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Publication date
2024

Document Version
Final published version

Published in
Proceedings of the 75th International Astronautical Conference

Citation (APA)
Cervone, A., Speretta, S., Uludag, M. S., Busso, C., Vasile, M., Boumchita, W., Clemente, C., Feng, J., & Madi, M. (2024). Pre-Phase A Design of the 16U4SBSP Spacecraft: a Scaled Demonstration of Space-Based Solar Power in Earth Orbit using a Swarm of CubeSats. In *Proceedings of the 75th International Astronautical Conference*

IAC-24-B.4.7.7

Pre-Phase A Design of the 16U4SBSP Spacecraft: a Scaled Demonstration of Space-Based Solar Power in Earth Orbit using a Swarm of CubeSats

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Abstract

The 16U4SBSP mission concept is based on using a swarm of CubeSats to perform a scaled demonstration of Space-Based Solar Power (SBSP) from Earth orbit. In this demonstration mission, seven identical spacecraft of 16U format are used to provide wireless energy in the kW-scale using Radio-Frequency (RF) Wireless Power Transfer (WPT), and the spacecraft in the swarm are designed to be suitable to both space-to-ground or space-to-space WPT applications. The main objective of the mission is to validate the general concept of providing SBSP using a swarm of satellites instead of a monolithic configuration, as well as some of the involved miniaturized technologies, in view of full-scale missions which could serve users in remote areas with low power requirements or support emergency operations in blackout zones affected by unpredicted hazards (e.g. natural disasters). More in general, the mission would represent a low-cost precursor towards MW-GW scale SBSP to supply clean and affordable energy from space to large areas on the Earth surface. A pre-Phase A study of the mission, funded by the European Space Agency (ESA) through the Sysnova campaign "Innovative Missions Concepts enabled by Swarms of CubeSats", has led to encouraging results on the feasibility of the mission concept.

This paper presents in detail the final outcome of the pre-Phase A design effort for the 16U4SBSP spacecraft. The trade-off studies conducted to select all sub-systems and components are presented and their final outcomes detailed and justified, together with the technical budgets and the main areas of attention for the spacecraft design. Particularly critical for the success of the mission are the choices related to: the power transmission payload (DC-RF converter, transmitting antenna and heat dissipation system); the ADCS subsystem and in particular the sensors required to provide sufficient accuracy in the knowledge of the 3-axis attitude (both absolute and relative to the other spacecraft in the swarm); the relative navigation system, based on inter-satellite link between the spacecraft in the swarm and on a beacon link to the receiving station on ground, for efficient beaming coordination; the main propulsion system for continuous formation flying control through the whole mission lifetime; the electric power system, based on orientable solar arrays by means of a SADA mechanism and a set of batteries with sufficient capacity for beaming the required amount of power while in eclipse conditions.

Keywords: 16U4SBSP, Swarms of CubeSats, Space Based Solar Power, Radio-Frequency Beaming, Wireless Power Transfer, ESA SysNova challenge

1. Introduction

16U4SBSP is a mission concept intended to perform a scaled in-orbit demonstration of Space-Based Solar Power (SBSP). In this demonstration mission, a swarm of 7 identical 16U CubeSats is intended to provide wireless energy at power levels in the kW-scale (total amount of received power on the Earth surface), using Radio-Frequency (RF) Wireless Power Transfer (WPT). Although the mission is mainly intended for space-to-ground demonstrations, the spacecraft in the swarm are designed in such a way to also be suitable to space-to-space WPT applications. The main objective of the mission is to validate the concept of providing SBSP by means of a swarm of satellites instead of a monolithic configuration. At the same time, however, the mission is also intended as an in-orbit demonstration for a number of miniaturized technologies. The ultimate objective is to pave the way to full-scale missions which could serve users in remote areas with low power requirements or support emergency operations in blackout zones affected by unpredicted hazards (natural or man-made). The mission therefore represents a low-cost precursor towards larger scale SBSP, where the actual transferred power would be in the MW-GW range, to supply clean and affordable energy to large areas on the Earth surface.

16U4SBSP was one of the proposals submitted to the European Space Agency (ESA) SysNova challenge "Innovative Missions Concepts enabled by Swarms of CubeSats", intended to generate new concepts for CubeSat swarm missions in Earth orbit and to quickly verify their usefulness and feasibility via short concurrent studies [1]. After the first phase of the challenge (open call for ideas), 16U4SBSP has been one of the seven proposals selected for performing a pre-Phase A analysis, funded by ESA.

This paper, after shortly introducing the mission analysis, concept of operations and phases, will discuss in detail the system and subsystem requirements and the consequent 16U4SBSP spacecraft design obtained at the end of the pre-Phase A study.

2. Mission analysis, phases and modes

Based on the above description, the general scope of the 16U4SBSP mission has been set as follows: *to design, develop, commission and launch a commercial space-based solar power (SBSP) demonstrator based on CubeSats in a distributed swarm configuration*. This flows down to the primary mission objective, which is: *to validate with a small-scale mission the beamforming power transmission model developed by the consortium and, in this way, confirm that it is feasible and convenient to provide SBSP by means of a larger constellation of spacecraft (larger both in terms of number, and size)*.

