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An innovative deployable solar panel system for Cubesats

Fabio Santoni [a,*], Fabrizio Piergentili [b], Serena Donati [a], Massimo Perelli [c,1], Andrea Negri [c,1], Michele Marino [c,1]

[a] Dipartimento di Ingegneria Astronautica, Elettrica ed Energetica – DIAEE, Università di Roma "la Sapienza", Via Salaria 851, 00138 Rome, Italy
[b] Dipartimento di Ingegneria Meccanica e Aerospaziale – DIMA, Università di Roma "la Sapienza", Via Eudossiana 18, 00184 Rome, Italy
[c] IMT Srl, Via Carlo Bartolomeo Piazza 8, 00161 Rome, Italy

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ABSTRACT

One of the main Cubesat bus limitations is the available on-board power. The maximum power obtained using body mounted solar panels and advanced triple junction solar cells on a triple unit Cubesat is typically less than 10 W. The Cubesat performance and the mission scenario opened to these small satellite systems could be greatly enhanced by an increase of the available power. This paper describes the design and realization of a modular deployable solar panel system for Cubesats, consisting of a modular hinge and spring system that can be potentially used on-board single (1U), double(2U), triple (3U) and six units (6U) Cubesats. The size of each solar panels is the size of a lateral Cubesat surface. The system developed is the basis for a SADA (Solar Array Drive Assembly), in which a maneuvering capability is added to the deployed solar array in order to follow the apparent motion of the sun. The system design trade-off is discussed, comparing different deployment concepts and architectures, leading to the final selection for the modular design. A prototype of the system has been realized for a 3U Cubesat, consisting of two deployable solar panel systems, made of three solar panels each, for a total of six deployed solar panels. The deployment system is based on a plastic fiber wire and thermal cutters, guaranteeing a suitable level of reliability. A test-bed for the solar panel deployment testing has been developed, supporting the solar array during deployment reproducing the dynamical situation in orbit. The results of the deployment system testing are discussed, including the design and realization of the test-bed, the mechanical stress given to the solar cells by the deployment accelerations and the overall system performance. The maximum power delivered by the system is about 50.4 W BOL, greatly enhancing the present Cubesat solar array performance.

1. Introduction

The development of small satellites, characterized by the involvement of small personnel teams, very short time from concept to launch and limited on-board resources, requires non-traditional design approaches [1–4]. This article is the result of a cooperation established between the Laboratorio di Sistemi Aerospaziali of University of Rome "La Sapienza" and IMT Srl, Ingegneria Marketing Tecnologia, with the aim to complement the academic and industrial expertise and skills in the development of a nanosatellite subsystem. This experience follows a previous initiative undertaken in the framework of the education and outreach program Edusat [5,6].

* Corresponding author.
E-mail addresses: fabio.santoni@uniroma1.it (F. Santoni), fabrizio.piergentili@uniroma1.it (F. Piergentili), sere.dona@gmail.com (S. Donati), massimo.perelli@imtsrl.it (M. Perelli), andrea.negri@imtsrl.it (A. Negri), michele.marino@imtsrl.it (M. Marino).
[1] http://www.imtsrl.it

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The diffusion of the Cubesat nanosatellite standard bus, described in [7], has made access to space affordable by lowering the launch cost through the use of specifically developed nanospacecraft dispensers allowing for multiple spacecraft launches [8,9]. This gave impulse to on-board component miniaturization and in particular to the development of standardized components and innovative missions [10–13].

The Cubesats capabilities could be greatly enhanced by increasing the available on-board power, while maintaining the compactness and volume limitations imposed by the standard. Technologies and methods for efficient power generation and storage on-board micro and nano-spacecraft have been sought, including the possibility of using commercial solar panels in space [14–17], commercial Li-ion batteries [18,19] and ways to maximize solar array power output by efficient interfacing with the load and batteries [20]. Concerning Cubesat power systems, the available surface for body mounted solar panels is so low that there are no alternatives to using high efficiency triple junction solar cells. In this case, the typical maximum delivered power is in the order of 10 W for triple Cubesats. The main advantage of body mounted solar panels is that no particular attitude pointing is required. However, the most performing Cubesats developed recently, including the 1U, 2 U and 3U size, can take advantage of current state of the art miniaturized attitude control systems [21], guaranteeing accurate pointing and maneuvering capabilities, as required by typical high performance and power demanding, missions.

