POWER SYSTEM CHALLENGES FOR SMALL SATELLITE MISSIONS

Craig S. Clark, Alejandro Lopez Mazarias

Clyde Space Ltd.,
6.01 Kelvin Campus, West of Scotland Science Park, Glasgow G20 0SP Scotland,
Email: craig.clark@clyde-space.com

ABSTRACT

For the last few decades, the focus of power subsystem development and advancement has revolved around the needs and requirements of large, high power missions, mainly for telecommunications applications. Whereas the challenges to increase efficiency and to reduce mass and volume in these applications are valid, the needs of the small satellite mission have often been over shadowed. The growing number and utility value of small satellites highlights the need for careful and measured consideration for the specific power requirements on small satellite missions.

The challenge for mission designers and, in this context, small satellite power system specialists, is to produce an efficient and flexible design that avoids the need for system redesign with each set of mission requirements. This paper sets out the various challenges facing small satellite mission designers and evaluates common power system architectures against an innovative approach to power management for small satellites.

1. INTRODUCTION

From a power systems perspective, it is relatively safe to say that power challenges and requirements are common across the full spectrum of small satellite missions. Small satellites can have a range of power requirements from as little as a watt or two, to a few kilo watts. Despite this extensive range of power requirements, a typical mission will be launched into a low Earth orbit and the majority will have fixed solar arrays that encounter varying solar illumination characteristics across a typical orbit. Most will also experience frequent eclipse periods.

For spacecraft within this category, there is a definitive need to have a flexible solar array interface that can adapt to the changing solar array characteristics whilst at the same time providing the ability to perform a fast recharge of the spacecraft secondary battery system.

With the growing utility value of small satellites, primes are looking to produce multiple missions for either constellations of small satellites or through the use of a reusable small satellite platform. There is a need for the power system electronics to be compatible with a multitude of mission profiles and be scalable in power handling capability. In addition, mission designers are consistently vying for a reduction in mass and more flexibility and modularity in the spacecraft make-up. Some spacecraft manufacturers are even investigating plug and play small satellite kits for fast response missions.

At Clyde Space, we are developing power systems capable of meeting these demanding requirements but with a focus on cost effectiveness as well as performance. This paper outlines the systems under development at Clyde Space and compares their performance against the current systems used on today's small satellites.

Clyde Space are based in Glasgow, Scotland and offer cost-effective, off-the-shelf and tailored small satellite power management systems, solar arrays and batteries. Our team has extensive industry experience having worked on over 25 small satellites. Clyde Space is Glasgow's first indigenous space company and is proud to carry on the Scottish tradition of innovative, reliable engineering.

2. COMMON POWER SYSTEM APPROACHES

In this paper we will evaluate the three most common power system implementation approaches found on today's small satellites. These power systems are as follows:

- Direct Energy Transfer (DET) with Battery Bus.
- DET with Regulated Bus.
- Maximum Power Point Tracker with Battery Bus.

The following sections describe the operation of each of the above systems.

2.1. Direct Energy Transfer (DET) with Battery Bus

One of the most simple power system configurations, this topology is often selected for its mass advantages. The system is low in mass because it has only a solar array regulator interface with no switch mode power supply element.

Whilst this apparently seems a desirable system to use on a small satellite, it is a false economy. The coupling of the battery and array voltages directly, results in the need for much larger solar arrays, resulting in a more expensive and heavier spacecraft.

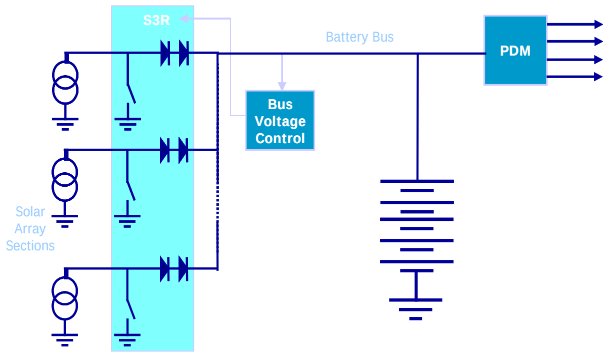

Figure 1 Direct Energy Transfer with Battery Bus

The main functional drawback of this system is that 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).

