Project: JB3-CBS1

Mechanical, Power, and Propulsion Subsystem Design for a CubeSat

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by

Keith Cote

Jason Gabriel

Brijen Patel

Nicholas Ridley

Zachary Taillefer

Stephen Tetreault

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Prof. John Blandino, Project Advisor

Abstract

This project explores Worcester Polytechnic Institute's (WPI) initial venture in experimenting with a type of picosatellite called a CubeSat. Three Major Qualifying Projects (MQP) representing seven subsystems collaborated on the construction of a ground-based CubeSat to test current technologies and investigate the feasibility of future CubeSat projects at WPI. Of the seven CubeSat subsystems, this report outlines efforts of the power, propulsion, and structure subsystems. Research on previous and current CubeSat projects provided baseline information, giving teams the ability to select components for a "Lab Option" as well as "Flight Option" CubeSat.

Although construction and testing of a full Lab Option CubeSat was beyond the scope of this project, each of the three subsystems teams were able to design and/or construct a baseline set of components for their subsystem and perform rudimentary testing. The extensive research and recommendations detailed herein will be used by future groups to prepare a space-ready satellite. In addition, this project (in conjunction with two other CubeSat design teams) resulted in a fully defined Flight Option CubeSat, including component selection and mission planning, for a 3U CubeSat carrying an Infrared Spectrometer.

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Associate Professor, Aerospace Engineering Program

Department of Mechanical Engineering, Worcester Polytechnic Institute

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Professor Michael Demetriou, Ph.D.

Professor, Aerospace Engineering Program

Department of Mechanical Engineering, Worcester Polytechnic Institute

Professor Nikolaos Gatsonis, Ph.D.

Director, Aerospace Engineering Program

Department of Mechanical Engineering, Worcester Polytechnic Institute

Authorship

Nicholas Ridley and Jason Gabriel wrote the Abstract, Lists, Executive Summary, Introduction, Mission and Payload Literature Review, Goals, Methodology chapters, and the Power Literature Review. Keith Cote and Brijen Patel wrote the Mechanical Structures Literature Review. Stephen Tetreault and Zachary Taillefer wrote the Propulsion Literature Review. All group members edited and revised various sections. In its final form, this report contains equal contributions from all group members, and each section represents the collaborative effort of multiple authors.

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Executive Summary

In 1999, professors at California Polytechnic State University (Cal Poly) and Stanford University outlined a set of specifications for a simple picosatellite, and the CubeSat was born. A CubeSat is a small, relatively easy-to-construct, and relatively low-cost, satellite based on a standardized design. The set of specifications is meant to "provide a standard for design of picosatellites to reduce cost and development time, increase accessibility to space, and sustain frequent launches" [42]. The target audience for this satellite standard would be universities, who would construct satellites as a way to introduce students to a realistic and practical spacecraft design and mission launch process.

This project represents the work of three of the seven subsystem teams responsible for the design, construction, and testing of a ground-based CubeSat. The collective Aerospace MQP student group, consisting of three teams broken into seven smaller subsystem teams was required to design a satellite to house the Argus 1000 IR Spectrometer in a circular orbit with altitude 680 km and period of 98.2 minutes. Teams researched laboratory and flight-qualified options for satellite components, accounting for mission and scientific payload requirements. The "Lab Option" satellite will be constructed and tested in WPI's vacuum chamber by future MQP Groups, while a set of recommendations will be put forth by all teams to comment on the requirements for a space-ready "Flight Option" satellite to be built by future teams.

This report presents the research and design of the power, propulsion, and structural subsystems. Our team spent the first of three seven week terms conducting research into previous and current CubeSat technologies, which created a baseline understanding of the technology and allowed us to explore technology applicable to our particular satellite and mission.

1 Introduction

Space exploration and research is one of the most alluring and prestigious endeavors within Aerospace Engineering. However, many engineering students do not get the opportunity to work on space-oriented research until, at the earliest, the start of their professional career. Moreover, the cost of sending vehicles and satellites into space compounded with the enormity of work involved make for infrequent missions, meaning engineers working on space systems often do not get many opportunities for the practice of launch and flight operations. However, in 1999, the California Polytechnic State University, San Luis Obispo [43] and Stanford University developed specifications for a class of picosatellites. These picosatellites were given the term “CubeSat,” whose small design (1-3 liters) and relatively low cost (construction and launch: $65,000-80,000) appealed to universities and companies worldwide [4]. Moreover, a standardized deployment system, the Poly-Picosatellite Orbital Deployer (P-POD), allows for any CubeSat to be carried into orbit (on a space-available basis) and deployed, as long as said satellite adheres to the CubeSat criteria [4]. The ease of creating an operational satellite using readily available electronics offers students the experience of mission planning and spacecraft design long before they would receive similar on-the-job experience, making the construction and launch of CubeSats an attractive tool for academia. Additionally, CubeSats are often outfitted with a variety of scientific instruments, although it is important to note that due to the size and power available to a CubeSat, satellites typically support only two instruments at most. This allows a CubeSat to serve a practical purpose in addition to its educational value. Moreover, CubeSats provide researchers not affiliated with the university developing the CubeSat with a low-cost space vehicle with which to conduct research. In some cases, these external researchers provide CubeSat teams with the funding support.

In the spring of 2010, Professors Gatsonis, Blandino, and Demetriou, of the WPI Aerospace Program (Mechanical Engineering Dept.) initiated the university's first effort in the area of CubeSat research and development. An eleven person team of fourth year undergraduate Aerospace students was formed to research and develop the various subsystems of a satellite as part of their Major Qualifying Project (MQP), exploring the potential of this technology through the construction and testing of a ground-based engineering model. Individuals were divided into three MQP teams representing seven subsystems, with each team assigned a different advisor. This report outlines the work done by the Power, Propulsion, and Mechanical Structures subsystem teams (other subsystems include Thermal Control, Payload, and Attitude Control and Dynamics). Teams were responsible for researching Lab and Flight Options for the satellite, coordinating efforts and tasks for satellite construction, and eventually testing the Lab Option satellite. The Lab Option is defined as a satellite constructed primarily with off-the-shelf parts to fit within the project's limited budget (approx. $2000). In addition, the Lab Option involved other cost-saving measures such as replacing the scientific payload or other expensive components with "black box" components (to simulate mass properties), or the use of a power umbilical to simulate different solar cell and battery power sources using laboratory power supplies.

The results of this MQP will lay the groundwork for future CubeSat groups. A set of conclusions and recommendations will be published, which will allow groups to apply lessons learned to development of a space-ready, or "Flight Option" satellite in the future.

1.1 Project Goals and Objectives

The primary goal of this project was to coordinate with two other MQP projects comprising seven subsystems to design, build, integrate, and test a single ground-based CubeSat, which incorporates key elements from each of the included subsystems. This allowed us to establish a baseline design for the CubeSat subsystems, and lay the groundwork for future CubeSat projects at WPI, which could lead to space-ready satellites. Our objectives for this project were to:

- Select components for both a "lab" and "flight" option CubeSat
- Integrate these subsystems
- Construct a Lab Option satellite as a "proof of concept" which can be used for hardware/software testing and construct a test fixture to support the Lab Option CubeSat in a vacuum chamber
- Perform testing of the completed Lab Option CubeSat in a vacuum chamber
- Create a set of recommendations for the Flight Option CubeSat for future groups to

reference

As the initial CubeSat project at WPI, much of our work will lead to improvements in the organizational structure and planning of the project, as well as the establishment of a baseline design for future groups.

1.2 Power Subsystem Objectives

The purpose of a power system on a satellite is to produce, store, manage, and distribute power to the systems that need it. In the case of this project, the power team's objectives were twofold. First, the power subsystem team was responsible for designing both a lab and flight option power system to include the four necessary functions stated above. This design needed to include specific details regarding the power system, including the amount and type of power provided, power needs of users, and specific components such as DC-DC converters, on/off switches, and battery management components. Moreover, the design needed to show the appropriate circuitry required to make each component function. Secondly, the power team needed to construct and test the Lab Option power system. While the Flight Option plan was intended for project continuity, the Lab Option needed to be constructed to allow preliminary testing of the satellite hardware and software. Without a working power system, many of the other subsystems cannot not be tested, and the overall project objectives will not be met.

1.3 Propulsion Subsystem Objectives

The preliminary design of a Flight Option propulsion subsystem was completed as a recommendation for future MQPs. Specific objectives for the Propulsion Subsystem are listed below:

1. Review previous work and available information for CubeSat propulsion.
2. Identify candidate technologies for laboratory and flight-qualified versions (e.g. cold gas, pulsed plasma thruster, etc.)

3. Generate a complete system design schematic for the baseline Lab Option (to aid in assembly planning and component selection).
4. Define power and command requirements for baseline Lab Option.
5. Collect mass, volume, power and cost information for all Lab Option components (and as many of the flight option components as possible).
6. Assemble the Lab Option and work with other team members to integrate the components
7. Define test(s) to be performed in vacuum chamber
8. Support testing and document results.
9. Incorporate all research, design, and test results into final report with other subsystems.

A major objective for this subsystem team was to design, build, and test a fully functioning prototype of a cold-gas propulsion system for a CubeSat. This system, designed for ground-based testing in a vacuum chamber, needed to be capable of demonstrating spacecraft control about one axis of rotation.

It was not possible to build an actual flight model with the time and budget available to this MQP, so the subsystem team focused on designing and building a working lab prototype of the CubeSat propulsion system. This lab option provides a proof-of-concept propulsion subsystem capable of maneuvering the satellite in Low Earth Orbit (i.e. providing primary ΔV) and supporting the minimum pointing requirements for the satellite's scientific payload (i.e. providing attitude control).

1.4 Mechanical Structure Subsystem Objectives

Design and Construct a CubeSat Lab Model and Test Fixture

Foremost, the main objective for the Structure Subsystem team was to design and construct a working prototype for a 3U CubeSat Structure for the purpose of performing laboratory tests as well as to design and construct a one degree-of-freedom (1DOF) rotation test fixture. Candidate designs for the lab model CubeSat structure and test fixture were created using computer-aided design (CAD) software, which will then be fabricated and assembled using computer-aided manufacturing (CAM) software as well as WPI's computer numerical controlled (CNC) machine tools located in Washburn Labs. Both assemblies will be designed and constructed for use inside a vacuum chamber and will be used for testing of both hardware and software.

Make Recommendations for a Flight Option CubeSat and Test Fixture

Secondly, since the Lab Option CubeSat will be treated as a proof of principle for a future Flight Option CubeSat proposal, the key objective will be to design optimal flight model designs for the CubeSat structure as well as the test fixture using CAD software. Optimal flight models will be designed implementing alternative lightweight materials as well as optimized structures that provide minimization of mass while allowing for the maximization of structural integrity. Recommendations will be given regarding Flight Option CubeSat structure and the test fixture designs and they will be incorporated into future proposals for a Flight Option CubeSat structure.

Mechanical & Structural Support for other Subsystems

Lastly, using the technical expertise with regards to mechanical & structural systems gained as part of the background research, the final objective will be to support other subsystems with structural hardware design, fabrication, and assembly as needed. This is done through creating an Integrated 3U CubeSat Assembly Model, which includes the primary structure as well as all the different subsystem component parts. Therefore, design decisions can be made regarding the placement, size, and mass of the different components allowing for an integrated assembly.

2 Background

The CubeSat is a standardized picosatellite[1] developed as part of a collaborative effort between California Polytechnic State University, San Luis Obispo, and Stanford University's Space System's Development Lab [4]. The goal of the CubeSat program is to provide standardized design specifications and deployment systems so that universities can design, build and launch satellites more affordably [4]. The basic CubeSat consists of a 10 cm cube with a mass of up to 1.33 kg [4]. Other common CubeSat designs consist of two or three of the 10 cm cube units oriented linearly [4]. Some companies that sell prefabricated CubeSat structures offer models in increments of 0.5U ranging up to configurations as large as "6U"[2] but to date, no CubeSats exceeding 3U have been launched [7].

During launch, the CubeSats are loaded into a deployment vehicle called a P-POD, which stands for "Poly Picosatellite Orbital Deployer" [4]. The P-POD is three units long, so multiple configurations of CubeSats can be loaded such as three 1U or one 3U satellites for example [4]. For cases in which CubeSats are larger than 3U, custom-made P-PODS must be built or purchased [4]. In order to ensure successful integration with the P-POD and standardization of all CubeSats, stringent design specifications have been defined for developers by Cal Poly.

Figure 1 – 1U and 3U CubeSats [4]

[1] Satellite with a wet mass between 0.1 kg and 1 kg

[2] The nU nomenclature is used to describe the size of a CubeSat in multiples of the unit CubeSat

2.1 General CubeSat Specifications

A master document called "CubeSat Design Specification" which outlines all of the requirements that must be met in designing a CubeSat is updated and distributed by Cal Poly [4]. The specifications are classified as general, mechanical, electrical, and operational design constraints. All of the mechanical and some of the general specifications apply to the design of the CubeSat structure. The specifications document also describes the waiver process that must be followed if for any reason, the satellite deviates from the set specifications. Finally, the document defines the testing requirements that must be met by each CubeSat in order for the satellite to be accepted for launch. These testing requirements include Random Vibration, Thermal Vacuum Bakeout, Visual Inspection, Qualification, Protoflight, and Acceptance and are explained in detail in Section 2.1.3.

2.1.1 Power Subsystem Specifications

Compared to other subsystems, there are very few requirements for the electrical system set by Cal Poly. The document requires only the CubeSat be able to undergo a "Dead Launch", meaning that all electronic systems are deactivated during the launch phase and all batteries are either disconnected or fully discharged. The electrical system must have a "Dead Switch" that is actuated upon ejection from the P-POD, activating all electrical systems in the satellite. The CubeSat must also have a "Remove Before Flight" pin to prevent any electrical systems from inadvertently activating during ground testing.

2.1.2 Propulsion Subsystem Specifications

The CubeSat Specifications Document does not put any restrictions explicitly on a propulsion subsystem. However, under the "General Requirements for CubeSats", for any vessel, a maximum pressure of 1.2 atm (0.12159 MPa) is set and a factor of safety no less than 4. This limits the pressure at which the propellant can be stored, which in turn limits the amount of propellant that can be stored. In addition, it can limit the specific impulse (Isp) and thrust capabilities, if the thrust level relies heavily on the storage pressure of the propellant. This section also disallows the use of pyrotechnics of any form onboard a CubeSat. Pyrotechnics are widely used for chemical propulsion as an igniter. Occasionally, pyrotechnic valves can be used to isolate propellant in a propulsion system as well. The restriction of pyrotechnics onboard the CubeSat effectively eliminates these propulsion options from consideration. Further restrictions on use of hazardous materials implicitly limit the allowable propellant types.

2.1.3 Mechanical and Structural Subsystem Specifications

The bulk of the specifications set for the structure of the CubeSat consist of dimension requirements in order to ensure compatibility of CubeSats with the P-POD. The critical dimensions for each basic CubeSat configuration are listed in Table 1 and a schematic diagram of a 1U CubeSat is shown in Figure 2. As shown in the diagram, The CubeSat consists of six 10 cm by 10 cm walls assembled into a cube and rectangular rails along the corners which make contact with the P-POD during integration [4]. A coordinate system defined in the design specifications [4] orients the Z-axis parallel to the four rails.

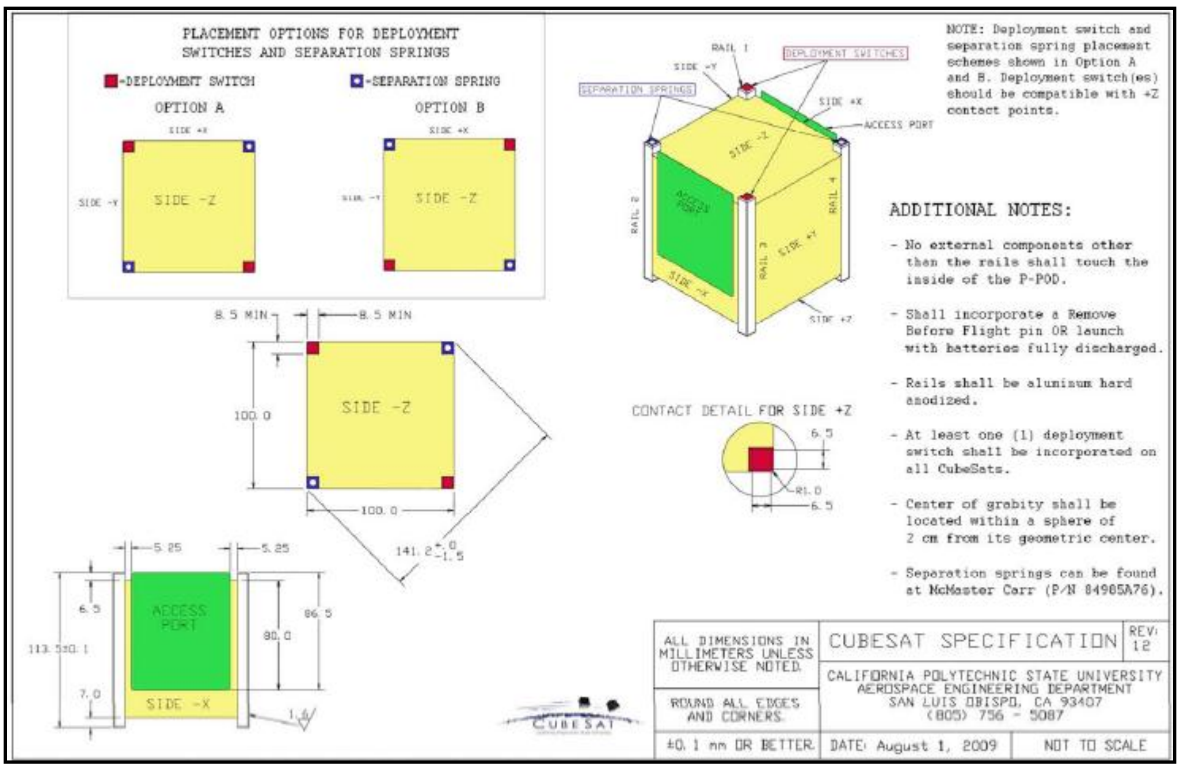

Figure 2 – 1U CubeSat Specification Diagram [2]

CubeSat Size	1U	2U	3U
X and Y Dimensions [mm]	100 ± 0.1		
Z Dimension [mm]	113.5 ± 0.1	227 ± 0.2	340.5 ± 0.3
Rail Width [mm]	8.5 x 8.5 mm MIN		
Rail Contact w/ P-POD (75 % of Z Dimension) [mm]	85.1 (minimum)	170.2 (minimum)	255.4 (minimum)
Component Protrusion normal to cube surface [mm]	6.5 mm (maximum)		
Mass [g]	1330 (maximum)	2660 (maximum)	4000 (maximum)

Table 1 – Critical Dimensions for 3 Primary CubeSat Sizes [2]

Also, as specified in the document, the only components of the CubeSat that may make contact with the P-POD are the four rails. This means that all deployable components of the satellite must be constrained within the CubeSat, so as not to interfere with the P-POD interface. In order for individual 2U and 1U CubeSats to separate from each other after deployment, they must use separation springs built into the ends of the rails. 3U CubeSats do not require separation springs since only one 3U CubeSat can fit into a P-POD. A diagram of a P-POD is shown in Figure 4. To reduce the amount of additional space debris introduced with each launch, all parts shall remain attached to the CubeSat through launch, ejection, and operational phases. In order to prevent cold welding[3] of the surfaces of the CubeSat to the P-POD and to ensure that the satellite maintains a coefficient of thermal expansion similar to that of the P-POD, the document specifies that the material for rails and primary structure of the satellite to be hard anodized Aluminum 7075 or 6061. Finally, the document specifies that for each CubeSat configuration, the center of mass shall be located within a radius of 2cm from the geometric center of the satellite.

[3]Cold welding- "The joining of materials without the use of heat, can be accomplished simply by pressing them together. Surfaces have to be well prepared, and pressure sufficient to produce 35 to 90 percent deformation at the joint is necessary, depending on the material. Lapped joints in sheets and cold-butt welding of wires constitute the major applications of this technique". [17]

[4] Aluminum 7075 is a stronger alloy that can be machined thinner consisting mostly of Zinc as the primary alloying element, but Aluminum 6061 is a cheaper, lighter alternative with Magnesium and Silicon as the primary alloying elements. [18, 19]

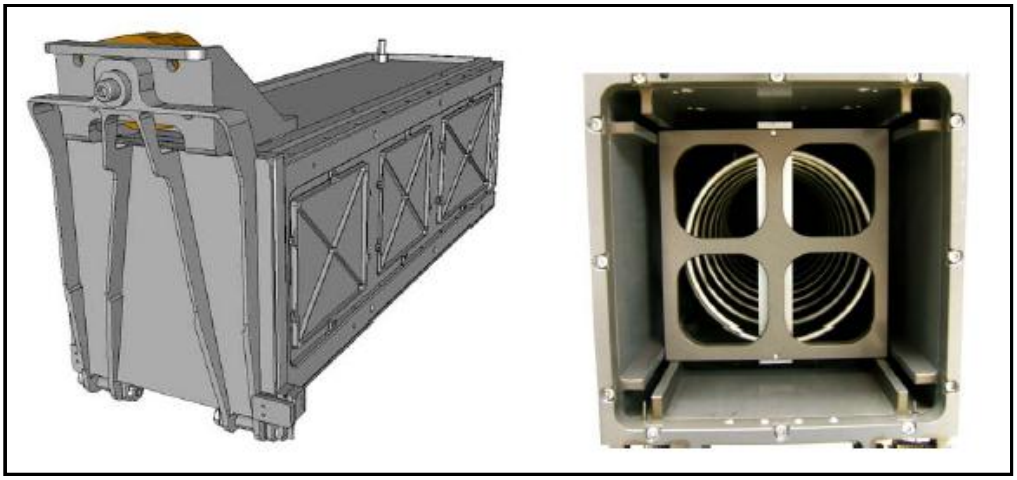

Figure 3 – P-POD Exterior and Cross Section [2]

Before a CubeSat can be approved for launch and integrated into the P-POD, it must first pass certain tests as listed in the CubeSat Design Specification document [4]. The launch provider may also require additional tests not specified in the document. The launch provider could be a private company or government agency [4]. For example, as recently as the summer of 2010, NASA has been offering launch opportunities for CubeSat developers in the 2011-2012 timeframe if the CubeSat and mission met certain specifications such that it would be of benefit to NASA [10]. If the launch environment is unknown, the GSFC-STD-7000 standards as defined by NASA shall be used instead. "This standard , prepared by NASA's Godard Space Flight Center, provides requirements and guidelines for environmental verification programs for GSFC payloads, subsystems and components and describes methods for implementing those requirements" [22].