Deployable solar arrays have been developed for micro and nano-spacecraft in order to improve the on-board power generation capability (e.g. [17,22]). Some have been tested in orbit and are commercially available as a standard “building block” for newly developed Cubesat systems. These systems are based on many different deployment geometries and mechanisms and some have been developed having in mind particular satellite orbital and attitude stabilization scenarios. The simplest systems are based on single deployable solar arrays, connected to the main satellite body by one single hinge (e.g. [17]). In more complex configurations, the system is based on a number of interconnected solar panels [22–26].

A performance parameter expressing the solar array capabilities that can be used in general, not being related to a particular mission geometry, is the maximum delivered power, when the sun is orthogonal to the solar array in free space (AM0). The increase of on-board power generated by deployable solar panel with respect to body fixed ones is significant. Several deployable solar panel solutions for 3U Cubesat have been developed, with total delivered power ranging from 22 W to 56 W (e.g. [22–26]), increasing the performance with respect to body mounted solar panels from 160% to 400% [27].

A broad classification of these systems can be performed based on the deployment geometry and solar panel's orientation once deployed. Three categories have been determined based on this classification: (i) the deployment involves single solar panels, which are connected to the satellite body by one single hinge; (ii) the deployment involves a number of solar panels connected in a chain and one solar panel is connected to the satellite body by one single hinge, without the possibility of steering the solar array once deployed; (iii) the deployment involves one or more solar panels connected in a chain and one solar panel is connected to the satellite body by a system of two hinges, allowing to rotate the solar panel system relative to the satellite body once the solar panel is deployed. The system described in this paper is of type (ii), but it has been designed as a building block of a SADA (Solar Array Drive Assembly).

The deployable solar array system described in this paper has been dimensioned for the maximum obtainable solar array power, compatible with the standard Cubesat dimensions and Cubesat dispenser limitations. As far as the single solar panel size is concerned, the obvious choice is to make it as large as the Cubesat lateral face. The number of solar panels is limited by the maximum number of staked solar panels that can fit between the Cubesat structure and the dispenser in the stowed configuration during launch. The typical solar cell, wires and hinge thickness allow in practice only for three stacked solar panels in the ISISPOD dispenser [28], or two solar panels in the P-POD dispenser [8]. A study on the deployment mechanics of this system can be found in [29].

The solar panel performance strongly depends on the relative orientation of the sun and the orbit, as well as on the attitude motion, as discussed in Section 2. The solar panel deployment system mechanical and electrical designs are depicted in Section 3 and the prototype realization and test is dealt with in Section 4.

2. Geometrical configuration

The geometrical configuration has been selected based on orbit and attitude stabilization considerations and on the deployment sequence reliability and impact on the future SADA system and other satellite subsystems.

2.1. Attitude and orbit configuration

The overall solar array performance in orbit depends strongly on the history of the angle of the sun direction with respect to the solar panel normal and on the total eclipse time. These are strictly related to the particular mission configuration, including orbital dynamics and attitude stabilization. Hence it is not possible to determine one solar array configuration which is optimal for all missions. However design drivers can be indicated by trade-off analysis referred to the most frequent orbital configurations of past Cubesat missions and to typical attitude stabilization of high performance Cubesats, based on the assumption that installing high performance steerable solar array on-board is profitable in particular for three axis stabilized spacecraft.

The first design analysis is devoted to comparing the performance of fixed and steerable deployed solar panels. The analysis is limited to circular sun-synchronous orbits, in which the relative motion of the sun with respect to the orbital plane can be described by one single parameter, the local time of ascending node (LT). We also assume the satellite attitude is nadir pointing, which is the most common among Cubesat missions.

