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23rd IAA SYMPOSIUM ON SMALL SATELLITE MISSIONS (B4)
Generic Technologies for Nano/Pico Platforms (6B)

Author: Mr. Ronnie Nader, Sys. Eng. EXA Cosmonaut
Ecuadorian Civilian Space Agency (EXA), Ecuador, rnader@exa.ec

Mr. Manuel Uriguen, Mec. Eng.
Ecuadorian Civilian Space Agency (EXA), Ecuador, muriguen@exa.ec

Mr. Sidney Drouet, Elec. Eng.
Ecuadorian Civilian Space Agency (EXA), Ecuador, sdrouet@exa.ec

Mr. Gerard Nader Drouet,
Ecuadorian Civilian Space Agency (EXA), Ecuador, gnader@exa.ec

HIGH ENERGY DENSITY BATTERY ARRAY FOR CUBESAT MISSIONS

During the development of the first Ecuadorian satellite once mission objectives and payload design was complete, the power budget calculations indicated that we would need a large amount of energy to run the main payload which was a real time video transmission system, our system design guidelines dictated that such power matrix should be robust, redundant and would need a backup system in order to ensure a continuous operation over the longest period of time possible, considering that our solar arrays were composed of solar cells with an efficiency of only 19 percent.

We needed a power supply of at least 26.64 Watts per bank, and as per our system safety guidelines the power matrix turned into 4 of this banks, giving a total of 106.56 Watts, the challenge was to pack this much power into an space small enough to fit into a 1U structure. The benefits of having this much power available for the spacecraft became obvious as we calculated the expected life of the power matrix and simulated/tested the illumination-eclipse cycle and charge-discharge periods, thus reducing the load on each cell and maximizing the expected battery life, each array was composed of 16 cells each, and our spacecrafts carry 2 of this arrays on board, also each array uses the waste heat of the spacecraft electronics to warm itself by the use of a carbon nanotubes based thermal transfer system and a micro MLI layer that allows the arrays to avoid radiating this heat back into the neighboring internal electronics.

Now after more than 3 years operating in space in 2 spacecrafts, NEE-01 PEGASUS and NEE-02 KRYSAOR, this battery array design has demonstrated to exceed the expectations of the system design guidelines. This paper will describe the system; discuss testing and operation data as well as a new thin design to flight in one upcoming U.S. cubesat mission next year and more follow-up missions of this program.

Introduction: EXA is the Ecuadorian Civilian Space Agency, a civilian NGO created in 2007, in charge of the administration and execution of the Ecuadorian Civilian Space Program – ECSP[1].

As part of the ECSP, Project PEGASUS was developed to demonstrate indigenous satellite building, testing and operation capabilities and to this end a ground station called HERMES-A[2][3][4][9][11] was built and tested successfully from 2009 to 2013 and 2 satellite flight models were also built in house, tested and launched successfully, both satellites are still in operation at the time of writing this paper.

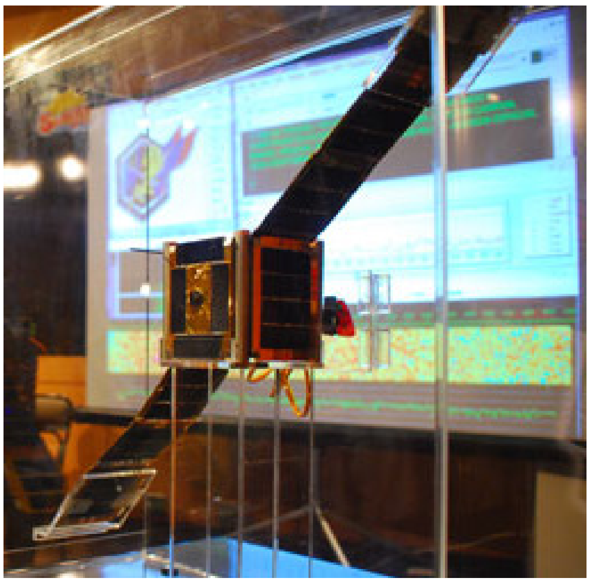


Fig-1 The NEE-01 PEGASUS in orbital flight configuration with its 2 DSA Multipanel solar wings deployed during its maiden presentation on April 4, 2011

NEE-01 is the Ecuadorian registry number meaning 'Ecuadorian Space Ship – 01' in Spanish, so the first spacecraft was christened **NEE-01 PEGASUS,** launched on April 25, 2013[19] and the second one was christened **NEE-02 KRYSAOR,** launched on November 21, 2013[30], both satellites were NEE-class spacecrafts as we named this architecture.

The design and development team was led by Cmdr. Ronnie Nader and composed by Sidney Drouet, Manuel Uriguen, Hector Carrion, and Ricardo Allu[23]

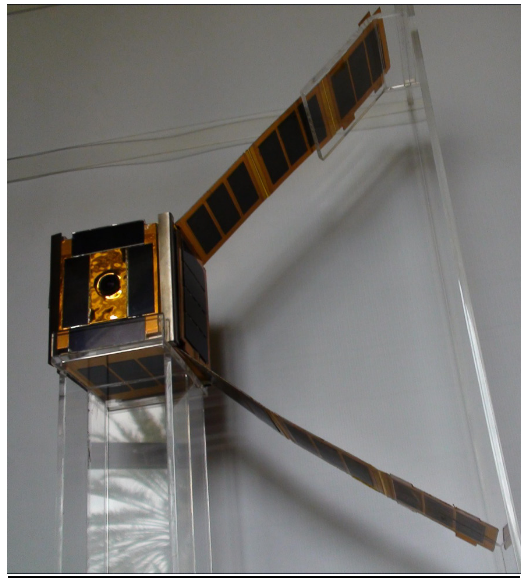


Fig-2 The NEE-02 KRYSAOR in orbital flight configuration with its 2 DSA Multipanel solar wings deployed

Design: The NEE-class satellites were designed as a 1U cubesat form factor, however, as soon as the initial design was complete, a limitation was discovered in the power budget calculations: A lack of space for enough solar cells, so we decided to add a pair of multi-panel solar arrays[22] or 'wings' to address this deficiency. However this solution required the use of big batteries, bigger than any battery array ever built or available for a 1U cubesat and even for bigger cubesats.

