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DEVELOPMENT AND TEST OF LOW COST SOLAR PANEL TECHNOLOGIES FOR SMALL SATELLITES

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This paper presents the activities carried out in collaboration between the University of Pisa and Alta SpA for the development, testing and integration of an efficient, yet inexpensive photovoltaic panel for microsatellite applications. The approach adopted, aimed at reducing cost and developing "low tech" techniques to assembly and qualify solar panels for small satellite applications, uses a printed circuit board designed to optimize the use of external surfaces partially occupied for power generation, where bare cells are installed by means of a double-sided insulating adhesive tape and each cell is covered with cerium doped borosilicate glass, using a controlled volatility silicone. Bonding was performed with a dedicated vacuum bag technique, developed in-house. This method achieves a significant cost reduction with respect to traditional techniques, while retaining high performance and reliable repeatability and avoiding complex technological procedures during the integration. A prototype solar panel was manufactured, tested and integrated on the UniSat-5 small spacecraft by GAUSS Srl in preparation of a flight scheduled for late 2013. Thorough mechanical testing was performed as a part of the integration with UniSat-5. The panels manufactured during the development programme were subject to electrical characterization to evaluate the current-voltage characteristic curve and the efficiency of the array and to thermal vacuum tests according to ECSS standards to estimate the outgassing properties of the protoflight model. For both tests, a low cost experimental setup was developed on purpose. The recorded flight unit total mass loss (TML) is well under the acceptable limits, so that the panel was accepted for space flight. In-orbit validation of the panel is expected with the upcoming flight of UniSat-5. The techniques and procedures developed under this programme allow for quick and inexpensive manufacture of reliable solar arrays, specially suited for micro- and nano-satellites. To improve the thermal and mechanical properties of the solar array, a substrate in carbon fibre composite laminate is under investigation. A thermal analysis is developed to characterize and compare the thermal response of the solar array with different substrate subjected to space heat flux.

I. INTRODUCTION

For low cost small satellites, solar panels are usually among the most expensive components[1]. Even small improvements in the development, production and qualification process may result in a significant saving for low cost small satellite missions[2]. "Low cost", "low tech" methods to assemble the panel and the development of a flexible and simple array to be mounted on generic small satellites are still open research topics[2].

Our research focuses on the possibility to increase the in orbit performance of low cost solar panels. One possible way to produce more affordable solar panels is to use bare and low-performance cells, but this might lead to modest in-orbit performance, fast degradation and reduced lifetime[3]. This because more than the 70% of the whole panel cost is represented by the photovoltaic cells and the devices needed to protect them in space. The production of covered solar cells is a very critical part, usually performed by specialized companies for the solar cell provider.

With the aim at reducing cost but retaining high performance, the cover glassing of solar cells has been performed by internal means through a "low tech" technique.

Another project goal is to develop a solar panel that can be mounted on a generic small satellites surface. As various payloads and experiments may need to be mounted on the spacecraft external surface, special attention has been given to optimizing the use of external surfaces partially occupied for power generation purposes[2]. The approach adopted uses a printed circuit board (PCB) as substrate of the solar panel and after various iterations towards an optimized disposition of solar cells and other elements on the PCB is possible to meet the electrical and mechanical requirements for the integration of the solar panel with the spacecraft. The other possible alternative to improve the surface available for power generation is to use deployable solar panel with consequent complexity in the spacecraft design[2].

In line with the "low tech" philosophy for the integration of the solar panel, the solar cells were installed on the PCB by means of a double-sided insulating pressure sensitive adhesive tape. This is a “low tech” but very effective method to bond the cells, requiring no curing, high temperatures or specific instruments[4], therefore suitable also for university programs.

The solar panel described in this paper is a protoflight model designed to be mounted on UniSat-5 (Figure 1), a scientific micro satellite designed and manufactured under the coordination of the GAUSS Team at the University of Rome[5]. The spacecraft is approximately a cube of 0.5 m side. UniSat-5 is scheduled to fly in November 2013 in a sun-synchronous 700 km orbit with constrained sun-pointing attitude to maximize the solar panel power production. The surface allocated for our solar panel is already partially occupied by antennas, sensors and the adapter for the Dnepr Launch Vehicle.

As additional development, a carbon fibre panel substrate is currently under development with the aim at reducing the panel weight, improving the mechanical properties of the assembly and allowing for a more uniform thermal distribution for the solar cell operation, so to increase performance. A thermal analysis of the solar array is developed to characterize the thermal response of the array subjected to space heat flux and compare the temperature field of the solar array with the two different substrates.

