Modelling and power management of a CubeSat electrical power system

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Abstract– ***The Electric Power Subsystem (EPS) is a critical subsystem of any satellite. It provides the electric power needed for satellite subsystems. Any failure in EPS leads to satellite mission failure or reduction of the satellite capabilities. When designing an electrical power system for CubeSats, the available energy is usually estimated first, then the energy budget is calculated, taking into account the load consumption.***

As is known [1], to estimate the energy budget, we need to know the orbit average power, generated by SA. It depends on many factors: orbital parameters (duration of illuminated and shadow areas), electrical characteristics of solar cells, their temperature, and state of health; this parameter can be calculated or measured fairly easily. The power of SA also depends on the illumination which is the amount of solar power that reaches the solar cell per unit area. This energy is determined by the orientation scenario (and attitude control method) of a CubeSat, since it depends on the angle between the direction of sunlight and the normal to the SA surface. This paper aims to design the Electrical Power System (EPS) for CubeSat LEO satellites and investigate it power capability to satisfy the mission requirements. In this regard, accurate solar irradiance determination for the nadir-orientation scenario, Multi-Junction (MJ) solar cells calculations, backup batteries type and number.

Designing highly-efficient power supply module circuits is done. The power supply module circuits are designed based on commercial on the shelf components. The design of those circuits is satisfying the modularity concept and perform sample of the whole function of the power supply system for CubeSat.

I. Introduction

The combination of commercial-off-the-shelf microelectronic technologies developed for terrestrial use and adapted to the space environment and the increasing capabilities of low-power microelectronics, has encouraged the development of a new class of highly capable smaller, faster, cheaper satellites complementing the conventional large satellite systems [1], [2]. As a result, it is observed an increasing interest in small and micro satellites design and, more specifically, in the use of simulation analysis for the deployment the control of the satellite subsystems [3]–[6]. The electrical power system (EPS) plays a crucial role in the lifetime of small satellites. Indeed, the EPS should be effectively compliant to several requirements in order to provide power generation, management, storage, control, protection and distribution to the spacecraft payloads and platform equipment's during the entire mission life.

From the regulation point of view the EPS shall manage peak, pulse and transient power demands and the battery charge/discharge cycle by avoiding spacecraft instability and performance degradation.

At a certain level of abstraction, a quite general architecture of an EPS can be decomposed into four main blocks: a primary power source, an energy storage, a power management unit that deals with power conditioning and charge/discharge control, and a power distribution unit [7],[8].

The sizing of the subsystems and the design of an efficient power management strategy are complex and critical tasks usually undertaken in the satellite design phase [9], [10].

A conceptual design of a spacecraft power system involves an optimal selection of available technologies of different components, such as solar cells, solar arrays, batteries, and bus voltages. However, the electrical architecture of spacecrafts is not standard and shall, in general, need to be adapted nearly case by case. Thus, the identification of the topology is the preliminary step for the EPS design. There are several basic topologies, that can be classified based on two main criteria: the energy transfer and the voltage main bus regulation [11], [12].

Power necessary for subsystem and payload operation at necessary current and voltage levels must be supplied by the electric power system. On-command power on/off for the S/C subsystems and payloads. Ensures that payloads and satellite components are safeguarded from power failures in components that harm the entire system. Telemetry is used to provide voltage, current, and temperature measurements for power management and status. guarantees that the necessary power will be available for the anticipated mission duration [11],[12].

The goal is to create a small earth observation satellite's electric power management system Gerber file of (printed circuit board-PCB) to be send to manufacture of PCB to implement it (by going through the following steps:

- Design the operating scenario of the cub-sat which considered the worst-case scenario

- Perform a power budget calculation for the worst-case scenario to verify the design of the power supply configuration.
- selecting the optimum configuration of the commercial of the shelf components subsystem.
- Develop the schematic diagram of the power management unit (printed circuit board-PCB)
- Perform a simulation-by-simulation environment (Linear Technology simulation Environment) to estimate the behavior of the designed circuit
- Develop the Gerber-file of the design power management unit (PCB) to be send to manufacture of PCB to implement the design

The power regulation technique used in this spacecraft is DET [3],[4].

II. SPACE MISSION SPECIFICATIONS AND POWER SUPPLY SYSTEM DESCRIPTION

A. MISSION ANALYSIS

The mission of the proposed satellite (MENA SAT) was earth observation and remote sensing. The orbit used is sun-synchronous orbit because the satellite in this orbit constantly sees the sun at the same angle and maintains a constant angle between the direction of the sun and the orbit at altitude of 600 km and inclination between 97:99°. The maximum eclipse time of the orbit period was 35.45 min and the orbit time was calculated 96.6 min. The bus voltage of the spacecraft was 18v. The mission's lifetime needed is 2 years [1]. The table 1 contains the required space parameters for the mission. The following figure 1 shows the spacecraft layout.

Orbit used	Sun-synchronous	
Lifetime	2	Year
Orbit type	Polar orbit	
Inclination	97:99	Deg
Altitude	600	Km
Orbit period	96.6	Min
Max eclipse time	35.4556	Min
Min sun time	61.144	min
Bus voltage	18	volt
Bus regulation	3/3.3/5/18	Volt
Solar constant	1356	W/ m²

Table 1:Space mission parameters for the proposed spacecraft

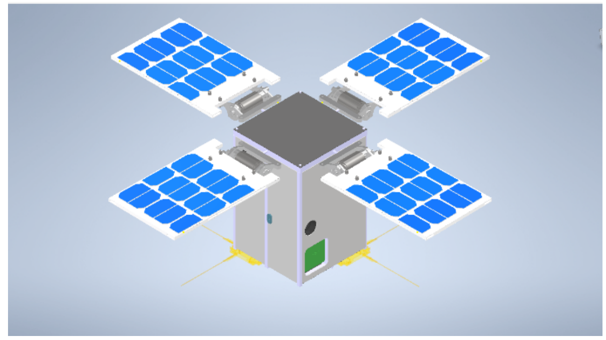

Fig. 1 MENA Sat layout

Figure 2 shows a simple configuration of the electric power subsystem, as it consists of primary energy source, secondary energy source, PDCU and loads.

The primary energy source is the sun, that it's energy is transformed into electric using solar cells arranged in 4 solar deployed panels.

When the primary source of energy is unavailable, such as during eclipses or when it is insufficient to power the spacecraft loads, or when we use all of the payloads and radios simultaneously and the peak power consumption exceeds the power produced by the solar panels, the secondary source of energy is used.

The PDCU is responsible for delivering power from solar arrays and batteries and distribute the power to the loads.

The loads are the rest of the satellite that consume power to achieve the mission requirements.

