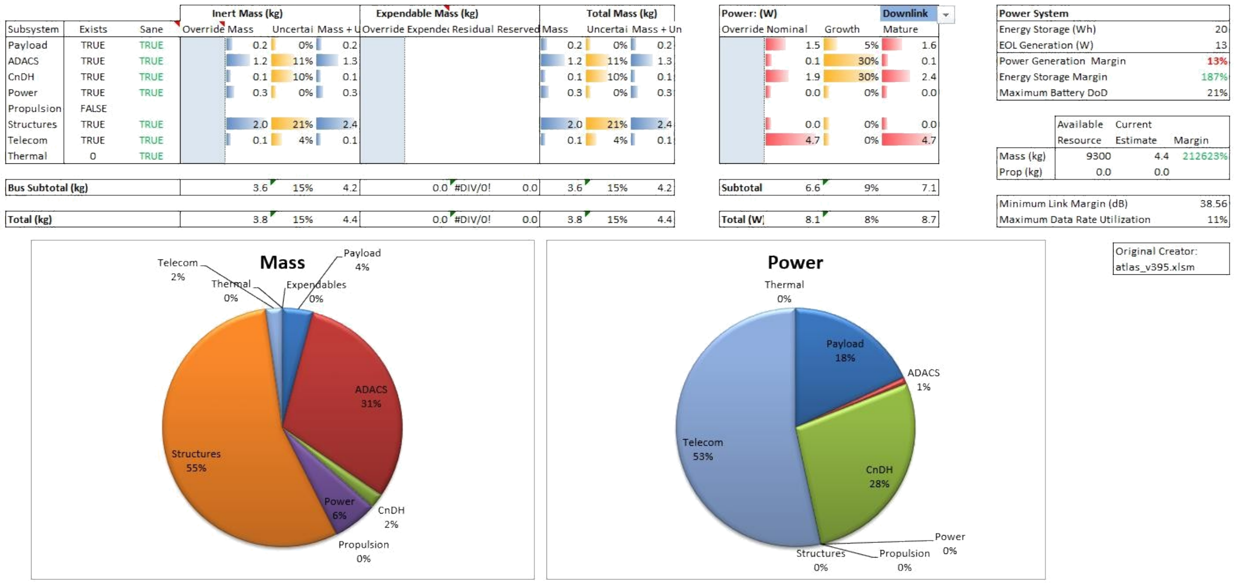


Figure 6: Power and Mass budget of Downlink Mode

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 is exceptional.

2. Science

The second mode to be analyzed is the Science mode, which repeated more than the Downlink. This will be

