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Towards Validation of Battery Mission Lifetime for Nano-satellites: Fast, Cheap and Accurate Through a Representative Mission Profile

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***Abstract*—Satellites' mission lifetime is limited by the lifetime of the battery. Thus, it is necessary to assess in advance on the ground that the applied battery design will support long-term commercial missions. So far, the applied approaches are either over-simplified and cannot be used for long-term predictions, or they are too expensive (time wise or testing resources wise). A synthetic mission profile is developed in this work based on telemetry data, to create a representative profile considering real battery mission conditions and to allow for low cost and effective lifetime testing.**

***Index Terms*—battery, nano-satellite, mission profile**

I. Introduction

Small size (nano-, micro-) satellites are a rapid growing branch of space industry [1]. They are based on commercial-off-the-shelf (COTS) components, which allow for low cost solutions [2]. However, further analysis and qualifications are needed for their use in space [3]. The lifetime and the usability of a satellite is critically dependent on its energy source, which is the most typically for nano-satellites a combination of solar panels and batteries. During a sunlight, solar panels supply the load and charge batteries, while during an eclipse batteries are used to cover the power demand. Thus, a nano-satellite mission lifetime can be considered to be limited by the battery lifetime. Until recently, the purpose of majority nano-satellite missions has been scientific, including scientific earth observation, or In-Orbit Demonstration (IOD). Typically, the required lifetime for these missions is short (in the range of months, up to one or two years). However, as the interest for commercial applications is growing, these missions, focused on communication, earth observation and remote sensing, require much longer lifetime [4]. Will the batteries be able then to support these missions?

A nano-satellite mission lifetime can be considered that is limited by the battery lifetime. Are COTS batteries able to support not only short-term IOD missions, but also long-term commercial missions?

This question has been attempted to be addressed in various works, mentioned later. Since the most common deployment of nano-satellites is low-earth orbit (LEO), then the considerations will target it, and its related properties. Based on the height placement in LEO, the orbit time will be roughly between 88 and 127 minutes, which can be translated into 4100 – 6300 cycles per year. In the approach used by Navarathinam et al. [5], they cycled a battery by 0.5 C and 100% depth-of-discharge (DOD) in standard temperature and pressure (STP) conditions. Then, by a ratio relation, considering the mission DOD to be 30% and the acceptable loss in capacity being 50%, they predicted the battery cycle life and found it sufficiently for meeting the one year requirement. While this approach might work well for a shorter target lifetime, in the case of a longer target lifetime (e.g., 5 years) it would result in an extremely over-sized battery, which might not be economically or design-wise feasible. This is due to the battery degradation being dependent on the cycle's DOD and as it was shown by Omar et al. [6], this dependency is not linear. Anderson et al. [7] performed lifetime tests considering two DOD conditions: 20% and 100% DOD, so lifetime information closer to a target LEO operation can be obtained. The tests were carried out on 1.3 Ah Varta lithium-ion polymer cells at 20°C by 0.2 C. However, the limitation of this approach comes from very low applied current, which results in 2 hours for a 20% DOD cycle, which results in a period of 1.14 years of testing, to validate 1 year of LEO life (assumed 5000 cycles). Pearson et al. [8] presented an approach accommodated at AEA Technology to 1.5 Ah Lithium-ion SONY 18650HC cell. A large amount of tests was performed at various conditions (DOD, temperature, charge/discharge profiles) and resulted in a collection of 40 million hours of life-test data, which have been assumed to support expectations of the cell to deliver up to 6.5 years at LEO. The specific test conditions were not presented, though based on the amount of data they had to be very extensive and/or the tests were performed for a long time. Even though this seems as a desired approach, providing a high confidence in mission lifetime prediction, if it is assumed that data was collected for example over a period of 5 years, then it would require a test facility with over 900 battery test channels, which is simply considered to be too many (too expensive) to ask from smaller/medium size companies or universities, whom main focus and activity is not in batteries (but for example the whole product, i.e. satellite).

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A generic LEO profile for battery lifetime testing is defined in the ESA standard [9], as repetitive cycles with 60 minutes CC-CV charging (sunlight) and 30 minutes CC discharging (eclipse) at a representative temperature related to the mission. This real time test can be further accelerated by increasing the currents or the temperature, which have to be carefully selected in order to avoid cells over testing. Otherwise, no representative cycle, closer to real battery mission conditions, is established.

