Spirit CubeSat Electric Power System

by

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Abstract

Western University was provided a grant from the Canadian Space Agency to design and build a nanosatellite called a CubeSat that will be launched from the International Space Station in 2021. The goal of the satellite is to take photos and videos of the Earth and transmit the data back to the Western ground station. Two designs were considered based on communication protocols: S-Band and UHF. As the S-Band Electrical Power System team, our project addressed the problem of providing adequate and reliable power for the satellite to perform its mission while adhering to financial and weight parameters. Our design is comprised of 1U solar panels, a power distribution system and a 40Whr battery bank. The performance of the Spirit CubeSat is dependent on its ability to produce above 65% of its maximum power generation to supply nominal mode. The power system can supply two 20-minute videos, 10 still images, and two 10-minute data transmission windows over a four-orbit period. This yields a net loss of 7% in battery capacity, which can be recovered through 80 minutes in safe mode or 133 minutes in nominal mode. The bottlenecks found include hindrances on power generation and the S-band transmission window duration.

Contribution of the Team Members

After starting the project and confirming the three components being used in the electrical power system, the team assigned each member one part and the fourth would be responsible for modelling the power generation. This included in selecting the components used in the prototype phase of the project.

Ze Xu Zhu was responsible for building and testing solar panels for the project. To select the ideal solar cell, the properties of semiconductor materials will be considered. Different semiconductor properties will dictate trade-offs between efficiency, price, current-voltage and power-voltage characteristics. Ze was selected as the best candidate for this role because of his previous experience working with solar panels at his internship employment during the last year.

Jamal Bouibaoune was responsible for the design of the power management system. The power management and distribution system design had the opportunity to work with other subsystems of the CubeSat to find rated voltages and their power consumption requirements. Jamal excelled at this role because of his strong intrapersonal skills for communicating clearly and efficiently between teams.

Alec Parhar was responsible in determining the ideal battery for our mission, assessing the mass and material property. The battery has a significant role during space missions so there was a lot of combined work between other members during the research phase to find a battery that matched our desired objectives in terms of cost, physical size, capacity and efficiency. Alec had worked intimately with batteries in his first-year design project and was adamite that he be responsible for this portion of the design.

Adam Dunn was responsible for modelling the behaviour of power generation under various circumstances to ensure the CubeSat was versatile. This included testing all operational modes under different amounts of power generation as well as the recovery time for the battery to return to moderate charge. Adam was selected for this task because of his knowledge in using Matlab and Excel software to efficiently track multiple variables.

In hindsight, this approach limited group members attention to one component of the project and doesn't give a holistic view. It appears difficult to separate tasks equally through another method so perhaps this was the fairest approach for this Capstone project.

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1. Introduction/Background

1.1 Problem Statement

Satellites have been a rapidly growing technology used by communications industries, governments and media to support a global transfer of information. Constructing a satellite still remains challenging due to the complexity of working with several subsystems, however the bulk of difficulties revolves around prohibitive costs, reliability, and unit lifetime.

Modern satellites are often built for a 15-year mission[1], resulting in proportionally expensive equipment, fuel and labour for construction and launch. Transitioning to a smaller satellite can aid with cost feasibility and in-orbit reliability for customers. The downsides with this approach are a reduced orbital lifetime, smaller payload and a stricter power budget[2].

Developing and launching a functional miniature satellite can demonstrate viability for use in industry. However, small satellites are burdened with reduced mass budgets, limiting the use of large solar panels and batteries for power capacity. This minimalist approach places emphasis on selecting optimal power system components and providing accurate modeling to ensure that the satellite can be operated for maximum orbit lifetime and supply payloads onboard[2].

1.2 Background Information/ Detailed Literature Review

The electrical power system for space vehicles is the most fundamental requirement for a satellite as failure results in the direct loss of a mission. In the 1980s, there was great focus on large power systems with the infrastructure associated with the Space Station, however political and technical issues have alternated the design to a smaller approach. Advances in Very Large Scale Integration (VLSI)[3] led to the possibility of sophisticated functions being built into small volumes, with low mass and requiring minimal electrical power. This led to the emergence of the nanosatellite, which dramatically reduced costs for owners due to fewer components at a fraction the cost. A nanosatellite tends to have the same electrical design regardless of function. The three elements include power distribution module, primary and secondary energy sources (Figure 1 & 2).

The purpose of the primary energy source is to convert a fuel into electrical power. On early space flights, batteries used to provide this function however with longer missions that use more advanced payloads, solar arrays became the industry standard. The fuel in this case is the solar radiant energy, which is converted via the photovoltaic effect into electrical energy. Other options for primary energy are hydrogen fuel cells. Fuel cells have provided power for the Shuttle Orbiter with a proven history of reliability as well as had significant improvement in the technology[3]. A fuel cell converts the chemical energy of an oxidation reaction directly into electrical energy with minimal heat change. This technology is advantageous in space due to its flexibility of generating power during sunlit or eclipse orbits and current research has improved its efficiency to 60%. The drawback is the requirement to carry fuel onboard the satellite. CubeSat devices carry no wet fuel so hydrogen fuels cells would be incompatible for our design. Other researched primary power sources include Radioisotope Thermoelectric Generators (RTG) and Nuclear Fission Systems[3]. RTGs (Figure 3) use the thermoelectric effect to generate a voltage between two materials while a temperature difference is maintained. It uses the heat of a decaying isotope and the outside environment to provide this difference. Fission would operate similar to conventional ground-based nuclear power with fissile uranium-235 as a heat source. Both of these sources provide energy for space missions over 5 years, which would extend the mission life but the fuel sources are unobtainable nuclear material so it can be neglected from the design process immediately. Solar panels are the only reliable primary power source that can fulfill the mission duration, provide sufficient energy and remain in financial budget.

Solar arrays are an assembly of thousands of individual solar cells that are connected to provide DC power levels. Each solar cell has a semiconductor p-n junction (Figure 4). For spacecraft applications, the base material typically has a resistivity between 10^-3 and 10^2 Ωcm. With no illumination, the junction achieves an equilibrium state where no current flows. When it is illuminated, photons with sufficient energy will create electron hole pairs, and the radiation is converted to a potential across the cell with usable electrical power. The photon energy required must exceed a band gap that varies with the material used (Figure 5), with any excess energy being dissipated as heat loss and reduced efficiency. Characteristic efficiency/temperature curves were also investigated for Si and GaAs (Figure 6), being the two most commonly used in space, to find an ideal candidate for a nanosatellite. It demonstrates a sensitivity to temperature in Silicon cells but an improved performance in Gallium Arsenide. Radiation damage is another problem to consider in space power systems. Silicon cells have a higher base resistivity (10 Ωcm) offering them significant tolerance to radiation[4]. Research into p-n junction shows that cells having p-type material as the upper region rapidly suffered from radiation damage. Thin cells suffer less than thicker ones, but have a lower conversion efficiency. Gallium Arsenide cells are more radiant tolerant than Silicon for this reason.

Rechargeable batteries have been used extensively for secondary power sources, providing energy during periods when solar is unavailable. It acts as a backup for the primary energy source meaning that it must be able to completely power the CubeSat during eclipse and recharge while in sunlight. Also, spacecraft in Low Earth Orbit (LEO) experience around 800 - 1,000 charge/discharge cycles every year, demonstrating a common low depth of discharge of the battery. A principle factor in selecting a battery is its material composition. The present trend for LEO spacecraft is using Nickel-Cadmium cells for their reliability[1], high energy density and large range of operating temperatures. Lithium batteries have been the subject of heavy research recently in smartphones, so they have been investigated for viability. The two leading types are Lithium-ion and Lithium-polymer. The benefits of lithium batteries are significantly higher energy density than Ni-Cad which is essential for meeting the mission goals in space (Figure 7). Lithium polymer batteries emerge as the ideal fit for a battery used in space due to its high energy density and reliability until 1,000 discharges charges. Lithium polymer batteries tend to have a higher cost as a result and do not suffer from memory effect (reduced performance over time).

The power management unit is required to deliver appropriate voltage-current levels to all loads when necessary. It must operate with primary and secondary power systems whose characteristics vary with time. The electrical bus is required to provide multiple voltage levels to meet the needs of different components on-board. Modern power systems are designed with higher bus voltages (150V) to reduce resistive losses, but it is unnecessary to have this implemented on a nanosatellite. DC converters are still required to provide a variety of voltages, taking place at the equipment level rather than at a central location. Another option is using a hybrid system with both AC and DC buses for improved performance of the latter. It can provide mass savings since fewer components are needed in conversion. An AC distribution is mainly applicable to high-power spacecraft where a large number of DC voltage points are needed. Its use in nanosatellites are also unnecessary since it has a relatively low power demand and number of voltage levels.

1.3 Project Objectives

The scope of this project is to design a system that will reliably generate and distribute power to other subsystems on-board a 2U CubeSat for a 1-year lifespan in orbit. This includes selecting solar panels, battery and power distribution unit that follow a $75,000 (total) and 0.96 kg (allotted mass for power system) financial and mass budgets. The component selection process will use decision matrix analysis to assess cost, long-term reliability, and product capability as prioritized factors.

A power budget will outline the consumption demands of five subsystems under peak and nominal conditions. These values will vary as other teams finalize their component selection over the entire design phase however not all configurations will not comply with the available power. Modelling the power profile of the CubeSat over multiple orbits and under different operational modes will offer assistance in selecting optimal parts that do not exceed energy limitations. This project will include a power profile for five operational modes and will be streamlined to account for internal and external factors such as:

- Beginning and End of Life Efficiencies
- Operating Temperatures
- Change in Orbit
- Battery Degradation

The minimum quantitative goal for the power systems team is to provide a maximum of 8.1 W of energy for sufficient time to gather 100 MB of data, and 6.6 W for two 10-minute periods to transmit the information to an S-Band ground station.

The scope of this project does not include building a functional CubeSat but will instead prototype a representative power system using similar components at a reduced cost. This simulates the power draw and generation while downscaling the cost to allow multiple iterations. This is meant to be a proof of concept to supplement the design associated with the CubeSat so component selection process and power modelling will not be analyzed or included for the prototype.

2. Design Approach

2.1 Concept Generation

The three main aspects that the power systems team explored are the power sources, energy storage, and power management. Previous CubeSat designs were investigated for successful techniques and new viable practices were researched to help make this configuration unique.

To determine the preferred concept of power generation for the S-band CubeSat, ideas were brainstormed from a technical standpoint. The potential concepts include a strictly battery powered CubeSat, a propulsion-based system, and a solar array in tandem with rechargeable batteries. All three of these systems were investigated to address the design requirements of the CubeSat.

Two design candidates were proposed for the energy storage system. First, the idea of a rechargeable battery to supplement the power generated from the solar arrays. In addition, an additional battery bank was incorporated for improved performance because of supplementary mass budget allowances. Second, the idea of a flywheel to generate the power and maneuver the

CubeSat as required[5]. Each system would meet the design requirements of the CubeSat, however, both systems have unique characteristics that would benefit the mission.

