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An orientable solar panel system for nanospacecraft ☆

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ABSTRACT

An orientable deployed solar array system for 1–5 kg weight nanospacecraft is described, enhancing the achievable performance of these typically power-limited systems. The system is based on a deployable solar panel system, previously developed with cooperation between Laboratorio di Sistemi Aerospaziali of University of Roma "la Sapienza" and the company IMT (Ingegneria Marketing Tecnologia). The system proposed is a modular one, and suitable in principle for the 1U, 2U and 3U standard Cubesat bus, even if the need for three axis attitude stabilization makes it typically preferred for 3U Cubesats. The size of each solar panel is the size of a lateral Cubesat surface. A single degree of freedom maneuvering capability is given to the deployed solar array, in order to follow the apparent motion of the sun as close as possible, given the mission requirements on the spacecraft attitude. Considerable effort has been devoted to design the system compatible with the Cubesat standard, being mounted outside on the external spacecraft structure, without requiring modifications on the standard prescriptions. The small available volume is the major constraint, which forces to use miniaturized electric motor technology. The system design trade-off is discussed, leading to the selection of an architecture based on two independently steerable solar array wings.

1. Introduction

The maximum delivered power of body-mounted photovoltaic systems is typically below 10 W in nanospacecraft-sized satellites [1–10]. This is a strong limitation for these kind of systems. Innovative and high performance, power intensive applications, such as electric propulsion, opening new scenarios for nanospacecraft systems, require high performance power systems. On the other hand, the current state of the art attitude control systems, developed for nanospacecraft applications, are able to guarantee accurate pointing capabilities [11–13]. Hence, there is a solid basis for the development of deployable steerable solar panels for this class of satellites. The system proposed in this paper is based on a deployable solar panel system previously developed with cooperation between Laboratorio di Sistemi Aerospaziali of University of Roma "la Sapienza" and company IMT (Ingegneria Marketing

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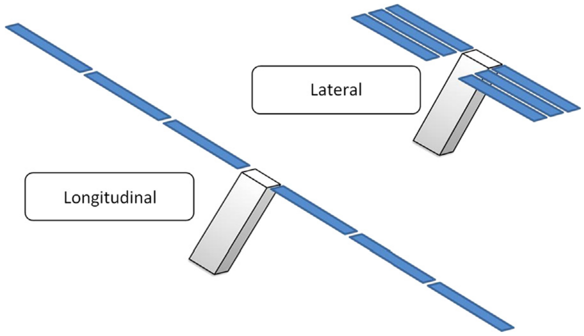

Fig. 1. Deployed solar panels' geometrical configuration.

Tecnologia), as described in [14,15]. The aim of this co-operation is to further develop the experience gained in the development of student-made nanosatellites at University of Rome "La Sapienza" [16–19], with the contribution of the expertise from the industrial environment. As an example, the microsatellite developed in the framework of the education and outreach program Edusat, represented a fruitful cooperation between academic and industrial partners [20,21].

Several deployable solar panel concepts have been proposed for Cubesats and some realizations have been implemented and tested in orbit (e.g. [22–26]). Most of these are based on the deployment of one single solar panel from the satellite's main body, increasing the area exposed to the sun and enhancing the geometrical performance, in terms of an average sun angle along the orbit. Some of these systems involve the deployment of multiple solar panels, with the aim of increasing the available surface for solar cells. The main limitation of these systems is that solar panels' orientation remains fixed with the satellite body orientation. Hence, when the satellite orientation is dictated by the payload requirements, for example nadir or inertial pointing, the solar panel orientation might be far from direct sun-pointing. A relevant improvement could be obtained providing the deployed solar panels with at least one rotational degree of freedom with respect to the satellite. For example nadir pointing Cubesats in sun-synchronous orbits could benefit from a rotating solar array deployed along the normal orbit and constantly pointed to the sun. Whereas this concept is commonplace in large satellites, for example geostationary satellites, where solar array drive systems (SADA) are used to maintain solar panels' orientation fixed to the sun while the satellite is nadir pointed, its extension to nanospacecraft must cope with the strong mass, volume and power limitations, which reflects the difficulty in obtaining the appropriate mechanical as well as electrical connection between the solar panel and the satellite body. A SADA system for Cubesats has been proposed in [27], in which the motion control is based on one electric motor controlling two deployed solar panel wings. This architecture, while saving space and power, has the main drawback that in case of one solar panel wing deployment failure, the whole system is blocked.

The availability of industrial grade miniaturized electric motors fitting with the dimensions constraints and torque requirements and the system operation are described in Section 2, which suggest to explore a more flexible architecture, based on two motors, as discussed in Section 3. The system has been developed according to the Cubesat standard [28], following a modular approach as much as possible. No modifications to the standard are required to install the system on a hosting Cubesat. The solar panel orientation management strategy and the sun angle measurement are described in Sections 4 and 5. The deployed and orientable solar panel concept and the related technological solutions are also applicable to non-standardized nanosatellites. Solar panel current and voltage measurements are provided to the host satellite upon request, as described in Sections 6 and 7.

2. System configuration and operational modes

The deployed solar panels system comprises two deployable Solar Panel Wings (SPW) and a SADA (Solar Array Drive Assembly), which sets the solar panel wings angular position, relative to the satellite.

The longitudinal configuration was preferred with respect to the lateral configuration, as shown in Fig. 1, mainly with the aim to improve the system modularity and scalability to different Cubesat sizes. In addition, the longitudinal configuration exhibits the minimum moment of inertia after deployment, so that there is less disturbance on the satellite attitude, once the system is deployed and operational.

