Thermal and Power Cubesat Subsystem Integration

A project present to
The Faculty of the Department of Aerospace Engineering
San Jose State University

in partial fulfillment of the requirements for the degree
Master of Science in Aerospace Engineering

By

Ernie Eugene Gabriel Alvarez

May 2014

approved by

Dr. Papadopoulos
Faculty Advisor

San José State
UNIVERSITY

Abstract

CubeSats were developed with the intent of giving those with very little to no experience at all the opportunity to participate in hands-on space missions. What makes CubeSats so accessible is that they incorporate commercial-off-the shelf components to reduce cost. CubeSats generally have lower bus voltages and power requirements. The power system can be operated primarily by batteries or directly charged by solar panels with secondary rechargeable batteries aiding. For CubeSats there is a greater demand for more peak and average power from the solar panels.

TechEdSat is a NASA Ames Research Center 1U CubeSat made by San Jose State University students in partnership with AAC Microtec. Its mission is to evaluate Space Plug-and-play Avionics (SPA) designed by AAC Microtec, and to perform a communications experiment utilizing the Iridium and Orbcomm satellite phone network.

The demand for more power has CubeSat designers researching deployable solar panel systems over the typically used body mounted cell configuration and other ways of increasing power. This in turn could require more attention to the thermal requirements of the CubeSat subsystems. Unlike conventional and larger satellites, the CubeSat employs passive rather than active control methods. One must account for the environmental heating loads on the exterior surface and the internal heat generated by electrical components. To confirm such thermal analysis and design, the CubeSat and its critical components are subjected to various environmental tests that simulate the space environment. This project will compare

a rough thermal analysis done under severe time constraints with the results attained from the TechEdSat thermal cycling tests as well as with in-flight decoded beacon packets from the TechEdSat regarding its nanoRTU temperature data.

Table of Contents

Background

The CubeSat is a miniature satellite with a standard volume is one thousand cubic centimeters (1 liter) and a mass not exceeding 1.33 kilograms. This volume and mass constitutes a 1U design which simply means a one unit CubeSat. The CubeSat was developed to allow people with very little experience to perform experiments in space exploration by using commercial-off-the-shelf parts at a significantly lower cost than components used for much larger satellites. Robert Twiggs developed the concept of the CubeSat while at Stanford University in collaboration with Jordi Puig-Suari from California State Polytechnic University in 1999. Cubesats are typically launched and deployed from a deployment mechanism system known as the Poly-Picosatellite Orbital Deployer (P-POD) which originated from Cal Poly. P-PODs are mounted to a launch vehicle and bring CubeSats into orbit.

While commercial-off-the-shelf parts can be used to make CubeSats, there exist many constraints in using a commercial electrical power system (EPS). Limitations include the lack of documentation on the EPS's true operational parameters and a lack of schematics to properly interface the EPS with the Cubesat's other subsystems. As the launch and delivery methods of CubeSats improve, Cubesats of larger sizes, 3U and over, may be more common. As CubeSats get larger, the power generation and complexity of the EPS will become greater to accommodate larger sized satellites being research and developed by commercial satellite makers (Dorn, 2009). As for the thermal subsystem, the most important thermal design considerations are to make sure that internal heat generating components have conductive thermal paths to the CubeSat frame for heat dissipation. Most launched CubeSats have operated in the nominal -20C to 30C environment of space without extensive internal or external thermal systems (Wertz, 2011).

Power Introduction

Power is extremely essential in order for a spacecraft to operate. The components and subsystems within the spacecraft would not be functional without electrical power. The challenges for space power systems focus around maximizing efficiency, safety, reliability, and radiation harness; while minimizing mass, volume, thermal requirements, and costs (Wertz, 2011). Addressing these needs improves the overall design of an electrical power system and can be applied spacecraft of all sizes. Standard CubeSats are limited in their power generation capability. More power generated by a CubeSat allows for more scientific study, data transfer, and the ability to improve a number of other technologies. Being able to increase the amount of power a CubeSat could generate would lead way to a greater number of missions which could lead to more options in space commercialization.

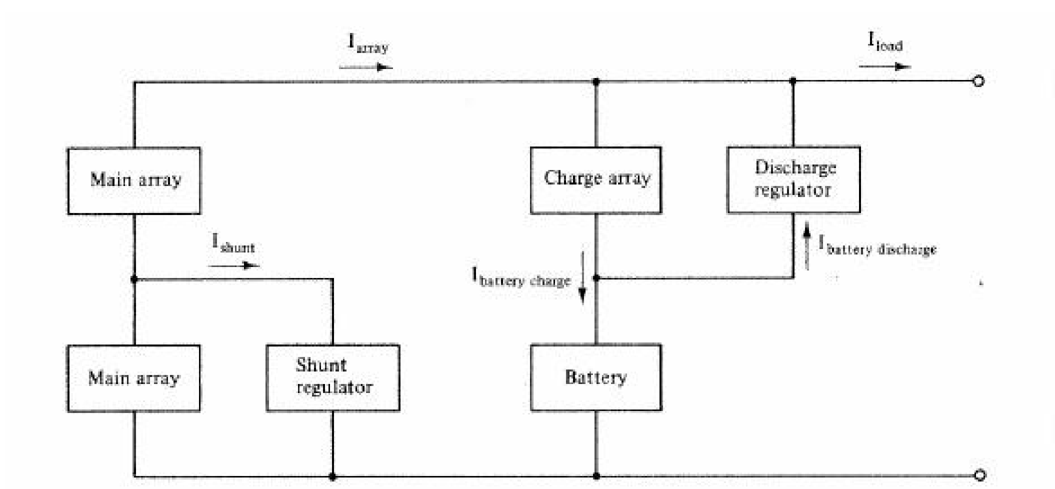


Figure 1-Block Diagram of Electrical Power Subsystem (Agarwal, 1986)

The electrical power system provides, stores, regulates, and distributes electrical power to spacecraft components and its subsystems. The following discussion pertains more to low earth orbit and geosynchronous orbit type spacecraft. Interplanetary and deep space missions often employ more sophisticated power systems such as electric propulsion, fuel cells and nuclear energy. This is because of the difference Earth orbits environments which take advantage of the sun providing energy. The primary source of electric power is from the solar arrays, in which the solar cells convert solar energy into electric power by photovoltaic conversion. During an eclipse, power storage devices such as batteries provide the power. These batteries are discharged during eclipses and are charged slowly during sunlight.

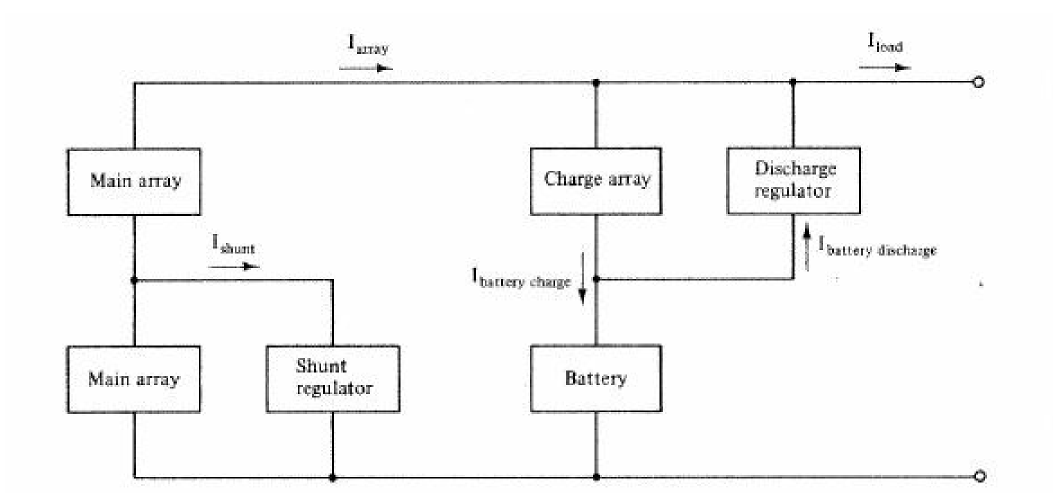


Figure 1-Block Diagram of Electrical Power Subsystem (Agarwal, 1986)

In the figure above a block diagram is shown to illustrate the basic components within a spacecraft’s power system and their relations to one another. The solar arrays and the batteries have highly unregulated power characteristics. Hence power control electronics are used to regulate spacecraft bus voltage and the charge rates for the batteries. The power from the main solar array is supplied to the spacecraft subsystems through the primary distribution bus. Voltage of the solar array is a function of load and temperature. One of biggest issues with voltage can occur when a satellite comes out of an eclipse. Out of an eclipse, the array is cold and its output voltage can double. Shunt regulators are employed to bypass some of the output current from the solar panel to control the bus voltage when the solar power exceeds the load (Agarwal, 1986).

Thermal Introduction

The typical CubeSat configuration results in a large number of radiative couplings between different internal components. If too many nodes are used in the thermal mathematical models then computing solutions will take too long. A compromise must be made between the level of detail and time it takes to provide the rest of design time. Adding deployable solar arrays would allow one to significantly control thermal aspects of the satellite. However, the limited envelope available for CubeSats often prevents the use of large deployable wings.

Thermal tapes can be applied with little to no damage to CubeSat surfaces. These thermal tapes can be implemented as a means of thermal insulation. The optical properties of such tapes can be arranged so that they exhibit desired thermo optical properties such as absorptivity and emissivity. Also, structural parts undergo a surface treatment that has to be taken into account for a thermal analysis of a CubeSat. The interface with the launch adapter guide rails that protrude from the main 10cm x 10cm x 10 cm cubic frame cannot be covered with thermal control material and the inside of the Tube chassis. This Material already has a relatively high emissivity with alodyned aluminum.

MLI is not often used on CubeSats because of the Cubesat’s small volume and subsystem integration. Conductive coupling between subsystem trays provides alternative heat transfer paths from the structural frame, standoffs, and other support plates. Conductive heat transfer from the exposed aluminum to the rest of the side panel reduces the effectiveness of the insulation.

Internal configuration of components are arranged so heat is dissipated to the surroundings of the satellite as efficient as possible. In CubeSats, individual subsystem elements are arranged to minimize conductive paths from highly dissipating items to areas where radiator panels would be and to maximize radiative coupling between them. Since the CubeSat has limited volume, the interior is packed with electronic box trays or circuit boards.

Another configuration aspect that affects thermal control CubeSats is PCB-stacking. By stacking circuit boards onto each other it is possible to obtain a very compact configuration of the internal systems. Also they can be interfaced by stack-through pin headers or flat cable. However, this approach reduces radiative coupling between a circuit board that is located in the center of the PCB-stack and the structure. (2007, Rotteveel). The conductive coupling of a PCB in the stack and its surroundings is also very limited. Radiative coupling between a circuit board and its adjacent boards is high, which makes it difficult to affect the temperature of a single board without altering the temperature distribution of the other boards. Place the highest power dissipation components on the top or bottom of the stack to maximize radiative coupling between the structure and the circuit board stack and place radiator surfaces near those components if CubeSat is willing to implement such radiator surfaces. If that cannot happen, then adjust spacing between the components and other boards to increase radiative coupling.

Since a CubeSat is typical more compact it has a higher specific density than a conventional satellite. This high thermal mass reduces transient effects significantly. In satellites without batteries, the gradients are large as the satellite is in and out of eclipse because there is no internal heat dissipation during this time. The battery system is usually one of the components with the most stringent requirements on allowable temperature range.

Electrical Power System Overview

Development of the electrical power system is broken down into four functions: power source, energy storage, power regulation and control, and power distribution. The three most important sizing requirements for the EPS are the average and peak power demands, orbital parameters such as inclination and altitude and mission duration. For many missions, the end-of-life (EOL) power demands must be reduced to account for a decrease in power source performance. This decrease in power source performance is usually associated with solar array degradation over time. The average electrical power required at EOL determines the size of the power source which in this case will be solar arrays.

For designing a power system for a CubeSat, the following steps must be carried out. One must identify mission requirements, select and size a power source, select and size and energy source, identify power regulation and control distribution (usually peak power tracking), and estimate the mass, average power and peak power requirements.

