Micro-Satellite Electrical Power System Design

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Micro-Satellites with a total system mass below 50 kg must prioritise efficiency and reliability within all subsystems due to the minimal scope for inbuilt redundancies and non-essential equipment. By designing a Micro-Satellite Electrical Power System (EPS) that is highly efficient at generating, regulating, storing and distributing the power required by a satellite, the chance of successfully completing a complex space mission is improved. This paper discusses the preliminary design stages of an EPS tailored to meet the demands of the Micro-gravity Experiment Recovery Satellite (MERS) re-entry concept mission. This paper also discusses an EPS design solution that utilises commercial off the shelf (COTS) products tailored to the power profile of the craft. Utilising proven COTS products decreases the chance of failure in what is an already complex mission. A range of candidate orbital and craft configurations are discussed in order to increase the flexibility of the mission, outlining the effects these choices have on maximum peak power, the beta angle of the orbit and the overall ability of the EPS to handle the crafts power needs. The report culminates with an EPS preliminary design architecture that is capable of operating in a wide variety of orbits, primarily the 47 and 98 degree inclination orbit with a minimum solar panel surface area of 0.24m^2.

Contents

Nomenclature

AU = Astronomical Unit
BOL = Beginning of Life
BCR = Battery Charge Regulator
COTS = Commercial of the shelf
EPS = Electrical Power System
h = Orbital height (Km’s)
ISS = International Space station
LEO = Low Earth Orbit
MERS = Micro gravity Experiment Recovery Satellite
PPT = Peak Power Tracker
Re = Radius of the earth (6378.1 Km’s)
Rs = Radius of the satellite (Km’s)

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S	=	Solar constant (Average 1370 W/m^2)
SOC	=	State of Charge
TCS	=	Thermal Control System
To	=	Orbital period
TT&C	=	Telemetry Tracking and Command
η_{pv}	=	Photovoltaic cell efficiency
η_T	=	Solar cell temperature efficiency
η_{pf}	=	Solar array packing factor
η_y	=	Total cell efficiency
ω_s	=	Satellite angular velocity (km/s)
μ_e	=	Earth's gravitational constant
$\hat{O}$	=	Orbital normal vector
$\hat{S}$	=	Sun vector
β	=	Beta angle

I. Introduction

The MERS is a concept space mission aimed at developing a craft with a total system mass of below 50 kg that is able to facilitate commercially viable access to a prolonged micro -gravity environment of up to 2 months for a wide range of experimental payloads. On completion of the micro-gravity experiments, the craft begins re-entry, passively degrading its orbit into a low trajectory decent allowing the craft to safely land back on earth, delivering the experimental payloads back to the consumer, intact. This need has arisen due to the fact that the main platform for conducting micro-gravity experiments the International Space Station, is forecast to decommissioned in 2020 (Chaplain, 2012).

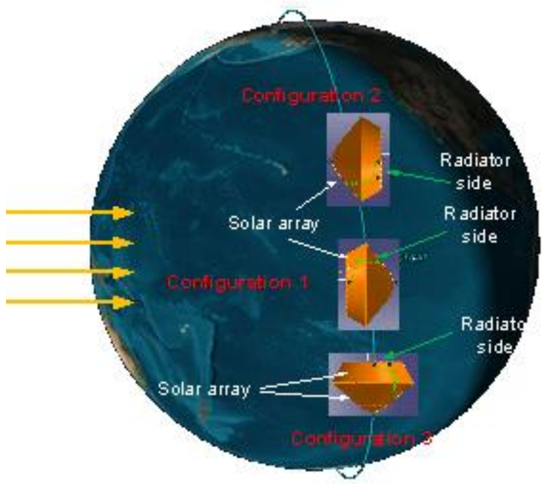

Figure 1. MERS Orbital configurations. Configuration 2 is the standard for the experimental operation stage of the

Three orbital configurations have been investigated in the design of the MERS. Configuration 1 positions the rear of the craft to be facing towards the sun during experiment mode, allowing the largest flat surface of the craft to be exposed to the sun. Configuration 2 is the current standard for all design and requires the front of the craft to be orientated to face the sun during the experiment operation stages of the mission, as shown in Fig. 1. This allows for maximum solar panel surface area to be exposed to the sun as well as ease for thermal control of the craft, with the rear acting to dissipate heat out to deep space.