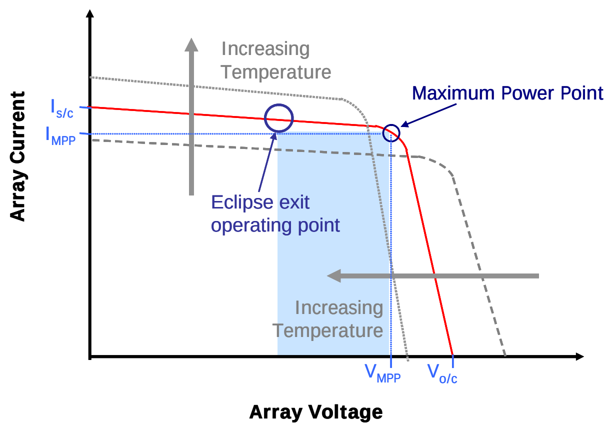

Figure 2 Solar Array Characteristic

As can be seen in *Figure 2*, the Maximum Power Point Voltage (VMPP) will increase as the solar panel cools and increase as it heats up. Similarly, the Maximum Power Point Current (IMPP) 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 arrays and a discharged battery, the array is clamped to the voltage of the battery. The blue area in *Figure 2* shows the unused array power when the battery voltage is low.

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.

2.2. Direct Energy Transfer with Regulated Bus

The most commonly found power system topology found on European spacecraft is the regulated bus. This is due to the influence of the European Space Agency (ESA) who have favoured this topology in different forms for many years. The power system in Figure 3 is a typical regulated bus topology and incorporates a sequential switching shunt regulator (S3R) on the array interface. This topology is most suited to applications where the spacecraft experiences extended periods of sunlight plus the occasional long eclipse period.

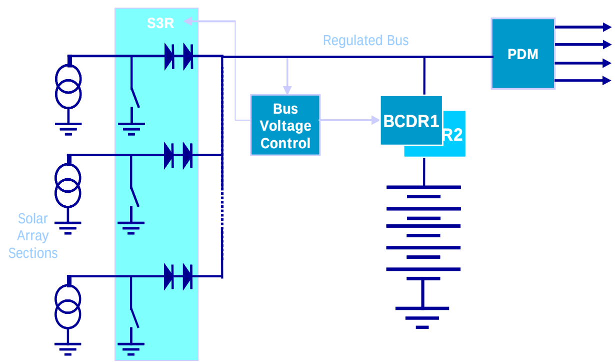

Figure 3 DET with Regulated Bus

Power is supplied directly onto the bus from the solar arrays via blocking diodes. During sunlight, the bus voltage is regulated, in most cases to 28V or 50V, by the S3R. For periods of high power demand, battery power is supplied to the bus at 28V via the Battery Discharge Regulator (BDR).

Although the power system is, as claimed by Olsson [3], more efficient than most small satellite power systems in the transfer of energy from the solar arrays to the bus, it is only when the arrays are at their maximum temperature, and at end of life, that the solar arrays operate at the maximum power point. For the vast majority of the spacecraft life, the full potential of the arrays will never be exercised. When used in LEO, this power system will require the use of solar arrays much larger than that required by an MPPT bus, but for GTO or GEO, the solar arrays can be sized for the equilibrium array temperature so that a smaller proportion of the potential array power is left in the panels.

The power system also suffers from the need to discharge the battery through a regulator when in eclipse, ensuring that the bus voltage remains regulated to the same fixed voltage. This introduces yet further inefficiencies into the system, especially significant for orbits where frequent or prolonged eclipse periods are experienced.

2.3. Maximum Power Point Tracker with Battery Bus

This power system architecture incorporates a Maximum Power Point Tracker (MPPT) between the solar arrays and the battery. As described by Denzinger [1] and Clark [2], this system works on the principle of charging the battery and supplying the bus during sunlight while setting the array voltage at the maximum power point.