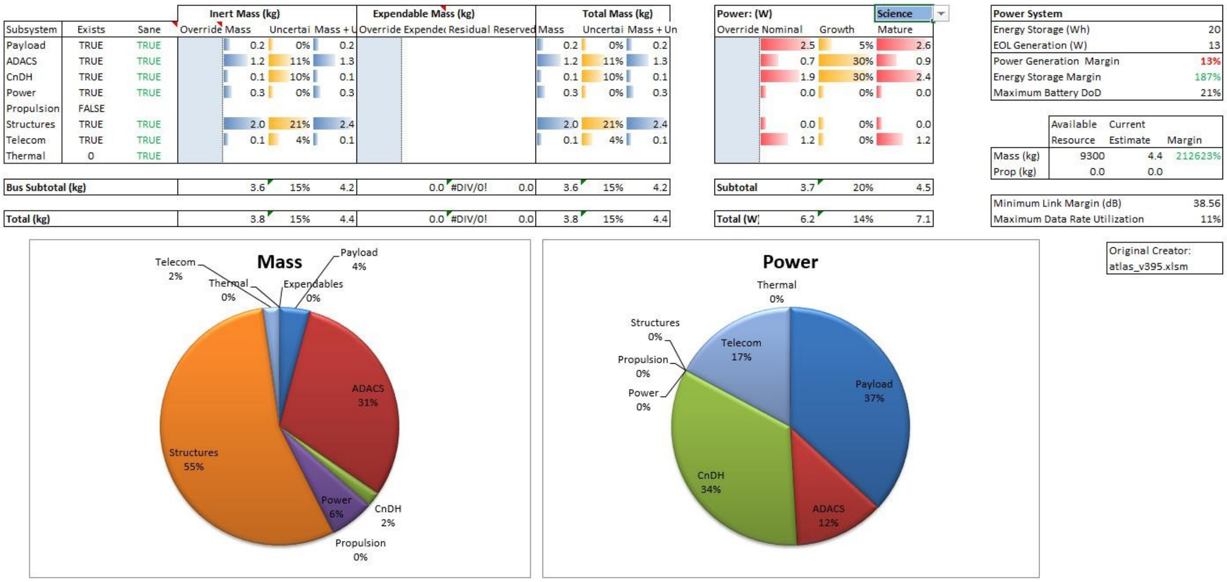


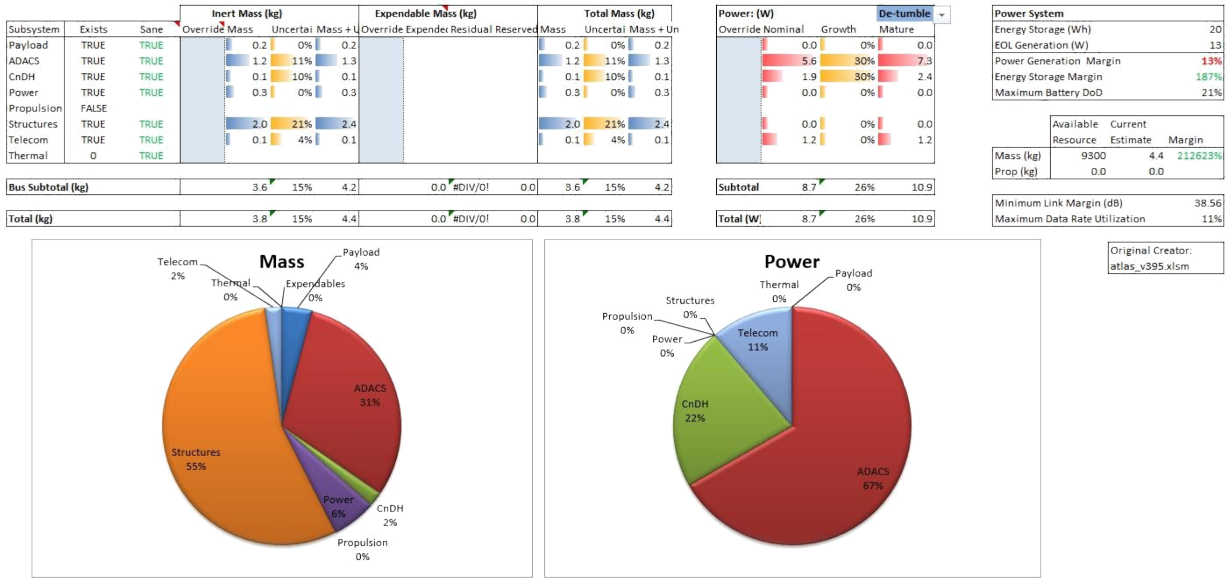
Figure 7: Power and Mass budget during Science Mode

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.

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.



			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%

Figure 8: Power and Mass 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.

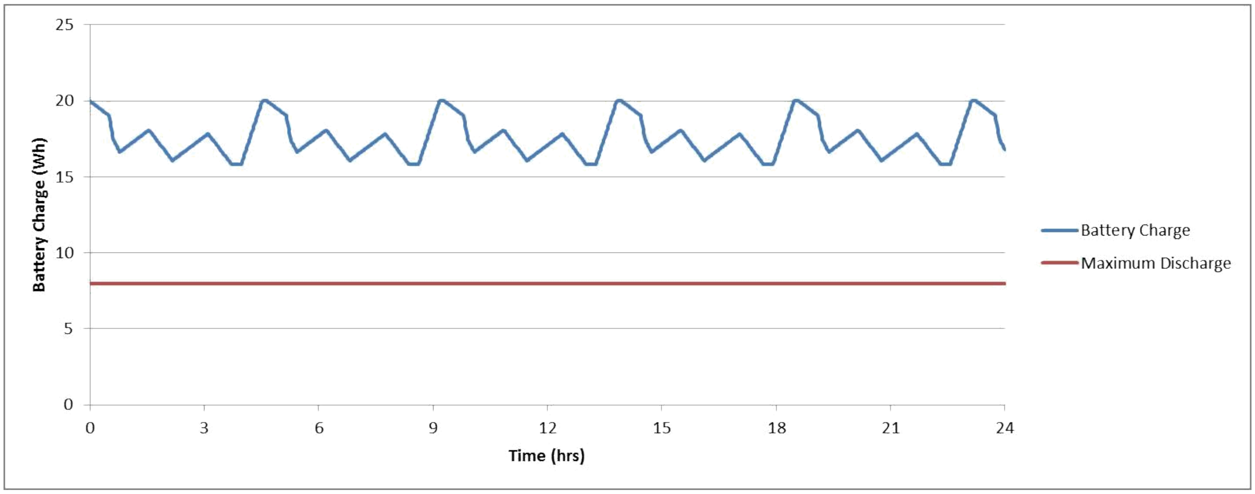


Figure 9: Battery Charge/Discharge during 24 hours of Operations

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	3	0.049	0.09	1	0

		Design)						

				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				

Table3: Main Equipment List (MEL)

VII. Test Planning

The testing of the cubesat is set to take place at NASA Ames Research Center, bldg N269, Spheres Lab.

