Satellite multi-functional power structure: feasibility and mass savings

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Abstract: A multi-functional structure saves mass from a spacecraft by incorporating other functional subsystems into the structure. By using the structural properties of a non-structural element, inert structure may be eliminated, and the requirement to allot internal volume to the subsystem in question is removed.

The current paper describes a multi-functional structure based on the secondary power system. By using commercially available plastic lithium-ion cells to form the core of a sandwich panel, inert mass is eliminated from both the structure and from the battery enclosure. The feasibility of the proposed multi-functional structure is demonstrated though vibration testing on a single cell, and the successful manufacture of a test panel.

The work goes on to quantify the potential mass savings that may be achieved by using a multi-functional structure of this type. By varying a set of spacecraft attributes, the study identifies that small spacecraft with high power requirements have the potential to gain the most benefit from using a multi-functional structure of this type.

Keywords: multi-functional structures, spacecraft structures, spacecraft power systems

1 INTRODUCTION

1.1 The spacecraft secondary power system

The vast majority of Earth-orbiting spacecraft use a primary power system based on solar photovoltaic cells, and a secondary power system based on an electrical battery to store energy for use during eclipse periods and peak power demands. A large power requirement leads to a large energy storage requirement, and hence a large power storage subsystem. The secondary power subsystem of a modern satellite may contribute significantly to the mass, and often to the volume, of the bus. Given that there is a roughly proportionate relationship between the mass of a spacecraft and its cost, notably the cost of its launch [**1**], this mass will contribute significantly to the cost of the mission.

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Using alternative battery chemistries, most notably lithium-ion [**2–4**], can help to reduce this mass. Lithium-based batteries have far superior energy and power per unit mass than traditional battery technologies used in spacecraft [**5**]. New battery types, however, not only provide better performance, but also offer potentially revolutionary changes in packaging configurations. The active elements of lithium polymer and plastic lithium-ion (PLI) batteries are composed entirely of solid materials. The mass of each cell is reduced, since the packaging of the cell needs to do no more than act as an oxygen barrier, as opposed to acting as a container as in a liquid-state cell. In addition, PLI cells are produced in a prismatic shape, making them easier to pack into electronic devices than cylindrical cells. Prismatic PLI cells are commonly used in applications where mass and volume are critical, such as mobile telephones.

Commercially available cells are often a viable alternative to aerospace-grade cells, particularly in low-cost microsatellite missions where the slight reduction in performance may be justified by a large reduction in cost. Commercial PLI cells have excellent electrical performance and have been space-qualified for the vibration, vacuum, radiation, and thermal environments [2]. The robustness of this cell configuration, along with its prismatic shape, makes packaging for a battery of such cells much simpler than for the cells currently used in satellites. Small PLI cells could be secured within a microsatellite simply by affixing them to available surfaces using aluminium tape or another adhesive, as suggested in reference [2]. This eliminates the separate battery enclosure, though it relies on there being adequate spare volume and areas to affix the cells within the spacecraft.

1.2 The multi-functional structure

An alternative to seeking free volume within the bus is to incorporate the cells into a multi-functional structure [6]. By using solid-state lithium cells to replace the inert core materials (usually aluminium honeycomb), the mass of the battery enclosure may be eliminated and the structural properties of the cells themselves may be exploited to become part of the structure. The structural properties of PLI cells have previously been utilized in unmanned aerial vehicles [7]. The mass density of the cells will, naturally, be higher than that of the core material they will replace, but they are a necessary part of the satellite and thus their mass is not removable. The core material, on the other hand, performs no function other than structural support and may thus be substituted for electrical cells without reducing the spacecraft's functionality, provided that the cells can perform the mechanical function of the core.

Distributing the battery of the spacecraft within its structure saves mass by replacing inert structural materials with functional ones, while simultaneously removing the parasitic mass of the battery packaging. In addition, the volume of the bus is reduced, as the battery now occupies volume that would otherwise be unused. The concept of the multi-functional structure, applied to a carbon fibre/aluminium honeycomb panel, is shown in Fig. 1.

The current paper shall describe work undertaken to demonstrate the feasibility and potential advantages of using Varta PoLiFlex PLI cells as part of the core of a carbon fibre sandwich panels. A cell has been tested to confirm its ability to survive the vibration of a spacecraft launch, and a trial panel has been produced to assess the feasibility of the manufacturing process. In addition, a study has been undertaken to identify spacecraft missions that could potentially benefit from the mass savings afforded by multi-functional structure technology.

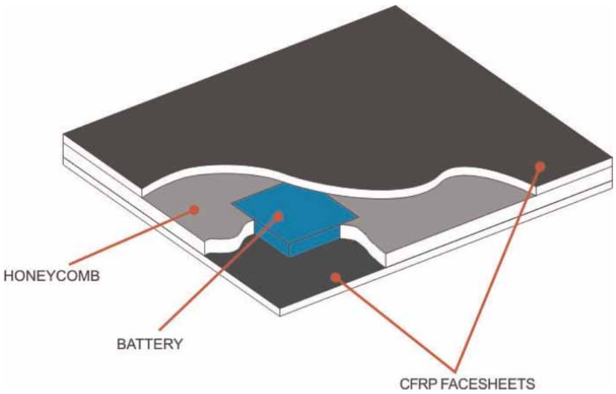

Fig. 1 Multi-functional structure concept

2 ELECTRICAL PERFORMANCE AND VIBRATION

Testing has been undertaken at the University of Southampton to assess a cell's ability to operate after intense random vibrations [8]. Varta PoLiFlex PLI cells (as shown in Fig. 2) have been subjected to random vibration testing. Between each vibration test, an electrical characterization test was conducted using a simple discharge logger to establish if the vibration had affected performance. Typically, the random vibration qualification test for a spacecraft application consists of a single high intensity random vibration test in each of the three principal axes, but repeated tests were used in this case in order to differentiate between performance loss due to aging and any losses due to vibration.