The use of representative cycles is no new concept. Driving cycles have been widely applied in automotive world to evaluate vehicles' emissions and consumption [10]. Consequently, with uprising of electric vehicles, the driving cycles have been used to evaluate battery performance and lifetime, because they provide higher accuracy than constant- current or power cycling [11].

Thus, the target of this work is to develop a lifetime test procedure, which resembles a real satellite use, is time efficient and is low cost. In order to provide this, a synthetic mission profile is developed based on nano-satellite telemetry data and design considerations.

II. Technology description

GOMX-3 and GOMX-4A/B satellites are used in this work as examples. Batteries used for nanosatellite battery packs are 2.6 Ah Li-ion cells [12]. GOMX-3 was equipped with a battery pack BP4, consisting of 4 cells in configuration 4s-1p [13]. GOMX-4A/B are equipped with battery packs BPX with 8 cells in configuration 4s-2p [14]. All presented telemetry data are scaled to one cell. The satellites are illustrated in Fig. 1.

A. GOMX-3

GOMX-3 was a 3U nano-satellite with missions of technology IOD such as attitude control, high-speed data downlink, radio frequency sensing, and aircraft monitoring ADS-B. It was launched in October 2015 and it was in operation for about a year, before it re-entered the Earth's atmosphere in October 2016.

B. GOMX-4

GOMX-4 mission consists of two 6U satellites GOMX-4A and GOMX4-B, launched in February 2018. GOMX-4A's task is Arctic surveillance, while GOMX-4B's objective is station keeping capabilities. Together, they prove a concept of inter-satellite linking and aircraft (ADS-B) and ships (AIS) monitoring. Both satellites are still in operation.

III. Mission profiles analysis

The current profile analysis is based on telemetry data from three satellites: GOMX-3 (from 5.10.2015 to 17.10.2016), GOMX-4A and GOMX-4B (from 2.2.2018 to 31.1.2019). For illustrative purposes, only GOMX-4A data are presented. The one-year-long battery current profile is shown in Fig. 2a). At first, it was visually inspected to determine general trends. It was found out that the battery charging profile is roughly uniform, while there are three distinguished levels of discharging

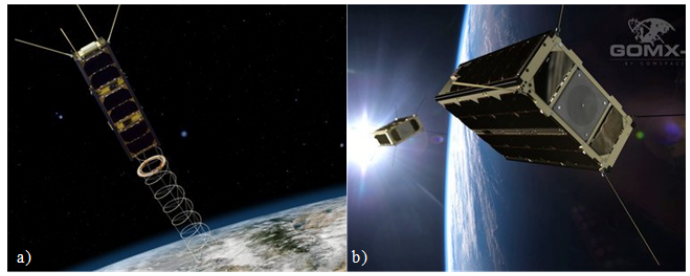

Fig. 1. Illustration of a) GOMX-3 and b) GOMX-4A/B nano-satellites (source: gomspace.com).

load current, as it is shown in Fig. 2b). The levels are labeled as base load, medium load, and peak load. To get an overview of total current distribution, a histogram representation was considered for charging and discharging current separately, shown in Fig. 2c). The histogram confirms the previously observed trends in the current profile. In order to simplify the current distribution, the histogram is refined considering a five step resolution, as shown in Fig. 2d). Since the third and fifth bins of discharging current have very small occurrence, they are for further calculations added into the fourth bin – the peak load. Moreover, dividing the current into three load levels can be supported by an analysis of the nano-satellite power budget. This is illustrated on an example of GOMX-4A power budget, presented in Table I. There are four categories of modules with power consumption according to the duty cycle: base (100% duty cycle), medium (50% duty cycle), peak (4.24% duty cycle) and unassigned (0% duty cycle). The power consumption can be translated to the load current experienced by a single battery cell by considering a battery voltage of 3.7 V and the nano-satellite configuration 4s-2p. Then, the summed current demand for low, medium, peak, and unassigned loads is 0.14, 0.02, 0.11 and 1.08 A, respectively. The cumulative current in this order is 0.14, 0.16, 0.28 and 1.36 A, respectively. Thus, the base load current identified in Fig. 1 corresponds to 100% duty cycle demand in Table I. For medium and peak load, there is a significant participation of the unassigned load (without determined duty cycle - on demand consumption) and the identified peak load current resulted to be approximately a half of total possible cumulative current demand.

