Study and Simulation of a Low Cost and Intelligent Electrical Power Sub-System Unit for Cube Satellite

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ABSTRACT
Satellite industry almost completely dependent on centralized Electrical Power System (EPS) designs. Maximum EPS is custom designed. There are a few manufactures that make their designs available for commercial use. Most of these designs conform to the most common standard that uses three distributed buses. Today large satellite follows single bus voltage distributed architecture. Therefore single bus voltage distributed architecture is now become a mainstream architecture for small cube satellite. Power system unit has distributed voltage which is more than one, such as a low voltage bus and a high voltage bus. There is generally one bus voltage within a subsystem. Every subsystem module is accountable for additional regulation or point-of-load regulation. With the developments of small, very efficient, monolithic dc-dc converters, this paper shows the design and simulation of possible implementation of distributed architecture at the cube satellite scale. The goal of this paper is to design an efficient electrical power system (EPS) with a great utility, capable of being used for several missions, without having to redesign the system every time.

Keywords
Power system of cube satellite, small satellite power system, power system model, satellite power system.

1. INTRODUCTION
The main source of power of a satellite is solar panel which gets power from the direct sunlight. And satellite also has a self-storage system which stores that power into the battery for backup power. This storage power has used for communication sub system, payloads, camera and other sensors as well as to operate all the sub systems of satellite that makes the satellite into its orbit and make its proper function. Before making any calculation, it needs to realize that the total power output from the solar panel directly depends on the area of solar cell which received sun light. In the case of 1U cube satellite which dimension is only 10cm×10cm×10cm the area of receiving the sun light is very narrow, therefore the output power is much small to use. That’s why it is very important to develop a better power sub system for cube satellite and cost efficient power unit which gives much better output and efficiency [1] [2].

The specific power system deals with all the requirements and specification of cube satellite components. As the orbit time is only half and an hour therefore it is about 15 minutes to get the direct sunlight and other time it has needed to depend on the backup power and storage system. Form the efficient point of view the power supply unit need to design such a way that when the solar panel gets direct sun light, the storage unit store that power using maximum power point tracker (MPPT)and MCU controls the equipment use of direct power rather than use of direct storage power. If these two systems simultaneously work together, the power distribution module can be able to deliver the required power to all the sub systems in the most efficient and effective way [3] [4].

2. CONVENTIONAL POWER SUB SYSTEM AND PROBLEM STATEMENT
The conceptual design of the power system for mini-satellites is essentially standard. While different satellites require mission specific components, due to the simple standardized philosophy the power system architecture is generalized and employs the use of commercial-of-the-shelf parts (COTS) [5]. From a simplified operational approach, energy is developed through the use of photovoltaic cells placed on each of the satellite’s faces. During the satellite’s orbit in the Sun, these cells generate the power required for proper operation of the various payloads and are also responsible for energizing the rechargeable batteries onboard the satellite. In the eclipse phase of the orbit, while the satellite is in the shadow of the Earth, the charged batteries then become the sole power source of the system. The conventional design allows the load to implement the second regulating converter rather than doing it on the EPS controller card. The advantages of the later design are listed below [6].

1. The regulated battery bus voltage is usually lower voltage than the subsequent unregulated battery bus voltages. As a result, there is small amount of I2R losses in the inter-connect cabling. Alternatively smaller gauge wire can be a solution.
2. The placement of the regulator at the point of load is another advantage to the designer. It is helpful for single load optimization.
3. The load variation at the point-of-load is usually smaller than at the system level. This allows a converter to be selected specifically for that load and then optimized.
4. Smaller capacitors and inductors are required for point of load regulators for its smaller size. On the other hand multi-load single bus regulators require larger size.
5. Point of load converters has the capability of isolating specific loads. Based on requirements, isolated converter topologies can also be used. Even without full isolation, each load is less subject to interference from other loads.
6. Simple and consistent ON/OFF control can be implemented. Since only one voltage is distributed, the switch design for each bus is the same.

The utility of this distributed architecture would be significantly increased if a common battery voltage standard could be established.