This lab contains a testing fixture utilized for semi-friccionless 6 DOF movements.

22
San Jose State University

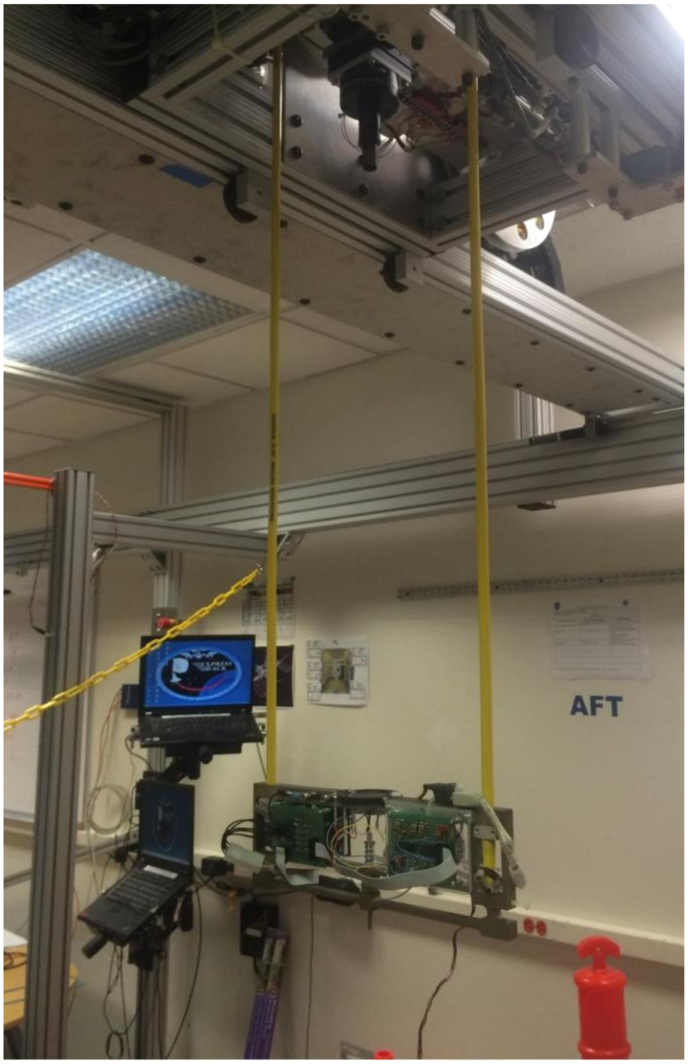


Figure 10: NASA ARC SPHERES Test Fixture Right

The test fixture was initially designed and manufactured for the Spheres ISS project. This fixture will allow coordinated and commanded maneuver, thus verifiying the control system designed. While a descope is being proposed for a spin stabilizing system, to replace the 3 axis system. Concurrent decisions are being made to get to a resolution as soon as possible.

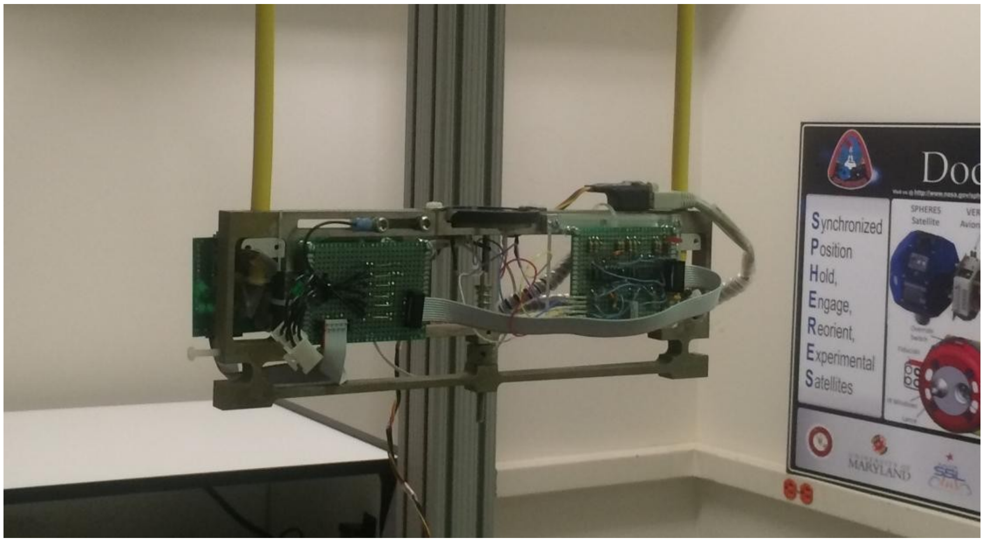


Figure 11: NASA ARC SPHERES Test Fixture

The testing is still in planning and further explination is to be added.