Calculations indicated that we would need a battery of at least 26.64 Watts per bank, and as per our system safety design guidelines the power matrix turned into 4 of this banks, giving a total of 106.56 Watts, the challenge was to pack this much power into an space small enough to fit into a 1U structure.

The design guidelines for the EPS system required:

-Capability of operation without batteries on solar power only.

-EPS capability to manage 32 battery cells distributed in 2 battery arrays for a total of 28.8A or about 107 Watts, with MCU-driven core and 8 solar input power channels each capable of supporting 6V@2A and 25ms switching capability.

-EPS card will manage all the power operations; we only needed a big power tank without any electronics on it.

Many challenges arose from these requirements:

A. Locate Li-Poly cells of the right geometry of length, width and thickness to fit into a PC104 form factor board with a maximum thickness of 7 mm rise per face for a maximum of 20 mm thickness per card.

B. Locate Li-Poly cells of the maximum possible energy density, constrained by the right geometry

C. A containment method to avoid the natural swelling of the Li-Poly cells in vacuum.

D. An affixing technique dependable enough to avoid separation of the cells from the PCB board due swelling.

Testing and Manufacturing:

As it turned out, no cells in the commercial or aerospace market could satisfy the A and B conditions, fortunately we could locate a cell manufacturer in Germany that allowed the design and fabrication of cells with unique or particular geometries: The result was a cell with the following characteristics:

Length: 42 mm.
Width: 19 mm.
Thickness: 5 mm.
Capacity: 900 mAh.

We ordered enough quantity of these cells and determined the right placement geometry, after a few prototypes we could start performance tests, such test results will be detailed ahead in this paper.

At this point we had solved A and B conditions, however, conditions C and D remained unsolved; we realized that there should be a technique that allowed us to solve both conditions at the same time. Encapsulation of the cells in a hard metal case was out of the question due weight and manufacturing considerations; also it will complicate the solution of condition D.

We solved the problem by shrouding each cell in a thin copper layer in order to be able to solder the cells to the copper substrate of the PCB and at the same time the tensile strength of the copper will help greatly to contain the swelling over the minor axis of the cells if we solder each cell to the PCB and then every cell to each other at the same time.

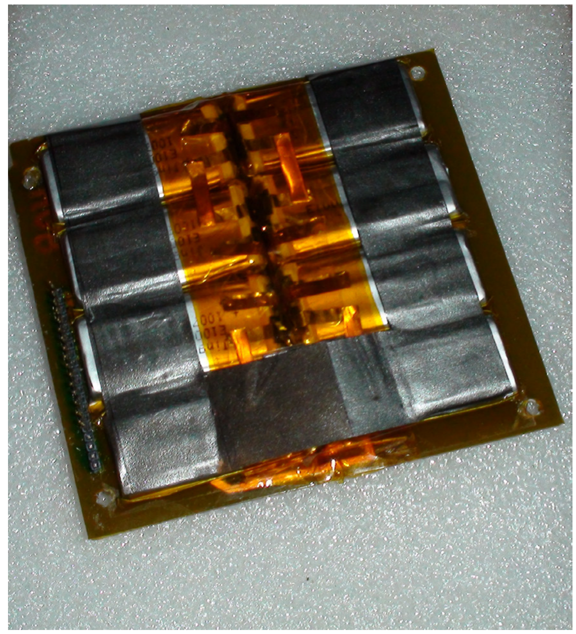


Fig-3 One of the NEE-01 PEGASUS battery array prototypes showing the shroud of carbon nanotubes.

However this technique being simple and easy to implement and solved the 2 remaining conditions, it had to be tested extensively, and it also had the advantage to allow us to use the shroud as a ground terminal for the electrical system of the spacecraft.

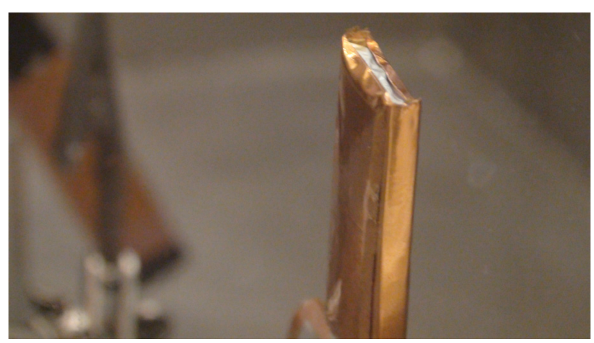


Fig-4 One of the Li-Poly cells being tested in thermal vacuum at 55 C in a 3.6x10E-6 mbar environment in our SESCA thermal vacuum chamber, it shows almost no swelling despite not being affixed to a PCB

After some more tests, it turned out that we needed to add some space grade epoxy to the borders of the seams to prevent any start of de-coupling of the cells from the board, besides soldering the copper shroud to the board itself. The final prototype demonstrates a ruggedness and strength beyond our expectations, being able to withstand crash tests of 300G without us being able to destroy the prototype. Other tests included freezing the cells down to -32C for 48 hours and keep them operating while at it and testing the same operation while being heated by radiating light to 65C for 2 hours. Both tests were successful with no damage detected in the cells, 5 years later, those very same cells are still operating in test benches here on Earth.