At this point, the current levels ($I_{mean,cha}$, $I_{mean,dch}$) and their ratios were determined for each satellite based on their telemetry data. The next step is to integrate these values by computing their means over the satellites in order to obtain a generalized current distribution. Then, the resulting distribution is used as the base for a synthetic mission profile (SMP). The current values and their ratios for each satellite and the resulting average are show in Fig. 3.

For further calculations, it is necessary to determine an orbit time (T_{orbit}) and a fractional duration of eclipse in relation to the orbit time (Ecl). T_{orbit} is around 5670 seconds for GOMX-4. For GOMX-3, T_{orbit} started at 5556 seconds and it decayed to 5290 seconds. The ESA standard defines a generic

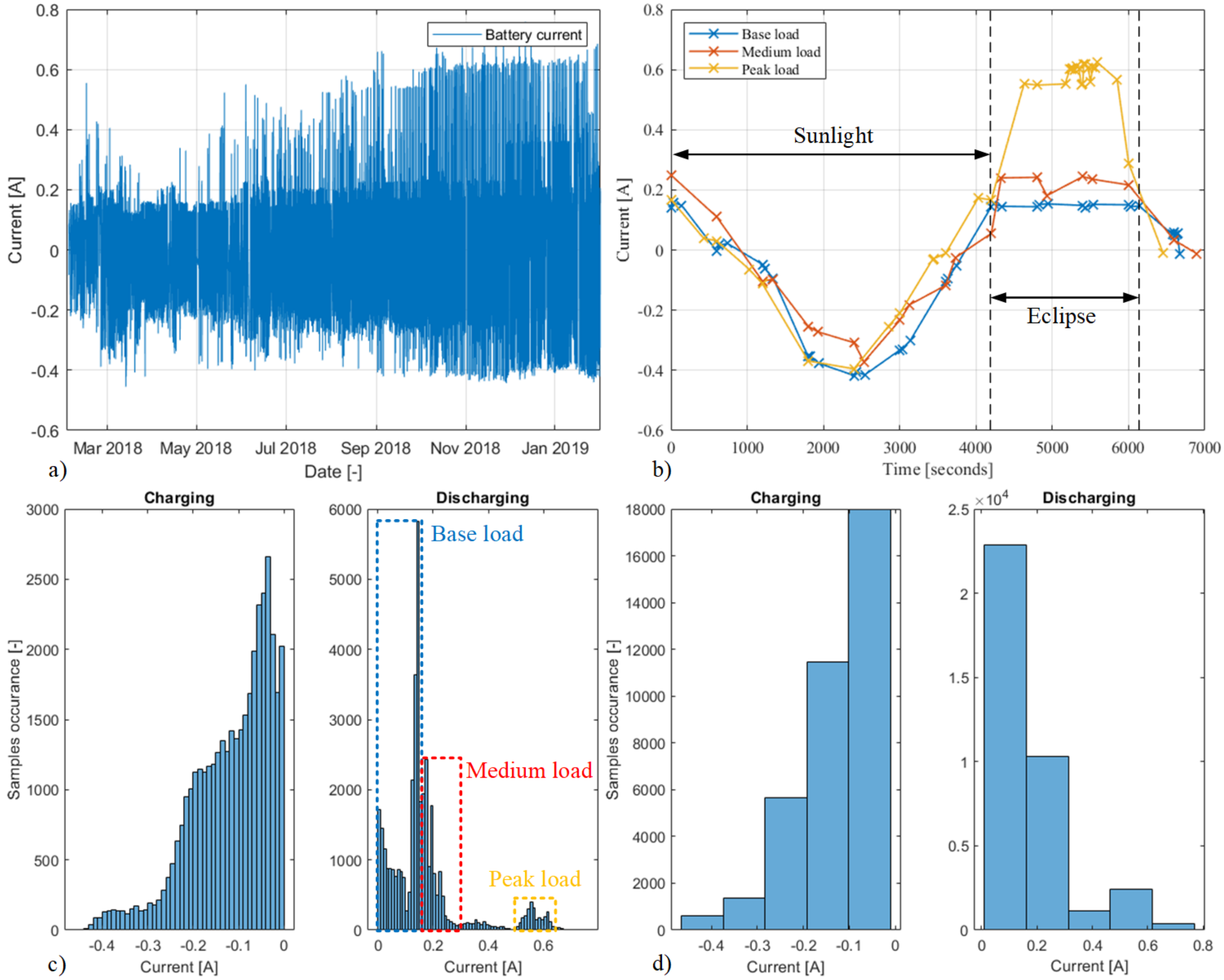

Fig. 2. Extracted and processed telemetry for GOMX-4A: a) battery current over a period of one year; b) examples representing a base load, a medium load and a peak load consumption; c) a histogram of battery current; d) refined histogram with respect of three load modes.

LEO profile orbit time as 5400 seconds [9] and this value is selected for the SMP. The fractional eclipse length towards the orbit time has been approximately at 0.39 for GOMX-3 and 0.31 for GOMX-4. The ESA standard specifies it to be 0.33, which seems as a good compromise and thus it is used for Ecl. An important mission design parameter is the DOD, which batteries are going to experience. Generally, smaller values of DOD are selected in order to reach a longer lifetime. In this case, a value of 10% DOD was selected. It relates to the 10% of battery capacity at its beginning of life, so for a 2.6 Ah battery it is 0.26 Ah for absolute DOD capacity ($aDODC$). Finally, we decide an accelerating factor (AF) for the SMP, by which the orbit time will shorten. In this study, an AF equal to 2 is used. Then by the procedure below, the new current values are obtained for the lifetime test:

for charging current:

$$T_{sunlight\_new} = T_{orbit} \cdot \frac{1 - Ecl}{AF} \tag{1}$$

$$I_{avg,cha,new} = \frac{aDODC \cdot 3600}{T_{sunlight\_new}} \tag{2}$$

$$I_{cha\_new} = \frac{I_{avg,cha,new}}{I_{avg,cha}} \cdot I_{mean,cha} \tag{3}$$

for discharging current:

$$T_{eclipse\_new} = T_{orbit} \cdot \frac{Ecl}{AF} \tag{4}$$

$$I_{avg,dch,new} = \frac{aDODC \cdot 3600}{T_{eclipse\_new}} \tag{5}$$

$$I_{dch\_new} = \frac{I_{avg,dch,new}}{I_{avg,dch}} \cdot I_{mean,dch} \tag{6}$$

TABLE I
POWER BUDGET CONSIDERATIONS FOR GOMX-4A

Module	Description	Duty cycle [%]	Power consumption [W]
NanoPower P60	Electronic power system's unit	100	1.11
NanoMind A3200 OBC	Onboard Computer	100	0.30
NanoCom AX100 Rx	UHF reception and transmission for communication	100	0.30
NanoMind A3200 ADCS	Attitude Determination and Control System component	100	0.30
NanoSense FSS (fine sun sensors)	Sensors to measure Sun's position relative to the satellite	100	0.05
NanoSense M315 (magnetometer)	Sensors to measure strength and direction of the Earth's magnetic field	100	0.01
NanoTorquer GSW600 x1	Reaction wheel to provide high torque and momentum storage capability	100	0.62
NovAtel OEM615/OEM719 GNSS Receiver	Multi-frequency positioning system	100	1.50
NanoTorquers 3-axis magnetorquers	Device to change attitude via the Earth's magnetic field	50	0.61
NanoCom AX100	Long-range VHF/UHF transceiver	4.24	3.37
Sensonor STIM210	Multi-axis gyro module	0	1.87
NanoTorquer GSW600 x4	Reaction wheel to provide high torque and momentum storage capability	0	2.09
NanoCom S100	VHF/UHF transceiver	0	4.76
Inter-Satellite Link	Satellite-to-satellite communication	0	6.80
High-Speed Link	Satellite-to-ground communication	0	6.80
NanoCom ADS-B receiver	Reception of Aircraft ADS-B transponder beacons	0	1.16
QubeAIS receiver	Automatic Identification System (AIS) receiver	0	1.08
NanoCam C1U	Camera payload for nano-satellites	0	0.74
NanoPower BPX Heaters	Battery heaters	0	6.73