VIII. Conclusion

A summary of the mission and results from the design process will be discussed.

Appendix

An appendix, if needed, should appear before the acknowledgements.

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References

[0]J.R. Wertz, W.J. Larsonl, “Space Mission Analysis and Design” Thrid Edition, Space Technology Library, Microcosm Press, Hawthorne, CA, 2010.

[1]112th Congress 1st Session, *To authorize the President to remove commercial satellites and related components from the United States Munitions List to certain restrictions, and for other purposes.* Washington, DC: In the House of Representatives November 1, 2011.

[2]Maryland Aerospace, Inc, “MAI-101 Miniature 3-Axis Reaction Wheel,” Budgetary Quotation MAI-QD-2015-04-01-MAI-101, April 6, 2015.

[3]Blue Canyon Technologies, “XACT, High Performance Attitude Control for Cubesats” Data Sheet, No Date.

[4]J. Farkas, “CPX: Design of a Standard Cubesat Software Bus,” California Polytechnic State University, June 15, 2005.

[5]Yost, B.,Hines, J., Agasid, E., Engelbert, D., Ricco, A., Kitts, C., “The GeneSat-1 Test Demonstration Project: A Unique Use of Smallsats,” 19th Annual AIAA/USU Conference on Small

[6]Haque, S., Teel, G., Keidar, M., Agasid, E., Tintore, O., Dono, A., Trinh, G., Uribe, E., “Micro Cathode Arc Thruster PhoneSat Experiment for Small Satellites,” IEPC-2013-409, 33rd International Electric Propulsion Conference, The George Washington University, Washington, DC, USA, October 6-10, 2013.

[7]Pottinger, S. J., Scharlemann,C.A., “Micro Pulsed Plasma Thruster Development,” IEPC-2007-125, 30th International Electric Propulsion Conference, Florence, Italy, September 17-20, 2007.

[8]EPSON, “M-V340-xx, IMU (Inertial Measurement Unit)” Data Sheet, 412572600, Rev.20130717, 2013

[9]SelexES, “S3 Smart Sun Sensor” Data Sheet, ASD MM07948 11-13 DRAFT, 2013

[10]eoPortal Directory, “TechEdSat (Technology Demonstration Cubesat),” eoPortal Directory, [Online]. Available at: https://directory.eoportal.org/web/eoportal/satellite-missions/t/techedsat [Accessed May 1, 2015]

[11]Nigg, D., Kinsey, R., Lewis, B., Vaughan, R., “Expanding Concurrent Engineering Design Center Capabilities to Meet Future Challenges,”, 4th International Workshop on System & Concurrent Engineering for Space Applications, SECESA 2010, pp 2-10.

[12]Lewis, B., Lang, J., Jolly, R. “Modular Concurrent Engineering Models: Enabling Alternative Models in Conceptual Satellite Design,” IEEEAC paper #1691, pp 1-7.

[13]J.R. Wertz, D.F. Everett, J.J. Puschell, “Space Mission Engineering: The New SMAD,” Space Technology Library, Microcosm Press, Hawthorne, CA, 2011.

[14]SpectroLab Inc., “Triangular Advanced Solar Cells (TASC)” Data Sheet, 4/10/02.

[15]Wayne, S., Schulz, P., Wise, J., King, B., “Intelligent Power Management System for a Nanosatellite,” No Information.

[16]Guarneros Luna, A., “New Communication Structure for Small Spacecraft,” Aerospace Engineering Department, San Jose State University, Master Project, spring 2013

[17]CubeSat Kit, “3D CAD Design, 3D Models of the CubeSat Kit,” CubeSat Kit, [Online] Available at: http://www.cubesatkit.com/content/design.html

[18]Advanced Mechanical Engineering Solutions, “Sectional Properties Calculator- Solid Rectangular Bar” AMES, [Online]. Available at: http://www.amesweb.info/SectionalPropertiesTabs/SectionalPropertiesSolidRectangle.aspx

[19]Westfall, A., “Design of Attitude Determination and Control System for HelioCube 3U Cubesat” Masters Thesis, San Jose State University, Westfall, A., San Jose, CA 2015 (unpublished)

[20]Amptek, “RoHS/WEEE Compliance X-ray Detector, Complete X-Ray Spectrometer, X-123” Data Sheet, http://www.amptek.com

[21]Names Withheld, “Power System Design and Trade Study for a Satellite Killer Spacecraft,” Massachusetts Institute of Technology, Cambridge, Massachusetts 02139, pp 7-9.

[22]ATLAS, Mission Design Division, Engineering, NASA Ames Research Center, Bldg N202, Moffett Field, CA, 94035