To meet these ambitious mission objectives, four different mission phases have been envisaged:

- **Deployment and Commissioning.** In this phase, after being released from the launcher, the spacecraft will perform an initial de-tumbling maneuver. Solar arrays will then be deployed and all subsystems commissioned. At the same time, all communication links will be established.
- **Formation Acquisition.** Each of the possible formation modes foreseen in the mission Concept of Operations will be acquired by performing all required orbital and attitude correction maneuvers. Reaction wheels will be desaturated, and the power transmission payload will be commissioned (first formation acquisition) or re-configured (successive formation acquisitions).
- **Operative phase.** After acquiring the formation, all subsystems (including payload) will be calibrated. When in range with the target ground receiver, power transmission operations will be performed. Outside these power transmission windows, other operations will be performed, including: formation maintenance, orbital maneuvers, reaction wheels desaturation, communication to ground.
- **End-of-Life.** After decommissioning all subsystems, any necessary End-of-Life maneuvers will be performed.

Based on the above mission phases, a comprehensive mission analysis has been conducted, with the final goal of determining the most convenient orbit for the demonstration mission and, consequently, the required Delta-V budgets. For the deployment from the launcher, it has been assumed that the CubeSats are deployed every 30 seconds, with a velocity differential (with respect to the launcher) ranging from 1.6 to 2 m/s. All CubeSats are then distributed, through dedicated maneuvers, within a range of 1 km from a central reference point. In this configuration, an observation campaign is conducted for a maximum of 10 days, in order to acquire sufficient data to define the required Delta-V to acquire the formation. A first formation at inter-satellite distance of 1000 m is then acquired, in order to perform a general test and troubleshooting of all sub-systems at safer intersatellite distance. Figure 1 shows the required steps to go from the initial along-track distribution of the seven CubeSats in the swarm, to this 1000 m intermediate formation.

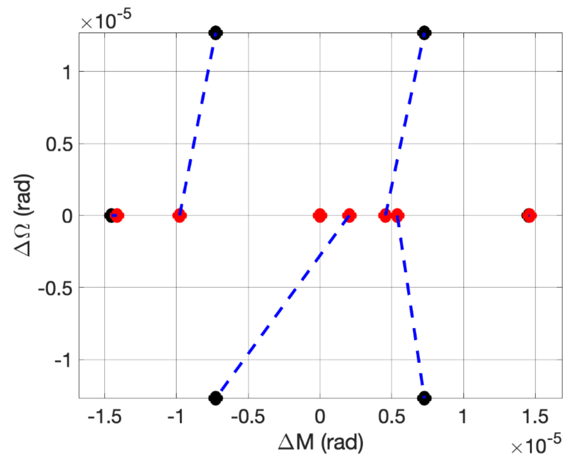

Fig. 1. Formation distribution from the along-track direction (in red) to the hexagonal formation (in black). The final and initial position in the *ΔM - ΔΩ* plane of each spacecraft is connected by a blue dashed line.

After this initial phase, the formation is then adjusted by decreasing the inter-satellite distance to 100 meters, maintained for the first part of the mission (3-6 months). For the second part of the mission and until End-of-Life, the inter-satellite distance is then further reduced to 10 meters.

A comprehensive analysis, based on all possible perturbation sources (atmospheric drag, solar radiation, gravity gradient), was performed to define how frequently the formation needs to be controlled. For an attitude knowledge accuracy of 0.1 degrees, the results of this analysis show that the orbital position needs to be controlled every 7 days for 100 m inter-satellite distance, and every 2.5 days for 10 m inter-satellite distance.

Finally, for the End-of-Life phase, it is predicted that with an initial Sun-synchronous orbit at 500 km altitude, the CubeSats will re-enter the atmosphere within 1.5 years, thus staying compliant with the most recent ESA Space Debris Mitigation requirements. With higher Sun-synchronous orbits (600 km or 700 km altitude), an additional Delta-V from the main propulsion system would be required to allow for sufficiently fast End-of-Life maneuvers.

Table 1 presents the calculated worst-case Delta-V budgets for two possible Sun-synchronous orbits, at altitudes equal to 500 km and 600 km. Given the highly stochastic nature of these calculations, in order to determine the Delta-V budget and derive the main propulsion sub-system requirements, a 100% margin have been considered on all calculated Delta-V values, in compliance to the current ESA margin policies.

Table 1. Current worst-case Delta-V budget for 16U4SBSP, for two Sun-synchronous operational orbits.

Phase	Δv [m/s] 500 km orbit	Δv [m/s] 600 km orbit
Formation acquisition	75	68
Operative phase	130	90
End-of-Life	0	220
TOTAL	**205**	**378**

Based on the above mission phases and analysis, several spacecraft modes have been considered, shortly summarized in Table 2.

Table 2. Spacecraft Modes for the 16U4SBSP mission.

Spacecraft mode	Envisaged operations
Nominal Mode	• Orienting solar panels towards the Sun • Tracking of desired attitude • Charging the batteries • Communications turned on
Safe Mode	• Orienting solar panels towards the Sun • Charging the batteries • Communications transmitting safe mode beacons • Communications turned on • AOCS is in Safe mode
Technology Demo Mode	• DC-RF conversion and wireless power transmission • Tracking of desired attitude • Tracking of relative and absolute navigation parameters • Communications turned on
Comms Mode	• Stored telemetry data downlink • Real-time telemetry data downlink • Telecommands uplink • Tracking of desired attitude • Orienting solar panels towards the Sun • Radiometric ranging and tracking
RWs Desaturation Mode	• Tracking of desired attitude • Orienting solar panels towards the Sun • Desaturation of RWs • Communications turned on
Maneuver Mode	• Tracking of desired attitude • Performing orbital maneuvers • Performing station keeping maneuvers • Communications turned on
Detumbling Mode	• Reduce the angular rate using ADCS or RCS • Acquire the final attitude • Communications turned on
Eclipse Mode	• All unnecessary subsystems are turned off or in standby mode • Communications turned on

3. System and Subsystem requirements

A comprehensive set of more than 400 requirements for the 16U4SBSP mission has been elaborated by the project team. Particularly relevant for the scope of this paper are the system and subsystem requirements, a selection of which is presented in Table 3 at the end of the paper. For a more detailed list of mission requirements, please refer to a companion paper presented at the same edition of the IAC conference [2].