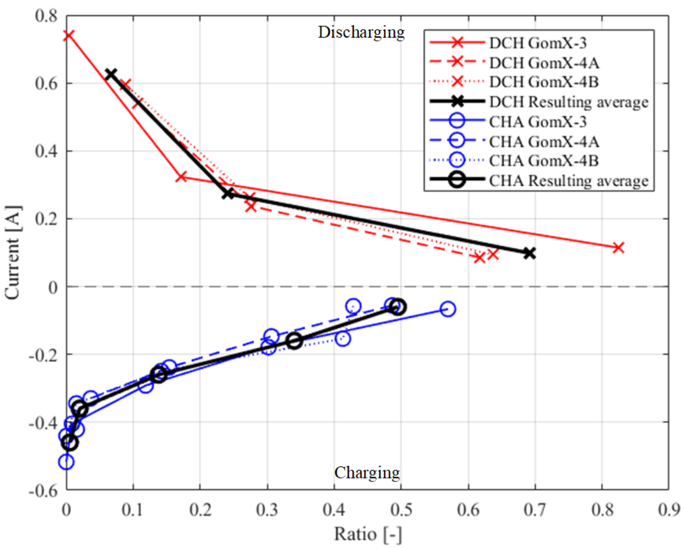

Fig. 3. Charging and discharging currents and their relative occurrence for GOMX nano-satellites and the resulting average values.

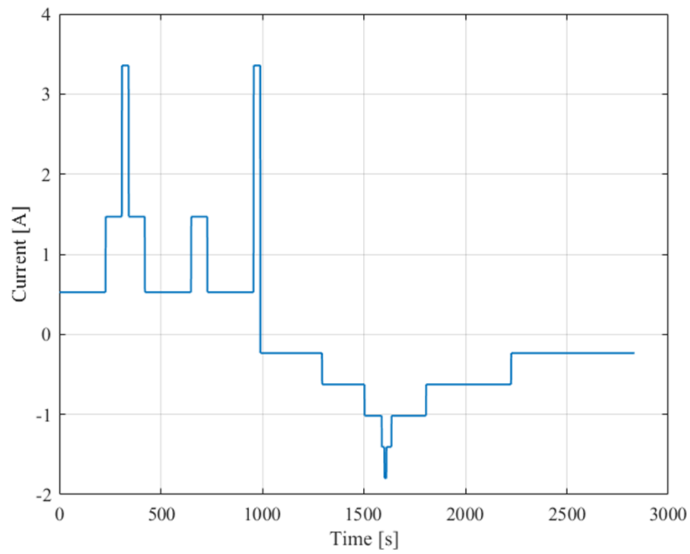

Fig. 4. A synthetic mission profile (SMP), which represents a nano-satellite battery usage. The negative values stand for charging.

A characteristic of the charging current profile is that it sharply increases in cca 1/3 of sunlight time and then it slowly decreases for cca 2/3 of the time. Thus a similar proportional profile is composed. For the discharging current profile, there is no specific lead of how it should be composed. It is considered that it should be simple enough, while beneficial for further use and representing a worst case. If the loading order will be only the base load, the medium load and the peak load, a lot of information about battery and its dynamic can be lost. Therefore, an indented profile was chosen, where one can find all the various transitions, as illustrated in Fig. 4. Moreover, this profile fulfills the worst case scenario, as it includes the peak load at the end of discharge (a placement, where will be the lowest SOC, thus the highest change to trigger discharge safety voltage limit). Since computed current values can have many decimal places, for practical reasons it is convenient to round them (depending on a range of a test device, typically two decimals); however, it has to be ensured that integral capacity over a cycle is zero or slightly negative (more charge input), so that SOC will not drift over many cycles. The final SMP is shown in Fig. 4.