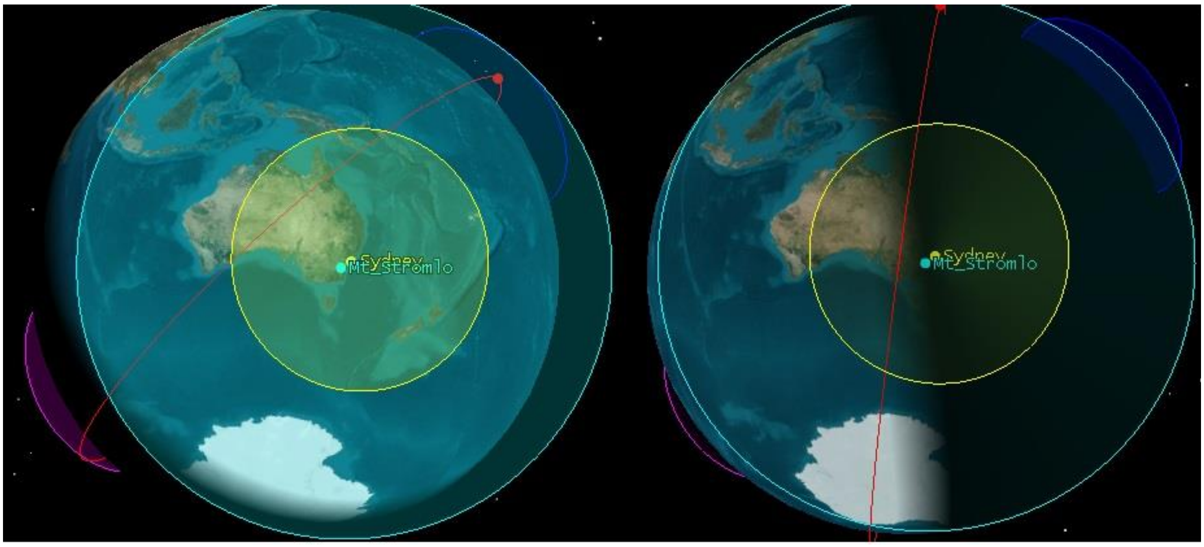

Figure 2. The two main candidate orbits for the MERS craft being the 47 degree inclination (Left) and the 98 degree inclination (Right) are shown in red. The ground station exposure periods for the proposed ground stations are represented by the yellow, turquoise, pink and blue shaded circles.

However configuration 2 comes at a cost, increasing the demand on the orbital control system to counter the increased drag placed on the craft, in order to keep the craft in the desired orbital configuration. Configuration 3 simply maintains the front of the craft facing in the direction of orbit and is utilized during re-entry.

In an attempt to maintain the commercial viability of the MERS mission, the craft has been designed to suit a range of orbits, at an orbital height of 450 km. The main focus is on the 47 degree inclination orbit and the 98 degree inclination sun-synchronous orbit, as shown in Fig. 2. The 47 degree orbit has been chosen to allow for maximum exposure for the communications system to a ground station located in Canberra. This means that the 47 degree orbit is ideal for data transfer. However, this comes at a cost to the EPS power generation consistency due to large orbital variations experienced. The effect orbital inclination has on EPS performance will be discussed in depth in the Beta angle component of the report. While the 98 degree orbit does not provide the optimal ground station exposure like the 47 degree orbit, unlike the 47 degree orbit it does however provide a more stable power generation profile.

II. EPS Definition

In general, EPS design is based around the development and integration of the four main subsystem components; power generation, regulation, storage and distribution. The power source, the subsystem component responsible for power generation, in this case a solar array, governs the maximum power usage of the craft as the power storage device is often not large enough to sustain a crafts power needs for the duration of a space mission.

Photovoltaic cells are the main means of power generation in space due to the readily available solar radiation. Solar panels currently can convert a maximum of 30% (Wertz, Everret, & Puschell) of the incoming solar radiation of approximately 1370 W per square meter (Kennwell & McDonald) Silicon semiconductors have an efficiency between 17-19% (Fraas & Partain, 1995) and Gallium Arsenide (GaAs) , a more commonly utilised photovoltaic cell, have an efficiency of around 26-29% (Clyde-Space, 2012).

The size of a solar array is typically determined by two main factors; the average power required for an orbit of the satellite system and the losses in the overall system. Photovoltaics work of the principle of using a P-type and n-type doped material physically separated to produce a p-n junction. The p type material is then excited by incoming photons to force electrons to flow from the p-type to the n-type material. The n-type and p-type materials are then connected in a circuit with an electrical load generating power. However solar radiation has a wide range of wavelengths other than visible light and only some wavelengths excite certain panels. For example, the most readily used panel is the triple junction GaAs cell that is only excited by light in the 700nm to 900 nm range (Bentham) The reason that only certain wavelengths activate solar cells is that light with two low a wavelength is absorbed by the material and those with too large a wavelength do not have the energy to excite the valence electrons. Power generated from photovoltaics is by no means constant, due to the efficiency degradation of cells as temperature increases as well as the inconsistency of solar radiation.

Regulation of the voltage generated by the solar arrays is necessary to ensure that the supplied voltage is at the required level and there are no peaks or troughs in the supply to the crafts system. Regulation is typically achieved using a DC-DC converter to accomplish a stable DC output from a varying DC input (Wertz, Everret, & Puschell).

Secondary batteries, capable of charging and discharging, are the typical choice of power storage device in space craft missions due to the ability to store generated power from sun exposed portions of an orbit and utilise this power during eclipse periods. Ideally without any losses or degradation to the battery state of charge and depth of discharge the system could continue to generate and redistribute necessary power as long as the system is serviceable. However, after a manufacturer determined amount of battery cycles the storage devices state of charge (SOC) degrades as well as the amount of power that can be discharged from the battery, the DOD. The rate of battery performance degradation can be rapidly increased from continuously overcharging the battery, hence making it important to not only produce enough power to maintain the system, but also not to produce too much so that the power system efficiency is degraded. However for short mission durations, like that of the MERS, overcharging effects on battery degradation can be ignored, as typically the mission duration is not long enough to have any noticeable effects on the system performance (Clyde Space, 2014).

The final component of the EPS, distribution, is achieved through a switching network and designated buses, operating at a level based on the nominal operating voltage of the subsystem components. The switching mechanism of the EPS operates to transfer power between the solar arrays, the battery and the subsystems, based on predetermined logic and the EPS operating mode. The usual allocation of buses is to operate a 'high' and a 'low' level voltage bus to cater for differing power requirements.