While some system constraints were established by the ESA SysNova Challenge rules (spacecraft wet mass of no more than 36 kg, CubeSat format not larger than 16U, mission lifetime longer than 1.5 years), a major part of the subsystem requirements derived from the functionalities needed to perform the ambitious in-orbit wireless power transmission demonstration. Particularly challenging to this respect, for the spacecraft design presented in next Section, were the very stringent attitude knowledge errors required to correctly perform the beam forming operations of the CubeSats in the swarm, the extremely accurate navigation requirements, the very high amount of thermal power dissipated from the payload during the power transmission phase, and the high total impulse required to the main propulsion system in order to perform all orbital control maneuvers (including End-of-Life ones when needed).

4. 16U4SBSP spacecraft design

The following sub-sections provide an overview of the pre-Phase A design of all spacecraft sub-systems. A summary of the spacecraft mass budget is shown in Table 4 at the end of the paper, while Figures 2, 3 and 4 on next page show renderings of the full 16U4SBSP spacecraft. The total margined mass of the spacecraft, after taking into account all margins as prescribed by the ESA margin policies, is approximately 35.3 kg, which is compliant with the 36 kg maximum mass constraint set by the SysNova challenge rules. The volume of all subsystems fits with sufficient margin into the prescribed 16U CubeSat format.

The power budget for the operative phase shows that, even in a worst-case scenario in which power transmission to ground is performed every 2 orbits during eclipse (thus requiring multiple power receiving stations on ground), the available solar arrays power is sufficient to perform all operations, with a battery DoD equal to 34%. In a more realistic scenario with a single power receiving station on ground, the actual required power conditions will be less demanding. Given the very high amount of instantaneous power required in the operative phase, any possible power peaks in sunlight require less than the maximum available power from the solar arrays (144 W) and can therefore be sustained without using the batteries. In safe mode, all vital sub-systems require a power of less than 20 W, which can still be provided by the solar arrays even with an orientation of 10° towards the Sun (instead of the nominal 90°). In eclipse, with initially fully charged batteries, the required safe-mode power can be provided for a total of 17 cumulative hours without any other power source available. Finally, when undeployed, the solar arrays are capable of providing 30 W of instantaneous power, which is sufficient for the OBC, SADA and ADCS together. Therefore, the deployment of solar arrays does not require in principle any use of energy stored in the batteries

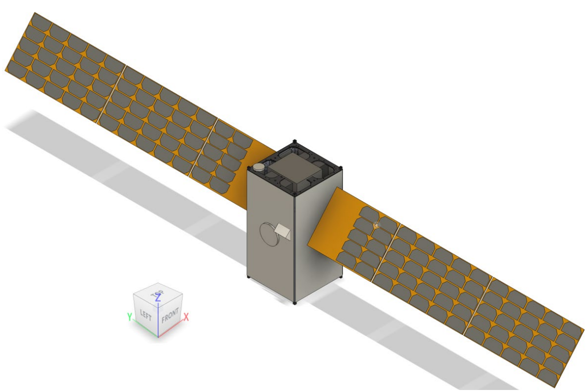

Fig. 2. Rendering of the 16U4SBSP spacecraft with cover panels, solar array wings deployed.

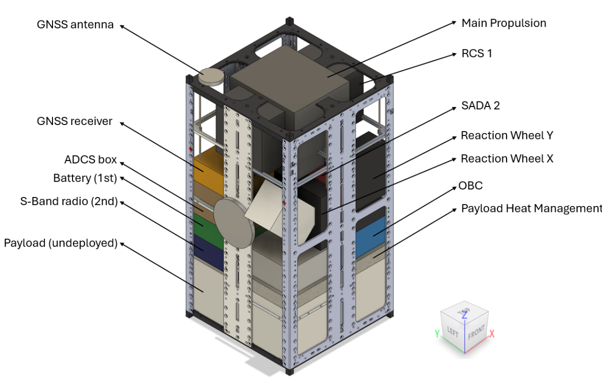

Fig. 3. X-/Y- view of the 16U4SBSP spacecraft without cover panels and solar array wings.

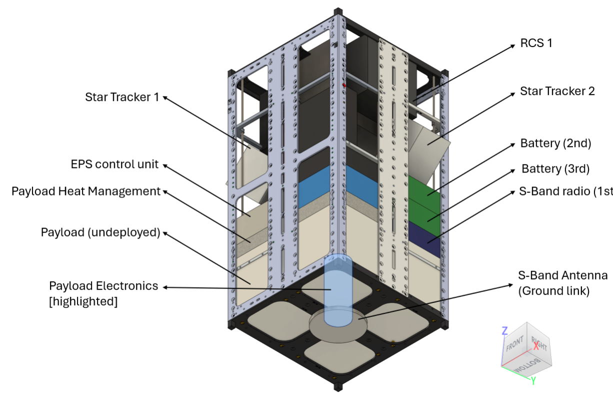

Fig. 4. Z- view of the 16U4SBSP spacecraft without cover panels and solar array wings.

