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Design and development of a COTS battery for space missions at the IDR/UPM Institute

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Abstract

The design process of an Electrical Power Subsystem (EPS) for a space mission is a key factor in order to have a successful project. During last decades, components included in this subsystem have been progressively released from a more and more wide range of suppliers. This fact has allowed access to the space power technologies to an increased number of users, resulting in the arrival of COTS elements to the design and development of space power subsystems (specially in nanosats). In the present work, the development of a Li-Ion battery at IDR/UPM Institute based on COTS elements for the UNION/Lian-Hé mission is described.

1. Introduction

The EPS design process is a critical milestone due to the fact the rest of satellite subsystems depends on it. In literature, a common power subsystem is the photo-voltaic (PV) configuration which includes a main source of energy such as solar array and a secondary battery. The latter is usually a lithium-ion model due to its high performance. These models are particular manufactured for specific space mission. However, nowadays Commercial-Off-The-Shelf (COTS) elements have make its appearance, providing a cost-effective way to supply energy.

2. Background

This mission is the third educational project based on a micro-satellite designed by Instituto Universitario "Ignacio Da Riva" from Universidad Politécnica de Madrid (IDR/UPM) within a collaborative framework with Beijing University of Aeronautics and Astronautics (BUAA). IDR/UPM has previous experience on similar projects such as UPMSat-1 [**upmsat1**] and UPMSat-2 [**upmsat2**] and a remarkable engineering background on space projects such as OSIRIS, ROSETTA, ExoMars and SUNRISE missions.

3. UNION/Lian-Hé mission

The purposes of this mission are the launching and tracking process of the satellite in an orbit with 400 km altitude and an inclination angle between 41° and 43°. In general terms, the spacecraft (S/C) is a 450 mm square base parallelepiped and 500 mm height with an approximately mass of 50 kg. The required power is supplied by a PV system compound of four solar arrays with Gallium Arsenide (GaAs) cells. In Table 1, the main orbit parameters of this mission are included.

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Table 1: UNION/Lian-Hé mission parameters.

Mission Duration	2 years
Orbit	LEO Circular h = 400 km i = [41°,43°]
Mass	50 kg
Dimensions	450 mm x 450 mm x 500 mm
ADCS	Active and pasive
Average Electrical Power	43 W
Communications	S-band /UHF
Launcher	To Be Confirmed (TBC)
Ground Station	UPM /BUAA

The mission is classified as a research spacecraft whose main payloads are experiments in order to test its correct function under orbit conditions [1]. In addition, UNION/Lian-Hé satellite consists of several subsystems whose descriptions can be found at reference documents [1] [2].

3.1 Mission requirements

As it has been mentioned previously, UNION/Lian-Hé mission is currently in a preliminary phase. Therefore, several requirements have been imposed from IDR/UPM work experience such as UPMSat-2 project. In following stages, mission requirements will be further developed.

3.1.1 Structural requirements

Battery model implemented in the spacecraft will be integrated in launcher payload interface. According to launcher user manual, minimal natural frequencies are specified in order to avoid dynamic coupling from satellite and its components. As an example, Ariane 5 and Soyuz frequencies specification are presented in Table 2.

Table 2: Minimal natural frequencies of Ariane 5 and Soruz launcher.

	Ariane 5	**Soyuz**
Longitudinal	90	45
Lateral	35	15

It can be observed that previous requirements affect the entire satellite. However, internal elements should overcome natural frequencies even with a higher order of magnitude.

In addition, battery system will also be subjected to external loads which are previously evaluated by different structural simulations such sinusoidal or random vibration where acceptance levels are defined by safety margins.

3.1.2 Thermal requirements

As its homologous, thermal requirements are established to ensure a proper working at space hazardous conditions. In this case, cells manufacturing data plays an important role due to operation temperature during charging and discharging process. Therefore, thermal requirements are typically focused on not exceeding a particular temperature range. However, UNION/Lian-Hé subsystems are still in a preliminary phase so inherited thermal requirements from UPMSat-2 project have been added as a reference.