2.1 Electrical testing

The purpose of the testing was to establish whether any notable changes in the performance had occurred. As such, a fairly simple electrical testing system was used to characterize the cell. The cell was discharged at 100 mA until 25 per cent of its nominal charge capacity (270 mAh) had passed through the circuit, the current and voltage being logged using the circuit shown in Fig. 3 throughout

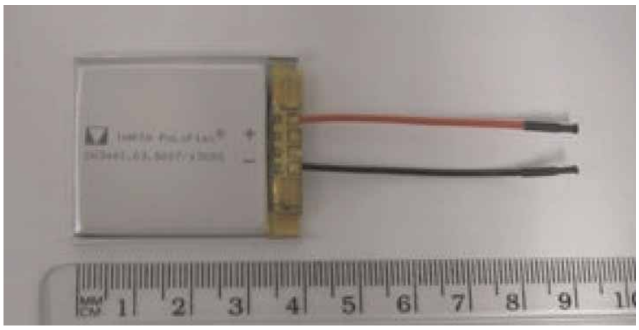

Fig. 2 Varta PoLiFlex 263441 PLI cell

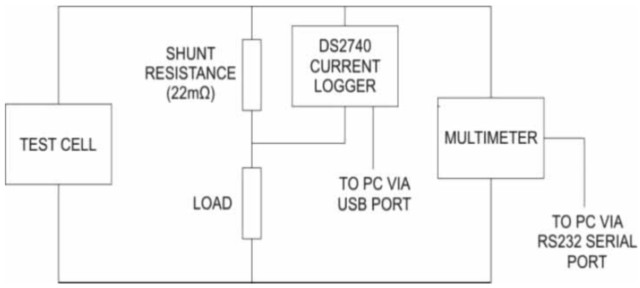

Fig. 3 Discharge logging circuit

this discharge. The Dallas Semiconductor DS2740 integrated circuit [**9**] measures current and integrates it to calculate the total accumulated charge. The voltmeter was an Extech MT330 multimeter. The voltage, current, and accumulated charge were sampled every 10 s. After discharge, the cell voltage was monitored for a further 10 min, during which the cell was left to return to its equilibrium state. Finally, the cell was recharged using a constant current-constant voltage profile (100 mA current until the cell voltage reached 4.2 V, then a constant charging voltage of 4.2 V).

In order to obtain a simple measure of any changes in the cell's performance before and after the vibration test, two simple performance indices were chosen. Reduction in the end of discharge (EOD) voltage (V_{EOD}, the terminal voltage of the cell after a discharge to a fixed depth of discharge (DOD)) is representative of reduction of the cell's capacity. An increase in the internal resistance (R_{INT}) of the cell is representative of a reduction in the cell's ability to deliver high currents, that is, a reduction in the maximum power it can produce. These two indices were used since they are relatively straightforward to measure, and because they are representative of the two most important specifications a cell must satisfy.

2.2 Vibration testing

The purpose of this testing was to investigate the effects of pure acceleration on the electrical performance of the cell, rather than allowing it to respond structurally, as this is the first step to assess the effect of the environment that the cell would encounter within a sandwich panel. It was necessary, therefore, to vibrate the cell in such a way that it did not undergo any external macroscopic deformation. The cell was clamped into an aluminium jig while being shaken, as shown in Fig. 4, which was then bolted to the top plate of an LDS electrodynamic shaker. The cell was then subjected to the 15 g_{rms} random vibration environment, described in Fig. 5, for 4 min in each of the three axes. The voltage was monitored throughout the test to ensure that there were no changes in the charge level of the cell due to the vibration.

A 15 g_{rms} test was conducted after every two complete electrical charge/discharge cycles, for a total of six vibration tests and thus 14 electrical cycles. In addition, a single test was conducted to investigate the effect of more intense vibration on the test subject. This test consisted of four complete charge–discharge cycles, with a vibration test between the second and third cycles. The vibration was of a similar profile to the repeated testing, but with a total acceleration of 25 g_{rms} as shown in Fig. 5.

2.3 Results and analysis

After the cell was cycled and vibrated as described, the voltage and current profile of the discharge were recorded and the relevant data extracted. The results of the experiments are shown in Figs 6 and 7. Each point on the graphs indicates a complete cycle of discharging and charging, and vibration tests are indicated by a dashed line and an arrow.