2.2. Body-fixed and steerable deployed solar panels

The solar array performance in terms of average delivered power in one orbit depends on the relative motion of the sun with respect to the solar panel normal. The solar panel orientation of body fixed deployed solar arrays can be selected in principle in order to maximize the average power, including eclipse times.

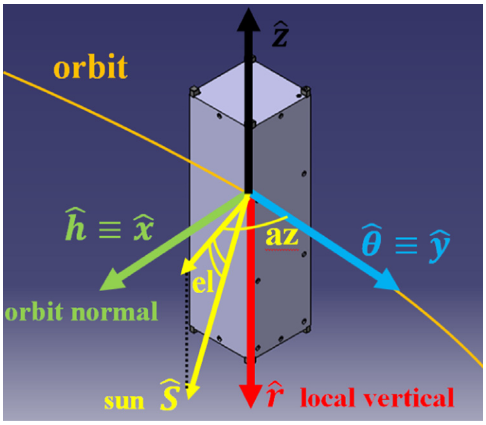

Fig. 1. Sun direction angles in the body fixed reference frame of a nominally oriented nadir pointing Cubesat.

The solution to this problem for nadir pointing Cubesats includes two limiting cases with a straightforward solution. The first one is when the sun is directed along the orbit normal. This situation, in which there is no eclipse is representative of sun synchronous early morning (dawn–dusk) orbits (LT=6:00 or LT=18:00). In this case the optimal orientation of the solar panel normal is along the orbit normal, so that the solar array is fully illuminated all the time. The second one is when the sun is in the orbital plane. This situation, in which there is the maximum possible eclipse time at the given altitude, is representative of sun synchronous noon/midnight orbits (LT=0:00 or LT=12:00). In this case the optimal configuration is having the solar panel normal directed upwards along the local vertical. Although the configuration is optimal for the given orbital and attitude geometry, the generated power is much less than the previous case, because the solar panel is illuminated only half orbit and because of the sun angle effect along the orbit.

In general, the sun moves along a cone in the body reference frame, whose aperture can be considered constant in one orbit, as shown in Fig. 2, with non-negligible seasonal variations due to the inclination of the ecliptic with respect to the Earth's polar axis. In these general cases, the optimal solar array orientation, taking into account eclipse times, can be obtained only numerically [30,31].

The solar array performance could be enhanced, depending on the orbital geometry, if the solar panels were allowed to rotate about a body-fixed axis.