The first test required for each CubeSat is random vibration testing in which the satellite undergoes dynamic loading that simulates the harsh loads experienced during launch. Additionally, "a thermal vacuum bakeout test shall be performed to ensure proper outgassing of components" [4]. The CubeSat must also pass a visual inspection by the launch provider in order to ensure that all specifications such as critical dimensions are met. The spacecraft must then pass qualification tests as defined by the launch provider. The Purpose of "Qualification tests are to demonstrate that the test item will function within performance specifications under simulated conditions more severe than those expected" so that deficiencies in the design and method of manufacture can be uncovered.

[4]. The qualification tests may either test "prototype" (any hardware of a new design not intended to be flown) or "protoflight" (any flight hardware of a new design) hardware [13]. Finally, the CubeSat must undergo acceptance testing to ensure that the satellite can be properly integrated into the P-POD. In acceptance testing, each component, subsystem, and payload that performs a mechanical operation undergoes a series of mechanical function tests in order to ensure proper performance and that previous tests have not degraded the spacecraft [13]. It is the responsibility of the CubeSat developer to perform all required testing except for the Acceptance testing prior to delivery to the launch provider [4]. California Polytechnic State University can assist CubeSat developers in finding test facilities if necessary or can perform the testing themselves for the developers and can charge the developers if deemed necessary [4].

2.2 Power Subsystem

The power subsystem is responsible for ensuring the power needs of the CubeSat are met. This includes generating power, conditioning and regulating power, storing energy for use during periods of peak demand or eclipse operation, and distributing power through the spacecraft. It is natural, then, that the power system be thought of as consisting of three basic building blocks: power sources, energy storage, and power management and distribution. A typical CubeSat design uses solar cells for power generation and a small battery for storage. The Power Management and Distribution (PMAD) system is responsible for many tasks, including conditioning the power to the specific voltage and current requirements of each component, making decisions about which systems should receive power when demand exceeds the power available, effectively distributing power to all subsystems at the appropriate time, and switching devices on and off [7].

2.2.1 Solar Cells

Solar cells essentially use the photovoltaic effect to convert the energy found in sunlight into electricity. Typically made from a semiconductor such as silicon (Si), gallium-arsenide (GaAs), or more advanced gallium-indium-phosphide, gallium-arsenide, germanium ($GaInP_2/GaAs/Ge$) compounds, solar cells on CubeSats are the main source of power when the satellite is in solar illumination (this includes powering the various subsystems and recharging the battery). These solar cells are constructed as either single junction or multijunction cells. Single junction cells work efficiently only over a certain part of the solar spectrum, while multijunction cells are multi-layered and consist of several materials, which allow them to have a higher efficiency over a wider range of the spectrum. Due to their greater efficiency, multijunction cells are typically used in space applications [37].

Many CubeSat projects order one of the pre-made panels produced by the Clyde Space Corporation (Glasgow, Scotland). Clyde Space obtains multijunction solar cells from EMCORE (Albuquerque, NM) and Spectrolab (Sylmar, CA), and creates standard solar cell assemblies for 1U, 2U, and 3U CubeSats, as well as custom arrays.

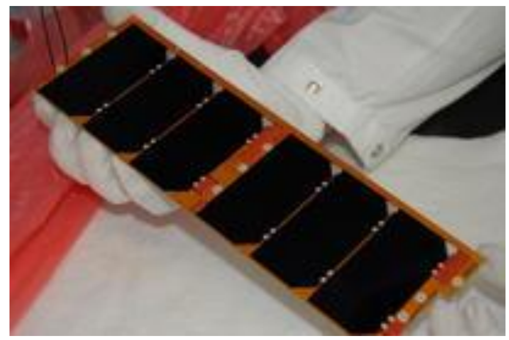

Figure 4 – Clyde Space Solar Cell [35]

2.2.2 Batteries

A battery is simply a cell that converts chemical energy into electrical energy. Due to their small size and short lifespan, CubeSats typically use secondary batteries (or rechargeable batteries) to fulfill energy storage requirements as these batteries are meant to be recharged multiple times. These secondary batteries are charged by power from the solar cells while the CubeSat is in illumination, and then discharged while in eclipse to power any systems that need power while in eclipse. Because these batteries typically cannot fully power all of the CubeSat subsystems by themselves, many components will go into a low-power (or zero-power) “standby” state while the satellite is in eclipse to allow power to be sent from the battery to components requiring constant power. Although less common, some CubeSats also use a primary (non-rechargeable) battery to execute one-time operations (i.e. extending solar arrays after launch).

The management of power flow through the battery, as well as the charging and discharging functions of the battery, are managed by the PMAD (see section 2.2.3). Logic decisions about when to switch between battery and solar power, and when to charge or discharge the battery, are typically made by the flight computer, and carried out by the

PMAD.

2.2.3 Power Management and Distribution System (PMAD)

CubeSats provide a unique challenge in their power requirements and limitations in that they have relatively limited energy sources (small area available for solar arrays, limited mass and volume to accommodate batteries, etc.), while still carrying scientific instrumentation and spacecraft subsystems that require power to operate. Because CubeSats operate on a strict power budget, the proper management and distribution of available power to all spacecraft systems is critical to the survival and operational capabilities of the CubeSat. Complex, integrated Power Management and Distribution (PMAD) systems are often employed on CubeSats to ensure proper allocation of power to onboard systems and prevent damage to electronics from voltage and current spikes [7]. PMADs also provide battery management, controlled capacitor charging/discharging, voltage signal conditioning, and voltage amplification.

Figure 5 – Flight-Ready PMAD from Clyde Space [35]

Every CubeSat currently on orbit employs some form of PMAD system. The most basic conceptual PMAD includes junctions to collect power from all power sources (usually solar arrays), a power conditioner, and a circuit to route power to a satellite's components independently. Most flight-ready PMADs, however, are circuit boards prefabricated with integrated circuits that are designed to meet mission-specific criteria, and are connected using a universal bus to the satellite's components. This allows connections to be made to numerous types of components from multiple manufacturers. Additional components are often added to provide more advanced capabilities: switching to battery power when power from solar cells is inadequate (and charging the battery when power is in surplus), the ability to "dead-launch" with none of the electronics receiving power during the launch but activating upon reaching orbit, and charging and discharging capacitors to provide short "bursts" of energy beyond what the batteries and solar cells can provide. Highly advanced PMAD systems use industry-standard "plug-and-play" power connectors that allow connections to components made by different manufacturers. Some "Smart PMADs" even output data about the health of the power system and status of each power client to be broadcast back to a ground station, and can give commands to the attitude control system to rotate the satellite to maximize solar illumination and "track" the sun along an orbit. These added features make the power system much more functional, but also add a much higher level of complexity to the concept of power management [35].

2.2.4 Sample CubeSat Power Systems

Below are four examples of CubeSat power systems that were designed with the intent to be used in space. Several design considerations and component concepts from these CubeSat designs were adapted to the design of the WPI CubeSat.

AAU CubeSat (University of Aalborg, Denmark)

Begun in September 2001, the AAU CubeSat was a 1U CubeSat initiated with the intent to provide students the opportunity to design and launch a small satellite. Unsurprisingly, power was provided by solar panels and batteries. Solar panels were triple-junction cells from EMCORE and placed in pairs on five of the six sides of the CubeSat (each cell measured 68.96mm x 39.55mm). What was unique was that four batteries from DANIONICS were used, considering the limited space of a 1U CubeSat. Unfortunately, the AAU CubeSat report did not include any more detailed data on their power system. While the AAU CubeSat did make it to space, after two and a half months, the battery capacity significantly deteriorated and satellite operations were unable to continue. [31]

SACRED

SACRED was a 1U CubeSat developed by over 50 University of Arizona students belonging to the Student Satellite Program to conduct radiation experiments. SACRED used six solar cells (one on each face) to provide power, with optimum power generation of 2W and an average of 1.5W. It was also mentioned that SACRED used several batteries, but locating any further data about the power system was futile as no official reports could be found. This could most likely be due to the fact that the satellite was destroyed shortly after takeoff when the launch vehicle failed, and subsequent continuity was not considered necessary. [32]

CAPE-1 and CAPE-2

CAPE-1 was designed as a preliminary CubeSat project to give students at the University of Lafayette the skills needed to design, build, and launch a satellite. CAPE-2 was a more ambitious project, with a primary mission to "develop a cutting-edge CubeSat Communication platform for the CubeSat community to improve data gathering" and secondary missions including "local educational outreach, deployable solar panels, peak power tracking, and software defined radio." While both are 1U CubeSats, these satellites are highlighted here for the developments in their power supply and management. In CAPE-1, solar cells were fixed to the body of the CubeSat, while CAPE-2 will have four deployable solar panels in addition to fixed cells. Additionally, CAPE-2 will be integrating a "peak power tracker" into its PMAD to assist the satellite in orienting itself and its solar panels to generate the most power possible. [33]

Cute-1.7 + APD II Project

Cute-1.7 + APD II is a continuation of Cute 1.7 + APD from the Small Satellite Program (SSP) at the Laboratory for Space Systems (LSS), Tokyo Institute of Technology. A notable improvement in Cute-1.7 + APD II is improved power generation, which had previously limited satellite operations. This will be achieved by increasing the satellite from a 1U to a 2U CubeSat, which will increase the area available for solar cell placement. The solar cells are 38.4mm x 63.2mm high-efficiency (23.2%) Gallium-Arsenide panels from EMCORE placed on all six sides of the satellite, which produce 2.12V at 363mA to power the satellite and charge the Lithium battery. The battery is a four-parallel configuration made by BEC-TOKIN with a nominal capacity of 1130mAhx4 and nominal voltage of 3.8V. Lastly, the PMAD (called the EPS or Electric Power System) is responsible for "detecting the voltage and current of the solar cells," "heating the Lithium Battery," "detecting the charge/discharge current of the battery," and load-leveling functions. [34]

2.3 Propulsion Subsystem

To the best of the author's knowledge, no CubeSat to date has flown with an onboard propulsion system to provide attitude control or perform orbital maneuvers. For this reason, and to increase mission and payload possibilities, propulsion systems applicable to CubeSats have garnered increased attention within the academic community and industry. CubeSats are often not placed in ideal orbits for their scientific payload simply because they are transported to their orbit as "stowaways" on a launch vehicle designed to transport a larger space vehicle whose orbital considerations take precedence. The ability to maneuver from these non-ideal orbits would greatly extend the capabilities of CubeSats.

2.3.1 Pulsed Plasma Thrusters (PPT)

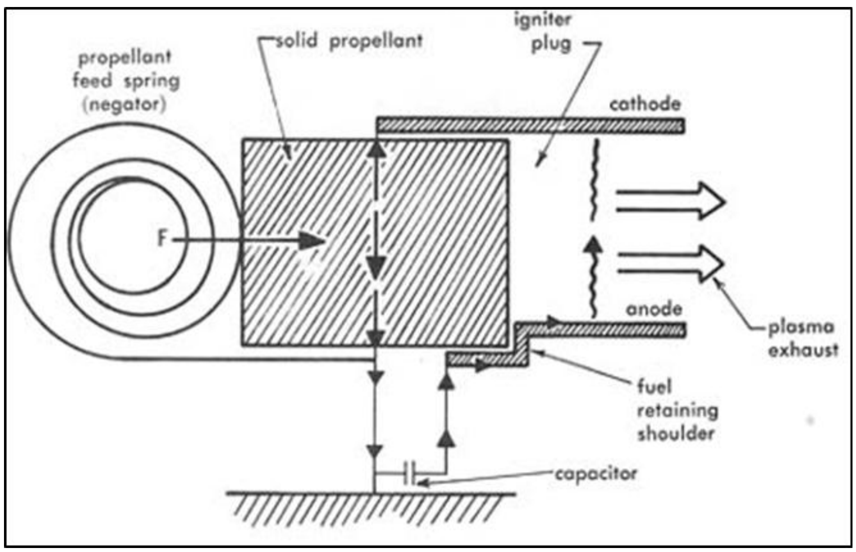

Figure 6 – Schematic of a typical PPT [6]

Pulsed plasma thrusters require low power but provide a high specific impulse. PPTs have been used on spacecraft to demonstrate their ability to provide attitude control and have been proposed for use on spacecraft to enable low thrust maneuvers. A PPT consists of two electrodes positioned close to a solid fuel source (Teflon), which is advanced towards the electrodes by a spring, as shown in Figure 6. Each pulse corresponds to an electric discharge between the two parallel electrodes and results in the ablation of the surface of the solid propellant. This eroded material is expelled out of the thruster at very high velocities due to the Lorentz force (2.1), which is created by the interaction of a magnetic field and an electric current [2].

$$\vec{F} = q\vec{v} \times \vec{B} \qquad \textbf{2.1}$$

Where F is the force (N), q is the electric charge (Coulombs), v is the velocity of the charge (m/s) and B is the strength of the magnetic field (Teslas) [19]. Despite the very low mass of the plasma expelled with each pulse, a useful impulse "bit" (approx. 860 μN-sec) is produced due to the high velocity (approx. 10,000 m/s) of the charged particles [2,3]. At a pulse repetition frequency of 1 Hz, the corresponding thrust for the aforementioned impulse bit would be 860 μN. Due to the large capacitor mass and volume, "conventional" PPT technology, such as the unit flown on EO-1 is much too large to be used on CubeSats [12]. However, a micro pulsed plasma thruster (μPPT) has been developed by the Air Force Research Laboratory (AFRL) (Edwards AFB, CA), which consists of two concentric conductive rods each containing Teflon fuel, see Figure 7 [21]. The fact that the electrode and Teflon fuel recede with each pulse eliminates the need for a spring to advance the propellant to the edge of the electrodes [22]. The inner conductive rod (Teflon) is consumed as fuel during thruster firing and recedes as a result of the erosion. Complications arise when scaling the discharge energy to meet the decreased fuel rod cross sectional area. If the discharge energy is too low, carbon neutrals in the plasma arc can return and collect on the fuel rod surface resulting in "charring". This charring can lead to electrode shorting resulting in thruster failure [21]. Another variation of the μPPT has been developed by Mars Space Ltd. (Southampton, United Kingdom) in collaboration with Clyde Space Ltd. which utilizes the conventional PPT design simply scaled down to meet the volume and power requirements of a CubeSat (Figure 8). The main goal, as stated by Mars Space, is to extend the lifetime of a 3U CubeSat from 3 to 6 years by providing drag compensation.

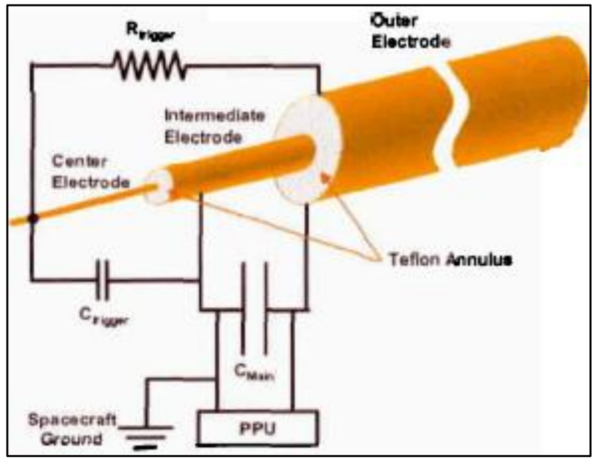

Figure 7 – AFRL Micro PPT concept [21]

Design challenges remain for μPPTs due to a high failure rate caused by electrode surface charring, a limited total impulse and the fact they can only offer pulsed, rather than continuous, thrust [22]. The fact that the electrodes are self-triggering or, charged until surface breakdown occurs resulting in a discharge and ablated material acceleration, leads to a large shot-to-shot variation in thruster performance [22]. However, with very small impulse bit and higher pulse frequency, the thrust produced approximates a "continuous" thrust. The pulse frequency must be high since small perturbations will have a larger effect on small spacecraft such as a CubeSat than on a larger spacecraft (>100 kg for example). Thorough analysis performed by the University of Washington (UW) on μPPT options for the Dawgstar spacecraft proved their feasibility on nanosatellites (discussed further in Section 2.3.2.4) [20]. With a total mass of 3.80 kg, the μPPT considered for the Dawgstar spacecraft is much too massive for use on a CubeSat. Remaining design challenges specific to CubeSats are a reduction in overall mass, miniaturization of the onboard electronics and component scaling.

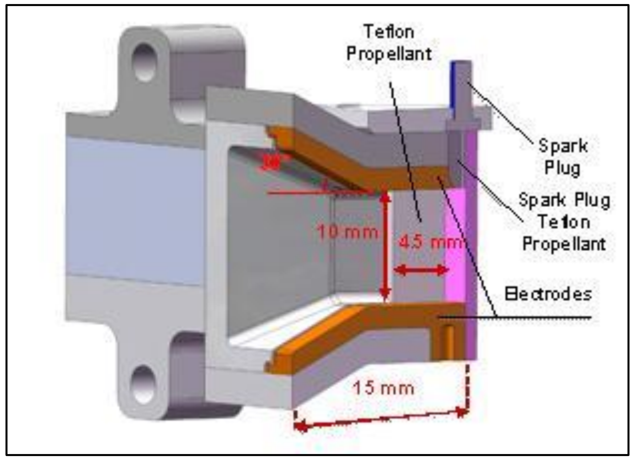

Figure 8 – Micro PPT CAD drawing [13]

2.3.2 Vacuum Arc Thrusters (VAT)

The Vacuum Arc Thruster is another type of ablative plasma thruster similar to a PPT, but one that uses thin, metal, film coated anode-cathode insulator surfaces as electrodes rather than conductive rods or advancing solid fuel. At a relatively low voltage (≈200V) the coated metal electrodes will break down, with a typical resistance of ~100Ω. The VAT uses a unique inductive energy storage (IES) circuit PPU to manage power and control inductor discharge [17]. An electric field is established when an inductor is discharged and current allowed to flow from anode to cathode. Plasma is generated by high electric field breakdown and expands into the vacuum between electrodes. The expansion of the plasma provides a path for current flow and is accelerated by the induced electric field between the two metallic electrodes [17]. A micro vacuum arc thruster (µVAT) was developed by Alameda Applied Space Sciences Corporation (San Leandro, CA) for use on board the Illinois Observing NanoSatellite (ION).

The ION spacecraft is a 2U CubeSat and the µVAT was designed to provide attitude control. The µVAT utilized the aluminum frame of the CubeSat as solid fuel to be consumed during thruster firings. Theoretical calculations performed by the ION team showed that 4 Watts -of power would produce approximately 54 µN of thrust, which enabled a 90 degree rotation in roughly 10 minutes [3]. Figure 9 shows a CAD model of the vacuum arc thruster designed for the ION spacecraft, dimensions of which were not provided.

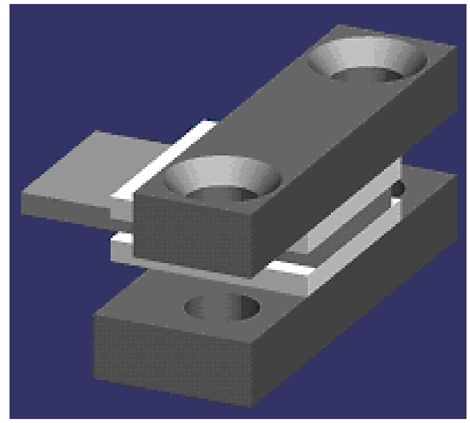

Figure 9 – Micro Vacuum Arc Thruster used on ION

Cathode (dark gray), Insulator (white), and Anode (light gray) [3]

2.3.3 Resistojets

Resistojets are conceptually the simplest of all electric propulsion systems, utilizing an electric heater to increase the temperature of the propellant to add extra energy, resulting in a higher exit velocity. This higher exit velocity (i.e. higher specific impulse) results in a higher thrust for the same propellant mass flow rate which can be a key feature when working with a strict mass budget. Reference 7 describes the design of a 2U CubeSat called RAMPART, presented at the 24th Annual AIAA/USU Conference on Small Satellites, whose flight date has yet to be established. RAMPART featured a resistojet propulsion system manufactured using Micro-ElectroMechanical System (MEMS) technologies, limited to a 1U section of the RAMPART [7]. The design also used rapid prototyping of components to allow them to conform to the exceedingly small volume constraints associated with a 1U CubeSat. The Free Molecule Micro-Resistojet (FMMR) was developed for attitude control of nanosatellites and microsatellites using water propellant and an integrated heater chip. The FMMR generates thrust by expelling water vapor from the plenum tank through a series of expansion slots located in the heater chip. The FMMR offers a specific impulse of 79.2 seconds with a thrust of 129 μN at a wall temperature of 580 K [21]. The dimensions of the theoretical satellite used in the analysis are 14.50 cm in diameter and 24.92 cm in height with an approximate mass of 10kg. The size of the theoretical satellite is comparable to CubeSats and with some component miniaturization the FMMR could be a viable option for CubeSats. However, the heater chip requires MEMS manufacturing technology.

2.3.4 Liquefied Gas Thrusters

Liquefied gas thrusters utilize the high vapor pressure of propellants such as butane or alcohol, which can be stored as a liquid, then upon expansion, phase transfer into a gas. This allows the propellant to be stored at a much lower pressure compared to a pressurized gas such as nitrogen. The main advantage however, is the higher density of a liquid versus a gas allowing much more propellant to be stored in a given volume. Liquefied gas thrusters generally consist of a liquid propellant tank and an adjacent plenum tank where the propellant vaporizes, allowing the vapor to travel to the valves followed by expulsion through exit nozzles [1].

Figure 10 – VACCO MiPS design for a CubeSat [4]

Recently, VACCO Industries developed a Micro Propulsion System (MiPS) designed specifically for use on CubeSats using their patented ChEMS™ (Chemically Etched Microsystems) technology, shown in Figure 10 [4]. The entire system has a mass of 509 g with a dry mass of 456 g and maximum propellant mass of 53 g of liquid isobutene (C_4H_{10}), and is roughly a 91 mm square. The MiPS is capable of 25 to 55 mN of thrust at 20°C, a total ΔV of 34 m/s and a specific impulse of approximately 65 sec [4]. The MiPS has a single axial primary thruster (E) and four tangential auxiliary thrusters (A-D). The performance characteristics of the MiPS is summarized in Table 2 below. It is important to note that mass ratios were not provided for the delta Vs listed in Table 2.

Thrust* [mN]	55	ΔV [m/s]	
Total Impulse [N·sec]	34	Total	34
Specific Impulse [sec]	65	+Z-Direction	26
Impulse Bit [mN·sec]	0.25	Pitch/Yaw	3
Pulse [msec]	10	Roll	4

Table 2 – Performance characteristics of MiPS [4]

*Thrust calculated with 40 psia plenum pressure

The VACCO micro propulsion system is ideal for use on CubeSats because of the integrated solid state valve system, the extremely compact design of the propellant and plenum tanks, and its ability to serve as a heat exchanger, for CubeSat thermal control. In this case, the required heat of vaporization is supplied by heat produced by components within the CubeSat, such as power dissipating circuit boards. In addition, the MiPS can function as a component of the structure, comprising one side of the CubeSat. The MiPS also conforms to all of the design specifications for CubeSats outlined in CubeSat Specification Document, including the limitations on power and maximum pressure of any storage vessel.