The solar array deployment system, described in detail in [14,15], is based on a modular hinge system, allowing for connection of any number of panels using the same hardware. The retain mechanism, keeping the satellites in a stored configuration during launch, is based on a polymer wire and a thermal cutter. Each solar panel wing is connected to the SADA by a drive shaft, allowing for one degree of freedom angular motion, as shown in Fig. 2. As discussed in [15], two solar panel wings are deployed at

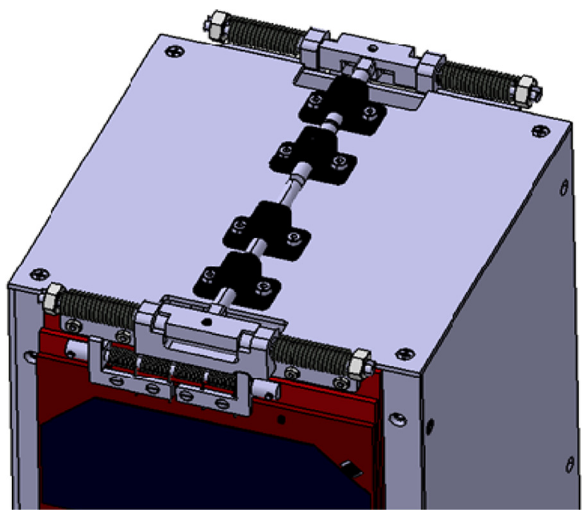

Fig. 2. Installation of solar panel wings on the Cubesat structure in the stowed configuration (SADA not installed). Each solar panel wing is connected to the structure by a shaft allowing for one degree of freedom angular motion.

Table 1
System operational modes.

ID	Operational mode
Before deployment	
OMB1	Wait for deployment command 1
OMB2	Deployment of SPW 1
OMB3	SPW1 autotest
OMB4	Wait for deployment command 2
OMB5	Deployment of SPW 2
OMB6	SPW2 autotest
After nominal deployment of both SPWs	
OMN1	Low power stand-by–motors OFF. Maintain solar array position relative to satellite
OMN2	GoTo commanded position
OMN3	Execute a scheduled sequence of orientations
OMN4	Autonomous sun tracking (one axis)
OMN5	Complete autotest
After failure of one SPW deployment	
OMF1	Failed SPW in LP stand-by. Deployed SPW in LP stand-by
OMF2	Failed SPW in LP stand-by. Deployed SPW GoTo commanded position
OMF3	Failed SPW in LP stand-by. Deployed SPW execute scheduled sequence of orientations
OMF4	Failed SPW in LP stand-by. Deployed SPW autonomous sun tracking (one axis)
OMF5	Deployed SPW complete autotest
After failure of both SPW deployment	
OMFF	Both SPW in LP stand-by

different times, so that the full battery current is available for every single thermal cutter operation.

2.1. Operational modes

The system's operational modes can be grouped in two main groups: the first refers to the deployment maneuver sequence, the second to the solar array system operation after deployment. In case of deployment failure of one solar SPW, the failed SPW is left in a Low Power (LP) mode, with the motor off. In case of deployment failure of both SPW, both motors are turned off. Table 1 lists the system operational modes.

It is expected that the hosting satellite, which must have three axis attitude control capability, has the ability to measure the sun's position and can detect the situation of a sun eclipse. A sun sensor can be installed on top of the SADA system, if required by the hosting satellite for the functionality of the satellite attitude control system. The SADA itself does not require the presence of a dedicated sun sensor, but it does have the capability to detect and measure the sun's position about the SPW shaft axis and autonomously track the sun's apparent motion about this axis, if required by the hosting satellite.

3. Motion control unit

The MCU (Motion Control Unit), which is the central part of the SADA system, includes the hardware necessary for the rotation of the solar arrays, once the deployment is complete. The MCU mainly consists of electric motors and related controllers, including power regulation and command/data exchange with the hosting satellite. The functional block diagram of the MCU is shown in Fig. 3.

One of the main design trade-off for the MCU is the selection of the number of motors. Using one single motor for controlling two SPW's angular positions, might be the simplest solution, involving one motor control electronics and one position measurement system. However, in case of failure of deployment of one SPW, also the other SPW remains stuck in a fixed position, completely losing the capability of orienting the solar panels, leading to a complete solar panel system failure. On the other hand, using two motors improves the system reliability by mitigating the risk of losing the mission in case of failure of one SPW deployment, at the cost of adding the complication of having two motors and two controllers in a small volume.

Concerning the motor technology, there are basically two options: brushless DC motor or stepper motor. From a functional point of view, the stepper motor represents the easiest solution. Once the starting position is known, without any further sensor, it is always possible to determine the position of the SPW simply by counting the number of steps performed by the motor. Applying a low current is also possible to generate a retaining torque capable of maintaining the panels in position in presence of disturbances, like satellite attitude or orbital maneuvers. The control circuit is more complex, but the lack of further circuits make this solution the simplest.

The other possibility is to use a brushless motor, paired with an encoder. The control circuit is simpler but it requires more components, due to the need of controlling three phases. Special care must be taken in reading the encoder output and the starting point, since the harness connecting the solar panels to the spacecraft moves with the panels themselves. It is mandatory to maintain the full SPW rotation angle within one full turn, to avoid the risk of cables being rolled upon the panels support and damaged.