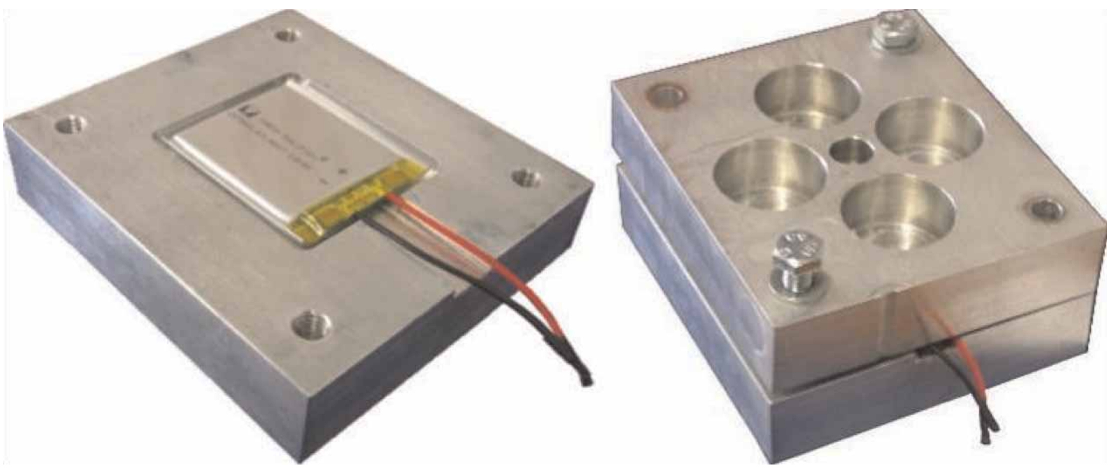

Fig. 4 The shaking jig

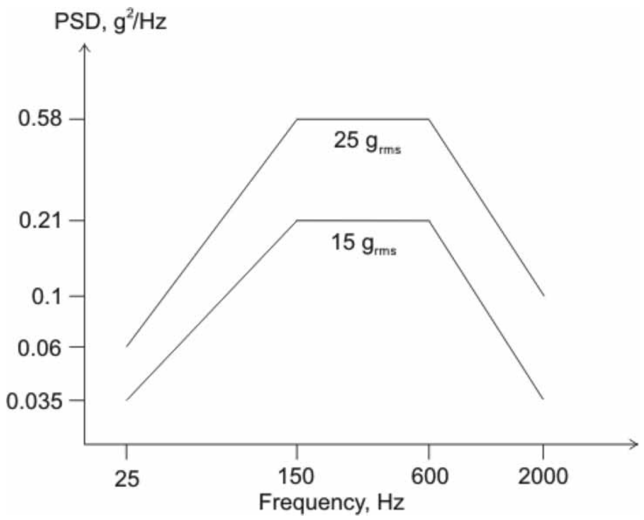

Fig. 5 Random vibration profile

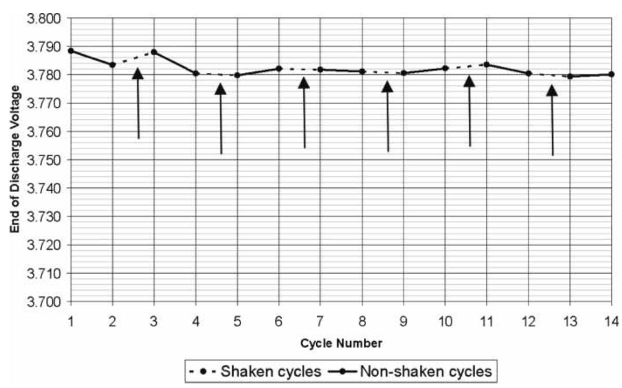

Fig. 6 Change in V_{EOD}

As was expected, V_{EOD} decreased and R_{INT} increased with repeated cycling, although there was considerable scatter in the results. However, is no notable difference in the loss in performance between cycles that included a vibration test and those that did not, for either V_{EOD} or R_{INT}. The cell still did not show any discernable loss in performance after the 25 g_{rms} random vibration, which is a good indication of the cell's robustness to vibration. If mechanical vibrations have any effect on the performance of the cell, then the effects are less than would result from a single electrical cycle, and thus do not result in any appreciable reduction in the cell's lifetime.

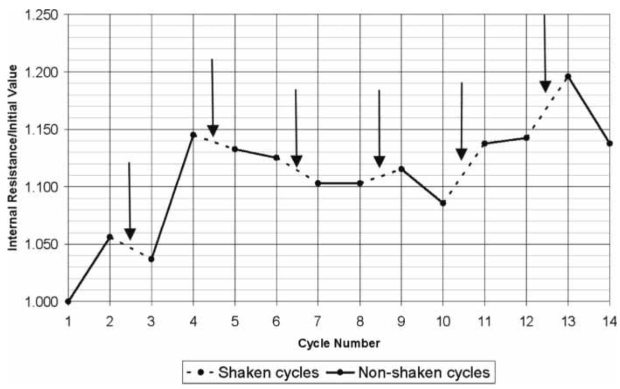

Fig. 7 Change in R_{INT} relative to the initial value

3 TRIAL PANEL

3.1 Design and manufacture

To produce a sandwich panel incorporating electrical cells as described using conventional means, the cells in question must be subjected to the heat and vacuum associated with the curing of epoxy resins. To assess the effect of this environment on the cells, a carbon fibre sandwich panel using both aluminium honeycomb and PoLiFlex cells as its core was manufactured. The panel was sized to fit as the lid of a Surrey Satellites Technology Ltd microtray and designed to have a natural frequency of around 150 Hz. These constraints were applied to give the panel a realistic design, but since the panel was intended merely to demonstrate the feasibility of the manufacturing process, it did not undergo any detailed design or optimization. The facesheets of the panel were made from 2 ply 600 gm^{-2} woven carbon fabric prepreg in a quasi-isotropic lay-up, and were cured using an oven and vacuum bag. The material used for the remainder of the core was aluminium honeycomb with a thickness of 2.6 mm.

Ordinarily, it is necessary to cure the resin that bonds the facesheets and core at an elevated temperature. The PoLiFlex cells which form part of the core, however, cannot be heated above 60 °C, which greatly limits the selection of resin systems that may be used. The selected resin film was Amber Composites' EF44 [**10**], which can be cured at temperatures as low as 50 °C. The sandwich panel was assembled using standard methods, with gaps being cut into the core to accommodate the cells, and cured for 15 h at 50 °C.