III. Methodology

Following the standard approach to EPS design outlined by the Space Mission Engineering (SMAD) text, (Wertz, Everret, & Puschell), outlines a design method that follows a top down approach, first identifying high

level system requirements and deriving lower level sub-system requirements. This design method has been adopted for this project. As shown in Fig. 3 the design method is a flow process that firstly identifies system requirements that allows for the selection and sizing of the power source, based on the power demands of the subsystems. After which the energy storage device can be selected and sized to ensure enough power during eclipse periods as well as an appropriate EOL charge. Identifying power regulation, control and distribution is the penultimate stage, tying together the power source, storage device and subsystem components, typically through a number of busses at different voltage levels to cater for different loads. Finally the mass, average power and peak power requirements of the system can be determined. The SMAD design method relies heavily on the establishment of clear, relevant system requirement, but also has scope for iterations of each step to produce the desired EPS. A major downfall in the design method is not determining the mass of the system until the last stage of the design. In a project were mass is one of the most vital components of the final system there is not enough control over this parameter. However this has been mitigated by outlining a strict requirement to produce a capable EPS with minimal subsystem mass.

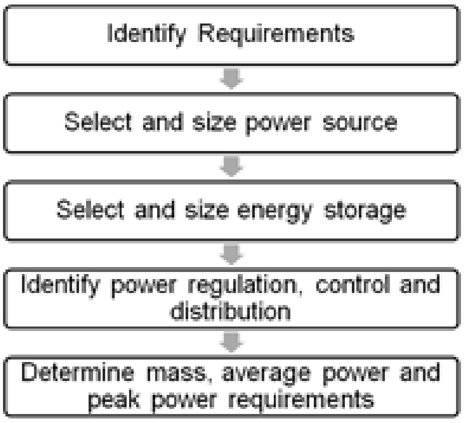

Figure 3. Space Mission Engineering Generic EPS design method (Wertz, Everret, & Puschell)

A. Requirements

The need statement of the EPS design is expressed as; to design an industry compatible Satellite electrical power system capable of efficiently providing sufficient power to the Microgravity Experiment Recovery Satellite sub-systems during a range of orbital parameters in a controlled manner, enabling the successful completion and return of microgravity experiments. In line with the above, the following requirements (Tuttle, 2013) were developed in the initial stages of the project;

1. The EPS shall be able to operate in a 47 degree and 98 degree Low Earth Orbit.
2. The EPS shall utilise proven COTS products where possible.
3. The EPS shall be compatible with Cube Sat components.
4. The EPS shall have minimal subsystem mass,
5. The EPS shall provide power during re-entry.

IV. Power Profile MATLAB Program

The power profile MATLAB program was developed as the main design tool used throughout the project to select and size both the power source and the energy storage device as defined by the SMAD EPS design method. As a basic overview, the program operates by taking orbital and craft parameters as inputs from the user and determines the crafts rotational position and hence the exposure to the sun per orbit, while taking into account the various efficiency factors that degrade the amount of power produced by the solar array, such as heat, packing factor and transfer efficiencies. The resulting output is a time dependent plot of the main EPS parameters being the battery SOC, the amount of power generated and the amount of power consumed. The Power Profile code in the MATLAB file UI.m follows the steps below to produce the power profile plots:

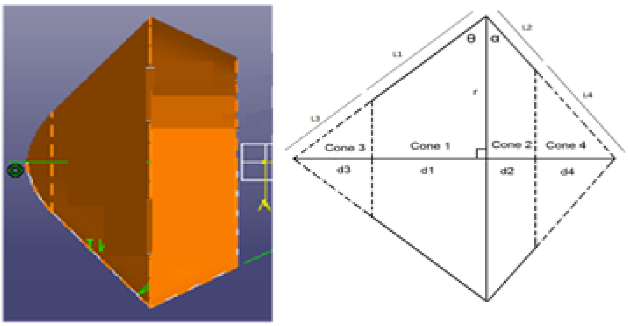

Figure 4. MERS craft side on view (Left) and the geometric model for the MERS craft used to determine surface area.

1. Orbital and craft parameters are input by the user. This includes the orbital height, number of orbits to be simulated, inclination, photovoltaic cell efficiency, packing factor of solar cells and the BOL SOC of the battery.
2. The input orbital parameters are used to determine the orbital period (T_0) of the satellite.

$$T_0 = \frac{2\pi}{\omega_s}, \text{ where } \omega_s = \left(\frac{\mu_e}{Rs^3}\right)^{\frac{1}{2}}$$

3. UI.m calls the MATLAB script surface_area.m to calculate the surface area of the MERS craft. The user inputs the craft parameters to determine the surface area available for solar panels to generate

power. This allows the user to test different craft sizes and the effect on power generation. Noting that the craft is modelled as two cones connected at the base, with the end of each cone removed. The spherical nose of the craft is not considered in the SA calculations as it will be utilised solely for the thermal protection system. This results in the surface area being calculated by:

$$A_{craft\_max} = \left(A_{cone1_{side}} - A_{cone3_{side}}\right) + \left(A_{cone2_{side}} - A_{cone4_{side}}\right) + A_{cone4\_base}$$
$$\therefore the\ A_{craft\_max} = (\pi r(l_1 + l_3) - \pi r_3 l_3) + (\pi r(l_2 + l_4) - \pi r_4 l_4) + \pi r_4^2$$
$$where\ the\ SA\ of\ a\ cone, A = \pi r^2 + \pi \mathrm{rl}$$

4. The crafts 360 degree orbit, $2\pi\ radians$, is then equally divided into a rotation array based on the angular rotation per minute of the craft. This allows the location of the craft in reference to the sun to be calculated. $Angular\ rotation\ per\ minute = \frac{2\pi}{T_0}$

5. A power generation array is produced based from the exposed surface area to the sun and the incident angle of the panels through its angular rotation. The power generation array is T_o long and has a power value for each location of the craft through the orbit.