3.1.3 Battery requirements

According to previous definitions in above sections, several requirements for the UNION/Lian-Hé battery have been defined. They are listed and identified as follows:

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- **RQ-ST-001:** The three first natural frequencies of the battery model shall be a higher order of magnitude compared to launcher specification.
- **RQ-ST-002:** Safety margin from structural analyses should be positive for all load cases.
- **RQ-TH-001:** UNION/Lian-Hé battery cells shall work under operating temperature conditions, as it mentioned in cell manufacturing data.

Requirements definition for the battery system are based on UPMSat-2 battery experience which has been remarkable helpful.

4. Prototype development

In following section, a prototype evolution is described from the first steps regarding a preliminary mechanical design to the last iteration where experimental test were established.

4.1 Preliminary design

In order to size the battery, several parameters are needed. Firstly, the battery will be the only element in charge of supplying power in eclipse period so from Table 1 it can be extracted the eclipse time (t_e = 34 min) Secondly, solar array configuration is defined which means that open-circuit voltage is also known (V_{oc} = 26.67 V). Lastly, maximal power specification must be determined in order to estimated the battery capacity. In Table 3 total power specifications are summarised [2].

Table 3: Mission power requirements.

	Average Power [W]	**Maximal Power** [W]	**Minimal Power** [W]	**Peak Power** [W]
Payloads	11.3	51.3	1.2	83.3
Subsystems	8.2	8.7	5.5	8.7
Total	19.5	60.0	6.7	92.0

In space applications, lithium-ion cells are commonly used since its higher specific energy and energy density compared to Nickel cells. In addition, these cells offer a high operation temperature range and costs reduction regarding the number of cells [2]. Several models were analysed in order to create a COTS battery assembly. However, the final selected cell was the Samsung INR18650-25R [3].

The total number of cells can be calculated where 6 is the number of cells in series (n_s) and 4, the number of cells in parallel (n_p).

4.2 6S4P Prototype

This preliminary configuration meets power satellite requirements according to mission specifications. However, an enclosure must be proposed in order to envelope cells and ensure all battery requirements.

4.2.1 Mechanical design

As it is mentioned before, this specimen has to be designed taking into account previous defined requirements. Therefore, 6S4P model, which was created by using a CAD software (CATIA®), consists of the following components:

- Samsung INR18650-25R cells.
- Support structure made of aluminium 7075.
- L-profile beams
- Closed panel where electronic elements will be integrated.
- Closed structural panel to envelope the whole model.

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In Figure 1, 6S4P battery prototype is shown including all components with the exception of the fixing elements.

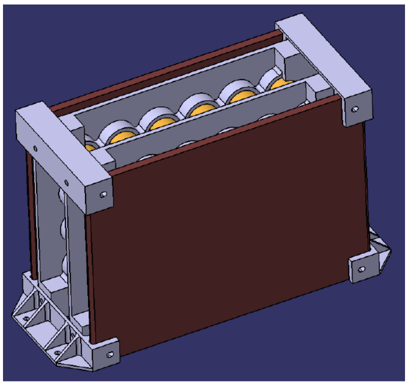

Figure 1: 6S4P prototype mechanical design.

4.2.2 Structural model and analysis

Structural modelling is approached by using an analysis software (PATRAN®) which is commonly implemented in the space sector. This kind of software allows geometrical simplifications in order to ease complex models integration. As a result, a representative model of 6S4P prototype based on finite element method (FEM) is shown in Figure 2

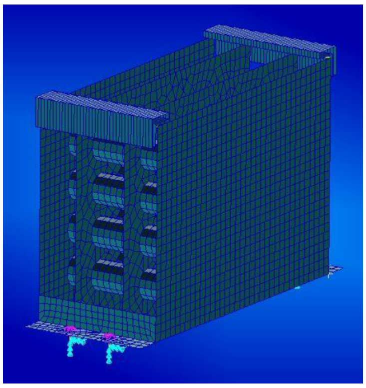

Figure 2: FEM model of 6S4P prototype [1]

Once a model has been created, particular analyses can be performed by using a structural solver (NASTRAN®) commonly linked to its counterpart. A normal modes analysis is carried out in order to verify model integrity and to obtain model natural frequencies. In Table 4, lateral and longitudinal frequencies are shown.