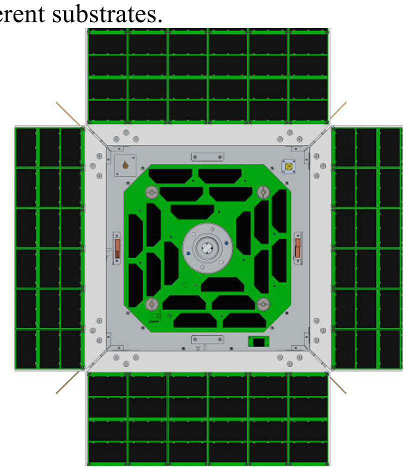

Fig. 1: UniSat-5 with Alta’s solar panel

II. PROJECT AND DESIGN CHOICES

II.I Coverglassing in-house of solar cells

The most critical part of the whole solar panel production, usually performed directly by specialized companies for the solar cell provider, is the application of cover glass to bare cells to improve their durability, radiation resistance and overall performance[2]. The main project goal is to develop a low tech technique for the in-house coverglassing of cells to limit cost and to develop specific know-how. The cell coverage used is a Qioptiq CMG150 glass, 0.15 mm thick, with anti-reflection protection. This borosilicate glass incorporates also cerium oxide that improves the resistance to darkening caused by radiation or ultraviolet light. Since cerium oxide absorbs ultraviolet light, it also prevents the yellowing of the adhesive between the glass and the cell. Yellowing is a mechanism by which the spectrum of light reaching the active part of the solar cell is altered, resulting in reduced efficiency and in degraded power production[2]. The glass is applied by means of a controlled volatility silicone produced by NuSil. This silicone was used for its low volatility and its transparency to a wide range of the electromagnetic spectrum[6]. This glue, while keeping costs low, has also a remarkable resistance to the yellowing caused by absorption of ultra violet radiation. In line with the "low tech" philosophy, after the application of the NuSil silicone on solar cells with a technique developed in-house; a simple vacuum bag system is used for realizing the cover glass. It provides a homogeneous pressure on both solar cells and glasses and reduces the air bubbles formation during the bonding.

II.II Design of printed circuit board

In order to make the maximum use of the surface area available, which is constrained in size and shape by the presence of several other devices, it was chosen to use a PCB as substrate. The PCB provides also the electrical connection between the solar cells and the other electrical components, mechanical support to the panel, and the proper electrical and thermal insulation from the spacecraft. The PCB is made of 2.4 mm thick FR-4, a standard glass-reinforced epoxy laminate (E-glass) used in spacecraft electronics.

The solar cells used for the panel prototype, AzurSpace model TJ Solar Cell 3G30C, are GaInP/GaAs/Ge triple junction solar cells on Ge substrate with efficiency class of 30%. Cells are of trapezoidal shape measuring 2.64 mm x 8 mm x 150 µm and weighting about 2 g each. In nominal conditions, these cells have an open circuit voltage V_{oc} of 2700 mV and a short circuit current I_{sc} of 330 mA. These cells are equipped with an integrated bypass diode which protects adjacent cells in a string against hot spot damage when the cell is shaded. The bottom side of the solar cell is a single extended positive contact. To maximize the reliability of the electrical connections of the cells, the PCB features two positive and two negative contacts in hot redundancy for each cell. An open circuit voltage Voc of 50 V is required to make the panel compatible with the spacecraft power bus. With the solar cells available, this requirement can be met by arranging at least 20 cells in series. For improved reliability, an external bypass diode for each cell and one block diode at the end of all strings (for the whole panel) were implemented.

Further requirements were imposed by the system integrator on size (max 310 mm x 310 mm), height (max 5 mm), mass (less than 500 g) and mechanical interfaces (connection point location). The PCB has five holes to be placed on the interface between the satellite and the launcher adapter and twelve holes to fix the panel to the satellite. The disposition of the elements on the PCB was optimized as shown in Figure 2.

With the aim at reducing costs, the solar cells were electrically connected with the PCB using standard terrestrial solar cell interconnects that consist in a copper conductor with thin solder alloy on both sides[7]. This solution has a much lower cost than the silver coated interconnectors usually found in spacecraft solutions. In addition, the ribbon can be easy procured in different thicknesses and widths[2].

III. SOLAR PANEL INTEGRATION

The solar panel was assembled in a class 100.000 clean room available at Alta's premises. The process started with the soldering of the diodes on the PCB and verifying their functionality. Afterwards, two interconnectors were soldered at the positive contact (rear of the cells) of each cell. So, the NuSil tape was cut at the cell size, excluded the area with positive contacts, and placed on the PCB. After verifying the absence of trapped air bubbles under the tape, the cell was installed in place, then the negative contacts (front of cells), the integrated diode and positive interconnectors were soldered on the PCB[2]. A preliminary electrical check is then performed to ensure absence of short circuits, correct cell behaviour after the soldering and efficient electrical connection between cell and PCB.

Afterwards, the vacuum bag was prepared and sealed (Figure 3). After 24 hours the vacuum bag was opened and the panel cured for 45 minutes at 75°C. A final electrical check was performed. The integration ends with the installation of connection wiring and one J type thermocouple, requested by the spacecraft integrator for monitoring the panel temperature[2]. Figure 2 shows the final result.