The design of the deployable CubeSat to be possibly used for future missions will employ solar cells as its energy source, lithium ion as its power storage, and a premade electronics power system board with peak power tracking capability for the power regulation, control, and distribution. The main approach for using solar cells is to use smaller sized pieces so that enough cells can be wired in series to achieve a desired voltage. Lithium Ion is said to be the best battery option for CubeSats to date because of its high capacity weight ratio. The range of regulated voltages for a CubeSat may be from 3-5V DC. Unregulated voltage for two Lithium Ion cells in series may be 7.2V. To charge these two cells would require a voltage of 8-10V from the solar panels. (Wertz, 2011) Peak power tracking is a new method utilized for CubeSat solar panel control to optimize the incident sunlight angle and the power generated. This method uses active control to vary the load on the solar panels in order to keep them at maximum power.

Premade

power boards with the feature of peak power tracking shall be compared in order to select one that appropriately meets the power design requirements.

Step	Reference	FireSat Example
1. Determine requirements and constraints for power subsystem solar array design		
• Average power required during daylight and eclipse	Input parameter, Secs. 10.1, 10.2	110 W during daylight and eclipse
• Orbit altitude and eclipse duration	Input parameter, end papers	700 km 35.3 min
• Design lifetime	Chaps. 2, 3	5 yr
2. Calculate amount of power that must be produced by the solar arrays, P_{sa}	Step 1 Eq. 5-5, end papers (Orbit period − T_e) Eq. 11-5	$P_e = P_d = 110$ W $T_e = 35.3$ min $T_d = 63.5$ min Assume a peak power tracking regulation scheme with $X_e = 0.6$ and $X_d = 0.8$ $P_{sa} = 239.4$ W
3. Select type of solar cell and estimate power output, P_o, with the Sun normal to the surface of the cells	*Si: $P_o = 0.148 \times 1{,}367$ W/m² $= 202$ W/m² *GaAs: $P_o = 0.185 \times 1{,}367$ W/m² $= 253$ W/m² *Multijunction: $P_o = 0.22 \times 1{,}367$ W/m² $= 301$ W/m²	Si solar cells $P_o = 202$ W/m²
4. Determine the beginning-of-life (BOL) power production capability, P_{BOL}, per unit area of the array	Table 11-35 Eq. 5-7 Eq. 11-6	$I_d = 0.77$ $\theta = 23.5$ deg (worst case) $P_{BOL} = 143$ W/m²
5. Determine the end-of-life (EOL) power production capability, P_{EOL}, for the solar array	Performance degradation Si: 3.75% per yr, GaAs: 2.75% per yr, Multijunction: 0.5% per yr Eq. 11-7 Eq. 11-8	Performance degradation is 3.75% per year $L_d = 0.826$ for 5 yr mission $P_{EOL} = 118.1$ W/m²
6. Estimate the solar array area, A_{sa}, required to produce the necessary power, P_{sa}, based on P_{EOL} an alternate approach	Eq. 11-9 Eq. 10-12†	$A_{sa} = 2.0$ m² $A_{sa} = 2.5$ m²
7. Estimate the mass of the solar array	Eq. 10-13†	$M_a = 9.6$ kg
8. Document assumptions		

Figure 2-Solar Array Design Process (Larson, Wertz, 1999)

Solar Array Design Process

The solar array design process is procedural as outlined by the book *Space Mission Analysis and Design*, (SMAD). It starts off by defining the power requirements and constraints for the power subsystem solar array design. In this step, required daylight power (P_d), required power eclipse power (P_e), orbit altitude,

eclipse duration (T_e), and mission duration serve as inputs for the next step. The eclipse duration is actually found from a step before that involves finding the period of a circular orbit defined by the equation where a is the orbit’s semi-major axis (R=6371km + orbit altitude) and μ is the standard gravitational parameter for Earth=398600.5 km^3/sec^2.

$$P = 2\pi\sqrt{R^3/\mu}$$

Equation 1-Period of a Circular Orbit

That value, period of a circular orbit, is inserted into Equation (5-4b) which φ is found from Equation (5-4a) to get the time of the period spent in the eclipse (T_e). The next step shows the how to determine the

amount of power the solar array must provide in illumination.

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

Equation 2-Power Produced by Solar Arrays

In the equation shown above, the daylight power (P_d), required power eclipse power (P_e), daylight

period, eclipse period (T_e) are plugged in as well as the transmission efficiencies of 0.6 and 0.8 for eclipse

and sunlight, respectively, for a peak power tracking regulation configuration. The next step (step 3), shows how to calculate the power output P_0, with the Sun being normal to the surface of the solar cells. The equation for power output is shown below:

$$P_0=\eta*(1367W/m^2)$$

Equation 3-Power Output

In the example, Silicon cells are chosen but for most CubeSats, multijunction solar cells would be used. The next step requires one to determine the beginning-of-life (BOL) power per unit area. In doing so, an inherent degradation (I_d) is assumed to be 0.77,

the beta angle is assumed to be 23.5 degrees as shown

in the example. Using the value of inherent degradation, the beta angle assumed, and the power output attained from the third step, we can attain the power per unit area for the BOL as shown below by equation 4.

$$P_{BOL}=P_0 I_d \cos\theta$$

Equation 4-Beginning of Life Power

Next, one must determine the end-of-life power per unit area. In doing so, the performance degradation for the solar cells are determined. For CubeSats, we would be using the value for multijunction cells rather than silicon cells. The value for performance degradation along with the mission duration are inputted into equation 5 to attain the lifetime degradation as shown below.

$$L_d = (1 - degradation/yr)^{satellite\ life}$$

Equation 5-Lifetime Degradation

The lifetime degradation from equation 5 and the BOL power from equation 4 are inserted into equation 6 shown below to attain the end-of-life power per unit area.

$$P_{EOL}=P_{BOL}L_d$$

Equation 6-End of Life Power

To finally estimate the area and mass of the solar array, we take equation 2 to be divided by equation 6 to get the area of the solar array shown by equation 7 below.

$$A_{sa}=P_{sa}/P_{EOL}$$

Equation 7-Solar Array Area

For the purposes of this CubeSat design we will want to determine the mass of deployed solar arrays. To find the mass of deployed solar arrays, we would take the power produced by the solar arrays from equation 2 divided by the solar array power density.

$$M_{sa}=P_{sa}/(\text{Power Density of Solar Array})$$

Equation 8-Solar Array Mass

Solar Array Configurations

Solar cells are usually in series-parallel combinations to form a solar array. The number of series-connected solar cells in one string established the bus voltage required at EOL, at the operating temperature: the number of parallel strings depend on the current output. Isolation diodes within the solar array typically minimize the effects of shadowing and reversed-biased solar cells.

Solar array configurations can be either planar or concentrated and can be body or panel mounted. Most photovoltaic applications have used a planar array in which solar cells are mounted onto a surface (typically insulated aluminum honeycomb) with an adhesive. Kapton, Kevlar, or fiberglass usually insulates the solar cell from the aluminum honeycomb support structure. Concentrator arrays increase a solar cell's output by utilizing mirrors or lenses to focus solar energy onto the cells.

An assembled solar array is less efficient than single cells due to design inefficiencies, shadowing and temperature variations. All of these factors account for an array's inherent degradation. Solar cells are applied to a substrate, usually honeycomb aluminum, and interconnected, resulting in losses of 10% of the array’s substrate area.

I-V curves indicate the current-voltage characteristics of a solar cell. There exists three important points on the I-V curve for solar-array design. The short-circuit current is where voltage is equal to zero. The peak-power point is where power is a maximum value. The open circuit is where current is equal to zero. From the figure below (11-6), you can see that the BOL power is slightly larger than the EOL. Temperature also affects the I-V curve. A change in the operating temperature of the solar cell or array causes an alteration of the I-V curve along either the current or voltage axis and could possibly change the curve shape that affects the peak-power point. An array often provides peak power coming out of an eclipse because its temperature is at its coldest.

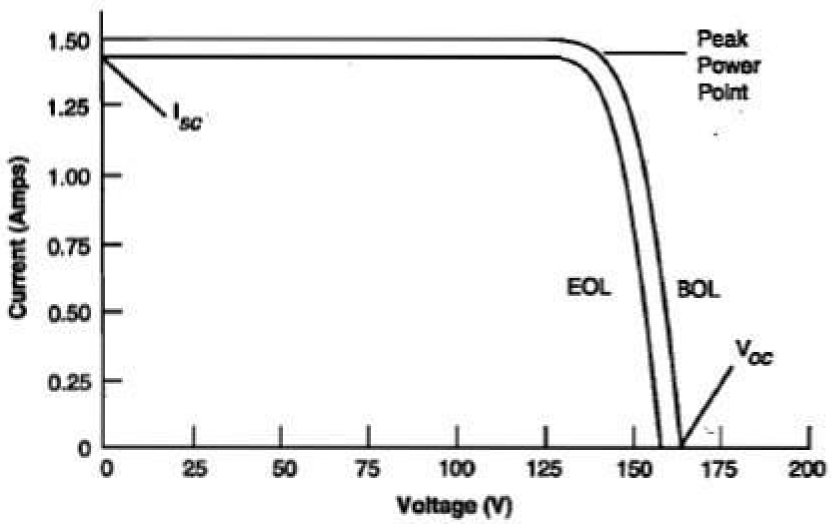


Figure 3-Current-Voltage Plot for a Planar Array

Shadowing considerations are also important because a solar cell will go into open circuit when not illuminated. This is dangerous for a series-connected configuration because if one of the cells is shadowed then the entire string of cells is lost.

Shadowing can be reduced by actively pointing and tracking arrays using diodes, or designing series-parallel arrays. The beginning-of-life, the array’s power per unit area is equation (11-6) where theta is the Sun incidence angle between the vector normal to the surface of the array and the Sun line. The solar array should be designed to minimize the cosine loss.

Panel-mounted arrays usually apply to only 3-axis stabilized spacecraft. The panel-mounted approach tracks and points the solar array to get the best Sun incidence angle. The body-mounted approach reduces tracking and pointing on any spacecraft but the less effective Sun incidence angle and increased array temperature of body-mounted cells produces a lower efficiency in orbit. Panel-mounted arrays are usually mounted on a boom. Deployable panel arrays are either flexible or rigid, according to the type of substrate material that is used. The solar array is often placed away from the payload and the other spacecraft subsystems because of the variable and often high temperature of the solar cells.

Body-mounted cells are used for spinning spacecraft, which provide thermal control by radiating heat to space as the spacecraft spins. Body-mounted arrays use cells inefficiently because of higher temperature and reduced voltage and thus produce less power than panel-mounted arrays. The exact configuration for the solar panels of the CubeSat will later be determined.

Energy Storage Design

Any spacecraft that uses solar cells as a power source requires a system to store energy for peak-power demands and eclipse periods. Energy storage typically occurs in a battery. A battery consists of cells connected in series. The number of cells is determined by the bus-voltage. Batteries can be connected in series to increase the voltage or in parallel to increase current. The energy-storage voltage difference between the end of charge and end of discharge determines the range of the bus voltage. We typically want a positively inclined slope for charging and a flat slope for discharging. We would want to perform the following three steps to determine the battery capacity for energy storage.

1-Determine the energy storage requirements.

Mission length, secondary power storage, orbital parameters such as the eclipse frequency and eclipse length, power profile information such as the voltage, current, depth of discharge, duty cycles, and the battery charge and discharge cycle limits.

2-Select the type of secondary batteries (lithium-ion for CubeSats).

Primary batteries convert chemical into electrical energy but cannot convert electrical into chemical. Thus they are not rechargeable. Primary batteries are used for short missions or long-term task such as memory backup that uses little power. Secondary batteries can also convert chemical into electrical energy as well as vice-versa thus making them rechargeable. It can repeat this process for thousands of cycles. A secondary battery provides power during eclipse periods on spacecraft with solar cells. These batteries charge during illumination and discharge during eclipse. Geosynchronous spacecraft encounter few charge/discharge cycles during eclipse periods, thus allowing a fairly high (50%) depth-of-discharge. Low Earth Orbit spacecraft have at most one eclipse period each orbit or about 15 eclipse periods per day thus accounting for their lower depth-of-discharge (about 15-25%). Depth of discharge (DOD) is simply the percent of total battery capacity removed during a discharge period. The higher the DOD, the shorter the life cycle is for a battery. For some missions, the peak power loads may drive the required battery capacity rather than the eclipse load.

