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Miniaturised Power and Propulsion for Small, Low Cost Space Exploration Missions: Current Status and a Look at the Future

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Abstract

Space exploration has historically been a costly endeavour. In the 1990s low cost 'microspacecraft' using miniaturised components such as modern microprocessors, drawn from the stocks of commercial manufacturers became feasible. Microspacecraft with a mass typically between 10 and 100kg can carry out useful missions such as regular Earth imaging of natural disasters, inspection of larger spacecraft and testing components, and have demonstrated useful lives of several years. The simultaneous availability of relatively inexpensive Russian launchers in the 1990s has enabled highly cost-effective space missions, for example where several spacecraft operate together in a 'constellation'. Microspacecraft typically use the traditional photovoltaic / chemical battery combination common to the majority of Earth orbiting spacecraft, however this is less practical for very small spacecraft and those operating in unusual environments such as far from the sun (e.g. Europa) or in the clouds above Venus' surface. The scope of this problem, and potential solutions which might allow cost-effective microspacecraft to explore space and offer new services in the 21st century are outlined.

Keywords: space microsatellite cost-effective exploration photovoltaic

1. INTRODUCTION TO SPACECRAFT POWER

Spacecraft design in general and the demands of orbiting the Earth:

- Vacuum with typical pressures <10^{-6}Pa,
- Extreme thermal cycling, as the spacecraft passes in and out of the Earth's shadow, typically between -50 and +90°C every 90 minutes for several years,
- Radiation, with exposure to a high flux of electrons and protons trapped by the Earth's magnetic field, as well as high energy cosmic rays from outer space,
- Extended vibration to several times Earth's gravity during launch, and short shock's orders of magnitude greater on separation from the launcher,
- Erosive species such as atomic oxygen, coupled with hypervelocity impacts of space dust, dirt and debris in other orbits,
- A general inability to inspect, repair/replace components or refuel once in orbit. In context, how many cars are designed to drive for 100000km over several years without refuelling or being serviced?

The last point is particularly acute when considered in the context of most terrestrial machinery, which either draws power from conveniently sited grid outlets, or is regularly refilled with hydrocarbon based fuels such as diesel or kerosene, which are then combusted with oxygen from freely available air. Fortunately in space the energy of the sun is considerable and has been tapped since the earliest space missions. At the Earth's distance from the sun of about 144Mkm, the radiant solar flux across the entire spectrum is 1343W/m^2 (AM0). This compares to <1000W/m^2 on the surface of the Earth (AM1). In theory a relatively small collector could harvest sufficient energy for many spacecraft, which are considerably less power hungry than, say a domestic automobile. For example a Toyota Corolla 1.6litre engine draws about 75kW, and a Toshiba laptop about 60W. However broad spectrum solar energy conversion is relatively inefficient, typically 5-20% of theoretical. Further, the nature of spacecraft orbits around the Earth results in shadowing of the sun for around 35 minutes in a 95 minute orbit. A means of storing and temporarily dispensing power is needed for space missions to continue without interruption.

Early spacecraft such as Sputnik used a non-rechargeable battery and operated for days or weeks. The space industry supported the development of higher efficiency photovoltaics, which were then coupled to rechargeable battery cells such that these could be charged during sunlight, while simultaneously producing power for spacecraft operations: the 'solar battery'. Switching to stored energy in the batteries takes place in eclipse, power conditioning modules ensuring a constant voltage uninterrupted power supply. Maximum solar incidence is assured with array area has to be maximised and the incidence kept as normal as possible to the cells. Shadowing of cells e.g. by parts of the spacecraft structure is also undesirable. Solar battery requirements have lead to two principle types of spacecraft design:

Figure 1: Large spacecraft with deployable, steerable solar arrays

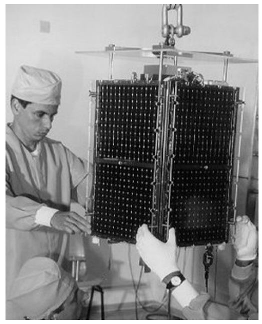

Figure 2: Small spacecraft with fixed arrays mounted on body(PoSat)

Figure 1 shows a typical large spacecraft with a pair of photovoltaic array panels folded against the sides of the platform or bus (central cube in this case), and deployed after spacecraft separation from the launcher in orbit. These then track the sun using 1 or 2 axis solar array drive mechanisms which also transmit power into the body of the spacecraft. This configuration is common for large (several tons) communications spacecraft operating in fixed positions 36000km above the Earth. Figure 2 shows a small, 360 x 360 x 690mm body (excluding antenna) 60kg microspacecraft pioneered by SSTL, with 4 fixed body mounted solar panels. Avoiding deployment or steering mechanisms on the panels lowers complexity and hence risk, and is ideally suited to a low cost spacecraft environment. Power can also be obtained in almost any attitude to the sun, although the average power is much less than from an equivalent area tracked array, Figure 1.