3.2 Effect on electrical performance

While the manufacturer's data indicated that the battery cells could survive temperatures higher than those encountered during curing, it was uncertain whether or not the lengthy period of high temperature exposure, coupled with the vacuum applied during the cure, would affect their performance. To investigate this, four of the eight cells were electrically characterized before manufacture, and then again afterwards. This testing showed a slight increase in R_{INT} and a slight decrease in V_{EOD}; however, during the period between the two tests, a similar degradation was observed in a control specimen. This effect is likely, therefore, to be due to aging or self-discharge.

This testing demonstrates the feasibility of the manufacturing process and may also be viewed as a basic qualification test for the thermal and vacuum environments in outer space. A temperature of 50 °C is as high as any spacecraft component will be likely to encounter, and the vacuum bagging procedure exposes the cell to pressures considerably lower than atmospheric, though not hard vacuum.

3.3 Alternative manufacturing method

Although the manufacturing qualification test indicates that the cells are able to survive the resin curing environment, it is nevertheless possible that a cell could fail at some point while being incorporated into the panel. Since even a single failed cell would probably result in an entire panel being discarded, it is desirable to delay irrevocably bonding the cells into the panel until as late as possible in the manufacturing process. To allow this, the cells could be cold-bonded into to the panel after the core had been cured at high temperature, although this would result in slightly inferior mechanical properties and would restrict the placement of the cells in the panel.

4 MASS SAVINGS

Qualitatively, the benefits of using a multi-functional structure have been described earlier: a saving in mass results from eliminating the mass of the inert battery components, principally the battery enclosure, and the volume of the battery pack from the bus. However, these benefits will necessarily be offset by the costs associated with designing a more integrated spacecraft, and the need for qualification of new technology. It is thus necessary to quantify the mass savings achievable in order that a trade-off may be made.

This section shall assess the mass savings that may be achieved by using a multi-functional structure for various spacecraft missions, which are described by a series of performance parameters. A range of values are used throughout in order to address spacecraft with various attributes and to indicate how effective at eliminating inert mass the multi-functional structure design is.

4.1 Spacecraft parameters

Several important parameters have been identified that affect the mass that may be saved using a multi-functional structure. The parameters considered in the current study are as follows.

4.1.1 Parasitic mass fraction

Defined as the mass of the inert parts of the battery enclosure divided by total battery mass, this parameter relates the mass of the active battery elements (i.e. the cells) to the mass of any inert components that support the battery. Since the principal aim of using a multi-functional structure is to eliminate this parasitic mass, it is important to ascertain how much mass may be saved by this means. If the structural properties of the cells are harnessed, it is possible to eliminate mass from the structure, which would effectively make this parameter negative. Values between the maximum and minimum indicate that inert mass has been eliminated but that the structural properties of the battery cells have not been fully harnessed.

$$\eta_{\text{para}} = \frac{M_{\text{para}}}{M_{\text{batt}}} \tag{1}$$

4.1.2 Specific energy capacity (SEC)

This is a fundamental property of the type of cell chemistry used in the spacecraft, defined as the nominal energy capacity of a cell (its nominal voltage multiplied by its nominal Ah capacity) divided by the mass of a bare cell. Recent advances in battery technology mean that this number varies enormously, from 40 Whkg^{-1} for a nickel–cadmium cell to well over 200 Whkg^{-1} for the latest lithium-based chemistry. Variation of this parameter allows a comparison of the mass saving achieved by using a different type of cell and that achieved by using a multi-functional power structure

$$\text{SEC}_{\text{cell}} = \frac{C_{\text{nom}} V_{\text{nom}}}{M_{\text{cell}}} \tag{2}$$

4.1.3 Specific energy requirement (SER)

The SER of a satellite is defined as the total energy storage requirement (i.e. the total energy capacity of the battery) at beginning of life divided by the launch mass of the satellite. Relating this parameter to SEC_{cell} allows the mass of the battery to be calculated as a function of the mass of the satellite, and hence, using η_{para}, the saving in parasitic mass may be calculated

$$\text{SER}_{\text{sat}} = \frac{E_{\text{sat}}}{M_{\text{sat}}} \tag{3}$$

4.1.4 Structural mass density

This parameter allows the volume reduction achieved by removing the battery from the spacecraft bus to be translated into a saving in structural mass. It is defined as the mass of the satellite structure divided by the volume of the bus (in stowed configuration if appropriate)

$$\delta_{vol} = \frac{M_{stru}}{V_{bus}} \tag{4}$$

4.1.5 *Packing efficiency*

This quantity represents the fact that the removal of the battery effectively eliminates more volume than that of the cells themselves. In addition to the inert elements of the enclosure itself, a spacecraft configuration typically wastes a large amount of internal volume, due to the need for harness clearance, balancing, safety margins and the fact that it is virtually impossible to fit a series of items with mounting feet and irregular protuberances into a single space without leaving gaps. The removal of the battery, therefore, will also eliminate a corresponding amount of this wasted volume.

$$\eta_{pack} = \frac{V_{cells}}{V_{batt}} \tag{5}$$

4.2 Parameter values

Table 1 shows the ranges of values used for the parameters in the study.

The range for η_{para} is based on historical data from previous satellite missions. The values of this parameter for conventional battery enclosures vary from 0.2 to 0.25, and so 0.25 is chosen as the highest value. The lower limit is a theoretical value for an ideal multi-functional power structure, where part of the structure is replaced by the battery – hence, the mass of the structure itself is reduced and this is indicated by the parasitic mass term becoming negative. A value of 0.05 is selected as a minimum since this would represent the cells (mass density of around 2000 kgm^{-3}) replacing aluminium honeycomb (mass density of around 100 kgm^{-3}).