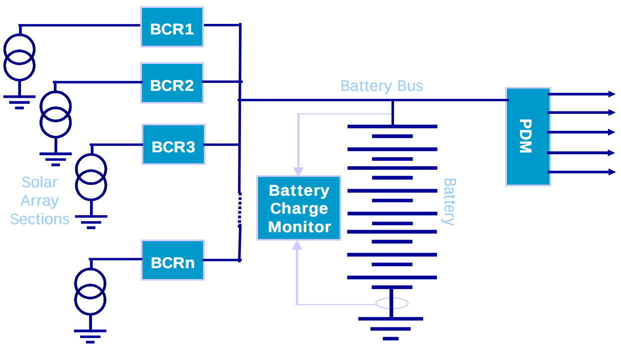

Figure 4 MPPT with Battery Bus

Once the battery has reached its end of charge state, the MPPT fixes the bus voltage at the end of charge voltage and allows the battery current to naturally taper off to a trickle charge level, at the same time backing off the power from the arrays. By backing off the arrays, there is no need for shunt regulators to dissipate unwanted solar array power. A block diagram of the topology is shown in *Figure 4*.

Although array power is maximised, there is an inefficiency associated in the MPPT system, and approximately 5-10% of the array power will be lost before it reaches the bus. This is because the MPPT system is a DC-DC converter, stepping down the array voltage to the bus voltage, and using a control loop to track the maximum power point. The efficiency of this converter is about 90-95%, and introduces the main inefficiency in the power system.

The benefits of this MPPT system are only realised in situations where the maximum power point (MPP) of the array is changing significantly whilst the spacecraft is in sunlight. This is the case in low earth orbit (LEO) where the array temperature, and hence MPP, changes considerably over the sunlight period of the orbit. In an orbit such as geostationary orbit, where there are extended periods of sunlight and solar arrays at equilibrium temperature, the inefficiency of the MPPT would make the use of this topology impractical.

During eclipse, however, energy is directly transferred from the battery to the bus. This is especially relevant for spacecraft in orbits that experience frequent or long duration eclipse.

3. INNOVATION IN SMALL SATELLITE POWER

Clyde Space are currently developing two systems for small satellites that address the issues of cost, performance and flexibility of design for future small satellite missions. The first is a Modular MPPT with integrated battery for a battery bus. The second is a modular MPPT bus with an integrated battery. Operationally, the difference between the two systems is that one has a battery bus and the other is a direct energy transfer system using a MPP bus.

The two systems are described in this section. For the Modular MPPT with integrated battery for a battery bus we will use the Clyde Space CubeSat power system as an example.

3.1. Modular MPPT with Battery Bus and integrated battery

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:

(a) It allows the use of different solar cell technologies and string lengths on each panel.

(b) 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.

(c) It provides a graceful degradation in the system design with the loss of a panel or an MPPT

(d) 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.

Of the available solar cell technologies, the GaInP2/GaAs/Ge multi-junction cell is the only real alternative for CubeSats. 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 a large number of cells in series to reach a useable voltage.

For 1U CubeSats, a single 100mm x 83mm panel can easily accommodate two 4cm x 7cm solar cells. This equates to a power of greater than 2W and a terminal voltage of above 4V.

The CubeSat power system 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 and 3.3V regulators. In order to ensure compatibility, a two series cell lithium polymer battery was 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[2] and ABSL [4].

The integrated two cell lithium polymer battery provides a minimum BOL capacity of 2 x 3.6V x 1Ah = 7.2Whrs. For higher power (i.e. deployed arrays) and larger CubeSats (3U), 2, 3 or 4 CubeSat power systems can be used in parallel to boost the capacity to the required level, or an additional 1Ah can be added to a single module.

The CubeSat Power MPPT uses a Flyback dc-dc converter topology. Clever design of this circuit enables us to achieve conversion efficiencies of close to 90% even at such power levels of 3W per MPPT.

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 can be combined in parallel (up to 4 identical modules).