Table 4: Minimal frequencies of 6S4P prototype.

	Frequency [Hz]
Longitudinal	90
Lateral	35

It can be observed that obtained frequencies are higher than launcher specification in Table 2. However, theses results do not meet battery requirements (RQ-ST-001). In consequence, a new prototype definition is considered to increase model stiffness without increasing mass.

4.3 6S2P Prototype

As a consequence of the obtained results from its predecessor, a specimen is proposed to increase system stiffness, meet power requirements and not to add any extra restriction.

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4.3.1 Mechanical design

This new iteration suffered several modifications to be compliance with structural requirements:

- Initial 6S4P model was divided into two separate modules (6S2P) offering a lower position for the centre of mass. Besides, energy mission requirements are still compliance with this configuration.
- L-profile beams were replaced by two structural closed panels allowing a more suitable holding and providing an interface with satellite tray (lower closed panel).
- As result of these two modifications, general dimensions were updated in order to match previous total mass.

Having in mind this new configuration, both modules follows similar mechanical aspects which consist of the following components (Figure 3):

- Samsung INR18650-25R cells.
- Lower closed structural panel which offers an interface between satellite tray and battery
- Upper closed structural panel.
- Main structural support.
- Closed panels where electronic will be integrated.

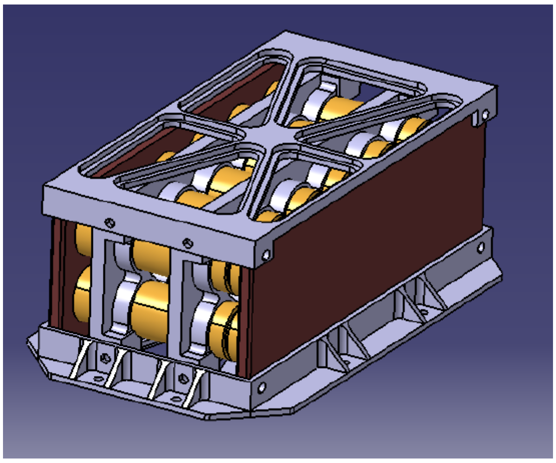

Figure 3: 6S2P prototype mechanical design [4].

In Figure 3, 6S4P battery prototype is shown including all components except of fixing elements. From reference [4], it is obvious that comparing mass budget from first iteration after modification, there is a mass saving of 0.093 kg.

4.3.2 Structural model and analysis

As its precursor, a representative model was performed in order study different load cases using similar procedure. Nonetheless, there were also modification due to recent components appearance. Hereafter, main adjustments are listed:

- As a consequence of the lower and upper closed structural panels, 2D elements were added to some section of the elements due to the existing interface with the satellite.
- Closed structural panels also show different geometry at its stiffeners, so different beam profiles were implemented.
- Fixing components were modelled by using rigid elements.

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As a result, in Figure 4 a new structural model is proposed and hence, it will be properly studied.