3-Determine battery capacity

This is shown by equation 9 below where the battery capacity is a function of the power required and period of the eclipse, the batteries depth of discharge (DOD), number of batteries, and transmission efficiency.

$$C_r = \frac{P_e T_e}{(DOD)Nn}$$

Equation 9-Battery Capacity

Power Distribution Overview

The power distribution consists of cabling, fault protection, and switches. The power distribution unit also represents individual power loads and switching requirements. Power switches are often mechanical relays because they dissipate less power. The load profile of a spacecraft is a key factor in the design. Some loads may require low to high voltage dc, high-voltage single-phase ac, or high-voltage three-phase ac . Voltages may need some regulating, leveling up or down, and possibly, inverting through dc-dc converters. Certain loads require a voltage that is different from the bus voltage. Power converters often connect loads susceptible to noise or requiring voltage conversion to the distribution system. Converters isolate the load from the noise on the bus and regulate the power provided to the load against disturbances from the load and the bus. They also prevent load failures from damaging the power distribution system and provide control to certain loads. The harness or cabling that connects the subsystems is about 10-25% of the power system’s mass. Harnesses should be kept short to reduce voltage drops and to regulate the bus voltage.

Power distribution systems have been typically dc because spacecraft generate direct current power. AC conversions are used more for higher power spacecraft and would require more electronics, which would add mass to the EPS. Power distribution systems are either centralized or decentralized, depending on the location of the converters. Decentralized power distribution systems has the converters at each load separately, whereas centralized distribution systems regulates power to all spacecraft loads within the main bus. Decentralized systems relate to an unregulated bus because distributed converters regulate power. A regulated bus usually has some converters at the load interface because electronics may require different voltages. Larger spacecraft typically use decentralized systems with an unregulated bus because of the variety of different applications the EPS is to perform. For smaller satellites, power distribution systems are usually centralized and a regulated bus.

Fault protection detects, isolates, and corrects faults. Its primary purpose is to isolate a load that has failed from others to prevent the failure from affecting other subsystems within the spacecraft. A failed load usually indicates a short circuit that attains too much power. Faults are often isolated by fuses. Usually, all these things are accounted for in the design of power boards for CubeSats. Next semester will mention the interaction of all these different attributes of the power distribution feature of the power board and how it interacts with the other minor subsystems within the electrical power subsystem. Also this will be shown with some power electronics theory and calculations.

Power Regulation and Control Overview

Power regulation is divided into three main categories: controlling the solar array, regulating bus voltage, and charging the battery.

Electrical power generated at the array must be controlled to prevent battery overcharging and spacecraft heating. There exist two main power control techniques which are the peak-power tracker (PPT) and a direct-energy transfer (DET). A PPT is a nondissipative subsystem because it extracts the exact power a spacecraft requires up to the peak power of the array. The DET subsystem is a dissipative subsystem because it dissipates power not used by the loads. The DET subsystem can dissipate this power at the array or through shunt resistors to avoid internal power dissipation.

A PPT is a dc-dc converter operating in series with the solar array. It dynamically changes the operating point of the solar-array source to the voltage side of the array and tracks the peak-power point when energy demand exceeds the peak power. It allows the array voltage to boost to its maximum power point; then the converter transforms the input power to an equivalent output power, but at a different voltage and current. A peak power tracker replaces the shunt-regulation function by backing off the peak power point of the arrays toward the end of the battery's charging period. PPT's have an advantage when it comes to mission under five years that require more power at BOL than at EOL.

A shunt regulator is used in direct energy transfer system. It operates in parallel to the arrays and shunts the array current away from the subsystem when the loads or battery charging do not need power. Power subsystems with shunts are very efficient and dissipate little energy by shunting extra power at the array or through shunt resistor banks. A shunt regulated system has fewer parts, lower mass, and higher total efficiency at EOL.

Controlling the bus voltage can either be done unregulated, quasi-regulated, or fully regulated. In an unregulated subsystem, the load bus voltage is the voltage of all the batteries. A quasi regulated subsystem regulates the bus voltage during battery charge but not during discharge. The fully regulated power subsystem is inefficient, but it will work on a spacecraft that requires low power and a highly regulated bus. It uses charge and discharge regulators. An advantage to this configuration is that when loads are connected, the system behaves like a low-impedance power supply, making design integration a simple task. However, it is the most complex type of power subsystem, with low efficiency and high electromagnetic interference when used with a PPT or boost converter.

We can charge batteries individually or in parallel. Parallel charging is simple and has low costs but is hard to integrate into the spacecraft. It can also degrade batteries faster. Also the voltage is the same but the current and temperature are not. This means one battery can receive all the current and possibly overheat if the bus voltage is not controlled. Parallel batteries eventually end up balancing out, so we could use them for missions under five years. For missions of greater than five years, it is better to use independent chargers. Also, batteries should be independently charged as little as possible to avoid degrading them. Individual charging optimizes the battery use by charging each battery to its limit. However, individual chargers add impedance, electronic piece parts, and thermal dissipation. For future work, a thermal analysis of the power board shall be done for the peak-power tracking configuration.

Modes of Operation

It is still uncertain exactly how many modes will be featured in the solar array deployable CubeSat design but the two main modes of operation used in many CubeSats are the active and stand-by modes. The active mode allows for normal operations of the CubeSat. The active mode can be broken down into transmission, eclipse, and sun modes. The transmission mode allows the satellite to send signal to targeted locations whether it be a ground station or another satellite within a reachable orbit. During this mode, the satellite consumes the most power. When in sun mode, the satellite is generation energy via sunlight illumination. In turn, this thermal energy is being converted into electrical energy that keeps the satellite functional till it reaches an eclipse. In eclipse mode, the satellite is being no longer powered by thermal energy absorbed by the solar cells. Instead, the satellite is now functioning via its battery. The battery discharges a voltage which creates a current to supply power to all of the necessary subsystems of the CubeSat. While in standby mode, the power requirements are really low. At most probably a quarter of a watt is dissipated and hence the power board regulates the system to allow for the battery to supply power and the solar cells to temporarily stop generating power for the CubeSat.

The table below give a rough rundown of how the software for TechEdSat is supposed behave under a certain operation level and the components that are active during the operational level.

Operational Level	**Summary**	**Active Components**
0 – Storage	RBF pin is inserted and or the deployment switches are depressed. This causes a physical disconnection of main power from the battery, preventing electrical operation within the satellite	None
1 – Safe Mode	Default mode, safe mode processor operates the Stensat Radio Beacon and charges the batteries using the solar panels.	• Safe Mode nanoRTU • MPDU • Solar Array • Stensat Beacon
2 – Nominal Mode	Only activated when battery charge level reaches a certain amount. In this mode the RTU Lite becomes the main processor, and the safe mode processor, Iridium and Orbcomm subsystems become SPA devices.	• Safe Mode nanoRTU • MPDU • Solar Array • Stensat Beacon • Iridium nanoRTU • OrbComm nanoRTU • Iridium Modem • OrbComm Mode

Table XX-TechEdSat Software Modes of Operation

Environmental Thermal Testing Overview

To ensure that a CubeSat is able to endure the harsh space environment, it must be subjected to a series of environmental tests. Environmental tests are usually broken down into a thermal and vibrational category. For the purpose of this project, only the thermal testing shall be focused on and specifically testing performed on TechEdSat. Environmental tests were conducted at NASA Ames Research Center in the Engineering Evaluation Laboratory (EEL) located in building N244. The facility was able to provide equipment for thermal cycling, vibration, and humidity testing. In order to write the documentation necessary for environmental testing, a number of other documents were referred to as guides. These documents were: TechEdSat ICD, JAXA JEM Payload Accomodation Handbook (JMX-2011073NC), Ames Procedural Requirements (APR 8070.2), ISS Pressurized Volume Hardware Common Interface Requirements Document (SSP 50835 Revision C), Environmental Testing of PhoneSat 1.0, and the ELaNa CubeSat to P-Pod ICD 1.0.

The TechEdSat ICD, interface control document, provided the required tests to satisfy the ISS safety requirements. The JEM Payload Accomodation Handbook provided environmental test levels that the CubeSat needed to be tested to in order to endure launch in a Common Transfer Bag (soft stow) on an HTV-3. The APR 8070.2 provided detailed information on thermal cycling tests. The SSP 50835 provided detailed information on the required tests that the Interface Control Document referred to. The Environmental Testing of PhoneSat 1.0 document gave a comparison of the testing needed to be done for TechEdSat. Thermal cycling testing levels were referenced from this document. The ELanA4 CubeSat to P-Pod

ICD 1.0 included information about the sine wave testing as well as information on other CubeSat Environmental Testing.

Because of the budget and time being severely constrained for this project, environmental testing was only conducted on the flight model at protoflight levels as opposed to qualification and acceptance levels. The initial proposed thermal testing were the thermal vacuum cycle test and battery thermal tests. The Thermal Vacuum Cycle Test, done per agreement with the NASA Ames Chief Engineer, the thermal vacuum cycling test will cover a temperature range of -10 to 50°C. This temperature range is standard for the ISS environment. Also, the test was proposed to run for two cycles. The other proposed tests were the Battery discharge tests. These tests proposed to be conducted for a spare 3000 mAh flight battery and a 4100 mAh trial battery. The spare flight battery tests were conducted at 60, 25, -10, and -15°C. The trial battery tests were conducted at 60, 25, -10, -20, -30°C. These tests measured the time it takes for the batteries to fully discharge. The Battery Thermal Tests was done before Thermal Cycling Testing. Thermal Cycling Testing was the last major test done in the environmental testing plan. After the thermal cycling test, a functional test was performed to see if the cubesat's stensat beacon was sending data packets to a local computer.

Thermal Vacuum Cycling Tests

The objective of the thermal vacuum cycling test is to ensure that TechEdSat will be able to function in the temperature extremes of the space environment. The requirements came from SSP 50835 in 3.9.1.2 (which will be shown in the appendix) and the test levels came from JMX-2011073NC Section 2.4.4 which has Launch (HTV) and Onboard temperatures at a range from -15 to 60°C. The equipment needed for this test were: thermal vacuum controller, data acquisition and control, thermal vacuum chamber, liquid nitrogen, kapton tape, data acquisition hardware and software and thermocouples. The test specifications are as follows:

Number of Cycles: 2 Cycles where a cycle constitutes a hot and cold phase. (Start on cold cycle and end on hot cycle to reduce risk of condensation and contaminants on test articles.)

Ramp Rate: 3˚C/minute (no less than 2˚C/minute, no greater than 5˚C/minute)

Hold Time: Hold the hot and cold extremes for a minimum of 45 minutes each.

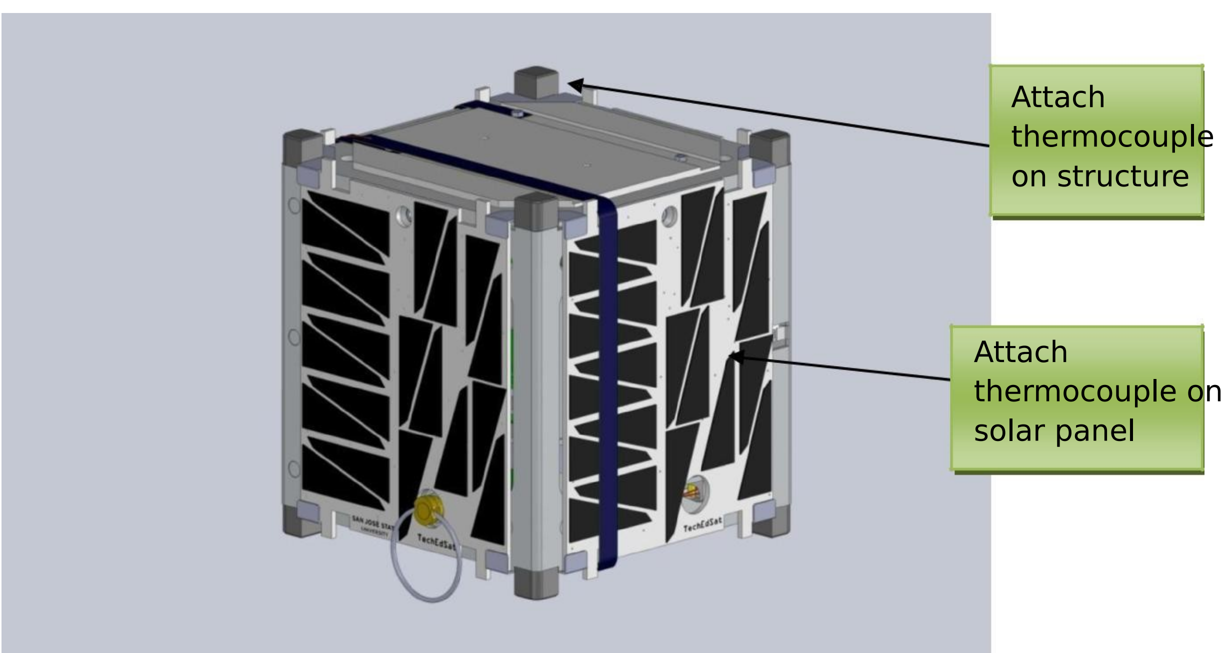
Total Cycling Time: 250 Minutes (4.167 Hours)=

Start off ramp rate from 20˚C and start holding from -10˚C (10 Minutes)+ Holding at each extreme (45 Minutes x 4 = 180 Minutes) +

Ramp rate from extreme to extreme (20 Minutes x 3 = 60 Minutes) *Pressure*: Ambient room Pressure to be brought down to hard vacuum.