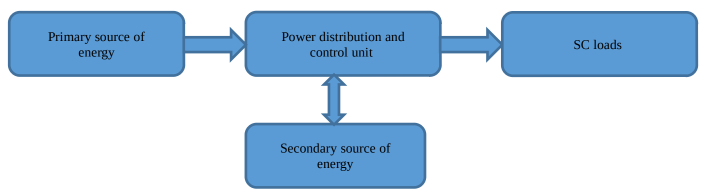

Fig. 2 simple configuration of EPS

B. SOLAR ARRAYS (SA) ASSUMED PARAMETERS AND ELECTRICAL REQUIREMENTS

The solar array (SA) could be in the configuration of body mounted or in deployable solar panels. The SA orientation has a great effect on the power generation profile [5].

In this mission, the SA configuration is a deployable solar panel as shown in fig 1. According to the calculations and the area available on the satellite.

The trade-off analysis of the solar cell characteristics should be performed to choose the most suitable solar cell technology. As a result of the trade-off analysis the ultra-Triple Junction (UTJ) Solar Cell was chosen with electrical and mechanical parameters suitable for the configuration of the solar panels [7],[8].

Tables 2,3 show the electrical and mechanical parameters of Ultra Triple Junction (UTJ) Solar Cell

Length	7	cm
Width	3.5	Cm
Area	24.5	cm²
Weight	84	mg/ cm²

Table 2 Solar Cell Mechanical Parameter

Voltage open circuit	V_{oc}	2.665	V
Current density short circuit	J_{sc}	17.05	mA/ cm²
Volt at max power	V_{max}	2.35	V
Current density at Max power	J_{max}	16.3	mA/ cm²
Max power	P_{max}	0.9384725	W
Efficiency	η	28.3%	

Table 3 Solar Cell Electrical Parameters

The power generated from one solar cell:

$$P = \text{solar constant} * \eta * \text{solar cell area} * \cos(\alpha)$$

Where α … angle between the solar flux and the normal to the surface of solar cell as shown in fig 3

normal

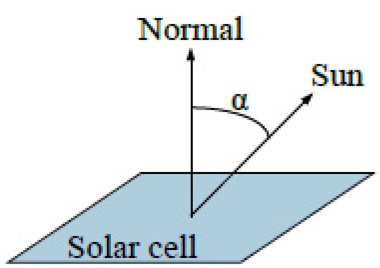

Figure 3 the angle alpha

Next table will show the degradation factors that affect the solar cell and reduce it’s efficiency and performance

Mismatch & fabrication	0.98
Wiring & diode loss	0.96
Packing factor	0.85
Temp. loss factor	1
Shadowing losses	0.9
Ultraviolet degradation	0.98
Radiation degradation	0.95
Fatigue (thermal cycling)	0.98
Micrometeoroid loss	0.98
Additional margin	1

Table 4 Solar Cell performance Degradation factors

The power that can be generated from the solar arrays can be calculated using equations [1].

$$P_{SA} = (((P_d * T_d)/X_d) + ((P_e * T_e)/X_e))/T_d)$$

$$P_0 = \text{solar constant} * \eta$$

$$P_{BOL} = P_0 * \textit{inherent degradation} * \cos\theta$$

$$P_{EOL} = P_{BOL} * \textit{life degradation}$$

$$\text{Area of arrays} = P_{SA}/P_{EOL}$$

$$\text{No of solar cells in series} = \frac{\text{Bus voltage} * 1.25}{\text{solar cell voltage}}$$

$$\text{No of strings} = \frac{\text{No of solar cells}}{\text{No of solar cells in series}}$$

Where:

P_0 ...... Power output with sun normal to the solar cell

P_{SA} ..... Power required from solar arrays

P_0 …… output power with sun normal to the solar cell

P_{BOL}..... Power produced at beginning of life

P_{EOL} …. Power produced at end of life

The calculations are gathered in table 5

parameter	value	unit
Power generated from solar panels	59.2315	W
Power required from solar panel (Psa)	62.3508	W
Power output with sun normal to the solar cell (Po)	383.748	W/ m²
Power produced at beginning of life (PBOL)	222.253	W/ m²
Power produced at end of life (PEOL)	220.0368	W/ m²
Area of solar array (4 panels)	0.28336561	m²
Area of one solar panel	0.070841403	m²
Total no of solar cells actually	63	cells
No of solar cells in series	9	cells
No of strings (in parallel)	7	strings

Table 5 The calculations of solar panels of the proposed spacecraft

C. Battery Description

In recent years, numerous LEO missions have demonstrated the viability of Li-Po batteries. They were a strong candidate because of their wide market accessibility, low cost, high specific energy, and little maintenance requirements, such as low self-discharge current and storage at room temperature. The satellite batteries were constructed from a number of cells coupled in series and parallel, adding up to the desired voltage, maximum currents for charge and discharge, and overall capacity. The most crucial quantity to determine is the total capacity, which is the sum of the capacities of each individual cell, whether they are in series or parallel. Parallel and series configurations must be defined concurrently in accordance with the limits of other subsystems. The method used to choose these values for the MENA will be discussed in the sections that follow.

D. BATTERIES CONFIGURATION

There are two main drivers for selecting battery voltage:

1. Battery charger's voltage
2. Bus requirements

Through these drivers, choosing the battery voltage and the number of batteries in series for one string can be calculated. For the MENA sat 5 batteries in series was the chosen due to achieve the bus voltage and the battery charger.

The charge and discharge current of the batteries must be determined to prevent any damage would occur to the batteries and loads.
The value of the current needed was calculated from loads analysis, 3 parallel strings of batteries.

The batteries sizing calculations [9] is summarized in the following table (table 6)

parameter	Energy [Wh]
Energy required at eclipse	10.89077847
Energy required at day	20.8196682
Average energy produced at day	32.19082814
Accumulated energy	11.37115994
Energy for batteries	45.48463978
Voltage of batteries	18
Capacity for batteries	2.526924432
No of batteries in parallel	Almost 3
No of batteries in series	Almost 5

Table 6 The calculations of batteries sizing of the proposed spacecraft

E. MISSION SCENARIOS WITH POWER BUDGET CALCULATIONS

The next step is obtaining information on how much power each load will demand and how it will be used during operation. By dividing the orbit into 8 modes, this was accomplished [13]. Table 3 explain the modes of operations.

DTS	daylight with transmission and with science
DTnS	daylight with transmission and without science
DnTS	daylight without transmission and with science
DnTnS	daylight without transmission and without science
ETS	eclipse with transmission and with science
ETnS	eclipse with transmission and without science
EnTS	eclipse without transmission and with science
EnTnS	eclipse without transmission and without science

TABLE 7 The modes of operation of the proposed spacecraft

For the mode DTS it's the most consuming power that all sub systems are working together at the same time at sun time. For the mode DnTnS it's the safe mode at sun time that it's the least consuming power at sun time where the payload and the transmitter are off and the necessary subsystems are working. Thus also for the eclipse time modes.