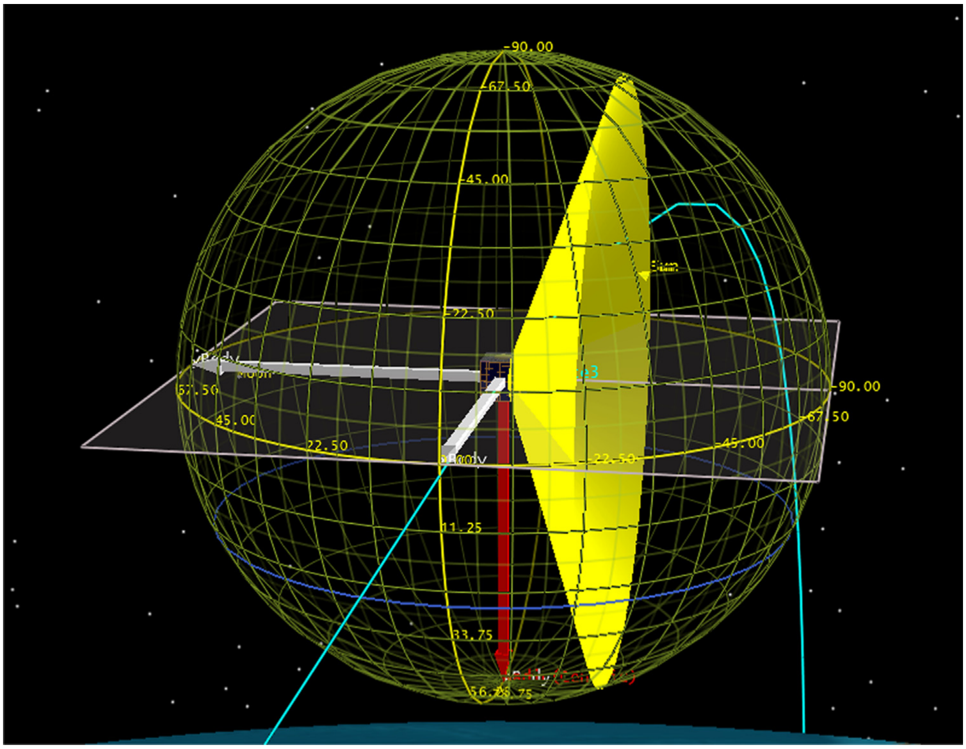

Fig. 2. Motion of the sun in the satellite greference frame for nadir pointin Cubesat, in a sun-synchronous orbit.

A comparison of the power generated by body-fixed and steerable deployed solar arrays in terms of percentage of solar power increase of steerable solar panels with respect to body-fixed is shown in Fig. 3, as a function of ascending node local time. It is assumed that no yaw maneuvers are performed, maintaining exactly the attitude geometry indicated in Fig. 1. The comparison is based on the average total power over one year. As expected, the maximum improvement is obtained with steerable solar panels, as high as 134%, is obtained in noon/midnight orbits, in which the attitude motion does not allow to maintain the solar arrays exposed to the maximum sun illumination. On the contrary, in dawn/dusk orbits no relevant improvements (2%) are observed.

This result indicates that in some situations, the average solar array performance can be more than doubled by using a steerable solar panel system.

2.3. Deployed solar panel configuration

The solar array size is mainly determined by geometrical constraints and Cubesat dimensions, not leaving significant design margins for the single panel configuration. Hence, the main design trade-off are in the overall solar panel system geometrical configuration and deployment architecture. The two selected configurations for design trade-off are shown in Fig. 4. In the longitudinal deployment configuration the solar panels are connected on the side parallel to the main hinge (the one connecting the solar panel system to the satellite main structure). In the lateral deployment configuration the solar panels are connected on the side perpendicular to the main hinge.

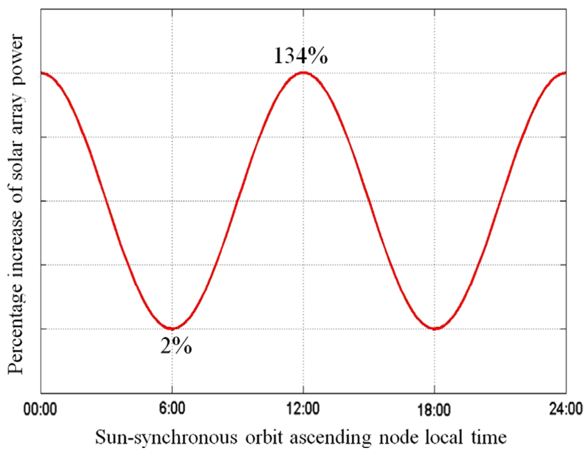

Fig. 3. Comparison of performance of body-fixed and steerable deployed solar panels for nadir pointing Cubesats in sun-synchronous orbits.

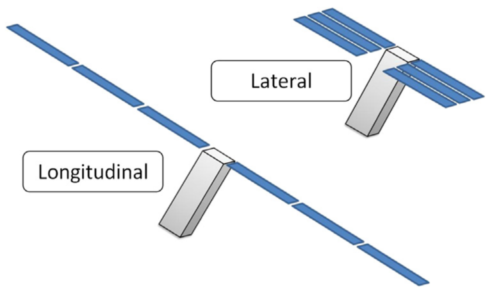

Fig. 4. Deployed solar panels' geometrical configuration.

The longitudinal configuration has been selected based on superior performance with respect to the lateral, concerning the following system aspects:

- Less disturbance torque during deployment, because of symmetry.
- Less moment of inertia of the deployed solar panel about the steering axis (SADA motor size and continuous operation power).
- Less disturbance on the satellite attitude due to SADA operation and solar panel motion.
- No possibility of solar panel impacts on the structure, in case of malfunctioning during the deployment phase.
- Larger space for installation of secondary hinges, considering that standard triple junction solar cells cover the solar panel almost completely widthwise.
- Scalability to 1U and 2U Cubesats.