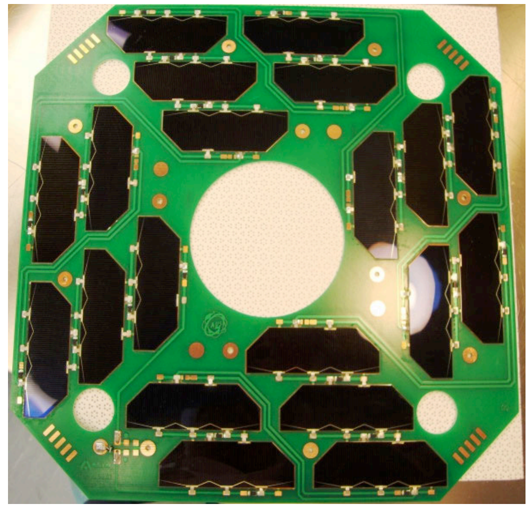

Fig. 2: Alta's solar panel with all components.

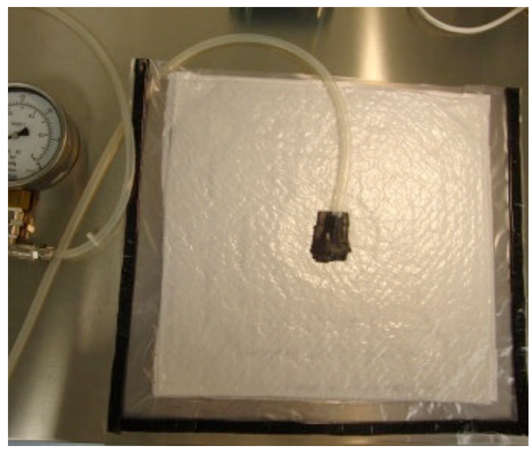

Fig. 3: Vacuum bagging of solar panel.

IV. TEST CAMPAIGN

In this paragraph a description of the main tests performed to qualify the panel is given. The integrator required two main tests: an electrical characterization to verify the correct operating of panel and a thermal vacuum test to evaluate the outgassing behaviour of the panel. The tests here reported, were carried out on two protoflight models, including the flight unit itself. For both tests, a low cost experimental setup was developed on purpose.

IV.I Electrical characterization

The current-voltage (I/V) curve characterizes the solar panel operation points. As no Sun simulator was available and with the aim at containing the testing costs, field tests were conducted with solar light during several sunny days, with a reduced power input with respect to what would be available in orbit. The experimental setup (Figure 4) includes a "low cost" electrical circuit developed in-house to vary the current produced by the solar panel and a pyranometer to measure the solar radiation flux density on the panel, to evaluate its efficiency[2].

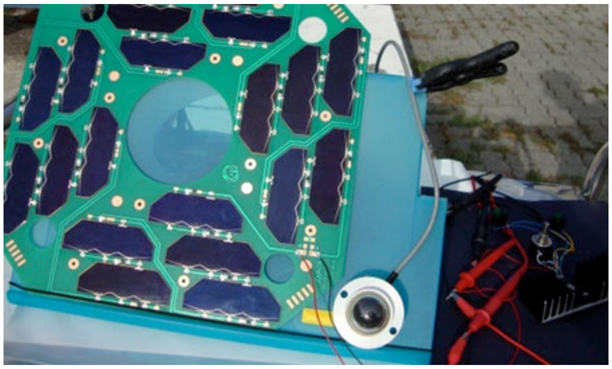

Fig. 4: Experimental setup of electrical test.

The I/V curve of the protoflight model is shown in Figure 5. Special attention was paid to the accurate evaluation of the open circuit voltage that was found to be in compliance with the requirements for the electrical integration with UniSat-5. The values of current and voltage of the whole panel, compared to the behaviour of a single cell characterized under similar conditions in previous internal tests, showed that the methods and techniques development did not damage the solar cells. This is confirmed by an efficiency of conversion of 25% (in ground conditions, quite different from the Air Mass Zero space environment) and by an open circuit voltage Voc of 52 V, in compliance with the electrical requirements imposed by system integrator.

<u>IV.II Thermal vacuum outgassing test</u>

Outgassing is the release of gaseous species from a material and the kinetics of the outgassing process is influenced by vacuum and temperature conditions[8]. This phenomenon is problematic because the released gases can potentially condense on spacecraft components, such as the coverglass of the solar cells, causing degradation of performance and functionality[2].

The standard normative se used as reference for our qualification is the ECSS-Q-ST-70-02[9] .

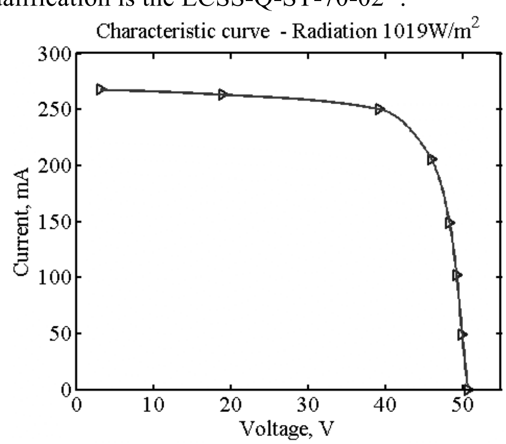

Fig. 5: Curve current-voltage of solar panel.

