

Plug-n-Play, Reliable Power Systems for Nanosatellites

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For most companies and organizations planning their own space mission, the prospect of producing a reliable, yet affordable power system for their mission is a non-trivial problem. Some non-traditional spacecraft manufacturers, such as Universities, are finding out the importance of a well designed power system the hard-way and are experiencing on-orbit power system failures.

The most obvious solution to removing concern over the reliability performance of a power system is to buy one from a recognized source. However, for most nanosatellite producers, the cost of doing so is perceived as being prohibitive. This is no longer the case: Clyde Space Limited has developed a plug-n-play power system that is not only designed by highly experienced small satellite power specialists; it is also affordable and available off-the-shelf.

The power system uses an innovative approach to interfacing to the spacecraft solar arrays, providing a maximum power point tracking system that can interface to up to six solar arrays of differing characteristics simultaneously. In addition, a regulated 5V and 3.3V is supplied for the satellite on-board systems.

The power system is based on CubeSat requirements and uses the PC/104 format. The system is modular and scalable to accommodate creeping power budgets or larger buses. The power system has been designed to cost with the objective of supporting low-cost space access and educational missions by providing mission designers with an alternative to an in-house design and to reduce the risk to mission success.

INTRODUCTION

The power subsystem could possibly be the most under-appreciated and forgotten of all of the on-board electrical subsystems. There maybe several reasons for this, but the most likely is that most people just don't find the subject interesting enough. There are, of course, exceptions to this generalisation (me for one), but I am confident that I am correct in saying that no one is currently planning a small satellite mission to demonstrate a new power management technique.

Grabbing the small satellite headlines currently are more advanced communications systems, on-board data handling, high speed data links, imaging systems, micro-propulsion, attitude control algorithms, sensors and actuators. It is natural that the best people in a small organisation focus on the more exciting aspects of a mission; this is the differentiator of that organisation's space mission from the rest of the world. However, it is also clear that these systems need power and power that is delivered reliably and efficiently.

As all miniature spacecraft require some sort of power management system, and that this system will differ little from mission to mission, it makes sense to provide an off-the-shelf solution for common buses such as that used for the CubeSat Kit. By providing such a system, the responsibility of design of the power system within smaller organisations can be removed,

allowing the mission design team to focus on the design of the rest of the spacecraft.

Another added bonus of buying an off-the-shelf system is that the power system has been competently designed by an engineer with more than a fleeting interest in the problem of on-board power provision. In the case of the power system described in this paper, the developers have over 25 small satellite missions worth of experience in power subsystems design and are passionate about providing the optimum systems and electrical design to suit the platform.

Our objectives in the design of our CubeSat Power subsystem were as follows:

- To maximise the power available from the solar arrays.
- To provide a high efficiency interface between the solar arrays and the rest of the spacecraft to minimise losses.
- Include an integrated lithium ion battery.
- Provide common regulated voltages of 5V and 3.3V.
- Provide a standard digital interface that can deliver telemetry and telecommand functions for the power system.
- To be independent of user set-up requirements with no need for modification – straight integration with the spacecraft bus. To be truly 'plug-n-play'.

POWER SYSTEM ARCHITECTURE SELECTION

Choice of power subsystem topology is often a point of contention amongst power management engineers. The final decision for the topology used on a mission usually boils down to personal preference or the need to use an existing system rather than develop a more optimum solution.

As someone who has encountered many methods of implementing power management systems over my 12 years of working on small satellite missions, I have generated an opinion on most types of commonly used architectures.

For CubeSats and miniature spacecraft in general, the challenge is to find an architecture that is efficient and also has a low mass and volume.

The Direct Energy Transfer Battery Bus Topology

I have seen many spacecraft use a simple direct energy transfer system where the solar arrays are connected directly to a battery bus. Out of all of the architectures I have come across, I have never understood why spacecraft designers continue to use this method.

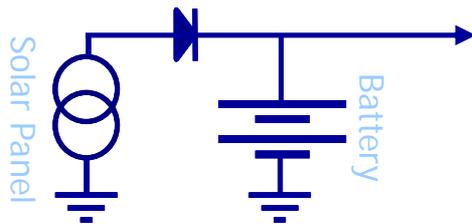


Figure 1 Direct Energy Transfer with Battery Bus

There is one main reason for my confusion and that is because the solar array performance is only at its maximum when the panels are at their maximum temperature and the battery is fully charged – these are usually the conditions when you don't actually need the power anymore. The reason that this is the case is that the characteristics of an array change significantly with the change in temperature seen by body mounted arrays in Low Earth Orbit (LEO).