The main drawback associated with the longitudinal configuration is the larger increase in the overall satellite moment of inertia, which might negatively affect attitude maneuvers.

2.4. Deployment sequence

The deployable solar panel consists of three solar panels, stacked on each other in the stowed configuration. The deployment sequence develops in two steps, as shown in Fig. 5. First the solar array pack deploys from the Cubesat body, when the thermal cut TC1 is activated. Then the pack opens, when the thermal cut TC2 is activated, releasing the solar panels in the final configuration.

The main hinge, connecting the solar panels assembly to the satellite structure (red in Fig. 5) must allow (and limit) the rotation of the whole assembly by 90°, from the stowed to the first step of the deployment sequence. The secondary hinges, connecting the solar panels each other (green in Fig. 6) must allow (and limit) the rotation of the solar panels by 180°.

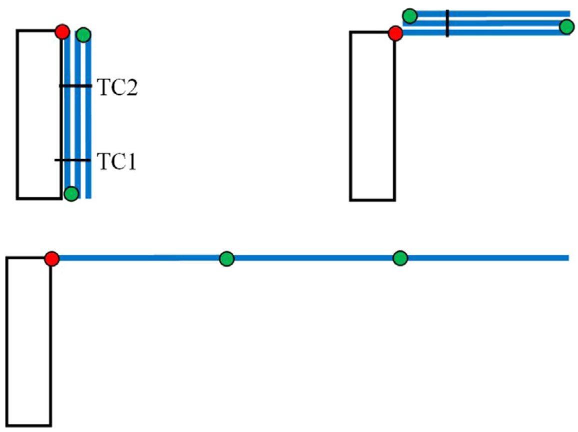

Fig. 5. The two phases of the deployment sequence.

The solar panel assembly during and after deployment is shown in Fig. 6.

3. Deployable solar panel design

3.1. Requirements

The deployable solar panel system has been designed based on the overall geometrical configuration selected above and taking into account of the main system requirements imposed by the Cubesat structure, Cubesat dispenser, commercial advanced triple junction solar cell size and thickness, mechanical and electrical interfaces. No requirements have been fixed a-priori concerning the system mass. The goal is to have the system as light as possible, compatible with the other requirements.

An additional requirement is that the system should be installed externally from standard Cubesat structure without modifications.

Taking the system geometrical constraints and the typical high performance triple junctions solar cells dimensions into account [32], a string of seven solar cells fits in each solar panel. According to [32], the maximum power delivered by one solar cell at AM0, room temperature of 28 °C, is about 1.2 W. Hence the solar array system is capable of delivering about 50 W in the same environmental operative conditions.

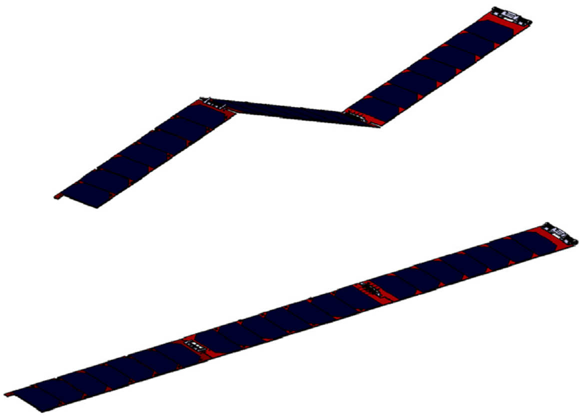

Fig. 6. Solar panel assembly during and after deployment.