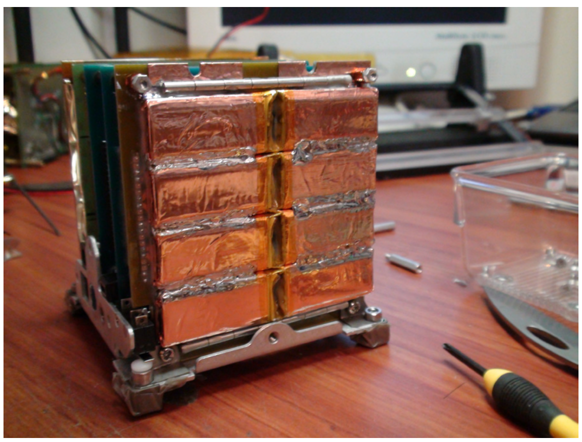


Fig-5 One of the final battery arrays mounted on the NEE-01 engineering model showing the copper shroud the solder and seam structure of the cells,

Each battery array packed 8 cells per side, for a total of 16 cells per pack with a final nominal power deliverance of 3.7V@14.4A = 53.28 Watts, both satellites included 2 of these arrays for a total nominal power of 106.5 Watts of capacity. One of the most important factors in a battery is the ability of not only being able to pack a very high energy density but also being able to deliver it safely, to this goal our interface to the spacecraft bus was made of very high purity copper pins coated with high grade gold, custom made for this project by Samco.

Each battery array had 32 pins so our EPS was able to control the charge of each cell in order to maintain a balance charge of the whole array and at the same time distributing the load on each pin pair

Using the battery arrays as radiation shielding and thermal regulators:

Due the CoM and CoG design of our spacecraft, the battery arrays were mounted in the outer sides of the satellites to achieve a CoM/CoG within the 2 CC of the center of the spacecraft, this indeed had the advantage of acting as last line of defense against radiation, besides the already very effective SEAM/NEMEA[13] anti-radiation and thermal regulation shielding designed for the project, being Lithium only the 3rd element in the periodic table it offered very good radiation shielding capabilities.

Fig-6 Battery arrays mounted on the sides of the payload acting as last line of defense against radiation on the NEE-01 flight model.

It also occurred to us that we could use the already big mass of each array (nearly 200 grams) as a thermal sink for the electronics waste heat and at the same time we will be solving the problem of heating the batteries during eclipse.

To this end, we designed a Carbon Nanotubes Thermal Transfer Bus[13] to route waste heat from the camera, the video transmitter and the OBC and shunt it directly to the batteries; such bus will require us to cover the battery arrays with a thin layer of carbon nanotubes over a layer of graphite applied directly over the copper shroud of the battery arrays and then transfer that heat to the other side of the battery array mass opposite to the side in contact with the bus using the same type of layers.

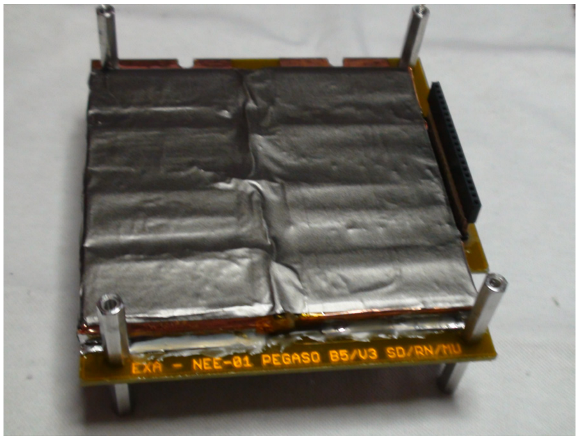


Fig-7 One of the final battery arrays mounted on the NEE-01 flight model showing the CNT=graphite layer over the copper shrouds,

Many thermal vacuum tests confirmed this design to be sound, allowing the array to operate with full payload electronics turned on continuously and no external charge, as a discharge only cycle, with only one battery array, it lasted 4.7 hours sustaining a full load of 3.42V@2.8A = 9.4 Whr before the charge declined to depletion levels as seen in Figure 8.

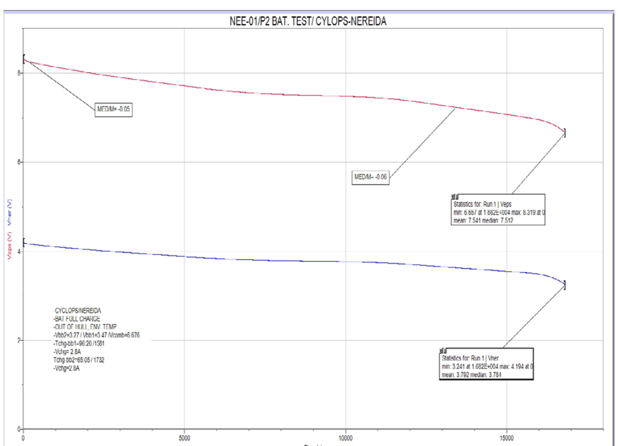


Fig-8 Battery array depletion test, under a continuous load of 9.4Whr lasted 4.7 hours with no external charge. Detailed view in page 11.