6. The MATLAB script Temperature.m is called to calculate the temperature effects of the orbit on the solar panels. The temperature calculation is based off the surface area of the solar panels, the solar constant of 1370 W per square meter (Kennwell & McDonald) and the thermally coupled craft body as part of the heat dissipation. The temperature of the panel is first calculated starting at the steady state sun exposed temperature of the panel, 87.5 degrees assuming the panel reached equilibrium before entering an eclipse period at the starting time of 1 second. The temperature of the panel degrades until the panel exits the eclipse stage and starts to heat up. Solar panel efficiency is a maximum below a temperature of 28 degrees and degrades by 0.5 % for every 1 degree Celsius increase above the nominal temperature. This relationship between temperature and efficiency is used to generate the panel efficiency array. Other efficiencies such as packing factor and power transfer efficiency are also applied to the efficiency array to produce a more realistic power generation result. The packing factor is estimated at 80% from computer aided design images and is due to the difficulty of placing the panels in close proximity to each other on a conical shaped surface. The power transfer efficiency is set to 95% as defined by the use of a Peak Power Tracker (PPT) (Strain, 2010).

7. UI.m calls the power_usage.m MATLAB script. This script calls for user inputs to determine the sub system power requirements of the craft such as payload power, attitude and orbital control power usage, thermal power and transmission power. A full list of the components drawing power can be seen in Fig. 10. An associated power usage array is created, identifying the power draw at any particular point in time. The SOC of the battery can then be determined iteratively for time *t* with the following equation:

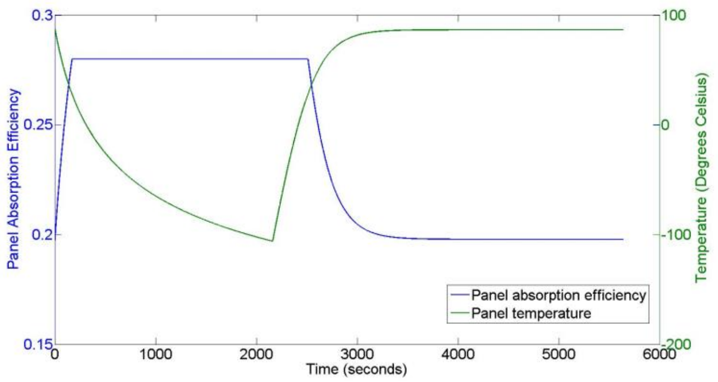

Figure 5. The effect temperature has on solar panel power generation efficiency. The plot shows that as temperature increases the efficiency of the solar panel decreases.

$$Battery\ SOC(t) = BatterySOC\ (t-1) - power\ usage\ (t) + power\ generation(t)$$

8. The Battery SOC for any time *t* in the given orbit is now known. The fluctuation of the beta angle is calculated from the solar vector (ŝ) and the orbital normal vector (Ô)through the following formulae:

$$\beta = \mathrm{asin}(\hat{O} \times \hat{s})$$

9. Plots of the battery SOC and yearly power generation are produced and important parameters are printed to the user such as peak power, maximum power and battery SOC.

A. Power Profile outputs

A power profile typically displays the main parameters of an EPS, being the battery state of charge, the amount of power generated and the power demands of the system. The graphs in Fig. 6 and Fig. 7 depict the power profile outputs for the two candidate orbital inclinations being the 47 degree and 98 degree orbits.

A 24 hour period is displayed indicating the power generated, power consumed and the resultant battery state of charge in the given orbital configuration. Both power profiles utilise a maximum battery power of 38 Watts and a solar array size of 0.24 square meters.

The consumed power is based off the nominal power requirements of candidate sub-system components, with the orbital control system actuating every 10 minutes to simulate the stabilisation of the craft in configuration 2. The seemingly random peaks in the power profile consumed power correspond to realistic transmission times for the communication system as the craft becomes exposed to one of the three candidate ground stations. The 47 degree orbit has an eclipse period of 36 minutes, indicated by the periodic drop of the generated power. It can be seen from the 47 degree power profile that the system has a net positive difference in the power generated to that consumed ensuring that the power utilised during the eclipse period can always be recharged during the sun exposure periods of the orbit. Minimal over charging occurs as the battery spends little time at the fully charged position. This indicates that the solar panel array size is well matched to the power needs of the craft. The power generation reaches the maximum of 38 Watts immediately after the craft exits the eclipse period due to the decreased temperature of the solar panel. When the panel is below the n**o**minal