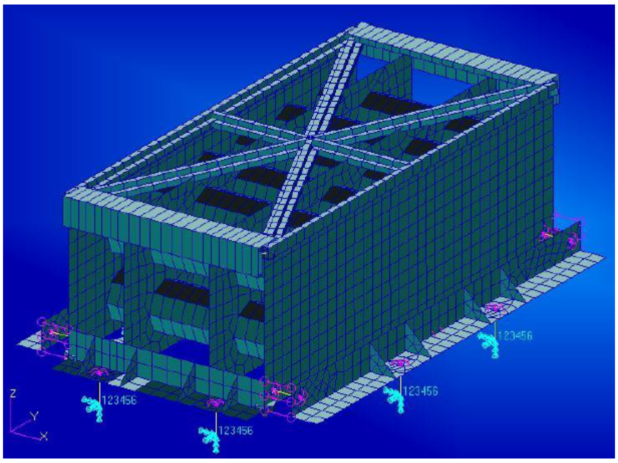

Figure 4: FEM model of 6S2P prototype [4].

In order to properly address distinct load cases, reference document from UPMSat-2 battery [5] and others from qualification test in lithium-ion cells [6] were used. Therefore, normal mode, sinusoidal and random analysis were carried out at the entire design. Nevertheless, safety margins were only calculated for the structure support since fixing element selection remains to be defined.

Firstly, a normal mode analysis was carried out in order to verify model integrity and to obtain model natural frequencies. From reference [4] lateral and longitudinal frequencies are obtained which fulfil launcher vehicle and battery system requirements.

Secondly, a sinusoidal analysis was performed where applied acceleration and frequency range are depicted in [4]. In Figure 5, sinusoidal analysis in longitudinal direction is shown.

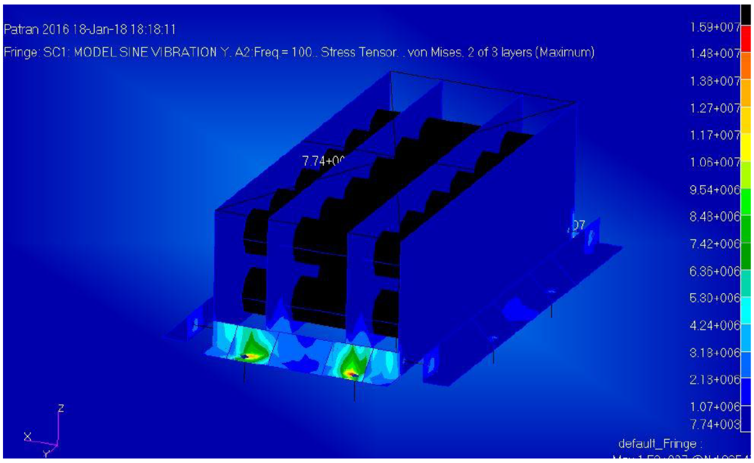

Figure 5: Sinusoidal analysis in longitudinal direction [4].

The obtained results satisfy battery requirement (RQ-ST-002) since all safety margins for the sinusoidal analysis are positive [4]. However, results also showed that closed panels dedicated to electronic were significant affected by the load case, giving the lowest safety margin. Therefore, these elements will be inspected in case configuration suffers a further alteration.

Lastly, a random analysis was executed where power spectral density (PSD) and frequency range is specified for lateral and longitudinal direction in [4]. In Figure 6, the random analysis in longitudinal direction is shown.

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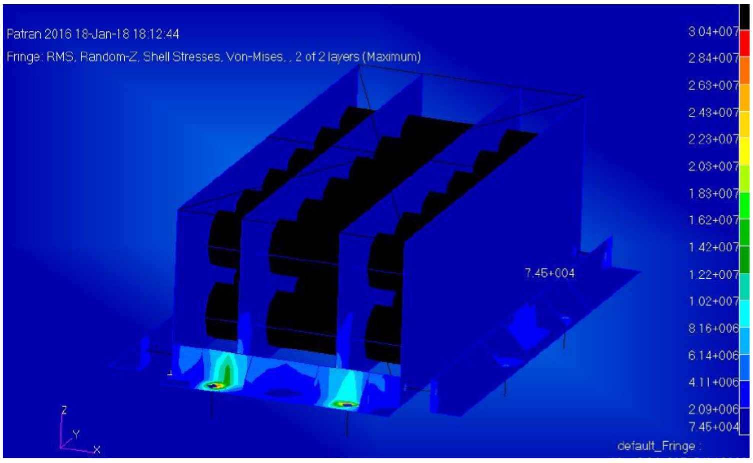

Figure 6: Random analysis in longitudinal direction [4].