4.1 Payload – Wireless power transmission system

In order to maximize the antenna size and, consequently, improve the beamforming efficiency and the beam pattern, a deployable membrane hexagonal antenna is proposed for the 16U4SBSP mission. Figure 5 shows the fully deployed antenna, constituted of individual triangular components and based on a concept jointly under development by Sirin Orbital Systems AG and its partner company Cosmobloom Inc, consisting of a hold/release mechanism, a membrane film phased-array antenna, and electronics. Each triangular component has 240 antenna elements, and a full antenna consists of 6 triangular components, each with a size of 1 m. Microstrip lines are the preferential method for feeding the RF signal into the antenna components. Initial analyses performed by the project team showed that this deployable hexagonal configuration, compared to a non-deployable rectangular antenna body-mounted on the spacecraft, allows for improving the beam efficiency by a factor in the order of 10-12. The full hexagonal antenna, including the hold/release mechanism, has an estimated mass of 4.16 kg. In its undeployed configuration, it fits in the lower part of the 16U CubeSat, as shown in Figures 3 and 4. The control electronics, including DC-RF conversion, has an estimated mass of 0.5 kg.

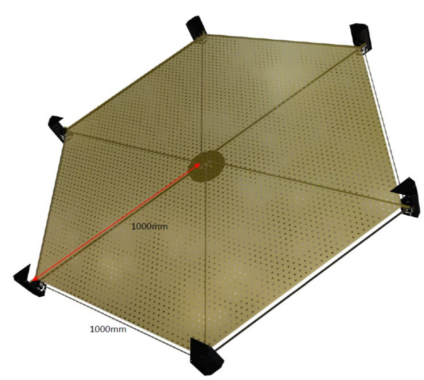

Fig. 5. Rendering of the fully deployed hexagonal 16U4SBSP antenna.

4.2 Propulsion

As indicated by the subsystem requirements in Table 3, the envisaged task for the main propulsion system is to perform slow orbital control maneuvers for which no specific maximum maneuver duration is required, but a significantly high total impulse (30000 Ns) is dictated by the requirements. A first trade-off was performed to decide which type of propulsion should be used, as shown in Figure 6. This trade-off involved the requirements on thrust level, mass and volume, as well as more general considerations related to schedule/TRL and costs. Given the high total impulse requirement and its implication on the mass and volume of the propulsion system, the final selection necessarily converged towards a single possible option, i.e. electric propulsion. In a second trade-off, involving several available COTS options, the Micro R^3 system produced by Enpulsion was selected, being the only one among all considered options, potentially capable of meeting all key subsystem requirements.

Options \ Criteria	Thrust Level	Mass	Volume	Schedule/TRL	Cost	
Chemical	*Allows for fast maneuvers*	*Not compatible to the required Delta-V*	*Probably not compatible to the required Delta-V*	*Several flight qualified options available*	*OK*	
Cold Gas	*Allows for relatively fast maneuvers*	*Not compatible to the required Delta-V*	*Not compatible to the required Delta-V*	*Several flight qualified options available*	*Generally low-cost system*	
Warm Gas	*Allows for relatively fast maneuvers*	*Not compatible to the required Delta-V*	*Probably not compatible to the required Delta-V*	*Good flight qualification*	*Generally low-cost system*	
Resistojet	*Allows for relatively fast maneuvers*	*Not compatible to the required Delta-V*	*Probably not compatible to the required Delta-V*	*Development and qualification efforts still needed*	*Higher costs for development and qualification*	
Electrical	*Low-thrust option, requires adjustments to the mission concept and Conops*	*Excellent propellant mass (high Isp), doubts on the dry mass*	*Acceptable*	*Several flight qualified options available*	*OK*	***Winner***

Excellent, exceeds requirements; Good, meets requirements; Correctable deficiencies; Unacceptable

Fig. 6. Main propulsion trade-off (type of propulsion system).

In the current spacecraft configuration, the presence of a RCS system is not strictly necessary, since not directly functional to the accomplishment of the mission needs: as described in the next sub-Section, attitude control tasks can be fully accomplished by the reaction wheels, and wheels desaturation is performed by magnetic torquers. Nevertheless, a RCS system has been included in the spacecraft for additional 6DOF control authority (for example, to support detumbling and reaction wheel desaturation if needed), and as a higher-thrust system for emergency collision avoidance maneuvers. The total impulse required for the RCS sub-system, as shown in Table 3, is calculated assuming a total of 7 emergency collision avoidance maneuvers per spacecraft during the whole mission lifetime, each modifying the orbital altitude by 0.5 km. At the nominal mission orbit this translates into a Delta-V per maneuver equal to approximately 0.27 m/s which, combined with an assumed spacecraft mass of 36 kg, translates into a total impulse for all maneuvers of approximately 70 Ns.

Similarly to what has been done for the main propulsion system, an initial trade-off for the RCS system was focused on the type of propulsion to be used, showing in this case three options as possible candidates: cold gas, warm gas or resistojets. In a second trade-off among available COTS systems, the IANUS 6DOF system (cold gas) designed by t4i for the Milani mission was selected, mainly due to its current maturity/development stage compared to the other candidates.

4.3 Attitude Determination and Control System (ADCS)

The ADCS trade-off considered several commercial systems, some under development and some with (partial) flight heritage. The critical requirements considered for the selection came from the capability of the system to cope with a 16U satellite bus with external appendages (solar panels and deployable antenna) and the size to fit in the available volume onboard. Finding a system with TRL of at least 8 for a 16U satellite bus is possible as several European small-satellite integrators have already demonstrated at least in part such capabilities. Besides the maximum satellite volume and inertia, the maximum required torque and slew were considered, based on the mission analysis and consequent subsystem requirements as shown in Table 3. Eventually the EnduroSat 16U system, including an attitude control computer and three CW5000 reaction wheels, was selected as trade-off winner. Since a star-tracker needs to be used to meet the required pointing accuracy under eclipse, the Sodern Auriga CP has been selected: this system provides a pointing accuracy better than 0.1 deg/s, while meeting a rotational rate of more than 1 deg/s.