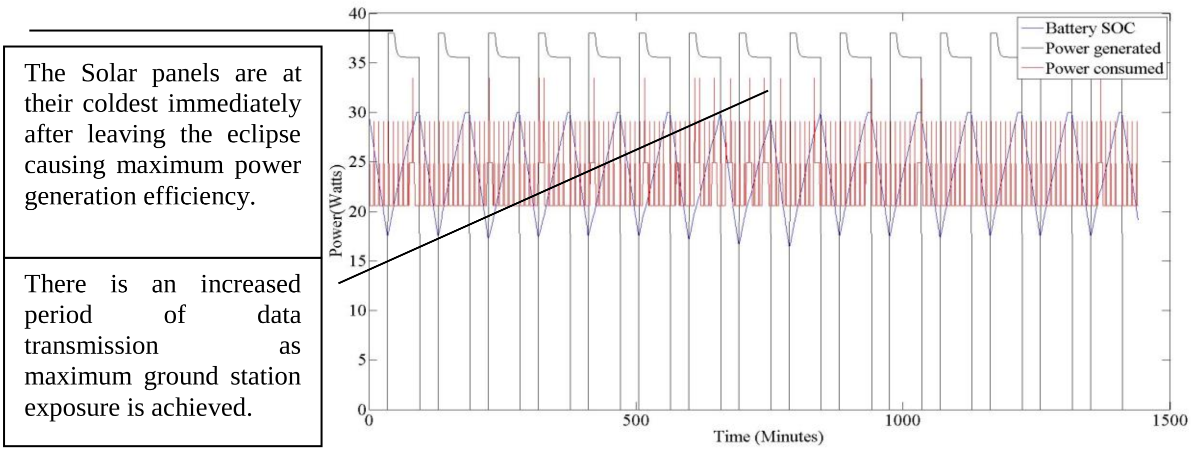

Figure 6. Power profile for MERS 47 degree candidate orbit. Peaks in the power consumption (red) correspond to transmission of data between MERS and one of the three candidate ground stations. The curved peak of the power generation plot (Blue) is caused by the increase in efficiency of the panels immediately after the eclipse period.

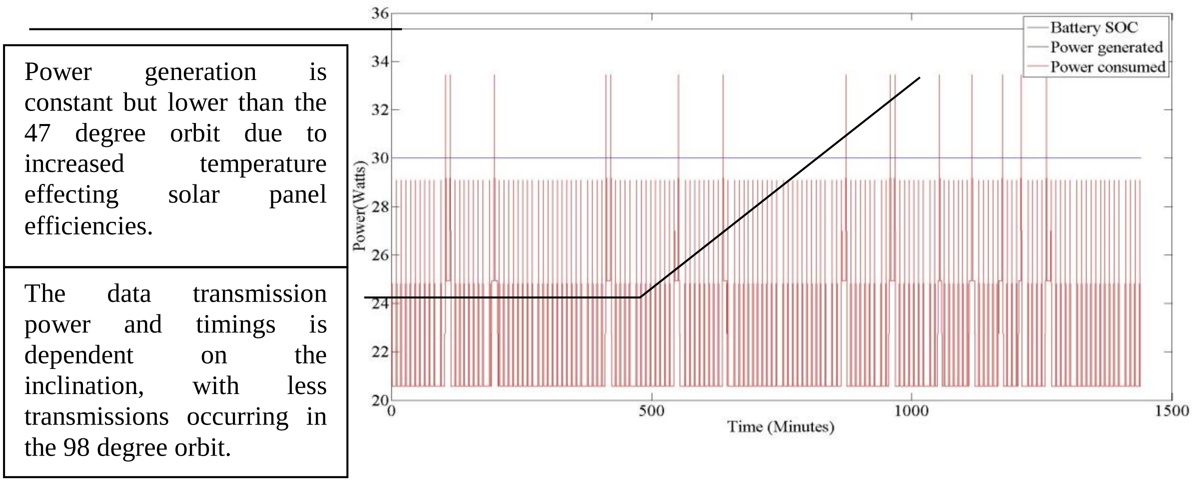

Figure 7. Power profile for MERS 98 degree candidate orbit. The power generated plot (Blue) and battery state of charge is constant due to the constant sun exposure in a dawn to dusk sun-synchronous orbit. The data transmission times have been determined specifically for each inclination.

temperature of 28 degrees the panels operates at maximum efficiency. After approximately 6 minutes the panel begins to heat past the nominal temperature and the power generation degrades by 0.5% for every 1 degree Celsius increase in temperature.

In contrast, the 98 degree orbit has no eclipse period allowing for constant power generation but at a lower efficiency due to the sustained heat in the panel from constant sun exposure. Although the efficiency of the panel is degraded the constant exposure allows for a greater amount of power generated. It can be noted from the 98 degree inclination power profile that the state of charge of the battery and the power generated are constant. At no point, does the power consumed exceed the power generated, hence the battery is not utilised during experiment mode. However the battery still serves a purpose later in the MERS mission to power systems during re-entry. Both power profiles indicate that the craft produces sufficient power in the given configurations for the best case orbit. However throughout the year, the amount of power that can be generated in particular orbits can differ greatly due to degradation in orbits and an increase in beta angle.