There is no need for the user to specify the

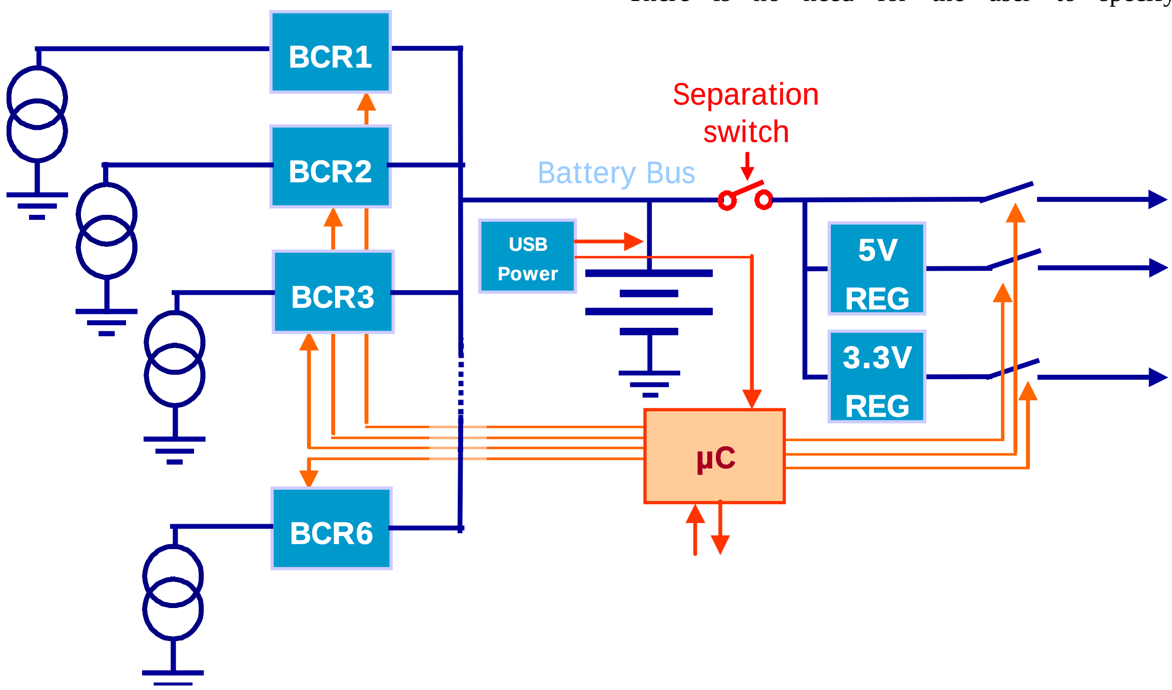

Figure 5 CubeSat Power System Block Diagram

characteristics of the solar arrays prior to integration. The design is plug-n-play and the BCR will track the MPP of the array as long as it is above 3.6V and below 25V.

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.

This power system meets the cost, performance and flexibility challenges by providing the ability to scale up the power handling capability and energy storage capacity of the system by simply adding more identical power boards.

3.2. Modular MPPT Bus with integrated battery

Figure 6 shows a block diagram of the Modular MPP Bus power system topology. The main principle behind the power system operation is the fact that the bus voltage is regulated to the maximum power point of the array during periods of high power demand (such as when charging of the battery and/or operating high power payloads such as the transmitter). When more power is available from the arrays than is required by the spacecraft, power from the array is 'backed-off' by allowing the maximum power point to be spoiled and the array voltage drift towards the open circuit voltage. In order to maintain a usable voltage level, the bus must stay within certain limits; these limits are generally 22V to 40V or similar to that of a 28V Unregulated battery bus.

Solar Array Requirements
The first and possibly most important aspect of the design of this power system is the specification of the solar arrays. Due to the solar array bus being controlled to either equal the maximum power point of the array or the backed-off voltage of the array, care must be taken in the solar array design to ensure that the bus voltage stays within the set limits.