The selection of such a large reaction wheel model as compared to the spacecraft size (500 mNms momentum storage capability, 37 mNm maximum torque per wheel) was mainly dictated by the possibility of performing even the most demanding de-tumbling and slew maneuvers using the wheels only, without any need of support from the RCS thrusters. For the de-tumbling maneuver, as per requirements, a worst-case scenario with initial tumbling rate of 30 deg/s is considered; with the maximum spacecraft moment of inertia in the initial (undeployed) configuration, estimated to be equal to 0.6 kg·m^2, this is corresponding to a total stored momentum of 314 mNms. For the slew maneuvers, requirements indicate a maximum slew rate of 3 deg/s; since the most demanding slew maneuvers are expected to require a spacecraft rotation of 90 deg, for example to re-orient the main propulsion system before performing orbital control maneuvers, this results in a maximum momentum stored in the wheel equal to 235 mNms.

4.4 Navigation

The navigation subsystem represents one of the most challenging aspects in the design of the 16U4SBSP spacecraft, mainly due to the quite stringent requirements in terms of position determination and pointing accuracy. Similarly to what has been done for the propulsion system, a first higher-level trade-off was performed to define which navigation technique is best suited to meet the given requirements, see Figure 7. The trade-off led to the conclusion that realtime GNSS (either absolute or relative) is the only possible choice to this respect, due to its performance features combined with still acceptable mass and volume characteristics. Among the available COTS option, the final selected GNSS receiver was the FUGRO SpaceStar, which meets all subsystem requirements although still at a relatively early development stage.

Criteria / Options	Pointing Accuracy	Features	Mass	Volume	Schedule/TRL	Cost	
Realtime GNSS absolute	10m - 10cm RMS	2-3 frequencies or Augmentation Service	< 0.5 kg	2U - 0.2U	Flight qualified	Expected OK	*Winner*
Realtime GNSS relative	~1cm RMS	ISL needed, RTK	< 0.5 kg	0.5 U	Flight qualification unclear	Expected OK	*Winner*
Post-processing GNSS	mm-level	Raw-data downlink, 2-5 days lead time	< 0.5 kg	0.6 U	Flight qualified	Expected OK	
Radio link	meter-level	Datarate >> 1 Mbps, special data encoding	Limited impact, ISL anyway needed	ISL required, marginal increase	High resolution to be developed	Expected OK, development required	
Optical link	cm-level	Very large datarate	~ 1-2 kg	Large	High resolution to be developed	Expected OK, development required	
Radar	cm-level	Difficult to achieve precise distance over satellite body	Sub 1kg	Large	No space heritage	Development required	

Excellent, exceeds requirements; Good, meets requirements; Correctable deficiencies; Unacceptable

Fig. 7. Navigation method trade-off.

4.5 Electrical Power System

To meet the quite challenging requirements of 16U4SBSP in terms of power generation and storage (mainly driven by the needs of the wireless power transmission system), a complete package from Kongsberg/NanoAvionics has been selected for the spacecraft, including batteries, power conditioning and power distribution units. A total of 4 stacked Li-ion battery units are used, which allows for a total battery capacity of 340.8 Wh BOL. Each battery unit provides a peak power of 100 W, which allows to achieve a total of 400 W peak power with the full four units. The power conditioning and distribution electronics is compatible to a wide range of solar panel input voltage, up to 42 V.

To optimize the power generation capabilities of the spacecraft in function of the specific needs of the 16U4SBSP mission, the project team opted for a fully customized design of the solar array wings, built upon the 30% Triple Junction GaAs Solar Cell Assembly from AzurSpace. The final solar arrays configuration consists of 2 separate wings, each with a total of 60 cells; additionally, in one of the two wings, 25 cells are present on the back side. With this configuration, a total power generation capability of 144 W (72 W per wing) is available from the deployed wings, while 30 W are available in folded configuration before deployment.

4.6 Communications

Given the specific characteristics of the mission and its concept of operations, two sets of radio and antenna need to be used by each spacecraft: one for inter-satellite link and one for ground communication. The two communication systems are expected for simplicity to be identical, therefore the same trade-offs considerations applied to both.

Since the subsystem requirements leave the door open to two possible frequency bands (S-band or X-band), an initial trade-off was performed to decide between these two frequency bands. Data throughput in either downlink or uplink conditions is not expected to be the major driving factor for the selection, considering that the primary mission objective is not associated to generating specific science products, thus only basic telemetry and telecommand information (including spacecraft position coordinates) will need to be exchanged with ground or with other spacecraft in the swarm. Therefore, the trade-off was instead focused on the mass/volume/power requirements and on other aspects such as bandwidth availability, TRL of currently available COTS systems and expected costs. The outcome is that, to meet the requirements and needs of the mission, communicating in the S-band is a preferrable option. Starting from this outcome, a representative combination of COTS radio and antenna operating in the S-band have been traded-off.

The selected S-band radio is the Syrlinks EWC31 model. It is a flight-proven model, with integrated diplexer and capable of operating in the 2200-2290 MHz frequency range (transmission) and the 2025-2110 MHz range (reception), with data rate in the range from 8 to 512 kbps and output power in the range from 27 to 36 dBm. The selected antenna is the Anywaves S-Band TT&C antenna, a patch antenna with strong flight heritage and full duplex telemetry & telecommand capabilities.