To avoid the heat from escaping into space, due laminar distribution and the big radiating surface area of the battery arrays we applied a layer of Kapton-Mylar and at the same time such layer will avoid radiated terminal heat from the exterior panels to leak into the arrays themselves. For the derived constrains when using MLI layers, proper venting had to be performed in the CNT and Kapton-Mylar layers.

Also, performance tests were run for many hours as depicted in Figure-9 for 24 orbits test and in Figure-10 for 16 orbits test.

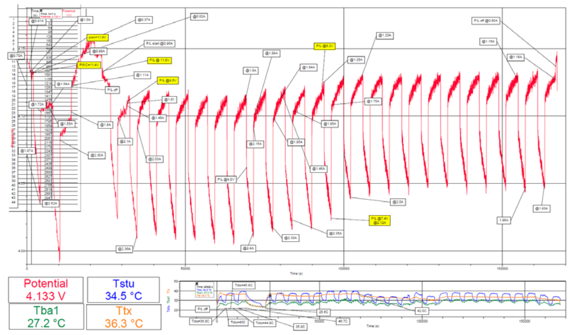


Fig-9 24-orbit performance test on NEE-01 battery arrays integrated with full payload. Detailed view in page 12.

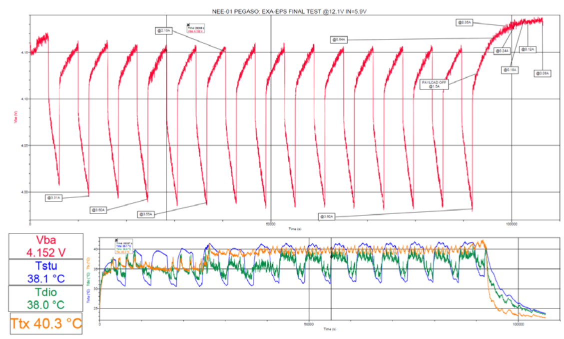


Fig-10 16-orbit performance test on NEE-01 battery arrays integrated with full payload. Detailed view in page 13.

Once completed, more thermal vacuum tests proved that the swelling containment technique worked better than expected as shown in Figures 11 and 12 with negligible or no swelling that could be observed.

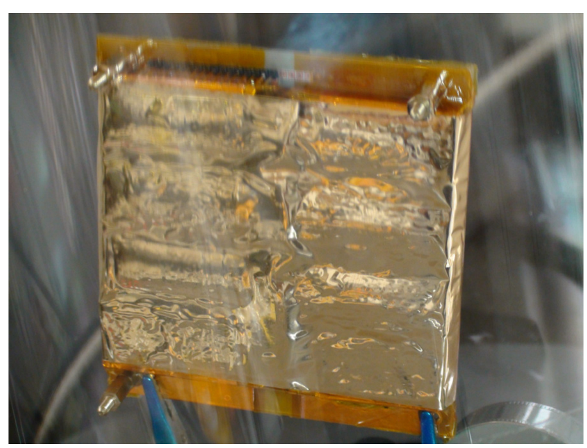


Fig-11 Final battery array in a 3.6x10E-6 mbar vacuum environment showing the Kapton-Mylar external layer

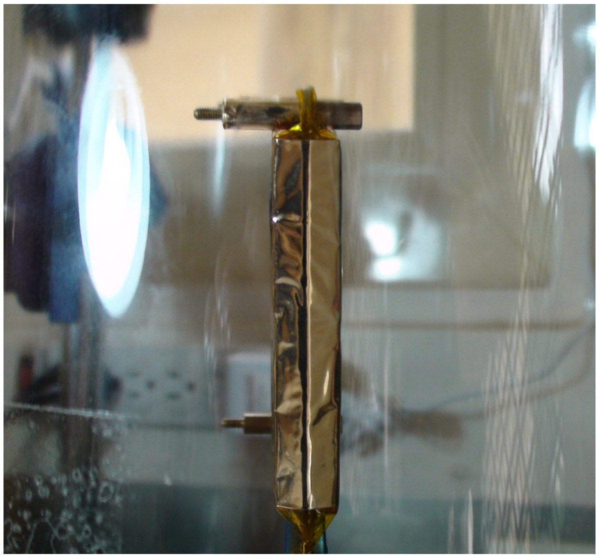


Fig-12 Final battery array in a 3.6x10E-6 mbar vacuum environment showing negligible or no swelling at all.

Final Parameters:

- 53.28 W/hr nominal
- 57.20 W/hr average
- 3.7V nominal, 4.2V max.
- 14.4A nominal
- -30C to +65C (tested)
- Self Discharge Rate of 1.18x10E-4 Volts/hr
- Interface:
  - 32 pins gold over copper
  - 2 mm pin pitch
  - Individual cell charge control
- Mass: 198 grams
- Dimensions: PC104 FF
- Copper / CNT-Graphite / Kapton-Mylar shielding

In-Orbit Testing:

The first satellite, the NEE-01 PEGASUS was launched on April 25[15][18] on a LM2D Chinese vector and worked properly[21] until an in-orbit anomaly with an unknown object caused the spacecraft to lose attitude control[26][27][28][31][32], damaged one of the deployable solar arrays and deformed the main antenna, the satellite survived the event, but remained out of contact and without attitude control for about 6 months until we were able to recover the signal[26][34][37][40], but an attitude control loss also meant that the satellite could not charge the batteries as it could not point the large solar arrays properly to the sun, so the batteries were actually depleted.