2.3.5 Cold Gas Thrusters

Cold gas thrusters generally consist of a pressurized tank containing gaseous propellant, such as nitrogen, and a solenoid actuated valve system leading to exit nozzles. Since the propellant is unheated and relies solely on the enthalpy of the stored gas, the velocity at the nozzle exit is relatively low resulting in a low specific impulse, typically around 60 sec, useful for small attitude adjustments and low ΔV maneuvers [14]. Other more advanced cold gas systems use a propellant tank, typically kept at a very high pressure relative to the desired pressure at the solenoid valve leading directly to the nozzle, and a smaller, intermediate tank to contain a limited amount of propellant for multiple thruster firings at a much lower pressure than the propellant tank pressure. Even with the secondary pressure reducing tanks, conventional valve designs are too massive or consume too much power for application onboard a CubeSat [1]. A cold gas system studied for the Dawgstar Spacecraft program at the University of Washington (UW) featured a miniature cold gas thruster, latch valve and pressure regulator, which had already been developed for the Pluto Fast Flyby Mission. The Dawgstar Spacecraft was a nanosatellite (~15kg) with a hexagonal prism design [20]. The miniaturized cold gas thruster was the Moog 58E135, developed by Moog Space Products (East Aurora, New York) in collaboration with Jet Propulsion Laboratory (JPL) [15]. Experiments performed at JPL measured the thrust of the Moog 58E135 to be 4.5 mN and minimum impulse bit of 100 μ-s [16]. Table 3 was taken from the analysis performed by UW on the performance characteristics of a μPPT and the Moog 58E135 thruster.

Propulsion System Type	Total Mass [kg]	Specific Impulse [sec]	Impulse Bit [μN·sec]	Thrust [mN]	Propellant Mass per ΔV [g·sec/m]	ΔV Time Duration [sec^2/m]	Energy per ΔV [J·sec/m]	Peak Power [W]
μPPT †	3.80	500	70	0.14	2	$1.43 \cdot 10^5$	$17.9 \cdot 10^6$	12.5
Cold Gas	4.58	65	100	4.5	16	$2.22 \cdot 10^3$	$1 \sim 5 \cdot 10^4$ ‡	10.1

Table 3 – Comparison of μPPT and cold gas propulsion systems (single thruster performance) [20]

† The performance of the μPPT was analyzed assuming a 1 Hz firing frequency.

‡ The energy per ΔV requirement for a cold-gas thruster depends on the firing mode, pulsed or continuous.

The μPPT was ultimately chosen due to concerns of propellant leakage and overall mass of the cold gas option. However, the team noted that both the μPPT and cold gas propulsion systems were feasible for the Dawgstar. With a total mass of 4.58 kg, the cold gas system considered for the Dawgstar is far too massive to be used on a 3U CubeSat whose maximum mass cannot exceed 4 kg.

The CubeSat Specifications Document limits an internal pressure vessel to 1.2 atmospheres (0.12159 MPa) [2]. This is an extremely low pressure for a cold gas thruster and makes pressurized gas systems much less attractive options for CubeSats. Waivers can be granted to exceed the 1.2 atm limit, which would be necessary for a cold gas system with realistic performance characteristics.

A cold gas propulsion system with miniaturized components would be the simplest system to implement into a CubeSat. A summary of the performance characteristics for the propulsion systems considered in this literature review is shown in Table 4.

Propulsion System Type	μPPT	VAT	Resistojet	Liquefied Gas Thruster	Cold Gas Thruster
Specific Impulse [sec]	500	>1000	79.2	65	65
Thrust [mN]	0.14	0.054	0.129	55	4.5
Total Mass [kg]	3.80 (including PPU)	<0.20 (including PPU)	n/a	0.509 (system)	4.580 (system)
Classification	Electromagnetic	Electromagnetic	Electrothermal	Chemical	Chemical

Table 4 – Summary of performance characteristics for propulsion options applicable to CubeSats

2.4 Mechanical and Structural Subsystem

The CubeSat program initiated at Cal Poly and Stanford University has been ongoing since the year 2000. During this time, over 40 universities, high schools, and private firms have participated in the program to create many different satellite designs [2]. From analyzing different trends in the design of the CubeSat structure, it can be determined which types of designs are best suited to meet various needs such as low price, low mass, simplicity of machining, and ability to support deployable components. With the knowledge of these trends, a new CubeSat can be designed with similar characteristics to suit the specific needs of a particular mission. Each of the characteristic listed in Table 5 were investigated in the review of previous CubeSat designs then compared in order to determine any design trends.

Design Style	There are a few distinct ways that the primary structure can be built. It can be machined out of a single block of aluminum so that the primary structure is one solid piece, or it can be assembled from multiple panels and components.
Structural Materials	The primary structure is limited to two aluminum alloys, but it can be determined if one of the two alloys is preferable over the other or if past CubeSat developers frequently apply for a waiver to deviate from the material specifications.
Structural Mass Fraction	There is a high variance in the structural mass of past CubeSats which reflects that various structural designs and configurations are possible.
Assembly Techniques	Some assembly techniques may be preferable over others in the designs of past CubeSat structures such as the use of screws or epoxy to fasten plates together or to attach additional components to the primary structure.
Fabrication Techniques	Some fabrication techniques such as computer numerical controlled (CNC) in which the machining is controlled by computers, are more beneficial than others in the machining of complex shapes, minimizing internal stresses during fabrication, and minimizing material loss.

Table 5 – CubeSat Structural Design Trend Categories

2.4.1 Mass Produced CubeSat Structures

Satellite developers can purchase prefabricated CubeSat structures and various components from companies that specialize in standardized CubeSat structure manufacturing. Two of the companies that provide CubeSat structures are Pumpkin Incorporated (San Francisco, CA) and Innovative Solutions in Space (ISIS), (Delft, Netherlands). Both companies sell sets of CubeSat structural components for different size satellites, which must be assembled by the developer.

Pumpkin Incorporated offers the CubeSat Kit to developers which contains the entire structure and all components necessary to allow the satellite "to be developed in as short time as possible and at low cost" [9]. The CubeSat Kit design is in its fourth generation, and has been delivered to more than 150 customers since 2003.It is claimed to be "the defacto standard in the CubeSat universe" [9]. The primary structure consists of six panels of 5052-H32 sheet aluminum fastened together with ten M3x5mm non-magnetic stainless steel flathead screws. The cover plates on the outside surface are made from approximately 1.5 mm thick sheets of 5052-H32. No deviation waver needs to be submitted for using Al 5052-H32 since the CubeSat Kit design is already preapproved. All other components are made from aluminum 6061-T6. The panels are designed to be compatible with a wide variety of subsystem components and payloads. The approximate mass of the primary 1U CubeSat structure is 241 g, which would yield a structural mass fraction of 0.18 if the total CubeSat mass is at a maximum. The cost of a 1U CubeSat structure from CubeSat Kit is about $1725 (US dollars). A model of the skeleton structure of a 3U CubeSat is shown in Figure 10Figure .

Figure 10 – CubeSat structure provided by the CubeSat Kit (left) [9] for a 3U model and by ISIS [7] for a 2U model

ISIS "a company which specializes in miniaturization of satellite systems with a particular emphasis on the design and development of subsystems for micro- and nanosatellites", offers CubeSat structures "as a generic primary satellite structure based on the CubeSat standard" [7]. The design of the ISIS CubeSat structure is more basic than the CubeSat Kit in that it consists of two modular side frames connected with four ribs for a 1U model assembled with M2.5x6 screws. The ISIS CubeSat structure also consists of a secondary structure, which incorporates a circuit board stack to enhance the structural integrity of the satellite. The primary structural mass of a 1U model is estimated to be 100 g and the estimated combined mass of primary and secondary structures is 200 g. The cost of the combined primary and secondary structures for a 1U CubeSat from ISIS is $3200 (US dollars). A 2U model CubeSat with both primary and secondary structures is shown in Figure 10.

2.4.2 Custom-Designed CubeSat Structures

A large number of CubeSats have been independently or custom-designed and built at different universities and organizations encompassing a variety of designs. A small selection of these CubeSats was reviewed in order to identify any specific trends in the satellite design. These independent designs differ significantly from those provided by ISIS and Pumpkin Inc. due to the limited budgets and manufacturing capabilities of the organizations.

The Stensat Group CubeSat was one of the original satellites designed for the first collaborative set of CubeSat missions by a team of engineers and amateur radio operators [12]. The initial goal in the design of this CubeSat was to keep the recurring cost of future CubeSats below $1000, to use standard commercial components, and to keep the design simple. The primary structure consisted of a snap fit and screw assembly of two types of 0.125-inch thick aluminum panels. "The center area was machined out to allow for mounting of a solar panel and magnetorquer coil" [12]. The inner surfaces of the panels were machined so that circuit boards could be snugly mounted.

Another satellite that was part of the first CubeSat mission was designed by students and faculty of Dartmouth College [15]. This design consisted of an assembly of four posts connected together with thin sheets of aluminum. Instead of using screws, this structure was assembled using epoxy. This CubeSat was designed so that the circuit boards also contribute to the structural strength.

California Polytechnic Institute at San Luis Obispo designed a prototype CubeSat in order to validate the tight constraints for picosatellites and to ensure proper integration with the P-POD deployment vehicle. This CubeSat was not launched and was purely a proof of concept design [15]. The design consisted of six individual panels of Aluminum 7075-T6and was strong enough to endure typical launch loads. The total structural mass of this prototype design was approximately 0.2 kg, or a mass fraction of at least 0.15.

The AAU-CubeSat was a satellite designed as a project by the students of Aalborg University (Aalborg, Denmark) [31]. One of the important design goals was to keep the structure as simple as possible. The primary structure consisted of a “frame cut from one piece of aluminum 7075-T6 and side panels made of carbon fibers attached with Epo-Tek U300-2 epoxy in order to conserve mass” [31]. The electromagnetic coils for the three magnetorquer were also incorporated into the structural design in order to further save mass. The aluminum frame had a total mass of 123.8 g or a mass fraction of at least 0.09.

SwissCube was a joint CubeSat project undertaken by various laboratories and universities in Switzerland with the goal of providing a “dynamic and realistic learning environment” for students in the development of small satellites [36]. In designing the structure of the SwissCube, the overall objective was to keep the design as simple as possible while minimizing cost and maximizing usable interior space. The resulting primary structure consisted of a monoblock design machined out of a single block of aluminum using wire electrical discharge machining (EDM). This machining method uses a rapid series of repetitive electrical discharges so that complex and thin shapes can be cut without excess cutting tool pressure. With this method, the resulting primary structure had a mass of 95 g (mass fraction of 0.07), which makes SwissCube one of the lightest CubeSat satellite structures ever produced. Another structural concern addressed by SwissCube was the prevention of Lithium-ion polymer battery cell expansion, a process in which these batteries expand and lose performance in a vacuum. This effect was counteracted though the use of a rigid battery box milled from aluminum and the use of epoxy resin in the interface between the block and the battery.

The DTUSAT-1 was a CubeSat designed and built by students from the Technical University of Denmark [31]. The primary structure of the DTUSAT-1 consisted of a monolithic wire-frame cube milled from a solid block of aluminum. The secondary structure consisted of a monolithic semi-cube (a cube with four faces instead of six) constructed from four printed circuit boards soldered together creating a sturdy structure with high resonance frequencies which minimized the need for additional assembly within the satellite due to the simplicity of the design. The outside faces were cut from 1.5 mm thick aluminum and fastened to the primary structure with screws. The face which supports the payload was milled from 2mm thick aluminum to support the heavier load.

The Canadian Advanced Nanospace eXperiment 1 (CanX-1) was a CubeSat built by graduate students of the Space Flight Laboratory at the University of Toronto [21]. The primary structure consisted of both Aluminum 7075 and 6061, alloys which are the two materials permitted by the CubeSat specifications. The total mass of the frame, exterior surfaces and mounting hardware was 376 g resulting in a heavy structure with a first natural frequency of approximately 800 Hz, meaning that the satellite had a very rigid design. Stress analysis of the structure with 12 g test loads revealed a 30 % margin on the maximum allowable stress in the satellite.

The CUTE-1 CubeSat was a satellite developed by students from the Tokyo Institute of Technology [34]. The primary structure consisted of four aluminum pillars and walls made from both circuit board stacks and individual circuit boards mounted against the interior walls to improve the structural integrity. Some of the secondary structural components such as fastening brackets for individual hardware components were actually made from magnesium alloys in order to minimize structural mass.

2.4.3 Summary of Structural Design Approaches

From analyzing each of the previous CubeSat projects, several design trends could be observed and then applied to selecting a CubeSat design as a starting point for the present work. The structural designs come in two flavors: models formed from a solid block of aluminum, and those assembled from multiple frames. There are pros and cons associated with each design approach. Solid body designs tend to be lighter and more rigid because they do not experience concentrated stresses due to fasteners during assembly. Forming thin shapes from solid blocks of aluminum, however, can leave residual internal stresses in the structure, which can be difficult to detect. Machining models in this manner may also be very difficult or even impossible depending on the available machining capabilities. Another drawback from forming shapes from a solid block of aluminum is that the material is not used efficiently, resulting in excessive waste of aluminum. This type of design would be ideal for a flight option CubeSat which would benefit from mass savings, assuming that it can be fabricated with available resources. A model assembled from multiple panels will typically be easier to machine and experience less residual stresses during assembly. This type of design is more suitable to a lab option that will be built and tested in the laboratory but not flown.

Another evident trend is that the primary structures of all past CubeSats were constructed of aluminum and did not use any exotic materials. Most CubeSat developers did not state specifically which aluminum alloy was used in the primary structure, so it can be assumed that either 7075 or 6061 alloys were used as specified by the CubeSat standards document [4]. However, the primary structure of the CubeSat Kit did use aluminum 5051, requiring a waiver had to be submitted in order to deviate from the official design specifications. Additionally, some satellites used materials such as carbon fiber composites and magnesium alloys as secondary structural support in order to save weight. The structural masses that are listed for each CubeSat vary in that some incorporate just the structural skeleton model, while some included the weight of external panel walls. The variance in structural mass is between 95 and 376 g (structural mass fraction between 0.07 and 28) depending on what parts are listed in the CubeSat structural mass. From these trends, it can be inferred what the proper materials for a WPI CubeSat should be, and that the structural mass fraction can vary depending on how the satellite is designed.

Most of the CubeSats investigated did not mention methods used in fabricating parts for the primary structure. One method that was mentioned is milling, which was used to form the monoblock design in the DTUSAT-1. Another more sophisticated method that was used to form the monoblock structure of the SwissCube CubeSat is the wire EDM method described earlier, which resulted in a very low structural mass. The most common assembly method consisted of using stainless steel screws to attach multiple parts of the CubeSats. The CubeSats that were of the solid monoblock design required less assembly than the multiple panel models. A few designs however, used epoxy adhesives to assemble parts in order to minimize weight. Overall, it can be concluded that there is no one way to fabricate and assemble a CubeSat, so that the construction of the satellite can vary depending on available resources.

3 Methodology

3.1 Research

During A-Term, each subsystem team conducted research on past and current CubeSats to determine what technologies and approaches have been used for each subsystem and what components would be required. Once the science payload and orbit were specified by the project advisors, subsystem teams focused their research on the specific components required to accomplish the overall mission.

3.2 System Engineering Group (SEG)

The System Engineering Group (SEG) consisted of (at least) one representative from each subsystem, and created a forum to discuss the physical integration of all systems into the CubeSat platform, as well as to collect critical design data (such as power requirements or component dimensions) from each subsystem, and discuss common issues related to the interplay of the subsystems.

The Power and Structural subsystem teams made particular use of this forum, as they required a substantial amount of information from all other subsystems, and facilitated the "give and take" of finite resources onboard the satellite (in this case power, volume, and mass). The power subsystem team collected power allocation "requests" for the ideal amount of power needed by each subsystem, and facilitated the allocation of power to each based on mission needs and careful consideration of each electrical component. The Structural subsystem also used the SEG as a vehicle to collect size and mass data for all components, determine component location according to need, and mission priority.

The SEG also provided a forum for presenting new ideas and theories about the design and construction of the CubeSat to other students and project advisors to collect valuable input and suggestions that were implemented during the design phase.

3.3 Construction

Various CubeSat models were developed by members of the Mechanical Subsystem of WPI using computer-aided design (CAD) software for each of the basic sizes (1U, 2U, & 3U) in order to propose models to be machined and used in laboratory tests. Constructing models in CAD is beneficial because it allows for visualizing how the various components will fit inside the satellite. Various types of analysis (thermal, dynamic etc.) can be performed in CAD programs such as SolidWorks (Dassault Systèmes SolidWorks Corp., Velizy-Villacoublay, France). Four different types of models were developed in SolidWorks for each of the three sizes. Following the design and modeling phase, the next step was to implement computer-aided manufacturing (CAM) software such as ESPRIT (DP Technology Corp, Camarillo, California), in order to map out the machining process, which in turn was converted into programmable machine code to be used by the computer numerical controlled (CNC) machine tools in WPI's Washburn Shops.

The first design consists of a basic monoblock structure which is based on previous CubeSat models that were reviewed in the literature. This design was modeled in each of the three sizes before it was confirmed that the WPI CubeSat will be a 3U size in order to accommodate larger payloads and a propulsion subsystem. The material for the initial design was selected to be Aluminum 7075 because it is the stronger of the two specified materials in the CubeSat Design Specification document [4] and has previously been used in other CubeSat structures. Non-critical dimensions (those that are not constrained by CubeSat specifications) can be modified later in order to optimize mass while meeting loading requirements. The construction of this design does not use material efficiently, and exceeds the capabilities of WPI's machining capabilities, so it was not a feasible design for a lab option model. 2U and 3U CAD models of this design are shown in Figure 11.

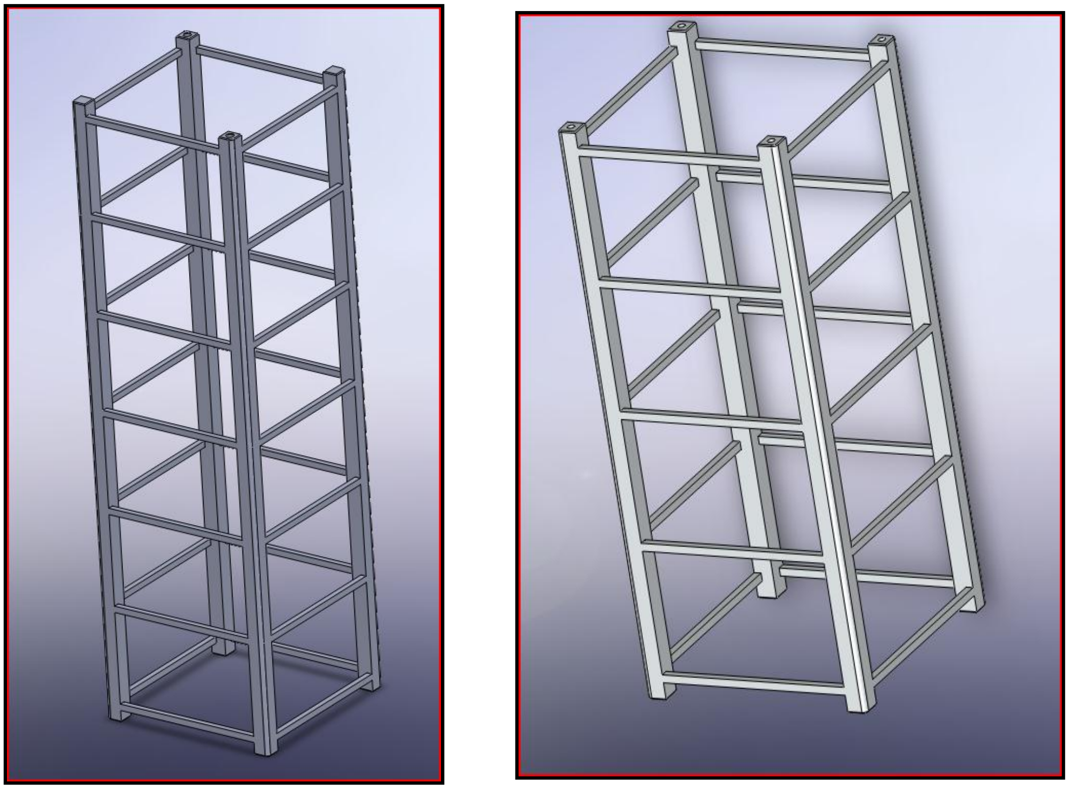

Figure 11 – Monoblock CAD models for the CubeSat for the 3U size (left) and 2U size (right)

The second iteration of the Lab Option or "Modular Design 1" is based on the ISIS CubeSat model which is an assembly of two railed panels with brackets that connect both panels together. The struts on the railed panels as well as the brackets are recessed inward by 1 mm so that the outside walls can be fastened to the CubeSat and be flush with the rails. The connecting brackets have additional threaded through holes which allow for assembly of various components into the satellite. In order to make the design more modular, each of the outside walls can be removed and redesigned with various features that can accommodate unique assembly needs of certain components. In contrast to a monoblock design which requires no assembly for the skeletal structure, this design will require sixteen screws to fasten the two-railed panels and eight connecting brackets together. This type of design uses material more efficiently and can be manufactured with WPI's machining capabilities. However, the vast amount of intricacies and sharp corners increases the number of stress concentrations throughout the structure and would take a large amount of time to fabricate with the available machining capabilities. A 3U CAD assembly model of Modular Design 1 is shown in both the exploded and collapsed configurations in Figure 12.