4.7 Structures and Mechanisms

A representative 16U CubeSat structure has been chosen for the current 16U4SBSP spacecraft architecture, i.e. the fully flight qualified 16U structure provided by EnduroSat. This structure is fully built in Aluminium 6082 with hard-anodized surface and offers lightweight characteristics. Some modifications to the available COTS structure will likely need to be made to accommodate the propulsion unit and the deployable antenna payload in the bottom 4U of the structure. For the deployer, the recently developed EXOpod NOVA 16U S1 dispenser from ExoLaunch is envisaged, since this specific dispenser is already qualified for a spacecraft mass up to 36 kg.

Given its very challenging power generation requirements, the 16U4SBSP spacecraft needs to be equipped with a solar array tracking mechanism, for which the μSADA system produced by the company IMT has been selected. It is based on a pointing mechanism with ± 0.3 deg pointing accuracy and full 180° rotation capabilities of both wings in both directions. It also offers a simultaneous deployment function for the two solar array wings, requiring a total power of 4.5 W during deployment, and can therefore also be used as solar arrays deployment mechanism.

4.8 Thermal Control

A preliminary thermal analysis for the 16U4SBSP spacecraft was performed, focusing on two equally important aspects: a simplified single-node steady-state analysis for the whole spacecraft, to predict the range of temperatures expected during operation and define a coating strategy on the external surface of the spacecraft for passive thermal control; and the definition of a strategy and sub-system design for dissipating the significant amount of heat produced by the payload during wireless power beaming.

The simplified thermal equilibrium analysis has been performed by modelling the spacecraft as a single-node steady-state lumped mass and the solar array wings as parallelepiped shapes, with no thermal interactions between spacecraft body and solar arrays. For the external surface of the spacecraft, after an iterative trade-off conducted based on the results of the simplified model, the following combination of materials has been assumed: 65% vapor-deposited Silver coating, 30% Silvered fused silica (Optical solar reflector), 5% Aluminized aclar film. Worst-case equilibrium temperatures of 43.5 °C for the spacecraft body and 89.8 °C for the solar array wings (sunlight), -8.9 °C for the spacecraft body and -72.3 °C for the solar array wings (eclipse) were estimated by the analysis. Although the predicted thermal conditions are generally acceptable for a major part of the sub-systems and components, potential criticalities are still present for some components, namely the batteries, main propulsion system, RCS thrusters, and star trackers, mainly under cold/eclipse conditions. For this reason, the currently envisaged spacecraft architecture assumes that a total of 5 active heaters are included in the spacecraft, each with 5 W heating power available, to be strategically placed close to the most critical sub-systems and components, for active thermal control in emergency or non-ideal conditions.

With an input transmission power while beaming of 331 W and an assumed 60% efficiency of the DC-RF conversion process, a total power of 132.4 W has to be dissipated on board of the 16U4SBSP spacecraft in Technology Demonstration Mode (i.e., while beaming power to ground). The maximum duration of each power beaming demonstration is assumed to be 10 minutes, compatible with the expected passing time over potential receiving stations on ground, leading to a maximum dissipated heat energy equal to 22.06 Wh. This considerable amount of dissipated energy requires a dedicated power management system for the payload, for which two different options have been considered: radiators and heat storage units based on phase-change materials.

With an assumed radiator emissivity equal to 0.95 and radiator temperature of 300 K, the required radiator area to dissipate an instantaneous power of 132.4 W is equal to approximately 0.3 m^2. This is incompatible with the available external surface on a 16U CubeSat, even under the unrealistic assumption that the entire surface on the lateral faces of the spacecraft would be fully covered by radiators. Therefore, a deployable radiator would be required. However, given the presence of other mechanisms and deployables on the spacecraft, and given the intrinsic design complications associated to the use of a deployable radiator, this solution has been considered too risky in terms of reliability and therefore discarded. The other option assumes that a phase-change material is used to store the dissipated energy, in the form of latent heat while melting the phase-change material. The energy would then slowly be released back during the remaining part of the orbit in eclipse, thus also contributing to mitigating the effects of the cold environment in this part of the orbit. The phase-change material selected for the preliminary design of the heat management system is Tetracosane ($C_{24}H_{50}$), a paraffin characterized by a good compromise between melting temperature (50.6 °C) and latent heat for melting (255 kJ/kg). Worst-case calculations showed that the required mass of Tetracosane, to dissipate the given 22.06 Wh of generated heat energy while beaming power to ground, is 0.312 kg. Considering the density of Tetracosane, and including a 50% margin to take into account possible variations of density due to the phase change and other internal metallic elements (such as fins or protrusions) to better distribute the dissipated heat through the whole mass of phase-change material, this leads to a total volume of the payload heat management box equal to 0.584U. Including the external part of the box, assumed to be made of aluminium, a total mass of 0.802 kg is estimated for the full payload heat management box.

4.9 Command and Data Handling

Among several identified COTS options for the On-Board Computer, the Argotec FERMI OBC was selected as trade-off winner, mainly due to its superior data storage properties which might result in a good added value in case particularly high data-demanding science goals are chosen as possible secondary mission objectives. This system is originally designed for deep-space CubeSat missions, which allows for superior properties in terms of radiation hardening. The CPU features a dual-core processor with a wide range of available data interfaces and supply voltages, and dedicated on-board software including, among other features, a built-in support to FDIR functions. A total of 16 GB are available as embedded mass memory.

