CubeSat: The Need for More Power to Realise Telecommunications

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1. Introduction

Conventional satellites are generally becoming larger, with increased capabilities and multifunctional roles [1]. However, this is costly both financially and in terms of time. Small satellites, weighing less than 500kg (see Table 1) come in at a much lower cost and lead time. This opens up many opportunities for space exploration to smaller and less specialised organisations. CubeSats are one type of satellite that fit into the pico- and nano-class of small satellites. CubeSats are only capable of performing one or two tasks as their size limits the amount of instrumentation that can be installed [2]. However, as they are relatively low in cost and quick to deploy, they allow more satellites to be deployed that can carry the latest technology and obtain results in a quicker time frame. One key limitation to CubeSats is the lack of available power, which can result in limiting the performance of other subsystems. One such subsystem is the communications system, which in turn restrict the opportunity to utilise CubeSats as an interlinked system, such as in a satellite telecommunications constellation.

2. Aims & Objectives

The overall aim of this report is to critically review the literature related to energy storage in CubeSats. This is divided into several smaller objectives; investigating the design criteria and applications of CubeSats (section 3), the current state of the art and requirements for energy storage devices on board small satellites (section 4) and a breakdown of the typical power demands that an energy storage system would be required to deliver, with a closer look at the power requirement of communication systems (section 5).

3. Overview of CubeSats

The CubeSat was devised in 1999 and first launched in 2003. It now accounts for 76% of all pico- and nano-satellites launched since 2003 [3].

A CubeSat is a small satellite that is based upon a standardised specification. The original CubeSat, known as 1U or 1 Unit, is a 10cm cube with a mass less than 1.33kg [4]. Larger CubeSat configurations are available, see Table 2, by effectively combining several 1U together. The most common sizes are 3U and 1U respectively, while sizes up to 27U are being developed to bridge the "gap between 3U CubeSats and traditional separation system restrained secondary payloads" [5].

The standardised specification allows the development cost and time to both be reduced. This is due to the fact that many of the components of CubeSats, including the battery energy storage system, are available as commercial off-the-shelf (COTS) components. Such components are available from various specialised and unspecialised suppliers (see [6]).

Table 1. Classification of satellites by mass and cost [1].

Type	Mass (kg)	Cost (US $)	Time of Development From Proposal to Launch (years)
Conventional large satellite	>1000	0.1-2 B	>5
Medium satellite	500-1000	50-100 M	4
Mini-satellite	100-500	10-50 M	3
Micro-satellite	10-100	2-10 M	~1
Nano-satellite	1-10	0.2-2 M	~1
Pico-satellite	<1	20-200 K	<1
Femto-satellite	<0.1	0.1-20 K	<1

Table 2. CubeSat specifications and Launch Statistics.

CubeSat Size	Dimensions (cm)	Max Mass (Kg)	№ Launched 2012-2015 (a)	
			№	%
1U (b)	10×10×10	1.33	28	29
1.5U (b)	10×10×15	2.00	24	25
2U (b)	10×10×20	2.66	5	5
3U (b)	10×10×30	4.00	37	39
6U (c)	12×24×36	12.00	2	2
12U (c)	23×24×36	24.00	n/a	n/a
27U (c)	35×35×36	54.00	n/a	n/a

Sources of information: (a) number of CubeSats launched (no launches detailed for 12U or 27U) [7], (b) specifications for 1U-3U [4], (c) specifications for 6U-27U [5].

As well as this, the fact that CubeSats are launched as an auxiliary payload on larger missions negates the high cost of a dedicated launch. CubeSats therefore open up access to space for organisations with smaller budgets, such as universities, private companies and developing nations.

3.1 Applications of CubeSats

CubeSats are used for many applications in the areas of scientific research, technological demonstration and commercial applications [8]. The capability of CubeSats in terms of scientific measurements and performance of communications systems are limited compared to large satellites, due to the restricted volume and power availability. However, they are popular due to their low cost and short mission lead times, which allows developers to put more satellites into space with the latest technology on board. Aside from this, a key application CubeSats can provide, that conventional satellites cannot due to cost, is to provide telecommunications for developing and emerging countries. In doing so, among other benefits, can help to improve education in said region. However, for a satellite communication constellation to be put in place developments in formation flying, communications and multi-CubeSat launch are required [8]. Understanding the limits of current energy storage on small satellites can help improvements in other subsystems, such as communications.