The minimum value of SEC_{cell} is based on data for cells currently in use in space applications: The earlier lithium-ion cells like the Sony 18650HC used by ABSL Power Solutions Ltd for spacecraft battery packs, with an SEC of around 130 $Whkg^{-1}$ [**11**]. Since this type of cell is commonly used in space applications, while being part of the new generation of lithium batteries, it is used as the baseline cell. The upper limit is based on the performance of the latest PLI cells (such as the Varta PoLiFlex [**12**]).

Table 1 Parameter values

Parameter	Units	Limits: Lower	Limits: Upper
η_{para}	None	−0.05	0.25
SEC_{cell}	$Whkg^{-1}$	120	220
SER_{sat}	$Whkg^{-1}$	<0.5	>5
δ_{vol}	kgm^{-3}	0	350
η_{pack}	None	0.5 throughout – not varied	

The range of SER_{sat} is based once more on historical data. There is no real lower limit on this quantity; in the case of geostationary observation satellites, for example, the few eclipses and low eclipse power requirement mean that SER_{sat} may be less than 0.5 $Whkg^{-1}$ (for example, Meteosat 5 has a battery capacity of 270 Wh, and a launch mass of 681 kg – giving an SER_{sat} of less than 0.4 $Whkg^{-1}$). Spacecraft that operate in low-earth orbit (LEO) require larger battery capacities in order to account for capacity fade due to repeated cycling, and hence have a higher SEC for a given eclipse power requirement. However, most LEO spacecraft are observational, and so have comparatively small power requirements in eclipse (for example, ERS-1 has a 2650 Wh battery and a launch mass of 2150 kg – leading to an SER_{sat} of 1.2 $Whkg^{-1}$). The highest SER_{sat} occurs for satellites which have large power requirements which continue during eclipse – most notably geostationary communication satellites. Such craft have lifetimes of up to 15 years, which limits their batteries' DOD to 40–60 per cent, even in geostationary earth orbit (GEO), and extremely high power requirements. GEO communication satellites have SER values of as much as 5 $Whkg^{-1}$.

δ_{vol} is also based on data from previous satellite missions. For large spacecraft (1–2 ton and over), this term is very small – typically less than 30 kgm^{-3}. However, as spacecraft mass decreases, δ_{vol} increases sharply, exceeding 150 kgm^{-3} for spacecraft under 100 kg. For very small spacecraft (of the nanosat or 'cubesat' type, with masses down to around 10 kg), the parameter can even exceed 500 kgm^{-3}. The current study shall consider structural mass densities from 0 up to 300 kgm^{-3}. The lower end of this range represents the largest spacecraft, while the lowest values equate to satellites with masses of the order of 50 kg.

η_{pack} is not easily quantified. Since it acts, to all intents and purposes, as a coefficient to the δ_{vol} term, it is not varied. A value of 0.5 is used throughout the study.

4.3 Methodology

The principle of this study is to calculate the mass savings that may be achieved for various values of the parameters listed in Table 1. To this end, a baseline spacecraft design shall be considered with given values of SER_{sat} and δ_{vol}. The baseline values for

SEC_{cell} and η_{para} represent those of the most typical battery designs in use (120 $Whkg^{-1}$ and 0.25, respectively). The mass saved purely by using a different cell type, for values of SEC_{cell} up to 220 $Whkg^{-1}$, shall be plotted, as shall the mass saving achievable through using a multi-functional structure – the saving due to both parasitic mass elimination and bus volume reduction being indicated. All mass savings shall be calculated relative to the total spacecraft mass at launch.

The subscript '0' indicates the current baseline value for the parameter in question throughout.

The reduction in battery mass achieved through using an alternative cell chemistry in a conventional battery pack (i.e. through varying SEC_{cell}) is calculated as follows

$$\frac{\Delta M_{batt}}{M_{satt}} = \left(\frac{SER_{sat,0}}{SEC_{cell}} - \frac{SER_{sat,0}}{SEC_{cell,0}}\right)\left(1 + \eta_{para}\right) \tag{6}$$

A lighter battery pack will also have a smaller volume, and hence the structural mass may be reduced. Since the battery's mass is calculated in terms of the spacecraft mass, this must be converted into a volume using the mass density of the cell, ρ_{cell}. The value of ρ_{cell} varies slightly for different cell types, and so to take account of this, the relationship within the range studied herein is approximated as follows, based on data for several different cells (though this relationship cannot be applied outside the range of SEC_{cell} values considered in Table 1)

$$\rho_{cell} = 3400 - 7.5 \times SEC_{cell} \tag{7}$$

The total mass saving achieved purely through changing SEC_{cell} is thus

$$\frac{\Delta M_{SEC}}{M_{satt}} = \frac{SER_{sat,0}}{SEC_{cell,0}}\left(1 + \eta_{para,0} + \frac{\delta_{vol}}{\rho_{cell,0}\,\eta_{pack,0}}\right) - \frac{SER_{sat,0}}{SEC_{cell}}\left(1 + \eta_{para,0} + \frac{\delta_{vol}}{\rho_{cell}\,\eta_{pack,0}}\right) \tag{8}$$

If the battery pack is redesigned or a multi-functional structure is used, additional mass is saved through the reduction of the battery's parasitic mass as follows

$$\frac{\Delta M_{para}}{M_{satt}} = \frac{SER_{sat,0}}{SEC_{cell}}\left(\eta_{para,0} - \eta_{para}\right) \tag{9}$$

Finally, the mass saving which results from removing the battery from the bus, and hence its volume is calculated

$$\frac{\Delta M_{vol}}{M_{satt}} = \frac{SEC_{sat,0}}{SEC_{cell}}\,\frac{\delta_{vol}}{\rho_{cell}\,\eta_{pack,0}} \tag{10}$$