5. Conclusions

The 16U4SBSP mission, one of the selected concepts of the ESA SysNova challenge “Innovative Missions Concepts enabled by Swarms of CubeSats”, has the primary objective to validate, with a small-scale mission, the Wireless Power Transfer (WPT) model developed by the mission consortium and, in this way, confirm that it is feasible and convenient to provide space-based solar power by means of a larger constellation of spacecraft. In order to achieve this ambitious objective, the mission aims at designing, developing, commissioning and launching a commercial space-based solar power demonstrator, based on a distributed swarm of 7 identical CubeSats.

The 16U4SBSP spacecraft is a 16U CubeSat equipped with a deployable hexagonal antenna constituted of triangular elements each of 1 m size, for improved beamforming efficiency and beam pattern. The mission implements a sophisticated formation flight strategy and is possible only if extremely good accuracy in terms of spacecraft attitude, pointing and orbital position knowledge is ensured. Nevertheless, in the current design, the 16U4SBSP spacecraft is mostly based on advanced COTS CubeSat technologies. In this way, the mission will also serve as a demonstrator for the use of viable, low-cost CubeSats platforms for ambitious swarm missions.

In this paper, the 16U4SBSP spacecraft design as resulting at the end of the pre-Phase A study has been described in detail, highlighting the main choices for the different subsystems and their motivations. According to the currently envisaged development and integration plan, in case all necessary funding and support is secured, the mission could be launched as early as the first half of 2029.

Table 3. Selection of pre-Phase A system/subsystem requirements for the 16U4SBSP mission.

ID	Title	Requirement
SYS.010	Mission Lifetime	The system shall have a design life longer than 1.5 years (TBC).
SYS.040	ITAR	The system should be free of components as described in ITAR Regulations.
SYS.050	System mass	Each spacecraft shall have a wet mass of no more than 36 kg.
SYS.060	Spacecraft format	Each spacecraft shall be based on the 16U CubeSat format.
ADCS.010	De-tumbling	The ADCS shall be able to de-tumble the spacecraft from tip-off rates of 30 deg/s (TBC) on each axis down to at least 1.5 deg/s on each axis.
ADCS.032	APE power transmission	The spacecraft shall provide an Absolute Performance Error (APE) with respect to the body-fixed frame lower than 0.05 deg 1-sigma half cone (TBC) when performing wireless power transmission.
ADCS.043	RPE power transmission	The spacecraft shall provide a Relative Performance Error (RPE) with respect to the body-fixed frame lower than 0.25 deg 1-sigma over 1 s (TBC) when performing wireless power transmission.
ADCS.050	AKE	The spacecraft shall provide an Absolute Knowledge Error (AKE) lower than 72 arcsec 1-sigma half cone (TBC).
ADCS.060	Solar array pointing requirement	The spacecraft attitude shall ensure that the solar arrays point towards the Sun with an accuracy better than 15 deg 1-sigma half angle cone.
ADCS.070	Slew rate	The ADCS shall perform slew maneuvers with a maximum slew rate of 3 deg/s (TBC).
CDH.020	On-board monitoring	The C&DH shall allow to monitor on-board parameters, raising an event when the parameter value exceeds defined limits
CDH.070	FDIR standard	The C&DH shall implement FDIR (Fault Detection Isolation and Recovery) functionality for subsystems and operating mode failures, using the ECSS-E-ST-70-11C standard (Section 5.7.5) as a guideline.
CDH.090	ADCS control	The C&DH shall command the ADCS and propulsion system to control the spacecraft attitude and orbit.
CDH.460	Mission data	The following data shall be time-stamped and stored on the spacecraft for later downlink to ground during the mission: Full state vectors; Attitude sensor data; Navigation sensor data; Control actuator data; Platform housekeeping data; FDIR data.
COMM.010	Ground link	The spacecraft shall support a radio communication link with the ground.
COMM.020	Inter-satellite link	The spacecraft shall support radio communication link with the other spacecraft in the swarm.
COMM.030	Uplink and downlink frequency	The spacecraft shall support X band and S band frequency (TBC), for both uplink and downlink communication links.
COMM.040	Ranging	The spacecraft shall support two-way ranging with Earth.
NAV.010	Navigation method	The mission shall implement a GNSS receiver (TBC) for coarse navigation tasks, and inter-satellite link for accurate navigation.
NAV.020	Knowledge uncertainty relative position	The knowledge uncertainty on the relative position between two CubeSats in the swarm shall be 0.05 m 1-sigma (TBC).
NAV.030	Relative position controllability	While performing wireless power transmission, the relative position between two CubeSats in the swarm shall be controlled with an uncertainty of no more than 1% (TBC) of the inter-satellite distance.
NAV.040	Update rate relative position	The relative position between two CubeSats in the swarm shall be known with an update rate of 10 Hz (TBC).
NAV.050	Control window relative velocity	The relative velocity between two CubeSats in the swarm shall be kept in the range 1-10 cm/s (TBC).
NAV.060	Coarse navigation accuracy	For coarse navigation tasks, the 3-sigma error in the range measurements shall be better than 50 m (TBC).
PLD.010	Transmission frequency	The payload shall transmit power at a RF frequency of 5.8 GHz (TBC).