To determine the appropriate power regulation and distribution system of the CubeSat, information was exchanged with the Communication and Ground Station teams to obtain a viable solution that would benefit all team's requirements. Two concepts were conceived, requiring further consideration. The first concept combines the power management system with the communication antenna for easier transmission between the CubeSat and ground station. The second concept has the power management system function independently from the antenna, allowing for more freedom in voltage conversions and it saves the inconvenience of tracking mixed budgets between subsystems.

2.2 Concept Evaluation and Selection

A purely battery powered CubeSat would provide several benefits over the other design choices. It would have consistent power unaffected by eclipse conditions. It would also provide adequate redundancy with multiple batteries that would ensure a successful mission even with a battery failure[6]. A constraint with this design is the excess mass of the additional, large batteries would limit other subsystems of the CubeSat. The propulsion-based system would provide the CubeSat with a surplus amount of power that would support all required activities of the mission. However, a limited financial budget disqualifies this option. A solar array in combination with rechargeable batteries would provide the CubeSat with reliable power and a significant amount of redundancy if an additional bank was added. This option has power limitations during eclipse mode hindering the capabilities of the CubeSat by forcing regular power management to maintain adequate charge on the battery (Table 1). With the described design, mission objectives can be reliably achieved for the desired one-year lifespan.

The rechargeable batteries act as the energy storage system and provide the CubeSat with energy. They must have a reasonable mass to not overburden the structure. Over time the batteries experience memory loss effect, limiting the energy available for use[7]. The concept of a flywheel to act as a power source and maneuver the CubeSat would help lower the mass and area required by the energy storage system. The down-side of this method is that it's confined by its inability to turn off and provide specific power when needed. The best option would be to use the batteries as the energy storage system. The ability to flexibly provide precise and consistent power is too valuable to discount for benefits in size and mass budgets (Table 2).

After meetings with project advisors, the combined power management system with the Communication and Ground Station teams was not be pursued due to poor product offerings available to purchase. Therefore, an independent power regulation and distribution system will be used. This system would need to consist of several bus voltages to be able to supply power to all subsystem components as well as the ability to effectively transfer power from the solar arrays to the components and batteries with low internal losses. In addition, the power management system must comply with the size and mass restrictions that were provided by the Structures team.

3. Design Analysis

3.1 Engineering Techniques/Software tools

There are several areas that require engineering techniques and software tools. The first was using Excel and Matlab to model power generation, power consumption during different operation modes and the capacity of the battery. The power requirements under all operational modes were plotted for one orbital cycle (Figure 8). Matlab was used to investigate the power flow time spectrum during short intervals. This is more useful for predicting inrush currents when switching modes. Excel was used to model the power system for multiple concurrent cycles to monitor the effects of changing operational modes on the battery.

A second example uses techniques learned from classes and laboratories to develop a functional prototype. These techniques include knowledge of PCB design, soldering electronics boards, using C++ in the Arduino code (Figure 22) to measure output voltage, and applying Kirchhoff's current and voltage laws to real circuits while debugging problems for a verified solution.

A circuit simulation software, called MicroCap, was used for planning the electrical schematic (Figure 9) of the prototype. The software offered common electrical components in the libraries, however the parts being used in the design were too specific to find so placeholders were used to represent them in the schematic. Had all components been accounted for, the program would simulate the steady state and transient characteristics of the circuit. This would be highly unnecessary for the prototype but useful for the full electrical build of the CubeSat.

A renewable energies course exposed all of the team to the details of optimized solar power generation such as ideal mounting configurations. This helped to bypass a lot of initial research because there was already an adequate foundation of knowledge concerning solar panel material and efficiency ratings. Knowledge of trigonometry helped tremendously while determining the ideal angle of incidence of solar panels exposed under the sunlight for maximum power generation.

3.2 Complete Analysis/Calculations

To start the power generation analysis for this mission, two limiting factors from the mission requirements were investigated. The low earth orbit (LEO) nature of the orbit, and the dimensions of the Spirit 2U CubeSat (20x10x10cm) are both fixed parameters that affect power generation.

In LEO there is an average available power from the sun of 1,368 W/m^2 and by choosing the triple-junction GaAs cells with an efficiency of 30%, an expected output power P_o from the solar cells is[10]:

$$P_o = 0.3 * 1368 = 410.40\ W/m^2$$

The available surface area on the CubeSat is two smaller sides that will hold 4 solar cells of 27cm^2 each giving a total of 2x108cm^2 (Figure 10) and the two full sides will hold 5 of the same solar cells giving a total of 2x135 cm^2 (Figure 11).

To study the power generation profile of the CubeSat, calculating the beginning of life (BOL), end of life (EOL) and the expected power generation facing the sun offers a starting point to begin modelling more complex behaviour. BOL power generation is defined as:

$$\mathbf{P_{BOL}=P_o * I_d * \cos(\theta)}$$

where $I_d = 0.77$ is the inherent degradation due to the assembly of solar cells to make an array, and θ is the cosine loss[10]. The EOL power generation is defined as:

$$\mathbf{P_{EOL}=P_o * L_d}$$

where L_d is defined as the performance degradation of the solar cells per year, for triple-junction cells it is 0.5%, so $L_d = (1-0.5\%)^1 = 0.995$ for one year of mission life[10].

The total power generation is defined as:

$$\mathbf{P_{sa}=A_{sa} * P_{EOL}}$$

where A_{sa} is defined as the surface area facing the sun.

Table 3 summarizes the worst-case and best-case scenarios possible, as the CubeSat will be rotating in a barbecue roll to expose all the solar cells to the sun equally. In the worst-case scenario, we will be having only one camera side with solar cells area of $0.0108m^2$ facing directly the sun at θ = 0° deg perpendicularly. While in the best-case scenario, we will have one camera side and one non-camera side with a total solar cells area of $0.0243m^2$ both facing the sun at an angle of θ = 45°. By using the average of power generated in both scenarios a power generation profile was constructed by modelling the CubeSat rotation around its own axis (Figure 8).

3.2.1 Power Consumption

Based on the information provided by the Spirit subsystem teams regarding their power requirements the power consumption was calculated under different operational modes (Table 4).

During communication and imaging modes, battery power will be required due to a net loss between generation and consumption. A double bank rechargeable battery of 40Whr was integrated to offer more flexibility for extended mission life.

Using Excel, the battery state of charge, power generation and net power consumption of the CubeSat are investigated under different operation modes. The amount of power could be generated over four orbits in nominal mode (Figure 12) to charge the battery from 30% to 51.1% the CubeSat needs to be in nominal mode for four orbits.

The CubeSat was modelled during an imaging mode scenario to investigate its handling of high power consumption assuming 80% of maximum power generation as a margin of safety (Figure 13). The first orbit takes two pictures and a 5 minutes video, the following two orbits transmit data for the maximum available 10-minute interval and the last orbit is in nominal mode for recovery. Starting with an initial capacity of 70%, the system was able to charge the battery to 77% for a net gain of 7% despite taking two margins of safety on the starting charge and power generation. In nominal mode, the battery experiences a net energy gain while in view of the sun. Over four orbits (360 minutes) the battery charges approximately 20%, demonstrating that the CubeSat would be able to remain idle in nominal mode despite receiving 80% of maximum power generation. There is a net power gain for the 10-minute transmission window as well, showing that it will also have a negligible effect on the battery charge assuming reliable power generation.

3.2.2 Detumbling Scenario

Early in the modelling process, the team investigated the efficacy of the CubeSat taking video immediately after being deployed from the International Space Station. This process would require authorization from the Canadian Space Agency so before pursuing further, the scenario was modelled to evaluate how long the CubeSat could remain in imaging mode and the time required for the battery to recover.

According to the internal specifications of the camera being used on the Spirit CubeSat, there is 1 GB of internal flash storage space. This allows for approximately 16 minutes of 720p video to be taken. Also, while detumbling the CubeSat will be moving in an erratic behaviour and it is unreliable to quote a large amount power generation during this phase. The initial battery charge in practice will start at 100%, however for modelling purposes it will begin at 70% to accommodate for a large margin of safety. In the Excel model of this scenario the

CubeSat is able to take nearly 80 minutes of video (500% of the capable on-board storage) over two orbits. After the first two cycles it is assumed that the CubeSat slowly stops detumbling in space and normal power generation begins at 1/3rd the second orbit. After taking video the battery is at approximately 50% charge and requires 3.6 days to recharge to 80% (Figure 14).

The scenario should be pursued during the actual launch because despite the three margins taken around the model, there is still 20% battery remaining before reaching the critical depth of discharge (30%).

3.2.3 Power Generation Dependency

For further investigation into the CubeSat's reliability in the long-term, fractions of the total power generation were modelled to find when the battery would begin to consistently lose charge. The model assumes a starting battery capacity of 30% to simulate an emergency scenario, 50% of maximum power generation and has the system in the nominal operational mode for four orbits (Figure 15). Under these circumstances, the CubeSat loses 7% of battery capacity every four orbits and would be completely drained at the end of one full day (16 cycles). This offers little opportunity for a ground station to respond with commands so it is recommended that the on-board computer also have a safety measure to monitor power generation as well as battery capacity.

3.2.4 Emergency Safe Mode

The purpose of safe mode is to provide an emergency state that the CubeSat enters if the battery capacity drops below 30%. It commits the CubeSat to only vital functionality, which is the On-board computer. This severely cripples the capabilities while in orbit but allows time for the ground station to review data and debug any issues. The CubeSat will only make contact with the ground station once its battery reaches 40% capacity. This case was modelled to set an expected time before the CubeSat would make contact with the ground station. The initial conditions were a starting charge of 10%, and a power generation of 80% the maximum (Figure 16). The model demonstrated that it would take approximately 2.5 days before ground station would be able to make contact with the CubeSat at 40% battery, and over 5 days before reaching maximum capacity.

3.3 Design Documentation

First of all, all the devices built into the CubeSat have to be CSA (Canadian Space Agency) approved which required passing a series of tests. The power system of CubeSat design starts with thorough analysis of low earth orbital conditions, operation parameters, the power generation deviations and the load consumption estimations. Eclipse duration, orbit inclination and location determined the optimal solar panel material to be Gallium Arsenide. Maximum efficiency for GaAs triple junction solar cell can reach 33.5%, comparing to silicon's 26.3%. Because of the higher efficiency GaAs solar cell owns superior flexibility and size advantage. The structural team implant the camera in the middle (Figure 17), solar panels have to be customized to fit the CubeSat's customized structure requirements.

For the fabrication plan, each solar cell needs to be mounted individually with kapton tape and silver epoxy applied to the back to ensure electrical conductivity. Industry standard requires the cells to be placed into cover-glasses to minimize the degradation by the solar radiation in low earth orbit. Limitations of the solar panel are the degradation leads to potential power loss at the end of life and it accumulate cosmic dust over time with both being unpreventable.

