DESIGN OF THE ELECTRICAL POWER SYSTEM FOR THE ESTCUBE-1 SATELLITE

M. Pajusalu¹, R. Rantsus¹, M. Pelakauskas¹, A. Leitu¹, E. Ilbis¹, J. Kalde¹, H. Lillmaa¹, R. Reinumägi¹, K. Voormansik¹, K. Zālīte¹, V. Allik², M. Noorma¹,², S. Lätt¹,²

¹Institute of Physics, University of Tartu,
4 Tähe Str., Tartu, 51010, ESTONIA
e-mail: mihkel.pajusalu@ut.ee

²Tartu Observatory,
1 Observatooriumi Str., Tõravere, Tartumaa, 61602, ESTONIA

In this work, the design of the electrical power system developed for ESTCube-1 – the first Estonian satellite with the first test mission of electric solar wind sail – is presented. The mission is highly energy-intensive, and, since its complexity might lead to many possible failure cases, it requires a very efficient and fault-tolerant solution. The system has been developed from ground up, using only commercial-off-the-shelf components and student workforce. It includes several innovations, which help the system to be more reliable, efficient and controllable than earlier nanosatellite power system designs.

**Keywords:** nanosatellites, electrical power systems, fault tolerance, solar power.

1. INTRODUCTION

Space research is highly important for the future of mankind, being the forefront of science and technology. Actual space missions have historically been available only to larger political powers and corporations due to high costs involved. Still, recent advances in the related technologies, such as electronics miniaturization and higher power efficiencies, make it possible to fit greater and greater amounts of functionality into the same volume. This has led to the emergence of nanosatellites, such as CubeSats (cubical satellites with a volume of one liter and a mass of less than 1.33 kg). These solutions allow the circle of entities participating in space research to be broadened by providing cost-effective research platforms [1]. Low cost also makes these platforms ideal for educating spacecraft engineers. CubeSats are promising solutions for many space research and educational projects – from solar sails [2, 3] to the Earth observation hyperspectral imaging [4] and tether experiments [5].

Space missions involving nanosatellites are becoming increasingly complex, which also imposes increased requirements on their electrical power systems (EPSs). At the same time, small size of CubeSats places very strict restrictions on the available volume, mass and power. This means that the management of elec-
Electrical power becomes more of an issue, requiring complex and dependable power harvesting, storage and distribution solutions (see, e.g. [6]). The details of other nanosatellite EPSs can generally be found in their mission reports, which often are open to public.

This paper is focused on the design of an EPS for the ESTCube-1, a single unit CubeSat with one of the most complex and ambitious missions in the history of nanosatellites: proof-of-concept testing of the Electric Solar Wind Sail, or E-Sail [7, 8], which could provide a very cost-effective mode of propulsion for further missions within the Solar system. Previously known experiments with charged tethers have all failed [1, 5]. In addition to complex mission requirements, the satellite has to be constructible from commercial-off-the-shelf (COTS) components. To increase its educational value, the CubeSat has been designed from ground up by students at the University of Tartu.

2. MISSION REQUIREMENTS

2.1 Brief overview of the mission

The main mission of the ESTCube-1 satellite, planned to be launched in the beginning of 2013, is to conduct the first conceptual test mission of the Electric Solar Wind Sail [7, 8]. The test involves deployment of a 10 m long conductive tether from the satellite, charging it using electron guns, and monitoring its interaction with the Earth’s magnetospheric plasma.

The satellite’s mission starts with deployment from the launch vehicle. After a period of time the satellite will deploy antennas for radio communications. If previous steps are completed, the satellite can start broadcasting radio communications to and receiving them from ground stations. The satellite will then de-tumble using magnetotorquers to achieve a controlled orientation of the satellite.

The main scientific mission involves deploying the conductive tether and charging it to a high voltage (±450 V). The deployment is to be done using a motor in the payload section. During the deployment and prior to it the satellite will be spun up in order to use the centrifugal force to keep the tether straight. To amplify this effect, an end-mass will be attached to the end of the tether. A secondary mission is to take pictures of the Earth for publicity materials using an onboard camera. To achieve specific subsections of the mission goals, the satellite system is divided into several subsystems, each occupying a proportion of the satellite’s volume:

- Attitude Determination and Control System (ADCS): used for determination of the satellite’s attitude in orbit using sensors and to adjust it using magnetic torquers.
- Command and Data Handling System (CDHS): the satellite’s main computer, used for orbital calculation and for conducting the main mission.
- Camera (CAM): an on-board camera module that can be used to take photographs in space; its main function is to take photos of Earth and of the tether and end-mass during the main mission.
- Communications System (COM): handles both receiving radio communications from the Earth and sending them to Earth.
- Electrical Power System (EPS): harvests, stores and distributes power.
- Payload (PL): tether reel, electron guns, and driver electronics.
2.2 Solar power conditions

The satellite is planned to be launched to Low Earth Orbit (LEO), and this is likely to be a sun-synchronous midday-midnight orbit. On this orbit the intensity of solar radiation is \( \approx 1350 \text{ W/m}^2 \). Since the satellite is a cube with dimensions of 10 cm x 10 cm x 10 cm, this means that the total power of incident radiation when the satellite is illuminated will range from 13.5 W (when one side of the cube is faced directly towards the Sun) to \( \approx 19 \text{ W} \) (when one of the cube’s edges is directed towards the Sun). A smaller contribution comes from Earth’s albedo [9], but this is generally less than 10% of the total power production. The exact orbit-average power production will depend on the orbit and the satellite’s orientation in respect to the Sun. To collect this energy, solar cells are going to be used. For this mission, the satellite is equipped with two 30% efficient GaAs solar cells (from AZUR SPACE Solar Power GmbH) per side, bringing the beginning-of-life (BOL) solar power production when illuminated from 2.4 W when one side is directed towards the Sun to \( \approx 3.4 \text{ W} \) when one of the edges is directed towards the Sun (at 28 °C). Due to the changing power demands of the satellite and the illumination of the solar cells, the power peak of the cells has to be continuously tracked through Maximum Power Point Tracking (MPPT), see [10] for a review.

2.3 Radiation environment

The satellite is to be deployed in LEO, which means that it is within Earth’s magnetosphere but will still be subject to radiation (the total ionizing dose per year being 4–40 kRad [11]). The time of planned launching of the satellite (the beginning of 2013) coincides with a solar activity maximum, making the radiation tolerance a design requirement.

Radiation can cause several different types of problems for the satellite; a study of its impact on CubeSats’ electronics is presented in [11]. It is unlikely that the total dose accumulated during the expected mission (less than a year’s duration) would cause the satellite to fail, whereas single event effects – most notably Single Event Latch-ups (SEUs) from the power viewpoint and Single Event Upsets (SEUs) from the data viewpoint – will likely be the main issues. The SEL is an effect where a particle hitting a semiconductor circuit can cause the formation of a parasitic thyristor, which stays operational until it is powered, potentially causing extreme local overheating and physical damage of the integral circuit dies. SEUs are flips of data bits causing data corruption, leading to corruption of memory and, possibly, unexpected program execution. To ensure the mission’s success, both of these have to be addressed.

2.4 Power budget

The preliminary power budget for the satellite is presented in Table 1. It can be seen that the average power consumption when the satellite is idle is \( \approx 700 \text{ mW} \) (including a 20% safety margin). When the estimated thermal variations during the orbit (−20 °C to 80 °C) are taken into account, the solar panels will produce on average a minimum of \( \approx 2.2 \text{ W} \), of which 70% (with a margin) can be harvested, giving \( \approx 1.5 \text{ W} \) of average power during the illuminated part of the orbit. Due to the expected orbital parameters, the satellite will be in this phase 2/3 of the time (approximately, since the exact orbit is not known), giving the worst case average power production per orbit of \( \approx 1.0 \text{ W} \), which is higher than 700 mW even if the
Converter losses are taken into account (greater than 90% efficient converters are used in most cases). This gives in the worst case ≈ 0.2 W of power to charge the batteries. Still, this shows the need for precise management of power to avoid a catastrophic power failure.

Most of the energy-intensive operations need to be performed on battery power, and therefore battery management is a high-priority task during the whole mission.

Table 1

<table>
<thead>
<tr>
<th>Satellite subsystem</th>
<th>Peak power, mW</th>
<th>Mission average power, mW</th>
</tr>
</thead>
<tbody>
<tr>
<td>ADCS</td>
<td>960</td>
<td>170</td>
</tr>
<tr>
<td>CAM</td>
<td>240</td>
<td>*</td>
</tr>
<tr>
<td>CDHS</td>
<td>250</td>
<td>180</td>
</tr>
<tr>
<td>COM</td>
<td>3600</td>
<td>320</td>
</tr>
<tr>
<td>PL</td>
<td>2500</td>
<td>*</td>
</tr>
<tr>
<td>ADS (Antenna deployment system)</td>
<td>4200</td>
<td>*</td>
</tr>
</tbody>
</table>

*Systems that are used only during specific mission phases can be scheduled according to the power available, and therefore do not affect the average power budget directly.