PLD.020	DC-RF conversion efficiency	The DC-RF conversion efficiency shall be no less than 60% (TBC).
PLD.030	Transmission efficiency	The RF transmission efficiency (excluding free space losses) shall be no less than 90% (TBC).
PLD.071	Thermal control payload	The payload thermal control system shall be able to dissipate a maximum power of 132 W (TBC) while the payload is active.
PLD.072	Beam steering	Electronic beam steering and control shall ensure phase control and pointing along elliptical or circular orbits, with an accuracy of 40 milliradian 1-sigma.
PROP.001	Main propulsion tasks	The main propulsion system shall be used to perform slow orbital control maneuvers, for which no specific maximum maneuver duration is required.
PROP.010	Overall total impulse	The main propulsion system shall provide a total impulse of at least 30000 Ns (TBC) for nominal orbital manoeuvres during the spacecraft lifetime, plus end of life manoeuvres including space debris mitigation.
PROP.020	Max Thrust	The maximum thrust delivered by each thruster shall be no more than 5 mN (TBC).
PROP.022	Min Thrust	The minimum thrust delivered by the propulsion system shall be at least 0.5 mN (TBC).
PROP.120	Propulsion telemetry	The main propulsion system shall send its telemetry to the ADCS subsystem.
POW.010	Solar panels tracking mechanism	The EPS shall be equipped with a solar panels tracking mechanism.
POW.020	Total power generated	The EPS shall allow for power generation of at least 100 W (TBC) BOL while in sunlight with the solar panels at 340 K (TBC), measured at the flight connector of the solar panel tracking mechanism.
POW.030	Power storage	The EPS shall allow for a power storage capability of at least 200 Wh BOL (TBC).
POW.031	Battery Instantaneous Power	Battery shall be able to supply at least 350 W (TBC) of instantaneous power.
POW.033	Battery state of charge	The design DoD of the batteries shall be 40% (TBC) or lower.
RCS.001	RCS propulsion tasks	The RCS propulsion system shall be used to perform 6DOF reaction control tasks (including support to de-tumbling and RWs desaturation), and for orbital control maneuvers requiring fast response time, such as collision avoidance maneuvers.
RCS.010	Overall Total Impulse	The RCS propulsion system shall be able to provide a minimum total impulse of 70 Ns (TBC).
RCS.020	Thrust	The RCS propulsion system shall be able to guarantee a thrust of no less than 10 mN (TBC) per thruster.
RCS.030	Number of thrusters	The RCS propulsion system shall have at least 4 thrusters.
RCS.130	RCS propulsion telemetry	The RCS propulsion system shall send its telemetry to the ADCS subsystem.

Table 4. 16U4SBSP mass budget (margined) as resulting from the pre-Phase A study.

Component	# units	Margin	Mass/unit [kg]	Mass/margined [kg]
Power transmission antenna	1	20%	4.16	4.992
Power transmission electronics	1	20%	0.5	0.6
Heat dissipation system	1	20%	0.802	0.963
On-Board Computer	1	5%	0.534	0.561
GNSS receiver	1	10%	0.45	0.495
S-Band radio	2	5%	0.17	0.357
S-Band antenna	2	5%	0.139	0.292
Structure	1	10%	3	3.3
Active thermal control	5	10%	0.005	0.028
Power distribution/conditioning	1	10%	0.15	0.165
SADA + release mechanism	2	10%	0.56	1.232
Solar arrays	2	20%	1.17	2.808
Batteries	4	10%	0.5	2.2
ADCS box	1	10%	3.9	4.29
RCS propulsion (dry)	2	5%	0.5	1.05
Main propulsion (dry)	1	5%	2.6	2.73
			Total dry mass (without harness)	26.062
			Harness (assumed 8% of the total dry mass)	2.085
			Total dry mass	28.147
			System margin	20%
			Dry mass (margined)	33.776
			Propellant mass	1.5
			Total Wet Mass	**35.276**

Acknowledgements

The work described in this paper has been funded by the European Space Agency under the contract n. 4000142405/23/NL/AS/ov. The authors would like to particularly acknowledge the contribution of the Agency in reviewing and iterating the pre-Phase A 16U4SBSP design. The input received from external experts throughout the design process has been extremely valuable and their role is highly appreciated, with a special acknowledgement to Cosmobloom, EnduroSat, Rocket Factory Augsburg, Kyoto University and University of Málaga for their expert contributions. Finally, the authors would like to thank all the students and collaborators who have given invaluable contributions to this early phase of the mission design.

References

[1] European Space Agency, 2023, “SoW Innovative Mission Concepts Enabled by Swarms of CubeSats – OSIP Campaign”, ESA-TECSF-SOW-2023-001164, is. 3.0

[2] M. Madi, et al. , 2024, “The 16U4SBSP Mission: a Swarm of CubeSats for Demonstrating Space-Based Solar Power in Earth Orbit”, IAF 75th International Astronautical Congress, Milano, Italy

[3] M. Vasile, et al., 2012, “Fractioned solar power satellite for regional coverage”, IAF 63rd International Astronautical Congress, Naples, Italy

[4] J. N. Pelton, 2019, “Space-based solar power satellite systems,” Space 2.0: Revolutionary Advances in the Space Industry, pp. 103–114

[5] S. Mihara, et al., 2018, “The Road Map toward the SSPS Realization and Application of Its Technology”, IAF 69th International Astronautical Congress, Bremen, Germany

[6] K. Itoh, N. Sakai, 2021, “Highly Efficient High-Power Rectenna Techniques,” Proc. RFIT2021

[7] A. Golkar, 2023, “Overview of the new space cubesat market,” in Next Generation CubeSats and SmallSats, pp. 605–620, Elsevier

[8] S. Liu, et al., 2022, “A survey on cubesat missions and their antenna designs,” Electronics, vol. 11, no. 13

[9] R. Trabert, et al., 2010, “The extendable solar array system: a modular nanosatellite power system,” in AIAA/AAS Astrodynamics Specialist Conference, p. 7654

[10] V. Giralo, S. D’Amico, 2019, “Distributed multi-gnss timing and localization for nanosatellites”, Navigation, vol. 66, no. 4, pp. 729–746