The primary disadvantage is that it takes more regulators to do the same thing. If there are four loads, it would require four separate regulators located at each load rather 25 than a single regulator located at the EPS. The currently available grouped low power regulators can mitigate this disadvantage. Each load regulator can be smaller and tuned for its specific application where the larger single regulator encounters difficulties.

The power, whether from the solar cells or the battery has to be converted in to different voltage busses for the various onboard subsystems which require specific voltages in order to operate efficiently, this is achieved through use of DC-DC converters. While the fundamental principles for the power system are the same, each cube satellite component has specific criteria to be met related to the specifications of the satellite system. These are dependent on various factors such as the specific payload, the mission objective, and the intended orbit [7].

In the case of cube satellite and other pico satellites, the life time directly depends on the power system unit. Most of the cube satellite average life time is about 1 year [8]. The main cause of this small period of life time is the failure of power system unit and therefore the communication system failure. That's why communication cannot be established with their ground station. The main cause of failure of the power system is over voltage, under voltage, over current, under current and voltage current fluctuation due to temperature variation [9]. To increase the life time of cube satellite it badly need to develop and implementation a smart and efficient power system with minimum cost and maximum efficiency. In the space the power system unit needs to deal with following environmental impacts [7].

1. Temperature variation (-1480C to +2550C)
2. Radiation for gamma ray and solar trails.
3. Heat radiations form the self-structure.

3. PROPOSED ELECTRICAL POWER SYSTEM

This paper has introduced a new method of power system over typical power system unit for cube satellite. When the solar panel gets direct sun light, MCU uses its internal coding to get the maximum power point tracker (MPPT) and stores that power in battery unit and other power consumption unit remains off except the altitude control, determination unit and GPS unit [5]. The other power below from the maximum power point is converted into DC-DC converter and the voltage level has risen to bus voltage 50V. The voltage sensing unit senses the voltage of the bus and send signal to the MCU. Here introduce a new bus voltage level 50V for power system unit. It is very efficient to reduce the thermal losses of the bus unit with previously used 28V or so on.

3.1 Power Generation

The satellite receives abundant solar energy while in orbit. This environment makes the use of solar cells the most effective means of power generation. However due to the limited surface area of the satellite, it is necessary to achieve maximum power output through maximizing available surface area for placement of the cells and also choosing the most efficient cells available.

These criteria have led to the selection of Triple Junction (XTJ) solar cells. Rated at 29.9% beginning-of-life (BOL) efficiency these cells are the most efficient to date and generate approximately 135 mW/cm^2. The satellite's external faces have a total surface area of 600 cm^2. It is however necessary to secure the upper and lower lids of the satellite to the chassis with external flanges protruding on each face. Therefore the allowable area for placement of the solar cells is equivalent to 420 cm^2. Two cells each of dimension 97 mm×34 mm are chosen to occupy each face of the satellite. These two cells are then connected in series with built-in bypass diodes eliminating the potential of draining power. This arrangement ensures that a minimum 2.7 W of energy is generated when at least one face is incident to the Sun. The electrical power generated by each cell is dependent on its efficiency and surface area [10] [11].

For each solar cell the incident power is calculated as shown below.

$$Q_{in} = AC_s \qquad (1)$$

Where the solar constant, C_s = 1367 Wm^{-2}

The electrical power generated by each cell is then represented as

$$Q_{elec} = \eta(T)Q_{in} \qquad (2)$$

Where $\eta(T)$, is the efficiency of the solar cell rated at an air mass index of 0 the conditions experienced in orbit. In this case the efficiency of the solar cells is above 25%.

3.2 Power Storage

The satellite's orbit has an eclipse period where it is in the shadow of the Earth. During this period, the satellites power supply comes solely from the onboard rechargeable storage batteries. These batteries are discharged during eclipse and charged during the Sun period. It is therefore vital that an efficient, appropriately sized battery be chosen to accomplish this task. There are several types of rechargeable batteries commercially available today but vital parameters which are evaluated when selecting a battery include maximum voltage supply, capacity, and efficiency and charge/discharge cycles [12].

From the comparison data tabulated below a high capacity Lithium polymer ion battery was evaluated and chosen to fulfill the mission requirements.