Operational State: System Power will be on during testing. Functional tests will be performed during cycles.

Data Acquisition: Satellite internal sensors and system, N244 lab data acquisition system with 4 thermocouples. Thermocouple data using the test lab DAQ should be logged every couple of seconds. Thermocouples are placed as followed on the figure below:



The test procedure will be listed in the appendix.

Thermal Cycle Testing Results

Thermal cycling testing shown below is to test the endurance of the CubeSat frame as well as the solar cells to see if they survive the space environment, particularly going in and out of an eclipse. Below is a figure tested to the JAXA JEM Payload Accomodation Handbook (JMX-2011073NC) standard of -15 to 60°C

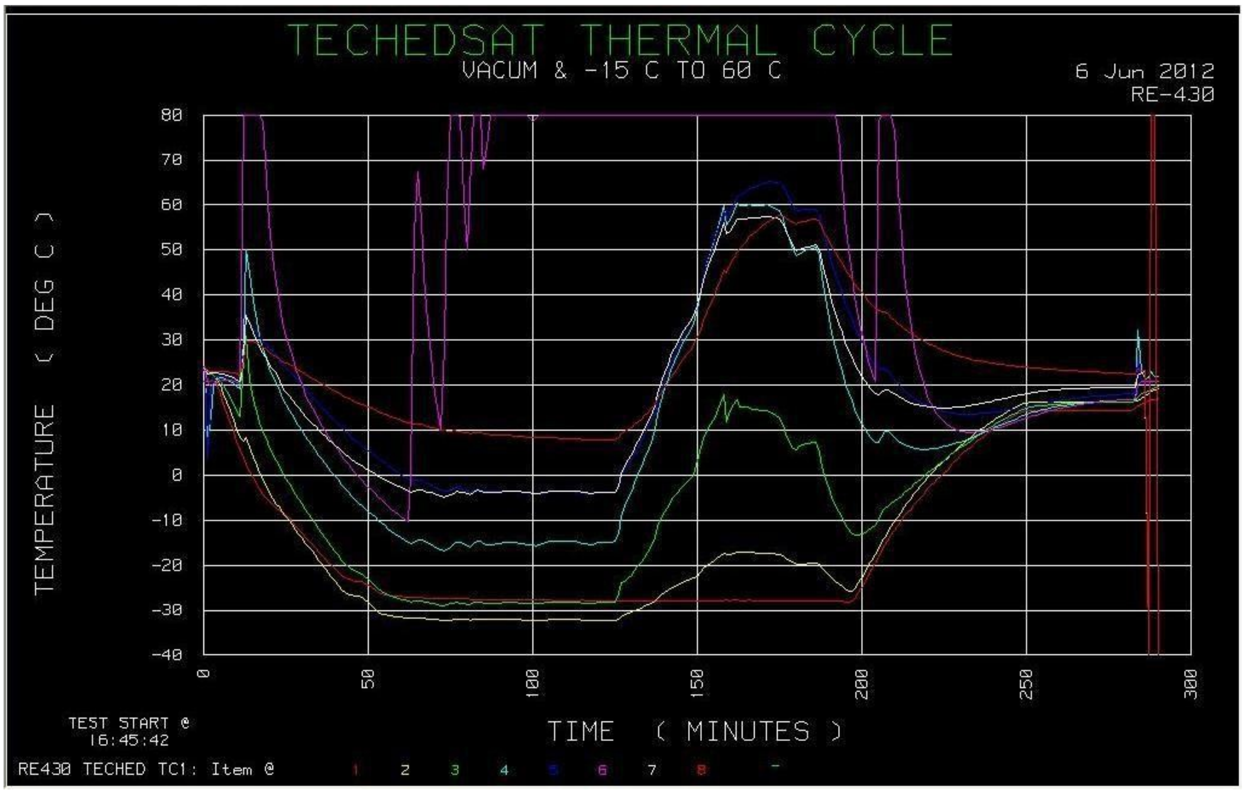


Figure XX-Thermal Cycle testing from -15 to 60°C

Eight plots are plotted are shown representing the liquid nitrogen bath (red), heat exchanger (yellow), shroud enclosing thermal chamber (green), top of chamber (teal), bottom of chamber (blue), lamp (purple), CubeSat frame (white), solar cells (also in red). Despite having the same color it is easy to distinguish the plot for the liquid nitrogen bath and the solar cells because the liquid nitrogen bath will always have a lower temperature than the solar cells. As expected the temperatures oscillate between their cold and warm cases starting off at around 20°C. The plots we are most concerned about are the CubeSat frame and the solar cells. The CubeSat frame stays within -5 to 55°C while the solar cells maintain a temperature range of 10 to 60°C. The lamp plot in purple is the most non-conforming curve because it provides the heat flux into the chamber system which accounts for its steep positive and negative slopes. The adjustment of the lamp's power simulates the conditions coming in and coming out of an eclipse.

In the second thermal cycle figure below, the Cubesat was able to withstand the thermal environment for the thermal vacuum cycle test. There were no anomalies from the test and TechEdSat passed the functional test after the thermal vacuum cycling test without any problems.

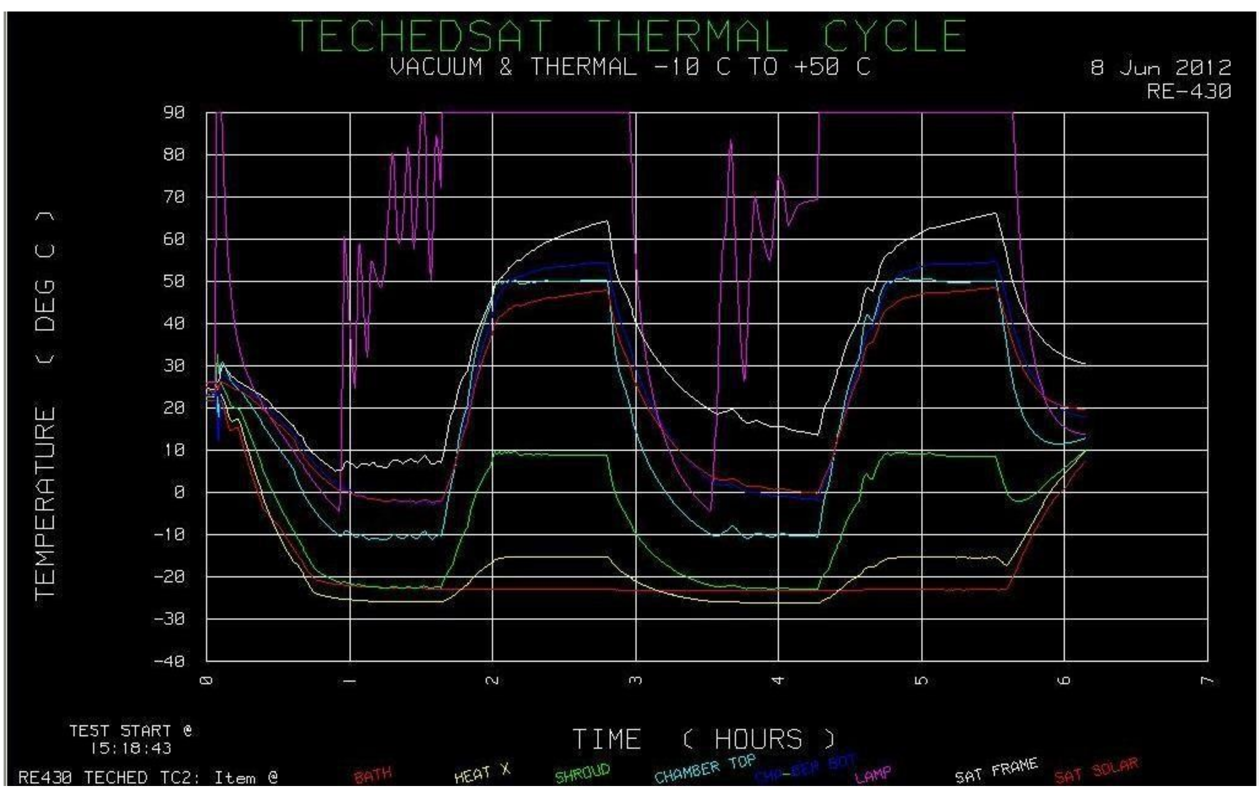


Figure XX-Thermal Cycle Testing from -10 to 50C

Here the figure looks a bit cleaner than the next. The cycling from cold to hot and hot to cold looks more periodically steady by making more adjustments to the lamp. The lamp has several oscillations in the beginning of cycling to make sure that heating and cooling of the vacuum chamber abides by the 3°C/min temperature slope. The CubeSat frame ranges from just under 10°C to about 65°C. The solar cells range from 0°C to 50°C.

Battery Thermal Test Results

This sections shows the battery discharge at varying temperatures. The thermal environment testing conditions did not present any complications with the battery operating. However, the battery ran out of power because it was running the entire day without charging when being subjected to all the other tests.

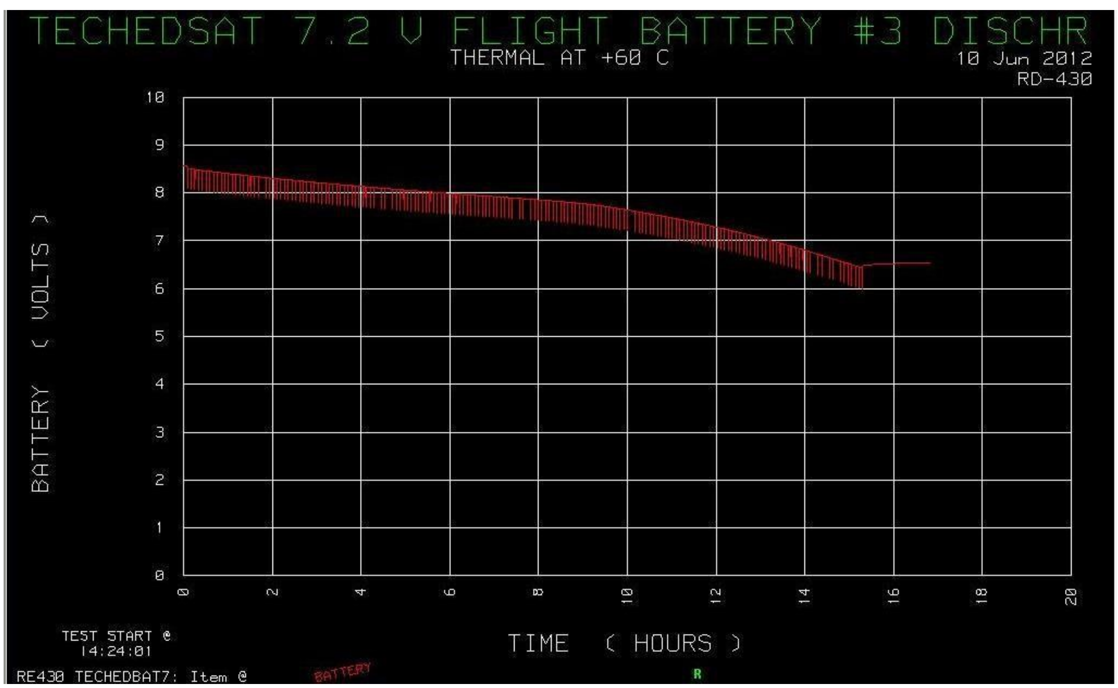


Figure XX=Spare Flight Battery Discharge Cycle at 60°C

The figure above shows the battery starting off at around 8.5V and going as low as 6V at the end of 15 hours. The battery is quickly charging and discharging in cycles of about half a volt between its peaks.