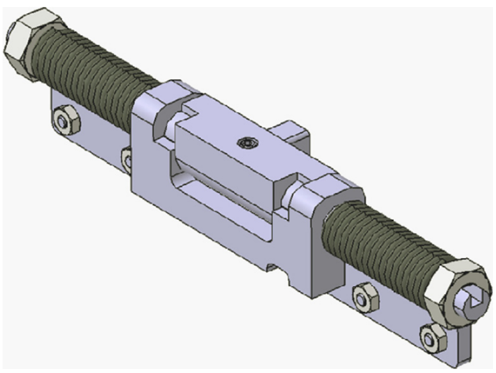

Fig. 7. Main hinge assembly (front view).

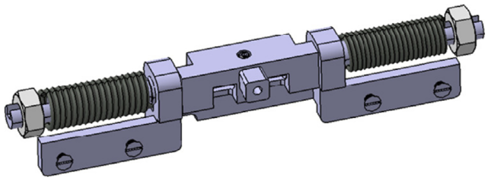

Fig. 8. Main hinge assembly (back view).

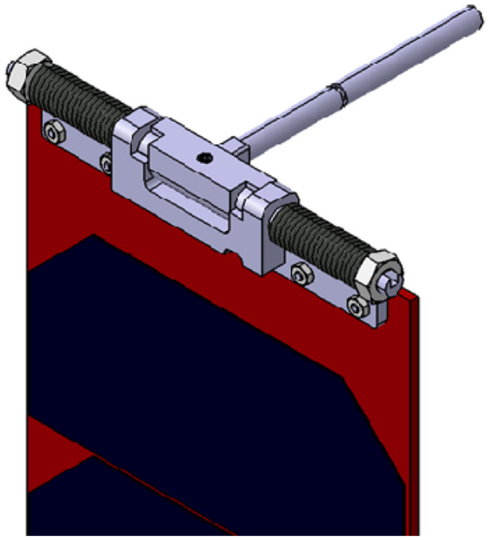

Fig. 9. Main hinge and solar array assembly (stowed).

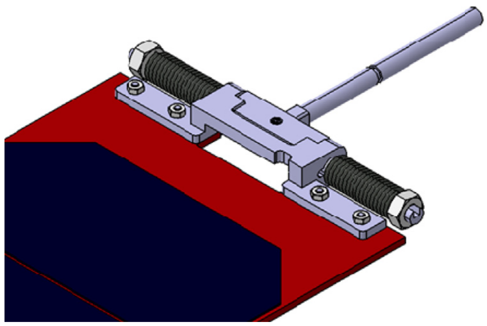

Fig. 10. Main hinge and solar array assembly (deployed).

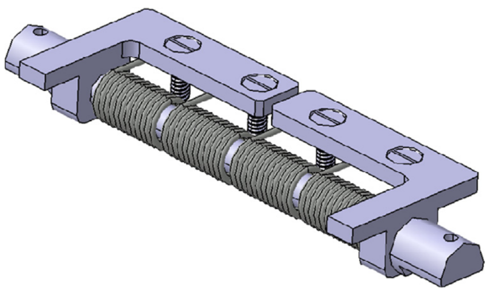

Fig. 11. Secondary hinge assembly.

3.2. Mechanical design

The system mechanical design has been focused mainly on the hinges connecting the solar panels and on the dimensioning of the springs for deployment.

The selected material for the solar panels is standard electronic board FR4, which has been preferred to aluminum because of its thermal stability, flexibility in the electrical interface design, realization and solar cell integration.

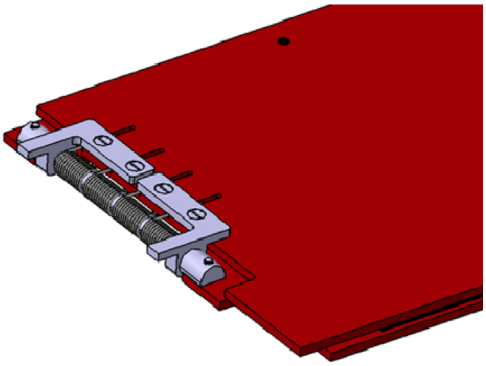

Fig. 12. Secondary hinge mounted on the solar panels in stowed configuration.