Table 1. Comparison Data for Different Battery Types

Type	Cell Voltage (V)	Energy Density (Wh/kg)	Power (W/kg)	Efficiency	Energy/ Cost (Wh/$)
NiCd	1.2	40-60	150	70-90	
NiMH	1.2	30-80	250-1000	66	1.37
Li-ion	3.7	160	1800	99.9	2.8-5.0
Li-Polymer	3.7	130-200	3000	99.8	2.8-5.0

The 4000mAh Lithium polymer battery chosen is a robust lightweight battery which has the highest energy density currently available, producing 3.7 V at 4000mAh. Weighing in at 50g, it is the lightest battery with the above specifications. While the life cycle is relatively less than traditional rechargeable batteries, the approximate four month mission life of the satellite makes it a suitable choice. Although being the lightest battery available to satisfy the operational requirements of the mission, the 50g weight consumes a considerable portion of the mass budget. It is therefore essential that the final placement of the battery maintains the center of mass requirements of the satellite [13].

The thermal requirements of the battery also require careful consideration as operation below the specified temperature will result in immediate failure of the component. Heating requirements may become necessary during the eclipse period. Charging Lithium polymer cells requires special charging characteristics. Current has to be gradually reduced while keeping the cell voltage maximum to ensure that the cell has attained maximum charging capacity. This process is achieved through the use of the Polymer Lithium ion battery charger which automatically reduces current to a trickle known as trickle charging when maximum capacity has been achieved [13].

3.3 Battery Charge Regulator

There are many manufacturer design and manufacture battery charge regulator. One of them that used in the DICE reference design is produced by Clyde Space Ltd. This regulator has been independently characterized for efficiency by measurements in the laboratory. The measured efficiencies are used throughout this analysis. For distributed electrical power system design, BCR is used which is presumed to have the same quality as the Clyde Space device.

3.4 Power Distribution

A new design model of power distribution board has designed and introduced here for to maintain fixed bus voltage and power distribution. The terminal voltage of distribution system is 3.3V, 5V, 6V, 9V and 15V with specific current limit and monitoring the status of voltage and current with over voltage, under voltage, over current and under current protection directly monitored by MCU [14].

The main power consumption units of cube satellite are payload camera and communication unit. In case of ellipse, payload camera system no longer needed but the communication systems need to be alive all the times. Therefore an efficient power distribution system has introduced here with minimum power consumption and distribution [15].

Table 2. Power Consumption Unit Specification

Module	Operating Voltage (V)	Operating Current (mA)	Power Consumption (mW)
OBC	5	0.5	2.5
ACDS	1.4 ~ 1.8	216	0.1 ~ 0.12
Camera	3.3	130	200 ~ 430
COM	6.5 ~ 12.5	24	300
MCU	5	160	8

4. DESIGN SPECIFICATION

Design specification of this new model has the following equipment and parameters.

Table 3. Power Budget

Power uses unit	Power (mW)	Peak Power	Duty Cycle	Orbit Average (mW)	10% Margin	Total Power (mW)
OBC	200	220	100%	200	20	220
ACDS	160	180	100%	160	16	176
Comm Tx	9300	10000	5%	279	27.9	306.9
Comm Rx	80	150	100%	80	8	88
GPS	950	1000	100%	47.5	4.75	52.25
EPS	285	320	100%	285	28.5	313.5
Camera	200	250	20%	40	4	44
Motor Control	100	120	10%	10	1	11