3.2 Orbit, Environment and Life Time

Nearly all pico- & nano-satellites are deployed into a Low Earth Orbit (LEO), which is defined has an altitude of 160km-2000km. In LEO they are most commonly (~50%) in a sun synchronous orbit, the rest being near circular or near polar, with some elliptical and low inclined orbits [3]. The period of a satellite in LEO is 90-105min with solar period lasting 2/3 of the orbit [8]. Temperature fluctuations can be as low as -40°C (eclipse period) to +85°C (solar period) [9]. The orbital characteristics and conditions of space play a key role in the management power systems and battery design, see section 4.1.

The target duration for pico- and nano-satellite missions vary from a few days to 5 years, with an average target duration of 8 months [3]. Life expectancies of CubeSat missions can be significantly longer than designed, e.g. YUsend-1 launched on 90 day mission but lasted at least 2 years [10]. The mission life fundamentally determines the battery life cycle requirement and in turn the sizing of the battery capacity, see section 4.1.

3.3 Design and Operation

Many of the COTS components used in CubeSats can withstand vibrations during launch; and the thermal and radiation environment of space with little to no modification [8]. However, the performance of batteries can be severely affected by temperature and the vacuum of space. Hence, these environmental effects have to be mitigated in some manner, see section 4.1.

The altitude of orbit plays a role in the necessity for radiation shielding and component choice. Bellow 3000km radiation levels are low and can be shielded by a few mm of aluminium, however at altitudes above this or for long duration missions the effects of radiation on components have to be considered [8]. As LEO is below 2000km it is the duration of CubeSat missions that is the main concern when considering radiation.

With respect to energy storage, suppliers of battery packs typically use cylindrical (18650) Li-ion cells [11]–[13] (2600mAhr @ 3.7V nominal [13]) or commercial polymer pouch cells (1300mAhr @ 3.8V) [14]. Purchasing a battery pack removes the need to test a batch of cells before selecting ones of similar internal resistance for better battery performance, as the pack manufacture does this.

3.4 Failures of CubeSats

Detailed information on failures of successfully launched missions is difficult to come by. However, in regards to mission failures relating to batteries, in the period from 2003 to date, only two mission failures can be attributed to battery faults. These are CAPE-1 due to an unknown battery fault [15], and AAU-CubeSat due to improper packing leading to battery pack expansion [16].

4. Energy Storage

Approximately 85% of all pico- & nano-satellites are equipped with solar panels in conjunction with rechargeable batteries [3]. The remainder, accounting for shorter missions lasting days to a few months, are powered purely by primary batteries [15].

The primary battery of choice in the last decade has been lithium batteries; while the rechargeable batteries of choice are currently Lithium-ion (Li-Ion) and Lithium-Polymer (Li-Po) [3]. Nickel Cadmium (NiCd) and Nickel Metal Hydride (NiMH) have had space flight heritage due to their robustness, however for applications in pico- and nano-satellites their energy density is too low [9], [17].

Energy storage on board CubeSats is typically Li-ion or Li-polymer batteries due to their availability as COTS component and their high specific energy density, a major requirement in small satellites where volume is severely limited. Other advantages of Lithium batteries over NiCd and NiMH are: low self-discharge rate, high cell voltage, no memory effect and long life cycles [18]. However, compared to other technologies such as Ni-Cd, Lithium batteries need to be managed more closely in regards to over dis/charge and temperature; and thus require more complex monitoring and protection circuitry. The characteristics of Li-Ion and Li-Po are very similar, however Li-Po have a slightly better energy density [17].

4.1 Design of an Energy Storage System

The construction of the cell also plays a role in efficient utilisation of the limited CubeSat volume. Cylindrical cells are lower cost than pouch cells, however pouch cells are advantageous due to the flexibility of cell packing and geometry allowing for better packing between other components [17].

As mentioned earlier, the characteristics of LEO have effects on the design of energy storage systems. The energy required of a satellite for an entire orbit has to be provided by the solar panels. The batteries have to be capable of storing a minimum of 1/3 of this energy so that it can be utilised during the eclipse period. However, to supplement high power demands (e.g. when communications are transmitting) during the solar period it is suggested that the batteries should be capable of providing a minimum of 70% of the orbit energy [17]. In LEO a satellite will orbit 14 to 16 times a day, or 5,000 to 6,000 times per year. As each orbit corresponds to one charge-discharge cycle, the length of the satellites missions directly influences the cycle life requirement of the battery. A fundamental characteristic of batteries is that you can increase the cycle life by reducing the depth of discharge (DOD) of the batteries. However for a given energy demand this requires the capacity of the battery to be increased, i.e. using more cells, hence taking up more volume. The maximum DOD is recommended to be 30-40% [17], [19], [20].