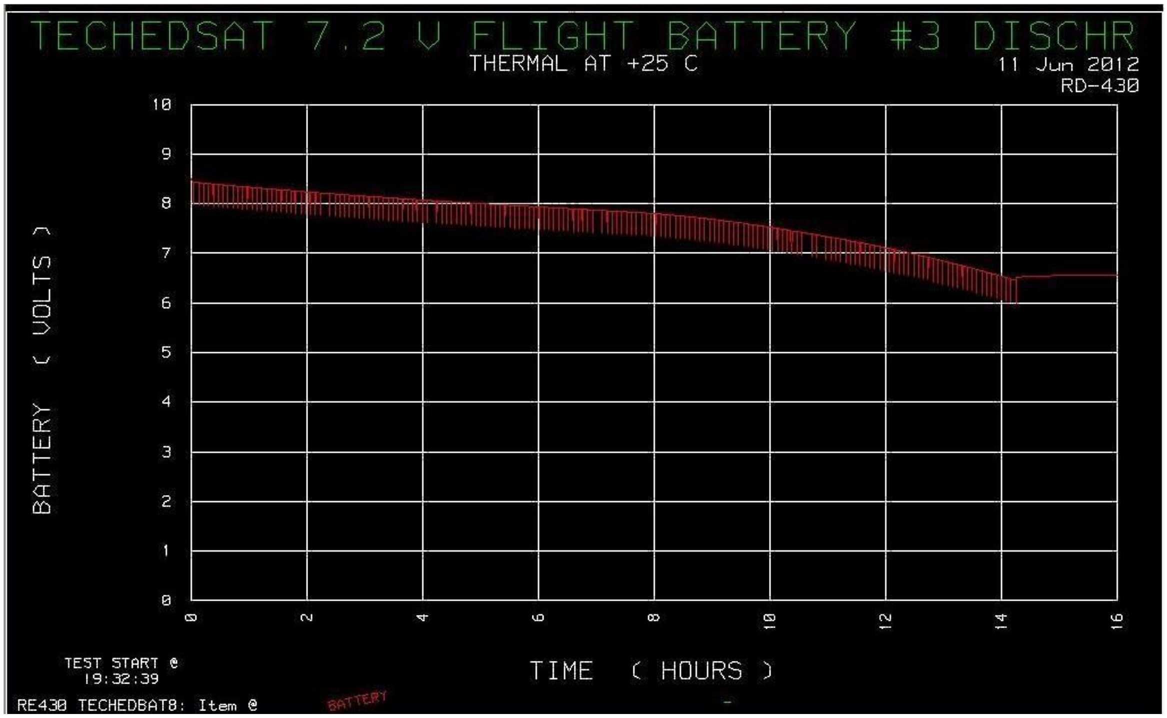


Figure XX-Spare Flight Battery Discharge Cycle at 25°C

The figure above shows that around 14 hours the battery reaches its lowest voltage at 6V while starting off at 8.5V. This figure is similar to the figure at 60°C

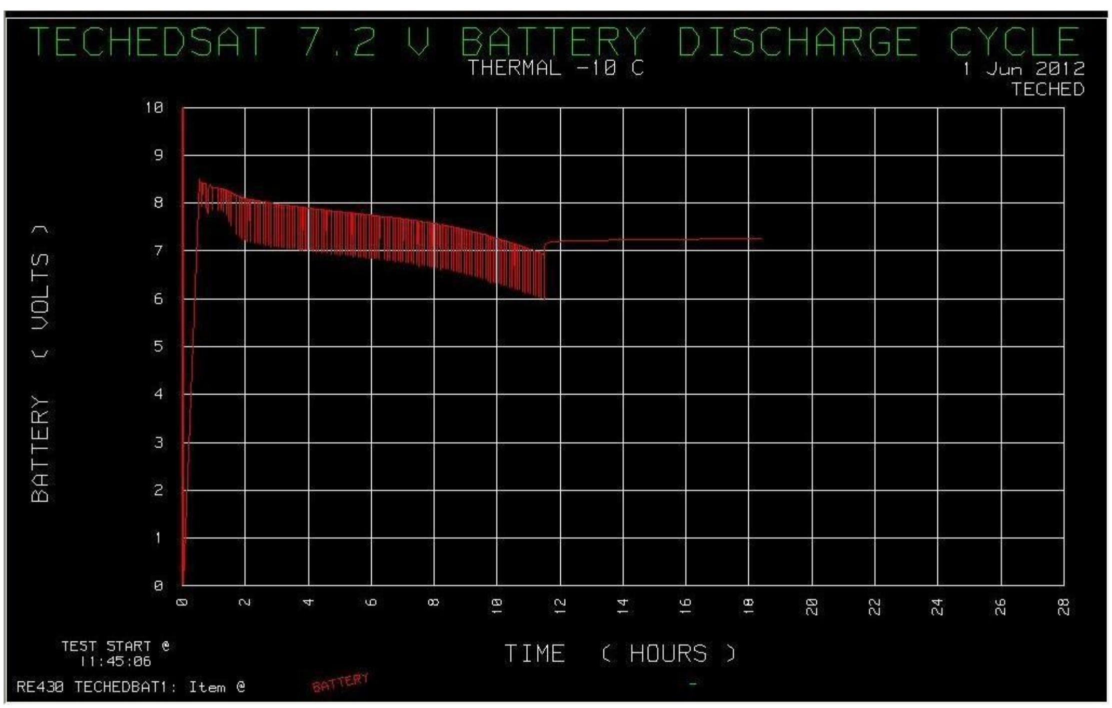


Figure XX-Spare Flight Battery Discharge Cycle at -10°C

The figure above shows how the battery voltage in the beginning of testing was quickly discharged to zero but was able to endure the past 12 hours and ended up discharging around 7 volts after that time.

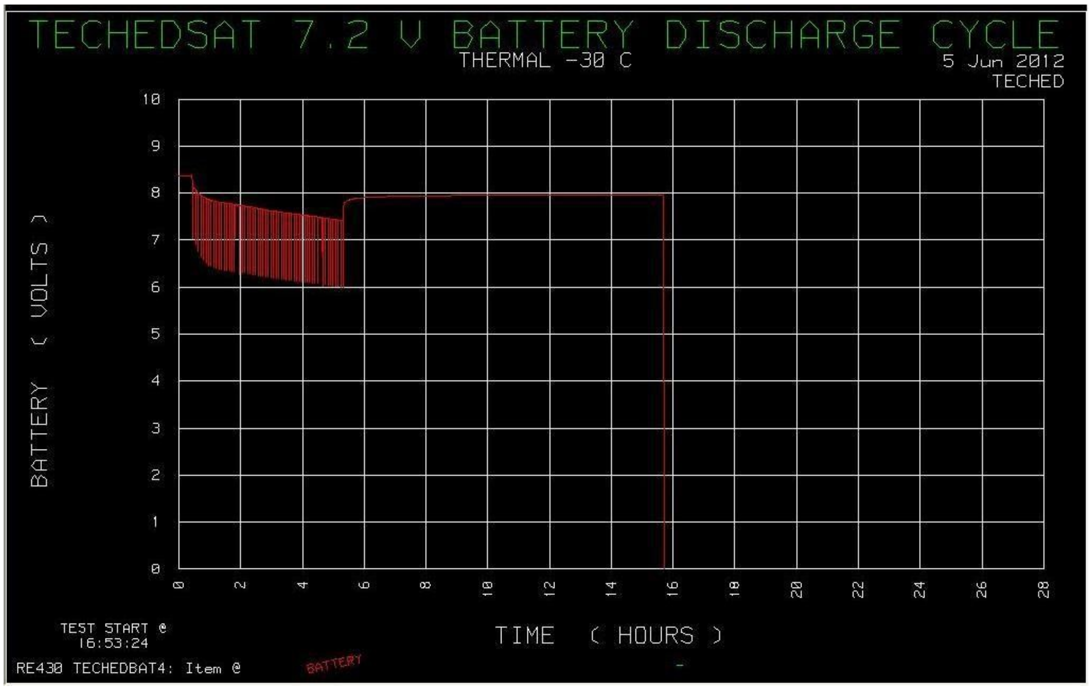


Figure XX-Test Battery Discharge Cycle at -30C

The thermal environment testing conditions did not present any complications with the battery operating. However, the battery ran out of power because it was running the entire day without charging when being subjected to all the other tests. Here the battery quickly discharges to around 6.5 V in an hour and rapidly oscillates between 6 and 7.5V for four hours before it reaches a steady state at 8V and quickly discharges to zero volts before 16 hours.

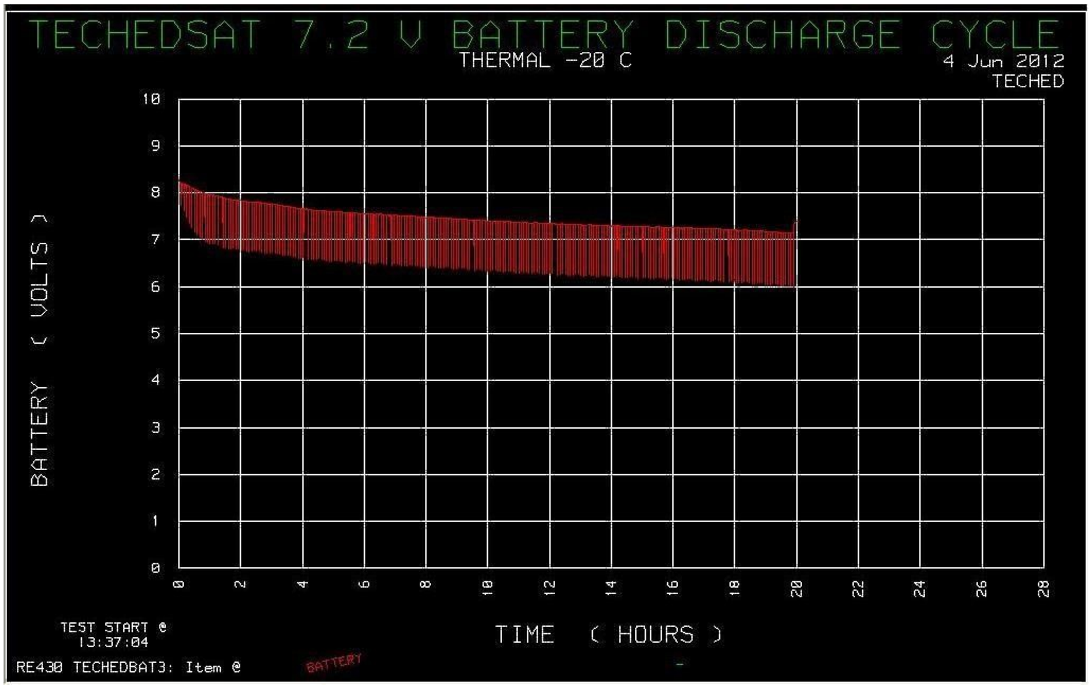


Figure XX-Test Battery Discharge at -20C

Here the battery is looking well and cycles in discharge with about a volt different in maximum and minimum voltage. The voltage at 20 hours reaches around 6 volts.