Battery selection is due to mission length and mass budget restriction. The ideal design is using Lithium Polymer battery instead of Lithium Ion. Lithium Ion battery is significantly cheaper but has a memory effect deficiency (harder to charge when time passes on). Lithium Polymer battery has a lower chance of leaking electrolytes and the appropriate fabrication method is to determine the incoming energy from the solar panels, develop an accurate power budget and balance out the overall power requirements for all the operating modes. The battery for the actual CubeSat contains a built-in power protection system, but for the prototype a battery management system protection board has to be integrated separately to protect over and under current situations. Limitations and constraints for the battery is unregulated battery temperature that will be posed a risk to shorten the battery life if heat flow is not properly managed.

Power distribution fabrication process ensures the input voltage is regulated and build in DC-DC booster to convert desired voltage levels. Most of the power distribution boards have their own under and over protection as well in order to fulfill the purpose of monitoring the power surge in space. The limitation is the CubeSat devices are often functioning in 3.3V, 5V and 12V, to supply some abnormal voltage levels; power distribution board has to be customized and it is very costly. Because of the budget limitation, the prototype's power management and distribution board consist of a DC-DC booster and current regulator which are operating at different voltage level than the CubeSat's.

4. Results and Validation

4.1 Prototype

The main principle behinds the prototype is to design a model that mimics the power flow inside the CubeSat. This requires some additional equipment incorporated in the prototype design that can assist with representation of the CubeSat's function (Table 5).

The prototype consists of two 6V car solar panels, three 3.7 V rechargeable lithium batteries, a boost DC-DC converter, a MAX745 3S lithium charging board, an arduino board and a phone charger. These devices were selected as a cost effective method to build a proof of concept. The purpose of using two series mounted 6V solar panels is to generate sufficient power to charge the battery while also being relatively the same size (4 x 5 inches) as implemented on the actual Cubesat. A lithium charging board regulates current to ensure the battery will receive a constant power by adjusting the input current range to a desired level. A DC-DC converter is used to amplify the input voltages primarily to support the lithium charging board that operates between 14.6V and 20V. This is also representative of the DC to DC converter inside the CubeSat which manipulates voltage generated by solar cells to the ranges of 3.3V-4.2V and 5V-9V. The battery pack is a combination of three rechargeable lithium battery cells in series, a 3-unit holder, and a battery management system (BMS) protection board. The protection board and current regulator are an inexpensive solution for mimicking the power flow of the actual satellite.

The lithium ion battery cells are rechargeable, have a capacity of 3000 mAh and the holding case allows more flexibility to test and solder various configurations. The BMS protection board prevents over-charge/discharge of batteries, short circuit, and overcurrent situations. Compared to other types of batteries, the cost of lithium ion battery is also one of the cheapest making it an ideal solution for prototyping. The entire system will simulate a similar setup to the CubeSat's where loads are prioritized to be powered by generated energy in the solar panels and any excess charges the battery. When loads exceed the generation capabilities, battery power is used to supply the difference.

Over and under charging has multiple redundancy in the prototype because on the actual CubeSat any damage to subsystems is irreversible and could cause EOL, so it is emphasized in practice. The BMS protection board and Arduino board mimics a power management and distribution system. The Arduino board will be sent terminal voltages of the three batteries to the computer, calculating the percentage of power remaining. The alternative would be to purchase another distribution board to measure the voltage after the load however this approach adds cost and would be difficult to solder.

In terms of a reliable and cheap DC power consuming source, a simple phone charger can be used as the primary load resistor as it draws a constant 12V from the battery and is easy to solder due to its plug that neglects any grounding wire. By building a prototype we can further investigate potential difficulties that will be encountered during the final design of the CubeSat.

4.2 Testing Strategy/ Validation Protocols

To test the prototype, the power flow cycle was investigated to be an accurate representation of the system on-board the CubeSat. A UV light simulated the Sun source due to the inconvenience of presenting indoors. Power will then be generated through the solar panels mounted on the prototype chassis and is validated by being able to provide sufficient power to supply the loads. This tests our system at a broad scope for mimicking the power flow from a source to load. An accurate readout of the real-time power draw or supply to the battery can be calculated through the Arduino board will also supplement.

To be able to verify the accurate use all components, a portable multimeter was used to measure the voltage and current at various points where the Arduino board cannot access. This primarily addresses the points before and after the DC-DC converter and current regulators.

Phone chargers were selected as the ideal load because they draw sufficient power to see changes live during the demo. Comparing the phone charger to other loads, such as LED lights, are too efficient to become high power consuming devices and also require a voltage divider to step down the input voltage to 4.5V making any loss of power difficult to track. A 3rd party application was download to monitor the current draw to the phone (in mAh) for verification that the load was drawing power.

The prototype design was not a perfect representation of the CubeSat. For instance, the solar panels generate DC voltage but due to object's nature to rotate in space, solar panels have different angles that are exposed under the sunlight. This results in a varied power generation throughout the orbit and was not considered during testing. An issue encountered while testing was the battery management system has the ability to shut down input current if any one of three battery's voltage drops below 2.4V. This is a symptom of having three independent batteries connected in series with embedded logic. The CubeSat battery banks would not experience the same issue in space,

Overall, all of the components are scaled down in terms of price compared to the on-board versions so testing the prototype for flight-like scenarios such as heat shock testing is unrealistic. However, the organization of our circuit is not embedded inside the physical chassis but rather displayed for easier visual sight of the components used (Figure 18).

4.3 Final Results and Validation

The testing results obtained from our design resonate with the objectives of the CubeSat project. Due to the limitations of being unable to test the real spacecraft components at this point in the project, we were able to successfully design a proof of concept that mimics the function of the electric power system onboard the CubeSat.

To verify power flow to the phone charger load, a readout of voltage via Arduino board/code combination (Table 6) and current draw through phone application were used. Afterwards power could be calculated using equation $P = VI$ to verify if the power received is reasonable. For the 30 minutes of measurements, the voltage and current were reasonably regulated to a small range. This was expected due to the components selected to ensure the load received a certain voltage. There was a maximum variance of 0.09 V and 54 mA, demonstrating the accuracy of the prototype. The power system provided an average of 11.4W for 30 minutes (5.7 Whr) with the majority of the energy being sourced from the batteries (Figure 19). The solar panels provided a maximum of 2W and the batteries 33.3W total, so this experiment could have been continued for 3 hours before the batteries would be completely drained.

Overall, the results show that the solar panel and battery combination could supply a DC load similar to on-board the CubeSat. The major limitation of this prototype test was that it assumed the same voltage at the point of the load and the Arduino board. It is reasonable to consider that the difference between these two values is low due to low internal losses in the wire.

5. Conclusions, Future Work and Recommendations

The design of the electrical power system for the Spirit CubeSat has the ability to produce sufficient power to fulfill the mission objectives. The original design goals of the project were to supply power to take photos and videos of the Earth and transmit the data back on a regular basis, with a 1-year lifespan in mind. With our design (Figure 24-26 and Table 9) of the electrical power system, sufficient power is available to take photos and videos regularly without significant strain on the battery with proper management of operational modes. Additionally, transmission with the ground station has a net positive gain in power signifying that the system will be bottlenecked by the communication window and not available power.

The advantages of this power system configuration are that the system is efficient in producing power as there is less than 10% power loss. The build is inherently lightweight, allowing the system to double the battery capacity to 40Whr and have excess mass for future considerations (Table 7). The components selected are well below the financial budget allocated to the electrical power system team (Table 8). The limitations of the design are that the solar panels have an inherent risk of accumulating cosmic dust and its reliance on attitude control to constantly position in the direction of the sun. Both factors can limit power generation capabilities with minimal solutions in place.

Furthermore, if the system is unable to produce power inside 65-100% of expected power generation, the CubeSat will become unstable due to the lack of power generated and the battery will drain in two days of nominal mode.

The CubeSat project offered insight from the intensive strategy and modelling required to design a system for a space vehicle. Additionally, the experience working within strict regulations set by the Canadian Space Agency helped expose the team to the difficulties of working within a highly regulated industry would entail. Overall, working on a project that will eventually produce a space vehicle provided the team with invaluable experience along with a better understanding of space systems.

Moving forward, our proposed design and modelling will be given to the CubeSat project's senior team members whom will continue next year. During the summer months, a decision for using the S-band or UHF design will be made after considering the work both teams have submitted. If S-Band is chosen our design and modelling will be passed onto the incoming CubeSat team to build and test. In the future, the CubeSat project will transition from the design stage to building and testing. By the end of 2020, it is anticipated that the CubeSat will be mission ready so that it may be transferred to the Canadian Space Agency for deployment.

There are two recommendations theorized for improving the electrical power system. First, the addition of redundancy would greatly benefit the CubeSat mission as there is no room for failure in space. Specifically, the addition of another low capacity battery could be used as a backup using the excess 200g of mass budget would provide adequate redundancy. This additional battery can be attached to the power management board for simplicity and does not need to be rechargeable. However, this addition may cause centre of mass issues for the structures team, so further investigation would be required. In the event of failure of the primary battery bank, the second battery would allow the mission to continue and potentially debug the original issue for a resumed mission rather than immediate EOL. Second, the placement of the boards for the electric power system on-board the CubeSat should be as close together as possible to limit the length of wires needed. This recommendation reduces the length of wire required and would ultimately reduce the weight of the electrical power system. Additionally, the smaller length wires would reduce the power lost due to internal resistance and the parasitic capacitance found between components. While the energy loss due to wire length would be minimal, maximizing efficiency is crucial in a low power budget. With these recommendations the Spirit CubeSat will have an improved performance in completing its mission objectives.