The test campaign was carried out in one of the high vacuum facility available at Alta (Figure 6), capable to reach an ultimate pressure as low as 10^{-8} mbar. The facility was operated at 10^{-5} mbar according to the normative[2]. The temperature of the panel in the chamber was increased by using four halogen bulbs of 100 W each. The same lamps were also used to illuminate the solar panel to verify the output voltage and then the functionality of the model in vacuum conditions (functional test). A dedicated mechanical structure was fabricated to sustain the bulbs and increase the efficiency of the setup by focusing the bulb light on the panel, avoiding dispersion towards the chamber walls (Figure 7). Temperature of the solar panel was monitored using two type K thermocouples.

The outgassing was measured via a weight check, carried out with a Mettler Toledo WM503-L22 precision balance with 1 mg resolution, before and after the exposition of the panel for 24 hours at a pressure of 10^{-5} mbar and temperature of 125 °C [9].

The recorded total mass loss was then compared with the requirements of the ECSS normative. The initial weight measured was 450.980 g while the final weight was 450.930 g, corresponding to a Total Mass Loss (TML) of 0.013%, well below the limit of 1% imposed. This part of the qualification for flight on UniSat-5 was therefore deemed fully successful.

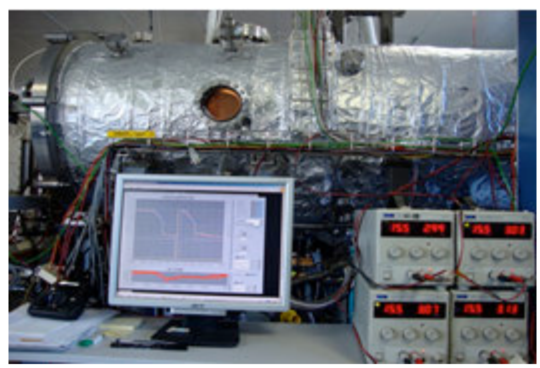

Fig. 6: Alta's IV1 vacuum chamber.

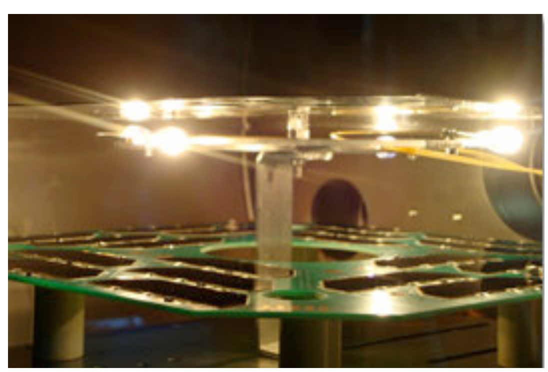

Fig. 7: Heating of the panel during the thermal vacuum test.

V.CARBON FIBRE SUBSTRATE

V.I Substrate in carbon fibre composite laminate

A key goal in spacecraft design is reducing the mass. In particular, for Alta's solar panel 88% of the whole panel weight is represented by substrate (substrate 399 g, whole solar array 450 g).

The recent activities are focussing on the development of a lightweight solar array architecture that improves the characteristics of efficiency, stiffness and thermal behaviour of the panel, still maintaining an approach based on the PCB as mechanical and electrical substrate. To pursue these goals and to increase the mechanical and thermal properties of the panel, the idea is to use the properties of the carbon fibre. Indeed the good properties of this material, such as high thermal conductivity, negative coefficient of thermal expansion, very low density and high tensile module, make carbon fibre a good candidate to realize the future generation of solar panels. Since carbon is electrically conductive, the composite is also electrically conductive. Thus, carbon fibre composite laminate cannot be used as a signal layer. As a consequence we choose to develop a hybrid substrate, where the carbon fibre has the function of mechanical support and thermal dissipation while a thin layer of the FR-4 is used to insulate electrically the copper from the carbon fibre, to realize the required electrical connections.

The facility used for the production of a prototype composite laminate is a press that applies pressure and heat to a package composed by three layers of pre-preg of carbon fibre, one layer of glass fibre and a thin foil of copper (Figure 8). Afterwards with a common technique it is possible to realize the electrical circuit on the substrate. The Figure 9 shows the final composite laminate after the chemical removal of some parts of copper. The thickness of the composite laminate is 1.3 mm.

A preliminary electrical test was performed to verify the electrical isolation between the copper on the top surface of substrate and the carbon fibre composite laminate. Then a preliminary thermal cycling test was performed on the final composite laminate to evaluate the mechanical properties after thermal cycling, in particular macroscopic changes in size and delamination.

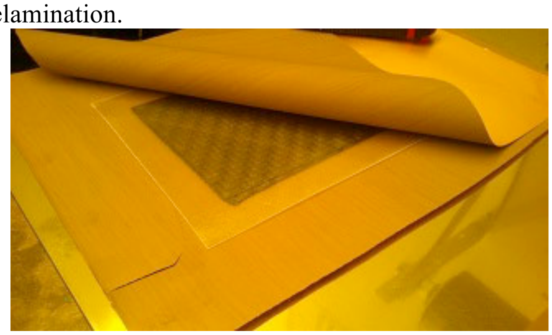

Fig. 8: Preparation of the package.

Fig. 9: The final composite laminate.