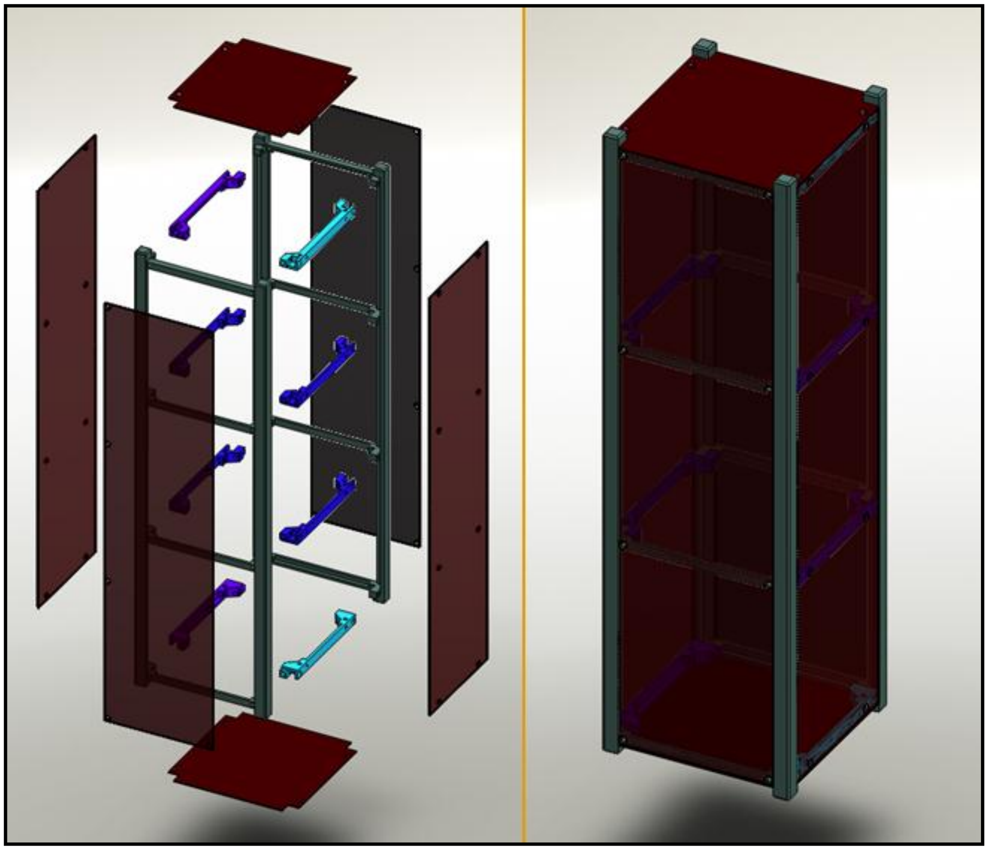

Figure 12 – Modular design 1 assembly CAD model for a 3U CubeSat in exploded (left) and collapsed configurations (right)

The third iteration of the Lab Option CubeSat is also a Modular Assembly design (Modular Design 2) similar to the previous, but has been simplified further, resulting in fewer stress concentrations making it quicker and easier to machine. This design consists of the two railed panels found in the previous design but without the protrusions that the brackets would snap into. Instead of using connecting brackets, this design includes two flanged walls that snap into the interior of the rails of the railed panels, which provide a snug fit. This design will also be modular in that it will allow for various designs of the outside panels, which can accommodate unique attachment requirements of subsystem hardware. This design, which is much simpler than the first modular design, still has a large number of intricacies and sharp corners, which would be very difficult to machine at WPI.

Exploded and collapsed configurations of the CAD assembly of Modular Design 2 are shown in Figure 13.

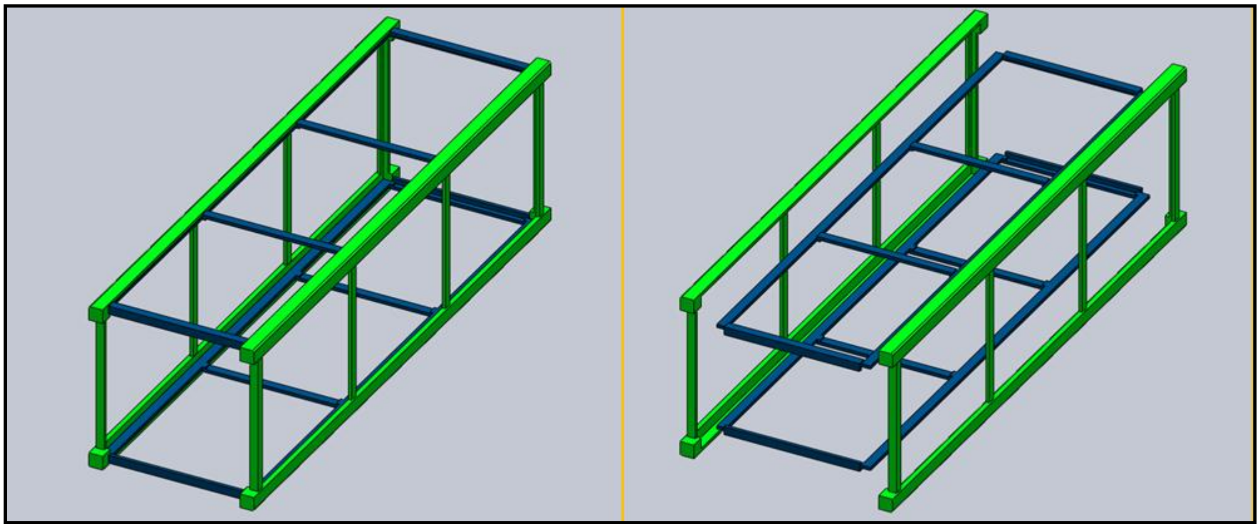

Figure 13 – Modular Design 2 CAD Assembly for a 3U CubeSat in exploded (left) and collapsed configurations (right)

The fourth and final iteration of the lab option structure or "Modular Design 3" is very similar to the Modular Design 2 model shown above, except certain changes were made in order to further improve machinability. The primary change in the design is that the "L" profile in the rails and struts, located at the ends, which would save mass and increase volume, were changed to square profiles due to limitations in WPI's machining capabilities. In addition, instead of connecting the two types of panels with flanged tabs, these were removed, so that so that a basic surface to surface mate (i.e. one using fasteners) of the two types of panels is used for assembly. Finally, 0.125-inch fillets were added to most of the internal corners so that they can be machined with the mill bits available. With each of these changes, this design could be machined with WPI's machining capabilities. Each railed panel system was machined out of a single plate of aluminum. Furthermore, since there are only two types of parts, it was machined using only two CAM operations. This greatly reduced manufacturing time and will make machining easier. Exploded and collapsed configurations of the CAD assembly of Modular Design 3 are shown in Figure 14.

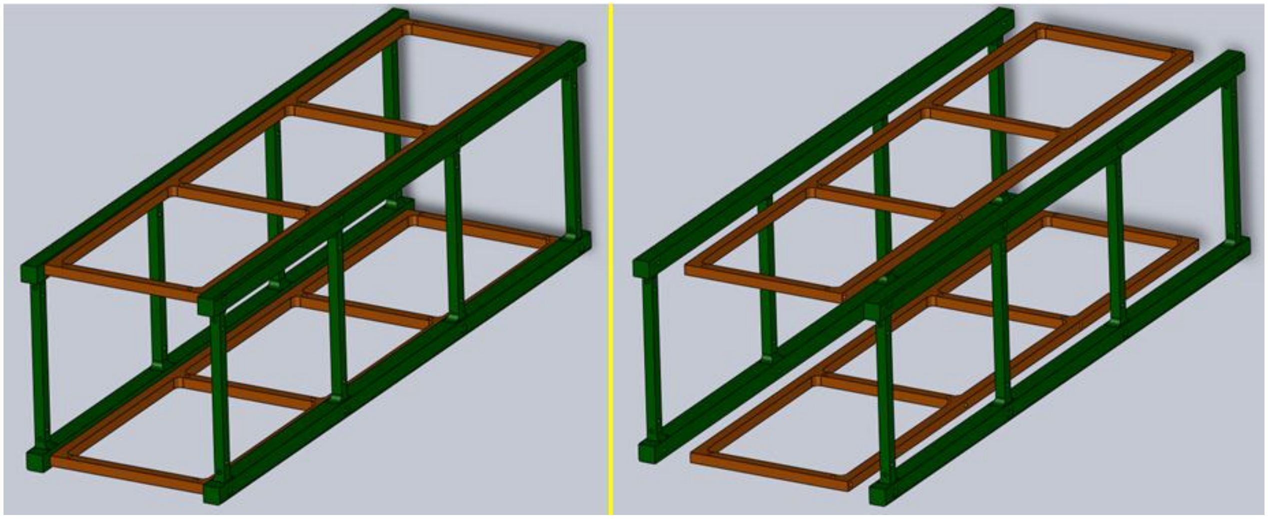

Figure 14 – 3nd Modular CAD Assembly for a 3U CubeSat in collapsed (left) and exploded configurations (right)

CAD assembly models have also been created which serve as schematics representing the placement of the individual hardware components within the CubeSat. The first iteration of the schematic assembly of "Lab Assembly 1" is uses modular design 1 to house the subsystem components. Exploded and collapsed configurations of Lab Assembly 1 are shown in Figure 23. This assembly highlights only one possible configuration of the components in the CubeSat, the final placement of components in the Lab Option may vary based on center of mass, passive thermal heating, and electromagnetic interference considerations. In this model, the modularity of the design is taken advantage of in fastening the propulsion system to the satellite. As shown in Figure 15, the outside wall, to which the propellant tank is attached, has two struts which have been added to ensure reinforcement for a rigid attachment. After examining this assembly, it was apparent that based on the current selection of components, there is additional space in the satellite that could accommodate an additional payload or a larger propulsion subsystem.

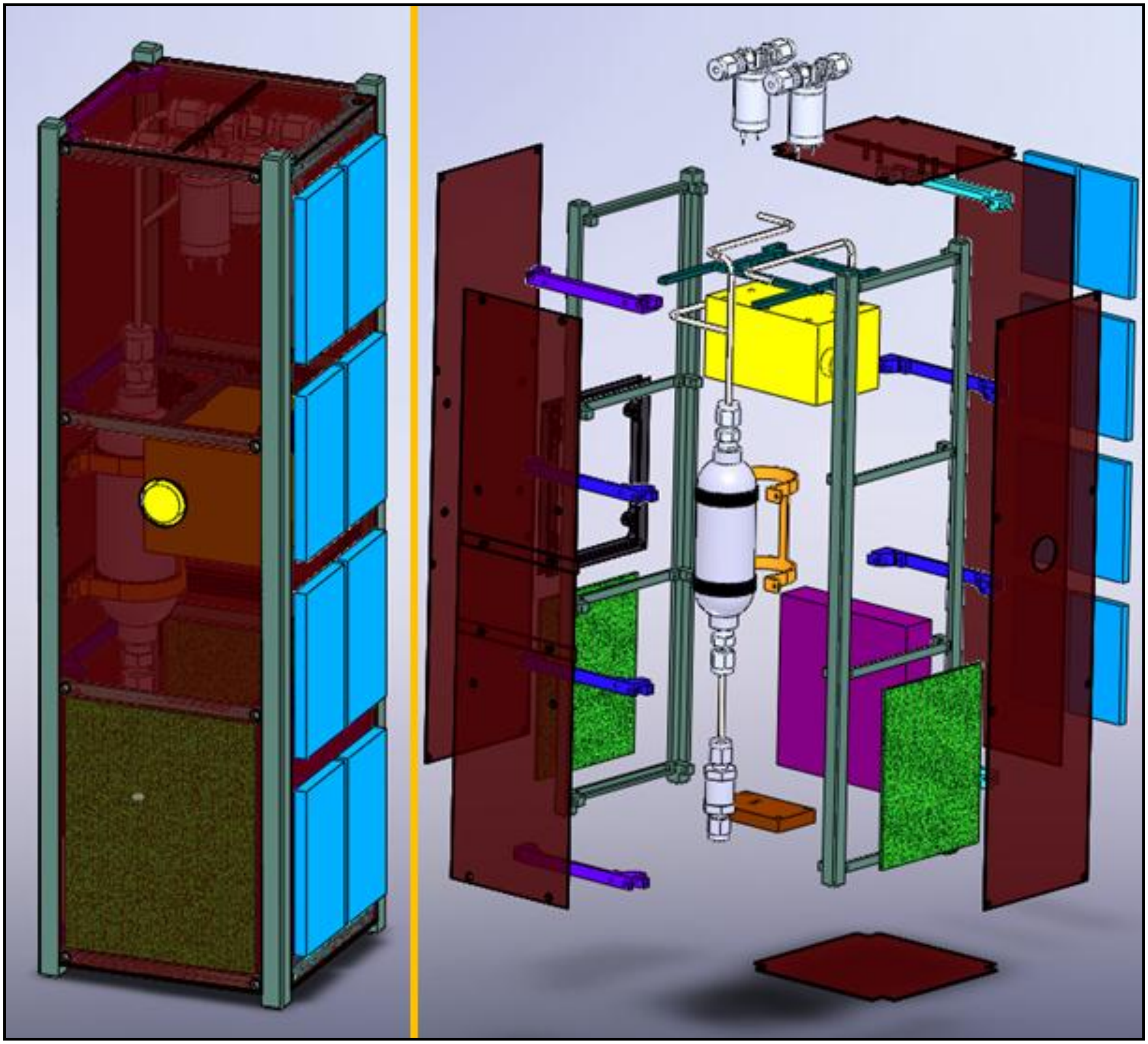

Figure 15 – CAD Assembly model of Lab Assembly 1 in exploded (left) and collapsed configurations (right)

The second and final configuration assembly (Lab Assembly 2) uses the structure of Modular Design 3 and will be used as a guide for assembling components into the machined structure. The exterior walls in this assembly are slightly thicker (1.5 mm) than in Lab Assembly 1, which is due to only certain aluminum panel thicknesses being readily available. This will make the Lab Option CubeSat slightly heavier, but it will better be able to support the loading of components mounted directly to the walls. In Lab Assembly 2, more attention is given to the specifics of the arrangement of each of the components. Also in this model, the circuit boards for the battery, PMAD, and OBC were assembled into a circuit stack, which is sandwiched between two of the side walls allowing enough room for the check valve. One of the side walls of Lab Assembly 2does not include any mounting holes so that it can easily be removed for access to the inside of the CubeSat. Exploded and collapsed configurations of Lab Assembly 2 are shown in Figure 16.

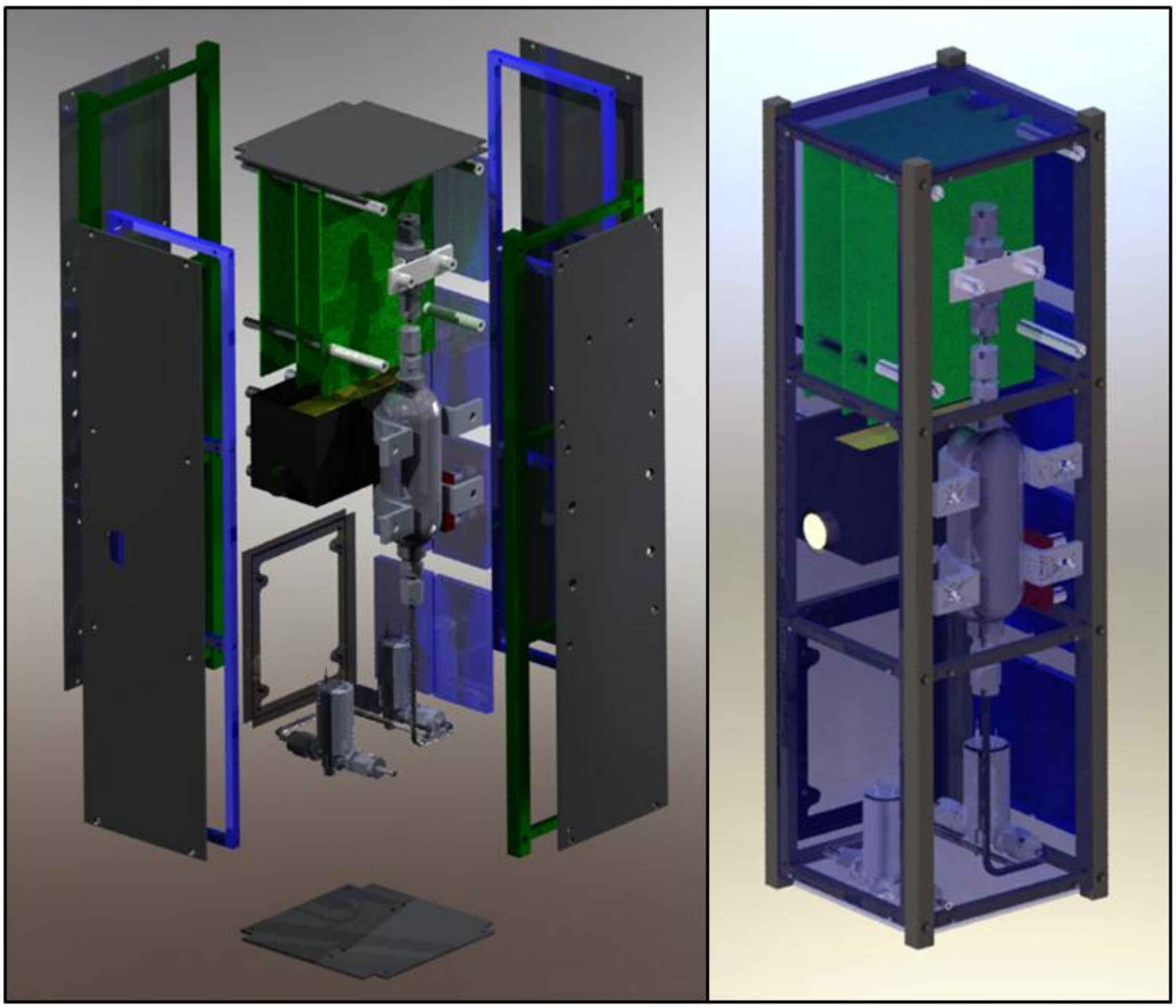

Figure 16 – CAD Assembly model of Lab Assembly 2 in exploded (left) and collapsed configurations (right)

Concerning the machining and assembly of the CubeSat, the CAD models of the CubeSat were input into the CAM software, ESPRIT, in order to create machine processes to machine the different parts using CNC machines. As for the Lab Option, Lab Assembly 2 was chosen to become the final lab design. The primary structure of this design is made up of two part types: the railed panel and the connecting panel. The part files for the final CAD models for Lab Assembly 2 were then inputted into ESPRIT in order to create machine processes. A total of three CAM operation files were created for each of the part types. The CAM operation files were then converted into numerical control (NC) code, which would be used by the Haas Vertical Machining Center Toolroom Mill (TM-1) shown in Figure 17 below.

Figure 17 – Haas Vertical Machining Center Toolroom Mill (TM-1) [51]

The TM-1 would then use the machine code to operate its tool bits and machine the part. Six tools have been identified to create the tool list for the machining of both panels: (1) 0.5-in End Mill, (2) 0.1875-in End Mill, (3) 3-in Face Mill, (4) #2 Center Drill, (5) 1.6-mm Drill - M2, and (6) 2.5-mm Drill - M3. Furthermore, before actual machining of the parts could begin, the raw Aluminum 6061-T6 material was prepared to the correct stock sizes using various band saws. An example of a pocketing operation can be seen in Figure 17, with each part taking 70-100 minutes machining time each.

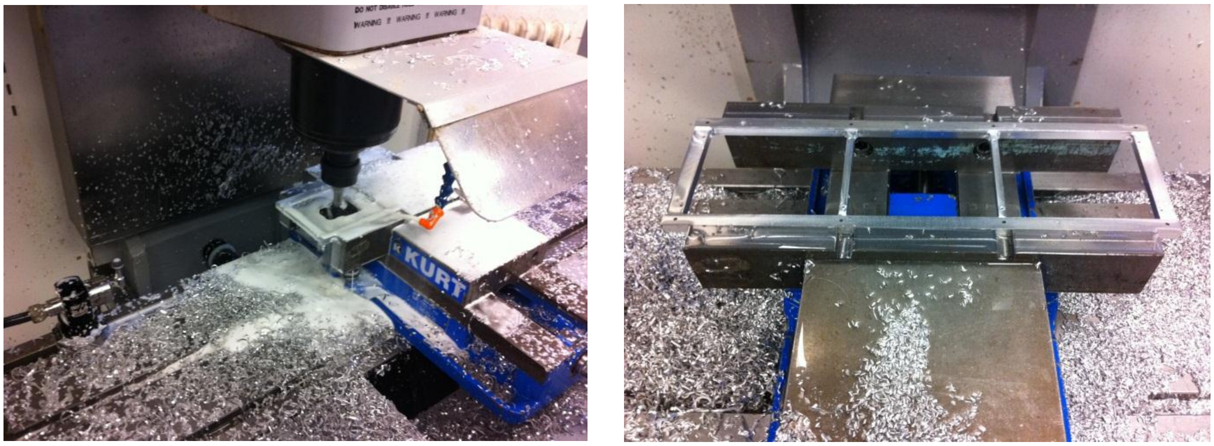

Figure 18 – Pocketing Operation [51]

Both railed and connecting panels have been machined completely, which can be seen in Figure 19. The use of the TM-1 allows for precision machining with very low tolerances needed for spacecraft construction.

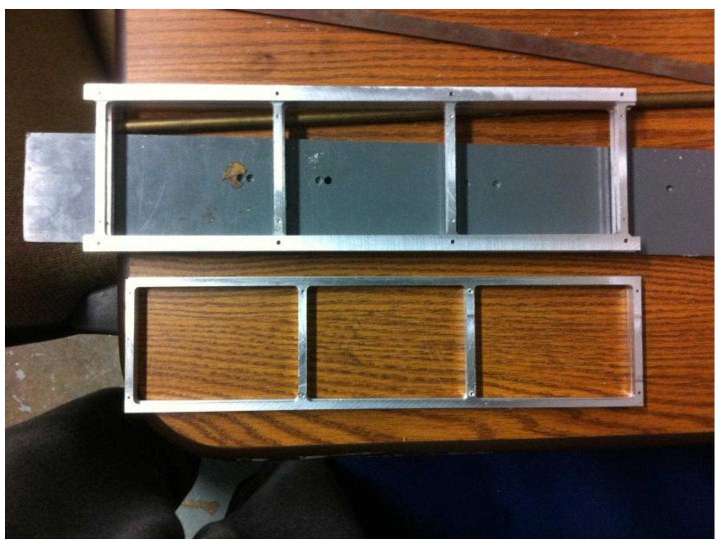

Figure 19 – Completed Railed & Connecting Panels [51]

3.4 Lab Option vs. Flight Option

This paper presents two different design considerations: "Lab Option" and a "Flight Option" CubeSat designs. The most time and consideration in this particular MQP were dedicated to designing and building the Lab Option. This set of design choices, components, and analysis, are specifically intended to satisfy the requirements of the CubeSat payload, but are not space flight qualified. These Flight Option components can be assembled and tested in a space environment simulator, but are not qualified for spaceflight. Some components of the Lab Option were ordered and constructed.

The Flight Option design is intended for operational space flight. The Flight Option component selections detailed in Chapter 5 will fully support the payload on orbit within the mission parameters detailed in Section 4.1. Flight Option components were not purchased during this project.

4 Lab Option Component Selection and Analysis

The components specified below (for both flight option and lab option) were selected based on specifications drawn from the mission requirements, payload specifications, and CalPoly CubeSat regulations. All components for the Lab Option were designed to fit within the physical dimensions of a standard CubeSat, to be consistent with mission criteria, and to be within the budget of this WPI project. Although the Lab Option components are not certified for space flight, they are based on the same specifications which would be used for a CubeSat designed to be flown.

4.1 Spacecraft and Payload Requirements

The mission requirements presented here are based primarily on the CalPoly specifications for a CubeSat designed to be deployed from their P-POD. Although the payload for the WPI CubeSat is still not finalized, the assumptions used to generate payload specifications are detailed below.