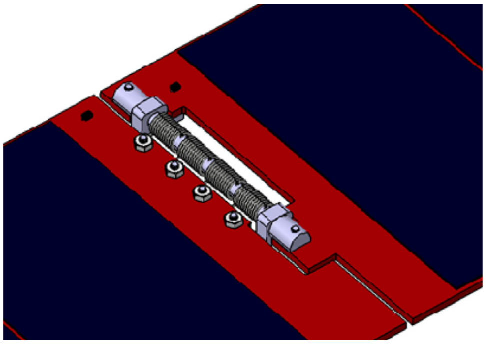

Fig. 13. Secondary hinge assembly mounted on the solar panels in deployed configuration.

The main hinge has been designed to connect the solar panel assembly to the SADA shaft in one single connection point. The hinge supports two torsion springs, as shown in Figs. 7 and 8 The solar panel is connected to the main hinge by four screws, as shown in Figs. 9 and 10. Here the SADA shaft connected to the main hinge is also shown.

The secondary hinge assembly, based on the same principle, is shown in Fig. 11. This hinge has a stop hinge functionality, allowing for a rotation of 180°. The stop function is provided by the hinge support shape, as shown in Fig. 12 in the stowed configuration. The hinge support connected to the upper panel has two small protruding elements, which limit the rotation of the lower panel in the correct open configuration. The lower solar panel FR4 plate gets direct contact with the stop elements made of aluminum. The deployment spring energy was calibrated for a sufficiently soft deployment, as discussed in [29]. As a result the FR4 plate has the ability to sustain the shock completely. The solar panels in the deployed configuration are shown in Fig. 13.

3.3. System assembly

The system assembly consists of two solar panels packs, comprising three solar panels each. The assembly of one pack in the stowed configuration is shown in Fig. 14, including the main external dimensions. These are compatible with the standard Cubesat dispenser ISIPOD [28].

The packs installed on the Cubesat structure are shown in Fig. 15. The shaft connecting the two packs is representative of the SADA shaft.

4. Prototype integration and test

A prototype of the system, assembled as shown in Fig. 15, has been realized and used for the deployment sequence ground testing.

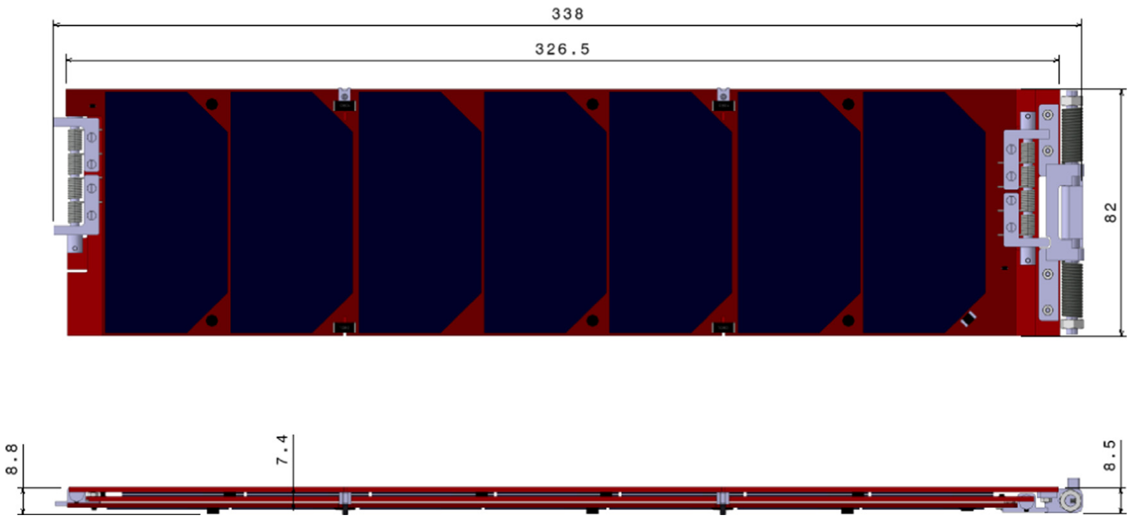

Fig. 14. Solar panel pack assembly in stowed configuration. Dimensions in millimeter.