Gallium Arsenide (GaAs) based solar cell technologies are best suited to use with this power system topology

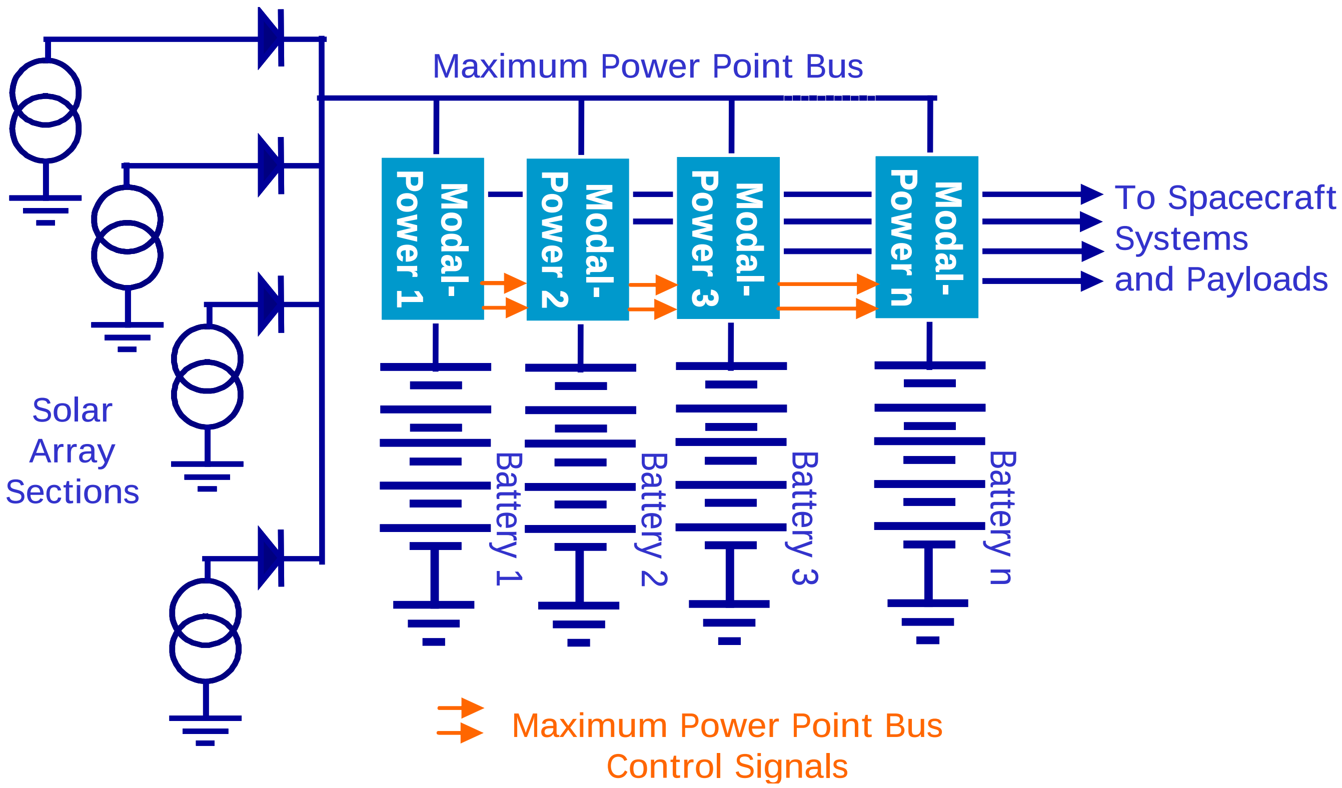

Figure 6 Modular MPP Bus Power System

due to their low maximum power point voltage (VMPP) temperature coefficient, higher cell voltage and superior radiation tolerance (compared to Silicon). The nominal cell load voltage for single junction GaAs is about 0.89V, with a temperature coefficient for VMPP of -2mV/°C.

Multi-junction cells have a higher cell VMPP of about 2V, but also have a higher VMPP temperature coefficient in the region of -6mV/°C. Over the expected temperature range (–40°C to +70°C), and for the solar array design selected for this mission, the maximum power point of the array will vary between 36.4V to 24V. This remains compatible with commercial dc-dc converters such as those produced by Interpoint [5].