The test was carried out in a climatic chamber available at Alta, the Sunrise SU 160 T Version C (Figure 10). The system was cycled for 11 times between -50 °C up to +100 °C with a dwell time of about 5 minutes at -50 °C and about 15 minutes at +100 °C and with a heating temperature rate of 2.5 °C/min and a cooling temperature rate of 1.5 °C/min. A visual check before and after the test confirmed that the composite laminate resists to the thermal cycle with no macroscopic change in size nor delamination.

VI.THERMAL ANALYSIS

In this paragraph a thermal analysis model of the substrate is presented with the aim at characterizing the thermal response of the panel subjected to space heat flux. The finite element method (FEM) is used to calculate the temperature field on a volume mesh of the model.

The preliminary analysis is developed to characterize the thermal response of the solar panel with two different substrate (FR-4, carbon/glass fibre) subjected to space heat flux. The aim is to obtain a qualitative comparison between the substrate in FR-4 and the hybrid carbon-glass fibre composite laminate.

In particular a low Earth orbit (700 km, the UniSat-5 reference orbit[5]) is analyzed. In this orbit the spacecraft makes one complete revolution in about 5900 s (eclipse 2100 s, 35%), so the solar array suffers severe thermal alterations. Sudden heating changes on a surface and on the thickness may induce temperature gradients that generate time-dependent bending moments inducing structural deformations[10].

The thermal environment modelled comprises the Sun radiation (1367 W/m^2), the Earth albedo (410 W/m^2), Earth heat radiation (239 W/m^2) and the heat irradiation of the components[11]. To evaluate the value of these contributes we assume the surface normal is always parallel to the solar radiation vector. Under this assumption we calculate the view factors of an orbiting solar array as a function of its flight altitude. The total external heat flux, which is the sum of solar radiation, Earth-emitted radiation and albedo, reaching the active surface of solar array is shown in Figure 10.

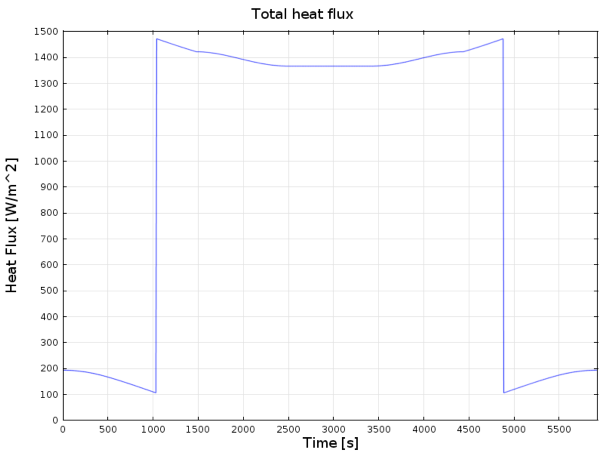

Fig. 10: Total incoming heat flux to the solar array (to be multiplied by the absorptivity of the material).

In additional, the top surface of substrate emits heat radiation toward the deep space, while the bottom surface is assumed to irradiate toward the spacecraft interior (not modeled). Because the thickness of the panel is much smaller than the other dimensions, the thermal exchange between the lateral walls and space is negligible.

The panel is assumed to have been manufactured without any imperfections (such as non-perfect bonding or air bubbles trapped inside the silicon). This assumption allows for considering nominal properties for the conduction matrix among the various elements. The adhesive between solar cells and substrate is sub millimetre thickness and thus not considered in the thermal model. The solar cells have an absorbance of 0.91 and an efficiency of 0.28. Accordingly 63% of incoming radiation causes the solar cells heating. The emissivity of the coverglass (0.90) is taken in account.

Under these assumptions, a FEM analysis is carried out on a solar array of the same shape of the protoflight model, one made in FR-4 with a thickness of 2.4 mm, the other made in carbon/glass fibre composite laminate with a thickness of 1.3 mm. We have chosen this different thickness because the strength-to-weight ratio of the glass fibre is twice that of carbon fibre.

Table 1 summarizes the main properties used for the thermal analysis.

	CTE (ppm/°C)	Thermal conductivity (W/m.k)	Density (kg/m^3)	ε
FR-4 / E-glass	X,Y:16/20 Z:60	0.3	1900	0.7
Carbon fibre	-0.41	X,Y: 24 Z: 1	1700	0.9

Table 1: Main properties of the materials.

Figure 11 shows the temperature field of the top and rear surface of solar array with substrate in FR-4 during the sunlight phase.