2. POWER DEMANDS FOR SMALL SPACECRAFT

Small spacecraft originated in response to opportunities to carry auxiliary or secondary payloads in residual space left by a larger spacecraft inside the shroud of a launch vehicle. Cost, volume, then mass rather than performance or lifetime in space were the main mission drivers. The spacecraft in Figure 2 was designed fit spare space on the European Ariane V and US Delta II rockets. Missions such as UoSat, PoSat, and TiungSat demonstrated the utility of small, <100kg spacecraft [1] for low cost Earth imaging missions, demonstration of commercial-off-the-shelf (COTS) technologies in space, and as training in satellite engineering for both students and engineers from developing nations. The more recent enhanced microsatellites, forming the basis of the Disaster Monitoring Constellation [2] have demonstrated regular (daily) Earth imaging with moderate resolution (32m ground sampling density) and over large areas (600 x 600km). Although this has required a growth in the size and mass of the spacecraft, towards 1m^3 and beyond 100kg respectively, SSTL and the University of Surrey Space Centre (SSC) have also developed much smaller spacecraft. Over a 9 month period between 1999 and mid 2000 the SNAP, or Surrey Nanosatellite Applications Platform was designed built and launched [3]. SNAP's, launch mass was 6.5kg, plus a 1.8kg adapter to connect it to the launcher. SNAP is shown in Figures 3 and 4 below

Figure 3: SNAP during testing, showing3 of the 4 solar panels (base area occupied by launcher separation system)

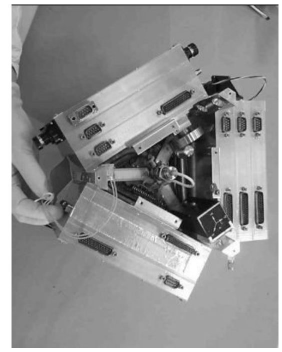

Figure 4: SNAP structure, prior to solar panel addition, showing triangular arrangement of module trays.

2.1. SNAP power system

SSTL and SSC's approach to SNAP design extended the Commercial-Off-The-Shelf (COTS)-based "modular" design approach applied so successfully to micro-satellites since 1979. Key principles of this approach include;

- Concurrent design in a small team of engineers;
- Modular design with standardised electrical and mechanical interfaces, based on stacks of electronics trays with simple connectors.
- Ease of assembly and testing.
- A robust design while using COTS technologies
- Simplicity!

A simple standard electrical interface was prescribed for each module, consisting of regulated 5V and raw battery (V_{batt} ~7.2V) power connections. A bidirectional Controller-Area-Network (CAN) bus was used for data transfer. Connections were provided via a 9-way D-type connector, standard to all modules. Each module was designed to house a standard Eurocard printed circuit board, 160mm × 100mm, module size has since evolved to 190 × 135 × (22 or 33mm). All the SNAP modules except the power system could be powered and tested using just the standard 9-way connector.

The SNAP spacecraft generated power through four body mounted solar panels, each populated with 18% efficient GaAs cells producing nominally 0.5A per panel at 12V, i.e. ~6W in total. Because of the mechanical configuration of the satellite (Figure 4), the total orbit average power available was also approximately 6W. However, the minimum *required* bus power was only 650 mW, representing only the receiver and power system operating. Therefore under nominal power conditions these systems, together with the flight computer, and the attitude control system (ACS) can be in continuous use. In addition, the GPS navigation system, payloads and transmitter can be activated periodically as required. The SNAP power system is shown in Figure 5 and a schematic of the component layout in Figure 6:

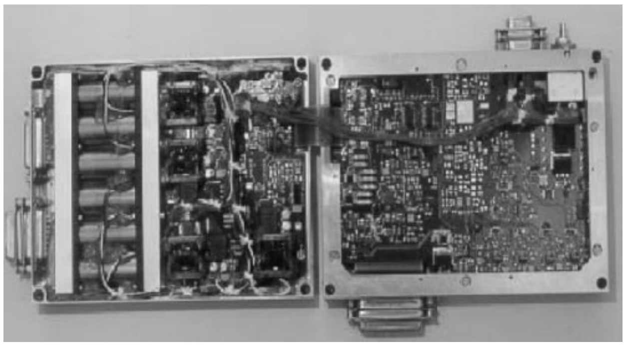

Figure 5: SNAP power system contained in two trays, each 160 × 100 × 30mm. Left side: 6 cell NiCd battery, battery charge regulators. Right side: Power conditioning and power distribution module. D-type connectors visible.

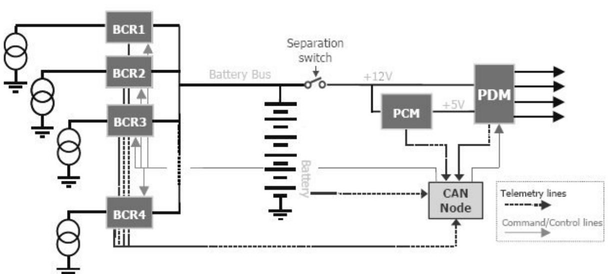

Figure 6: SNAP power system block diagram
BCR = Battery Charge Regulator, PCM = Power Conditioning Module; PDM = Power Distribution Module

Each solar panel supplies an independent BCR implementing maximum-power-point tracking in hardware for optimum power transfer efficiency. The BCRs charge a single 10 Whr battery consisting of six 'A' sized SANYO KR-1400AE Cadnica cells series-linked to give a nominal 7.2V. The battery pack mass is 270g, giving an energy density of 37 Whr/kg. Overcharging is prevented using a temperature compensated end-of-charge voltage trigger to switch the BCRs into a trickle-charging mode.

The PCM provides a regulated 5V supply, and an unregulated 12V supply for the spacecraft systems. The maximum total current that can be safely drawn from the PCM is ~3A. In total, the battery can sustain up to ~10A (i.e. ~60W) output for a few minutes. This feature was designed to allow future SNAPs to support high-power-demand payloads or thrusters. The PDM provides over current protection, ON/OFF control and current telemetry for all non-essential loads using the regulated 5V and battery bus lines.

2.2. SNAP power budget

A power budget for the SNAP nanosatellite is compared with a more typical microsatellite such as PoSat from Figure 2, in Table 1 below:

Table 1: SNAP and microsatellite Power budget

	Microsatellite	SNAP nanosatellite	
Module	Average power (W)	Power (W) × % duty	Avg. power (W)
Attitude control	2.0+ (2.8-4.5)	0.42@100	0.42
Wheel	1.2 × 3	0.75@100	0.75
Magne-torquer	1.0 × 3	3 × 0.1@10	0.03
GPS	6	1@5	0.05
Power conditioner	5 *(power system and dual hot redundant receivers)*	0.51@100	0.51
Battery charge reg.		0.51@100	0.51
VHF receiver		0.4@100	0.4
S-band transmitter	5.6 (low rate 38.4kbps)	3.25@10	0.33
Computer (double)	2.2-3.6	0.76@100 0.76@0	0.76
Propulsion	41 when in use (occasional)	5@0 (occasional)	
TOTAL	27.4 – 33.3 + Propulsion		3.75

Although SNAP required only 3.75W of power on average (and less than 1.5W for the essential power system and VHF receiver), compared to the 6W available, payloads are excluded from the table. The envisaged role for SNAP was to fly freely around other spacecraft inspecting them for damage, degradation and monitoring functionality e.g. during critical activities such as deploying solar arrays or antennae. SNAP therefore included a spread spectrum intersatellite link payload, requiring 2.5W and transmitting continuous position and orientation readouts taken from GPS and attitude control systems during rendezvous, and a CMOS camera system requiring 1W on a 66% duty cycle, An actual rendezvous (SNAP did not succeed in this task) would also required greater use of the propulsion subsystem, perhaps 10% of the time thus requiring 0.5W. It is easy to see how the meagre power budget could be exceeded with only modest manoeuvring, with the added complication that battery charging for periods of eclipse must also take place. A further complication is that SNAP had a very low ballistic coefficient, i.e. the ratio of mass to frontal cross sectional area and drag coefficient, which dictates how much atmospheric drag affects the spacecraft. At the orbit altitude of 696km, SNAP was found to decrease in altitude at 15m / day, almost 10m/day faster than the larger 60kg microsatellite it was launched alongside. Additional propulsion operation is also needed to mitigate this drag.