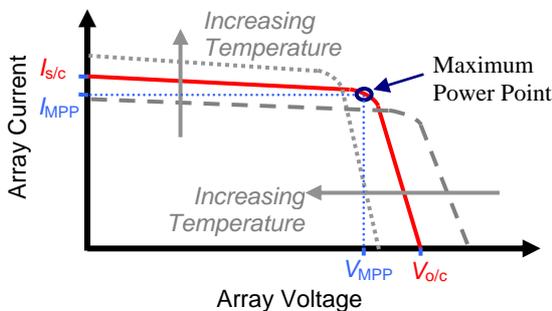


Figure 2 Solar Array Characteristic

As can be seen in Figure 2, the Maximum Power Point Voltage (V_{MPP}) will increase as the solar panel cools and decrease as it heats up. Similarly, the Maximum Power Point Current (I_{MPP}) decreases as the panel cools and increases as it heats up. The combination of these effects means that a fraction of the available power from the arrays is utilised during the sunlit period. As the spacecraft leaves eclipse with cold panels and a discharged battery, the array is clamped to the voltage of the battery.

For example, take a solar array of 7 multi-junction cells and at -40°C feeding into a 3 cell lithium ion battery at 25% depth of discharge (DoD). The battery and the array have a voltage of approximately 11.2V and the array current is 0.15A, this gives an array power of about 1.5W. At this temperature and operating at the array maximum power point, the panel would be delivering 2.6W. It is only when the battery is fully charged at 12.6V and the array is hot (i.e. towards the end of the sunlit period) that the panel power whilst clamped to the battery voltage (1.8W) and that of the maximum power point (2W) are close. However, now that the battery is charged, it is likely that power from this new found efficiency will need to be shunted in order to prevent over-charging the battery.

The battery bus direct energy transfer topology offers gains in mass and volume (when ignoring the need for a shunt to clamp the bus voltage and larger solar arrays to meet the power requirement), but falsely offers efficiency gains with only a diode drop of loss. It is clear from this basic analysis that this power system is unsuitable for most, if not all, mission scenarios.

The MPPT Battery Bus Topology

There are other topology options that can and have been used on small satellites, such as a regulated bus [4], but for miniature spacecraft these are not practical. The ideal topology for CubeSats and other miniature spacecraft is the Maximum Power Point Tracker (MPPT) with battery bus system. The most versatile implementation of this system is to use a dedicated MPPT for each solar panel.

This configuration has many advantages:

- It allows the use of different solar cell technologies and string lengths on each panel.
- The Maximum Power Point (MPP) of an individual panel can be tracked over the changing thermal conditions whilst in sunlight. The panels are likely to be at different temperatures and hence have different characteristics, so this is important.
- It provides a graceful degradation in the system design with the loss of a panel or an MPPT
- The battery typically needs to be charged for the majority of the sunlight period, so additional loss through having a switch-mode power supply (SMPS) in series with the array has little impact on the overall sunlight efficiency of the power system.

- (e) The direct connection between the battery and the bus provides maximum efficiency during eclipse.

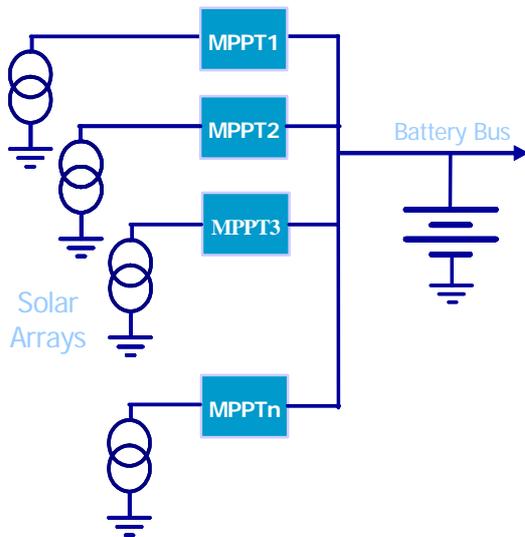


Figure 3 MPPT Battery Bus

Using this configuration it allows the power system to maximise the power from the arrays at the beginning of the sunlit period, replenishing the charge of the battery in a quicker more controlled manner. In addition, once the battery is fully charged and there is low power demand on the bus, the unwanted power is left in the solar array by moving off the maximum power point, so no need for hot and heavy analogue shunt electronics.

The MPPT is typically a SMPS and therefore can regulate its output to the end of charge voltage of the battery once this level is reached, removing the need for a complicated charge control system

For these reasons, the MPPT bus was selected for our off-the-shelf miniature spacecraft power system. This technique has also been proven on many missions designed by the author, including Surrey Satellite Technology Limited's (SSTL) highly successful SNAP-1 nanosatellite mission [5].