According to survey, it was found that the most familiar platform for the 12U cubesats consist of:

1. EPS
2. Payload
3. Command&data handling
4. S-band transmitter
5. VHF beacon transmitter
6. Transceiver UHF
7. GPS receiver
8. Propulsion system
9. ADCS

The following table containing the subsystems with their power consumption

subsystem	Power[w]
EPS	5
payload	7
Command&data handling	0.22
S-band transmitter	5
VHF beacon transmitter	1.5
Transceiver UHF	2.71
GPS receiver	1
Propulsion system	10
ADCS	5

Table 8 The power consumption of the proposed spacecraft subsystems

The calculations here depend on the power value of each subsystem and if it on or off in the mode
The next table will show the power budget calculation of the modes

The subsystem	DTS [w]	DnTS [w]	DTnS [w]	DnTnS [w]	ETS [w]	EnTS [w]	ETnS [w]	EnTnS [w]	Maneuver [w]
EPS	5	5	5	5	3	3	3	3	5
payload	7	7	0	0	7	7	0	0	0
Command&data handling	0.22	0.22	0.22	0.22	0.22	0.22	0.22	0.22	0.22
S-band transmitter	5	0	5	0	5	0	5	0	0
VHF beacon transmitter	1.5	1.5	1.5	1.5	1.5	1.5	1.5	1.5	1.5
Transceiver UHF	2.71	2.71	2.71	2.71	2.71	2.71	2.71	2.71	2.71
GPS receiver	1	1	1	1	1	1	1	1	1
Propulsion system	0	0	0	0	0	0	0	0	10
ADCS	5	5	5	5	5	5	5	5	5
total	27.43	22.43	20.43	15.43	25.43	20.43	18.43	13.43	25.43

Table 9 THE POWER BUDGET CALCULATION OF THE MODES

III. RESULTS AND DESCUSSION

A- Power generated : Using the above calculations and simulations, the power generated from the solar panels is shown in the next figure

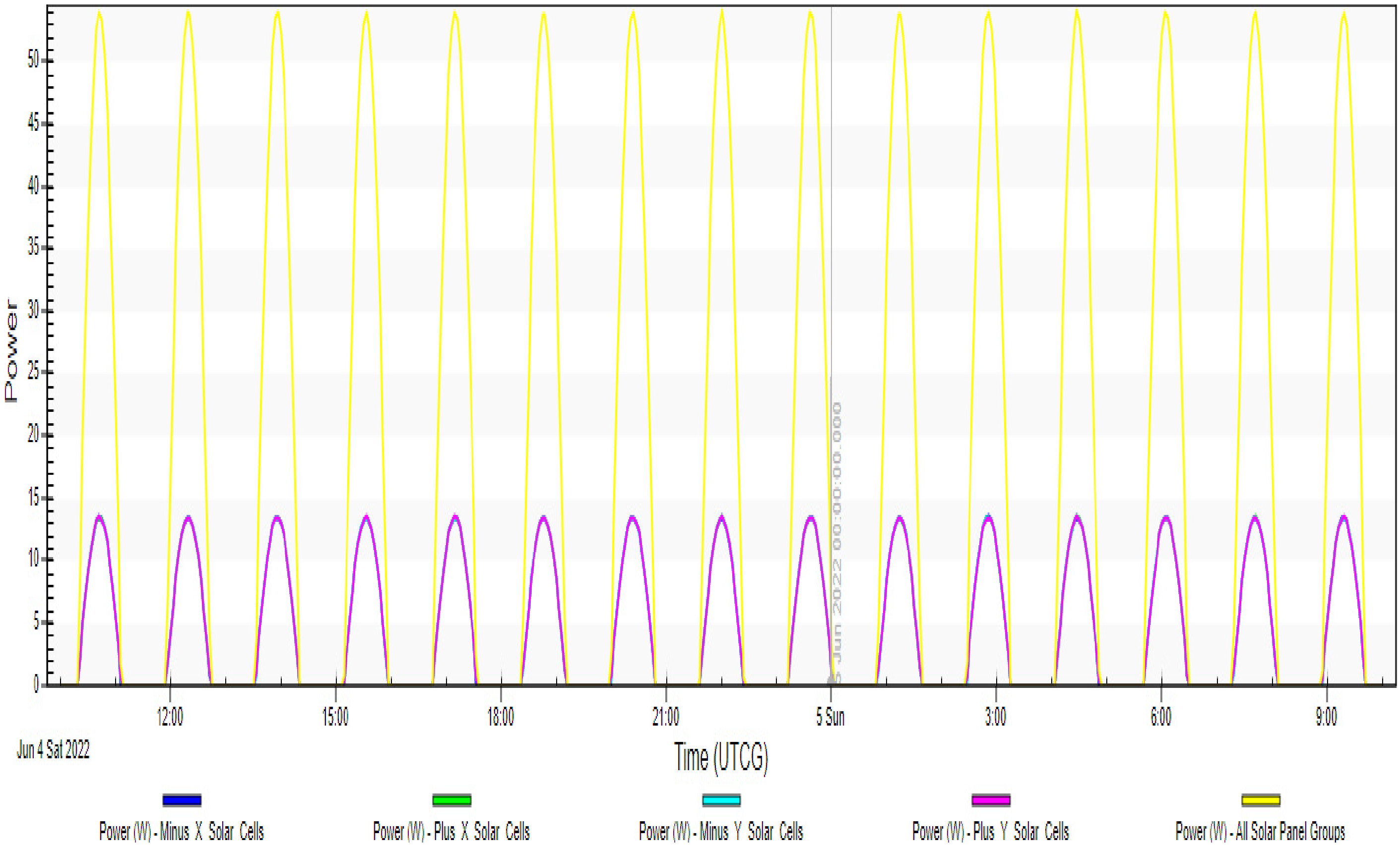

Fig. 4 power generated from solar panels

B - The load profile for one day: this graph shows the power consumptions of loads for one day, the maximum, minimum and average power generated [9].