The availability of extremely compact COTS (commercial off the shelf) industrial grade stepper motors and planetary gearheads, with diameters as low as 5 mm and total length (motor+gearheads) less than 4.5 cm, led us to the selection of the two-motors architecture. Stepper motors were preferred to brushless motors, which require an absolute encoder to measure the position. Using stepper motors, the encoder can be eliminated by counting the motor steps. However, this architecture requires particular care in case of loss of the step counts for any reasons, such as a system reset. To manage this situation, a mechanical limit switch can be added to the system, obtaining a reference “zero” angle. In this situation, which is not likely to happen often in the mission's lifetime, the solar panels are commanded to perform a full rotation towards the limit switch. Of course, the shaft will rotate only by the initial offset angle, stopping in the “zero” position. In this way the reference angle can be recovered.

The configuration selected implements a two motors system. The motors can be operated independently, in case one solar panel wing fails to open. In this way, control over the deployed wing is maintained. The installation of two stepper motors and gearheads within the SADA box

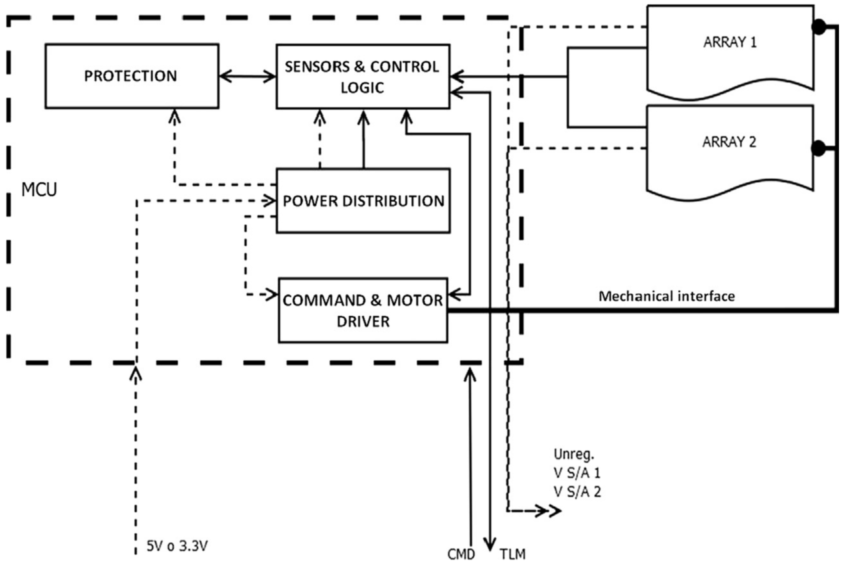

Fig. 3. SADA system architecture.

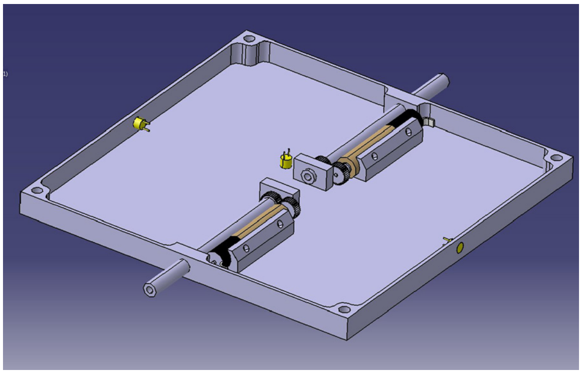

Fig. 4. Two stepper motors and planetary gearheads installed in the SADA box.

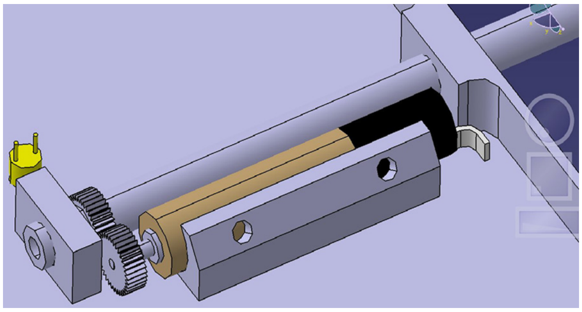

Fig. 5. Enlargement of one motor and planetary gearheads, showing spur gears and the motor fixing system.

(6mm × 100mm × 100mm) is shown in Fig. 4. The small size of motors and planetary gearheads leaves enough space to install control boards, sun sensors, and limiting switch.

Fig. 5 shows an enlargement of one motor installation, showing the spur gear coupling with the SPW shaft. This kind of mechanical coupling was selected for two main reasons: i) the SPW structural loads are transferred to a quite solid support, a shaft with external diameter of about 4.5 mm; ii) the shaft can be made a hollow shaft, so that the internal part of the shaft, which is actually a tube, can be used to pass electrical connection wires from the SADA box to the SPW.

Concerning the SPW shaft support, the design trade-off between ball bearings suspension and lubricated suspension without ball bearings is in favor of this second choice. Industrial grade ball bearings, as small as 4 mm in diameter, are available commercially. However, the internal diameter, which is in the order of 2 mm, forces one to use a stainless steel solid cylindrical shaft. In this way, a hollow shaft, useful for passing electric wires, cannot be installed.

The evaluation of the expected lifetime when operating in a vacuum at an extended temperature range is not trivial. The number of turns expected during the whole satellite lifetime depends on the mission profile and pointing requirements. A typical small satellite mission orientation is the nadir-pointing, in which the solar panels rotate by one turn per orbit back and forth. Assuming 15 orbit turns per day, typical of low earth orbits, the solar panels perform about 5500 cycles/year. With a planetary gearheads reduction of 1:64, one gets about 350 K cycles/year for the motor shaft. For a lifetime of several years in low Earth orbit, the system should therefore be dimensioned for a number of cycles in the order of 5×10^6.