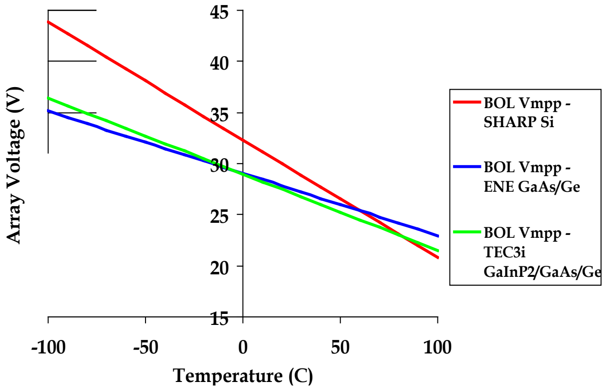

Figure 7 Solar Cell Technology Voltage Variation with Temperature

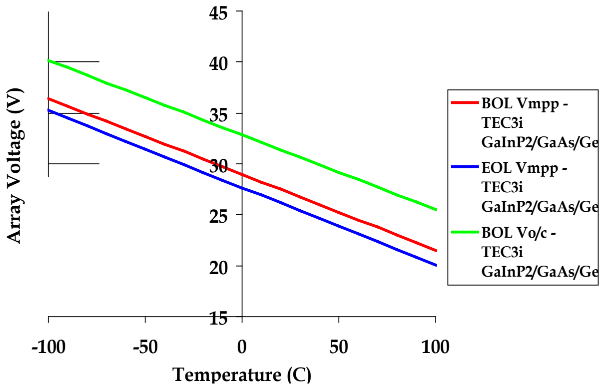

Figure 8 Multi-Junction Solar Cell Technology Voltage Variation with Temperature

Battery Technology

Another important factor in the design of this power system is the selection of the battery technology. All ESA missions now use Lithium Ion technology, abandoning previously favoured technologies such as Nickel Cadmium.

One of the features of Lithium Ion is its limited charge and discharge current rates. For terrestrial use, these rates are limited to 2C (where C is the cell capacity) – this rate is often derated for space use. It is therefore possible to design a set of charge/discharge electronics around this maximum current. This approach has the benefit of being extremely modular, but also of being inherently failure tolerant in the form of graceful degradation; the redundancy philosophy for each unit is based on the loss of a single component, whether it be a battery cell or an element of the converter.

In this case, the charge/discharge controller (or Modal Power Unit as described in the block diagram), is limited to a charge and discharge rate of about 2A (for use with a 1.5Ah cell string)

Control signals are connected between each power unit to maintain the bus voltage at the desired level. The operation and thermal control of each module is independent of the other units. Different battery types and strings at different temperatures can all be connected on to the same bus, meaning that the power system can actually be distributed around the spacecraft instead of in a single location.

Most importantly, the system is very mass and power efficient. The extent of the system performance will be examined in the next section.

4. Performance Comparison

For the purposes of comparing the system performance of the power systems described earlier in the paper, an typical LEO small satellite mission was selected; 650km altitude, fixed solar arrays, 3-axis stabilised, nadir pointing, sun-synchronous, polar orbit of LTAN at approximately 12pm.

Four power systems are evaluated for performance under these conditions:

- Direct Energy Transfer (DET) with Battery Bus.
- DET with Regulated Bus.
- Maximum Power Point Tracker with Battery Bus.
- DET Maximum Power Point Tracker Bus.

It is clear that these conditions favour a MPPT based power system, as the array characteristics are changing over most of the orbit. Figure 9 shows the results of the simulation. The data has been normalised based on the power requirement of the MPPT bus system.

The blue (back) bars indicate the amount of power captured from the arrays by the solar array interface. A 5% error was introduced to the MPPT bus as two or three arrays of differing characteristics could be in sunlight at the same time, each having differing maximum power point voltages. It is clear that the battery bus DET and regulated bus topologies do not perform well in these conditions. The next bars (pink) show the available power after losses in the array interface electronics – this is where the MPPT with battery bus losses out.

The most important data is the power margin. For the same size solar arrays, the DET Battery Bus falls significantly short of the power requirement. In fact, under these typical conditions, this topology provides 44% less power from the arrays that MPPT bus. This means that the solar array on the spacecraft will need to be approximately 50% larger for a DET Battery Bus system than for a system that incorporates a MPPT. This represents a significant cost and mass impact on the mission.

The Regulated Bus architecture proves to be unsuitable for this mission type. It is unable to adapt to perform efficiently within the mission parameters. This system is better used on spacecraft with sun tracking solar arrays where the equilibrium temperature of the array is quickly reached and maintained throughout the sunlight period of the orbit.