The harsh conditions of space have effects on battery performance. The low temperatures experienced in the eclipse period cause electrolytes to solidify and hence increase internal resistance [17]. Wang *et al* suggests operating at ambient temperature in the region of 10°C-30°C [21] for optimal performance. Hence temperature management systems will need to be used, however many suppliers of electrical power systems (EPS) for CubeSats incorporate this into the component. The vacuum of space can cause pouch cells to bulge leading to electrode layer separation, again increase internal resistance [22]. This is more prominent in liquid electrolyte pouch cells compared to Li-Po in which the polymer assists in holding the layers together [21]. However, pouch bulging can be easily mitigated by clamping cells between aluminium plates. Uno *et al* [22] state that vacuum conditions can also lead to electrolyte leakage from Li-Ion cells, in turn reducing the capacity of the cell. However, they go on further to state that cells "potted" in epoxy resin can prevent this leakage. Insufficient management of battery temperature; and inadequate mitigation of cell bulging and electrolyte leakage leads to increased rates of cell degradation and alterations of the voltage profile during charge and discharging [17].

4.2 Battery Testing

Many tests are undertaken in the characterisation of cells for satellite applications. Work by Navarathinam *et al* on the "characterization of Lithium-polymer batteries for CubeSat applications" demonstrates many of the necessary tests [17]. These include:

- *High discharge rate under standard temperature and pressure (STP)*: to simulate a worst case scenario at 1C discharge rate, 100% DOD. This is to evaluate cells under extreme conditions and to assess the capability of the cell to remedy itself later at lower rates (see next bullet point).
- *Normal discharge rate STP*: to assess the cells ability to recover lost capacity (0.5C).
- *Vacuum Testing*: to assess the effect of low pressure on the electrochemical and physical performance of the battery. Test undertaken at 0.5C discharge rate and pressure of 10^{-7} Torr.
- *Temperature cycling in a vacuum*: to simulate space environments. Battery charged and discharged under temperatures (charging @ 50°C, discharging @ -20°C and -10°C) simulating that to expected within the satellite structure. Charging/ discharging time to last 65/35min respectively to correspond with the Solar/Eclipse periods of LEO.
- *Life cycle testing*

Wang *et al* also demonstrate useful test including [20]:

- *Charge rate effects*: to evaluate cycle life performance due to rate of charge
- *Cycle-life test at different charge rates*: under LEO operating conditions at 40% DOD and ambient temperature
- *Cycle life testing with taper voltage effects*: varying charge and discharge rates to supress taper voltage
- Capacity verification and impedance measurements during cycle life testing.

4.3 Battery Safety

Battery safety is a key factor in any application. Neubauer *et al* [23] outlines factors that increase the likelihood of battery failure, the risks of Li-ion batteries and the precautions that should be undertaken to prevent battery failure. They state that an increased likelihood and severity of battery failure is due to the compounded effects of the satellite industry increasing energy levels on satellites, requiring batteries with greater energy density and reducing the time period for development. The risks of Li-ion batteries they outline are high current delivery and thermal runaway, while the use of COTS batteries presents a safety risk if they are not designed, handled or operated properly. Recommended precautions that should be undertaken include;

- *Cell protection devices:* positive temperature coefficient polyswitch, current interrupt device & vent, and shutdown separator;
- *Battery level protection strategies:* topology, cell size and thermal management; and
- *Operational protection strategies:* limiting voltage range, limiting charge/ discharge currents, limiting environment temperature, preventing short circuits.

They conclude by saying that thermal runaway of a battery in a small satellite (auxiliary payload) could lead to the destruction of the primary payload which would be hugely detrimental to the small satellite industry. Therefore the small satellite community must be as equally as thorough as conventional satellites developers to avoid safety incidents.

4.4 Novel Energy Storage Systems

Lithium-ion battery and super-capacitor hybrid energy storage systems for small satellite applications have been discussed by Chin *et al* [24], their findings are summarised here. They state that a hybrid system combines the high power of super-capacitors with the high energy density of Li-ion batteries. Super-capacitors, unlike li-ion batteries, perform well at low temperatures without degradation. This facilitates the hybrid system to have improved discharge capacity at lower temperatures. Chin *et al* also present a low temperature electrolyte solution that allows a battery capacity of 2.32Ah at 20°C compared to a COTS Li-Pol (from Clyde Space) which achieves 1.04Ah. Under test at -40°C and 1C discharge rate the low temperature electrolyte battery retains a capacity of 1.64Ah (compared to 0.20Ah for Li-Pol). Low temperature electrolyte Li-ion batteries used in conjunction with supercapacitors also increases the minimum cell voltage, at the end of a pulsed discharge, from 0.86V (COTS Li-Pol only) to 2.79V (hybrid). They state this is because of the super-capacitors "buffering the high loads and minimising battery voltage polarization which ultimately maintain excellent long term battery state-of-health". They also note, hybrid systems in other applications have also proven to increase "battery cycle life and power density under pulse load conditions with short duty cycles".