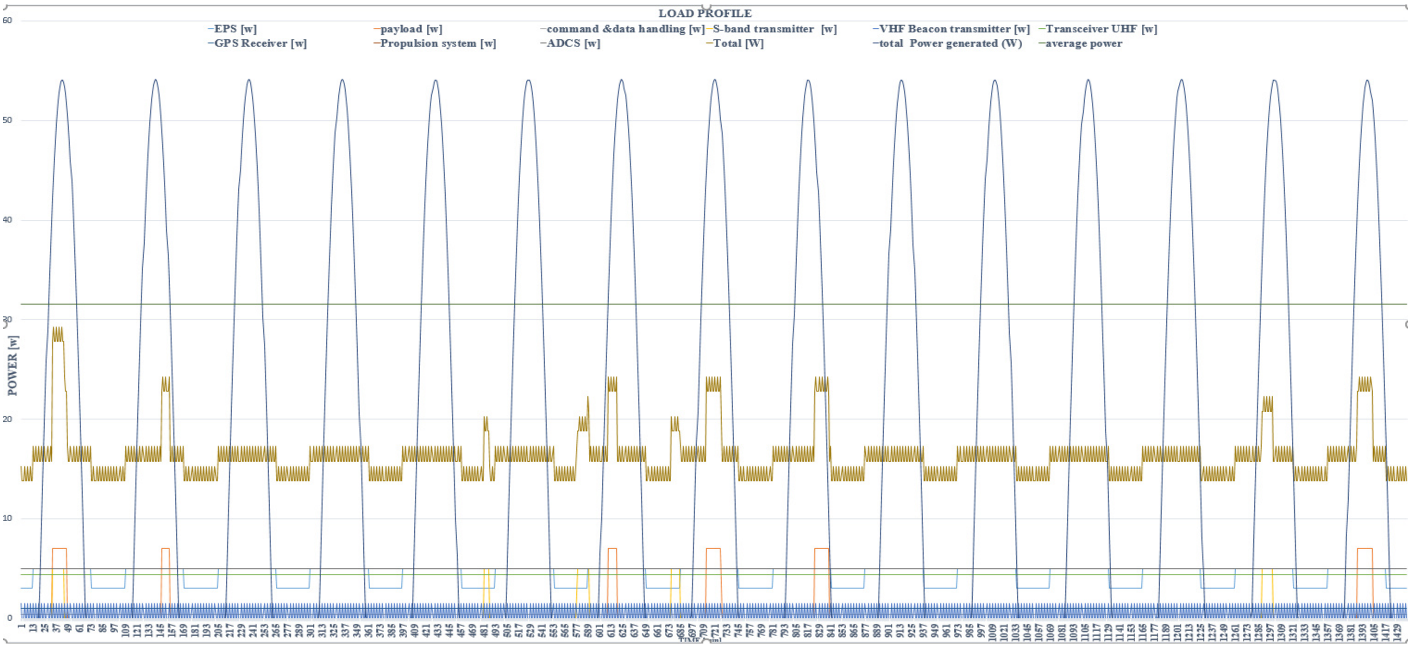

Fig. 5 loads profile for 1 day

C – The battery profile: The profile indicates the charge/discharge of the batteries during one day scenario, but the profile was created was the worst case scenario to make the batteries expose to hardworking to its fatigue.
The worst case scenario was that at session time we use batteries for providing power for the satellite not using solar panels at sun time.

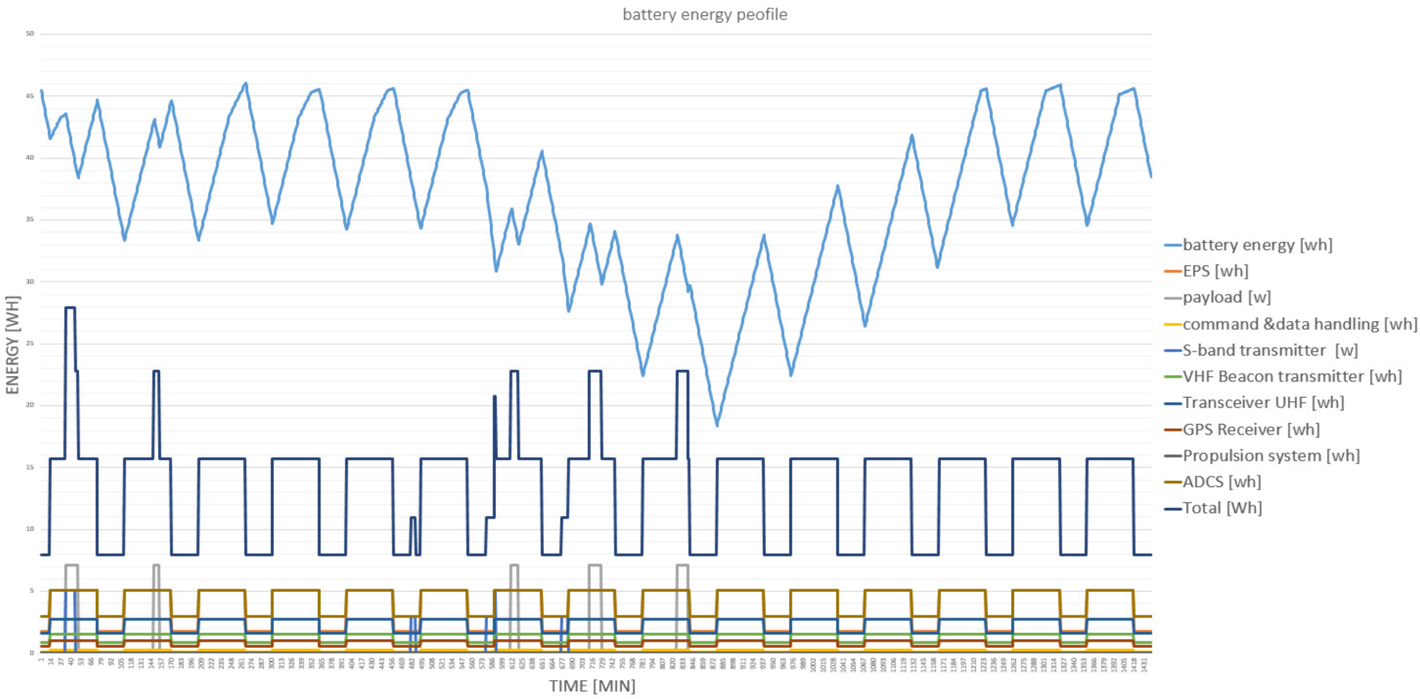

Fig. 6 battery profile

IV. The design of PCDU

The PCDU here is a printed circuit board that incorporates some chosen electronic components [6], [10].

PCDU-MTC Evaluation board block diagram

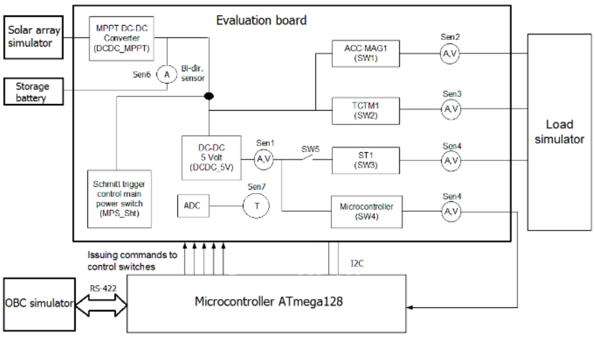

Fig.7 PCDU block diagram

The functional requirement

The board is designed to functionally perform the following:

- Perform the main bus under voltage protection through a hardware implemented Schmitt trigger IC.
- Converts solar array voltage to 3.3 volts using dc-dc converter and deliver the output to load switch.
- Deliver bus power directly before dc-dc converter to load switch.
- Voltage and current measurement through V-I sensor.
- Temperature measurement through temperature sensor.