The obtained results satisfy battery requirement (RQ-ST-002) since all safety margins for the random analysis are positive [4]. Nonetheless, as in the previous analysis, the lower safety margin was found at the electronic panels.

Once all simulations have been performed, it is noted that structural requirement applied at the battery system are compliance (RQ-ST-001 and RQ-ST-002).

4.3.3 Thermal model and analysis

The preliminary thermal analysis is a steady-state simulation assuming a cold case as the worst scenario due to previous work on UPMSat-2 battery. This simulation has been approached by FLUENT®and ANSYS®software.

The aforementioned cold case refers to the discharging battery process because satellite is in eclipse period and solar arrays are unable to supply energy. Besides, a low cell dissipation is considered which leads to the worst cold case and radiation effects are negligible due to battery is assumed to be covered by a multilayer insulation (MLI). Hence, conduction effects are only considered. Boundary conditions are depicted in [4].

In addition, thermal properties must be applied to the components material before simulation. However, this information does not appear in the cell manufacturing data. Consequently, an average value has been estimated from experimental test on similar lithium-ion cells [7] [8] [9]. Material thermal properties are included in [4].

A first simulation is performed (Figure 7) and it shows satellite-battery interface becomes in a significant heat sink so battery temperature does not increased and charging process will not start at a temperature lower than 0 °C, according to cell specification. For that reason, a second analysis is carried out.

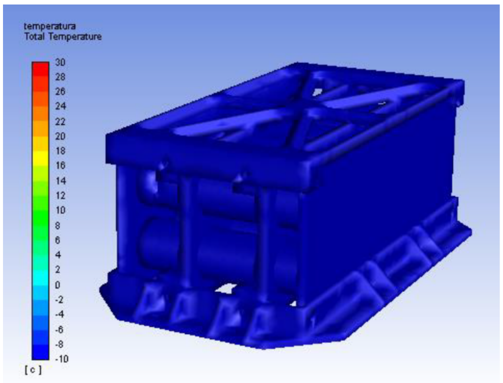

Figure 7: First thermal simulation in 6S2P prototype [4].

In this case, a Teflon lamina is placed at the satellite battery interface to isolate cells from environment. Teflon thermal properties are included in [4]. However, Teflon thickness has been calculated for different thermal cases and the corresponding relation is included in [4]. As a result, a minimal Teflon thickness of 5 mm is accomplished to overcome minimal cell temperature. In addition, results for this condition are shown in Figure 8.

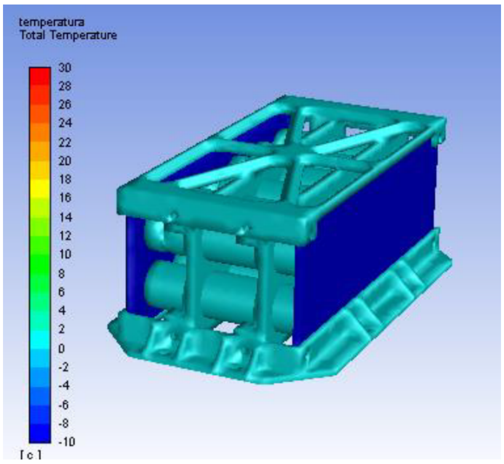

Figure 8: Second thermal simulation in 6S2P prototype including Teflon lamina [4].

In conclusion, minimal temperature for charging process is overcome. However, it is recommended that operating temperature range is between 10 °C and 20 °C. Despite of the fact that battery thermal requirements have not been met, preliminary thermal simulation have concluded and a detailed thermal control system will be proposed taking into account electronic equipment dissipation in further stages.