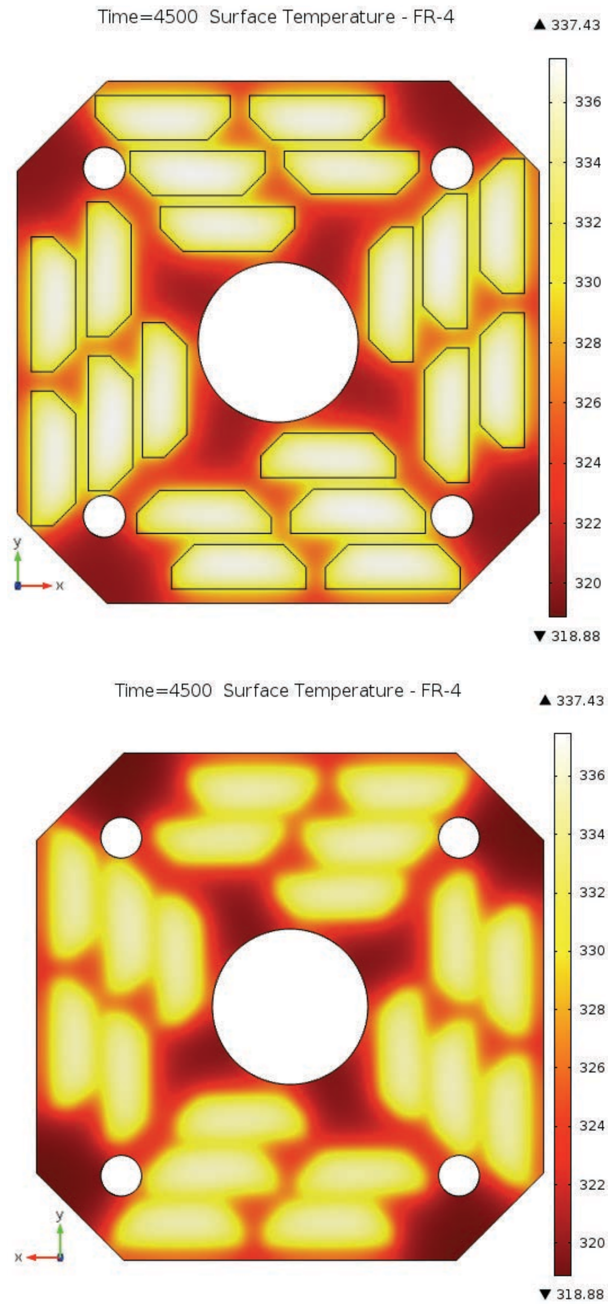

Fig. 11: Temperature field of solar array with substrate in FR-4 during the sunlight phase.

Figure 12 show the temperature field in the same time of the top and rear surface of solar array with substrate in carbon/glasse fibre composite laminate.

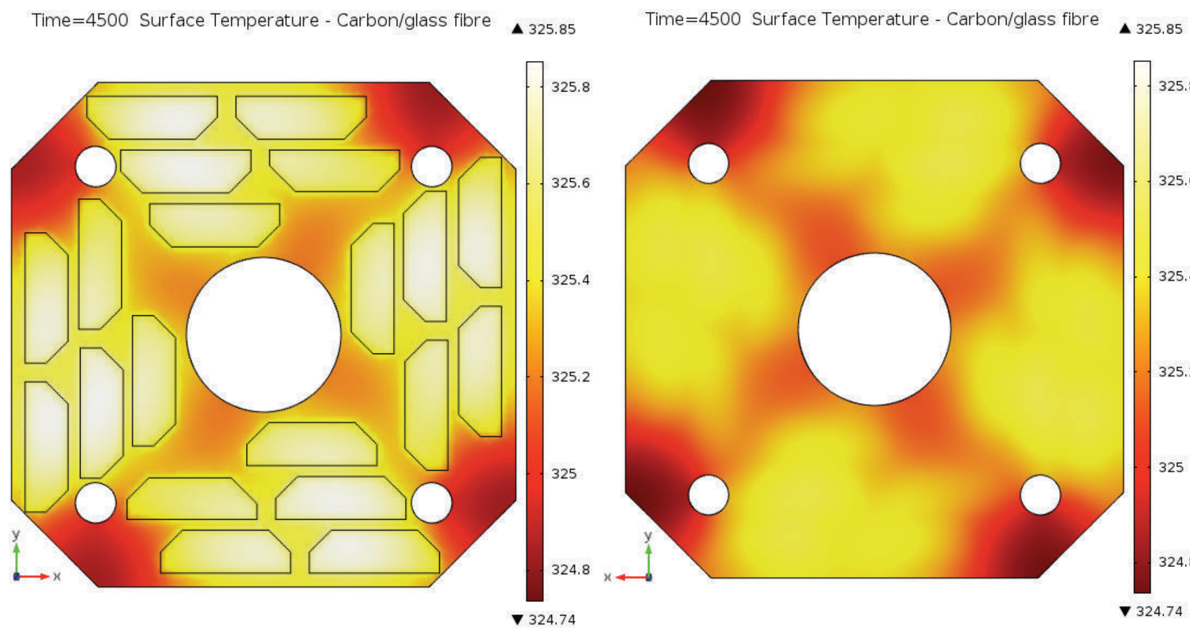

Fig. 12: Temperature field of solar array with substrate in carbon/glass fibre during the sunlight phase.