The MCU lifetime is limited by the wear in gears and bearings. The boundary lubrication regime (all asperity contact) is expected, considering that the operation of the system is typical of instrument gears, with very low loads and slow motion. Analytical as well as numerical models for wear prediction are available (e.g. [29,30]). However, in the boundary lubrication regime, lifetime predictions are not well-supported analytically and it is recommended [31] to conduct life testing experiments, which may be accelerated in this case. A rough evaluation of the expected

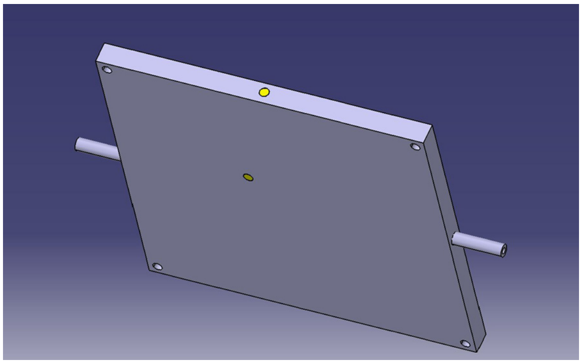

Fig. 6. Installation of sun sensing elements (yellow dots) on the SADA box. (For interpretation of the references to color in this figure legend, the reader is referred to the web version of this article.)

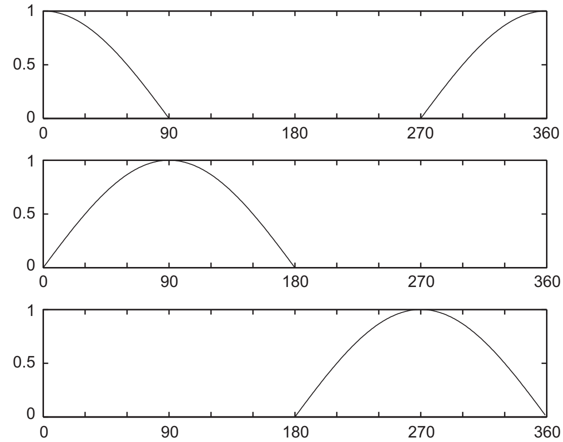

Fig. 7. Normalized sun-light sensors output as a function of the sun-angle (degrees), measures from the top surface normal.

performance, to be confirmed by appropriate testing, can be obtained by considering that industrial grade micromotors and gearheads are available commercially with both extended operative temperature range (-30 to $100\,^\circ$C) and dry lubrication for operation in a vacuum [32], hence the operation in orbit falls within these parts' nominal operational envelope. Detailed lifetime data for stepper motor and planetary gearheads are not available, but the lifetime of brushed micromotors is reported to exceed 1000 h [32]. This limit is mainly due to the motor brushes wear, suggesting that brushless motors and other system parts, such as gearheads and bearings, have lifetimes at least in the order of 1000 h as well. Assuming a typical operating speed of 1000 rpm, one obtains a lifetime of 60×10^6 cycles, which fulfils largely the MCU lifetime requirements. Therefore, though appropriate testing cycles in a relevant environment are mandatory to confirm functional performance and lifetime, no major failures or criticalities are expected, provided that parts and lubricants for operation in a vacuum have been selected.

4. Solar panel orientation management strategy

The solar panel orientation management strategy is aimed at maximizing the power output from solar arrays. Because of the relative motion of the sun with respect to the satellite, some form of sun tracking capability must be included in the system, without disturbing the satellite operations. The simplest option for the SADA would be delegating the computation of the sun direction to the host satellite and act only upon command from the host satellite. This is indeed an operative option selectable by the hosting satellite. In addition to this, an autonomous solar panel orientation capability is present in the system, which might facilitate the host satellite operations in some mission phases.

Few options for the solar panel orientation strategy were analyzed in the system design trade-off, including: (i) direct measurement of the solar array output power; (ii) measurement of the sun angle in the solar array relative rotation plane, by shadows of an obstacle perpendicular to the solar panel plane; (iii) measurement of the sun angle in the solar array relative rotation plane, using a system of one axis sensors installed externally from the SADA box; and (iv) measurement of the sun direction, using a coarse, two axis sun sensor.

(i) First option: direct measurement of the solar array output power.
This system has the main advantage of maximizing power by direct power measurements, without relying on external sensors, avoiding issues related to sensor alignment, calibration and correlation with the solar panel delivered power. A second feature of this strategy is that no additional hardware is required, besides the solar panel voltage and current measurement system present in the SADA. In principle, there are different ways to calculate and track the solar panels' maximum peak power. To maintain the system completely transparent to the satellite, it is necessary to know the voltage and the current delivered by the solar panel at any time. The designed system has the option to bypass the current sensing element, in order to avoid an unnecessary voltage drop and power dissipation in the mission phases when the solar panel voltage and current measurement are not required.
To search for the highest power output as a function of solar panel orientation, solar panels are periodically moved by small angles in opposite directions. It is an easy strategy to implement and it does not require the presence of a dedicated sun sensor. However, it presents limitations in case of rapid satellite rotations. It is also sensitive to variations in the satellite power absorption happening during measurements. In addition, it may create a disturbance on the satellite attitude, due to the frequent and quite “fast” solar panels' rotation involved in this strategy.