Figure 10 shows a mass comparison of the four systems evaluated. Again the data has been normalised. Due to the need for dc-dc conversion in the solar array regulator with the MPPT Battery Bus system, this topology is the heaviest out of the four. However, the gains in battery and solar array mass reduction more than compensate for this.

As suspected, due to the absence of any significant control electronics, the DET Battery Bus is the most mass efficient. However, this does not account for the addition mass required in the solar array which, for most solar array types, results in a significant increase in the overall mass and complexity (through the need to accommodate larger arrays on the structure).

The MPPT Bus system out performs both the Regulated Bus and MPPT Battery Bus in mass efficiency. The Regulated Bus still requires a shunt system to maintain regulation in sunlight, and therefore incurs a mass penalty in comparison to the MPP Bus, which instead leaves excess power in the arrays.

5. CONCLUSIONS

The demand for more power with lower mass and volume is increasing with the utility value of small satellites. Small Satellites are now capable of high performance missions and mission designers are understandably pushing the envelope of capability; greater payload duty ratios, faster downlink rates and highly agile platforms.

In addition, there is increasing pressure on cost and time to orbit, requiring responsive, flexible designs that are capable of adapting to a multitude of mission scenarios without the need for redesign.

We have shown in this paper that there are alternative approaches to power system configuration that can offer both increased electrical and mass performance, whilst at the same time providing the flexibility required to efficiently and effectively meet a myriad of small

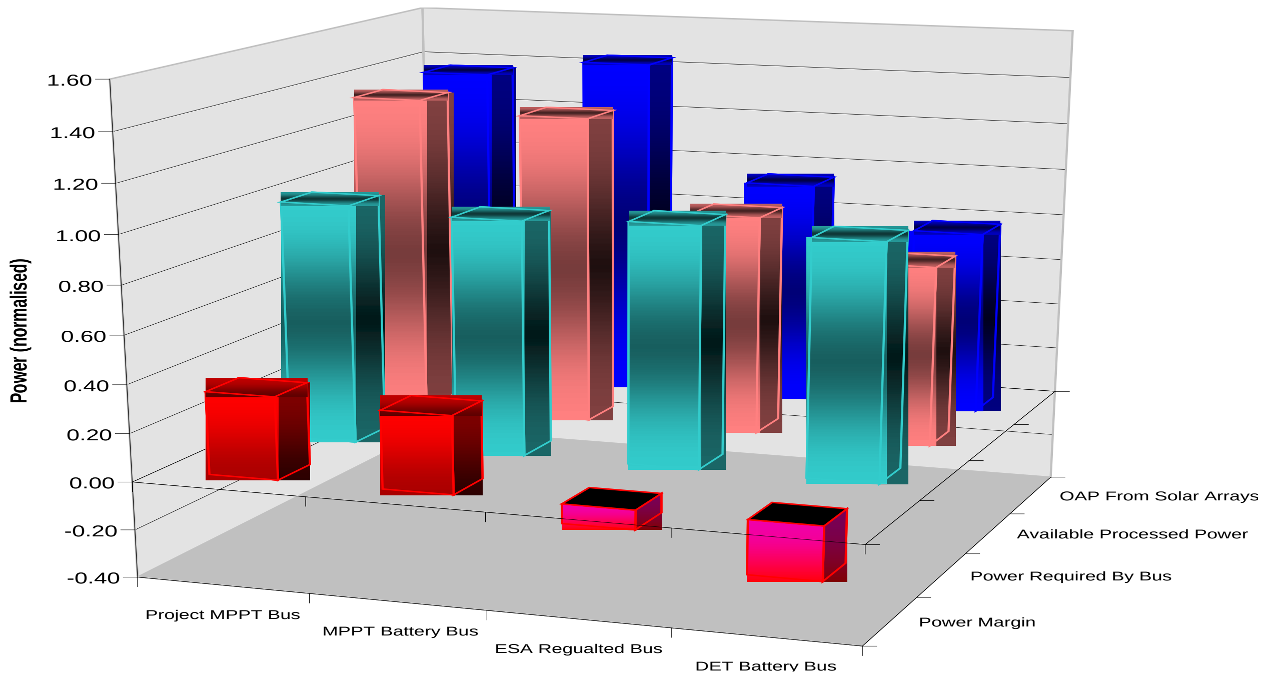

Figure 9 Performance Comparison Results

satellite mission scenarios.