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7. Figures and Tables

Figure 1 - Common Nanosatellite Layout

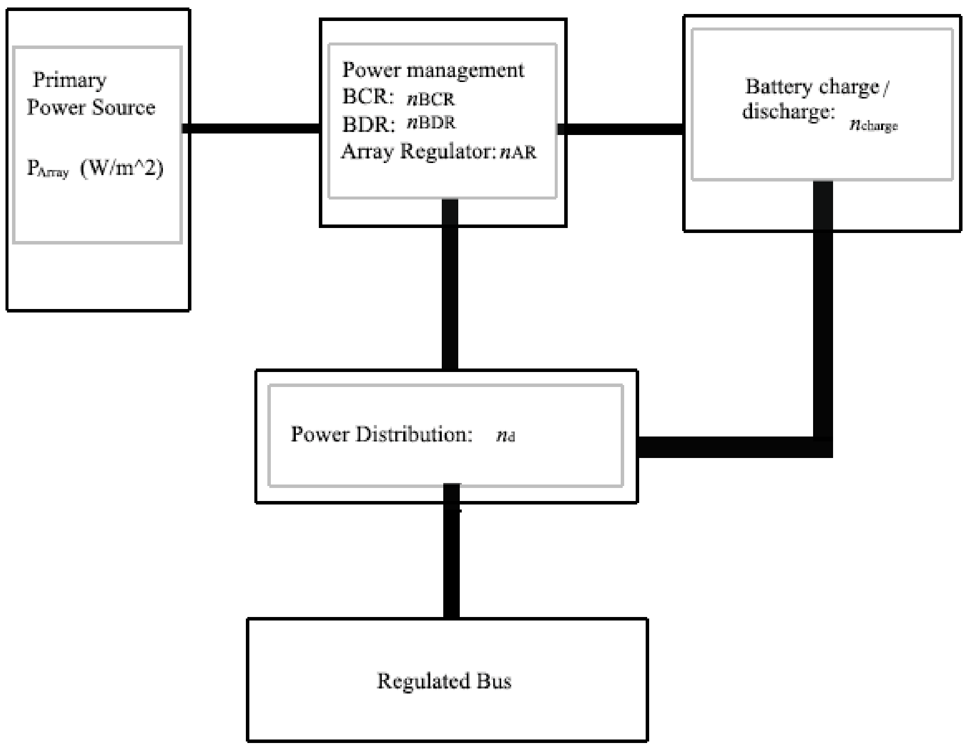

Figure 2 - Schematic of Typical Spacecraft Power System

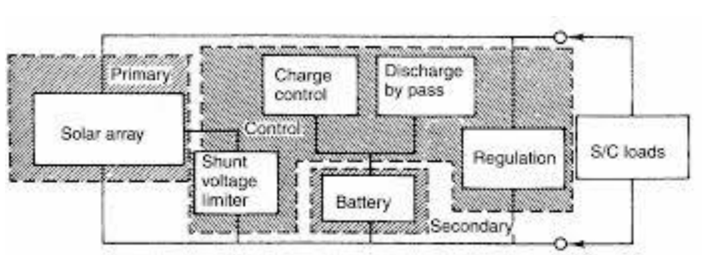

Figure 3 - Radioisotope Thermoelectric Generator Diagram

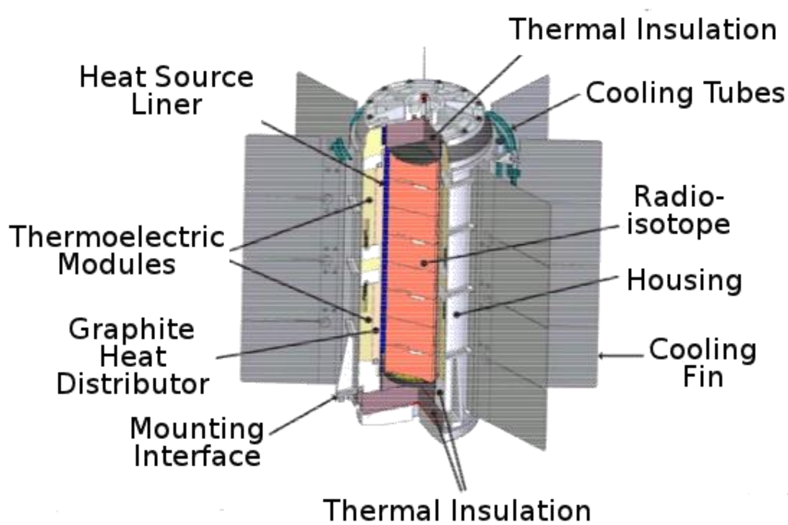

Figure 4 - Schematic of a Typical Solar Cell

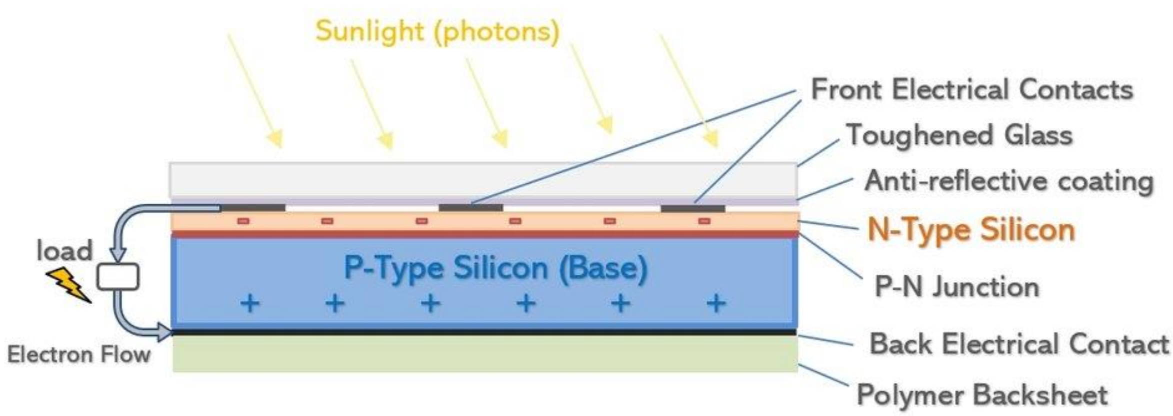

Figure 5 - Properties of Semiconductor Materials

Material	Band Gap (eV)	Maximum Wavelength (µm)
Si	1.12	1.12
CdS	1.2	1.03
GaAs	1.35	0.92
GaP	2.24	0.554
CdTe	2.1	0.59