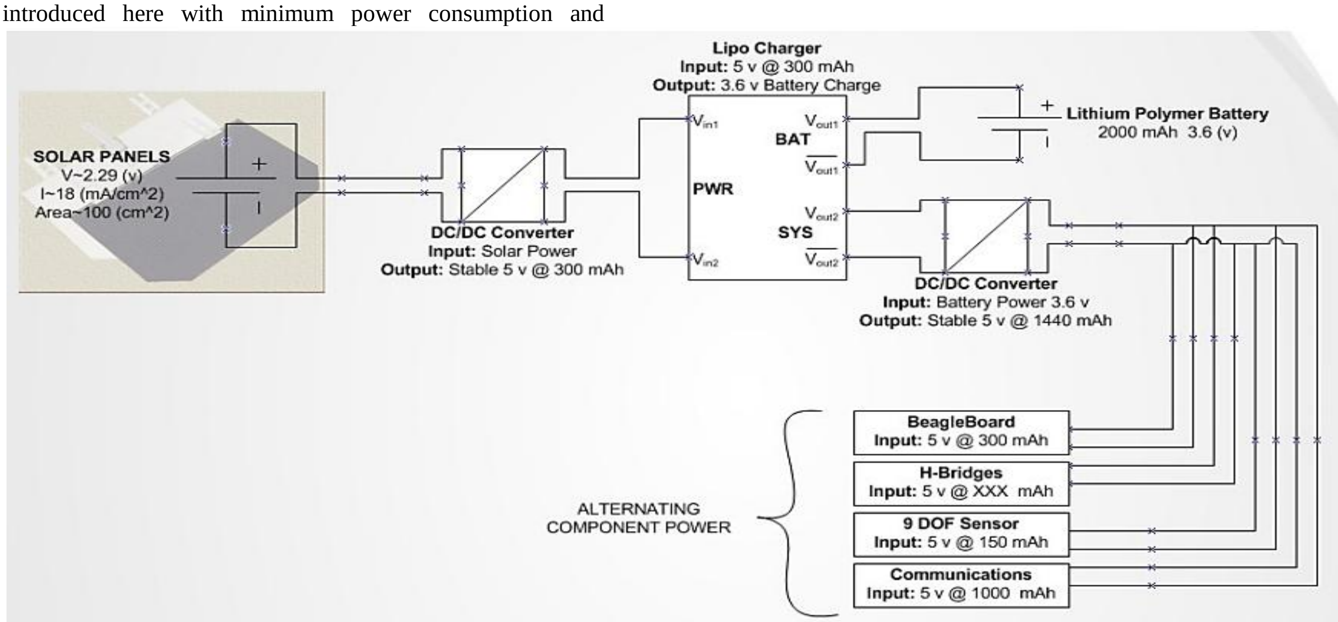

Fig 1: Power Distribution Structure

5. RESULT

Finally, for developing of small satellite power system, an estimation of load requirement of each component is necessary based on real time current and voltage data from the system. It gives an overall idea of the power budget allocation necessary and allows for the design engineer to accurately configure the power flow through the power system circuit. It is very insightful, for EPS designs, to perform full power system analysis. Looking at the power performance from the solar array down to the last converter before the load, gives you a very complete look at all of the power dissipation. It allows for identification of problem areas where further optimization can be made. Building a prototype design for each converter, with representative loads, allows you to completely characterize the performance of the selected converter. It helps identify issues early in the design process.

6. CONCLUSION

The distributed EPS design is very flexible with a high degree of utility. The efficiency of an optimized centralized design can be shown to be approximately close or equal to that of the distributed design. In the case of the reference design used in this analysis, the distributed design efficiency is better than others. The use of efficient, smaller size point-of-load converters, both inductor based converters and charge pumps facilitates single bus voltage architectures for nano class or cube class satellite applications. This architecture is popular and widely used in small satellite applications, and plays vital role for electrical power system design of cubesat or nanosat that are usually used among various platforms and missions.

Ultimately, if a distributed design is implemented, optimization can be done at a lower level. A series connected, two cell lithium-ion batteries was used in this analysis. The research would indicate that an 8.4 V (two series cells) battery bus is the most common. For example, the DICE radio initially required a higher bus voltage. They initially wanted greater than 9 V. The requirement was subsequently lowered to accommodate the DICE battery bus voltage. Using a higher bus voltage would reduce the number of boost converters required in a system. However, the higher the bus voltage, the lower the converter efficiency is when that voltage is converted to low level regulated voltages. A voltage exceeding 12.6 V is not suggested for this reason. Therefor a standard/typical voltage is selected to realize its maximum utility.

Full power system modeling is extremely insightful and useful for analyzing the power system performance. Further development of the EPS system level models to include system level dynamics would be valuable. Further work in this modeling arena could provide a very valuable tool for the EPS designer in not only evaluating the EPS architecture and optimizing the system, but it could be very useful in performing mission simulations for the power system. Bus switches could be implemented and controlled based on mission scenarios.

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