(ii) Second option: in plane sun angle measurement by shadows of an obstacle perpendicular to the solar panel plane.
Two small sensing solar cells can be placed closely on the solar panel, with a small obstacle casting a shadow on one of these two symmetrical solar cells, depending on the sun's position. The system tries to keep the two power outputs at the same level,

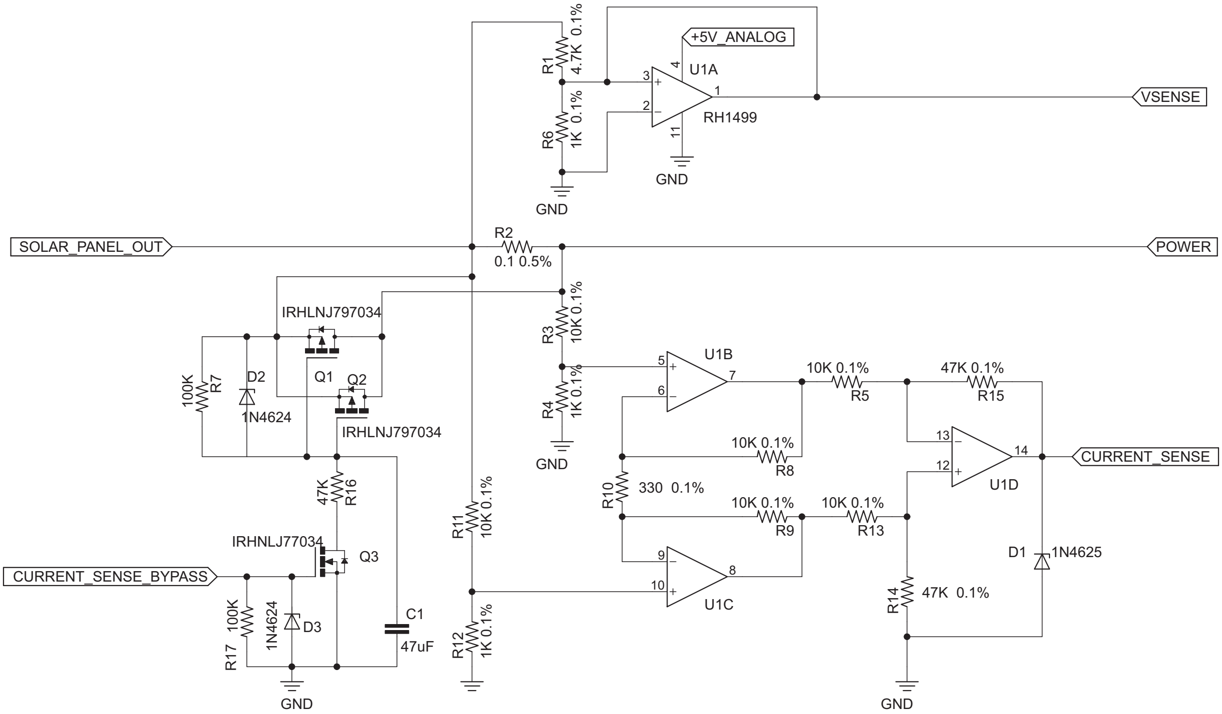

Fig. 8. Solar panels current and voltage measurement circuit, including bypass when measurements are not taken.

keeping the shadow centered, with the sun perpendicular to the obstacle. The main advantage of this system is that it is immune from the satellite load variation. It is easy to implement in principle, but it presents mechanical difficulties concerning the solar panel deployment system. The main drawbacks are: (i) small additional mass; (ii) sensing elements wasting some of the solar panel surface; and (iii) additional harness connecting the SPW to the SADA central unit.

(iii) Third option: sensors mounted on the SADA.
A number of small sensors, such as few square millimeter surface photodiodes or phototransistors, can be installed externally from the SADA box to sense the in-plane sun angle. The main advantages of this system are that it does not require harness going to the SPW and that no SPW motion is necessary to evaluate the orientation of maximum power. The main drawback is that accurate sensor calibration is needed on the ground and might be needed in orbit in case of long satellite lifetime, to evaluate differential sensor degradation in the space radiation environment.

(iv) Fourth option: two axis sun sensor.
A two axis, wide field of view, miniature sun sensor could provide the sun direction relative to solar arrays. This is a very high cost option, providing unnecessary information for the SADA operation, as the SADA has only one degree of freedom for the solar panel motion, while, at the same time, the satellite attitude is fixed by the mission objective and requirements. In addition, operation is limited by the sensor field of view. In case the satellite operation allows for complete sun tracking and maximum power is needed, then it is expected that the hosting satellite orients itself in such a way that the SADA box plane is orthogonal to the sun. Once this relative attitude with respect to the sun is reached, one axis pointing capability of the SADA is sufficient to orient the solar panels to the maximum power orientation.

At the present stage of the project, the result of the options trade-off for the solar panel orientation management strategy leads to the selection of third option, which gives the ability to autonomously detect the optimal solar paler angle, without requiring for additional mechanical deployment mechanisms and avoiding disturbance on the satellite attitude. This sensing system is intended for the SADA operations and it becomes effective once the satellite is stabilized. During the transient phase before stabilization, if required by the host satellite, it may provide a coarse information on the sun angle with respect to the top surface of the SADA. Of course this information is not sufficient for the satellite attitude determination, and it is expected that the host satellite has its own attitude sensing and control system to complete the attitude stabilization maneuver.