3. RESULTS AND DISCUSSION

3.1. General design

To fulfill the mission requirements, the EPS has been designed as shown in Fig. 1. Energy harvesting is done by three MPPT drivers, each connected to four solar cells. The main power bus (MPB) transfers the energy between EPS components, either for storage or distribution. The energy is stored in two batteries, which are interfaced to the MBP through two independent protection circuits. Voltages for other subsystems are converted using pairs of redundant converters for each voltage. The whole system is controlled by a microcontroller, which also communicates with other subsystems.

![Fig. 1. General design of the electrical power system.](image-url)
In reality, the system is divided into two printed circuit boards (PCBs). Power harvesting, storage and distribution are on one of them, and a separate PCB is allocated for the control. Here the novel design aspects of the EPS are presented (the actual testing results will be provided in a later publication).

3.2 Energy harvesting

To harvest energy from the solar arrays as efficiently as possible, a Maximum Power Point Tracking solution is to be used. Earlier nanosatellites either have not done this, connecting the solar panels directly to the batteries via diodes [6], or have used an MPPT driver actively controlled by a microcontroller, which runs the MPPT algorithm [12]. Our first approach was similar to the latter; however, this was later abandoned in favour of an independent MPPT chip (SPV1040 from STMicroelectronics). The chip is in itself a boost converter, which regulates its output voltage using an internal perturb-and-observe MPPT algorithm.

The use of this new system has many advantages over traditional MPPT designs. First, this makes the MPPT system completely autonomous. This means that the power production can continue even an event in which the satellite’s power is completely lost, thus improving fault tolerance. Secondly, it frees up the microprocessor’s resources. Besides, it is more efficient than traditional MPPT solutions — our preliminary tests have shown the efficiency to be ≈ 80%, i.e. 10% greater than that of most reported MPPT designs.

To achieve the best performance while using the least number of chips, a design with 3 MPPT modules was chosen, each collecting the energy from solar panels on the opposite sides of the satellite (from four solar cells in total). In practice, the satellite is illuminated by sun from one side only (albedo effects can be generally neglected [9]), meaning that the power is produced from one side only, thus providing a single power peak that can be efficiently tracked. The general schematic for a single MPPT module is seen in Fig. 2. Each module collects power from four solar cells connected in parallel from the opposite sides (denoted by “+” and “−” on the figure) of the satellite.

![Fig. 2. Block diagram of a single Maximum Power Point Tracking module.](image)

Generally, this configuration is not used, since the voltage drop is too large if blocking diodes are used. In this case, however, power flow controllers were employed (MOSFETs driven by LTC4352 chips from Linear Technology). This
ensures a low voltage drop compared with usual Schottky diodes. A downside of this system is its complexity, which makes single point failures a potential problem. To mitigate these, three MPPT modules are placed in parallel, all working independently. This kind of configuration is able to handle several possible failure modes. The system can provide power to the satellite even if a single solar cell with its MPPT driver is online, making a complete power production failure very unlikely.

3.3 Energy storage

The satellite stores its energy in two cylindrical P-CGR 18650C Lithium-Ion cells from Panasonic. In our earlier tests it was determined that these are suitable for the satellite’s requirements, providing a total of 9 Wh of energy storage in the worst case (\(-15 \, ^\circ\text{C}\)). This is enough to perform, e.g., the full tether unreeeling (taking \(\approx 3\) 4.5 Wh according to our latest estimates).

Both battery cells are connected to the MPB through independent protection circuits, which provide current limiting both in the charge and the discharge directions and can be used to switch either the former or the latter off. This system is based on two TPS2557 chips (Texas Instruments) connected in series in reverse to each other.

3.4 Power distribution

Power is distributed to other subsystems through switching regulators. The EPS outputs three voltages: 3.3 V, 5 V and 12 V, each of them produced by two parallel converters: Linear Technology LTC3440 buck-boost converters are used for 3.3 V and 5 V, while National Semiconductor LM2700 boost converters – for 12 V; the 12 V voltage can be regulated by \(\pm 10\%\) using a digital-analog converter on its feedback line to fulfil a payload requirement.