Suggestions for addressing potential power shortages in SNAP follow-on missions have included using higher efficiency triple junction photovoltaics (in excess of 26% for Emcore cells, and Spectrolab have reported a world record of 38% recently) and enlarging the solar panels slightly to match the platform size. However a recent design study for a SNAP platform for formation flying with the ESA PROBA small satellite mission found that these two solutions only increased orbit average power to around 11-14W depending on panel angle. However the desire to improve propulsion performance by heating propellant, thus allowing extended orbital manoeuvring, drag offsetting suggests that the modest increases in power generation could still be overwhelmed by the demands of manoeuvring. Increasing downlink data rates and camera scan rates would exacerbate this problem. A brief examination of alternative power sources, in particular fuel cells, or a hydrocarbon engine such as the micro-Wankel or rotary engine system pioneered at UC Berkeley [4] found that significant technical barriers exist to the use of such an engine outside of the Earth's atmosphere. Despite the attractive energy density of hydrocarbon fuels (>10000Wh/kg) compared to primary chemical batteries (up to 740Wh/kg for Li Thionyl Chloride $LiSoCl_2$), and rechargeable batteries, no oxidiser is available in space for a combustion engine or a fuel cell to use. Storage of oxidiser considerably reduces the fuel's energy density, and when coupled with the relatively poor thermal – electrical conversion efficiency, the primary chemical battery for at least short missions is still more attractive.

3. PALMSAT

The cost of a typical (sophisticated) micro/nano-satellite mission is of the order of $1-10 million, i.e. orders of magnitude lower than traditional space missions, but still beyond the reach of many organisations who would like to have access to space. Fortunately the continuing increase in capability of low-power microelectronics coupled with novel micro-electro-(optical)-mechanical systems (MEMS or MOEMS) technology have made it possible to design ultra-small (circa 1 kg) satellites, which have mission costs of just a few tens of thousands of dollars, thus enabling even educational institutes to have access to space within their limited financial resources. A key development in this process has been the "CubeSat" concept proposed by Robert Twiggs of Stanford University [5]. A CubeSat is an ultra small (~100 × 100 × 100mm), ~1kg platform with the following attributes:

- A standard physical layout and design guidelines.
- A standard, flight proven deployment system.
- Coordination of required documents and export licenses.
- Integration and acceptance testing facilities with formalized schedules.
- Shipment of flight hardware to the launch site and integration to the launcher.
- Confirmation of successful deployment and telemetry information.

The CubeSat programme is intended primarily as an educational vehicle. SSC and SSTL have begun their own programme, PalmSat, to both teach systems and specific engineering principles to under- and post-graduate students, but also to act as a carrier for testing advanced technologies. A PalmSat model is shown in Figure 7,and an exploded view of one possible interior layout in Figure 8. PalmSat aims to build upon the success SNAP taking the concept of spacecraft miniaturisation a step further, from a modular COTS technology based spacecraft formed of "Eurocard" (165 mm x 120mm) sized payload and bus-system modules, to one based on "credit-card" (90 mm x 55 mm) sized modules.

Figure 7: PalmSat model with deployed solar panels, compared with coffee mug.

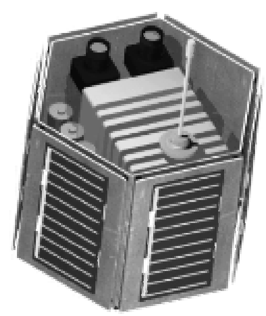

Figure 8: PalmSat CAD model showing credit card module stack, batteries (left), CMOS imagers (top), & antenna.

PalmSat will carry miniature attitude control systems, and ultimately propulsion. Its first mission will attempt to demonstrate spacecraft rendezvous and remote-inspection, using CMOS camera technology. Other miniature payloads are under development, including ionising particle detectors, magneto-resistive magnetometers, GPS receivers, thermal-infra-red micro-bolometer imagers, near ultra-violet radiometers and multi-spectral imagers, which should enable PalmSat-class spacecraft to carry out meaningful scientific investigations in a cost-effective manner, and test new small devices for SSTL [6].