4.3.4 Electrical design

As it was described previously, two identical models with 6S2P configuration were developed. However, a new difference is made. On the one hand, a specimen is subjected to experimental test simulating charging and discharging process without considering any circuit protection. On the other hand, a module is considered as the qualification model. Both designs are named, Prototype 6S2P-01 and Prototype 6S2P-02, respectively.

5. Prototype 6S2P-01

An experimental model is proposed in order to test charging and discharging process without cell imbalance protection or any kind of circuit protection. Under this condition, cells specified in previous sections will perform a power profile similar to UNION/Lian-Hé mission.

Prototype 6S2P-01 electrical model is shown in Figure 9 where transmission and imbalance protection signals are not included.

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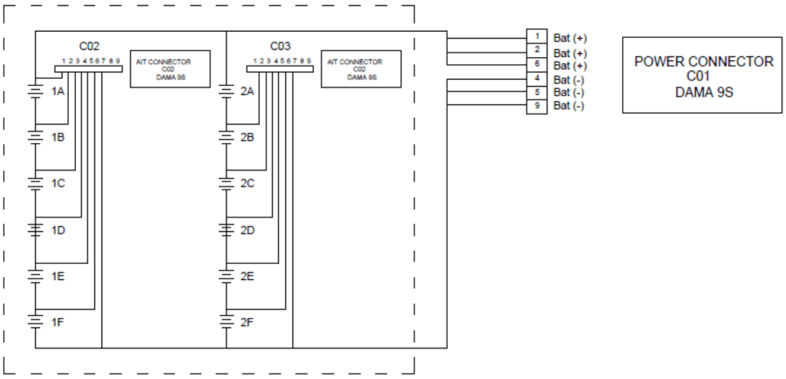

Figure 9: Prototype 6S2P-01 electrical design [10].

Although a simulation showed that structural requirements were met, a preliminary electrical model was implemented by manufacturing the support structure through 3D impression with ABS as material. In Figure 10 and Figure 11, entire experimental model is shown in two different views (lateral and front). The detailed manufacturing process has been omitted due to its large extension, which includes equipment and tools [10].

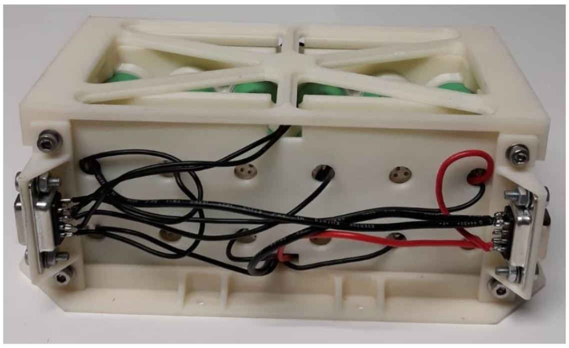

Figure 10: Prototype 6S2P-01 electrical model assembled in a ABS support structure (lateral view) [10].

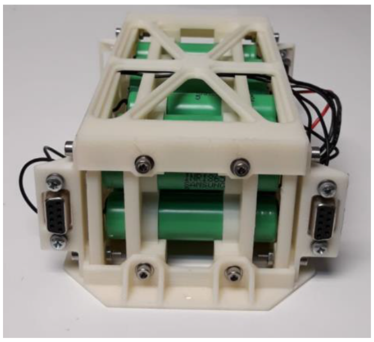

Figure 11: Prototype 6S2P-01 electrical model assembled in a ABS support structure (front view) [10].

5.1 Experimental tests

This section explains charging and discharging processes definition. The study´s aim is to analyse cell imbalance after a particular number of cycles. The recording voltage process is achieved by Mayuno®software [11].

Prototype 6S2P-01 is subjected to a power profile similar to mission orbit to study cells internal behaviour. According to mission power requirements and orbit parameters, a current profile has been estimated for experimental

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tests and the cycle duration is fixed at 90 min, respectively. Therefore, a test cycle is defined as follows:

- Quasi-static discharge (30 min) which corresponds with eclipse period.
- Latency period (5 min).
- Quasi-static charge.
- Dynamic stress tests (DTS) [12].