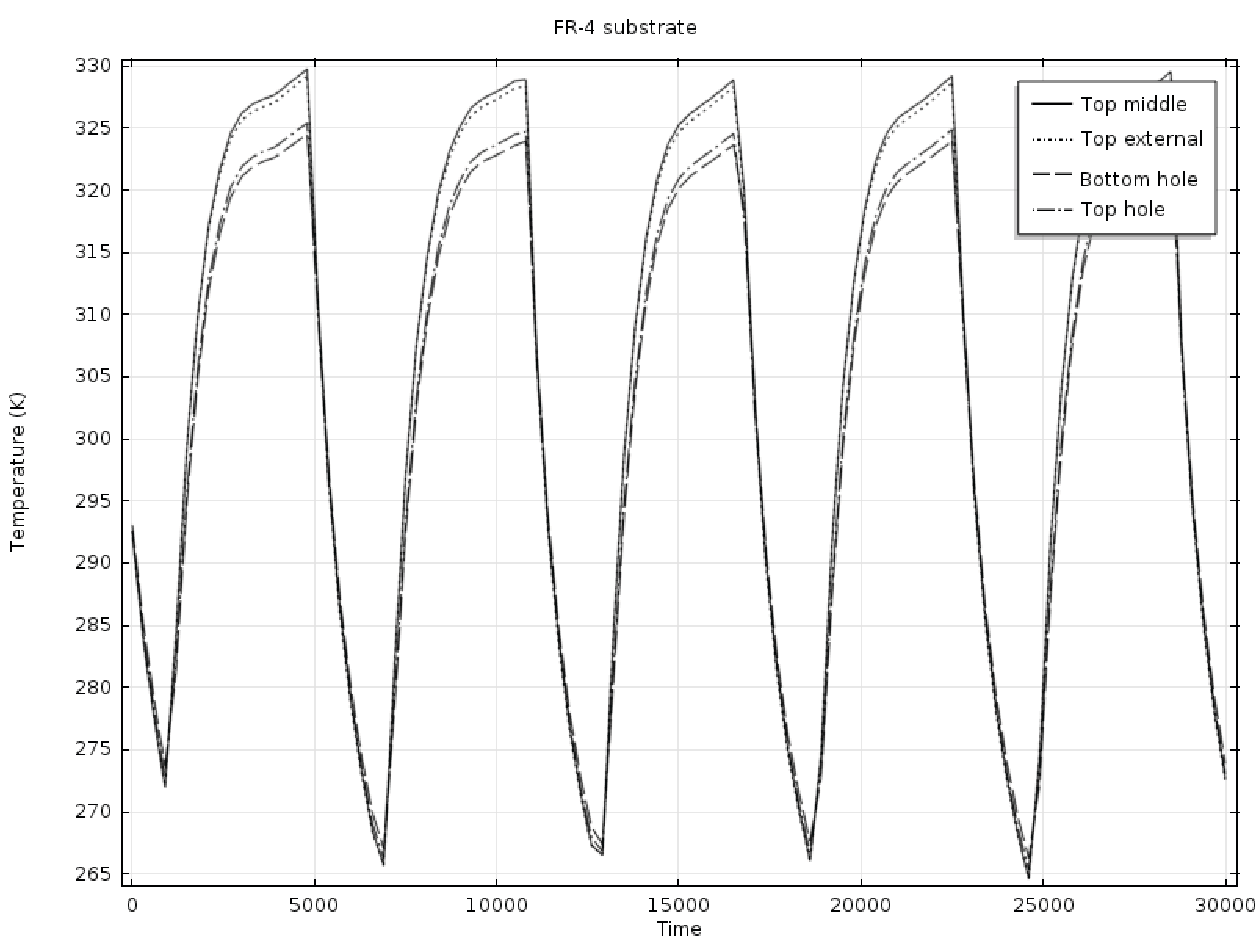

Fig. 13: The temperature vs. time graph of a point in the middle and in external area of solar cells on the top surface of solar array and of a point near a fix hole on the top and bottom surface.