Components technical specification

#	Component designation	Part number	Function	Q
1	DCDC_MPPT	TBD	Track MPP	1
2	Schmitt trigger	LTC4365 STD16NF06L	Schmitt trigger controller main power switch	1
3	DCDC_5V	TPS54331	5 Volt DC-DC converter	1
4	Sen1	LTC4151	I2C voltage and current measurement of DCDC_5V converter	1
	Sen2	LTC4151	I2C voltage and current measurement of DCDC-5V converter of SW1	1
	Sen3	LTC4151	I2C voltage and current measurement of DCDC-5V converter of SW2	1
	Sen4	LTC4151	I2C voltage and current measurement of DCDC-5V converter of SW3	1
	Sen5	LTC4151	I2C voltage and current measurement of DCDC-5V converter of SW4	1
	Sen6	LT1787	Precision high side current sense amplifiers	1
	Sen7	LM135z	Temperature sensor	1
	ADC	LTC2309	Analog to digital converter for temperature sensor	1
	SW1	SIP32419	Switch of ACC-MAG1	1
	SW2	SIP32429	Switch of TCTM1	1
	SW3	MAX890L	Switch of ST1	1
	SW4	LTC4361 IRF7910	Switch with Over voltage / Over current Protection of Microcontroller	1
	SW5	IRF7910	Switch to check voltage drop in case of using reserve converter	1

1 7	MC	ATmega25 60	Include EB firmware	1

Table 10 States the PCDU board electronic components technical specification

Control circuits design and simulation

Schmitt trigger: For the suitable working of all electrical and electronic devices, it is recommended to allow voltage at prescribed limits. Voltage fluctuations in electric power supply certainly have adverse effects on connected loads. These fluctuations can be of over voltage and under voltages.

Operation limits: To adjust the limits of over/under voltage the resistors values were adjusted according to datasheet of the IC.

The operation limits here are [10:21] Volt.

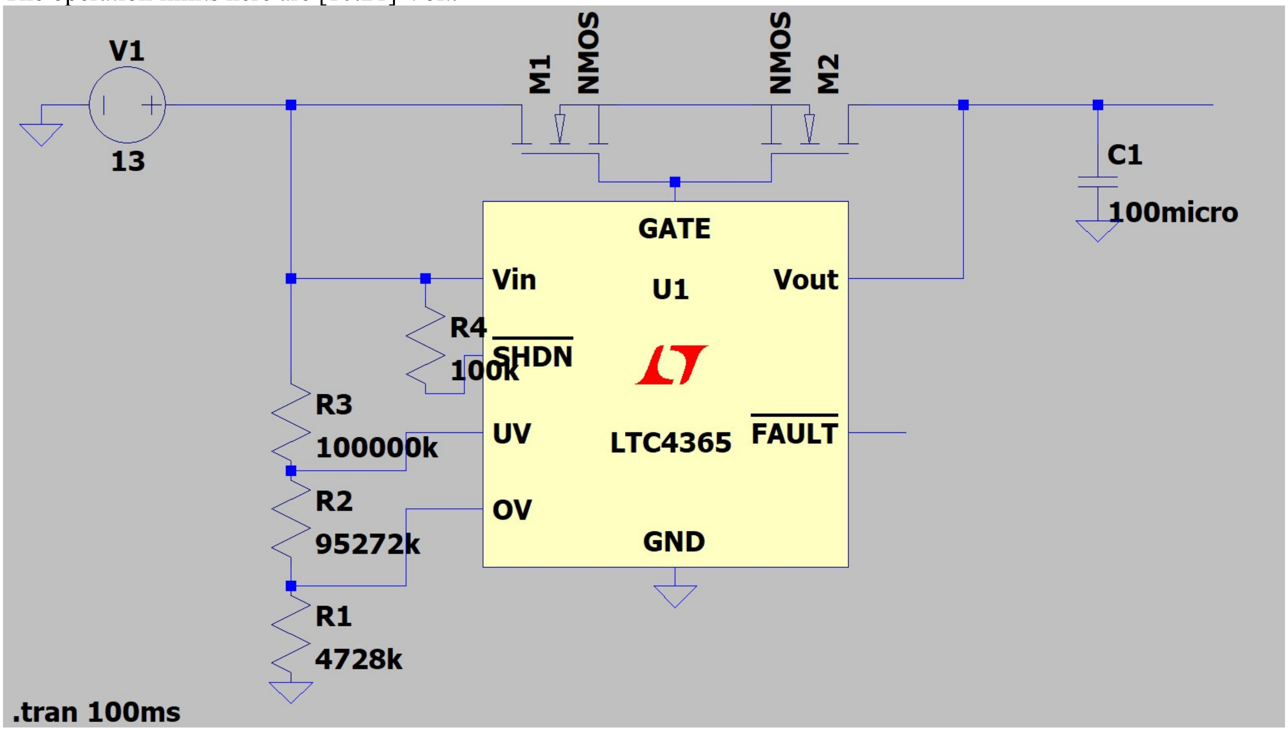

Fig. 7 Schmitt trigger IC

Simulations of Schmitt trigger:

Firstly operation limits: where the input value is the green line and it's value 13V, the output value is the blue line and it's value almost 13V.

(The X-axis is time in ms, the Y-axis is voltage in v)

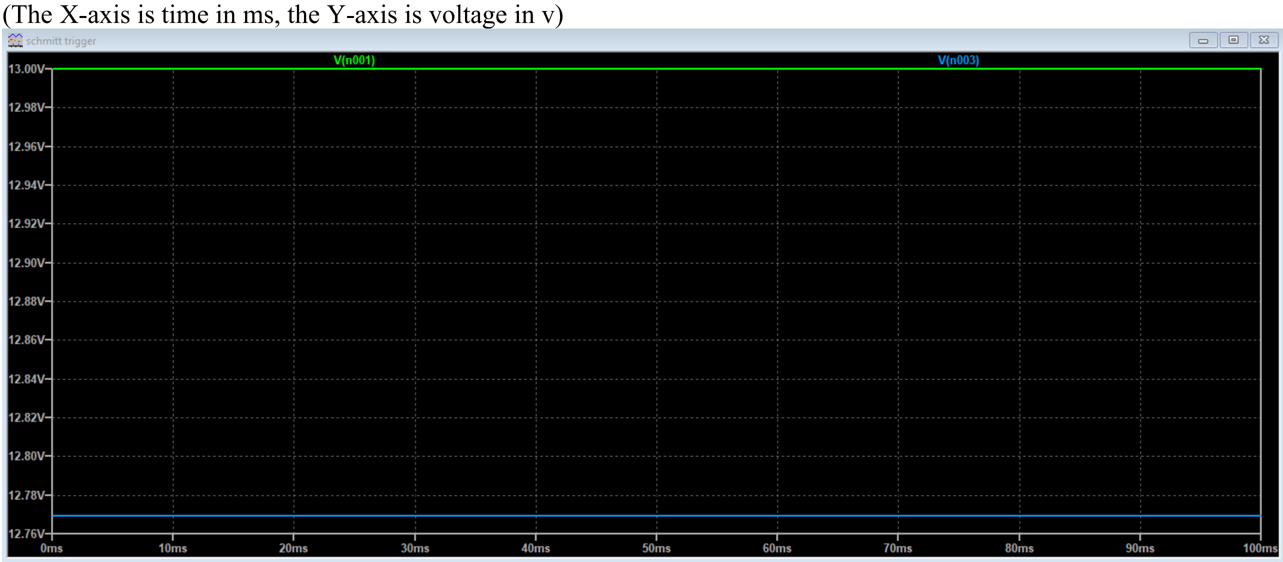

Fig. 8 operation limits

Secondly out of operation limits (over voltage): where the input vale is the green line and it's value 22V, the output value is the blue line and it's value 0V.