5. Sun angle measurement system

The sun angle in the plane orthogonal to the SPW rotation axis can be measured by installing three sun-light

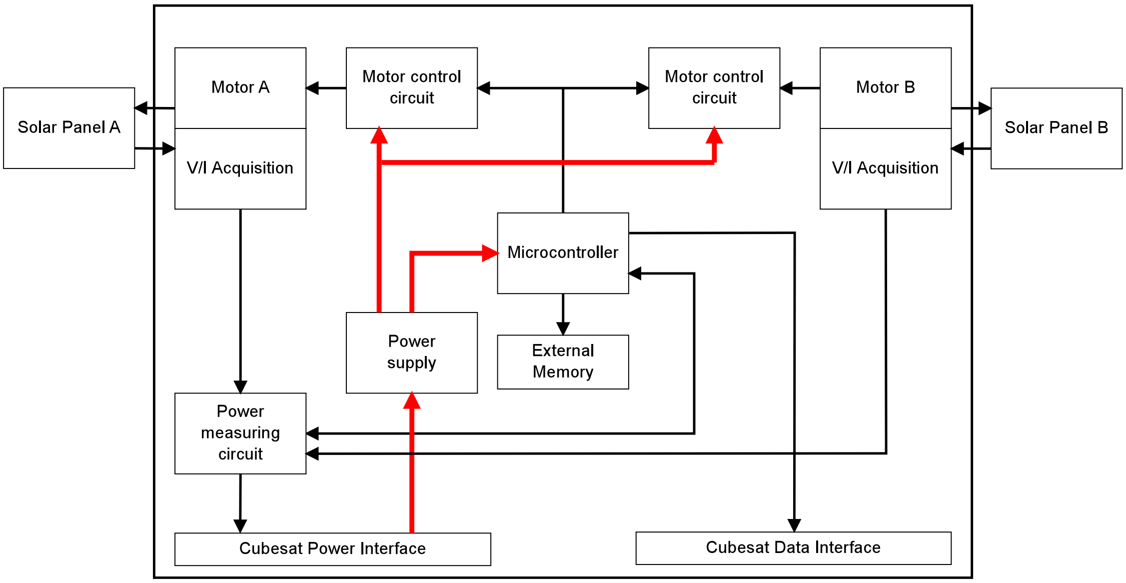

Fig. 9. SADA system block diagram including the motion control unit, solar panel status measurement system and interface to the hosting satellite.

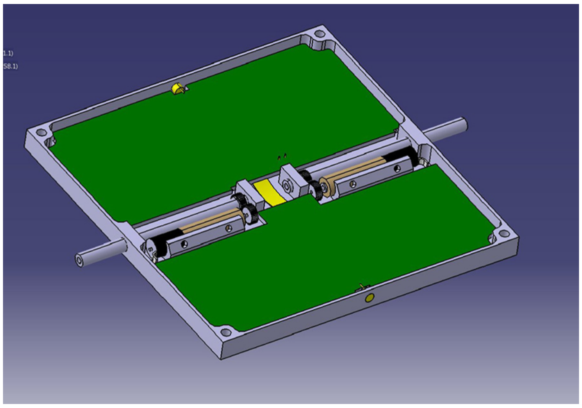

Fig. 10. Installation of two electronic boards within the SADA box.

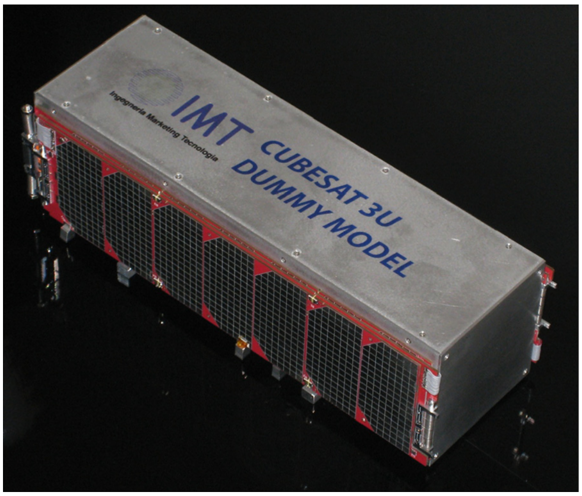

Fig. 11. Assembled satellite mock-up structure and solar array packs in the stowed configuration (SADA not installed).

sensing elements. One is mounted on the SADA top surface. The other two are mounted on the two opposite lateral surfaces of the SADA box which do not contain SPW shafts, as shown in Fig. 6.

The sun angle is measured by combining the outputs of three sensors. These are shown in Fig. 7, as a function of the sun angle, assuming that each sensor output is proportional to the cosine of the sun co-elevation with respect to the sensor itself. In the graph, the sun angle is measured from the normal to the SADA top surface. Hence a zero sun angle corresponds the sun orthogonal to the top surface, 90° and 270° sun angle correspond to maximum output from one of the two lateral sensors, 180° sun angle corresponds to zero outputs from all of the sensors.

6. Solar panel current and voltage measurement circuit

The SADA system has the capability of measuring the solar panel electrical status, that is its voltage and current. This information is used internally in the system test mode and it can be provided to the hosting satellite upon request. This system does not alter the solar panel operations and, being very simple in principle and realization, does not affect the system reliability negatively.

The measurement system's electric circuit is shown in Fig. 8. The U1A amplifier is used to sense the voltage at solar panel output. The voltage divider R1/R6 adapts the output voltage to the operational amplifier input range. Since this sensing element does not have any influence on the power bus, it is always connected. The sensing resistor R2 is used to read the current drawn by the satellite. The operational amplifiers U1B, U1C and U1D operate as an instrument amplifier, with a high differential value, capable to amplify the voltage drop on R2 and giving a precise reading of the current flowing through it. Due to its low resistance, the voltage drop on it is negligible and it does not interfere significantly with satellite operations, except for a very small power dissipation. Considering that the time intervals between two subsequent current and voltage readings are expected to be quite long, it is advisable to avoid this unnecessary power dissipation, even if very small. Two low Ron MOSFETs, Q1 and Q2, are used to bypass the sensing resistor when the current is not being measured.