B. Beta angle outputs

The beta angle is defined as the angle between the vector normal to the orbital plane (Ortiz Longo & Rickman, 1995) of the craft and the sun vector. The beta angle determines how much of the solar power can be utilised at a given point in time. Beta angle can span between -90 and 90 degrees with an angle of 0 degrees corresponding to maximum sun exposure. The beta fluctuates due to changes in the right node of ascension caused by the oblateness of the Earth For certain orbits this causes large fluctuations in the angle between the

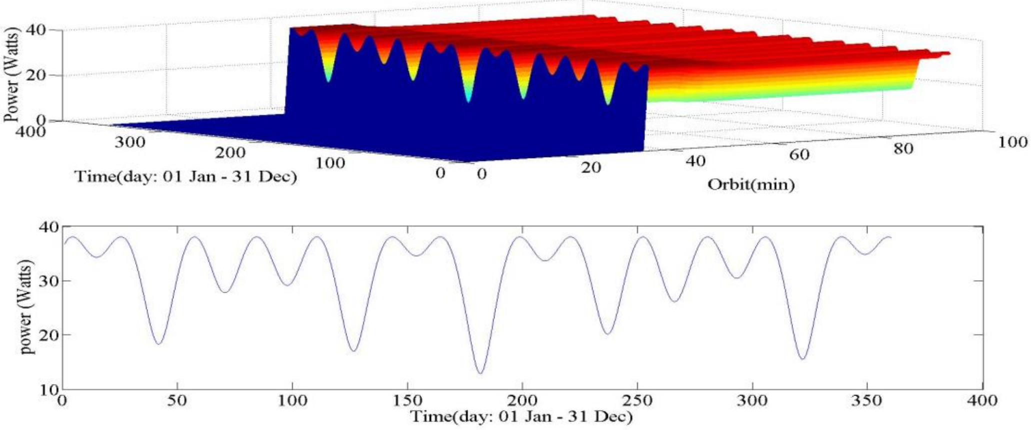

Figure 8. Beta angle plot for a 47 degree orbit over a year indicating the power fluctuation year is the beta angle changes.

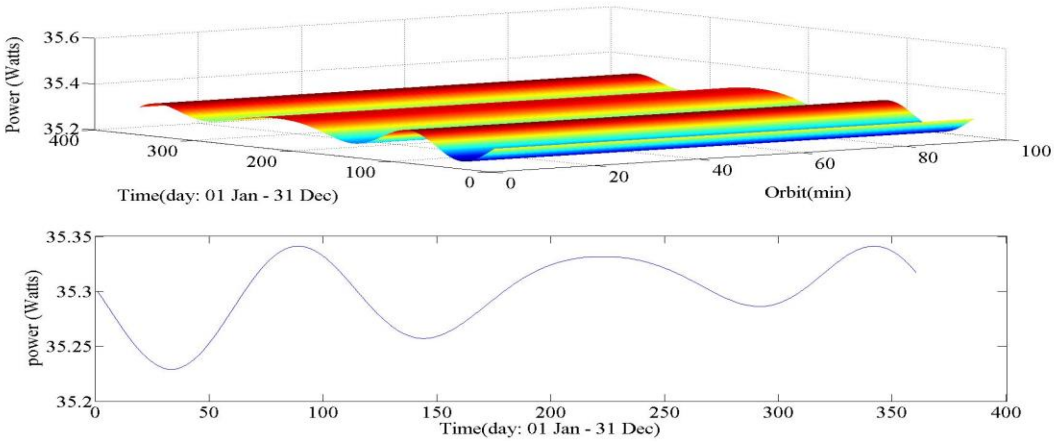

Figure 9. A Beta angle plot for a 98 degree orbit over a year, displaying a more consistent power generation than the 47 degree inclination orbit.

orbital plain and the sun vector.

The beta angle plot for the 47 degree orbit, Fig. 8, shows that is such an orbit where large fluctuations are experienced. Thus, deciding to enter a 47 degree orbit for the MERS would require careful planning to ensure

that the craft is not launched at a time period when the power generation is a minimum as the craft would not be able to sustain the power needs of all of the subsystems, resulting in a system failure or sub-optimal performance. Conversely the 98 degree orbit has a very consistent beta angle throughout the year resulting in a reliable power generation regardless of the launch date. Comparing the two orbits, the 47 degree and 98 degree have a 20 Watt fluctuation and 0.1 watt fluctuation respectively. Particular note should be taken when observing the power axis of Fig. 9 as the axis only ranges between 35.2 Watts and 35.6 Watts.

C. Derived Parameters

Parameter	47 Degree Orbit	98 Degree Orbit
Peak power (Watts)	38	35.3
Power generated in a 24 hour period (Watt Hours)	534	848
Power consumed in a 24 hour period (Watt Hours)	527	524
Maximum Power fluctuation over a year (Watts)	24	0.1
Maximum panel efficiency	28%	22%
Minimum Solar panel SA m^2	0.24	0.16
Maximum Sun exposure time (minutes)	58	94
Eclipse Duration (minutes)	36	0

Table 1. Derived system parameters.

Based on the above analysis of the two main candidate orbits, the following derived parameters indicate the main differences between the orbits and allow for the selection and sizing of the power source and storage device. For all values that require a solar panel surface area 0.24 square meters was utilized as it allows for adequate power generation in both configurations. The peak power generated by the 47 degree orbit is larger due to the solar panel temperature reaching subzero degrees Celsius temperatures during the eclipse period, allowing for maximum power generation on the initial 6 minutes of sun exposure before the panel increases above the nominal temperature of 28 degrees where the power generation starts to degrade. It can be seen that the 47 degree orbit, operates at a higher efficiency with a shorter exposure time than the 98 degree orbit that is constantly exposed to the sun but a lower efficiency. The values shown in table 1 allowed for the synthesis of a possible design solution to meet the sub-system and MERS requirements.