4.1.1 Orbit Specifications

The mission and payload for the project were specified by the project advisors and selected to represent a realistic set of mission requirements. As specified in the project requirements document [4] the CubeSat will follow a circular orbit at an altitude of 680km and a period of 98.2 minutes, where the argument of latitude u is defined as

Element	Value
Semimajor axis a (km)	7051
Eccentricity e	0.0
Inclination i (deg)	98.0
RAAN Ω (d g)	0.0
Argument of Latitude u (deg)	0 0

Table 6 – Orbital Characteristics

$$u \cong \omega + v \qquad \textbf{Eq 4.1}$$

Table 6 details the orbit characteristics.

4.1.2 Scientific Payload

The scientific payload is the Argus 1000 IR Spectrometer, an infrared spectrometer used to investigate greenhouse gases in the atmosphere [10]. Table 7 lists the technical specifications of the Argus 1000 IR Spectrometer:

Argus 1000 Specifications	
Type	Grating spectrometer
Configuration	Single aperture spectrometer
Field of View	0.15° viewing angle around centered camera bore-sight with 15mm fore-optics
Mass	230g
Dimensions	50mm x 45mm x 80mm
Operating Temperature	-20°C to +40°C
Survival Temperature	-25°C to +55°C
Detector	256 element InGaAs diode array with Peltier cooler (customized options available)
Optics	Gold with IR glass and coatings
Electronics	Microprocessor controlled 10 bit ADC with co-adding to 13 bit, 3.6-4.2V input rail 250mA-600mA (375 mA typical)
Operational Modes	-Continuous Cycle, constant integration time -Continuous cycle, adaptive exposure
Data Delivery	Fixed length parity striped packets of single or co-added spectra with sequence number, temperature, array temperature and operating parameters
Interface	Prime and redundant serial interfaces (RS232 protocol)
Integration Time	500 μs to 4 s
Calibration	Two-wavelength laser calibration
Handling	Shipped by courier in ruggedized carrying case

Table 7 – Argus 1000 IR Spectrometer Specifications [52]

4.2 Power Component Selection and Analysis

The project's scientific payload, the Argus 1000 IR Spectrometer, requires the power subsystem to provide a continuous feed of 572mA (375mA typical) at 3.5-5.0V. The Power Management and Distribution (PMAD) electronics will need to stabilize any current spikes within 10ms of detection, and the power feed to the spectrometer must be switched on or off as commanded by the computer based on the operations schedule. In addition, the power system is expected to meet the power requirements of other subsystems including propulsion, onboard computing, attitude control, and other sensors.

4.2.1 Solar Cells

As described in Section 2.2.1, solar cells will be the primary power source for the CubeSat, and will be used to charge the battery for use during eclipse or in times of peak power demand. Several factors influence the total power output of the solar cells: cell placement (fixed on body vs. deployable array), cell orientation relative to the sun, solar cell area, and any protective coatings on the cells.

The power density available from a solar cell will depend on the illumination (solar constant) and the cell efficiency as shown in Eq 4.2.

$$P_{01} = 1367\,\eta\,(\frac{W}{m^2}) \qquad \textbf{Eq 4.2}$$

where efficiency is defined as the percentage of the total energy absorbed by the solar cell that is converted to electrical power (i.e. 15% for a low-cost "hobby-shop" solar array). Alternatively, if the mean voltage V of the solar cell is multiplied by mean current, I, then the electrical power produced is given by:

$$P_{02} = I\,V \qquad \textbf{Eq 4.3}$$

where current is expressed in Amps and voltage in Volts. The power at beginning of life (BOL) can then be determined, taking into account the inherent degradation I_d and the reduction in power output with increase in angle to the sun, θ (measured as the angle θ between a vector normal to the solar cell and a vector extending from the solar cell to the sun):

$$P_{BOL} = P_0\,I_d \cos\theta \qquad \textbf{Eq 4.4}$$

where I_d is assumed to be 0.77 (or 77%) efficiency as a nominal power loss due inherent inefficiencies in a solar array power system. Eq 4.4 provides the power output of a solar cell (just as in Eq 4.2 and Eq 4.3), but Eq 4.4 also accounts for the angle of the solar cell in relation to the sun as well as system inefficiencies.

The Power at Beginning of Life can also be used to determine the output power density per unit area

$$P_d = \frac{P_{BOL}}{A} \qquad \textbf{Eq 4.5}$$

Eq 4.2 through Eq 4.5 can be manipulated to determine the area of solar cells required to produce a given amount of power based on the efficiency, angle to the sun, brand of solar cell, and solar cell area [7].

A variety of solar cell options were considered for use on the Lab and Flight options, assuming the total cell area would occupy a 10 cm x 10 cm area (equivalent to one side of a 1U CubeSat). For the Lab Option, solar cells from SolarBotics (Calgary, Alberta, Canada) and Solar World (Hillsboro, OR) were considered. As shown in Table 8, these solar cells are affordable but they will not supply adequate power for our satellite unless significant design changes are implemented (i.e. increasing the area of the solar cells either by creating deployable arrays or covering more of the satellite body with cells).

Brand	**SolarBotics**	**Solar World**
Dimensions	3.7cm x 6.6cm square cells	9.525cm x 6.35cm square cells
Price	\$7.15-\$11.00 per cell	\$7.95-\$9.95 per cell
Voltage per Area	6.7V at 1.2285mA/cm^2	0.5V at 13.2267mA/cm^2
Peak Power Output	0.6191W at 97.68cm^2	0.31W at 60.48cm^2

Table 8 – Lab Option Solar Cell Comparison [53] and [54]

4.2.2 Batteries

Batteries will be used to provide the CubeSat's energy storage for the duration of the mission. Due to the maximum practical mission length for a CubeSat (shorter than 3 years), the battery will only provide back-up power for periods of eclipse and peak power demand.

Typical CubeSat batteries provide either 3.3V or 5.0V. The scientific payload will require a 5.0V battery while the size of the satellite will limit the design to one secondary (rechargeable) battery. When choosing components for the Lab and Flight Options, battery storage capacity and total mission length were the highest weighted figures-of-merit to ensure that the flight battery is designed to endure the number of charge/discharge cycles required by the satellite while still providing adequate power.

Although time did not permit actual purchase and testing of a lab option battery, it was determined that a 5.0V (approximately 1500 mA-hr) rechargeable Lithium Ion battery would be the best option for lab testing, as it can provide a stable and reliable power source and is easy to integrate with the selected battery charging circuitry.

4.2.3 Power Management and Distribution (PMAD)

An integrated PMAD module is produced by the Clyde Space Corporation specifically for use on CubeSats. Clyde Space designs custom PMAD systems to integrate with specific scientific instruments to be launched on CubeSats. This option is ideal for space flight because the Clyde Space PMAD systems are specifically designed for flight aboard a CubeSat. At over $2500, however, this option's cost is prohibitive for the present project's lab option.

In order to closely simulate the operations of the CubeSat in a lab environment, it was necessary to design and build a Power Management and Distribution system that could simulate all the functions of a flight-option PMAD. This system had to be able to produce power, condition power to the correct voltage and current for each device, and switch devices on and off based on commands from the flight computer.

When the Lab Option is eventually completed, power will be provided first by "lab option" solar cells illuminated by bulbs in a Space Environment Simulator (the large vacuum chamber in WPI's Higgins Labs for the purposes of this CubeSat). The DC power from the solar cells will be used to charge the lab battery and potentially to power individual components (although the cells do not provide enough power to support all systems at once). Supplementary power will be provided through an umbilical attached to a simple "lab bench" DC power supply connected in parallel with the batteries and solar cells.

Figure 20 shows a functional block diagram of the PMAD for this CubeSat project.

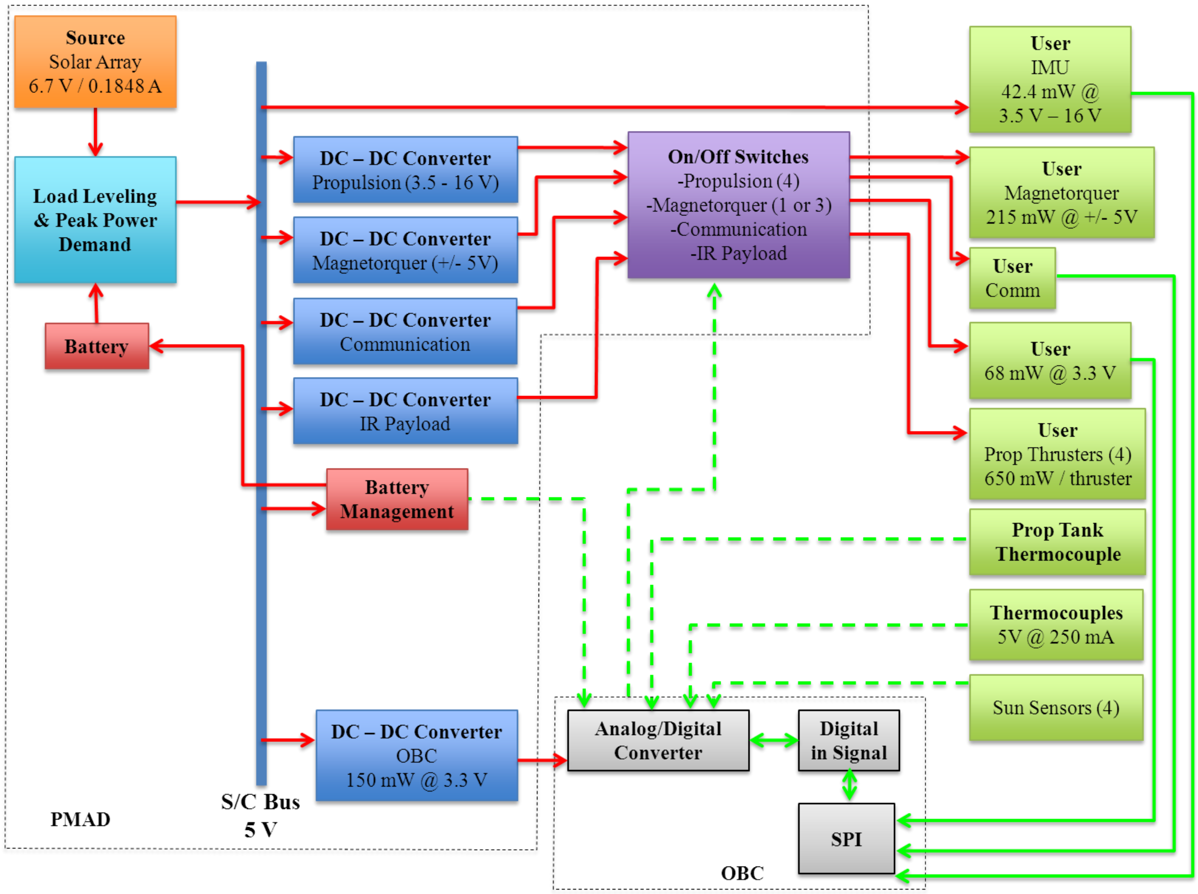

Figure 20 – PMAD Block Diagram

The power provided by the battery, solar cells, and umbilical, is received and conditioned by a series of circuits on IC chips for conditioning and conversion. The only basic functionality that will be carried out by Lab Option circuitry during the first phase of testing will be converting the voltage from the power source to fit the needs of each subsystem.

This power conditioning will be done with simple DC-DC converter circuits on IC chips. These chips will be connected directly to the power rail and component switches, and will modulate the voltage and current coming from the power source to the exact specifications of each power client. The PMAD will also provide battery charging and discharging capabilities. This will be done with a simple integrated circuit connected to the battery, power supply, and a timing (or "clock") chip.

Finally, the PMAD will provide switching capabilities to turn each individual component on and off. This capability will be particularly important in lab testing because the solar cells cannot produce enough power to run all subsystems simultaneously. Each individual subsystem will be switched on and off via a manual input (most likely via a laboratory desktop computer) into the PMAD for the first phase of testing.

Function	Component	Specifications	Notes
DC Conversion	LT1054CN8#PBF-ND	Boost/buck conversion, 3.5V-15V @ 100mA	DC Conversion and switching combined on one IC
Switching	LT1054CN8#PBF-ND	Simple high/low input signal (@ 5V) for on/off	
Battery Charging	LM3622MX-4.1-ND	Li-ION Battery, 24V max	Requires additional IC diode and timer

Table 9 – Lab Option Power Subsystem Components [55]

The parts listed in Table 1 were selected for compatibility with each other and to fulfill the power requirements of the other subsystems presented in this report. Although the parts were ordered and received, time did not allow for any significant testing. All parts were ordered from Digi-Key (Thief River Falls, MN).

4.3 Propulsion System Selection & Analysis

Upon completion of the literature review (Section 2.3) performed by the Propulsion Subsystem, in conjunction with the constraints set forth by the CubeSat Design Specifications document and WPI project scope, a cold gas propulsion system was the most feasible system to implement [22]. The propulsion system for Lab Option 1 is strictly a cold gas system. Two Lab Options were considered due to the restriction placed on pressure vessel by the CubeSat Specifications Document resulting in concerns of propellant storage limitations. The propellant is air compressed to approximately552 MPa (80 PSI). Compressed air was chosen because of its ease of use, safety and cost. Compressed air was readily available in the on-campus laboratory where all propulsion system testing will take place. Since the compressed air was a stock item in the laboratory there was no impact on the project's monetary budget. When handling compressed air, neither protective equipment nor specialized equipment or materials are required. Since the compressed air propellant was stored at a moderate pressure (~0.689 MPa or ~100 PSI) and temperature (25°C or 77°F) high pressure fittings and components were not necessary for the onboard propulsion system or refueling system. The pressure inside the propellant tank is the maximum provided to the solenoid valves.

Figure 21 – SERIES 411 Miniature Solenoid valves from ASCO Scientific, manifold mount option (left) and standard option [56]

Miniature solenoid valves, SERIES 411 shown in Figure 21, manufactured by ASCO Valve (Florham Park, New Jersey), act as the electrically actuated thruster valve [18]. Such solenoid valves are typical on a cold gas thruster propulsion system, and the "nozzle" consists of a constant cross sectional area tube exiting the sidewall of the CubeSat. Due to the low pressure and temperature characteristic of this design, a significant boost in thrust (or specific impulse) is not anticipated from adding a nozzle to provide gas expansion upon exit.

An impulse bit represents the smallest possible change in momentum deliverable by the thruster, which is significant when maneuvering small spacecraft because of their inherent low mass moment of inertia. The miniature solenoid valves are available for use with low voltage over a wide range (5 VDC to 24 VDC) and require very low power to open, from the normally closed position, and hold open (0.65 W for a two way, normally closed valve) with a response time, or the minimum time possible between opening and closing the valve, of approximately 10 ms. The mass of one miniature solenoid valve is approximately 50 g, which does not contribute significantly to the mass budget.

The solenoid valves also have a manifold mount option which will increase the volume consumption of the entire system but could also act as a component of the overall structure. The manifold mount option was not chosen because of overall component configuration constraints. The Series 411 have a relatively small orifice size (approximately

0.0125 in), which should be satisfactory for this application. In the Lab Option 1, propellant will be filtered prior to entering the propulsion system, therefore filters were not necessary onboard the CubeSat.

A thermocouple will be located on the propellant tank to monitor fuel temperature, which will be used to determine fuel pressure through calculation performed by the On Board Computer (OBC), see Figure 22. The pressure of the fuel will be used by the OBC in an algorithm to determine the "burn time" of the thrusters for a given maneuver. The surface mounted thermocouple is not ideal and inaccuracy in the actual temperature of the fuel is expected and will be compensated for in analysis and algorithms.

The propellant tank is refillable using a check valve to regulate the flow direction and seal one end of the propellant tank. A SolidWorks model of the Lab Option 1 propulsion system is shown in Figure 22. The propellant tank is attached to the sidewall with custom bracketing to minimize the propellant tank and lines from experiencing excessive vibrations and also to prevent the lines from supporting all the weight of the tank. The check valve, located at the bottom of the figure, will also have a custom mounting bracket to minimize vibrations and to prevent any damage to the propellant lines while using a wrench to attach/remove fill lines. The filling process consists of removing the side wall at the end of the CubeSat and attaching a fill line (connected to a supply tank in the laboratory) to the check valve via a secure tube fitting connection. The propellant lines consist of 0.125 in stainless steel tubing and will connect to the propellant tank, solenoid valves and check valve with off-the-shelf tube fittings.

The Lab Option 2 propulsion system was essentially a hybrid of a liquefied gas thruster system and a cold gas thruster system. Lab Option 2 was not constructed but considered as a laboratory option due to financial (component and propellant cost), safety (propellant handling and storage) and time to manufacture constraints.

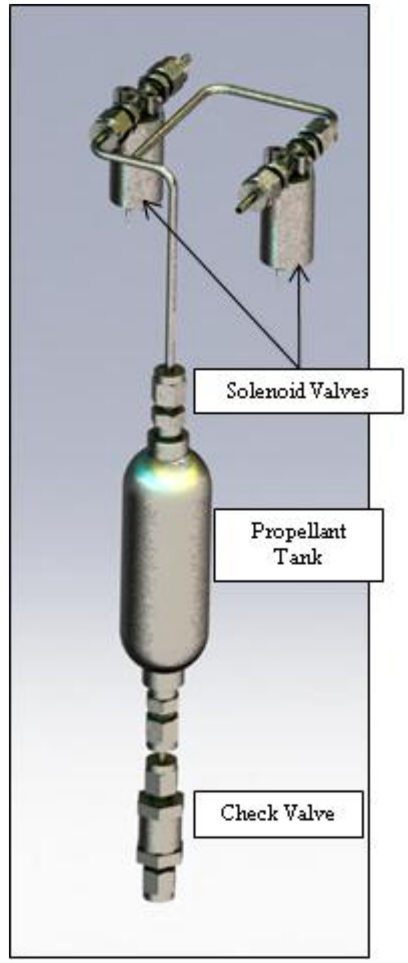

Figure 22 – Lab Option 1 SolidWorks model

Using liquid propellant such as butane or alcohol contained in a single tank would allow the liquid and vapor to reach equilibrium. The vapors would not be heated in any way and the vapor pressure inside the propellant tank would be the maximum pressure achievable by the system. Solenoid valves, identical to those considered for Lab Option 1, act as the thruster valves. A nozzle would not be implemented and the same fittings and tubing would be used. The thermocouple mounted to the propellant tank would provide the fuel properties for the OBC. The temperature of the fuel in this option is more critical because it will determine the vapor pressure, therefore an accurate fuel temperature is essential and more thermocouple mounting options may need to be considered, such as a probe inserted directly into the tank. Since the probe, either inserted into an end or through the sidewall of the propellant tank, would measure the temperature of the gas directly, rather than through the tank wall, it will provide a much more accurate gas temperature measurement. The propellant tank will only be storing gas at a maximum internal pressure of 1.2 atm (0.12159 MPa), as required by CubeSat Specifications Document [4]. The tank would have been manufactured on the WPI campus from Plexiglas or Lexan. Because the CubeSat will be mounted to a fixture limited to rotation about the vertical axis during testing, propellant displacement inside the tank should not be an issue.

4.3.1 Propulsion Analysis

The first step towards any analysis of the propulsion system capability involves calculating attainable values of ΔV, which can then be used to calculate possible orbital maneuvers. This was done using the *rocket equation*:

$$\Delta V = g\, I_{sp}\, ln\left(\frac{m_0}{m_f}\right) \qquad \textbf{Eq 4.6}$$

Where g is the gravitational acceleration constant at sea-level (9.81 m/s^2), I_{sp} is the specific impulse (approximately 60s), typical for cold gas thrusters, and m_0 and m_f are the initial and final masses of the CubeSat, respectively. Calculations were carried out for varying ratios of propellant mass to overall mass, with a realistic value close to 10% or even lower, the results of which can be seen in Table 10.

m_p/m_0	ΔV(m/s)
10%	62.02
25%	169.33
50%	407.99
75%	815.97

Table 10 – ΔV calculations for varying mass ratios.

4.3.2 Orbital Maneuvers

Having found the achievable ΔVs, these values can then be used to calculate orbital maneuvers which may be performed by the CubeSat. The two orbital maneuvers considered in this analysis were orbit raising and inclination change. Orbit raising would involve raising the initially circular orbit of the CubeSat to a higher circular orbit via a Hohmann Transfer. A Hohmann Transfer requires two engine firings, one to put the CubeSat on an elliptical "transfer" orbit, the second to re-circularize the CubeSat's orbit once it has reached the desired altitude. The equation for finding the necessary ΔV to perform a Hohmann Transfer in terms of altitude change (ΔA) and initial radius (r_0) is provided below [8], where μ_E is the standard gravitational parameter of Earth $\left(3.986 \cdot 10^{14}\, \frac{m^3}{s^2}\right)$

$$\Delta V = \sqrt{2\mu_E\left(\frac{1}{r_0} - \frac{1}{\Delta A + 2r_0}\right)} - \sqrt{\frac{\mu_E}{r_0}} + \sqrt{\frac{\mu_E}{\Delta A + r_0}} - \sqrt{2\mu_E\left(\frac{1}{\Delta A + r_0} - \frac{1}{\Delta A + 2r_0}\right)} \qquad \textbf{Eq 4.7}$$

The second orbital maneuver considered was an orbital inclination change. The ΔV required for an inclination change was found using Eq 4.8 below:

$$\Delta V = 2V \sin\left(\frac{i}{2}\right) \qquad \textbf{Eq 4.8}$$

$$V = \sqrt{\frac{\mu_E}{r}} \qquad \textbf{Eq 4.9}$$

where V is the initial velocity of the CubeSat in its circular orbit, found to be 7519m/s using Eq 4.9, where r is the radius of orbit (roughly 7050km, or an altitude of 700 km) and i is the inclination change. The changes in altitude and orbital inclination are plotted in Figure 23 as a function of propellant mass ratio.