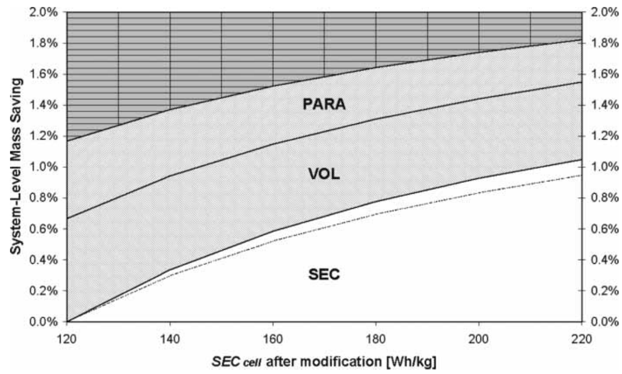

Fig. 8 Example graph showing mass saved versus SEC, using baseline values for SER_{sat} of 2 $Whkg^{-1}$, δ_{vol} of 100 kgm^{-3}, and η_{pack} of 0.5

Figure 8 shows a chart of the mass savings for a spacecraft with an initial η_{para} of 0.25, an $SEC_{cell,0}$ of 120 $Whkg^{-1}$ and an $SER_{sat,0}$ of 2 $Whkg^{-1}$. The mass saved by changing the cell chemistry (ΔM_{SEC}), eliminating parasitic mass and harnessing the structural properties of the cells (ΔM_{para}) and reducing structural volume (ΔM_{vol}) at each value of SEC_{cell} are indicated separately. A dashed line divides the ΔM_{SEC} saving into two areas: The lower area indicates mass saved by reducing the mass of the cells themselves, while the smaller upper area indicates the mass saved by reducing the volume of the bus.

4.4 Analysis

4.4.1 Variation of SER

It is evident without extensive calculation that all three mass saving terms are directly proportional to SER_{sat}. Hence, holding all other parameters constant, the size of the mass saving varies linearly with SER_{sat}, meaning that a spacecraft with a larger energy storage requirement benefits from a larger mass saving through the use of a multi-functional structure. Figure 9 shows how much mass may be saved for different values of SER_{sat}. The upper area represents the absolute maximum value of the mass saving that may be made by modifying the secondary power system, increasing the SEC from 120 to 220 $Whkg^{-1}$ and eliminating the parasitic mass (i.e. modifying η_{para} from 0.25 to -0.05) and volume of the battery pack (the area is used to show the variation of δ_{vol} from 0 to 350 kgm^{-3}); the heavy line indicates the saving that may be made by increasing the

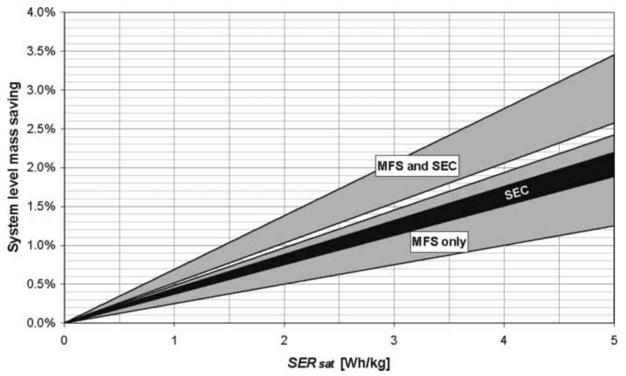

Fig. 9 Maximum achievable mass savings versus SER

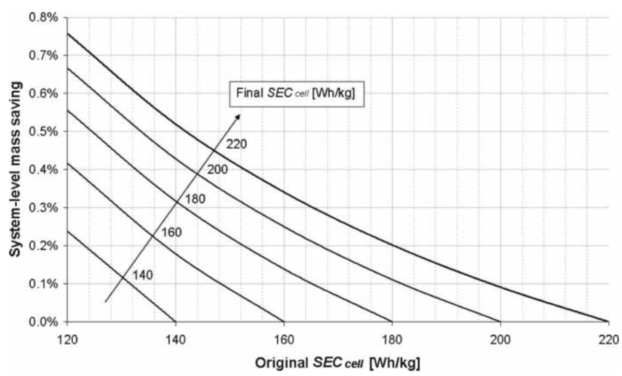

Fig. 10 Available mass saving from chemistry change

SEC but using a conventional battery pack; the lower area indicates the saving made by using a multi-functional structure alone.

As an overview, this chart indicates the following.

1. The savings made by using a multi-functional structure are roughly equivalent to changing the cell type from the more conventional cylindrical lithium-ion cells (depending on the value of δ_{vol}) to a state-of-the-art PLI cell.
2. For spacecraft with an SER of 1.5 or less, the maximum possible mass saving is around 1 per cent or less, which is probably not sufficient to warrant the investment that would be required to achieve this saving.
3. For spacecraft with an SER of 4 or higher, even the minimum mass saving is over 1 per cent.

Since the variation in mass saving with SER is linear, from here onwards it shall be fixed at a value of 2 Whkg^{-1}.

4.4.2 Mass savings from cell chemistry

Normally, when seeking to reduce the mass of a battery (assuming that the energy storage requirement is fixed) the simplest method is to seek a cell chemistry with a higher SEC. Assuming that commercially available cells are to be used, the cell chemistry is limited to an SEC of around 220 Whkg^{-1}.

Figure 10 shows how much mass may be saved by changing the cell type. From this chart the following may be concluded.

1. Increasing SEC results in diminishing returns – the reduction in mass made by increasing SEC by 40 Whkg^{-1} is more than half that made by increasing SEC by 100 Whkg^{-1}.
2. The obvious convulsion that, if the cell used already has one of the highest available SECs, then little to no mass may be saved by changing the cell type due to the fact that there are no available battery types with significantly higher performance.