V. Design Solution

A design solution that can be tailored to operate in both the 47 and 98 degree inclination orbits is centered around the Clyde space 38 W flexible EPS (Strain, 2010) The system is designed to handle a combined peak craft power demand of 28 W generating a peak of 34 W through 6 separate solar arrays with a combined surface area of 0.24m^2.

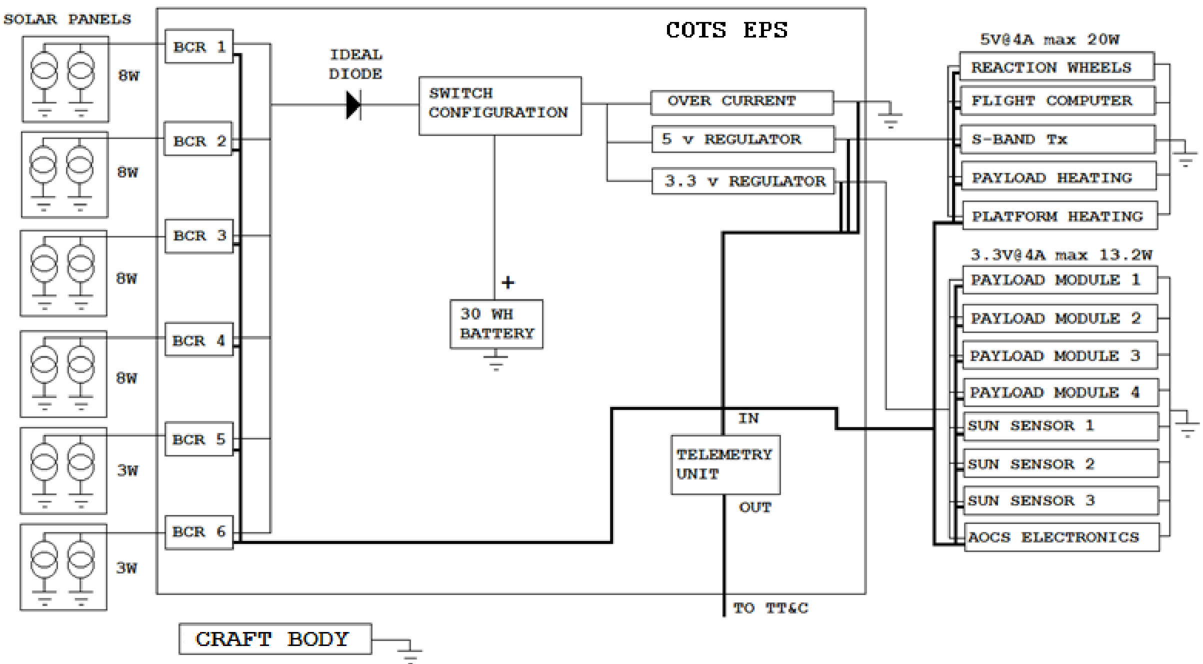

Figure 10. MERS EPS circuit block diagram utilising the Clyde Space CSXUEPS2-42 flexible EPS.

The power system utilises a commercial of the shelf product manufactured by Clyde Space being the CSXUEPS2-42 flexible EPS (Strain, 2010). Proven COTS products are utilised for the MERS EPS to ensure minimal chance of failure in an already complex mission. Each solar panel is attached to a Battery Charge Regulator (BCR) design to ensure interface between the internal electrical system and the power generated at the solar panels. The 8W BCR’s (1-4) utilise a buck converter and the 3W BCR’s (5-6) utilise a SEPIC converter. Each BCR has a Maximum Power Point Tracker (MPPT) used to position the electrical point of operation of the solar panel ensure maximum power transfer between the solar panel arrays and the EPS.

The entire EPS has an estimated mass of 1,440g. Breaking down the mass total, the solar panels have a designated mass of 500g, the CSXUEPS2-42 ¬140g, the battery 300g and the wiring and harnessing 500g. The estimated financial cost for all of the components of the EPS is $45,450 US with the solar panels costing approximately $30,000, the CSXUEPS2-42 $9,800, the battery $3,850 (Clyde Space CS-SBAT2-30) and the wiring $1,800. The solar panels are a triple junction GaAs cell with an efficiency of approximately 28% and a packing factor of 80%. (Clyde-Space, 2012) The solar panel must be a minimum size of $0.16m^2$ to sustain the power requirements of the craft with a surface area of $0.24m^2$ used to ensure excess power is produced to handle budget blow outs. Due to the unique shape of the MERS the solar panels must be positioned on a 45 degree angle to the incident solar rays causing a large decrease in the effective power generated by the solar cells as only a portion of the visible light is absorbed. The peak power generated is 21% greater than the peak requirement of the MERS system however this allows for future increases in power demands and allows redundancy in the system if a panel is damaged by debris.