Figure 6 - Theoretical Cell Efficiency as a Function of Temperature

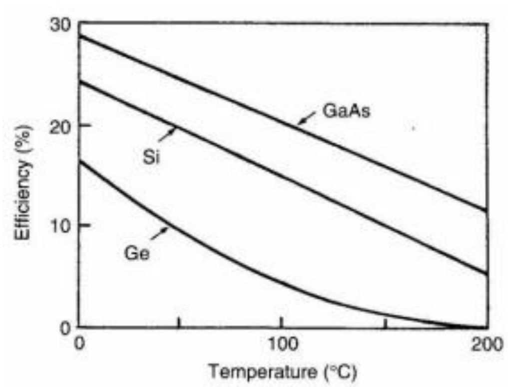

Figure 7 - Battery Performance Comparison

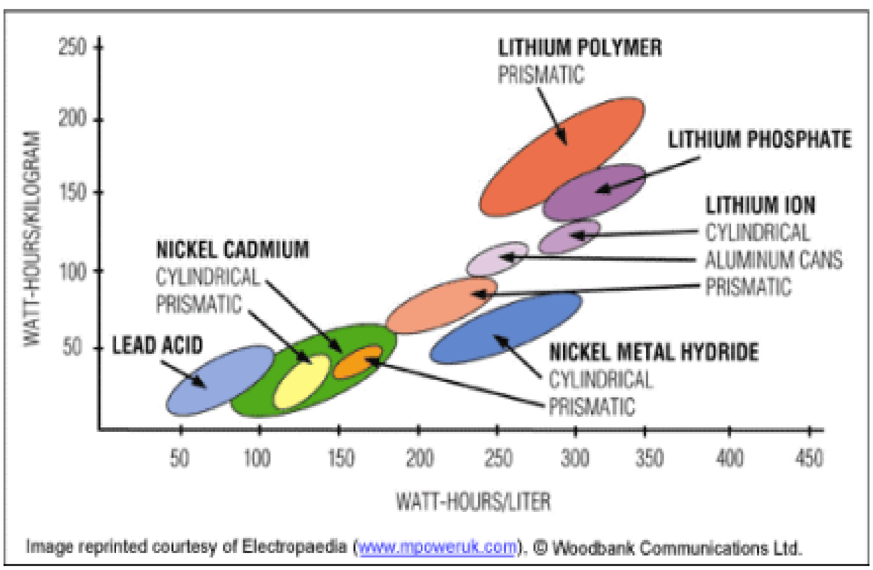

Figure 8 – Power Generation Over One Barbecue Roll Rotation of CubeSat

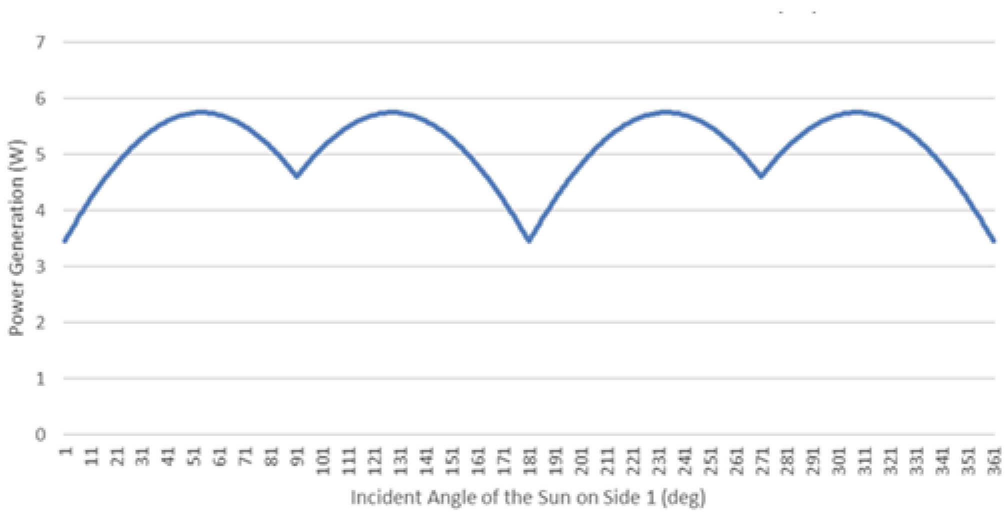

Figure 9 -Planning Schematic of Prototype

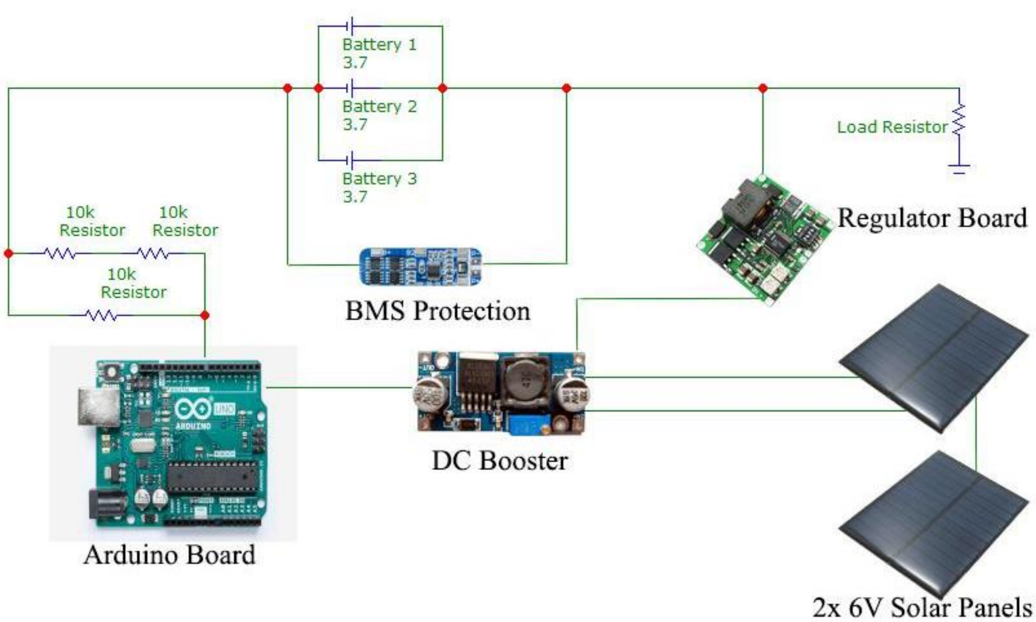

Figure 10 – Side of CubeSat with Reduced Surface Area and 1U Solar Panels

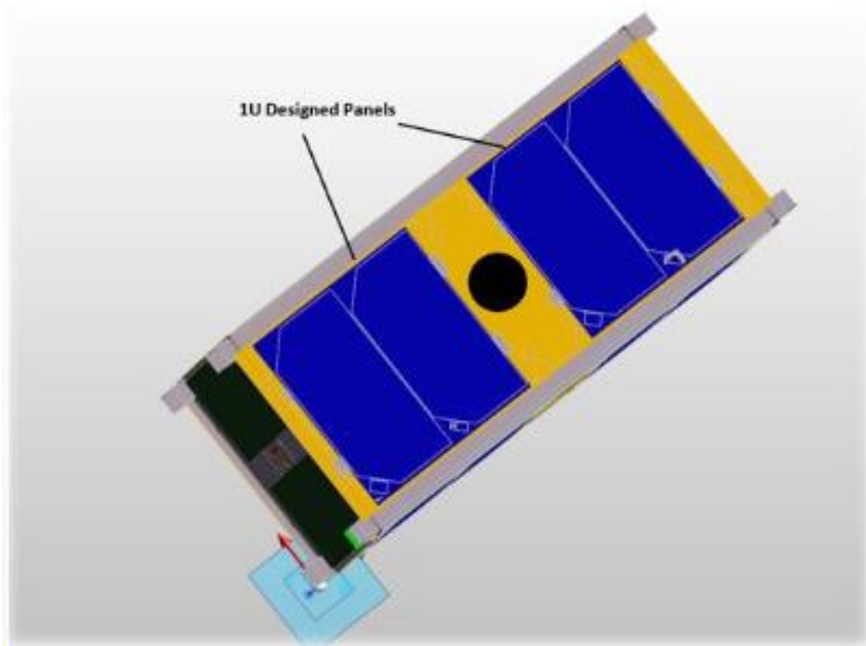

Figure 11 – Full Side of CubeSat with 2U Solar Panel

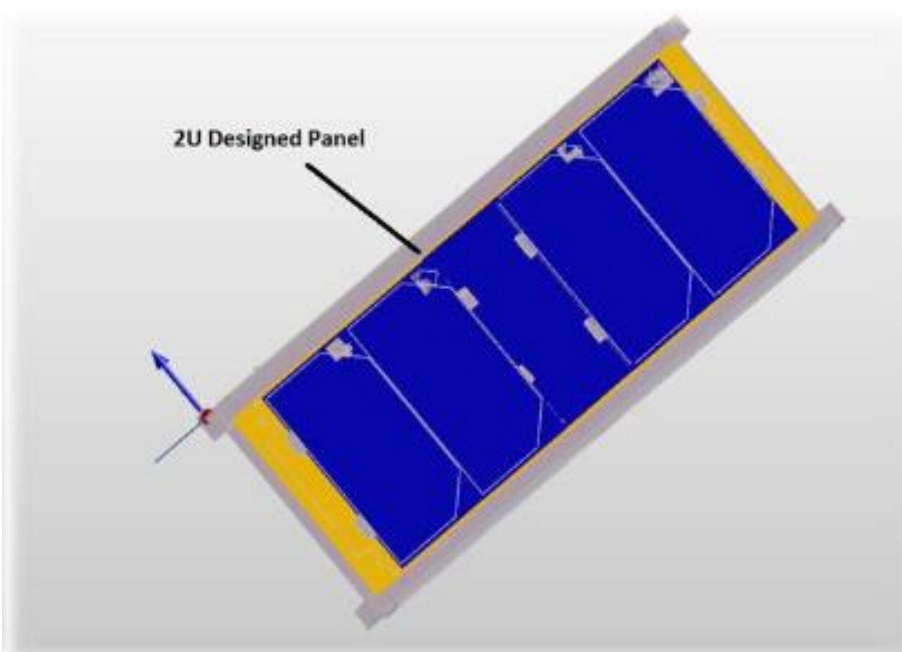

Figure 12: Power Generation During Nominal Mode

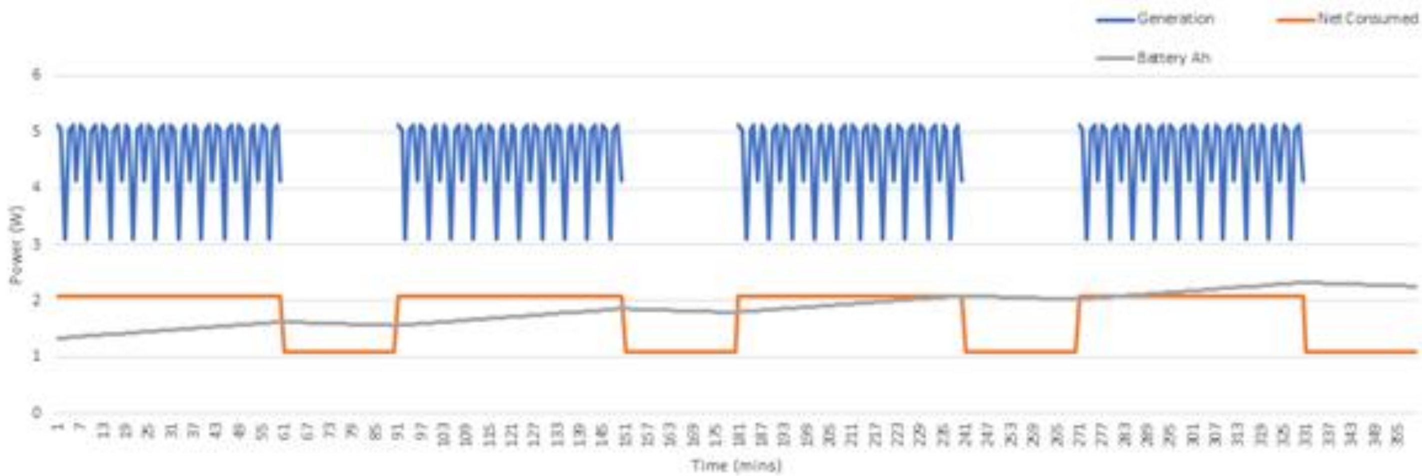

Figure 13: Power State After Imaging and Communication Modes

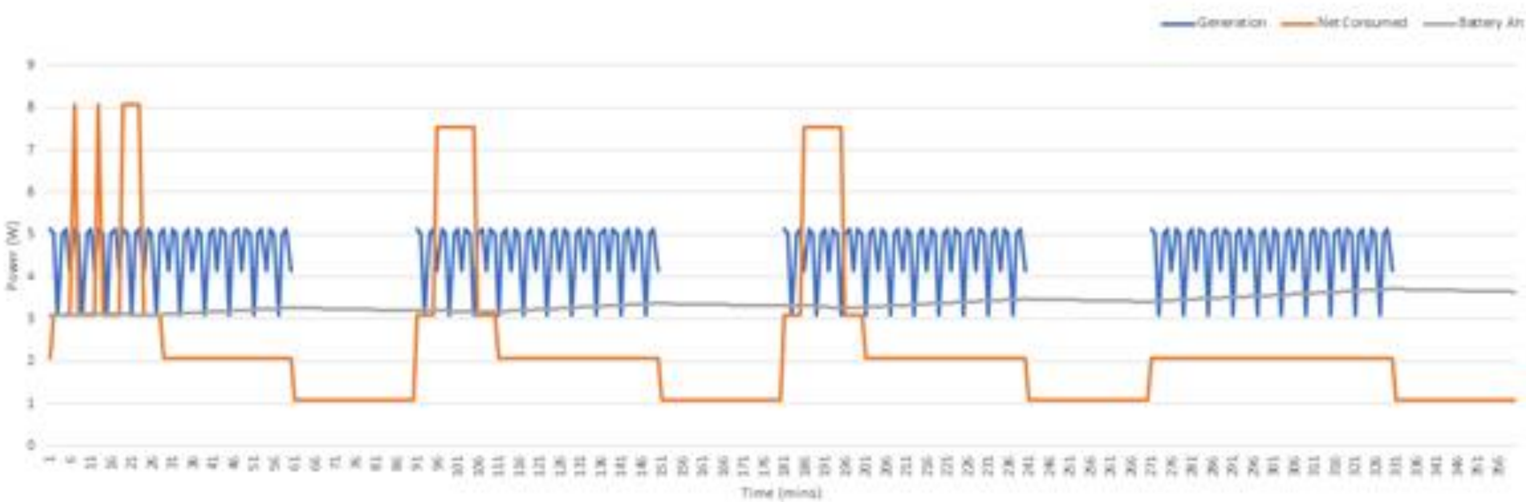