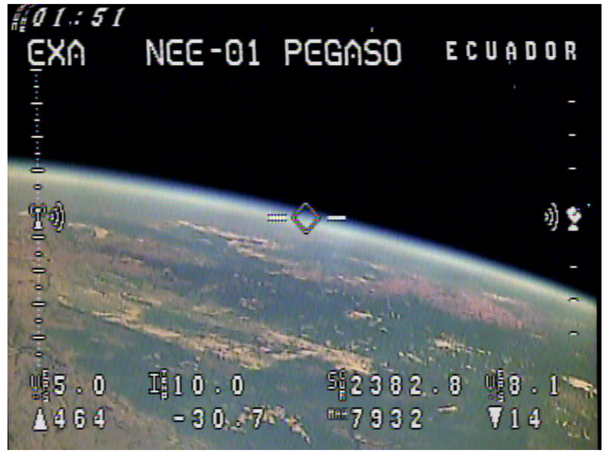


Fig-13 A snapshot of the first public video transmitted by the NEE-01 during May 16 2013, 10h41m pass over South America

Surprisingly for everyone, once the satellite could regain attitude control, the batteries started to charge again, although the antenna damage remained, the spacecraft continued transmitting the beacon as our EPS included a feature that allowed the system to operate on solar power only in the event of the batteries to get depleted beyond recharge.

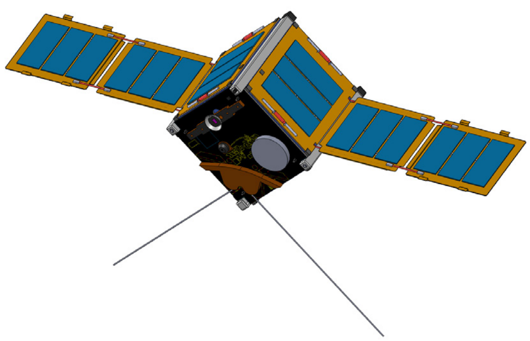
At the same time, the satellite operating on solar power provided the electronics waste heat needed to prevent the batteries from freezing thanks to the thermal transfer bus system thus greatly contributing to the battery life and ulterior recovery

Fig-14 Screen capture of one of the NEE-02 live video transmission in January 25, 2014.

This unintended and harsh test of our battery arrays showed how the rugged engineering worked in ways that we do not foresee at the time of design and build, however, the design guidelines proved successful in arresting problems as serious as the ones we faced during this anomaly.

The New Battery Array Generation:

In March 2016, the EXA was selected by the Irvine Cubesat Stem Program[(14)], a U.S. Public schools based satellite program comprising twelve 1U cubesat launches among 14 years to provide the Deployable Solar Arrays (DSA), titanium infrastructure, NEMEA shielding and battery arrays for this program, the maiden launch of IRVINE01 is to take place in March 2017 to a sun-synchronous orbit. For this specific need we were asked to design a new class of battery array to go in line with a new class of DSA, the new array was designed with the same principles as the original arrays but scaled down to 20Whr of capacity an only one side of the PCB populated with cells and this side had only 6 cells instead of 8.



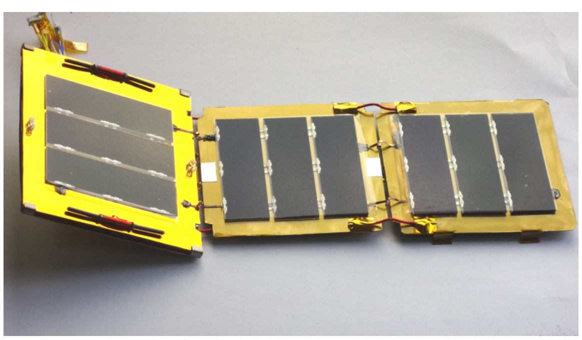


Fig-15 IRVINE01 final design showing the scaled-down DSA arrays in deployed configuration and the actual DSA built for the spacecraft.

This scale-down design was to have only 6.5 mm thickness due IRVINE01 using a new DSA design that only had 2 panels per array instead of the original 3 and the advantage of a high rate discharge port

This new battery arrays were named BA01S being the ‘S’ stand for ‘Slim’ as they were only 6.5mm thick. As in our satellites, these arrays were positioned inside the structure to serve as last line of defense against radiation, however this spacecraft design did not included a thermal transfer bus and therefore this battery array were not equipped with the CNT/Graphite layer needed for this option to work.

Each battery array packed 6 cells on one side only, with a final nominal power deliverance of 3.7V@5.4A (6x0.9A) = 19.98 Watts, the satellite includes 2 of these arrays for a total nominal power of 39.96 Watts of capacity. Actual power is a little more if we take into account the real charge capacity, so it is 4.15V@5.4A (6x0.9A) = 22.41 Watts per array.

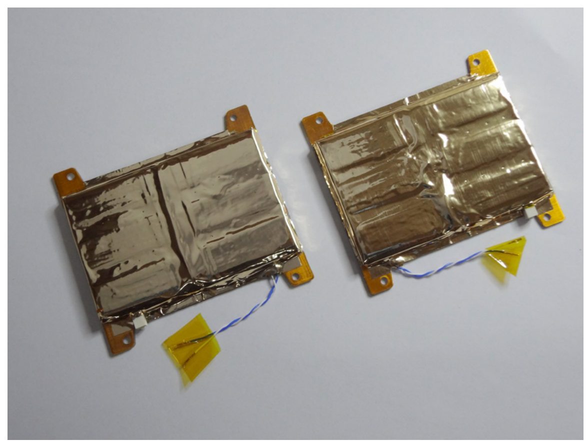


Fig-16 IRVINE01 battery arrays finished and ready for the connectors to be installed.