Both voltage and current sensing outputs are routed to the microcontroller's internal ADC for auto-testing purposes, if required, or provided to the host satellite upon command.

7. SADA management architecture and control board

The overall SADA system' block diagram is shown in Fig. 9, including the supervising microcontroller, the MCU, the solar panel status measurement system and the interface to the host satellite.

The use of miniaturized motors leaves sufficient space to install two electronics boards. Correspondingly, electronics will be separated in two parts, connected by a flexible circuit, to permit a safe exit of the harness wires going to the solar panels, which will be routed inside the SPW hollow shaft. Fig. 10 shows the final aspect of the assembly.

This arrangement will leave enough space to mount components on both sides of the PCB, with the highest ICs on the top side and the small components, like SMD 0805 or 0603 format passive components, on the lower side.

A prototype of the system has been realized and used for the deployment sequence ground testing. The prototype built, at present, includes the deployable solar panel system and the necessary commands installed on the SADA.

Several successful deployment sequences were conducted on a virtually frictionless lubricated silicon support. The assembled prototype ready for testing is shown in Fig. 11.

8. Conclusions

A deployable solar panel system for nanospacecraft has been developed in the framework of a cooperation established between the Laboratorio di Sistemi Aerospaziali of University of Rome "la Sapienza" and IMT srl, Ingegneria Marketing Tecnologia. The system mainly consists of two distinct parts: the deployable solar panels and the motion control unit which controls the solar panels orientation. The system design has been finalized for the 3U Cubesat, scalable to any Cubesat size. The design trade-off was discussed in the paper, highlighting the main advantages and drawbacks of possibly identified technical solutions, both at the architectural and component level. The final architectural configuration consists of two independent deployable solar panel systems, improving the system reliability in case of failure of one solar panel's wing deployment. The final component technology selection, mainly referred to the electric motor technology, was in favor of stepper motors, which leads to a simpler motor driving circuit and simplifies the solar panels angular position measurement. At the present stage of the project, a prototype of the deployable solar panels has been built and successfully tested.

References

[1] J. Bester, B. Groenewald, R. Wilkinsin, Electrical power system for a 3U CubeSat nanosatellite incorporating peak power tracking with dual redundant control, Przeglad Elektrotechniczny (Electrical Review), ISSN 0033-2097, R. 88 NR 4a/2012.

[2] A.E.Kalman, Enhanced Power Systems for CubeSats, CubeSat Developers' Workshop 2012, CalPoly, San Luis Obispo, CA, April 18-20, 2012.

[3] F Santoni, F Piergentili, Analysis of the UNISAT-3 solar array in-orbit performance, J. Spacecr. Rockets 45 (N.1) (2008), http://dx.doi.org/10.2514/1.32392. (Jan–Feb).

[4] Santoni, F., Piergentili, F., Graziani, F., In orbit performances of the UNISAT-3 solar arrays, in: Proceedings of the 57th International Astronautical Congress, Volume 9, pp.5789-5797, paper IAC-06-C3.4.-D3.4.04, Valencia, Spain, 2-6 October 2006.

[5] Santoni, F., Piergentili, F., Bulgarelli, F., Graziani, F., "UNISAT-3 power system", ESA SP-589, in: Proceedings of the Seventh European Space Power Conference, 9-13 May 2005, Stresa, Italy, pp. 395–400.

[6] Santoni, F., Piergentili, F., UNISAT-3 attitude determination using solar panel and magnetometer data, in: Proceedings of the 56th International Astronautical Congress, Volume 5, pp. 2812-2819, paper IAC-05-C1.2.06, Fukuoka, Japan, 17–21 October 2005.

[7] Santoni, F., Piergentili, F., Design and test of a maximum power point tracking system for UNISAT-3 microsatellite, in: Proceedings of the 55th International Astronautical Congress, Volume 10, Pages 6668-6677, paper IAC-04-R.2.01, Vancouver, Canada, 4–8 October 2004.

[8] Santoni, F., Tortora, P., Alessandrini, F., Passerini, S., Commercial Li-ion batteries for nanosatellite applications: a flight experiment, in: Proceedings of the Sixth European Conference on Space Power, ESA SP (502), p. 653–658, 2002, ISSN 0379–6566.

[9] Santoni, F., Ferrante, M., Graziani, F., Ferrazza, F., In orbit performance of the UNISAT terrestrial technology solar panels, in: Proceedings of the IEEE Aerospace Conference, Big Sky, MT, 10-17 March 2001, Volume 1, 2001, pp.1363-1371.

[10] Santoni F., Bolotti F., Attitude determination of small spinning spacecraft using three axis magnetometer and solar panels data, in: Proceedings of the IEEE Aerospace Conference , Vol. 7,2000 p.127–133, March 2000.

[11] G.P. Candini, F. Piergentili, F. Santoni, Miniaturized attitude control system for nanosatellites, Acta Astronaut. 81 (1) (2012) 325–334, http://dx.doi.org/10.1016/j.actaastro.2012.07.027.

[12] L. Vaccari, M. Altissimo, E. Di Fabrizio, F. De Grandis, G. Manzoni, F. Santoni, F. Graziani, A. Gerardino, F. Perennes, P. Miotti, Design and prototyping of a micropropulsion system for microsatellites attitude control and orbit correction, J. Vac.Sci. Technol. B, Microelectron. Nanometer Struct. 20 (2002) 2793–2797.