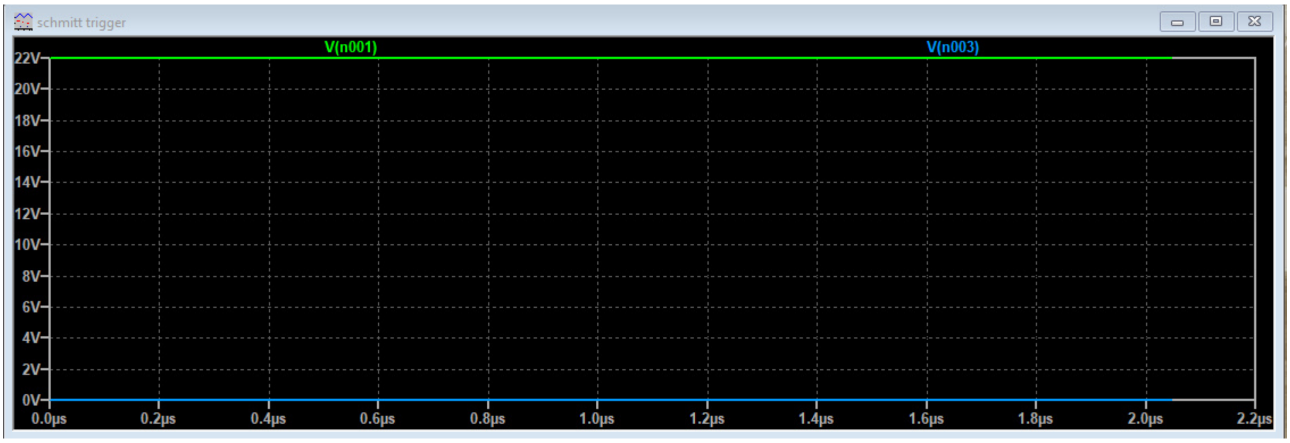

Fig. 9 out of limits

DC-DC converter

The converter efficiently converts power from an input voltage source to a lower output voltage

In the CubeSat the loads, voltage vary from one to another.

According to the high bus voltage, every load should have step down converter for regulating the incoming voltage.

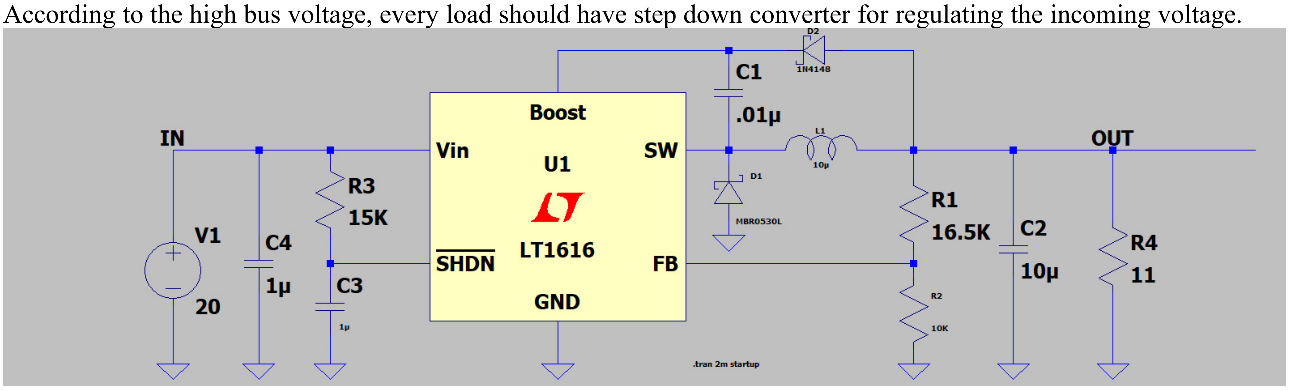

Fig. 10 DC-DC converter IC

The input value is the green line it's value 20V and the output is blue line it's value 3.3v after response time of 0.95 ms (The X-axis is time in ms, the Y-axis is voltage in v)

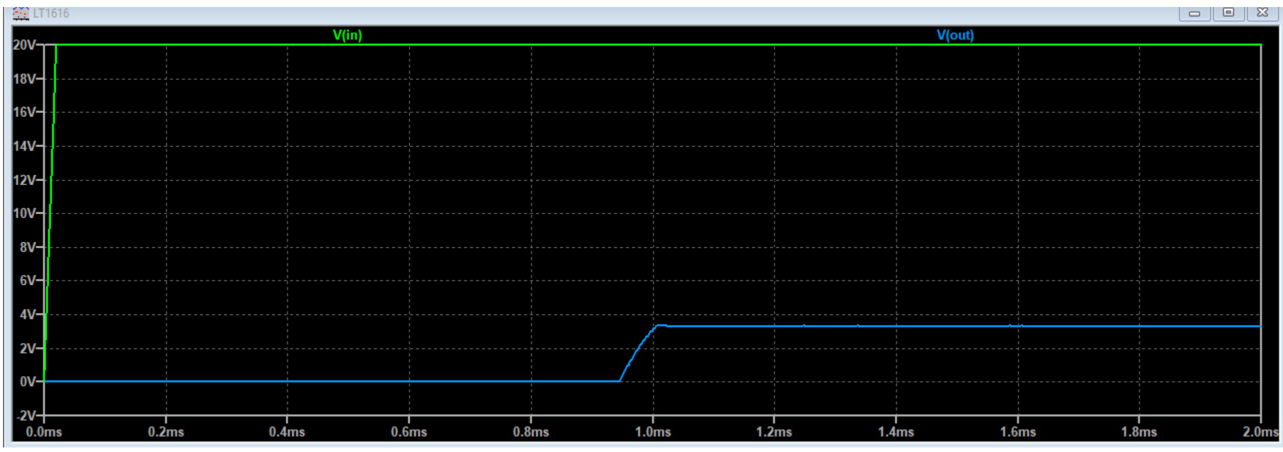

Fig. 11 DC-DC converter simulation

The design of pcb

The Altium Designer is used to implement the design. The following circuit diagram with describe the power management unit module of Cube-sat. the technique used is direct energy transfer (DET) as there is no dynamics on illumination for the designed Cube-sat. The input from solar array is from connector J2. The IC (INA226) is direct connected to the input of SA as it wors as voltage-current sensor to estimate the output of the SA at any time. The IC The LTC2309 is used as analog to digital converter (ADC) which is a low noise, low power, 8-channel, 12-bit successive approximation ADC with an I^2C compatible serial interface. This ADC includes an internal reference and a fully differential sample-and-hold circuit to reduce common mode noise. The LTC2309 operates from an internal clock to achieve a fast 1.3µs conversion time. The IC LTC4365 is used as The LTC4365 protects applications where power supply input voltages may be too high, too low or even negative. It does this by controlling the gate voltages of a pair of external N-channel MOSFETs to ensure that the output stays within a safe operating range. The LTC4365 can withstand voltages between –40V and 60V and has an operating range of 2.5V to 34V, while consuming only 125µA in normal operation.