Figure 14 - Detumbling Scenario with Video Upon Deployment

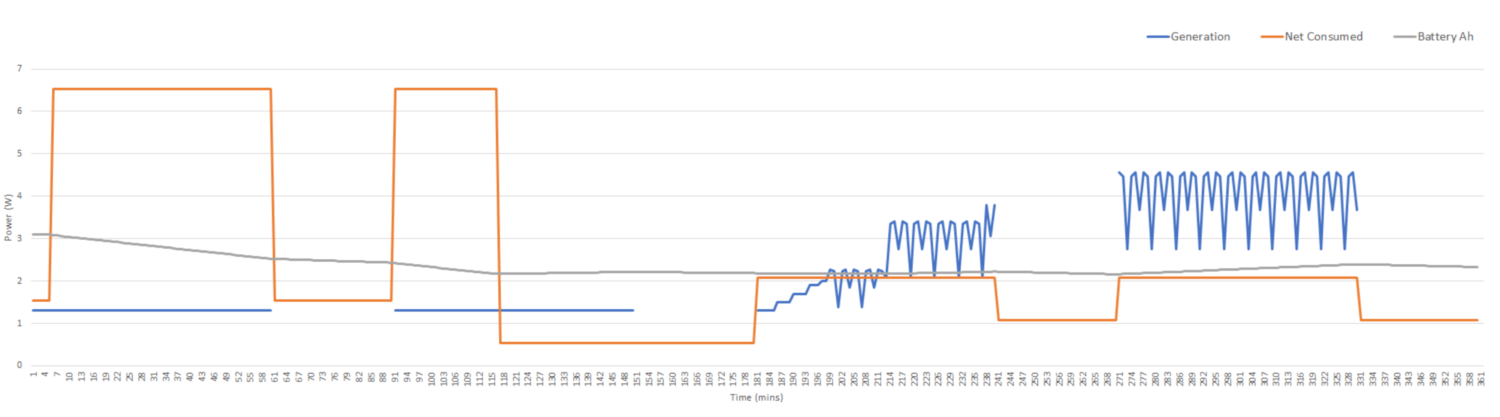

Figure 15 - Nominal Operation Mode for 50% Power Generation

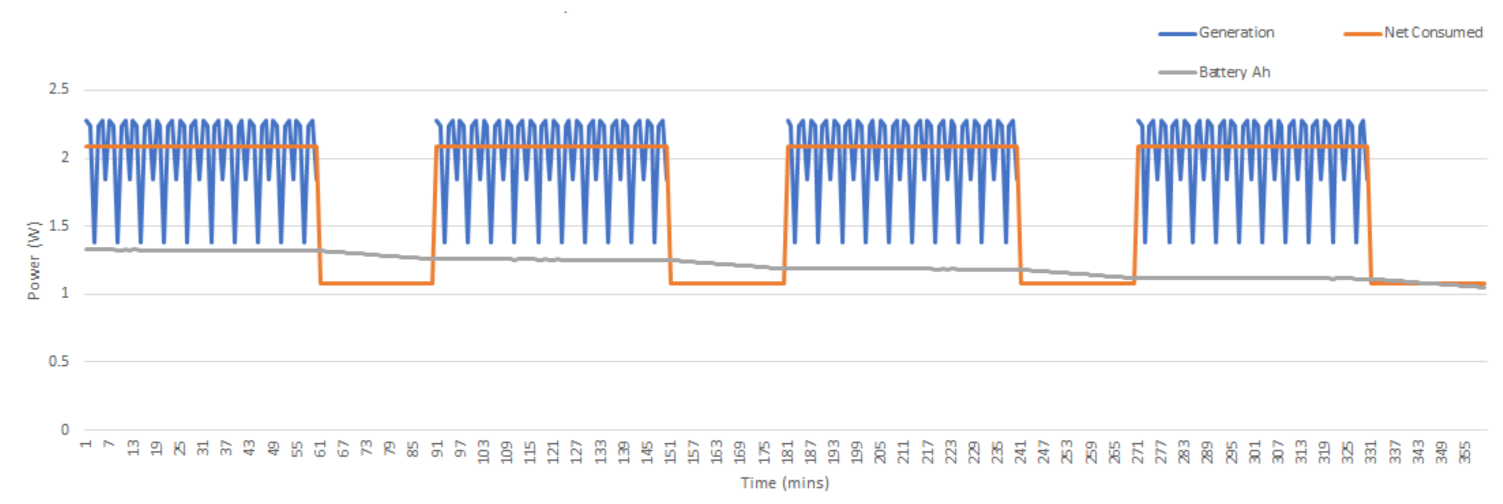

Figure 16 - Safe Mode Recharging from 10% Battery Capacity

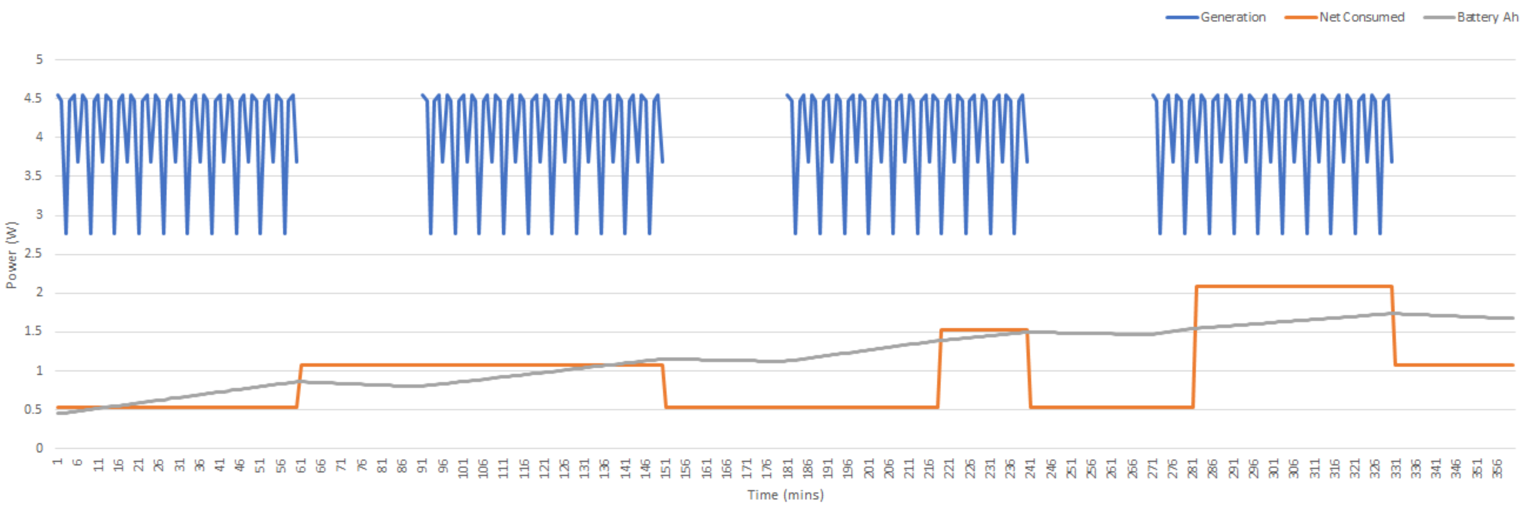

Figure 17 – Mechanical CubeSat Prototype

Figure 18 - Final Power System Prototype

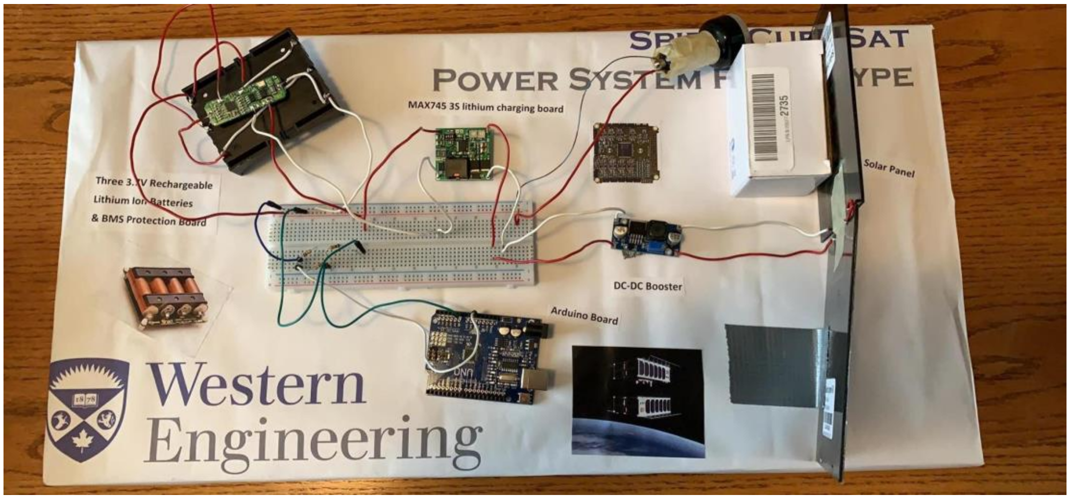

Figure 19 - Prototype Power Output in Half Hour

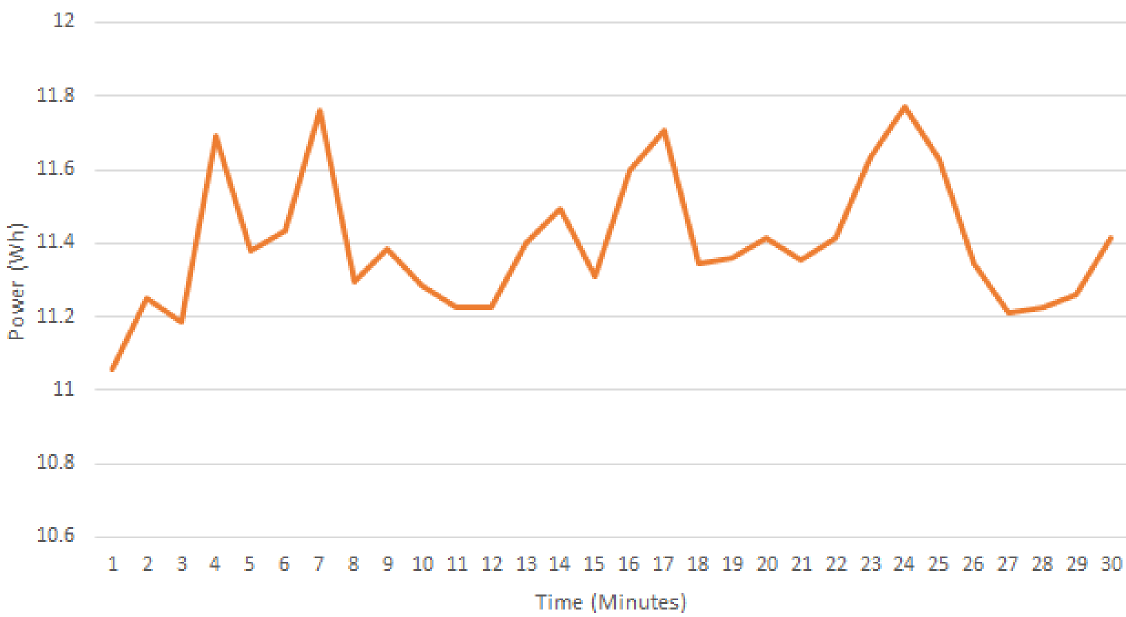

Figure 20 – Gaant Chart

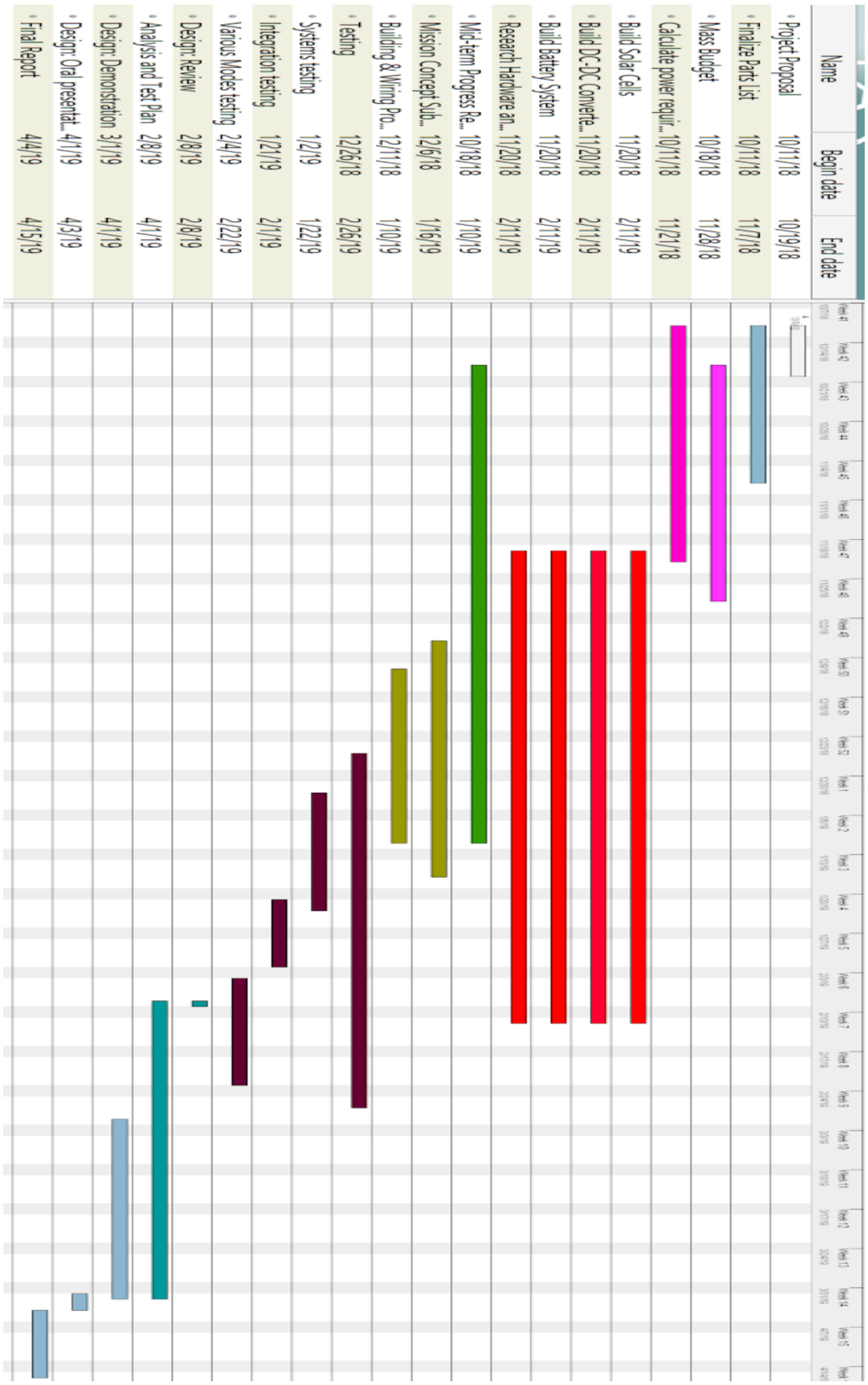

Figure 21 – Subsystem Interaction Flow Chart

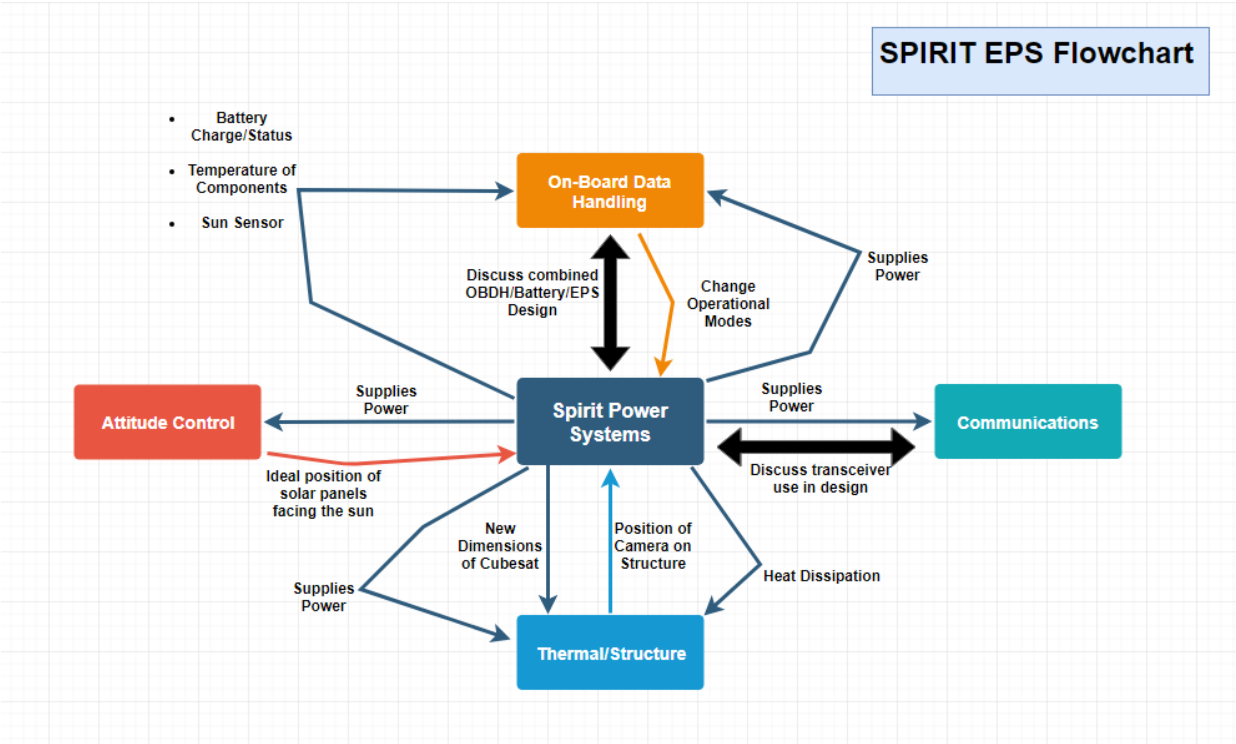

Figure 22 – Arduino C++ Code

File Edit Sketch Tools Help

run_test_feb24 §

```
//Input Pin
int analogPin = 0;

void setup() {
  // put your setup code here, to run once:
  Serial.begin(2400);
}

void loop() {
  // put your main code here, to run repeatedly:
  // read the raw data coming in on analog pin 0:
  int voltageOUT = analogRead(analogPin);
  // Convert the raw data value (0 - 1023) to voltage (0.0V - 12.0V):
  float voltage =  voltageOUT*(5.0 / 1024.0)*6.04; //maximum voltage from batteries is 11.1V, converted from binary
  float currentJ = 1.10;
  float powerJ = voltage*currentJ;
  // write the voltage value to the serial monitor:
  Serial.println("Voltage: " + String(voltage) + " V");
  Serial.println("Power(W): " + String(powerJ) + " W");
}
```

Figure 23 – Solar Panel Website Snap Shot[14]

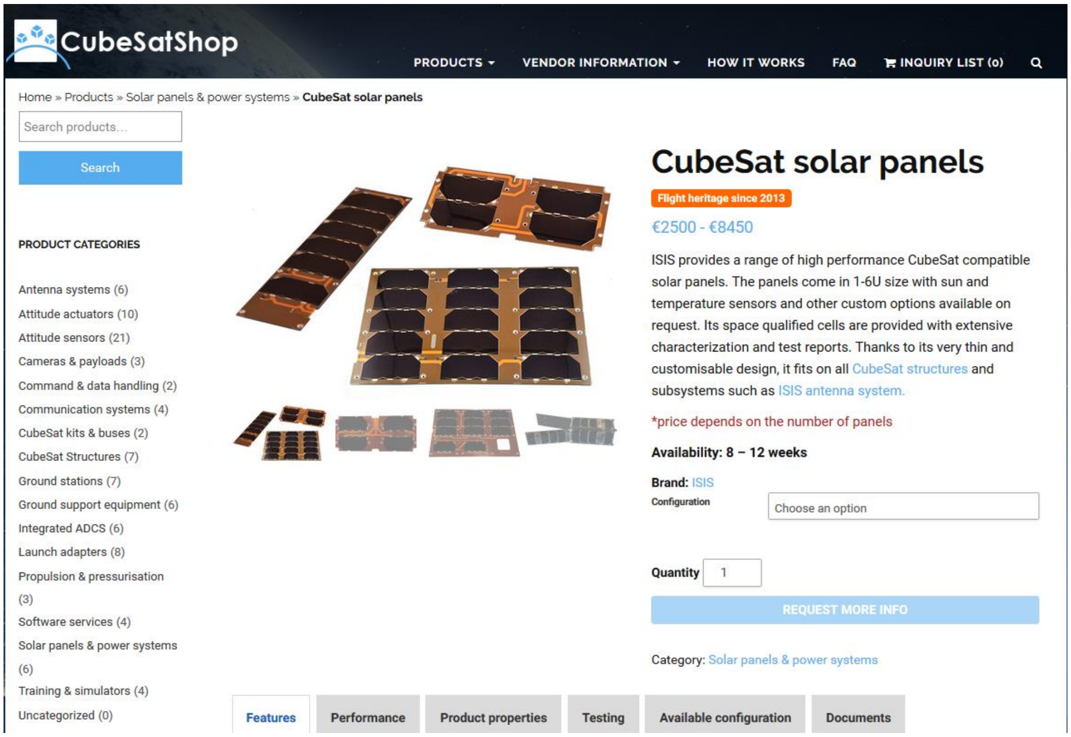

Figure 24 – Power Management System Website Snap Shot[15]

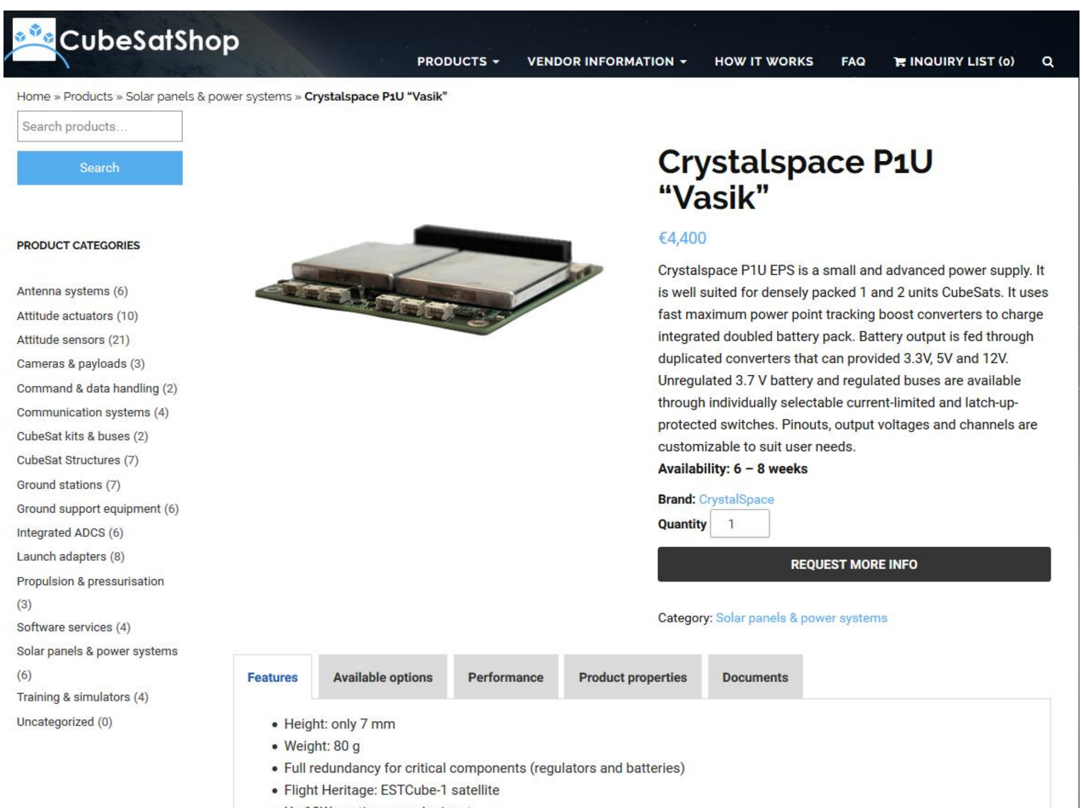

Figure 25 – Battery Bank Website Snap Shot[16]