At present the commercial interests of SSTL have focused on larger (100kg+) missions so PalmSat has yet to be tested in space, although substantial work has been carried out on all the relevant subsystems. PalmSat's power bus is likely to be unregulated using a peak-power tracking topology. Triple-junction solar-cells will be used to maximize the power available, although 6 double sided fold-out panels are still required to give an average of 4W in low Earth orbit sunlight, depending on attitude. Each panel gives ~210 mA at ~4.2 V under load and at 28°C, therefore the input power will vary between ~1.7W and 5.2 W. Two cells per panel (18 panels in all, front and rear of deployable) gives a raw solar-panel operating voltage of approximately 4V, sufficient to drive DC-DC power converters. This raw voltage is stepped-up to ~6-8 V by a PIC-controlled boost BCR. This regulator also sets the operating point of the solar panels. Students have produced and examined BCR designs using a discrete power-FET solution, and an integrated COTS power regulator. The discrete solution gives better control over the maximum power-point-tracking of each individual panel and could obtain almost a 10% improvement in power availability. However, the COTS solution is simpler to implement. The same Sanyo KR-1400AE 'AA' type NiCd batteries which flew on SNAP are still preferred. These cells have a storage capacity of 1400 mAhr at 1.2V per cell, thus a 5-cell pack provides ~ 8.4 Whr of energy in a 120g package (i.e.~70 Whr/kg). In a typical sun-synchronous low Earth orbit, the eclipse time is 35 minutes, which at the 4W generated by the solar cells would only 27% discharge the cells. Low depths of discharge enable a NiCd battery to survive several thousand cycles, to date Ni-MH and Li-ion cells have not acquired the same heritage.

Although PalmSat looks like it has little need of more advanced power systems, particularly energy storage, a recent study by SSTL and AEA Technology [7] showed the benefits obtainable from replacing NiCd cells with commercial Li-ion polymer cells such as the Ionity LiP 345585. This is a flat, 85 × 55 × 3.5mm cell which might be much better suited to packing within the tight PalmSat volume than the cylindrical AA NiCd cells, shown in Figure 8 alongside the credit card module stack but since rearranged to pack in the stack. Each Ionity cell weighs 33g, stores 1600mAh and discharges at 3-4.2V. Two cells could easily be placed in the space allocated to the NiCd battery pack, and would offer an energy reserve of ~12Wh, an increase of almost 50% compared to NiCd cells, for a mass of almost 50% compared to the NiCd cells. Alternatively, the small thickness of the Ionity cells in the space 20mm wide vacated by the NiCd cells allows up to 5 Li-P cells to be placed in the volume, offering a potential capacity 3.5× that of the NiCd cells for no net gain in mass. Alternatively an additional card, such as an extra receiver / modem or payload could be accommodated in the space saved by replacing Ni-Cd with Li-ion polymer.

4. CHIPSAT

CHIPSat, or 'satellite-on-a-chip' is a possible next step to PalmSat, the latter making use of advanced discrete COTS components tightly packed in the smallest possible volume, the former utilising the integration benefits of MEMS to add as much functionality as possible onto a single wafer or into a package. The ultimate goal is an entire spacecraft fabricated onto a silicon wafer or bonded set of wafers, for an envisaged mass between 0.1 and 1kg and opening up the possibility of mass producing large numbers of spacecraft for potentially the cost of a mobile telephone today. Launch costs, at thousands of € or $ per kg to low earth orbit, are also reduced as one launch vehicle could inject literally thousands of these satellites into their mission orbits. Production schedules are also reduced as the number of component interfaces has been reduced to a single development system level. Current research at the SSC aims to establish the feasibility "of the smallest possible satellite built as a monolithic integrated circuit that can be launched into space to perform a mission while communicating with a ground station" [8].