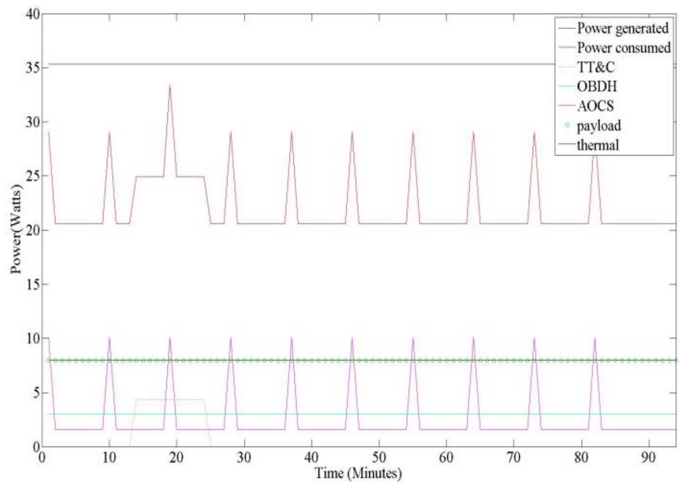

Figure 11. Power profile for a 98 degree inclination for a single orbit.

Fig. 11 shows the power profile of the MERS for the designated 450km Dawn to Dusk sun synchronous orbit. The power generated by the solar panels is indicated in blue and the total power consumption of the craft is shown in green. Note that the power demand only reaches above the generation during the activation of the stabilisation wheels. The individual craft sub-systems power requirements including the Telemetry tracking and command (TT&C), On Board Data Handling (OBDH), Altitude and Orbital Control System (AOCS), payload and thermal requirements are also shown in the plot.

The 10 peak points indicate the activation of the stabilisation wheels for a minute to re-orientate the craft.

The craft utilises a 30 Watt Hour battery to handle peak power surges as well as providing adequate end of life power for re-entry. The Clyde Space CS-SBAT2-30 has an 8.2 V terminal voltage and a verified cycle life of 5000 cycles. With a planned mission life of 2 months and an orbit of approximately 94 minutes the battery is likely to undertake a maximum of 935 cycles having minimum degradation in the DoD. For the majority of the mission the generated power will be enough to sustain the craft with the power being transferred directly to the crafts sub-systems bypassing the battery. The battery will mainly provide power for the beginning and the end of the mission when solar power is unavailable. To ensure the battery is not degraded by increased temperature it will be located close to the rear center of the craft being the most stable cool temperature environment of the craft, close to the payload modules.

The EPS has a 5V at a maximum of 4A bus (20W) and a 3.3 V at a maximum of 4A bus (13.2W) to handle the varying power needs of the subsystem components. The wiring to be used is 26AG high quality wire that will be more than capable of withstanding the power demands of the craft. Some or all of the payload modules may potentially have larger power demands depending on the type of experiment being conducted so the EPS must have the flexibility to accommodate one or more of the payload modules being connected to the 5V bus. As mentioned previously the solar array inputs are regulated by individual BCR’s that control the voltage transfer from the solar arrays to the battery. These BCR’s operate in two possible modes being MPPT mode and End of Charge (EoC) mode. The MPPT mode is used when the battery SOC is well below the designated pre-set EoC state and the maximum power from the solar panels is harnessed being transferred to the battery. Once the EoC voltage is reached the BCR enters EoC mode where the battery is kept at a constant output voltage and the current is tapered to top up the battery. The EoC mode moves the point of the solar array operation away from

the maximum power point to only generate the amount of power required. The downside for this is that the excess power is left in the solar array as heat. The CSXUEPS2-42 also has inbuilt over current protection to ensure that if there is a surge in the system the sub-subsystem devices are not impacted.

V. Conclusions

The preliminary design stages of the EPS have resulted in a flexible preliminary architecture capable of operating in variety of orbits, mainly the 47 and 98 degree candidate orbits. Supplementary to the design architecture a MATLAB program capable of determining the power profile for given orbital and craft parameters was successfully developed and utilised as a design tool throughout the design.

In terms of system performance, of particular note, was the large variations in the power profile for the craft in a 47 degree orbit compared to the 98 degree inclination, is the maximum annual variation of 24 Watts and 0.1 Watts for the 47 and 98 degree inclination orbits respectively. Choosing to launch the craft in a 47 degree inclination would require careful planning to ensure that the mission does not coincide with low power generation periods caused by an increased beta angle. Alternatively the 98 degree inclination orbit allows for more stable power generation at all points throughout the year and a much larger power generation producing 314 Watt hours more than the 47 degree orbit in a 24 hour period. If the craft is launched in the 98 degree orbit the solar panel surface area can be reduced to a minimum of $0.16m^2$ from $0.24m^2$, decreasing the mass of an already light subsystem.

The EPS solution discussed is a low weight commercially viable solution with an estimated cost of $45,450 US and a weight of 1.5 Kg. This system is well within weight requirements for the craft and utilizes entirely COTS products, increasing the reliability and likely hood of successful performance.

VII. Recommendations

The project has a significant scope for future development, focused primarily on developing and testing low level requirements of the EPS as well as the construction and testing of the components. Aspects that have not been addressed properly include the switching configuration, grounding strategy and the feasibility of producing the EPS in house at UNSW@ADFA.

While the switching configuration is important to the distribution facet of the EPS, it has not been explored at this stage due to the large scope for subsystem components to change as the design continues. Thought has been given to the potential problem of photovoltaics damaging the craft upon re-entry due to the uneven distribution of heat placed on the surface, but the idea has not been investigated beyond that.

Testing of COTS products outlined above to further validate project findings would be required before proceeding any further into the development of the craft. Physical positioning of electrical components has been considered, but not finalized.

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