5.2 Results

In this section, obtained results are exposed after 45 test cycles. On the one hand, in Figure 12, battery voltage evolution is shown

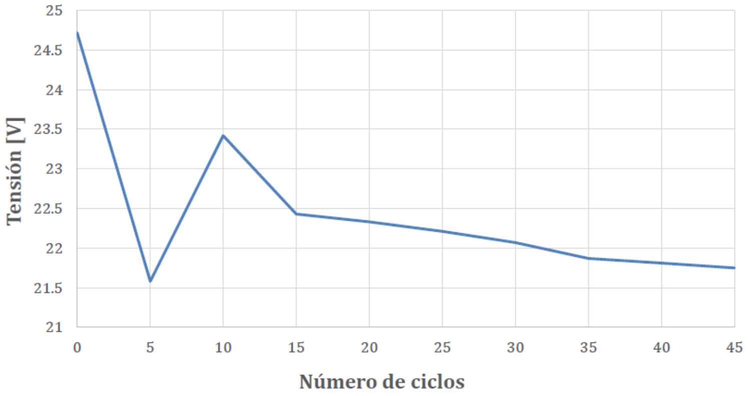

Figure 12: Prototype voltage level after 45 test cycles [10].

where it can be seen that there is a wide variation at the beginning, but after approximately 15 tests cycle battery voltage is still decreasing. At this point, it is obvious that after a considerable period the battery need to be charge not to reach minimal voltage level. On the other hand, in Figure 13 and Figure 14, S3 and S4 voltage level is shown, respectively.

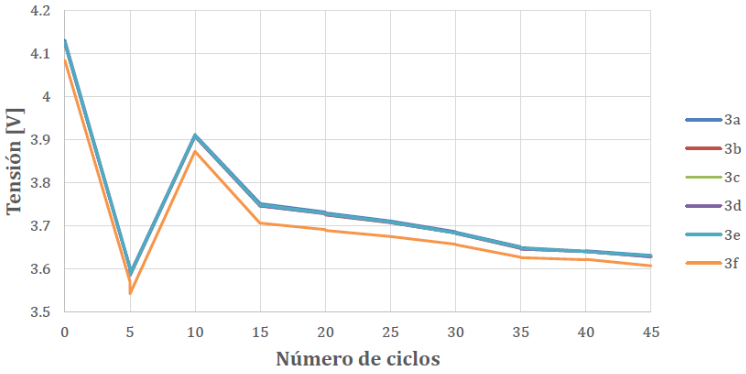

Figure 13: Cell voltage level after 45 test cycles [10].

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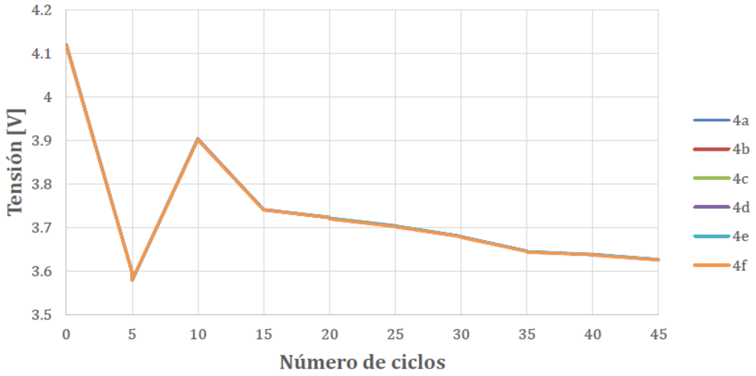

Figure 14: Cell voltage level after 45 test cycles [10].

From Figure 13, it is observed an extensive variation between each cell, specially obvious at *3f* cell whose voltage level is the lowest of the row. This variation is due to cell imbalance which provides a higher degradation. On the contrary, in Figure 14, there is no significant variation between each cell.