4.1. Applications of CHIPSat

It is not yet clear which space missions are uniquely enabled by CHIPSat technology, although considerable thought is being devoted. The vast distances , and consequent large quantities of propellant required for robotic interplanetary exploration favour the lowest spacecraft mass possible and hence CHIPSat. A large number of CHIPSats could also be launched simultaneously while still keeping mass low, thus reducing mission risk due to critical subsystem failures. Slightly closer to Earth, wide area field monitoring such as mapping the earth's magnetotail or magnetic field shape downstream of the sun and its response to changes in solar activity is a possible science goal. A swarm of CHIPSats distributed over a wide are could enable high resolution and wide area field measurements for a low cost. Each CHIPSat would only require a magnetometer and ranging device to spatially locate its readings. Distributed space based radar, where a number of CHIPSats form nodes of an extremely large synthetic aperture is also attractive. The complications of formation flying and intra-satellite communication must be demonstrated first, and the associated propulsion, ranging and transceiver technologies made tractable at a wafer scale.

4.2. Technology drivers for CHIPSat

The approach currently being taken at SSC, supported by SSTL, to enabling CHIPSat (while the mission applications are being refined) is to use an iterative spacecraft subsystem design process designed for larger spacecraft. This is mapped onto a physical configuration based on a modern semiconductor process available for prototype runs, such as the IBM 0.13m mixed-mode process available through the MOSIS prototyping service [8]. A 200 mm (or 8in ∅) reference silicon wafer with a mass of 55g, which can hold up sixty 18 × 20mm dies or in this case spacecraft components or even subsystems will be the basis of a wafer scale integrated spacecraft. Figures 9 and 10 show an 8in. silicon wafer, and an example of 'smart dust' a highly integrated power source, RF transceiver and sensor package as an example of what MEMS integration can achieve today.

Figure 9: 8inch diameter (300mm) silicon wafer

Figure 10: 'Golem dust' solar powered mote with accelerometer, light sensor and RF transceiver (UC Berkeley, BSAC)

Examining a spacecraft from the viewpoint of power source, energy storage, power distribution, and power regulation and control, it is assumed that power distribution, regulation, and control design can be met with wiring, switching, and regulation circuitry easily be placed on a wafer [9]. The real challenge lies in implementing an integrated power source and energy storage. Integrating solar power with digital circuitry is of increasing interest for "Smart Dust", shown in Figure 10 and designed as wireless sensor nodes for monitoring buildings, tracking goods and surveying remote sites. Since solar cell fabrication on CMOS is fairly straightforward, an efficiency of at least 5% has been assumed. A simple 2.2Ghz downlink, employing binary phase shift keying and turbo decoding allows 300bps to a 3.7m ∅ ground station, requires 16 mW on average assuming a 1% efficiency. This translates into a solar array area of 7 cm^2. A two-sided array design has to be considered at this point, to ensure solar illumination regardless of orientation. Provision of power to retain minimal system functionality during eclipse must still occur. Currently integrated capacitors are only reasonable way to store energy in CMOS, however analysis suggests that an on-chip capacitor would not have anywhere near the required capacity. Although confining operations to sunlit passes over the ground station, or reducing the power requirements by selecting a less power hungry payload might address this, expanding the design space to allow a commercially available thin-film rechargeable battery to be placed between wafers is attractive. Ultra capacitors, nuclear batteries and other power MEMS are other options to be explored during further research.

Additional challenges for CHIPSat include the need for propulsion to offset the drag from the very low ballistic coefficient of a high low aspect ratio wafer (stack in orbit), which will further increase power requirements if the SNAP and PalmSat experiences hold true. Better understanding of wafer defects in manufacturing and the required fault tolerance to allow wafer scale integration of several subsystems, with those that do not normally lend themselves to CMOS processing such as thin or thick film batteries is also needed.

5. CONCLUSIONS

This paper outlined the drivers for power generation and storage in small cost effective spacecraft which are being used more widely for a range of missions such as earth observation, low rate communications and technology testing. Two example highly miniaturised space missions developed by SSTL and the University of Surrey Space Centre, SNAP and PalmSat have been described. These have demonstrated the capability of spacecraft well below 10kg, but do not use MEMS and have not placed undue demands on power systems. However the future of ultra-small spacecraft, especially for exploration of the solar system may be CHIPSat, a concept being researched at SSC with support from SSTL. Provided silicon wafer scale integration of the majority of spacecraft subsystems proves possible, efficient thin film energy storage during eclipse is one of the main barriers to be overcome for CHIPSat.

6. ACKNOWLEDGEMENT

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