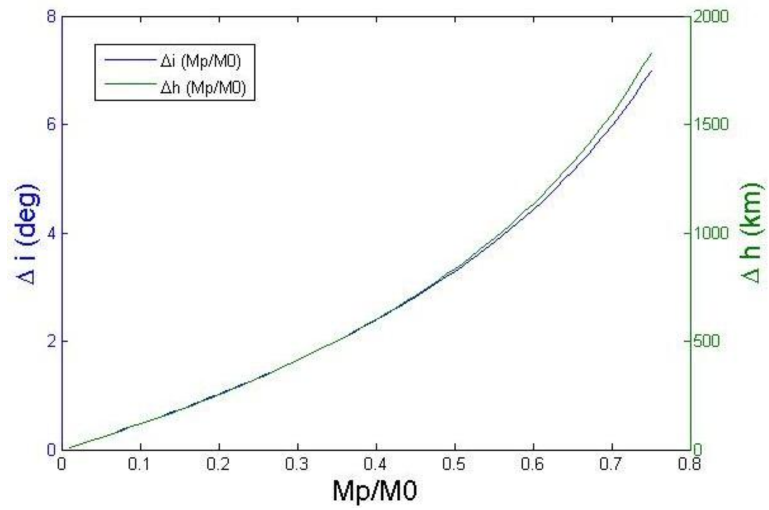

Figure 23 – Change in altitude or inclination for varying mass ratios

4.3.3 Propellant Volume

The next step in the analysis involved looking more in depth into the requirements set forth by the CubeSat program at California Polytechnic State University and described in the official requirements document [4], which states that no pressure vessel can exceed 1.2atm (0.121 MPa). Taking this into account, the Ideal Gas Law was used to relate different amounts of propellant mass to the volume needed to store that propellant at the maximum allowed pressure, seen in Figure 24. The propellant assumed in this analysis was nitrogen, a typical choice for cold gas thrusters, which has a specific gas constant of 287 J/Kg-K. The volumes required were then found at the minimum and maximum operating temperatures for the ASCO 411 Series valves used in the lab option design, 0°C and 60°C respectively [12].

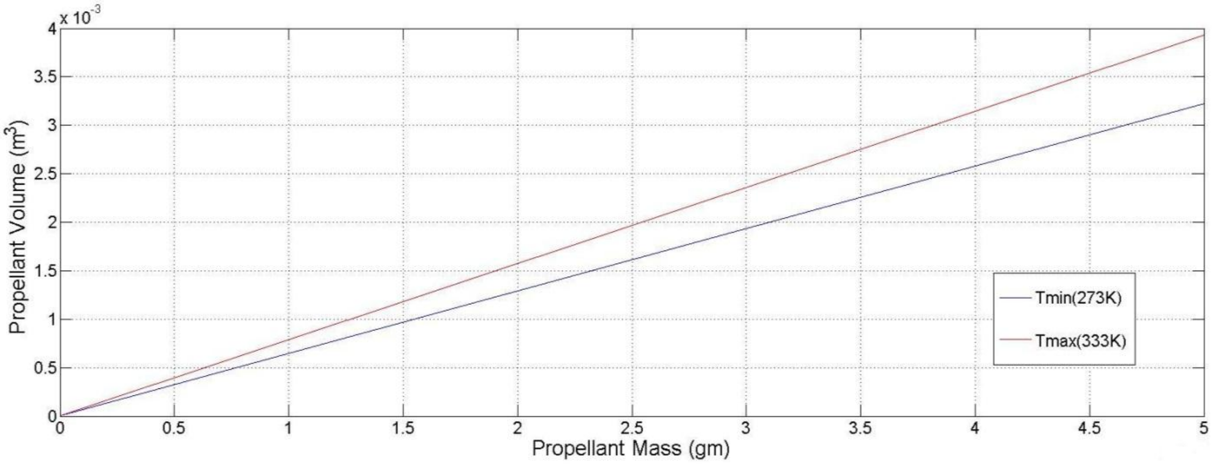

Figure 24 – Propellant Mass vs. Volume

Assuming a generous propellant volume to overall volume ratio of 50%, the propellant mass comes out to be roughly 0.75 g at the minimum operating temperature for a 1U CubeSat. This corresponds to a negligible ΔV of 0.332 m/s. For the same volume ratio with a 3U CubeSat, the propellant mass is roughly 2.5 g, with a corresponding ΔV of 0.368 m/s.

Given the possibility of applying for a waiver to go beyond the1.2 atm limit in mind, the Ideal Gas Law was again used to find volume requirements for varying amounts of propellant in a 1U CubeSat over a range of pressures much higher than the 1.2 atm limit. This can be seen in Figure 25.

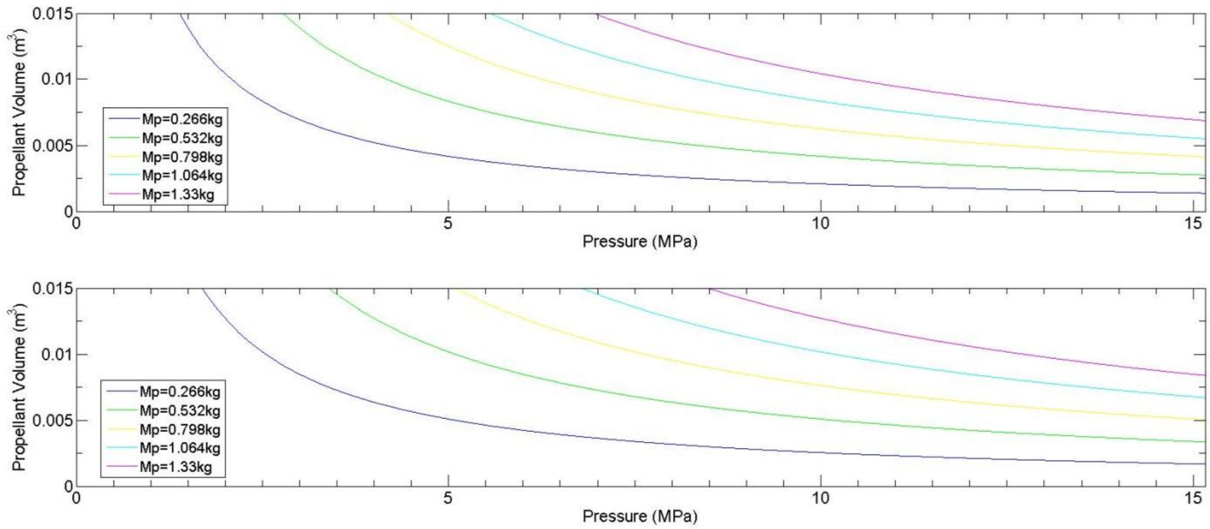

Figure 25 – Volume vs. pressure for different propellant masses at T_{min} (top) and T_{max} (bottom) for 1U

4.3.4 Atmospheric Drag

The last propulsion analysis involves considering the effects of atmospheric drag on the CubeSat's orbit. There is no simple, closed-form analytical model to accurately predict the atmospheric density at high altitude as a function of time due to the large number of uncertainties in gas composition, temperature, and solar activity. There are however, several atmospheric density models, one of which is the Mass Spectrometer and Incoherent Scatter (MSIS) model used in this analysis [10]. The MSIS atmospheric model uses tabulated values found by various measurements to predict atmospheric conditions over a period of time throughout various levels of the atmosphere. With the tabulated density, it is possible to estimate the change in semi-major axis height per revolution with the following equation [8]:

$$\Delta a_{rev} = -2\pi \left(\frac{C_d\, A}{m}\right) \rho\, a^2 \qquad \textbf{Eq 4.10}$$

Where C_d is the drag coefficient of a CubeSat in a rarefied gas [11], A is the frontal area (in this case 100 cm^2), m is mass (100 g), ρ is atmospheric density (assumed to be 4.914E-14 kg/m^3) [10], and a is the semi-major axis of 7091 km (radius of Earth plus altitude). For a 3U CubeSat, the values for drag will be the same as 1U due to the fact that all parameters remain the same, except for the frontal area and mass which scale proportionally to each other, (in other words; for a 3U CubeSat the frontal area becomes three times that of the 1U and mass also becomes three times that of the 1U). This calculation assumes that the CubeSat is orbiting with the largest frontal area perpendicular to the flow direction, in other words, "sideways" as opposed to "end first". With the above parameters, the initial change in semi-major axis height per revolution was found to be roughly 25 cm. However it is important to note that this is just a rough estimate, using an average value for atmospheric density based off of the MSIS atmospheric model. It is also noteworthy to add that this value does not remain constant. As the CubeSat descends with every orbital revolution, the density will continue to increase, causing the loss in semi-major axis altitude to grow exponentially until it finally reaches the point where the orbit decays rapidly. This "lifetime" of the satellite can be estimated using Satellite Tool Kit (STK), a software suite designed by Analytical Graphics, Inc, which allows mission planners to simulate the orbit of a spacecraft, providing important information on the satellite's environment, ground tracks, and many other details vital to mission design [13]. Upon completion of the lifetime calculation by STK, it was found that the CubeSat's expected lifetime will be approximately 60 years, which exceeds the required lifetime set forth by the CubeSat's mission.

4.4 Mechanical Structures Design Selection & Analysis

Analysis

In order to properly design and construct a CubeSat, analysis must be performed on CubeSat models. Examples of such "virtual tests" can include a manufacturability test, stress analysis test, and dynamic response analysis test, among others. Performing such studies on the models helps to optimize parts for improved performance in the intended environment and provides a low-cost solution to testing, in which the computer-based model is tested rather than machining the actual CubeSat and testing it multiple times, essentially eliminating multiple field tests. Furthermore, parts can be optimized for mass by performing stress analysis tests on the models to determine the minimum mass needed to have adequate structural strength.

Property Name	Value	Units
Elastic Modulus	7.2e+010	N/m^2
Poisson's Ratio	0.33	–
Shear Modulus	2.69e+010	N/m^2
Mass Density	2810	kg/m^3
Tensile Strength	5.7e+008	N/m^2
Yield Strength	5.05e+008	N/m^2
Thermal Expansion Coefficient	2.4e-005	1/K
Thermal Conductivity	130	W/(m·K)
Specific Heat	960	J/(kg·K)

Table 11 – Properties of Aluminum 7075 [15]

Before virtual stress analysis tests can be performed on the CubeSat models, it must first be known what types of force loading the spacecraft will undergo from initial transportation to end of operation. For most spacecraft, including CubeSats, the greatest force loading occurs during launch. In order to accurately estimate the loading on the WPI CubeSat during launch, the typical launch loading of three frequently used launch vehicles for past CubeSats were reviewed. The three launch vehicles reviewed are the Dnepr, the Eurockot, and the Minotaur I [12]. The most frequently used launch vehicle is the Dnepr [12], which is a Russian developed rocket that has been converted from an Intercontinental Ballistic Missile (ICBM) [11]. The Eurockot is a launch vehicle from Russian commercial launch provider Eurockot Launch Services and has been designed for launching satellites into low earth orbit (LEO) [13]. The Minotaur I is an American commercial launch vehicle for small satellites designed by Orbital Sciences Corporation [14]. The various types of launch vehicle loading reviewed include longitudinal and lateral g-loading, as well as random and harmonic vibration loading over different frequency ranges. With the load values acquired for each of these launch vehicles, accurate vibration load testing can be performed and the structure of the CubeSat can be optimized in order to withstand the greatest loading with as little mass as possible. Table 12 lists the maximum lateral and longitudinal loads for each of these three launch vehicles. Table 12 also shows when the maximum loading occurs during ascent. As shown below, the highest overall loading occurs with the Dnepr launch vehicle, so the WPI CubeSat flight option CubeSat will be designed to withstand similar loading.

Launch Vehicle	Max. Longitudinal G-Loading and Time	Max Lateral G- Loading and Time
Dnepr [11]	+ 8.3 g's at 2nd Stage Burn	0.8 g's after LV exit from transport launch canister
Eurockot [13]	+8.1 g's at Stage I engine Cut-Off	+/- 0.9 g's due to max. dynamic pressure
Minotaur I [14]	+6.6 g's 2nd Stage Ignition	+1.6 g's at Liftoff

Table 12 – Typical Launch Loads of Past CubeSat Launch Vehicles [14]

Once all the launch loads and characteristics are known, the structure can be tested using various simulation modeling tools. SolidWorks offers many modules to test the manufacturability and perform structural analysis of any models or assemblies created. In turn, this helps to optimize parts or assemblies to make them more efficient for use in the environments for which they are designed to operate. Manufacturability tests aid in determining how easy-to-machine a part will be so users can see how much time, effort, and cost parts will take to make. This allows the users to make educated decisions with regards to what designs and materials they want to use for the end product. Structural analysis tests aid in determining if assemblies can withstand the launch and flight characteristics specified. Lastly, simulation analysis aids in providing testing for different static and dynamic environments to model how the assembly will perform under adverse conditions.

To perform stress analysis, the SimulationXpress Analysis Wizard module inside SolidWorks was used to determine a model's performance under static loads. The SimulationXpress determines the von Mises Stress[4], displacement, deformation, and factor of safety of a module and provides results in graphical form. Using simulated restraints or fixtures, static loads such as forces or pressures can be applied to the model allowing for analysis under different stress conditions. This allows the designer to view which areas in the model are critical regions with regards to stress as well as view which areas of the model are being deformed by the stress. Depending on the factor of safety needed for the

[4] von Mises Stress – used to predict yielding of materials under any loading condition from results of simple uniaxial tensile tests

model, the designer can optimize the model by adjusting different parameters to create greater performance under stress while minimizing mass.

The DFMXpress module inside SolidWorks can be used to validate the manufacturability of the model by identifying design areas that might cause problems in fabrication or areas that might increase production costs. Specific examples of such problems are sharp interior corners, which are difficult to machine due to radii of the toolbits used to mill the material. Other areas of concern are stress concentrations that will result from machining very thin and long parts, since the toolbit will tend to deform the part which might cause failure and fractures. Using this module helped in ensuring the final Lab Option design could be machined here at WPI.

SolidWorks also offers a suite of Simulation modules: Linear Static Analysis, Frequency Analysis, Dynamic Analysis, Linearized Buckling Analysis, Thermal Analysis, Nonlinear Static Analysis, Drop Test Analysis, Fatigue Analysis, as well as Beam & Trusses Analysis. SolidWorks testing and analysis allows the creation of an optimal CubeSat model before machining an actual CubeSat structure.

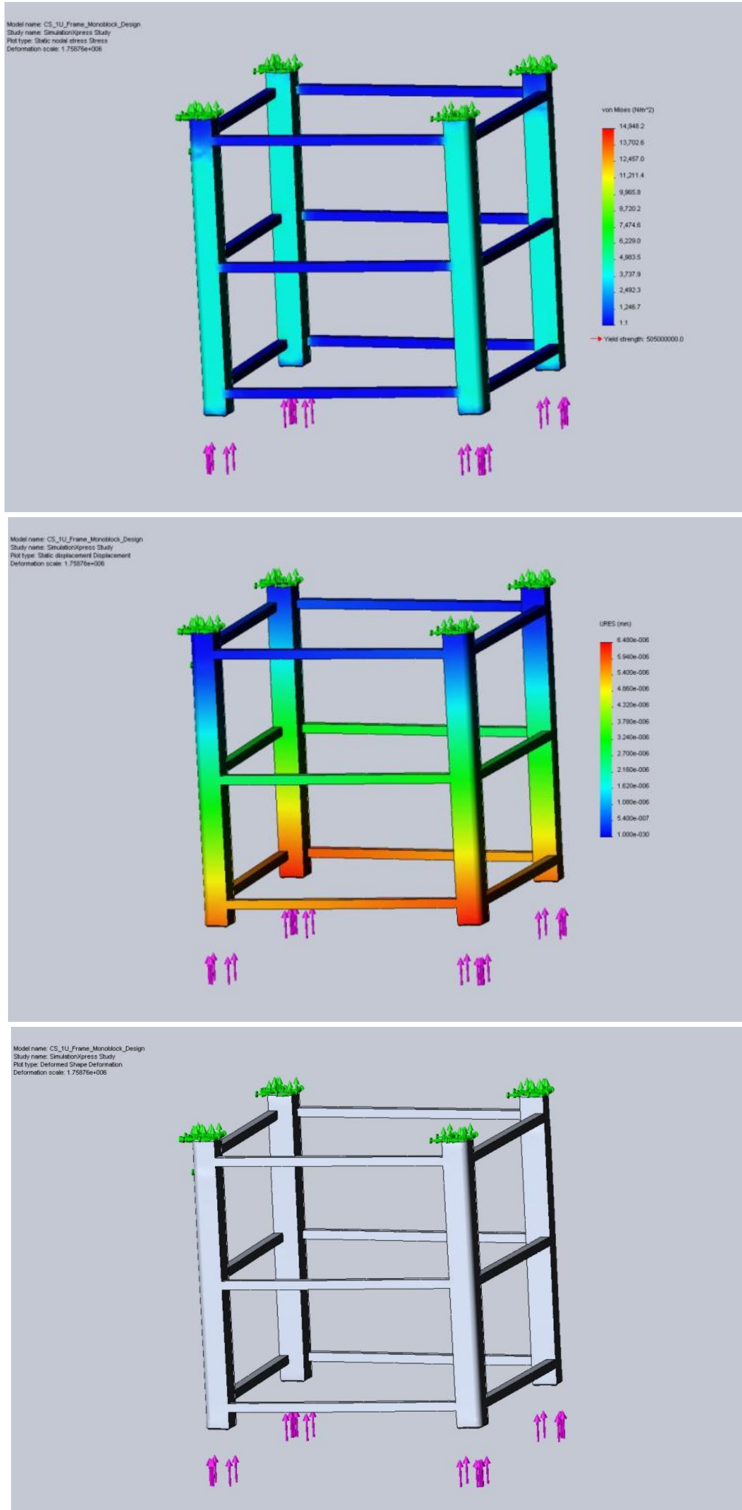

Figure 26 – CubeSat Structural Stress Analysis (von Mises Stress, Displacement, Deformation (top to bottom)

The results of the structural stress analysis are shown in Figure 26 were generated using the SimulationXpress Analysis Wizard module inside SolidWorks. A 1U Monoblock CubeSat structure was given 1 g static loads applied to the bottom rails directed upwards in order to simulate a launch environment. Three tests: the von Mises Stress, displacement, and deformation were performed and results used to investigate the structural integrity of the CS structure under launch environment conditions.

5 Flight Option Component Selection and Analysis

Although time and budget did not allow for purchase, construction, or testing of any Flight Option items, extensive research was performed to select ideal flight components for a space-ready CubeSat. These components were selected to perform the prescribed mission with the payload listed in Section 4.1.2, and to work with each other.

5.1 Power Subsystem

When selecting the power components (solar panels, battery, and PMAD) for the Flight Option CubeSat, two companies were considered: Clyde Space (Glasgow, Scotland) and CubeSat Kit (San Francisco, CO). After careful deliberation, Clyde Space was selected as the supplier for the Flight Option CubeSat components for several reasons.

Most importantly, Clyde Space's reliability and experience are unparalleled. Clyde Space has built and provided power system and solar panel components for the SOHLA-2 Panel ExTension SATellite (PETSAT); solar panels for the SumbandilaSat Satellite (which were constructed, tested, and shipped in a record 4 weeks) [4]; and a 1U EPS, two batteries, and solar panels for the PARADIGM UT CubeSat. In addition, Clyde Space supplies both off-the-shelf components, made in bulk, that conform to the CubeSat standards laid out by CalPoly, as well as custom made products that can be made to fit specific mission requirements. Finally, all of Clyde Space's components are space-ready and have undergone extensive testing. All Clyde Space components are compatible with one another, making it very easy to fully integrate a complete power system into a CubeSat.

The components selected for the Flight Option are all provided by Clyde Space, and can be ordered by the customer to perfectly match any mission parameters and specifications. These components, described in the following subsections, are clearly very expensive, but are the “industry standard” for flight-ready CubeSat components.

5.1.1 Flight Option Solar Cells

Clyde Space specializes in making custom CubeSat space components. Clyde Space creates custom 3U side solar panels that can incorporate seven large-area, triple junction, solar cells from Spectrolab for £3,750 ($6,120) [7]. It was determined that Clyde Space actually orders their solar cells from EMCORE and Spectrolab, and simply custom fits these cells to satellites.

The 29.5% efficiency NeXt Triple Junction (XTJ) Solar Cells from Spectrolab, which are constructed with a germanium substrate and cell structure of $GaInP_2/GaAs/Ge$, were selected for the flight option. These cells will cover an area of 271.825 cm^2, and at full illumination, will produce 10.962W with 2.33V at 17.32mA/cm^2. The amount of power these cells produce is far beyond the power requirements of the satellite payload and subsystems, as well as power needed for battery charging. These cells will still provide the satellite with enough power even including inefficiencies in the system or cell damage after launch. [9]

5.1.2 Flight Option Battery

Clyde Space produces a Lithium-Polymer battery for £950 ($1550). This battery has an energy density of 120-150Wh/kg, with a capacity of 1.25Ah at 8.2V for 10Whr. Moreover, the battery can operate for more than a year in LEO with over 5,000 charge/discharge cycles, which, like the solar cells, far exceeds the operational requirements of the CubeSat. [9]

5.1.3 Flight Option PMAD

Called an EPS (Electric Power System) by Clyde Space, the PMAD is specifically designed to integrate Clyde Space solar panels and batteries, making it an obvious choice for selection, priced at £2,600 ($4,240). Standard operating functions of the PMAD include load-leveling between the solar panels and battery, battery under-voltage and over-current protection, and max solar-panel tracking [37].

5.2 Propulsion Subsystem

Several options for flight-ready propulsion systems were considered for this CubeSat, and are detailed below.

5.2.1 Flight Option

The propulsion Flight Option recommendations are divided into two categories, Primary and Auxiliary propulsion. The Primary propulsion recommendation is the Micro Propulsion System (MiPS) manufactured by VACCO as described in Section 2.4.4. The MiPS is designed specifically for use onboard CubeSats and meets the volume and mass constraints inherent to a CubeSat. The MiPS offers high propellant storage mass with a low volume penalty since the propellant is stored as a liquid. The system is manufactured using patented ChEMS technology which allows plumbing connections to be avoided and a robust titanium design. The design also includes redundant valves to protect against leakage and does not include any sliding parts, which increases system reliability. The size of the propellant tank can be increased as needed to store more propellant and is compatible with a number of propellant types such as Nitrous Oxide (N_2O). As part of the Flight Option recommendation, the MiPS would only be used for primary propulsion (i.e. orbit raising or shaping) although it offers auxiliary propulsive capabilities. This is to allow for redundancy in auxiliary propulsion and to demonstrate the MiPS capabilities for orbit raising, which was not a requirement for the scientific payload.

It is recommended that four thruster couples (eight thrusters total) be used for auxiliary propulsion. Moog (model No. 58E142) thrusters were developed for the Pluto Fast Flyby mission and considered for the Dawgstar, a detailed description is provided in Section 2.3.5. The size (diameter 14mm, length 20mm) and mass (0.016 kg) of the thrusters will allow for simple integration into the CubeSat and a relatively low mass penalty. The thrusters are compatible with most cold gas propellant options and can be custom ordered.

6 Results & Conclusions

6.1 Power Conclusions

Although this report only provided a preliminary investigation into the lab and flight option systems for a CubeSat, it was possible to draw conclusions based on the research and system integration described above.

6.1.1 Solar Cells

It should be noted that solar cell area can be adjusted depending upon the size of the satellite (either 1U, 2U, or 3U), or if the decision is made to use some form of deployable array. Additionally, due to the significant performance variations between the Flight and Lab Option solar cells, testing with the Lab Option will most likely need to include a power umbilical during testing to simulate power level provided by the Flight Option solar cells. Otherwise, significant design changes would need to take place in order for the Lab Option to adequately model our Flight Option.