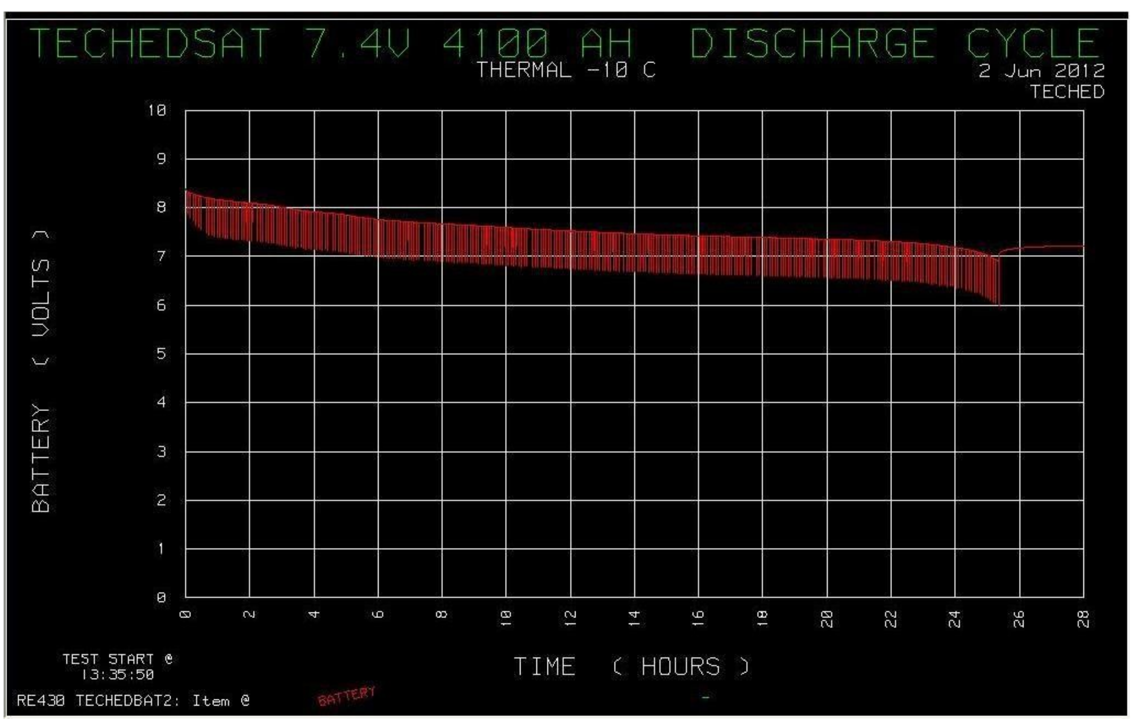


Figure XX-4100AH Test Battery Discharge Cycle at -10C

The figure here shows that the battery was able to endure 26 hours of continuous discharge and reached a steady state voltage of 7V.

Figure XX-Test Battery Discharge Cycle ramped to 60C

The figure above shows that the battery was subject to nearly 28 hours of continuous testing. At around 24 hours, the discharge cycles started to decrease as the voltage was as low as 6 volts at 26 hours.

CubeSat Analytical Thermal Analysis

The following approach is for a simplified thermal model for the thermal control of a CubeSat orbiting the Earth at 400 km altitude.

The satellite is assumed to be 3-axes stabilized, with a cubic main body of 0.1m. All walls are 1.5 mm thick and there are four faces exposed to sunlight. These faces are covered with Spectrolab Triangular Solar Cells 27% efficiency. Inside the CubeSat are the following critical components:

RTULite, NanoRTU-1, NanoRTU-2, NanoRTU-3, Quake Q1000, Iridium 9602, StenSat Beacon, Solar Panel Face 1, Solar Panel Face 2, Solar Panel Face 3, Solar Panel 4, Positive Z Frame, Flight Battery, Battery Plate, Bottom Plate (that is articulated with the Patch Antenna and one of the Nano-RTU's).

This accounts for the geometry that is to be used for the model development. Also needed information about each component are its optical and thermophysical properties. Optical properties consists of absorptivity, α, and emissivity, ε. Below is a figure showing optical surface properties by the types of materials that are used.

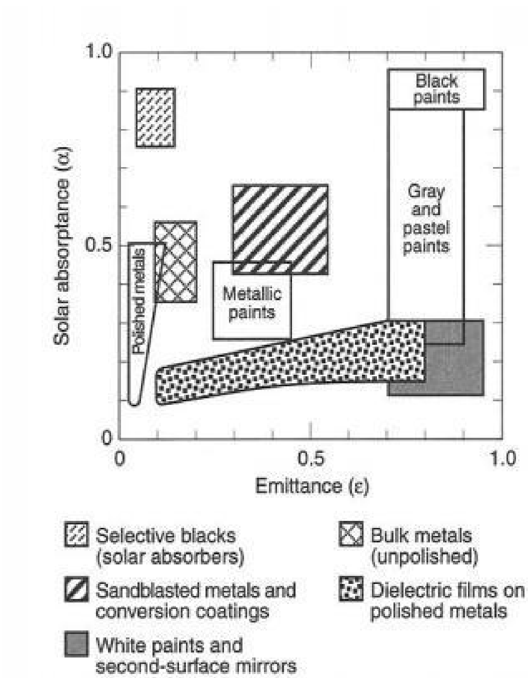


Figure XX-Surface Properties by type of Finish (Spacecraft Thermal Control Handbook, Gilmore, pg. 141)

For the thermal model, the following nodes are to be considered; one at each of the four faces of the

main box covered in solar cells and others for components. Thermo-optical properties of surfaces

should be adequately selected.

For CubeSats, especially of the 1U variety such as TechEdSat, there is limited optical property design that takes place as most of the variance or design space for configuring optical properties is minimized by the use of body mounted solar cells in order to produce as much power as possible.

The other material property that is taken into consideration is thermophysical. Thermophysical properties give information regarding a material’s thermal conductivity, density and specific heat.

Before the Nodal Equations can be developed for these 15 components there are two other cross-disciplinary fields a thermal designer must consult with and that is those handling the orbital mechanics subsystem and the electrical power subsystem.

In regards to orbital mechanics, the following depicts typical orbital elements that are associated with orbital mission design and can be of use to thermal designers that are taking into account the environmental fluxes that go into a satellite.

Orbital Mechanics

In regards to orbital mechanics, the following depicts typical orbital elements that are associated with orbital mission design and can be of use to thermal designers that are taking into account the environmental fluxes that go into a satellite.

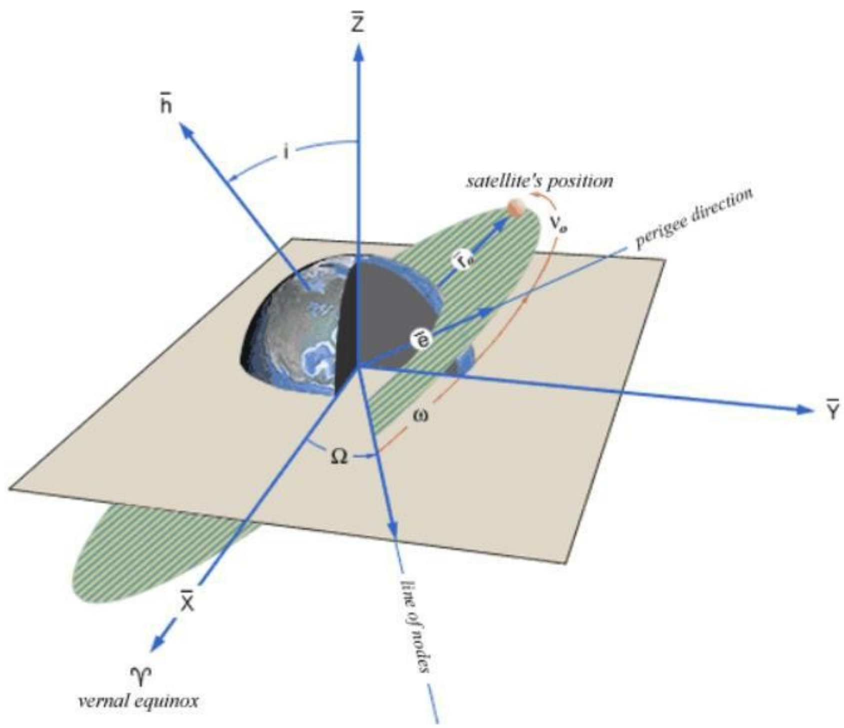


Figure XX-Orbital Mechanics Elements

Sun to Earth Distance, Earth Radius, Earth Orbit Period, Inclination Angle, Earth Surface Temperature, Earth Emissivity, Solar Irradiance, Albedo, Beta Angle, Relative Altitude, Satellite Orbit Period

The semi-major axis indicated by a, defines the size of the orbit. Eccentricity, indicated by e, defines the shape of the orbit. Inclination, I, defines the orientation of the orbit with respect to the Earth's equator. Argument of perigee, ω, defines the low point, perigee, of the orbit is with respect to the Earth's surface. Right Ascension of the Ascending Node, Ω, defines the location of the ascending and descending orbit locations with respect to the Earth's equatorial plane. The true/mean anomaly, γ, defines where the satellite is within the orbit with respect to perigee.

CubeSat Data:

Main cube side, Aluminum Plate Thickness, Cube face area (each), Solar panel effective area , Solar

Panel Efficiency, Solar Panel Absorptance, Solar Panel Emissivity, Component Size (each),

Component Mass (each), Component C_p (each)

In order to come up with rough temperature plot estimates one must find the external heat loads (solar, albedo, and infrared) as a function of orbit position.

An energy balance of all CubeSat components are created by a finite difference scheme of non regular isothermal elements also called nodes. These nodes are found by the temperature evolution at each point in the satellite along the orbit.

E_ele is the energy stored by the batteries

W_net is the net electrical work rate received

Q_net is the net heat input rate

Q_s, in=total solar energy absorbed, including the part that generates electricity in the PV panels, but not including the reflected part; and similarly for the planet albedo input

Total solar input power depends on time due to changes in area subjected to sunlight (including eclipse shadowing: irradiance can be considered constant for a satellite)

$$\dot{Q}_{solar} = \alpha E A_{exp}$$

Power input on the CubeSat from albedo is:

$$\dot{Q}_{albedo} = \alpha A_e F_{e,CS} \rho_e E F_a = \alpha A_{CS} F_{CS,e} \rho_e E F_a$$

This assumes a planetary reflectance ρ_e to solar irradiance E, the maximum power reflected over a unit

area of the planet is $\rho_e E$, and the average from a point of view moving along the orbit can be taken as

$\rho_e E F_a$, where the albedo factor is approximated from the subsolar point to the entrance into an eclipse.

The fraction of that solar reflected power incident on the CubeSat is $\rho_e E F_a$, multiplied by the area of

Earth, Earth's view factor from itself to the CubeSat and the average solar absorptance of the CubeSat is $\alpha A_e F_{e,CS}$. The power input on the CubeSat from Earth's emission would be:

$=\alpha_{IR} A_e F_{e,CS} \varepsilon_e \sigma^4 = \varepsilon A_{CS} F_{CS,e} \varepsilon_e{}^4$. While the CubeSat emits $=A_{CS} \varepsilon \sigma^4$

When more than one node is chosen to represent the temperature of a specific component, one has to account for all the heat and work exchanges that take place amongst the nodes themselves. The energy balance equation that best represents this is:

$$m_i c_i \frac{dT_i}{dt} + \frac{dE_{ele,i}}{dt} = \dot{W}_{ele,net,i} + \dot{Q}_{solar,i} + \dot{Q}_{albedo,i} + \dot{Q}_{e,i} + \left[\sum_j \dot{Q}_{cond,j,i} + \sum_j \dot{Q}_{conv,j,i} + \sum_j \dot{Q}_{rad,j,i} \right] - \dot{Q}_{out,i}$$

For purposes of simplification, the orbit will be assumed to be a circular Keplerian orbit and a spin rate factor will decrease the amount of heat flux on nodes 1 and 2 by 0.25 to assume that heating time on all

four sides about an orbit is equal. Electrical and thermal energy balance equations are written for each node in order to find the mean temperatures at the nodes along the orbit.

After the nodal equations have been established, we want to get conductive couplings that enter into the heat input to node I from node j.

$$\dot{Q}_{cond,j,i} = C_{j,i}(T_j - T_i)$$

using a quasi one-dimensional approximation of heat conduction along two different geometrical and material properties in series, the conduction coupling for two nodes will look like this equation below:

$$\dot{Q} = k_1 A_1 \frac{T_1 - T_{int}}{L_1} = k_2 A_2 \frac{T_{int} - T_2}{L_2} = \frac{T_1 - T_2}{\frac{L_1}{k_1 A_1} + \frac{L_2}{k_2 A_2}} => C_{12} = \frac{1}{\frac{L_1}{k_1 A_1} + \frac{L_2}{k_2 A_2}}$$

where A is the area normal to heat flow, L is the length along the conduction path. Solve C_ji for the other n nodes and arrange them into an n by n matrix.