SOLAR PANELS FOR MINIATURE SPACECRAFT

Of the available solar cell technologies, the GaInP2/GaAs/Ge multi-junction cell is the only real alternative for miniature spacecraft. Other than a significantly higher efficiency than other technologies, the most advantageous characteristic of this technology is that the terminal voltage of the cell is over 2V (at least double that of other cell technologies). Given the relatively small panel area available on a miniature satellite, the higher terminal voltage allows the mission designer to achieve a more useable array voltage. Single junction GaAs cells have a terminal voltage of

0.89V and Silicon 0.5V and, neglecting issues relating to their inferior efficiencies, will require the use of an unreasonable number of cells in series to reach a useable voltage of above, say, 10V.

For 1U CubeSats, a single 100mm x 83mm panel can easily accommodate six to eight 2cm x 4cm solar cells. This equates to a power of greater than 2W and a terminal voltage of above 12V. Most of the major solar cell providers in the USA, Europe and Japan can provide multi-junction cells either in this or a similar size.

For 3U CubeSats, it is possible to use the more readily available 4cm x 6cm cells. With two cells per panel and 3 panels per spacecraft x/y facet, a 6 cell string can easily be achieved and hence a useable voltage.

N.B. It is not viable to connect cells in series that cannot be guaranteed to be in sunlight at the same time (i.e. on different spacecraft facets) as this has the same effect as shadowing cells in a string and the string will not deliver any power to the spacecraft.

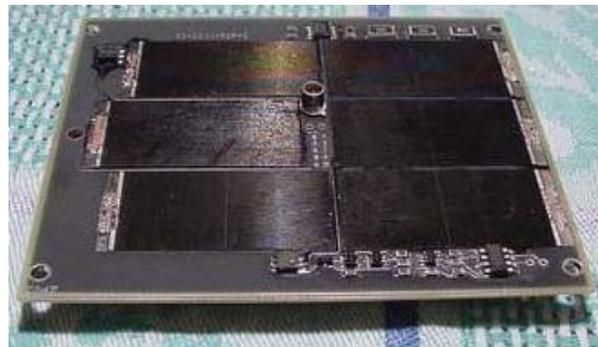


Figure 4 STENSAT CubeSat Solar Array [1]

OFF-THE-SHELF POWER SYSTEM DESCRIPTION

With the power system architecture selected and the potential solar array designs evaluated, it is now possible to describe in more detail the specifications of the CubeSat power system.

In order to ensure that the specifications of our design matched the requirements of the CubeSat community in particular, we liaised closely with key figures already producing CubeSat systems in the development of our specification.

CubeSat Power Battery

The battery is integrated with the power system electronics. The battery voltage is sized such that the voltage is compatible with both the minimum solar array voltage and the minimum input voltage to the 5V regulator. In order to ensure compatibility, a two series cell lithium ion battery has been selected. As with most small satellite missions, we propose to use a screened commercial cell for the battery. The use of commercial cells in space has been successfully practised for many years by companies such as SSTL [3] and ABSL [2].

For battery capacity requirements, this can tend to be mission specific, but there is certain general platform information that can allow a typical battery capacity requirement to be determined. The typical Orbit Average Power (OAP) for a 1U CubeSat in a sun synchronous orbit with maximum eclipse should be no more than 2W (the panels are 100mm x 83mm, so even with high efficiency cells the max power per panel is about 2.6W).

Therefore, the max battery capacity would be for 2W for 35 mins = 1.1Whrs. With a DoD of 20%, this equates to a battery capacity of 6Whrs. With Lithium Ion cells it is important to also consider that the capacity fades over the battery life. 40% fade is a reasonable assumption for a 3 year mission in this orbit, therefore a battery capacity of 10Whrs beginning of life (BOL) is preferable.

Our integrated two cell Lithium Ion or lithium polymer battery gives us a minimum BOL capacity of $2 \times 3.6V \times 1.5Ah = 10.8Whrs$. For higher power (i.e. deployed arrays) and larger CubeSats (3U), it is expected that 2, 3 or 4 CubeSat power systems will be used in parallel to boost the capacity to the required level.

The battery will have an integrated heater, controlled autonomously, that will ensure that the cells stay above 0°C.



Figure 5 *Left*, battery system for nanosatellites from ABSL [2]; *right*, Danionics lithium polymer battery.

Maximum Power Point Tracker

With so little power available on a CubeSat, it is essential that the interface between the solar arrays and the rest of the spacecraft is optimised for both energy transfer efficiency and systems design.