Two comparator inputs allow configuration of the overvoltage (OV) and undervoltage (UV) set points using an external resistive divider. A shutdown pin provides external control for enabling and disabling the MOSFETs as well as placing the device in a low current shutdown state. A fault output provides status of the gate pin pulling low. A fault is indicated when the part is in shutdown or the input voltage is outside the UV and OV set points. The MOSFT switch is used as main switch of the power management unite. The battery output is connected to the connector J5. The IC SiP32419 is a SiP32419 and MAX9611 are load switches that integrate multiple control features that simplify the design and increase the reliability of the circuitry connected to the switch. Both devices are 56 mΩ switches designed to operate in the 6 V to 28 V range. An internally generated gate drive voltage ensures good R_{ON} linearity over the input voltage operating range. Both those switches are used to connect the load to the designed power management unit (PCB Card). The loads are connected through connector J1. The following figure is the layout of the designed power management unit (PCB Card) with the whole connection between the SA and storage battery to the loads with the whole controlling and monitoring circuit for the designed PCB. Figure 14 is the PCB layout of the designed power management unit (PCB Card) which considered as one of the Gerber files which will send to the manufacture of the PCBs to realize the design.

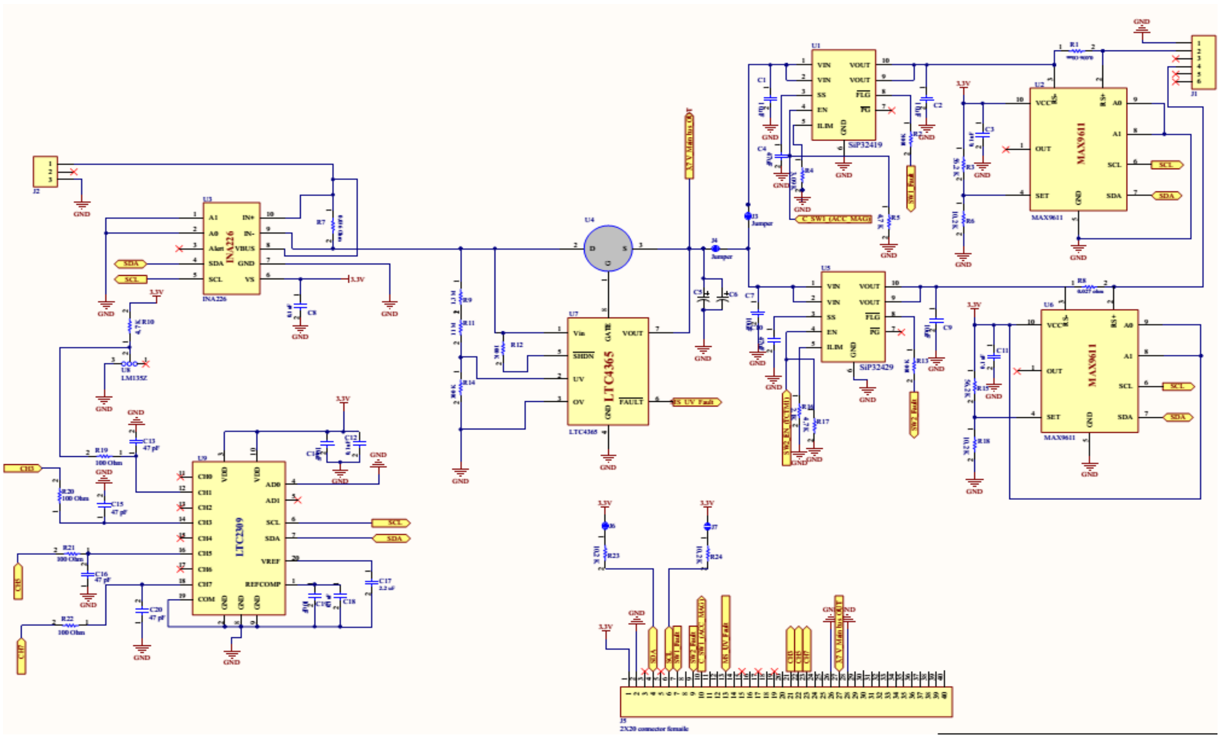

Fig. 13 scheme in Altium

Then updating the PCB

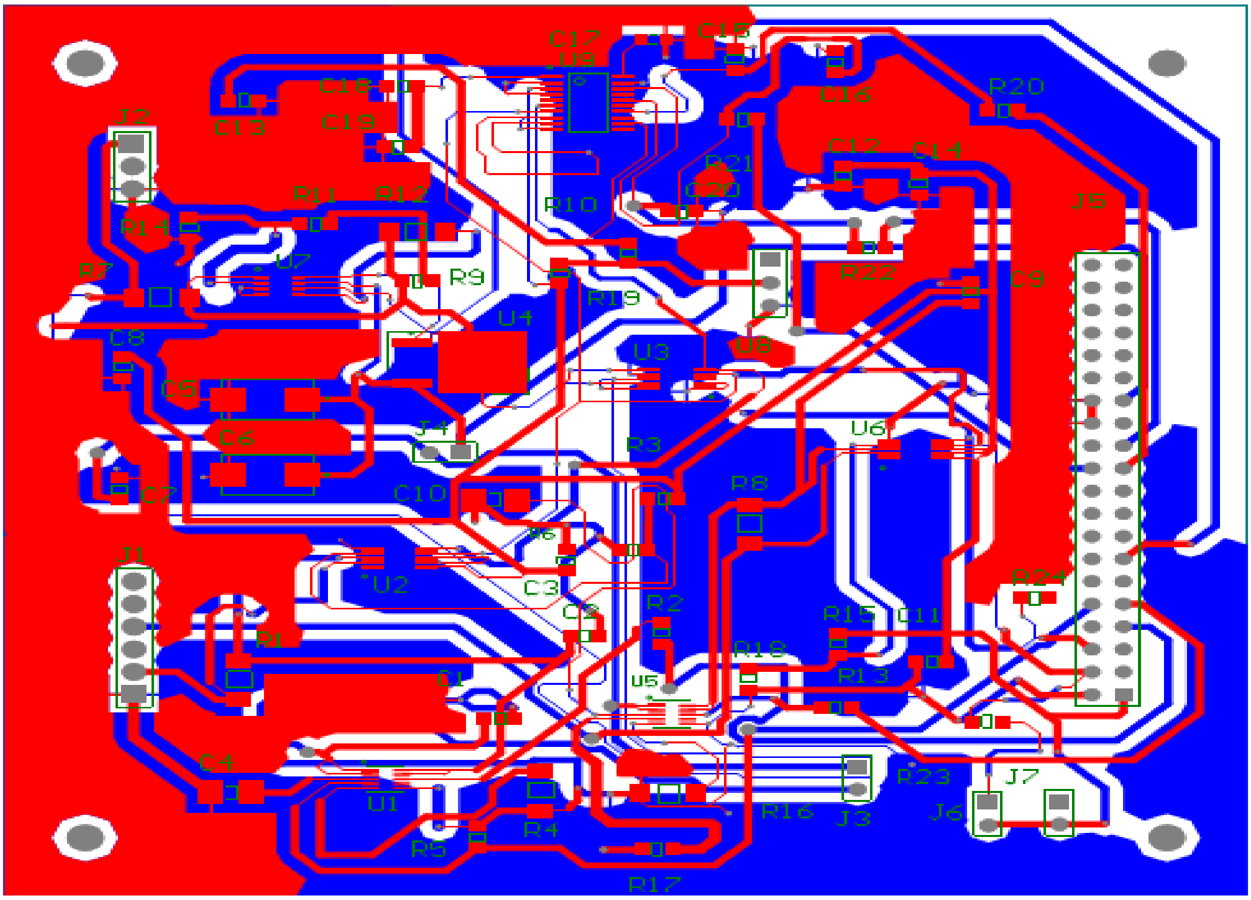

Fig. 14 PCB in Altium

Comment	Description	Designator	Footprint	LibRef	Quantity
MLCC Capacitor	MLCC Capacitor	C1, C2, C7, C9, C12, C19	0805	CC0805KKX5R8BB106	6
CL21B104KBCNNNC	MLCC Capacitor	C3, C8, C11, C14, C18	0805	CL21B104KBCNNNC	5
GRJ32ER71A476KE11	MLCC Capacitor	C4, C10	CAP_cer_1210	GRJ32ER71A476KE11	2
T495X476K035ATE300	Capacitor-Tantalum	C5, C6	2917 (7343 Metric)	T495X476K035ATE300	2
Capacitor	MLCC capacitor	C13, C15, C16, C20	0805	08052C473JAZ2A	4
GRM188R61E225KA12D _1	MLCC Capacitor	C17	GRM188R61E225KA12D	GRM188R61E225KA12D _1	1
M20-7910642R	Header, 6-Pin	J1	HDR1X6F	M20-7910642R	1
M20-7910342R	Header, 3-Pin	J2	HDR1X3M	M20-7910342R	1
Jumper	Jumper	J3, J4, J6, J7	Jumper	Jumper	4
2X20 connector femaile	40 Pin Stack Through Connector Femaile 2x20	J5	StackThrough40	M20-6102045	1
ERJ-8CWFR036V	Thick Film Resistor	R1, R7	1206	ERJ-8CWFR036V	2
ERA-6AEB104V	Resistor-SMD	R2, R13	0805	ERA-6AEB104V	2
CRCW040256K2FKED	Thick Film Resistor	R3, R15	0805	CRCW040256K2FKED	2
CRCW12063K09FKEA	Resistor-SMD	R4	1206	CRCW12063K09FKEA	1
ERJ-P06F4701V	Resistor-SMD	R5, R10, R17	0805	ERJ-P06F4701V	3
CRCW040210K2FKED	Thick Film Resistor	R6, R18, R23, R24	0805	CRCW040210K2FKED	4
ERJ-8CWFR027V	Resistor-SMD	R8	1206	ERJ-8CWFR027V	1
RC0805FR-071M3L	Resistor-SMD	R9	0805	RC0805FR-071M3L	1
RMCF0805JT11M0	SMD-Resistor	R11	0805	RMCF0805JT11M0	1
CRCW1206100KFKEA	Resistor-SMD	R12	1206	CRCW1206100KFKEA	1
ERA-6AEB304V	SMD-Resistor	R14	0805	ERA-6AEB304V	1