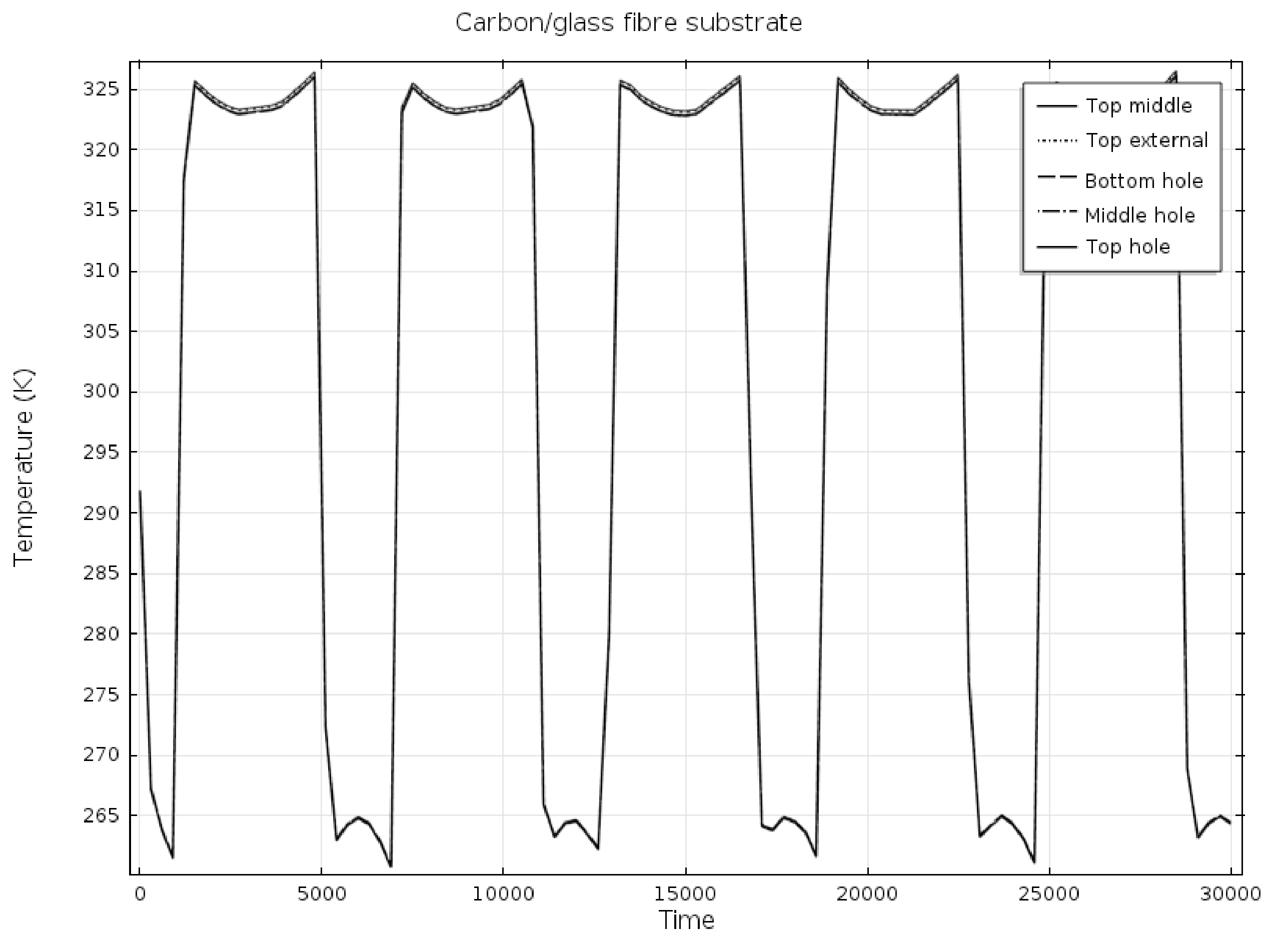

Fig. 14: The temperature vs. time graph of a point in the middle and in external area of solar cells on the top surface of solar array and of a point near a fix hole on the top, bottom and separation surface.

The temperatures in some points on the top face and the bottom surface of the substrate during five periods are show in Figure 13 and in Figure 14. Because of the orbit characteristics, temperature of the solar array varies obviously with the change of the heat flux.

The comparison of the results shows that the use of carbon fibre results in lower extreme temperatures of the substrate, indeed the highest temperature of the substrate in FR-4 in illumination conditions is about 338 K (under the solar cells) on the top surface, while the lowest one in shadow is about 265 K. For the substrate in carbon/glass fiber the same temperatures are about 325 K and 260 K.

The maximum temperature difference between the top and bottom surface is 2 K in the illuminated regions for the FR-4 substrate, while a near zero value is for the carbon/glass fibre substrate.

In additional the main result is the smaller temperature gradient (maximum 1 K in illuminated regions) on the top and bottom surface of carbon/glass fibre substrate. For the FR-4 substrate, the temperature gradient on both surface is about 20 K and it may generate time-dependent bending moments that may induce structural deformations. Also these results are of particular significance for electronic components (e.g. solar cells, sensors), as lower operative temperatures are desirable.

VII. CONCLUSION

A solar panel for small satellite applications has been developed, aimed at low cost and "low tech" manufacturing processes. The project includes solutions potentially improving the overall performance, flexibility and cost of the solar panel, avoiding the need to use dedicated, expensive technologies or facilities.

We have acquired the know-how and the techniques for the development and the construction of solar panels with a repetitive and reliable procedure. We have also acquired the competencies about the process for qualify solar panel in according to standard normative, with a dedicated low cost experimental setup.

The techniques for the application of cover glass have provided very positive results, with a definite cost saving. The approach of a PCB as substrate of panel allow us to optimize the use of surface partially occupied that aren't accessible for the traditional solar panel, with an improvement of the power budget available to the satellite. The electrical and mechanical requirements for the integration of Alta's solar panel on board the UniSat-5 spacecraft have been met.

From our test campaign, the flight unit total mass loss is well below the limit stated by the ECSS. Accordingly, the protoflight model results to be compliant with the requirements and qualified for space flight.

In additional, a carbon fibre panel substrate is currently under development with the aim to reduce the panel weight, improve the mechanical properties of the assembly and allow for a more uniform thermal distribution. For the protoflight model that was realized in this project, the use of carbon fibre composite laminate as substrate might reduce the weight of the whole solar array by about 48%. We have verified the advantages offered by the carbon fibre on the temperature distribution of the substrate by means of a dedicated thermal analysis. The know-how related to the production processes of composite laminates has been also gained.

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