Clyde Space is pushing forward with innovation in this field and will be flying CubeSat power systems on multiple CubeSat missions within the next 12 months, whilst at the same time developing the MPPT Bus system to offer as an alternative for future small satellites. We are targeting a flight validation of this concept sometime in the 2008-2009 timeframe.

6. REFERENCES

[1] *Electrical Power Subsystem of Globalstar*, W. Denzinger, Daimler-Benz Aerospace – Dornier GmbH, Fourth European Space Power Conference, Poitiers, France 4-8 September 1995.

[2] *Commercial Nickel Cadmium Batteries for Space Use: A Proven Alternative for LEO Satellite Power Storage*, Craig S. Clark, Alan D. Hill and Martin Day, Surrey Satellite Technology Ltd., European Space Power Conference, Tarragona, Spain 21-25 August 1998.

[3] A Power System Design for a Microsatellite, Dan Olsson, ESA/ESTeC, European Space Power Conference, Graz, Austria 23-27 August 1993.

[4] *'Small Cell Lithium-Ion Batteries: The Responsive Solution for Space Energy Storage'*, Chris Pearson, Carl Thwaite and Nick Russel, ABSL, 3rd Responsive Space Conference April 25–28, 2005 Los Angeles, CA

[5] Interpoint DC-DC Converter Data Pack, Interpoint Corporation, 1995.

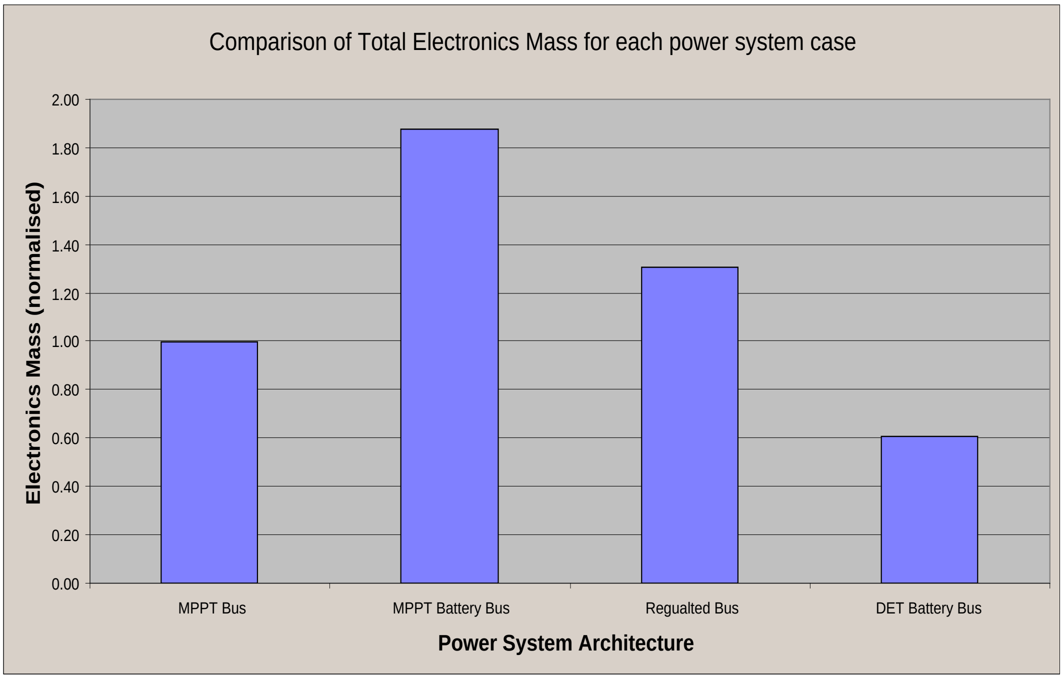

Figure 10 Mass Performance Comparison