The batteries passed the following tests:

-Thermal Bake out (10E-7 mbar @ 50C for 24 hours)
-Full vibration test for Dnepr and Long March 2D vibration profiles.

Overheat and Freezing tests as well as Depletion recovery tests were also performed as detailed in Table-1. Crash test were not performed in the flight models of the batteries due obvious reasons, but were done at 300G for the engineering model.

Test	QT	AT
Functional	✓	✓
Vibration	-	✓
Thermal Cycling	-	✓
Thermal Vacuum	-	✓
Overheat test	✓	✓
Depletion recovery	✓	✓
Swelling containment	✓	✓
Freezing Test	✓	✓
Performance	✓	✓

Table-1: Tests performed on the shipping units for IRVINE01

Both battery arrays, along with the full order of products for IRVINE01 were delivered on May 14[(16)] on Irvine, California, U.S.. Right now our DSA Deployable Solar Arrays are available through the CubeSatShop[(20)] website and we plan to make these batteries available to the cubesat market in Q4 2016.

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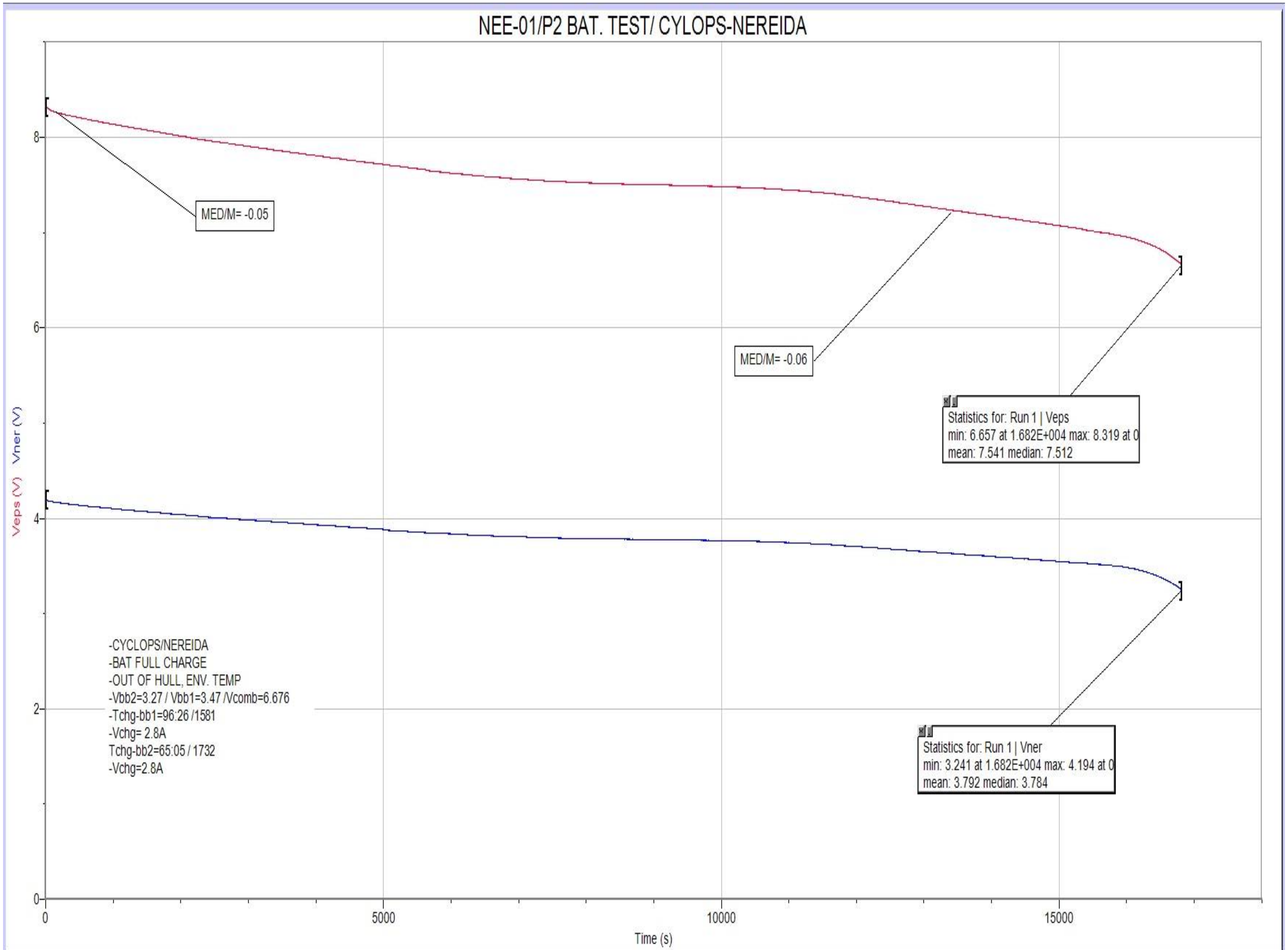
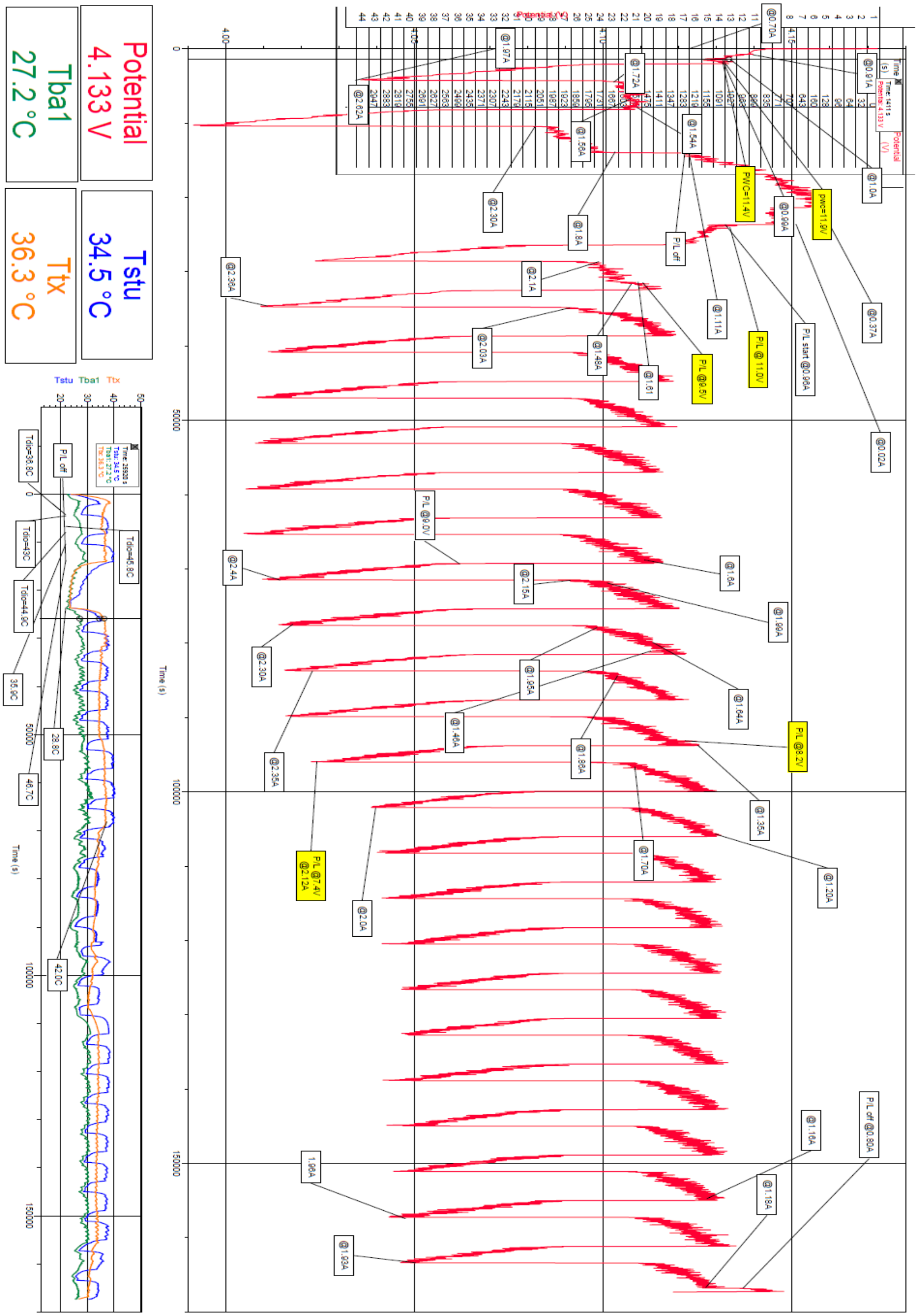
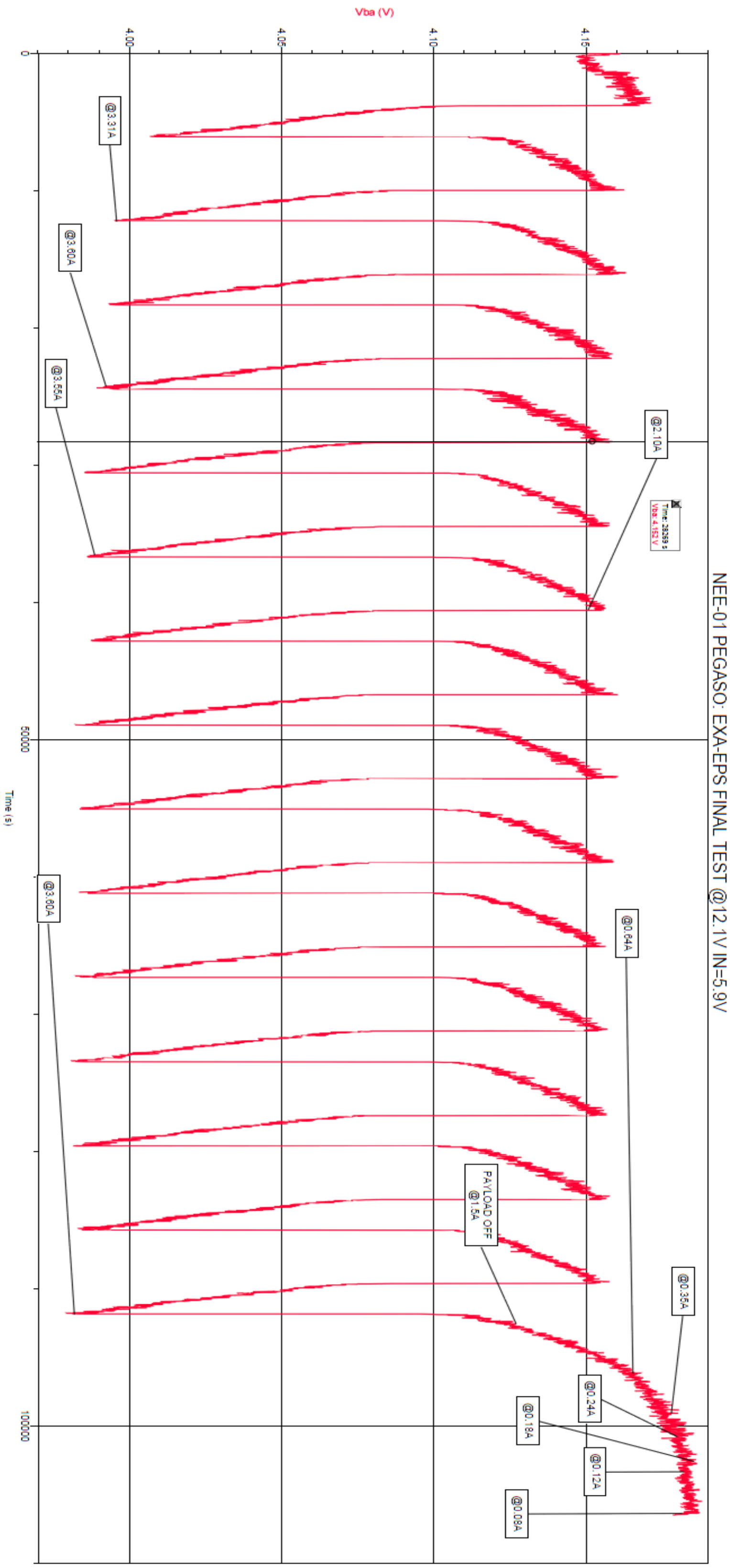
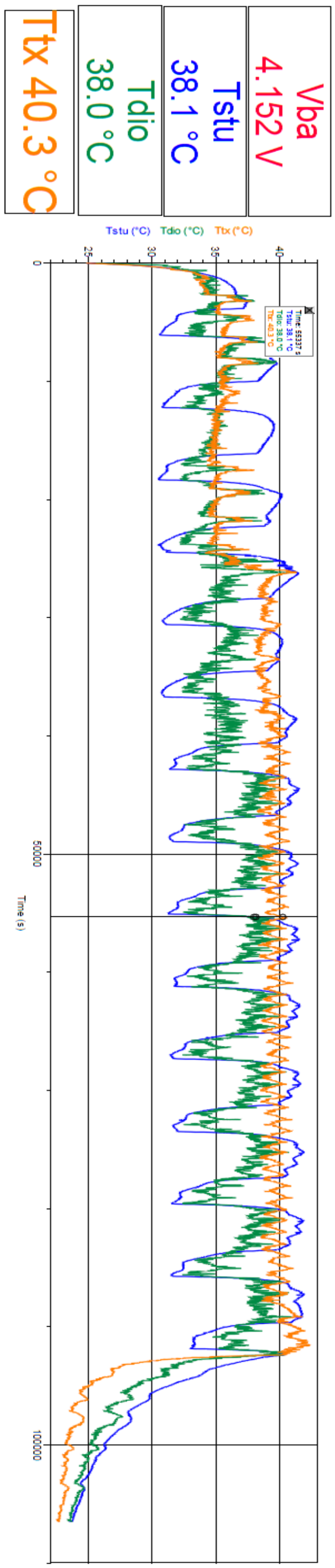
42- “*THE ECUADORIAN EXPERIENCE IN SPACE*”, 65th IAC 2014 – 21st IAA SYMPOSIUM ON SMALL SATELLITE MISSIONS (B4) 15th UN/IAA Workshop on Small Satellite Programmes at the Service of Developing Countries - Nader, Uriguen, Carrion.