4.4.3 Mass savings from parasitic mass removal

This section shall compare the savings made through the changing cell types with those made by eliminating the parasitic mass of the battery enclosure. This is representative of situations where δ_{vol} takes a small value (i.e. for spacecraft with masses of around 1 ton or over, where δ_{vol} is generally around 25 kgm^{-3}), or if the spacecraft volume cannot be reduced (for example, if the external panels are used for body-mounted solar arrays), meaning that the structural volume terms in equations (8) and (10) may be neglected.

Figure 11 shows the system level mass savings available through eliminating the parasitic mass of the battery pack (with the baseline η_{para} assumed to be 0.25) and compares this with the mass reduction achievable through increasing SEC_{cell} from 120 to the maximum value 220 Whkg^{-1}.

It is notable that, where SEC_{cell} is above 155 Whkg^{-1}, the saving that can be made by eliminating parasitic mass exceeds that deriving from changing the cell type (assuming once more that 220 Whkg^{-1} is the limit to the parameter). Even below this value, the saving made by eliminating

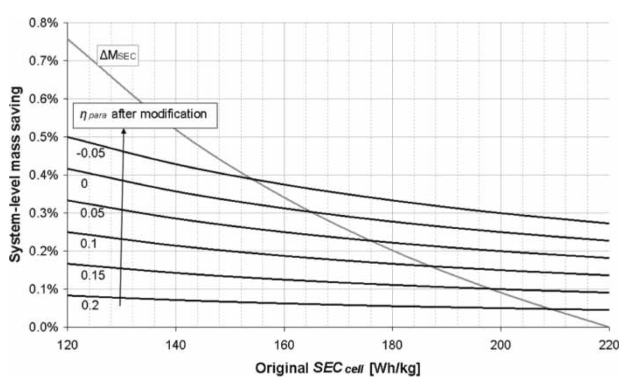

Fig. 11 Comparison of ΔM_{SEC} with ΔM_{para} for various values of η_{para}

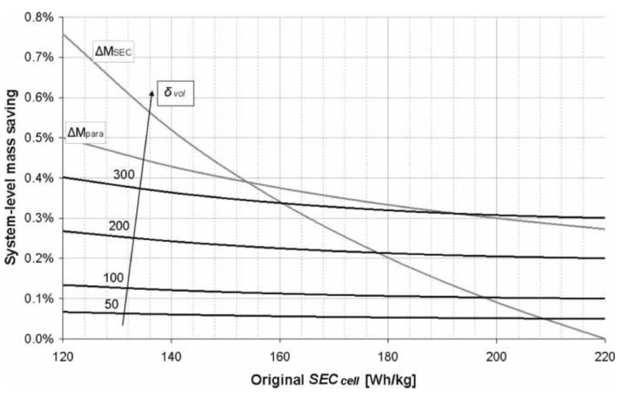

Fig. 12 Mass savings due to volume reduction versus SEC_{cell}, for various values of δ_{vol}

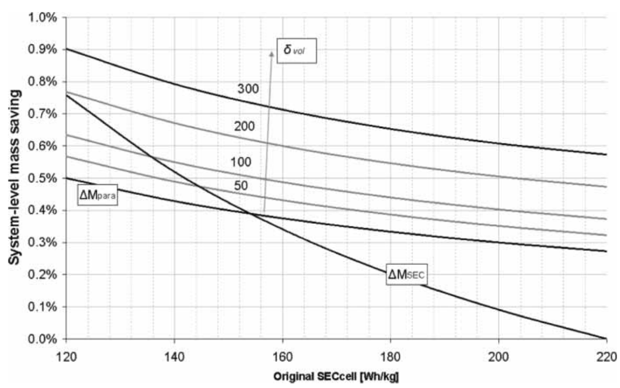

Fig. 13 Combined savings from parasitic mass and volume reduction versus SEC

parasitic mass is significant – 1.25 per cent for a spacecraft with an SER of 5 $Whkg^{-1}$.

4.4.4 Mass savings from volume reduction

For spacecraft with a high δ_{vol}, the third mass reduction element, that based on structural mass reduction, must be considered. The ΔM_{vol} term is highly variable according to the value of the δ_{vol} parameter, and varies slightly with the SEC_{cell} parameter. Figure 12 shows how much mass may be saved by eliminating the volume of the battery compared with the savings available from modifying SEC_{cell} (once more, from 120 to 220 $Whkg^{-1}$) and eliminating parasitic mass (baseline η_{para} of 0.25, final η_{para} of -0.05), for various values of δ_{vol}.

The highest value of δ_{vol} results in a similar mass saving to that achieved from eliminating parasitic mass and, when SEC_{cell} exceeds around 160 $Whkg^{-1}$ the potential saving exceeds that which may be achieved by changing the cell type.

4.4.5 Combined savings

Figure 13 shows the combination of the mass savings that arise from parasitic mass elimination (baseline η_{para} of 0.25, final η_{para} of -0.05) and volume reduction, compared with the available mass saving from changing SEC_{cell}. For medium values of δ_{vol} (around 200 kgm^{-3}), the saving achievable through using a multi-functional structure exceeds the available mass saving from changing SEC throughout.

4.5 Summary of results

Three particular qualities identify spacecraft which may benefit from the use of a multi-functional power structure: high energy storage requirement, high structural mass per unit volume, and high cell specific energy. It is a natural conclusion that a spacecraft, with a higher energy requirement will have a heavier battery, and hence will benefit from reduction in battery mass by any means. This correlates to spacecraft which have high power requirements in eclipse and/or very long lifetime requirements (a longer lifetime means that less of the battery's capacity is actually used, to allow for capacity fade).