As previously described, an MPPT system ensures that the voltage of the array remains at its optimum value when the power requirement demands, but it is also essential that the MPPT itself is highly efficient to take advantage of this.

The CubeSat Power MPPT uses a BUCK dc-dc converter topology (also known as a STEP DOWN Converter). Clever design of this circuit enables us to achieve conversion efficiencies of close to 90% even at such power levels of 2W per MPPT.

The MPPTs do not rely on battery power for their operation, instead they are powered via their dedicated solar array, allowing charge to reach the battery no matter what the battery state of charge.

The MPPT is controlled by the on-board microcontroller with the control algorithm operating in the firmware. In the case of a fully discharged battery, the MPPT will start up with a fixed usable input voltage, facilitating battery charge at a recovery level until the battery voltage has recovered sufficiently to enable the regulated 5V bus and hence the microcontroller to come back ON. (This functionality is described in more detail in the next section).

With 6 MPPT interfaces on a single board, there are sufficient inputs to ensure compatibility with a CubeSat with 3W solar arrays all six facets. This was considered the worst case. For missions that have more or higher power solar arrays, the system has been designed such that additional power units to be combined in parallel.

There is no need for the user to specify the characteristics of the solar arrays prior to integration. The design is plug-n-play and the microcontroller will track the maximum power point of the array as long as it is above 9V and below 25V.

Power Conditioning Module

The Power Conditioning Module (PCM) consists of two dc-dc converters; one regulating its output to 5V and the other 3.3V. Each converter can provide up to 1A on its output and, as with the MPPTs, can be paralleled with additional units on a sister power board if a higher current is required.

The PCM provides additional protection features that are essential for protection of the spacecraft from anomalous operational modes.

The first feature is simply to limit the output current of the PCM to a maximum level, and hence limiting the bus current, protecting the power system from faults elsewhere in the spacecraft.

The second feature is a hardware unloading function to back-up the battery under-voltage safety tasks that typically run on the on-board computer. The unloading function will disable the output of the 5V and 3.3V converters once the battery voltage reaches its minimum acceptable level (i.e. close to zero capacity). This not only stops power consumption on the 5V and 3.3V buses, but also turns OFF the power switch that supplies the battery voltage bus to the rest of the spacecraft. Built in hysteresis means that all three buses return to operation once the battery voltage has recovered to a reasonable level. This functionality prevents

the battery suffering permanent damage due to over-discharge.

Telemetry and Telecommands

As previously mentioned, the power unit's dedicated microcontroller provides a platform on which to control the maximum power point trackers. The microcontroller also provides a serial bus interface using the I2C standard through which system telemetry data can be monitored.

Due to the high number of telemetry channels on the power system, there is a need to interface the signals to the microcontroller via analogue multiplexers. The telemetry channels on the power system include:

- Solar Array Voltages (one channel per solar array)
- MPPT Currents (one channel per solar array channel)
- Solar Array Temperature (one channel per solar array)
- Battery Voltage
- Battery Current
- Battery Temperature
- PCM Input current
- 5V bus, 3.3V bus and Battery bus currents.
- Battery Bus switch status.
- Battery heater status.

There are a total of 27 telemetry channels on the power system.

The telecommands on the power system are as follows:

- Battery bus over-current protection switch command ON and OFF.

INTERFACES

The main power system interfaces are designed to be compatible with the standard CubeSat Kit bus via the stack Samtec ESQ connector. The main power system specific connections through the bus connector are as follows:

- Separation/activation switch. Using the NC and C pins of the plunger switch on the launcher interface facet, the spacecraft is held OFF during launch, coming ON only once the spacecraft have separated from the launch vehicle.
- The RBF switch is used to disconnect the battery negative from ground, hence isolating the battery. This is particularly important when putting the spacecraft into storage as it enables the battery to be stored in an unconnected, discharged state.
- Battery bus, 5V, 3.3V and ground connections are provided over the main bus.
- With the spacecraft connected only by the USB port, the power system provides the capability to charge the battery using USB power. The maximum charge current over this connection is 0.2 A per battery.
- Another feature whilst the spacecraft is connected by USB is the ability to receive power system telemetry from the microcontroller.

Other connections and properties are as follows:

- The solar arrays connect directly to the power board via dedicated connectors located on the board perimeter.
- All major interfaces have ESD protection to prevent accidental damage during handling or whilst in the stack (i.e. on the solar array input lines).
- The spacecraft ground is connected to the spacecraft structure on the power board to ensure that no currents flow through the structure and for EMC issue prevention. This connection would be made only on one power board where multiple boards are used.