After finding the conductive couplings, the radiative couplings R_ji that enter into the heat input from node j to node I must be found by:

$$\dot{Q}_{rad,j,i} = \sigma R_{j,i}(T_j^4 - T_i^4)$$

With the case of isothermal, opaque and diffusively radiating surfaces, radiation exchange can be worked out analytically and expressed in terms of view factors and emissivities as expressed by:

$$\dot{Q}_{rad,j,i} = A_i \sigma F_{j,i}(T_j^4 - T_i^4) \rightarrow R_{i,j} = \varepsilon_i A_i \varepsilon_j \sum \bar{F}_{j,k} F_{j,k}$$

where:

$$\bar{F} = \begin{bmatrix} 1-\rho_1 F_{11} & -\rho_1 F_{12} & -\rho_1 F_{13} & \dots \\ -\rho_2 F_{21} & 1-\rho_2 F_{22} & -\rho_2 F_{21} & \dots \\ -\rho_3 F_{31} & -\rho_3 F_{32} & 1-\rho_3 F_{33} & \dots \\ \dots & \dots & \dots & 1-\rho_N F_{NN} \end{bmatrix}$$

p is assumed to be 1-e. If all internal surfaces are assumed to be blackbodies then equation previously expressed in terms of view factors and emissivities is expressed by

$$\dot{Q}_{rad,i,j} = A_i \sigma F_{i,j}\left(T_j^4 - T_i^4\right) \rightarrow R_{i,j} = A_i F_{i,j}$$

This is used to get

After all is complete, we can now establish all the values for the nodal equations and find the steady and means orbital temperatures of the nodes. The power budget serves as the boundary conditions in creating the thermal model. The thermal boundary are described by associating a component with the heat it dissipates thru its power consumption.

Power Boundary Conditions

The internal energy generated within the CubeSat by its electrical internal components are what account for the conduction term in which electrical energy is converted into heat via power dissipation. These will be values that will be put into thermal desktop as heat loads and are calculated for on a per second basis.

Device	Volts	Amps (active)	Watts (active)	Amps (Standby)	Watts (Standby)	Amps (Sleep)	Watts (Sleep)	active cycle
RTULite	3.3		1.5		1.3		1.3	
NanoRTU -1	3.3		0.15		0.02		0.02	
NanoRTU -2	3.3		0.15		0.02		0.02	
NanoRTU -3	3.3		0.15		0.02		0.02	
Quake Q1000	12	2	24	0.07	0.84	0.000005	0.00006	.5 sec
Iridium 9602	5	1.5	7.5	0.195	0.975		0.022	.5 sec
Stensat Beacon	5	0.65	3.25	0.04	0.2		0.2	1 sec / min
	Battery 1	**Battery/sec**						
Voltage	7.4							
Amps-Hr	4.1	0.001138889						
Watts	30.34	0.008427778						
Orbit (min)	44	11	22	11				
Solar	**Eclipse**	**Partial**	**Full**	**Partial**	**Total W**		**Total W/hr in sec**	
Watts	0	1.35	1.8	1.35				

Table XX-Devices and Watt Calculations

	Eclipse	Partial	Full	Partial	Total W		Total W/hr in sec
Level 0							
NanoRTU -1	0.11	0.0275	0.055	0.0275			0.15
Stensat Beacon	0.039722222	0.009930556	0.019861111	0.009930556			0.054166667
Total used	0.149722222	0.037430556	0.074861111	0.037430556	0.299444444		
available	30.19027778	30.55006944	30.92513889	30.55006944			
Level 1							
NanoRTU -1	0.11	0.0275	0.055	0.0275			0.15
Stensat Beacon	0.039722222	0.009930556	0.019861111	0.009930556			0.054166667
RTULite	1.1	0.275	0.55	0.275			1.5
NanoRTU -2 (standby)	0.014666667	0.003666667	0.007333333	0.003666667			0.02
Quake Q1000 (R)	0.616	0.154	0.308	0.154			0.84
NanoRTU -3 (standby)	0.014666667	0.003666667	0.007333333	0.003666667			0.02
Iridium 9602 (R)	0.715	0.17875	0.3575	0.17875			0.975
Total used	2.610055556	0.652513889	1.305027778	0.652513889	5.220111111		
available	27.72994444	29.93498611	29.69497222	29.93498611			
Level 2							
NanoRTU -1	0.11	0.0275	0.055	0.0275			0.15
Stensat Beacon	0.039722222	0.009930556	0.019861111	0.009930556			0.054166667
RTULite	1.1	0.275	0.55	0.275			1.5
NanoRTU -2	0.11	0.0275	0.055	0.0275			0.15
Quake Q1000	0.006666667	0.006666667	0.006666667	0.006666667			0.018181818
NanoRTU -3	0.11	0.0275	0.055	0.0275			0.15
Iridium 9602	0.002083333	0.002083333	0.002083333	0.002083333			0.005681818
Total used	1.478472222	0.376180556	0.743611111	0.376180556	2.974444444		
available	28.86152778	30.21131944	30.25638889	30.21131944			

Table XX-Operation Modes and Watts per Second Calculations

Table XX-Devices and Watt Calculations display electrical performance values that are later to be used in the table below, Table XX-Operation Modes and Watts per Second Calculations, that are later translated into W/sec from the values in blue from Table XX as well as the orbital durations solved in Thermal Desktop.

In the design and analysis of spacecraft in the harsh space thermal environment, solar fluxes constantly vary. Most CubeSats are designed for short-term orbits and an average constant value can be used for a radiation analysis throught RadCAD. Thermal simulations for designs are typically performed for the hot case and cold case scenarios. A hot case scenario entails a satellite being in illumination nearing end-of-life, EOL, and operating at maximum power. A cold case scenario entails a satellite being in eclipse with minimal to no power being dissipated.

In Flight Results

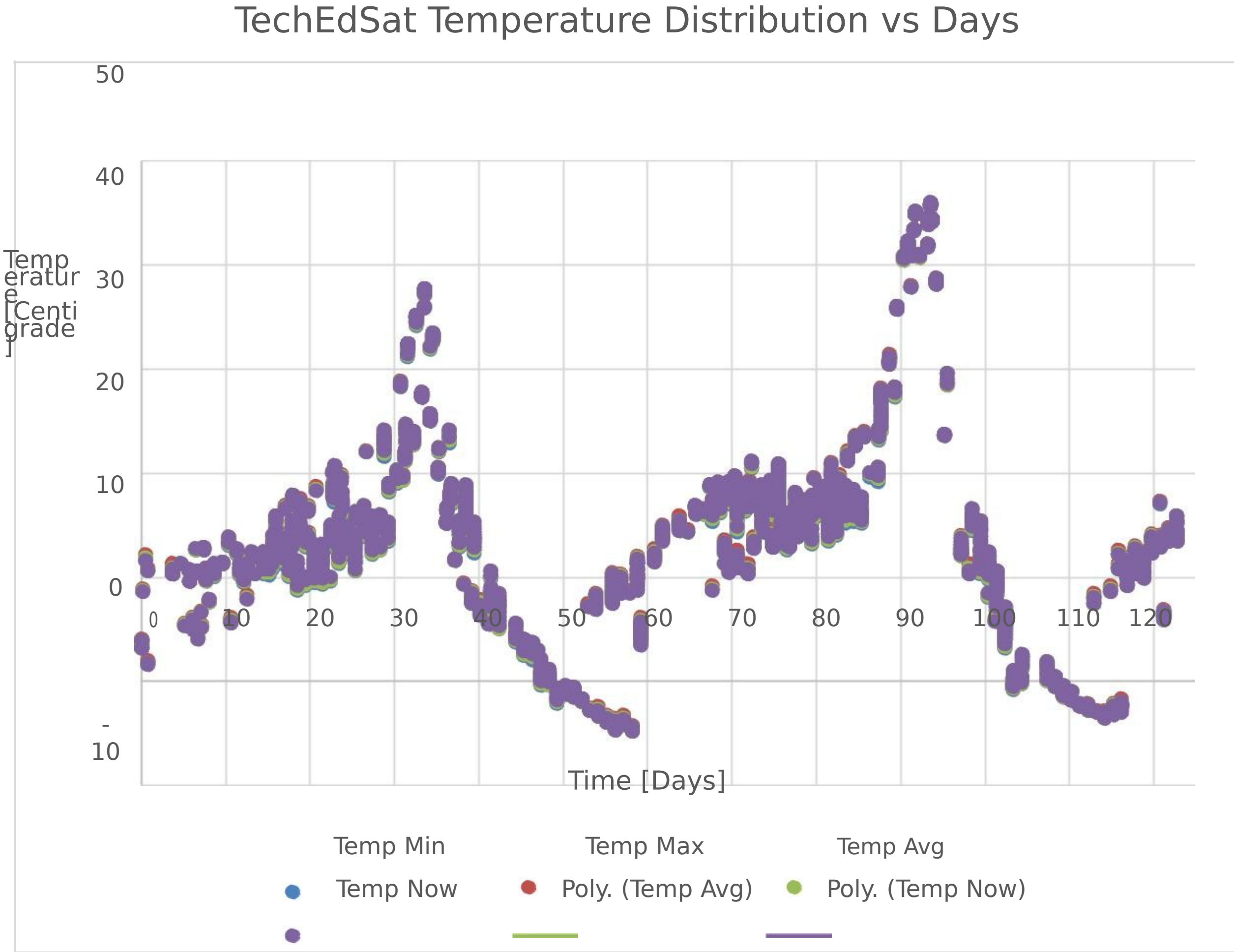


Figure XX-TechEdSat Temperature Distribution over Time

In the figure above, TechEdSat is oscillating in temperature as it completes orbits. What is discernible about this plot is that the temperature amplitude between high and low temperatures is starting to increase. One explanation could be that the magnetic hysteresis material that is keeping the satellite passively stable is starting to lose strength in its magnetic properties. If its magnetic properties are starting to diminish then its spin rate is possibly decreasing. With a slower spin rate, TechEdSat cannot dissipate heat more evenly along its periods of illumination and stays colder during eclipse.

References

Agarwal, B. N (1986). *Design of Geosynchronous Spacecraft.*

Prentice Hall

Bein. A.L. (2008). *NPS-SCAT (Solar Cell Array Tester) Construction of NPS' First Prototype Cubesat*.

(Master's thesis, aval Postgraduate School)

Dech, S., Elbrecht, A., & Gotscheber, A. (2011). *1U Cubesat Design for increased Power Generation*.

(1st AIAA Conference of University Satellite Missions and CubeSat Workshop, Rome, Italy).

Dorn Jr., L.T. (2009). *NPS-SCAT; Electrical Power System.*

(Master's thesis, aval Postgraduate School)

Everett, D.F., Puschell, J.J., & Wertz J.R. (2010). *Space Mission Engineering: The New SMAD* Microcosm Press.

Garg, M., Franki, M., Sembera, J. *Satellite Solutions Cubesat Design Team* (University of Texas at Austin)

Larson, W.J. & Wertz, J.R. (1999). *Space Mission Analysis and Design (3rd Edition*).

Microcosm Press.

Patel, M.R (2004). *Spacecraft Power Systems.*

CRC Press

Appendix A-Power Budget

	Power Modes			
Device	**Volts**	**Amps (active)**	**Watts (active)**	**Amps (Standby)**
RTULite	3.3		1.5	
NanoRTU -1	3.3		0.15	
NanoRTU -2	3.3		0.15	
NanoRTU -3	3.3		0.15	
Quake Q1000	12	2	24	0.07
Iridium 9602	5	1.5	7.5	0.195
StenSat Beacon	5	0.65	3.25	0.04

Device	**Watts (Standby)**	**Amps (Sleep)**	**Watts (Sleep)**	**active cycle**
RTULite	1.3		1.3	
NanoRTU -1	0.02		0.02	
NanoRTU -2	0.02		0.02	
NanoRTU -3	0.02		0.02	
Quake Q1000	0.84	0.000005	0.00006	.5 sec
Iridium 9602	0.975		0.022	.5 sec
StenSat Beacon	0.2		0.2	1 sec / min

	Battery 1			
Voltage	7.4			
Amps	4.1			
Watts	30.34			
Orbit (min)	44	11	22	11
Solar	**Eclipse**	**Partial**	**Full**	**Partial**

Watts	0	1.35	1.8	1.35

Level 0				
NanoRTU -1	0.11	0.0275	0.055	0.0275
Senstat Beacon	0.039722222	0.009930556	0.019861111	0.009930556
Total used	0.149722222	0.037430556	0.074861111	0.037430556
available	30.19027778	30.55006944	30.92513889	30.55006944

Level 1				
NanoRTU -1	0.11	0.0275	0.055	0.0275
Senstat Beacon	0.039722222	0.009930556	0.019861111	0.009930556
RTULite	1.1	0.275	0.55	0.275
NanoRTU -2 (standby)	0.014666667	0.003666667	0.007333333	0.003666667
Quake Q1000 (R)	0.616	0.154	0.308	0.154
NanoRTU -3 (standby)	0.014666667	0.003666667	0.007333333	0.003666667
Iridium 9602 (R)	0.715	0.17875	0.3575	0.17875
Total used	2.610055556	0.652513889	1.305027778	0.652513889
available	27.72994444	29.93498611	29.69497222	29.93498611

Level 2				
NanoRTU -1	0.11	0.0275	0.055	0.0275
Senstat Beacon	0.039722222	0.009930556	0.019861111	0.009930556
RTULite	1.1	0.275	0.55	0.275
NanoRTU -2	0.11	0.0275	0.055	0.0275
Quake Q1000	0.006666667	0.006666667	0.006666667	0.006666667
NanoRTU -3	0.11	0.0275	0.055	0.0275
Iridium 9602	0.002083333	0.002083333	0.002083333	0.002083333
Total used	1.47847222	0.376180556	0.743611111	0.376180556

	2			
available	28.8615277 8	30.21131944	30.25638889	30.21131944

Appendix B

Solar Panel Testing-Outside and Inside Chamber Test

Description: Determine performance characteristics of the solar panels, two tests were performed:

-Outside test, with the solar panel in direct sunlight

-Inside test, in a chamber in which the solar panel was tested both in ambient light and while illuminated by a lamp

Test Setup: The following setup was used for testing:

The following setup was used for testing:

-A TechEdSat solar panel

-A number of load resistors (300Ω, 150Ω, 5Ω and 16Ω)

-A multimeter

The setup for the inside chamber test is illustrated below. For the outside test, the solar panel was directed at the sun.