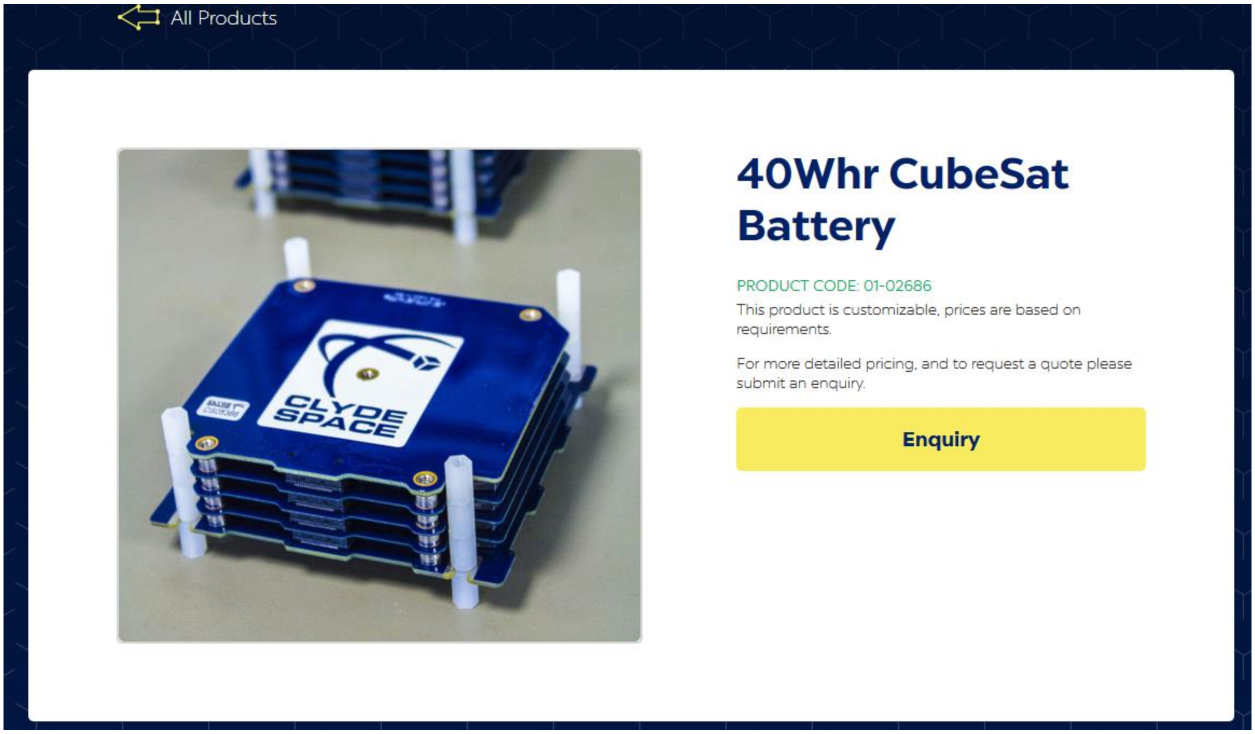

Table 1 - Power Generation Decision Matrix

(5 being ideal)	Power Generation (1-5)	Cost (1-5)	Practicality (1-5)	Redundancy (1-5)	Total
Criteria Rating	**3**	**2**	**4**	**2**	
Battery Powered	3	3	2	4	
Weighted Rating	9	6	8	8	**31**
Propulsion System	5	1	1	2	
Weighted Rating	15	2	4	4	**25**
Solar Array and Batteries	3	4	5	3	
Weighted Rating	9	8	20	6	**43**

Table 2 Power Storage Decision Matrix

(5 being ideal)	Storage Capacity (1-5)	Cost (1-5)	Practicality (1-5)	Redundancy (1-5)	Total
Criteria Rating	**3**	**2**	**4**	**2**	
Batteries	3	3	4	4	
Weighted Rating	9	6	16	8	**39**
Flywheel	1	3	2	2	
Weighted Rating	3	6	8	4	**21**

Table 3 – BOL and EOL Power Generation Calculations

	Worst-case scenario	Best-case scenario
Power output Po (W/m^2)	**410.40**	**410.40**
Inherent degradation I_d	0.77	0.77
Sun incidence angle θ (cosine loss)	0	45
BOL Power (W/m^2)	**316.01**	**223.45**
Life degradation L_d	0.995	0.995
EOL Power (W/m^2)	**314.43**	**222.33**
Solar cells area facing the sun (m^2)	0.0108	0.0243
Power Generation	**3.40 W**	**5.40 W**

Table 4 - Power Budget For Operational Modes

		Nominal	Communications	Imaging	Safe	Eclipse
Subsystem	**Component**	**Power (mW)**				
COMMS	Transciever	550	5000	550	0	550
C&DH	On-Board Computer	442	442	442	442	442
ADCS	Magnetorquer (3)	600	600	600	0	0
	Sun Sensor (3)	390	390	390	0	0
	magneometer	1	1	1	0	0
	Gyroscope	12	12	12	0	0
Payload	Camera 1	0	0	3000	0	0
	Camera 2	0	0	3000	0	0
EPS	Battery	89	89	89	89	89
Total		2084	6534	8084	531	1081

Table 5 – Prototype Component Budget List

Component	Cost ($CAD)
MAX745 Lithium Battery Charging Board	$ 10.00
Battery	$ 20.00
12V Car Solar Panel	$ 60.00
BMS Protection Board	$ 10.00
3 Pack Battery Holder	$ 5.00
XL6009 DC-DC Step-up Adjustable Buck Boost Power Converter Board	$ 5.00
Arduino Kit from Electronics Shop	$ 50.00
Total	$ 160.00

Table 6 - Measurement Data from Prototype Verification

Time (Minutes)	Voltage (V)	Current (mAh)	Power (Wh)
1	11.84	934	11.05856
2	11.84	950	11.248
3	11.8	948	11.1864
4	11.75	995	11.69125
5	11.84	961	11.37824
6	11.85	965	11.43525
7	11.76	1000	11.76
8	11.84	954	11.29536
9	11.82	963	11.38266
10	11.84	953	11.28352
11	11.84	948	11.22432
12	11.84	948	11.22432
13	11.8	966	11.3988
14	11.81	973	11.49113
15	11.84	955	11.3072
16	11.75	987	11.59725
17	11.79	993	11.70747
18	11.84	958	11.34272
19	11.82	961	11.35902
20	11.83	965	11.41595
21	11.84	959	11.35456
22	11.79	968	11.41272
23	11.76	989	11.63064
24	11.75	1002	11.7735
25	11.79	986	11.62494
26	11.84	958	11.34272
27	11.84	947	11.21248
28	11.83	949	11.22667
29	11.84	951	11.25984
30	11.78	969	11.41482

Table 7 - Electrical Power System Mass Budget Allotment

Component	Mass (g)	Percentage of 960 g
Solar Panels (2x 2U, 2x 1U)	300	31.3%
EPS Distribution Board	80	8.3%
Battery Bank	335	34.9%
Sum	715	74.5%

Table 8 - Electrical Power System Financial Budget Allotment

Component	Cost ($CAD)	Percentage of $75,000
Solar Panels (2x 2U, 2x 1U)	$17,700	23.6%
EPS Distribution Board	$6,700	8.9%
Battery Bank (x2)	$3,900	5.2%
Sum	$28,300	37.7%

Table 9 – Selected Component Properties

Component	Composition	Efficiency	Power Rating	Mass (g)	Dimensions (mm)	Operating Temperature (°C)	Operating Voltage (V)	Cost ($CAD)
2U Solar Panels	GaAs	30.0%	4.6 W	100	82.5 x 196 x 2.4	-55 to 125	3.0	$5,900
Battery	Lithium Polymer	95%	40 Whr	335	95 x 90 x 17.25	Charging: 0 to 45, Discharging: -20 to 60	7.6	$3,900
EPS Distribution Board	Copper, Radiation hardened	90-95%	3Ah/ 11 Wh for battery	80	96 x 90 x 13	-40 to 85	3.3, 3.7, 5.0, 12.0	$6,700

Table 10 – Gaant Chart Reference Table

Task Name	Start Date	End Date	Duration (Days)	Days Complete	Days Remaining	Percent Complete
Project Proposal	10/11/2018	10/18/2018	7	7.00	0.00	100%
Finalize Parts List	10/11/2018	10/31/2018	20	20.00	0.00	100%
Mass Budget	10/18/2018	11/17/2018	30	30.00	0.00	100%
Calculate Power Requirements (Power Budget)	10/11/2018	11/10/2018	30	30.00	0.00	100%
Build Solar Cells	11/20/2018	1/19/2019	60	60.00	0.00	100%
Build DC-DC Converter Circuits	11/20/2018	1/19/2019	60	60.00	0.00	100%
Build Battery System	11/20/2018	1/19/2019	60	60.00	0.00	100%
Research Hardware and Build Electrical Controller (PC104)	11/20/2018	1/19/2019	60	60.00	0.00	100%
Mid-term Progress Report	10/18/2018	1/10/2019	84	84.00	0.00	100%
Prototype Design	12/6/2018	1/5/2019	30	30.00	0.00	100%
Building & Wiring Prototype	12/11/2018	1/10/2019	30	30.00	0.00	100%
Testing	12/26/2018	2/9/2019	45	45.00	0.00	100%
Systems testing	1/2/2019	1/17/2019	15	15.00	0.00	100%
Integration testing	1/20/2019	1/30/2019	10	10.00	0.00	100%
Various Modes testing	2/4/2019	2/19/2019	15	15.00	0.00	100%
Design: Review	2/8/2019	2/9/2019	1	1.00	0.00	100%
Analysis and Test Plan	2/8/2019	4/1/2019	52	52.00	0.00	100%
Design: Demonstration	3/1/2019	4/1/2019	31	31.00	0.00	100%
Design: Oral presentation and aspect	4/1/2019	4/4/2019	3	3.00	0.00	100%
Final Report	4/4/2019	4/15/2019	11	11.00	0.00	100%