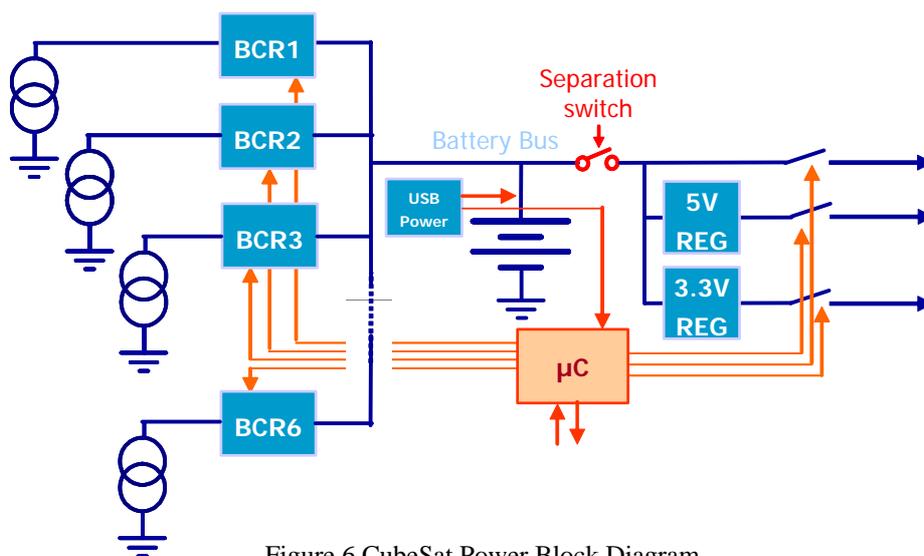


Figure 6 CubeSat Power Block Diagram

POWER SYSTEM PACKAGING

The complete block diagram of the power system is shown in the Figure 6.

This entire system and battery is accommodated within the footprint of the standard PC/104 card. The height of the unit remains within the 15mm standard for CubeSat Kit hardware, ensuring that a minimum volume is occupied by the power system within the spacecraft stack.

This configuration not only saves volume by having the battery and electronics on a single board, it also benefits the modular nature of the power system design by allowing mission designers to use 2 or more power system boards with little impact on the overall spacecraft volume.

The battery is located in the centre of the PCB to help with centre of gravity issues on the spacecraft.

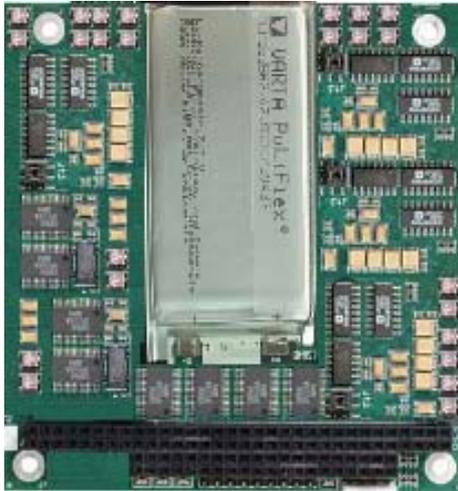


Figure 7 CubeSat Power example configuration.

With each battery cell weighing around 41g, the total battery mass is expected to be less than 100g (including fixings and epoxy). With the rest of the power system electronics expected to also weigh less than 100g, the total mass of the CubeSat power system (excluding arrays) is less than 200g. **Error! Reference source not found.** shows the planned configuration of the CubeSat power system, illustrating the integrated nature of the battery within the assembly.

Having worked through the requirements and the preferred configuration of the power system to arrive at our design specification, it was clear that PCB area was going to be precious. However, including so much functionality on a single CubeSat slice makes this power system a very attractive power management option for CubeSats in general, and particularly those that are pushing the boundaries of CubeSat utility.

CONCLUSION

The CubeSat Power design project was an extremely interesting exercise, especially given the team's already extensive experience in the design of small satellite power subsystems. The main challenges were to design a system that could meet the demands of a wide variety of mission profiles, to fit the electronics and battery on to a single PC/104 card without the need to increase the unit height and to design the system to cost and for on-orbit reliability. We are satisfied that we have met our objectives.

These power systems will be mass produced using automated assembly techniques in order to drive down production costs and increase the unit's affordability. Our first flight units will be available to buy towards the end of 2006 once the design has been verified.

The Clyde Space team are enjoying the challenge of producing cutting edge, innovative hardware for next generation CubeSats and other miniature spacecraft and are proud to be a contributor to be CubeSat phenomenon.

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