CRCW12062K10FKEA	Resistor-SMD	R16	1206	CRCW12062K10FKEA	1
RMCF0603JT100R	Resistor-SMD	R19, R20, R21, R22	0805	RMCF0603JT100R	4
SIP32419DN-T1-GE4	Load switch with progammable current limit	U1	DFN-10	SIP32419DN-T1-GE4	1
MAX9611AUB+T	IC OPAMP CURR SENS 2.5MHZ 10UMAX	U2, U6	10-MSOP_MAX9611	MAX9611AUB+T	2
INA226AIDGSR	INA226_IC MONITOR PWR/CURR BIDIR 10MSOP	U3	VSSOP_INA226AIDGSR	INA226AIDGSR	1
N-CHANNEL 60V - 0.060 ? - 24A DPAK/IPAK STripFET™ II POWER MOSFET	Power MOSFET	U4	STD16NF06L3N	STD16NF06LT4	1
SIP32429DN-T1-GE4	Load switch with progammable current limit (Autoreset)	U5	DFN-10	SIP32429DN-T1-GE4	1
Overvoltage, Undervoltage	Overvoltage, Undervoltage and Reverse Supply Protection Controller	U7	TS8 PACKAGE	LTC4365CTS8#TRMPBF	1
LM135Z	Temprature Sensor	U8	LM2135Z	LM135Z	1
LTC2309HF#PBF	8-Channel, 12-Bit SAR ADC with I2C Interface	U9	LTC2309HF#PBF	LTC2309HF#PBF	1

Fig. 15 List of components

CONCLUSION

Using the demo board components, the functionality of the circuit has been proven. The solar panels will provide ample power and the battery and supercapacitor bank both have the required energy storage to make the circuit operational during eclipse. There is a board design in place to implement the circuit in the satellite package. This being said, there still remains a vast amount of work to be done on the power system. Though the boards have been designed, there are still a wide range of tests that must be performed on the system to ensure that they not only function as intended, but are also able to withstand some of the harsh conditions and cycling requirements demanded by the mission. The first step will be to ensure that the circuit is in fact functional and that all of the connections have been made to ensure that power is delivered cleanly and efficiently. Once this testing and any necessary revisions have been made, the final testing process will be conditional testing which will have to done at the military technical collage (MTC) Labs. These tests will come as new teams continue to work on this CubeSat package and understanding that this work may change the system requirements this team is confident that it has delivered a system that is not only functional, but malleable as well. By using the designed converters as the voltage regulators, the system can be easily modified to accommodate changing load needs in the satellite's electronic components.

While every component of a satellite architecture is crucial and necessary, the power subsystem is always expected to be the first to operate without issues. This system's failure will probably result in a catastrophic mission failure. The paper proves the methodology to design, simulate, and model a power subsystem. Analyzing various battery topologies, solar panel configurations, and solar cell configurations is also rather simple.

The future work is to implement this design by manufacturing the designed PCB. A full scope of testing is also needed to validate the design and also to qualify the designed power management (PCB card) to work in a cube-sat in space environment.

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