In summary, cell imbalance for each cell after 45 test cycles is shown in Figure 15 where difference between each other are notably appreciable. Cells imbalance at S3 row is always higher than 20 mV whereas cells imbalance at S4 row is always less than 5 mV. In conclusion, a higher number of test cycles is required in order to validate previous results. However, S3 row will be under special attention in case cell imbalance can be increased.

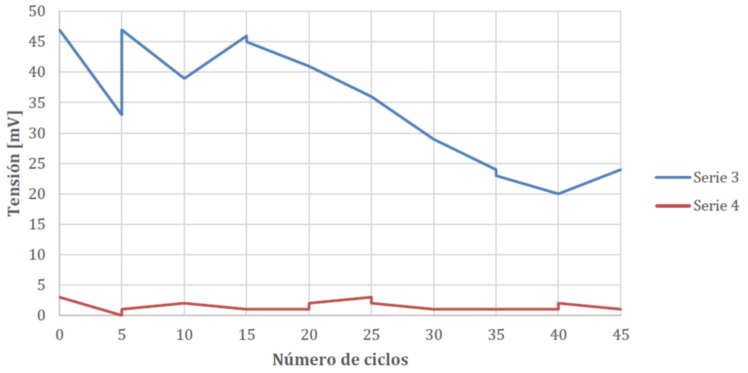

Figure 15: Row imbalance after 45 test cycles [10].

6. Conclusions

A battery for a space mission based on COTS elements is being designed at the IDR/UPM Institute. Several analysis on the mechanical design (vibrations), thermal properties, and the electrical design have been carried out within the last couple of years. According to this results the initial prototype was modified, the 24-cell configuration being changed to a 12-cell configuration. Obviously, the target mission that sets the requirements for this battery will require two 12-cell battery modules.

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References

[1] Aldea, Jorge García. “Diseño y caracterización de una batería de uso espacial”. Desarrollo y Ensayos Aeroespaciales DEA. May 2017.

[2] Aldea, Jorge García. “Desarrollo de una batería de uso espacial a partir de células comerciales”. July 2017.

[3] *SPECIFICATION OF PRODUCT:Lithium-ion rechargeable cell for power tools*. INR18650-25R. Rev. 1. Samsung. May 2014.

[4] Zaragoza Asensio, Juan Antonio. “Desarrollo de una batería para uso espacial a partir de células comerciales: Diseño mecánico, análisis estructural y térmico preliminar”. Desarrollo y Ensayos Aeroespaciales DEA. May 2018.

[5] *UPMSat-2 Battery Mechanical Tests Test Produce*. IDR/UPM. 2017.

[6] Semerie, J.-P. “Lithium-ion batteries for geosynchronous satellites. Qualification test results of the STENTOR battery”. In: vol. 1. Feb. 2000, 621–628 vol.1.

[7] “Methodology to determine the heat capacity of lithium-ion cells”. In: *Journal of Power Sources* 395 (2018), pp. 369–378.

[8] “Thermal analysis of lithium-ion batteries”. In: *Journal of Power Sources* 140.1 (2005), pp. 111–124.

[9] Muratori, Matteo *et al.* “A Model Order Reduction Method for the Temperature Estimation in a Cylindrical Li-Ion Battery Cell”. In: vol. 1. Jan. 2010.

[10] Zaragoza Asensio, Juan Antonio. “Desarrollo de una batería para uso espacial a partir de células comerciales: Diseño detallado y prototipo eléctrico = On the development of a Li-ion battery for space systems : Detail design and prototype”. Desarrollo y Ensayos Aeroespaciales DEA. May 2018.

[11] *Programmable DC Electronic Load - Models M98 Series*. Maynuo Electronics. 2012.

[12] *ELECTRIC VEHICLE BATTERY TEST PROCEDURES MANUAL*. INR18650-25R. Rev. 2. 1996.