Raffeaelle *et al* presents a so called "integrated power supply" which incorporates thin film li-ion energy storage into the structure of photovoltaic power generation structure [25]. This could present an ideal solution to reducing the volume the energy storage system requires.

5. Power

5.1 Solar Power

Satellites generate all their necessary power from solar panels. In the case of CubeSats, their small size limits the available area for solar panels and in turn limits the available power. The available average power generated by body mounted solar panels is in the range of about 2W for 1U and 5-7W for 3U, with peak power approximately 9W (3U) [8], [26]. However with the increasing development of deployable and sun following solar panels the amount of available power is increasing. For example, for a 3U system with deployable solar panels, continuous average available power in the region of 12W-23W with peak power approximately 15W-40W has been demonstrated [26], [27]. While larger CubeSats have solar power arrays that are increasingly greater in surface area.

5.2 Power Budgets

A power budget details how much power is required by the satellite balanced over the power generated from the solar arrays per orbit, less loses. A positive power budget means that there is available energy to charge the batteries while a negative power budget means there is insufficient energy supplied from the solar panels to recharge the batteries to 100% in a single orbit.

YUsend-1 (1U CubeSat), developed in 2011, is a technology demonstration unit carrying three payloads; "a micropropulsion unit, high speed data rate communication system and a star camera" [17]. Its theoretical power budget, Table 4, shows that under any mode in which a payload is in operation the average power (2.2W-2.8W) is greater than that delivered by the solar panels (max 2W as mentioned previously for 1U CubeSat). Hence, the power budget for YUsend-1 is negative, to recharge it will have to spend a certain number of orbits in safe mode (average power 1.3W), and therefore the satellite will not be able to operate a payload continuously. The maximum power draw is not specified, but it can be seen from the table that it would be over 4.5W if either the microthruster or high data communications link operating, while the standard communications was inactive. This clearly cannot be provided by the solar panels alone and therefore has to be supplemented by the batteries.

DICE (1.5U CubeSat) was launched in 2011 "with the goal to map the geomagnetic SED (Storm Enhanced Density) plasma bulge and plume formations in Earth's ionosphere." [10].

Table 4. Sample power budget for YUsend-1 under different operational modes [17].

Sub-system	**Component**	**Power (mW)**	**Duty cycle (%) and energy use (mWh) per orbit**							
			Safe		**Camera**		**Thruster**		**High Data Comm.**	
			%	**mWh**	**%**	**mWh**	**%**	**mWh**	**%**	**mWh**
ADCS	Magnetorquer	350	50	288	80	461	80	461	80	461
	Magnetometer	5	10	1	50	4	50	4	50	4
	Reaction wheels	300	0	0	20	99	20	99	20	99
	Rate sensor	60	0	0	50	49	100	99	50	49
Comm.	Receiver	200	95	313	80	263	80	263	80	263
	Transmitter	3000	5	247	20	988	20	988	20	988
CPU	CPU	400	100	659	100	659	100	659	100	659
Power	PDU	40	100	66	100	66	100	66	100	66
Payload	Microthruster	4000	0	0	0	0	5	329	0	0
	Star Camera	250	0	0	10	41	0	0	0	0
	High Data Com	4000	0	0	0	0	0	0	10	659
		Sum (mWh)	n/a	1573	n/a	2630	n/a	2967	n/a	3247
		Sum inc. path efficiencies (mWh)	n/a	2200	n/a	3679	n/a	4151	n/a	4542
		Average power (mW)	n/a	1337	n/a	2235	n/a	2521	n/a	2759

ADCS – attitude determination and control system, Comm. – communications, OBC – on board computer, PDU – power distribution unit.

Table 3. DICE power budget (adapted from) [28].