6.1.2 Batteries

Battery options were not rigorously investigated as a part of the lab option research in this report, but from the small amount of research above, it is clear that Li-ION batteries will be the best choice for lab testing of the CubeSat, as his type of battery provides sufficient power (approx.. 1500 mA-hr) and the battery charging circuitry for Li-ION batteries is relatively simple. A prefabricated and integrated battery pack from a space-manufacturing corporation (Clyde Space or similar) will be ideal for the flight option, primarily because it is designed to integrate with the other components and provides a relatively high power density and cycle life.

6.1.3 PMAD

The currently proposed lab-option PMAD will cost approximately $180-250 and will require an external power umbilical to simulate the power that will be provided by the flight-option solar cells (lab-option solar cells are not capable of producing enough power). The lab-option PMAD will be developed to the breadboard level (in this MQP) and its components eventually integrated into the lab test system(in a future MQP) to support all other components of the CubeSat lab test.

6.2 Propulsion Conclusions

Performing even these fairly basic analyses has led to important conclusions that will help facilitate the design of the CubeSat's propulsion system. Due to the strict volume constraints in a 10 cm x 10 cm x 30 cm satellite, it appears that the limiting factor for the effectiveness of this propulsion system will be the 1.2 atm limit for any pressure vessels on board. At this pressure, the mass of any gaseous propellant will be insufficient to perform any significant orbit raising maneuvers or inclination changes. However, it may be possible that this amount of propellant can be used to ensure the CubeSat does not exceed the orbital lifetime limit of twenty-five years. This result has also led to investigation of other forms of propellant such as liquid propellant, where the pressure would be the vapor pressure of the liquid (much lower than the pressure of compressed (non-vapor) gas in the conventional cold gas system). This would allow for the storage of much more propellant, while still staying below the pressure limit of 1.2 atm.

Information regarding the sensitivity of the scientific payload with respect to altitude is needed before a decision can be made as to whether or not drag's effects on the CubeSat's orbit are completely negligible. More analysis should also be done on gaining a better understanding of what the atmospheric density at time of launch will be. This could not be done directly using the online MSIS model, as it has information only up to June, 2010. However, given the periodic nature of the sun's activity, it is possible to look back into the past at a point that had similar solar activity as the expected launch date.

7 Recommendations

Through research, investigation, development, and testing of multiple CubeSat subsystems, the following recommendations are made for the future of this project.

7.1 Power Recommendations

The next logical step in designing the power subsystem for this CubeSat is to breadboard and test the lab option circuitry. Although time did not allow for these circuits to be built, they were fully designed and specified in this report, and are ready to be assembled and lab-tested.

Another area of Power Management that requires attention is the function of Load Leveling. This investigation of the capability to prevent power spikes and damage to flight hardware due to transient currents is critical to mission success, but was unfortunately beyond the scope of this report.

7.2 Propulsion Recommendations

To continue this project, the Lab Option propulsion system should be assembled and tested to determine the feasibility of performing an attitude control maneuver using a cold gas system. The propulsion subsystem team should work in collaboration with the structures subsystem team to develop a test fixture to support the CubeSat during testing and allow for as unrestricted motion as possible. If significant torques due to friction or other external forces cannot be avoided they should be accounted for in thrust calculations. Development of an algorithm to be used by the OBC for required valve open time will require the estimation of a thrust coefficient. The thrust coefficient is required to relate the chamber pressure to thrust which will determine attitude control maneuver parameters such as valve open time.

7.3 Mechanical Structures Recommendations

For the lab option CubeSat, it is recommended that the structure be assembled and all available subsystem components that are necessary to operate the cold gas propulsion system be configured into the satellite in a manner similar to that of Modular Design 3. After the Lab Option CubeSat is built, it is recommended that a test stand structure be built to be used in the vacuum chamber for testing the Lab Option propulsion capabilities. A schematic CAD model of a possible design for 1-axis test fixture is shown in Figure 27. The test stand consists of a base which will attach to connecting adapters in the vacuum chamber. There is a sliding bar attached to the base which could be used to adapt the test stand to fit 1U, 2U, and 3U CubeSat sizes. There are also two adaptor plates in which the rail ends of the CubeSat are inserted into. Attached to these adaptor plates are low friction bearings in order to allow for relatively unperturbed rotation during testing.

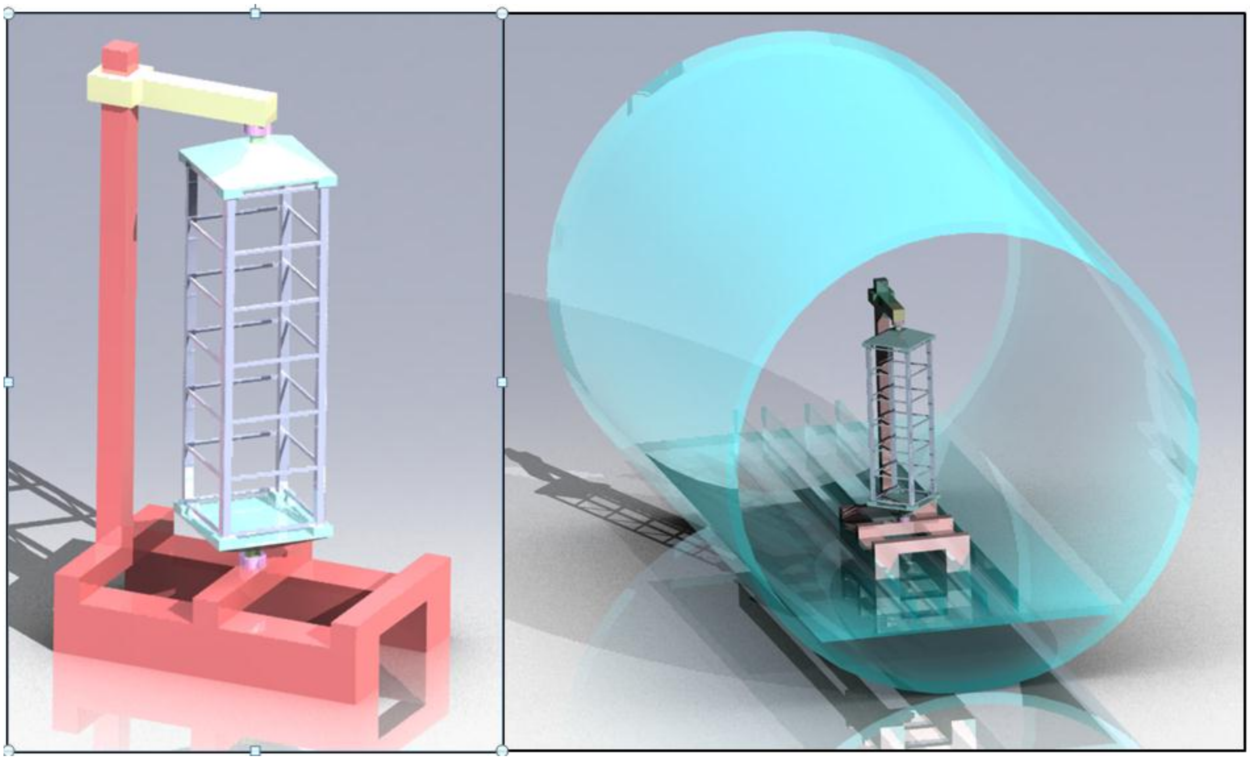

Figure 27 – Schematic CAD model of test stand fixture in close-up (left) and configured in the vacuum chamber (right)

For the Flight Option CubeSat, it is recommended that a monolithic design such as Monoblock Design 1 be used for the primary structure. This type of design requires the least amount of assembly for the structure, which can significantly reduce the overall weight. As previously stated, WPI's machining capabilities cannot support the complex design of a monolithic structure, so the machining will have to either be outsourced to more capable facilities, or a prefabricated CubeSat structure will have to be purchased from a specialized company such as ISIS or Pumpkin. It is assumed that the development of the Flight Option will involve a sufficient budget to accommodate the high expenses associated with obtaining an optimal CubeSat structure.

A second iteration of the monoblock design 1 has been created (monoblock design 2) which has a similar appearance in structure to the CubeSat Kit model in that the walls consist of a cross lattice design in order to optimize mass and structural integrity. Unlike the Lab Option models, monoblock design 1 does not use exterior paneling as part of the structure. Instead, the substrate that the solar cells are mounted to can act as the exterior walls of the CubeSat. In addition, thin aluminum walls can be mounted on the outside for thermal and environmental protection if necessary. A CAD model of Monoblock Design 2 is shown in Figure 28. The model shown contains all the recommended components from each of the subsystems.

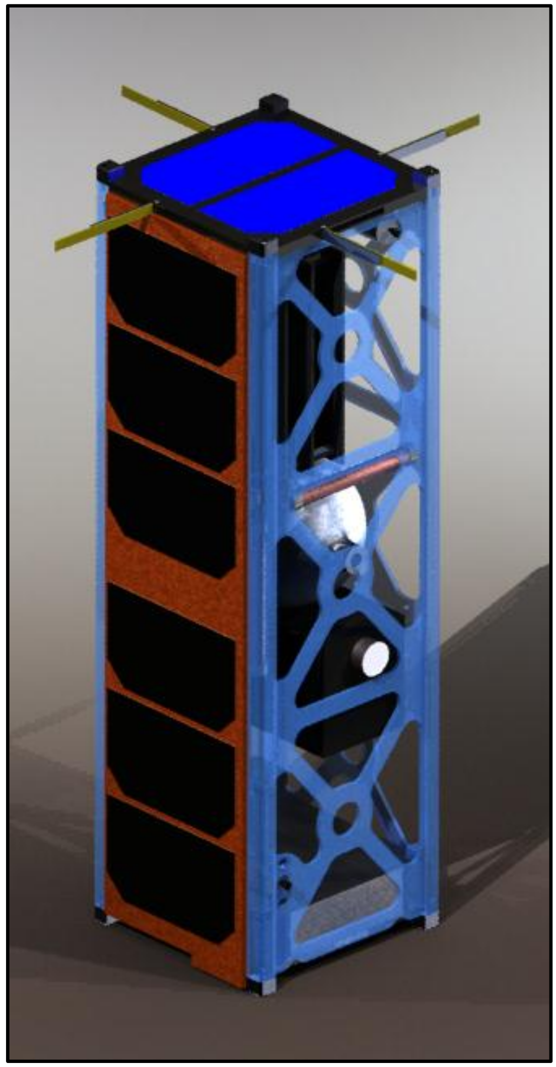

Figure 28 – CAD Assembly monoblock design with subsystem components configured inside

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Appendix A: Abbreviations and Variables

1U	One Unit, or One Liter CubeSat
2U	Two Unit, or Two Liter CubeSat
3U	Three Unit, or Three Liter CubeSat
BOL	Beginning of Life
Cal Poly	California Polytechnic State University
CubeSat	Cube Satellite
EOL	End of Life
I	Electrical Current [measured in Amperes unless otherwise specified]
I_d	Inherent Degradation
MQP	Major Qualifying Project
OBC	On-Board Computer
P_0	Power Density
P_{01}	Power Density
P_{02}	Power Density
P_{BOL}	Power Density at Beginning of Life
P-POD	Poly Picosatellite Orbital Deployer
PMAD	Power Management and Distribution
PPT	Pulsed Plasma Thrusters
SEG	System Engineering Group
V	Volts
W	Watts
WPI	Worcester Polytechnic Institute

Appendix B: CubeSat Design Specifications[5]

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Document Classification	
X	Public Domain
	ITAR Controlled
	Internal Only

CubeSat Design Specification

(CDS)

CUBESAT
California Polytechnic State University

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CHANGE HISTORY LOG

Effective Date	Revision	Author	Description of Changes
N/A	8	Simon Lee	N/A
5/26/05	8.1	Amy Hutputanasin	Formatting updated.
5/15/06	9	Armen Toorian	Information and presentation revised.
8/2/07	10	Wenschel Lan	Information updated.
10/02/08	11	Riki Munakata	Format, Design specification and Mk.III P-POD compatibility update.
8/1/09	12	Riki Munakata	Requirements update, waiver form requirements, and 3U CubeSat Specification drawing.

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APPENDIX

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List of Acronyms

CAC	CubeSat Acceptance Checklist
Cal Poly	California Polytechnic State University, San Luis Obispo
CDS	CubeSat Design Specification
cm	Centimeters
CVCM	Collected Volatile Condensable Mass
DAR	Deviation Wavier Approval Request
FCC	Federal Communication Commission
GSFC	Goddard Space Flight Center
IARU	International Amateur Radio Union
kg	Kilogram
LV	Launch Vehicle
mm	Millimeters
mW	Milliwatt
NASA	National Aeronautics and Space Administration
P/N	Part Number
P-POD	Poly Picosatellite Orbital Deployer
RBF	Remove Before Flight
Rev.	Revision
RF	Radio Frequency
SLO	San Luis Obispo
SSDL	Space Systems Development Lab
STD	Standard
TML	Total Mass Loss
µm	Micrometer

5 Official CubeSat specifications from http://cubesat.org/images/developers/cds_rev12.pdf

1. Introduction

1.1 Overview

Started in 1999, the CubeSat Project began as a collaborative effort between Prof. Jordi Puig-Suari at California Polytechnic State University (Cal Poly), San Luis Obispo, and Prof. Bob Twiggs at Stanford University's Space Systems Development Laboratory (SSDL). The purpose of the project is to provide a standard for design of picosatellites to reduce cost and development time, increase accessibility to space, and sustain frequent launches. Presently, the CubeSat Project is an international collaboration of over 100 universities, high schools, and private firms developing picosatellites containing scientific, private, and government payloads. A CubeSat is a 10 cm cube with a mass of up to 1.33 kg. Developers benefit from the sharing of information within the community. If you are planning to start a CubeSat project, please contact Cal Poly. Visit the CubeSat website at http://cubesat.org for more information.

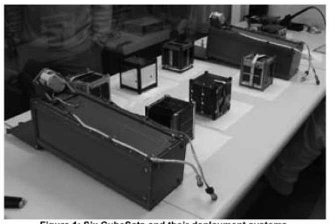

Figure 1: Six CubeSats and their deployment systems.

1.2 Purpose

The primary mission of the CubeSat Program is to provide access to space for small payloads. The primary responsibility of Cal Poly, as the developer of the Poly Picosatellite Orbital Deployer (P-POD), is to ensure the safety of the CubeSat and protect the launch vehicle (LV), primary payload, and other CubeSats. CubeSat developers should play an active role in ensuring the safety and success of CubeSat missions by implementing good engineering practice, testing, and verification of their systems. Failures of CubeSats, the P-POD, or interface hardware can damage the LV or a primary payload and put the entire CubeSat Program in jeopardy. As part of the CubeSat Community, all participants have an obligation to ensure safe operation of their systems and to meet the design and minimum testing requirements outlined in this document. Requirements in this document may be superseded by launch provider requirements.

1.3 Waiver Process

Developers shall fill out a "Deviation Waiver Approval Request (DAR)" (see appendix A) if their CubeSat is in violation of any requirements in sections 2 or 3. The waiver process is intended to be quick and easy. The intent is to help facilitate communication and explicit documentation between CubeSat developers, P-POD integrators, range safety personnel, and launch vehicle providers. This will help to better identify and address any issues that may arise prior to integration and launch. The DAR can be found at http://cubesat.atl.calpoly.edu/pages/documents/developers.php and waiver requests should be sent to standards@cubesat.org.

Upon completion of the DAR, the P-POD Integrator shall review the request, resolve any questions, and determine if there are any additional tests, analyses or costs to support the waiver. If so, the Developer, with inputs from the P-POD Integrator, shall write a Test Plan and perform the tests before the waiver is conditionally accepted by the P-POD Integrator. Waivers can only be conditionally accepted by the P-POD Integrator until a launch has been identified for the CubeSat. Once a launch has been identified, the waiver becomes mission specific and passes to the launch vehicle Mission Manager for review. The launch vehicle Mission Manager has the final say on acceptance of the waiver, and the Mission Manager may require more corrections and/or testing to be performed before approving the waiver. Developers should realize that each waiver submitted reduces the chances of finding a suitable launch opportunity.

CubeSat Standard Deviation Waiver Process

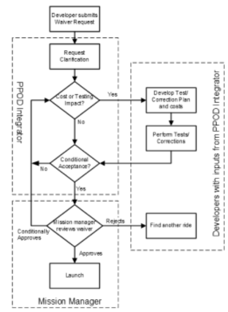

Figure 2: CubeSat Standard Deviation Wavier Process Flow Diagram

Poly Picosatellite Orbital Deployer

1.4 Interface

The Poly Picosatellite Orbital Deployer (P-POD) is Cal Poly s standardized CubeSat deployment system. It is capable of carrying three standard CubeSats and serves as the interface between the CubeSats and LV. The P-POD is a rectangular box with a door and a spring mechanism. Once the release mechanism of the P-POD is actuated by a deployment signal sent from the LV, a set of torsion springs at the door hinge force the door open and the CubeSats are deployed by the main spring gliding on its rails and the P-PODs rails (P-POD rails are shown in Figure 3b). The P-POD is made up of anodized aluminum. CubeSats slide along a series of rails during ejection into orbit. CubeSats shall be compatible with the P-POD to ensure safety and success of the mission, by meeting the requirements outlined in this document. The P-POD is backward compatible, and any CubeSat developed within the design specification of CDS rev. 9 and later, will not have compatibility issues. Developers are encouraged to design to the most current CDS to take full advantage of the P-POD features.

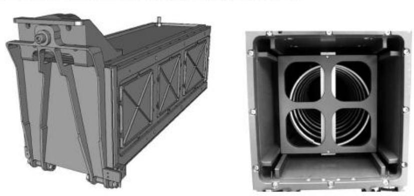

Figure 3a and 3b: Poly Picosatellite Orbital Deployer (P-POD) and cross section

2. CubeSat Specification

2.1 General Requirements

2.1.1 CubeSats which incorporate any deviation from the CDS shall submit a DAR and adhere to the waiver process (see Section 1.3 and Appendix A).

2.1.2 All parts shall remain attached to the CubeSats during launch, ejection and operation. No additional space debris shall be created.

2.1.3 Pyrotechnics shall not be permitted.

2.1.4 No pressure vessels over 1.2 standard atmosphere shall be permitted.

2.1.4.1 Pressure vessels shall have a factor of safety no less than 4.

2.1.5 Total stored chemical energy shall not exceed 100 Watt-Hours.

2.1.6 No hazardous materials shall be used on a CubeSat. Please contact us if you are unsure if a material is considered hazardous.

2.1.7 CubeSat materials shall satisfy the following low out-gassing criterion to prevent contamination of other spacecraft during integration, testing, and launch.

2.1.7.1 Total Mass Loss (TML) shall be $\leq$ 1.0 %

2.1.7.2 Collected Volatile Condensable Material (CVCM) shall be $\leq$ 0.1%

2.1.7.3 Note: A list of NASA approved low out-gassing materials can be found at: http://outgassing.nasa.gov

2.1.8 The latest revision of the CubeSat Design Specification shall be the official version (http://cubesat.calpoly.edu/pages/documents/developers.php), which all CubeSat developers shall adhere to.

2.1.8.1 Cal Poly shall send updates to the CubeSat mailing list upon any changes to the specification. You can sign-up for the CubeSat mailing list here: http://atl.calpoly.edu/mailman/listinfo/cubesat

2.2 CubeSat Mechanical Requirements

CubeSats are cube shaped picosatellites with a nominal length of 100 mm per side. Dimensions and features are outlined in the CubeSat Specification Drawing (Figure 5). General features of all CubeSats include:

2.2.1 **Exterior Dimensions**

2.2.2 The CubeSat shall use the coordinate system as defined in Figure 5. The Z face of the CubeSat will be inserted first into the P-POD.

2.2.3 The CubeSat configuration and physical dimensions shall be per Figure 5.

2.2.4 The CubeSat shall be 100.0$\pm$0.1 mm wide (X and Y dimensions per Figure 5).

2.2.5 A single CubeSat shall be 113.5$\pm$0.1 mm tall (Z dimension per Figure 5).

2.2.5.1 A Triple CubeSat shall be 340.5$\pm$0.3 mm tall (Z dimension per Appendix C).

2.2.6 All components shall not exceed 6.5 mm normal to the surface of the 100.0 mm cube (the green and yellow shaded sides in Figure 5).

2.2.7 Exterior CubeSat components shall not contact the interior surface of the P-POD, other than the designated CubeSat rails.

2.2.8 Deployables shall be constrained by the CubeSat. The P-POD rails and walls shall not to be used to constrain deployables.

2.2.9 Rails shall have a minimum width of 8.5mm.

2.2.10 The rails shall not have a surface roughness greater than 1.6 μm.

2.2.11 The edges of the rails shall be rounded to a radius of at least 1 mm

2.2.12 The ends of the rails on the +Z face shall have a minimum surface area of 6.5 mm x 6.5 mm contact area for neighboring CubeSat rails (as per Figure 5).

2.2.13 At least 75% of the rail shall be in contact with the P-POD rails. 25% of the rails may be recessed and no part of the rails shall exceed the specification.

2.2.13.1 For single CubeSats this means at least 85.1 mm of rail contact.

2.2.13.2 For triple CubeSats this means at least 255.4 mm rail contact.

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2.2.14 **Mass**

2.2.15 Each single CubeSat shall not exceed 1.33 kg mass.

2.2.16 Each triple CubeSat shall not exceed 4.0 kg mass.

2.2.17 The CubeSat center of gravity shall be located within a sphere of 2 cm from its geometric center.

2.2.18 **Materials**

2.2.19 Aluminum 7075 or 6061 shall be used for both the main CubeSat structure and the rails. If other materials are used the developer shall submit a DAR and adhere to the waiver process.

2.2.20 The CubeSat rails and standoff, which contact the P-POD rails and adjacent CubeSat standoffs, shall be hard anodized aluminum to prevent any cold welding within the P-POD.

2.2.21 The CubeSat shall use separation springs (Figure 4) with characteristics defined in Table 1 on the designated rail standoff. Separation springs with characteristics can be found using McMaster Carr P/N 84985A76. The separation springs provide relative separation between CubeSats after deployment from the P-POD.