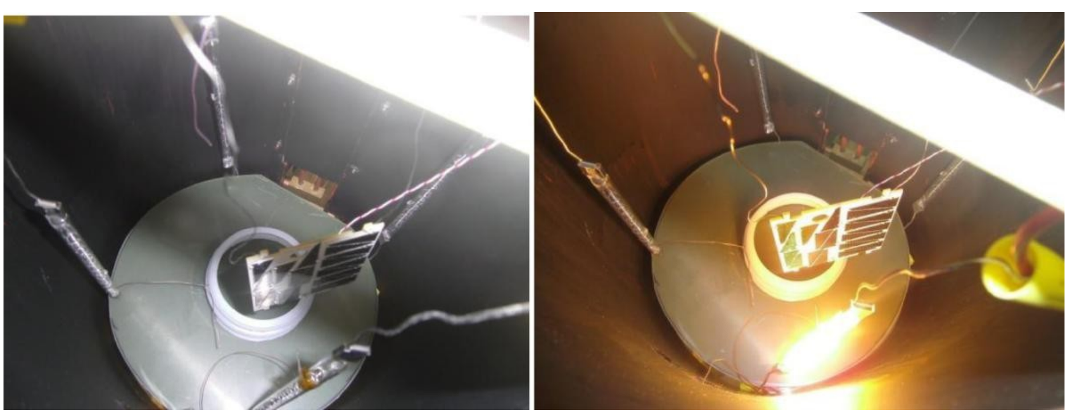


Figure XX-Pictures of solar panel in test chamber (with and without illumination)

Test Results

Outside Test: Following values were recorded

Outdoor Solar Panel Results

Open circuit	12.12	-	-
300	11.5	38.6	0.44
150	11.4	73	0.83
75	9.15	120	1.1
16	1.6	132	0.21

The following solar panel characteristic is then obtained:

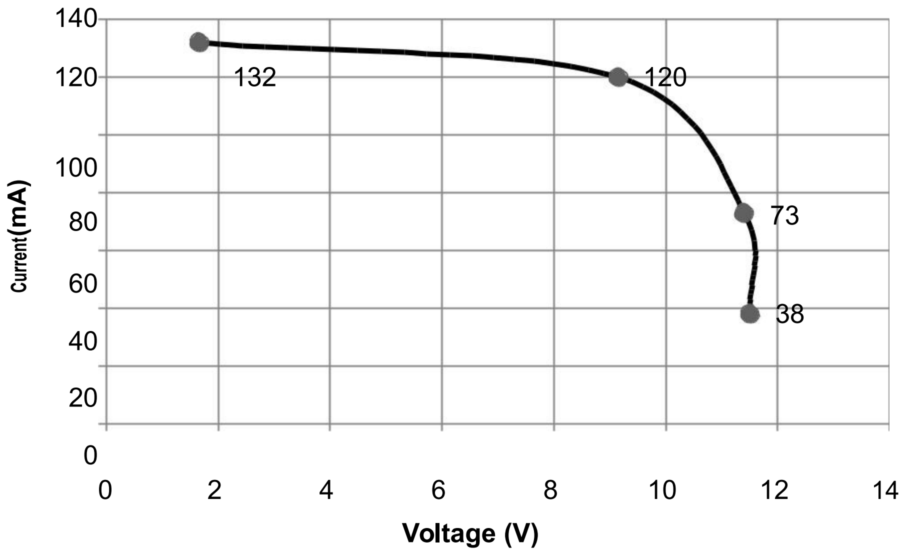


Figure XX-Power curve for outside test

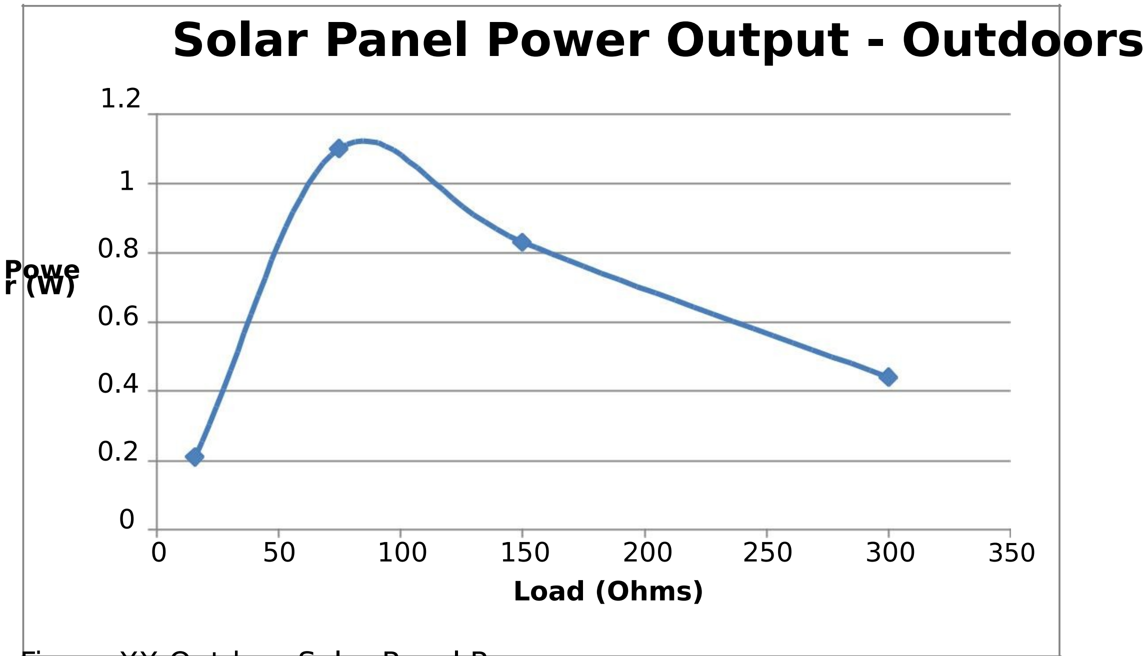


Figure XX-Outdoor Solar Panel Power

Inside Test

Similarly to the outside test, the following values were attained:

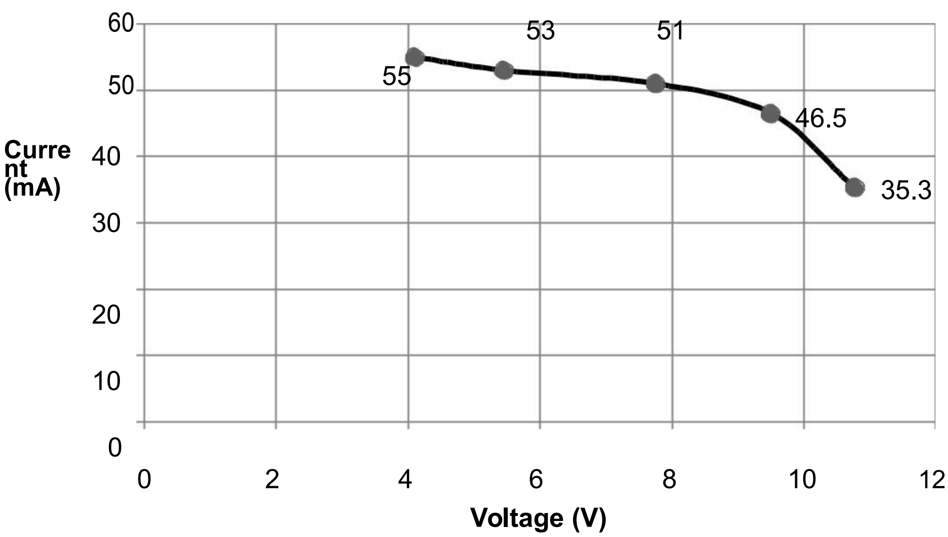


Figure XX-Power Curve for Indoor Test

The loads used for the inside test were: 75, 100, 150, 200, 300Ω. The values obtained are listed in the table below.

Open circuit	12.12	-	-

300	10.77	35.5	0.38
200	9.5	46.5	0.44
150	7.75	51.5	0.39
100	5.45	35.3	0.19
75	4.1	55	0.23

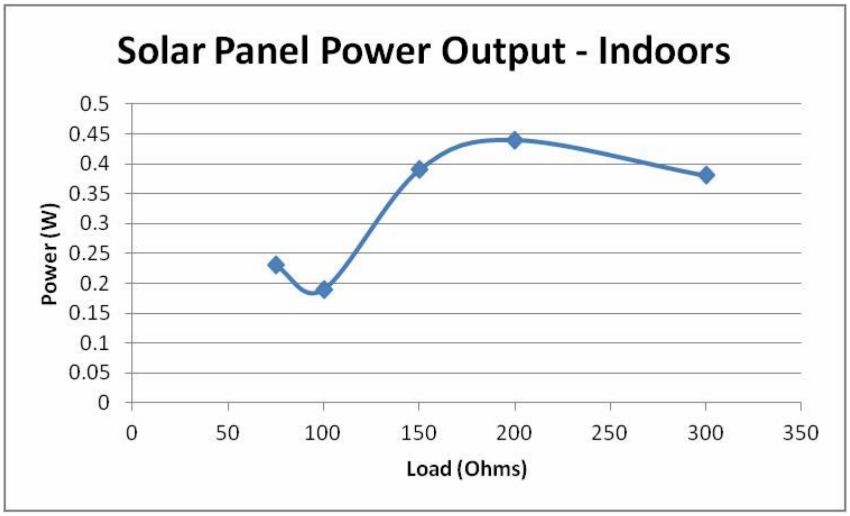


Figure XX-Indoor Solar Panel Power vs. Load Resistance

Appendix C-SSP 50835: 3.9.1.2

from pg. 3-87 of ISS Pressurized Volume Hardware Common Interface Requirements Document A- Integrated end items shall meet all performance requirements after being exposed to temperatures ranging from 0 to +50°C (32 to 122°F). This range is an envelope of the launch vehicle temperature environments shown in Table E.2.10-1. The end item may verify to a different temperature range listed in Table E.2.10-1 with approval from the VCB in accordance with the PMP.

Note: The ISS Program has decided not to levy the -50 to +50°C (-58 to +122°F) temperature range requirement for Russian Transport on end item developers. Temperature-sensitive end items planned for a winter launch on a Progress or Soyuz should request special handling with the JCCT.

B- Integrated end items shall meet all safety requirements when exposed to temperatures ranging from 0 to +50°C (32 to 122°F). This range is an envelope of the launch vehicle temperature environments shown in Table E.2.10-1. The end item may verify to a different temperature range listed in Table E.2.10-1 with approval from the VCB in accordance with the PMP.

Note: The ISS Program has decided not to levy the -50 to +50°C (-58 to +122°F) temperature range requirement for Russian Transport on end item developers. Temperature-sensitive end items planned for a winter launch on a Progress or Soyuz should request special handling with the JCCT.

C- Integrated end items shall meet all performance requirements when exposed to the ISS atmosphere temperatures ranging from 5 C to 45°C (41 to 113°F).

D- Integrated end items returning on a Soyuz vehicle shall withstand post-landing temperatures as low as -50°C (-58°F).