Component	**Power (mW)**	**Duty Cycle (%)**	**Power (mW)**[1]
ADCS Card	160	100	194
PIC CPU	60	100	73
Comm Tx	9300	3	338
Comm Rx	80	100	97
Magnetometer	10	0	0
GPS	950	5	58
Torque Coils	750	0	0
Sun Sensor 1	25	100	30
EPS	25	100	345
Payload	300	-	242
Magnetometer	90	100	109
DC-Probe	40	100	48
E-Field	40	100	48
Motor Control	100	0	0
PL Controller[2]	30	100	36
Total (orbital average) Power			1375

[1]Power consumed during duty cycle including a 10% contingency and a 10% margin
[2]PL – Payload controller

DICE' battery pack utilised two Li-pol cells in series, supplied by Clyde Space. The power budget for DICE, Table 3, shows that it would require ~1.4W orbital average power (OAP). However it could generate an OAP of ~1.7W [29]. Hence, in this case there is a positive power budget and therefore DICE would be able to operate continuously without going into a safe mode to recharge. The maximum power draw here is over 9W, due to communications, again this could not be supplied by the solar panels alone.

From these two power budgets, it can be seen that a large percentage of the total power requirements is utilised by communications transmission. This will have a major effect on the power budget if the duration of communication increases (i.e duty cycle increases), for example in telecommunications. The relationship between communications and power is discussed in the following section.

5.3 Communication Power Requirements

The power demands of the components in YUsend-1 and DICE, Table 4 and Table 3 respectively, illustrate the large power demands of communications systems compared to the majority of all the other components. This is supported by the general statement that communications systems, when transmitting, consume approximately 50% or more of the total power demand in satellites [8]. This is a concern when it is also stated that the communications capabilities of pico- and nano-satellites are limited, not by radio technology, but by the lack of available power [3]. See Table 5 for a comparison between available data rates and power requirements.

Table 5. Commercially available transceivers for small satellites.

Transceiver Frequency Band	Bit rate, bit/s (max)	Electrical power, W	Transmitter Power, dBm	Source
VHF	9.6K	1.7	22	[32]
VHF	19.2K	16	30-38	[33]
UHF	9.6K	4	27	[34]
UHF	9.6K	10	27-33	[35]
UHF	19.2K	16	30-38	[36]
UHF	200K	4	36	[37]
S	100K	3.5	<28	[38]
S*	153K	1.5	-	[39]
S	1M	5	<27	[40]
S	2M	6	21-30	[41]
S	6M	26	30-34	[42]

Comment: All transceivers have dimensions that allow them to fit into a 1U CubeSat besides * which would have to be in a minimum of 2U.

If we take, for example, broadband multimedia satellites in LEO we see that they are subject to "bursty and self-similar" traffic in which "burst[s] appear in scales ranging from a few milliseconds to minuets and hours" [30]. The traffic throughput of a satellite system can be defined as "the rate at which the bits are actually transmitted" [31], which is related to power and network load [30]. Hence, the traffic pattern affects how much power is required by the communications system at any given time.

We can assume that this bursty behaviour would be present for small satellites acting as (low data rate) telecommunications. Hence the work by Chin et al (see section 4.4) using a hybrid system incorporating super capacitors to deal the high power demands presents serious merit for consideration in such applications.

As the traffic pattern is coupled with power demands, an unpredictable bursty traffic pattern makes it difficult to predict the orbit cycle power demand. In doing so it makes it difficult to predict the charging/ discharging regime as the batteries "are subject to a lot of pulse discharge[s]" [30]. The energy per bit is dependent on many aspects within the circuit design of the transceiver as well as the operating and management of the transceiver [43]. The energy requirement of communications is therefore proportional to the number of bits processed in a given period of time [30]. Hence the amount of traffic has a role in the DOD of the battery, which in turn affects the battery life.

For a CubeSat undertaking constant bursty communications, it would be thought that the duty cycle percentage for communication transmission will be greater than that of the 1-5% in scientific research and technological demonstration CubeSats. Hence the OAP and energy demands will be greater. As we have already seen, the power budgets are already tightly constrained and hence present a barrier to the realisation of CubeSat based telecommunications satellite constellations.

6. Conclusion

The small size and restricted geometry of CubeSats pose strict constraints on the subsystem components, including the energy power supply (EPS) system, to be installed in the satellite. The satellites application, mission life and orbit characteristics are the key aspects in determining the EPS system requirements, including the average orbital energy, the peak power demands and the cycle life expectancy. The need for energy in such a restricted volume has led to Lithium rechargeable batteries been the most common battery used on CubeSats. However, as CubeSats are being utilised for more advanced applications their energy requirements are increasing. While systems such as communications have high peak powers and burtsy characteristics putting large strain on the EPS. These increasing power demands are leading to the need for new developments in energy storage for CubeSats, including the use of battery-supercapacitor hybrid systems and integrating energy storage into other components.

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