For minisatellites (spacecraft with a mass below around 500 kg), the structural mass density becomes higher and so the amount of mass that may be saved by reducing volume increases dramatically. In some cases, the saving in structural mass may exceed the saving made by eliminating the parasitic mass of the battery. In this case, significant savings in system level mass may be made even if the multi-functional structure is not structurally optimized, due to the large savings from volume reduction.

Finally, where a spacecraft already employs a high SEC_{cell} cell type, the multi-functional structure becomes more useful. Such a battery is comparatively light due to its high performance, and thus the system-level mass saving through using a multi-functional structure is less than for a battery that uses a lower performance cell. However, if a mass reduction is required, there is less or no possibility to save mass by increasing the already high SEC_{cell}, and so using a multi-functional structure is the only way to reduce the secondary power system's mass. This is indicative of a spacecraft with particularly high performance requirements. In addition, it is possible that, while a higher a performance cell type may be available, the cost of qualifying a new technology may exceed the cost of incorporating an existing cell into a multi-functional structure.

While these mass savings represent a corresponding saving in a satellite mission's overall cost, it is also likely that using a multi-functional structure will add complexity, and hence cost, to the design, as discussed in reference [**13**]. The benefit of using a multi-functional structure would need to be weighed carefully against the cost associated with its qualification. However, it should be noted that, if the desired mass saving requires a significant change in SEC_{cell}, then it is likely that a completely different (and new) cell would need to be used. Such a new cell type would need to be qualified, and it seems reasonable to assume that the best performing cells would also be the newest, and thus would require the most extensive and costly qualification – though, once again, resources expended in qualifying the multi-functional structure would need to be compared against that of qualifying new cells.

5 REQUIREMENTS FOR DEVELOPMENT

In order to harness the structural properties of the cells and thus effect a saving in the mass of the spacecraft, it is necessary to produce an optimized structure which has the best possible structural and power storage capabilities.

5.1 Mechanical characterization

In order to model the structural properties of a multi-functional panel, the structural properties of the cell must be known. Since the cells are used as a core material, methods used to test such materials form the basis of this testing. The principal structural properties that need to be determined (i.e. those that have been found to influence the structural behaviour of a core material) are the elastic modulus and shear moduli out of the plane of the panel – that is E_z, G_{xz}, and G_{yz} (where the x- and y-axes are in the plane of the panel) [**14**].

The failure of the cells may occur structurally or electrically – that is to say, the cells may lose their electrical function while retaining structural integrity. Thus, in addition to determining the structural properties of the cell, this testing must assess the effect of mechanical loading on the cell's electrical performance.

5.2 Optimization

Once the material properties of the cells are known, finite-element analysis models can be produced to optimize the multi-functional panel design based on case studies and/or historical data. The objectives of the optimization are to produce a panel that has structural behaviour (for example, highest possible first natural frequency) and strength comparable to non-multi-functional structure, to prevent structural damage during launch, that is capable of storing an amount of energy appropriate to the size of the structure (an energy capacity appropriate to a spacecraft of the given size, based on case studies or averaged historical data) and that would not result in electrical damage (that is, degradation or loss of electrical performance) due to the loading of cells during launch. A successful panel should have a total mass, which is less than the mass of its elements were they assembled separately (that is, the mass of the multi-functional panel should be less than the mass of a purely structural panel and conventional battery pack). If the cells make a structural contribution, the mass of the final panel will be less than the mass of the cells and the original panel added together, that is, the value of η_{para} will become negative as the cells replace part of the structure.

6 CONCLUSIONS

A multi-functional satellite structure system has been presented which will use commercial lithium-ion battery cells as a structural element. The cells have demonstrated an ability to withstand the vibration, thermal and vacuum environments to which they will be subjected in service, and the processes that will be used in the manufacture of the structure.

The design of such multi-functional structures presents a challenge to spacecraft designers due to the need for multiple subsystems to be designed concurrently; however, by using low-cost commercial materials, this multi-functional structure system should allow significant savings to be made in mission cost.

The cost of concurrently designing an integrated battery and structure is not easy to quantify, and may well render the multi-functional structure impracticably costly. However, quantifying the cost of designing a multi-functional structure was beyond the scope of this work. The parametric study presented in the current paper should identify spacecraft missions that may benefit from such a system, and help to establish the performance that is required of the multi-functional structure in order to make it commercially viable.

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APPENDIX

Notation

C_{nom}	nominal cell capacity (Ah)
E_{sat}	satellite energy requirement
E_z	elastic modulus (out of plane)
G_{xz}, G_{yz}	shear moduli (in plane)
M_{batt}	mass of battery
M_{cell}	mass of battery cell
M_{para}	parasitic mass of battery enclosure
M_{sat}	mass of satellite
M_{stru}	mass of structure
R_{INT}	internal resistance
$\mathrm{SEC}_{\mathrm{cell}}$	cell mass-SEC
$\mathrm{SER}_{\mathrm{sat}}$	satellite mass-SER
V_{batt}	volume of battery
V_{bus}	volume of satellite bus
V_{cells}	total volume of battery cells
V_{EOD}	end of discharge voltage
V_{nom}	nominal cell voltage
δ_{vol}	structural mass density
ΔM_{para}	mass saving due to parasitic mass removal
ΔM_{SEC}	mass saving due to cell chemistry change
ΔM_{vol}	mass saving due to volume reduction
η_{pack}	packing efficiency
η_{para}	parasitic mass fraction
ρ_{cell}	cell mass density