2.2.21.1 The compressed separation springs shall be at or below the level of the standoff.

2.2.21.2 The throw length of the separation spring shall be a minimum of 0.05 inches above the standoff surface.

2.2.21.3 Separation springs are not required for 3U CubeSats.

Table 1: CubeSat Separation Spring Characteristics

Characteristics	*Value*
Plunger Material	*Stainless Steel*
End Force Initial/Final	*0.5 lbs. / 1.5 lbs.*
Throw Length	*0.05 inches minimum above the standoff surface*

Figure 4: Spring Plunger

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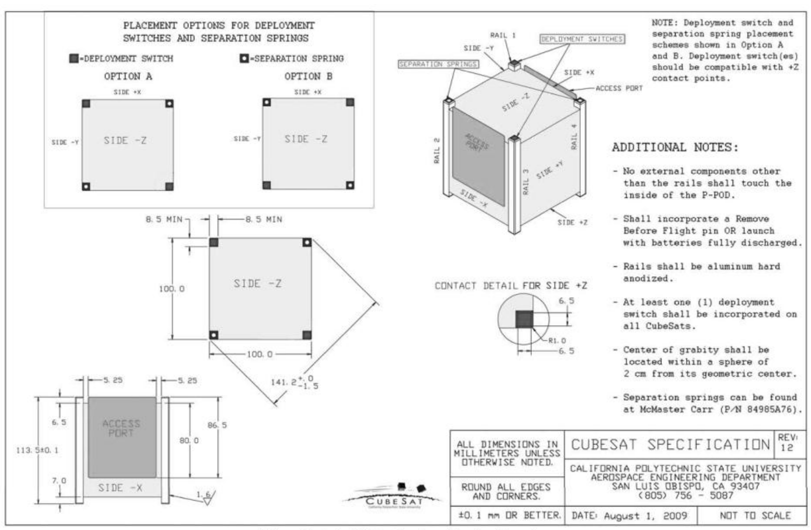


Figure 5: CubeSat Design Specification Drawing

2.3 Electrical Requirements

Electronic systems shall be designed with the following safety features.

2.3.1 No electronics shall be active during launch to prevent any electrical or RF interference with the launch vehicle and primary payloads. CubeSats with batteries shall be fully deactivated during launch or launch with discharged batteries.

2.3.2 The CubeSat shall include at least one deployment switch on the designated rail standoff (shown in Figure 5) to completely turn off satellite power once actuated. In the actuated state, the deployment switch shall be centered at or below the level of the standoff.

2.3.2.1 All systems shall be turned off, including real time clocks.

2.3.3 To allow for CubeSat diagnostics and battery charging after the CubeSats have been integrated into the P-POD all CubeSat umbilical connectors shall be within the designated Access Port locations, green shaded areas shown in Figure 5.

2.3.3.1 Triple CubeSats shall use the designated Access Port locations (green shaded areas) show in Appendix C.

2.3.3.2 Note: CubeSat deployment switch shall be depressed while inside the P-POD. All diagnostics and battery charging shall be done while the deployment switch is depressed.

2.3.4 The CubeSat shall include a Remove Before Flight (RBF) pin or launch with batteries fully discharged. The RBF pin shall be removed from the CubeSat after integration into the P-POD.

2.3.4.1 The RBF pin shall be accessible from the Access Port location, green shaded areas shown in Figure 5.

2.3.4.1.1 Triple CubeSats shall locate their RBF pin in one of the 3 designated Access Port locations (green shaded areas) show in Appendix C.

2.3.4.2 The RBF pin shall cut all power to the satellite once it is inserted into the satellite.

2.3.4.3 The RBF pin shall not protrude more than 6.5 mm from the rails when it is fully inserted into the satellite.

2.4 Operational Requirements

CubeSats shall meet certain requirements pertaining to integration and operation to meet legal obligations and ensure safety of other CubeSats.

2.4.1 CubeSats with batteries shall have the capability to receive a transmitter shutdown command, as per Federal Communications Commission (FCC) regulation.

2.4.2 All deployables such as booms, antennas, and solar panels shall wait to deploy a minimum of 30 minutes after the CubeSat's deployment switch(es) are activated from P-POD ejection.

2.4.3 RF transmitters greater than 1 mW shall wait to transmit a minimum of 30 minutes after the CubeSat's deployment switch(es) are activated from P-POD ejection.

2.4.4 Operators shall obtain and provide documentation of proper licenses for use of frequencies.

2.4.4.1 For amateur frequency use, this requires proof of frequency coordination by the International Amateur Radio Union (IARU). Applications can be found at www.iaru.org.

2.4.5 The orbital decay lifetime of the CubeSats shall be less than 25 years after end of mission life.

2.4.5.1 Developers shall obtain and provide documentation of approval of an orbital debris mitigation plan from the FCC or local agency.

2.4.6 Cal Poly shall conduct a minimum of one fit check in which developer hardware shall be inspected and integrated into the P-POD. A final fit check shall be conducted prior to launch. The CubeSat Acceptance Checklist (CAC) shall be used to verify compliance of the specification (Appendix B for single CubeSats and Appendix D for triple CubeSats).

3. Testing Requirements

Testing shall be performed to meet all launch provider requirements as well as any additional testing requirements deemed necessary to ensure the safety of the CubeSats and the P-POD. If launch vehicle environment is unknown, GSFC-STD-7000 (http://standards.gsfc.nasa.gov/gsfc-std/gsfc-std-7000.pdf) shall be used to derive testing requirements. GSFC-STD-7000 is a useful reference when defining testing environments, however the test levels defined in GSFC-STD-7000 are not guaranteed to encompass or satisfy all LV testing environments. Test plans that are not generated by the launch provider or P-POD Integrator are considered to be unofficial. Requirements derived in this document may be superseded by launch provider requirements. All flight hardware shall undergo protoflight and acceptance testing. The P-PODs shall be tested in a similar fashion to ensure the safety and workmanship before integration with the CubeSats. At the very minimum, all CubeSats shall undergo the following tests.

3.1 Random Vibration

Random vibration testing shall be performed as defined by LV provider, or if unknown, GSFC-STD-7000.

3.2 Thermal Vacuum Bakeout

Thermal vacuum bakeout shall be performed to ensure proper outgassing of components. The test cycle and duration will be outlined by LV provider, or if unknown, GSFC-STD-7000.

3.3 Visual Inspection

Visual inspection of the CubeSat and measurement of critical areas shall be performed per the 1U CAC (Appendix B) or 3U CAC (Appendix D) as appropriate.

3.4 Qualification

CubeSats may be required to survive qualification testing as outlined by the LV provider. If LV environments are unknown, GSFC-STD-7000 (NASA GEVS). Qualification testing will be performed at developer facilities. In some circumstances, Cal Poly can assist developers in finding testing facilities or provide testing for the developers. **Additional**

testing may be required if modifications or changes are made to the CubeSats after qualification testing.

3.5 Protoflight

All CubeSats shall survive protoflight testing as outlined by the LV provider. If LV environments are unknown, GSFC-STD-7000. Protoflight testing will be performed at developer facilities. In some circumstances, Cal Poly can assist developers in finding testing facilities or provide testing for the developers. CubeSats **SHALL NOT** be disassembled or modified after protoflight testing. Disassembly of hardware after protoflight testing shall require the developer to submit a DAR and adhere to the waiver process prior to disassembly. **Additional testing shall be required if modifications or changes are made to the CubeSats after protoflight testing.**

3.6 Acceptance

After delivery and integration of the CubeSats into the P-POD, additional testing shall be performed with the integrated system. This test ensures proper integration of the CubeSats into the P-POD. Additionally, any unknown, harmful interactions between CubeSats may be discovered during acceptance testing. The P-POD Integrator shall coordinate and perform acceptance testing. After acceptance testing, developers may perform diagnostics through the designated P-POD diagnostic ports, and visual inspection of the system shall be performed by the P-POD Integrator. The P-POD **SHALL NOT** be deintegrated at this point. If a CubeSat failure is discovered, a decision to deintegrate the P-POD will be made by the developers, in that P-POD, and the P-POD Integrator based on safety concerns. The developer is responsible for any additional testing required due to corrective modifications to deintegrated P-PODs and CubeSats.

4. Contacts

Cal Poly - San Luis Obispo
Prof. Jordi Puig-Suari
Aerospace Engineering Dept.
(805) 756-5087
(805) 756-2376 fax
jpuigsua@calpoly.edu

Roland Coelho
Program Manager
(805) 756-5087
(805) 756-5165 fax
rcoelho@calpoly.edu

Student Contacts
(805) 756-5087
(805) 756-5165 fax
cubesat@gmail.com

SRI International
Dr. Scott Williams, Program Manager
Engineering Systems Division
(650) 859-5057
(650) 859-3919 fax
scott.williams@sri.com

CubeSat Design Specification
Deviation Waiver Approval Request (DAR)

Date: August 1, 2009 Rev. 12

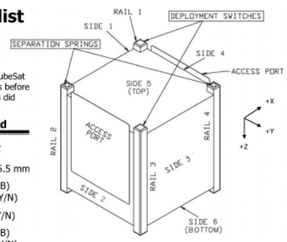
CubeSat Developers only fill out sections 1 through 9 and 15(optional). Email to: standards@cubesat.org

1. MISSION NAME:	**2. DAR NUMBER:**	**3. DATE:**
4. INITATOR	**5. INITIATING ORGANIZATION:**	
6. SPECIFIED REQUIREMENTS NUMBERS:	**7. JUSTIFICATION FOR DAR:**	**8. WAIVER TYPE** ☐ DIMENSIONS or MASS ☐ STRUCTURE ☐ ELECTRICAL ☐ OPERATIONS ☐ TESTING ☐ OTHER
9. DESCRIPTION OF DEPARTURE FROM REQUIREMENTS:		
10. CSEP DISPOSITION: ☐ ACCEPTED ☐ REJECTED ☐ CONDITIONALLY ACCEPTED	**11. ACCEPT/REJECT JUSTIFICATION:**	
CSEP AUTHORIZED REP.	**SIGNATURE**	**ORGANIZATION** **DATE**
12. ACCEPTANCE CONDITIONS		
13. LAUNCH VEHICLE INTEGRATOR APPROVAL AUTHORITY: ☐ APPROVED ☐ DISAPPROVED ☐ CONDITIONALLY APPROVED	**14. LVI APPROVAL/DISAPPROVAL JUSTIFICATION:**	
LVI AUTHORIZED REP.	**SIGNATURE**	**ORGANIZATION** **DATE**
15. APPROVAL CONDITIONS		

CubeSat Acceptance Checklist

Revision: 12
Revision Date: August 1, 2009
Author: Riki Munakata

This document is intended to be used concurrently with the CubeSat Integration Procedure (CIP) and should be used by developers before and after each testing phase to verify the CubeSat dimensions did not change.



List Item	Actual	Required
Mass	______	$\leq 1330g$
Remove Before Flight	______	Protrudes $\leq$ 6.5 mm
Spring Plungers	______	Option (A/B) Functional (Y/N)
Rails	______	Anodized (Y/N)
Deployment Switches	______	Option (A/B) Functional (Y/N)
Width [x-y], Top		
Side 1	______	$100.0 \pm 0.1mm$
Side 2	______	$100.0 \pm 0.1mm$
Side 3	______	$100.0 \pm 0.1mm$
Side 4	______	$100.0 \pm 0.1mm$
Width [x-y], Middle		
Side 1	______	$100.0 \pm 0.1mm$
Side 2	______	$100.0 \pm 0.1mm$
Side 3	______	$100.0 \pm 0.1mm$
Side 4	______	$100.0 \pm 0.1mm$
Width [x-y], Bottom		
Side 1	______	$100.0 \pm 0.1mm$
Side 2	______	$100.0 \pm 0.1mm$
Side 3	______	$100.0 \pm 0.1mm$
Side 4	______	$100.0 \pm 0.1mm$

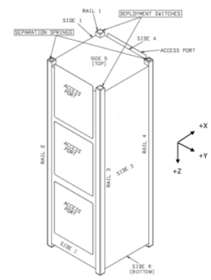
Authorized By:	Testing Info:
IT #1: ______________	Date: ___________
IT #2: ______________	Passed: **Y / N**

List Item	Actual	Required
Height [z]		
Rail 1	______	$113.5 \pm 0.1mm$
Rail 2	______	$113.5 \pm 0.1mm$
Rail 3	______	$113.5 \pm 0.1mm$
Rail 4	______	$113.5 \pm 0.1mm$
Diagonal [x-y]		
Top 1&3	______	$141.2^{+0}_{-1.5}mm$
Top 2&4	______	$141.2^{+0}_{-1.5}mm$
Bottom 1&3	______	$141.2^{+0}_{-1.5}mm$
Bottom 2&4	______	$141.2^{+0}_{-1.5}mm$
Protrusions		
Side 1	______	$6.5 + 0.0mm$
Side 2	______	$6.5 + 0.0mm$
Side 3	______	$6.5 + 0.0mm$
Side 4	______	$6.5 + 0.0mm$
Side 5	______	$6.5 + 0.0mm$
Side 6	______	$6.5 + 0.0mm$

CubeSat Acceptance Checklist

Revision: 12
Revision Date: August 1, 2009
Author: Riki Munakata

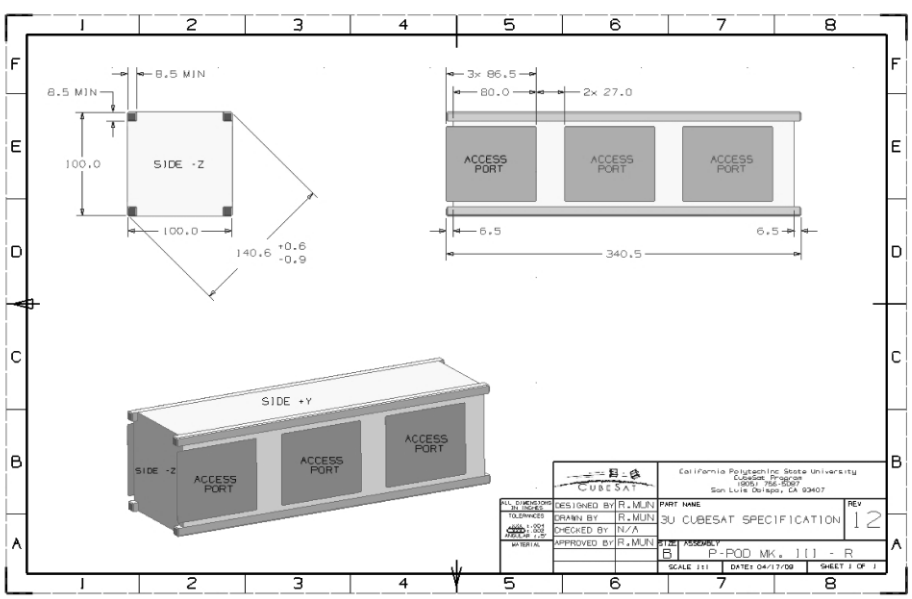
This document is intended to be used concurrently with the CubeSat Integration Procedure (CIP) and should be used by developers before and after each testing phase to verify the CubeSat dimensions did not change.



List Item	Actual	Required
Mass	______	$\leq 4000g$
Remove Before Flight	______	Protrudes $\leq$ 6.5 mm
Spring Plungers	______	Option (A/B) Functional (Y/N)
Rails	______	Anodized (Y/N)
Deployment Switches	______	Option (A/B) Functional (Y/N)
Width [x-y], Top		
Side 1	______	$100.0 \pm 0.1mm$
Side 2	______	$100.0 \pm 0.1mm$
Side 3	______	$100.0 \pm 0.1mm$
Side 4	______	$100.0 \pm 0.1mm$
Width [x-y], Middle		
Side 1	______	$100.0 \pm 0.1mm$
Side 2	______	$100.0 \pm 0.1mm$
Side 3	______	$100.0 \pm 0.1mm$
Side 4	______	$100.0 \pm 0.1mm$
Width [x-y], Bottom		
Side 1	______	$100.0 \pm 0.1mm$
Side 2	______	$100.0 \pm 0.1mm$
Side 3	______	$100.0 \pm 0.1mm$
Side 4	______	$100.0 \pm 0.1mm$

Authorized By:	Testing Info:
IT #1: ______________	Date: ___________
IT #2: ______________	Passed: **Y / N**

List Item	Actual	Required
Height [z]		
Rail 1	______	$340.5 \pm 0.3mm$
Rail 2	______	$340.5 \pm 0.3mm$
Rail 3	______	$340.5 \pm 0.3mm$
Rail 4	______	$340.5 \pm 0.3mm$
Diagonal [x-y]		
Top 1&3	______	$141.2^{+0}_{-1.5}mm$
Top 2&4	______	$141.2^{+0}_{-1.5}mm$
Bottom 1&3	______	$141.2^{+0}_{-1.5}mm$
Bottom 2&4	______	$141.2^{+0}_{-1.5}mm$
Protrusions		
Side 1	______	$6.5 + 0.0mm$
Side 2	______	$6.5 + 0.0mm$
Side 3	______	$6.5 + 0.0mm$
Side 4	______	$6.5 + 0.0mm$
Side 5	______	$6.5 + 0.0mm$
Side 6	______	$6.5 + 0.0mm$



Appendix C: CubeSat Database

General					Controls		
Date	Launch Vehicle	CubeSat	Company	Unit	Attitude Actuators	Attitude Sensors	Magnetometer Information
Jun-03	Eurockot	Aau Cubesat	Aalborg Univ.	1	magnetorquers	magnetometer (Honeywell components), sun sensor	Internal; 2 used in ADCS
		DTUsat-1	Technical Univ. of Denmark	1	magnetorquers	5 sun angle sensors, magnetometers (Honeywell components)	Internal; Built from 4 Honeywell sensors;
		CanX-1	Univ. of Toronto Inst. for Aero. Studies	1	magnetorquers	GPS, magetometer (Honeywell)	Internal; Honeywell HMR 2300 3-axis; other subsystems shut down during de-tumbling
		Cute-1	Tokyo Institute of Technology	1	?	Piezoelectric Vibrating Gyros, 2-axis accel, CMOS camera Sun sensor	
		XI-IV	University of Tokyo	1	permanent magnet, libration damper	?	
		QuakeSat	Stanford Univ. and Quakefinder	3	permanent magnet, hysteresis rods	magnetometer (Honeywell components), sun sensor	External; 0.701 m deployable boom; ELF magnetometer
Oct-05	SSETI Express	Ncube-2	Norwegian Univ. of Science and Tech.	1	magetorquers	magnetometer	Yes
		XI-V	Univ. of Tokyo	1	permanent magnet, libration damper	?	
		UWE-1	Univ. of Wurzburg	1	?	Gyro	
Feb-06	M-V-8	Cute-1.7+APD	Tokyo Institute of Technology	2	magnetorquers	Gyro, sun sensors, manetometer	Internal; Honeywell HMR 2300
Jul-06	Dnepr*	AeroCube-1	The Aerospace Corporation	1	?	?	
		CP1	Cal Poly, SLO	1	magnetorquers	sun sensor	
		CP2	Cal Poly, SLO	1	magnetorquers	magnetometers	Internal; 2-axis magnetometer on each side board
		ICEcube-1	Cornell University	1	gravity gradient boom, magnetorquers	Magnetometer, GPS	Yes
		ICEcube-2	Cornell University	1	gravity gradient boom, magnetorquers	Magnetometer, GPS	Yes
		ION	Universtiy of Illinios	2	magnetorquers, micro-vacuum arc thrusters	magnetometer,sun sensors	Internal (as far as I can tell); Honeywell HMC 2003
		HAUSAT 1	Hankuk Aviation University	1	permanent magnet, hysteresis rods	sun sensor, GPS	
		KUTEsat	University of Kansas	1	magnetorquers	magetometer, sun sensor	Yes
		MEROPE	Montana State University	1	permanent magnet, hysteresis rods	sun sensor (solar panels)	
		Ncube-1	Norwegian Univ. of Science and Tech.	1	magnetorquers	magnetometer (Honeywell), sun sensor (Solar panels)	Yes
		RINCON	Univ. of Arizona	1	spin stabilized via sunlight	?	
		SACRED	Univ. of Arizona	1	spin stabilized via sunlight	?	
		SEEDS	Nihon University	1	?	Gyro, magnetometer	Yes
		Voyager	Univ. of Hawaii	1	permanent magnet, hysteresis rods		
Dec-06	Minotaur 1	GeneSat-1	NASA Ames Research Center	3	permanent magnet, hysteresis rods	Gyro, accelerometer	
Apr-07	Dnepr	CSTB1	The Boeing Company	1	magetorquers	sun sensor, magnetometer	Yes
		AeroCube-2	The Aerospace Corporation	1	?	?	
		CP3	Cal Poly, SLO	1	magnetorquers	magnetometer	Internal/external; 2-axis magnetometer on each side panel
		CP4	Cal Poly, SLO	1	magnetorquers	magnetometers	Internal/external; 2-axis magnetometer on each side panel
		Libertad-1	Univ. Sergio Arboleda	1	?	GPS	
		CAPE1	Univ. of Louisiana	1	permanent magnet, hysteresis rods	?	

General					Controls		
Date	Launch Vehicle	CubeSat	Company	Unit	Attitude Actuators	Attitude Sensors	Magnetometer Information
		MAST	Tethers Unlimited, Inc.	3	?	GPS, magnetometer	
Apr-08	PSLV-C9	COMPASS-1	Aachen Univ. of Applied Science	1	magnetorquers	GPS, sun sensors, magnetometer	Yes
		Delfi-C3	Delft Univ. of Technology	3	permanent magnet, hysteresis rods	sun sensor (solar panels), wireless sun sensor	
		SEEDS-2	Nihon Univ.	1	?	Gyro, magnetometer	Yes
		CanX-2	Univ. of Toronto Inst. for Aero. Studies	3	magnetorquer, reaction wheel	sun sensors. Magetometers, camera (Earth, moon, stars sensor)	External; 20 cm extendable boom
		AAUSAT-II	Aalborg Univ.	1	magnetorquer, momentum wheels	Gyro, magnetometer	
		Cute-1.7+APDII	Tokyo Institute of Technology	2	magnetorquer	Gyro, magnetometer, sun sensor	
Aug-09	Falcon 1*	NanoSail-D	NASA Marshall Space Flight Center	?	Magnetic Passive stabilization**	?	
		PreSat	NASA Ames Research Center	?	?	?	
May-09	Minotaur 1	AeroCube-3	The Aerospace Corporation	1	permanent magnet, hysteresis rods	2-axis sun sensor, earth sensor	
		CP6	Cal Poly, SLO	1	magnetorquers	magnetometers	Internal/external; 2-axis magnetometer on each side panel; mission was to implement ADCS with only magnetometers/magnetorquers
		HawkSat	Hawk institute for Space Sciences	1	?	?	
		PharmaSat	NASA Ames Research Center	3	?	?	
Sep-09	PSLV-C14	BeeSat	Berlin Institute of Technology	?	micro wheels, magnetorquers	sun sensor, magnetometer (Honeywell), gyros	
		ITUpSAT	Instanbul Technical Univ.	?	permanent magnet	magnetometer, gyro, accelerometer	
		SwissCube	Ecole Polytecnique Federale de Lausanna	?	magnetorquers	magnetometer, sun sensor, gyro	
		UWE-2	Univ. of Wurzburg	?	permanent magnet	GPS, sun sensors, magnetometer, gyro, accelerometer	
Jul-09	Endeavour	Aggiesat-2	Texas A&M Univ.	1	?	GPS	Yes
		BEVO 1	Univ. of Texas at Austin	1	?	GPS	
* Launch Vehicle Failed							
**premanent magnet and/or hysteresis rods							