IV. EXPERIMENTAL

A 2.6 Ah Li-ion cell was tested using a MACCOR Series 4000 battery test station. The thermal chamber temperature was maintained at 25°C. At first, the cell was charged by constant-current of 1.25 A to 4.0 V and then by constant-voltage to 0.05 A. Then, 5 SMP cycles were applied and they are presented in Fig. 5, the data was collected with 1 second resolution. One can see in Fig. 5a) that the discharging part of SMP was performed exactly as defined. However, the charging current profile is somewhat distorted. This is a result of battery dynamics, namely a high charging current causing a high over-potential, which is exceeding the set maximal charging voltage (4.0 V), thus the current is limited and the cell being charged by a constant-voltage mode in those periods. A reduction of

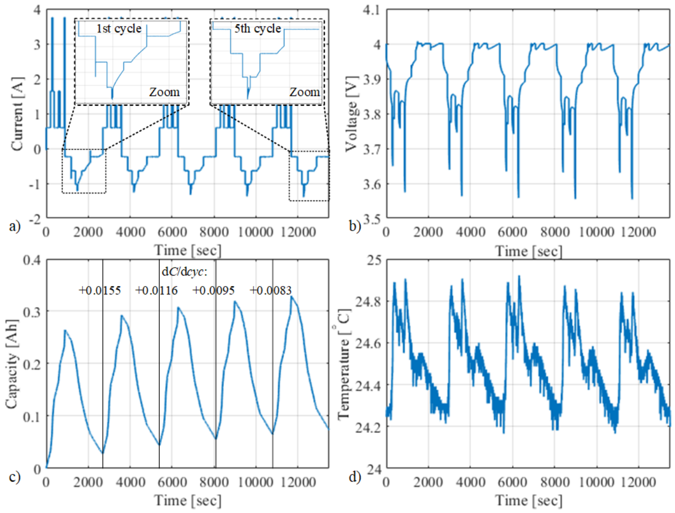

Fig. 5. Profiles from laboratory measurements of five consecutive SMP cycles at the 2.6 Ah Li-ion cell: a) current, b) voltage, c) capacity and d) temperature profiles.

charging current compared to the pre-defined SMP results into the cell being charged less than it is discharged and then the SOC is drifting. In Fig. 5c), one can see how the capacity of the cycles is drifting. This behavior is only temporary and the charge-discharge capacity over a cycle will get balanced, when the cell will reach steady-state, which will happen when no current limitation will be reached. Alternatively, to avoid or shorten the settling period, the cell's cut-off current in the charging step before the SMP procedure can be increased to a higher value than 0.05 A. This behavior can also occur in nano-satellites based on their mission preparation procedure. The thermal behavior during SMP is shown in Fig. 5d), where temperature spikes up during discharging phase due to higher discharging currents, while it drops down during charging phase. The same heating pattern and contribution is expected from cells in nano-satellites, as well. Moreover, the resulting temperature of a cell will be highly influenced by the environmental temperature and heat generation of other nano-satellite parts.

V. Discussion

The derived SMP will be used for accelerating aging tests to assess battery lifetime for nano-satellite applications. The SMP shall offer a good compromise between testing complexity, time efficiency, need for testing resources and matching of the real nano-satellite conditions. Nevertheless, further aging tests will be required to clearly determine relations of the SMP to Li-ion battery aging dependence on C-rate, to determine up to which accelerating rate are the results still valid to a nano-satellite use and when another aging phenomena are triggered. Moreover, the SMP can be used for the evaluation and a comparison of batteries' performance for nano-satellite applications. This work is considered to be the first step directing the discussion of testing standardization in this new industry segment. At the end, standardized test profiles shall be established, similar to the ones available for the automotive industry (e.g. Worldwide harmonized Light vehicles Test Procedures (WLTP), or New European Driving Cycle (NEDC)).

VI. Conclusion and Future Work

There is a need for a representative mission profile for batteries in nano-satellites. Thus, the telemetry data from three nano-satellites were analysed and the current characteristics determined. Three distinguished load levels were observed, while charging current shown rather a continuous evolution with faster increase and slower decay. SMP was composed based on the telemetry observation and ESA's definition of a generic LEO profile. Future work will focus on using the SMP for battery degradation and comparison against telemetry data and constant-current degradation tests. Moreover, the current influence on the battery aging has to be determined in order to apply the most possible acceleration factor, while staying close enough to the real operation.

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