Between each voltage line and the connected consumers a control & monitoring circuit is located (see Fig. 3), which consists of a current sensor and a current-limiting switch TPS2557 or TPS2551 (Texas Instruments), depending on the electrical current requirements. Current-limiting switches are used to provide current-limiting functionality in the case of a Single-Event Latch-up and are to be able to turn the consumers on and off. To control the switch in a fault-tolerant way, a FM1105 FRAM (Ferroelectric Random Access Memory) state saver (from Ramtron) is placed on the enable signal line.

![Fig. 3. Design of the control circuit applied in power distribution units. Solid lines describe the energy flow, and dashed lines – the data flow.](image)
3.5 Control and Communication

The EPS is controlled by its own microcontroller, ATmega1280 from Atmel. Previous publications report its radiation tolerance to be sufficient [11] for the current mission. The chip contains the power-distribution and housekeeping firmware for the satellite’s electrical power system.

To ensure the radiation-tolerance of the memory, FRAM parallel memory FM18W08 (Ramtron International Corporation) is used as a non-volatile and radiation-tolerant extension to the microcontroller’s internal random access memory. Since FRAM is non-volatile and highly radiation-tolerant [13], it can be employed to maintain the operation of the EPS by keeping important parameters safe even when power is lost. To store information safely for a longer time, FM25256 SPI FRAM chips from Ramtron are used. Apart from previous, an extra feature is the use of FM1105 FRAM state-saver modules (from Ramtron) that can save the state of switches, ensuring that the state of power distribution will not change when the processor is reset.

The microprocessor itself is powered from a 5 V secondary power bus (LTC3440), powered by two redundant converters, and backed up by a capacitor bank. The bus is intended to give the EPS microcontroller time to react in the case of a power failure. The most critical elements – the state-savers – are powered by a third power bus, which is fed by a charge pump and is meant to power up the state-savers that enable various power distribution switches.

4. CONCLUSIONS

The basic elements of the electrical power system have been designed for the ESTCube-1 satellite. High efficiency and reliability of the EPS is achieved by using novel autonomous Maximum Power Point Tracking integral circuits, which provide a redundant and efficient way to harvest solar energy. The power is transmitted within the EPS through the Main Power Bus, which connects energy harvesting, storage and distribution. The energy is stored in two lithium-ion
batteries, which are interfaced to the Main Power Bus through independent protection circuits, providing possibilities for current limitation and for disabling/enabling charging or discharging of the batteries. The energy is distributed to other subsystems through the power distribution converters, which provide 3.3 V, 5 V and 12 V voltages using parallel redundant converters. The energy distribution occurs via control circuits, which make it possible both to limit and to monitor the currents consumed and to turn the subsystems on and off.

The EPS is controlled by its own microprocessor powered through a semi-independent power supply, and FRAM is used for increasing the radiation tolerance of the control system.

It has been proven that an advanced power supply system for a nanosatellite could be constructed from COTS components and using only student workforce for designing.

Acknowledgements

The authors thank all of the people who have participated or are participating in the Estonian Student Satellite program.

This research was partially supported by European Social Fund's Doctoral Studies and Internationalisation Programme DoRa.

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**Kopsavilkums**

Nanosatelītu kosmiskās misijas klūst arvien kompleksākās, kas, savukārt, uzliek papildu prasības izmantoto satelītu elektropadeves nodrošināšanas sistēmām. To izstrādā sarežģī CubeSat platformu mazais izmērs, kas ierobežo risinājumam izmantojamo vietu, svaru un jaudu. Šajā rakstā izklāstīts Igaunijas pirmā satelīta ESTCube-1 elektropadeves nodrošināšanas sistēmas dizains. Misija ir pirmo reizi izvērīt kosmosā elektrisko saules buru. Uzdevuma sarežģītās un jaudas prasību dēļ ir iespējami daudzi klūdu scenāriji, tāpēc risinājumam jābūt ļoti efektīvam un noturīgam pret klūdām. Sistēmu izstrādāja studenti, izmantojot komerciāli pieejamos komponentus, un tā satur vairākus jauninājumus, kas nodrošina lielāku tās drošumu, efektivitāti un kontrolējamību salīdzinājumā ar iepriekšējām nanosatelītu elektropadeves nodrošināšanas sistēmām.

06.05.2012