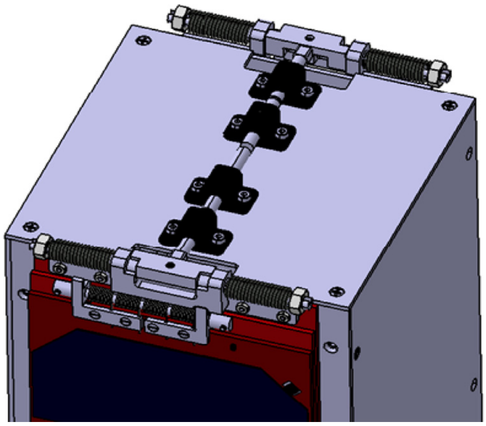

Fig. 15. Installation of the solar panel packs on the Cubesat structure in the stowed configuration.

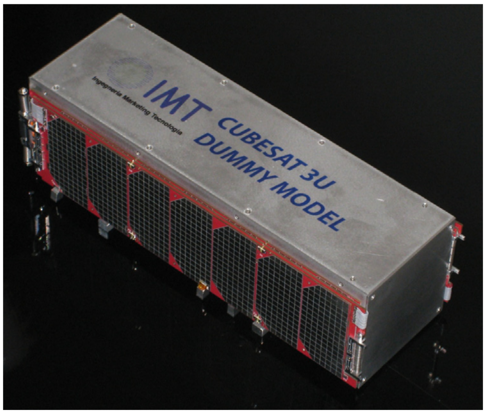

Fig. 16. Assembled satellite mock-up structure and solar array packs in the stowed configuration.

The prototype includes six solar panels assembled in two packs made of three solar panels each, as described above, and a dummy structure of a 3U Cubesat respecting all of the standard dimensions concerning the external shape. The assembled prototype ready for testing is shown in Fig. 16.

A special equipment set has been developed for the solar panel deployment sequence testing. The hinges and the springs are dimensioned for operation in a microgravity environment. Hence a deployment sequence in a regular gravity environment would not be representative of real conditions in orbit. To release the solar panels' weight from the hinges, a low friction support has been developed, based on small lubricated steel spheres located in appropriate spherical holes machined in the support. A steel ball and its support are shown in Fig. 17. To enhance the low friction motion, a special lubricated horizontal table has been employed. The overall view of the testing equipment and solar panel after deployment are shown in Fig. 18.

The deployment sequence has been performed several times, with no failures. Only slight changes in the sequence duration have been observed, due to the changing level of friction in the low friction support.

Fig. 17. Low friction support for solar panel deployment testing.

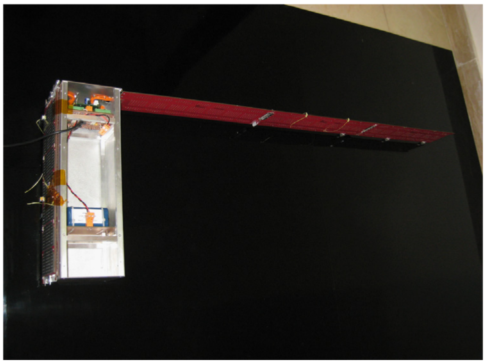

Fig. 18. Low friction lubricated horizontal support, dummy 3U Cubesat structure, testing equipment and solar panel after deployment.

5. Conclusions

A deployable solar panel system for Cubesats has been developed in the framework of a cooperation established between Laboratorio di Sistemi Aerospaziali of University od Rome "la Sapienza" and IMTsrl, Ingegneria Marketing Tecnologia. The system has been developed for 3U Cubesats, but it is conceived to be scalable to all Cubesat shape factors, including 6U, 2U and 1U Cubesats, respecting all of the geometrical constraints given by the Cubesat standard and by the deployment system. The system can be installed externally from standard Cubesat structures, without modifications. A prototype of the system has been developed, ready to be installed on a SADA (Solar Array Drive Assembly). The ground testing of the deployment sequence has been performed, with no failures or major problems.

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