[13] F. Santoni, P. Tortora, F. Graziani, G. Manzoni, E. De Fabrizio, L. Vaccari, Micropropulsion experiment on UniSat-2, in: Proceedings of the IEEE Aerospace Conference 2002, Big Sky, MT, USA, March 9-16, 2002. doi: 10.1109/AERO.2002.1036868.

[14] F.Santoni, F.Piergentili, S.Donati, M.Perelli, A.Negri, M.Marino, Desing and realization of an innovative deployable solar panel system for cubesats, paper IAC-12,C3,4,1,x14280, in: Proceedings of the 63rd International Astronautical Congress, Naples, Italy, 1–5 October 2012.

[15] F Santoni, F Piergentili, S Donati, M Perelli, A Negri, M Marino, An innovative deployable solar panel system for cubesats, Acta Astronaut. 95 (2014) 210–217, http://dx.doi.org/10.1016/j.actaastro.2013.11.011.

[16] Graziani, F., Santoni, F., Piergentili, F., Bulgarelli, F., Sgubini, M., Bernardini, S., Manufacturing and launching student-made microsatellites: "hands-on" education at the University of Roma, in: Proceedings of the 55th International Astronautical Congress, Volume 9, pp.5789-5797, paper IAC-04-P.5.A.02, Vancouver, Canada, 4-8 October 2004.

[17] F. Santoni, Risk management for microsatellite design, Acta Astronaut. 54 (3) (2004) 221–228, http://dx.doi.org/10.1016/S0094-5765(02)00291-6.

[18] Santoni, F., Piergentili, F., Bulgarelli, F., Graziani, F., The UNISAT program: lessons learned and achieved results, in: Proceedings of the 57th International Astronautical Congress, Volume 13, 2006, Pages 8930-8936, paper IAC-06-E1.1.10, Valencia, Spain, 2–6 Oct 2006.

[19] F. Graziani, F. Piergentili, F. Santoni, A space standards application to university-class microsatellites: the UNISAT experience, Acta Astronaut. 66 (9-10) (2010) 1534–1543. (ISSN: 0094-5765. doi: 10.1016/j.actaastro.2009.11.020).
[20] F. Graziani, G. Pulcrano, F. Santoni, M. Perelli, M. L. Battagliere, EduSAT: an Italian space agency outreach program, paper IAC-09-E.1.3, in: Proceedings of the 60th International Astronautical Congress, 12-16 Oct. 2009, Dajeon, Korea.
[21] M.L. Battagliere, F. Santoni, F. Piergentili, M. Ovchinnikov, F. Graziani, Passive magnetic attitude stabilization system of the EduSAT micro satellite, Proc. Inst. Mech. Eng. Part G: J. Aerosp. Eng. 224 (10) (2010) 1097–1106, http://dx.doi.org/10.1243/09544100JAERO732.
[22] Jansen, T., Reinders, A., Oomen, G., Bouwmeester, J., Performance of the first flight experiment with dedicated space CIGS cells onboard the Delfi-C3 nanosatellite, in: Proceedings of the 35th IEEE Photovoltaic Specialists Conference (PVSC) 2010, 20-25 June 2010, p. 1128–1133. http://dx.doi.org/10.1109/PVSC.2010.5614729.
[23] Senatore, P., Klesh, A., Zurbuchen, T. H., McKauge, D., & Cutler, J., Concept, design and prototyping of XSAS: a high power extendable solar array for CubeSat applications, in: Proceedings of the 40th Aerospace Mechanisms Symposium, NASA Kennedy Space Centre, May 12-14, 2010, p.431–444.
[24] Plaza, J. M., Vilan, J. A., Agelet, F. A., Mancheno, J. B., Estevez, M. L., Fernandez, C. M., Ares, F. S., Xatcobeo: small mechanisms for Cubesat satellites antenna and solar array deployment, in: Proceedings of the 40th Aerospace Mechanisms Symposium, NASA Kennedy Space Centre, May 12–14, 2010, p. 415–429.
[25] Reif, A. W., Hoang, V., & Kalman, A. E., “Recent advances in the construction of solar arrays for CubeSats”, CubeSat summer developer's workshop, in: Proceedings of the 24th Annual AIAA/USU Conference on Small Satellites, Utah State University, August 2010.
[26] Clyde Space Ltd, Small Satellite Solar Panels datasheet, March 2012.
[27] Passaretti, M., Hayes, R., Development of a solar array drive assembly for CubeSat, in: Proceedings of the 40th Aerospace Mechanisms Symposium, NASA Kennedy Space Centre, May 12–14, 2010, p. 445–453.
[28] H. Heidt, J. Puig-Suari, A. S. Moore, S. Nakasuka, R. J. Twiggs, CubeSat: A new Generation of Picosatellite for Education and Industry Low-Cost Space Experimentation, in: Proceedings of the AIAA/USU Conference on Small Satellites, Aug. 21–24, 2000, SSC00-V-5.
[29] P. Bajpai, A. Kahraman, N.E. Anderson, A surface wear prediction methodology for parallel-axis gear pairs, J. Tribol. 126 (2004) 597–605.
[30] L.S.Akin, D.P. Townsend, Wear consideration in geardesign for space applications, in: Proceedings of Fifth International Power Transmission and Gearing Conference, ASME, Chicago, Illinois, April 25–27, 1989, (NASA TM 101457).
[31] T.P Sarafin, Spacecraft structures and mechanisms, space technology series, Microcosm, Torrance, CA/Kluwer Academic Publishers, Dordrecht, The Netherlands, 2003.
[32] Faulhaber, Technical Information EN, 4th Edition, 2012–2013.