43- “*Realities of space debris: the Pegasus satellite*”, Nader, Carrion - "*Space Equities: The Role of the Americas in Building Norms of Behavior*", UNIDIR regional seminar, Mexico city – 2013


NEE-01/P2 BAT. TEST/ CYLOPS-NEREIDA
Veps (V) Vner (V)
Time (s)
MED/M= -0.05
MED/M= -0.06
Statistics for: Run 1 | Veps
min: 6.657 at 1.682E+004 max: 8.319 at 0
mean: 7.541 median: 7.512
Statistics for: Run 1 | Vner
min: 3.241 at 1.682E+004 max: 4.194 at 0
mean: 3.792 median: 3.784
-CYCLOPS/NEREIDA
-BAT FULL CHARGE
-OUT OF HULL, ENV. TEMP
-Vbb2=3.27 / Vbb1=3.47 /Vcomb=6.676
-Tchg-bb1=96:26 /1581
-Vchg= 2.8A
Tchg-bb2=65:05 / 1732
-Vchg=2.8A


Potential
4.133 V
Tstu
34.5 °C
Tba1
27.2 °C
Ttx
36.3 °C
Time (s)


NEE-01 PEGASO: EXA-EPS FINAL TEST @12.1V IN=5.9V
Vba (V)
Time (s)
@3.31A
@3.60A
@3.55A
@2.10A
@3.60A
@0.64A
PAYLOAD OFF @1.5A
@0.35A
@0.24A
@0.18A
@0.12A
@0.08A


Vba 4.152 V
Tstu 38.1 °C
Tdio 38.0 °C
Ttx 40.3 °C
Tstu (°C) Tdio (°C) Ttx (°C)
Time (s)