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INTERPLANETARY CUBESATS MISSION TO EARTH-SUN LIBRATION POINT FOR SPACE WEATHER EVALUATIONS

Maria Antonietta Viscio
Politecnico di Torino, Italy, maria.viscio@polito.it

Nicole Viola
Politecnico di Torino, Italy, nicole.viola@polito.it
Sabrina Corpino
Politecnico di Torino, Italy, sabrina.corpino@polito.it
Fabrizio Stesina
Politecnico di Torino, Italy, fabrizio.stesina@polito.it
Christian Circi
University "La Sapienza", Italy, christian.circi@uniroma1.it
Silvano Fineschi
National Institute for Astrophysics, Italy, fineschi@oato.inaf.it
Federico Fumenti
Deutsches Zentrum für Luft- und Raumfahrt e.V. (DLR), Germany, federico.fumenti@gmail.com

The paper deals with an interplanetary CubeSats mission to Earth-Sun Libration point. CubeSats are an interesting alternative to larger science satellites to accomplish both scientific and technological tasks in deep space, as proved by the growing interest in this kind of application within the scientific community and, most of all, at NASA. Indeed such systems allow less costly missions, due to their reduced sizes and volumes, and consequently less demanding launches requirements.

The CubeSats mission presented in this paper is aimed at supporting space weather evaluations that represent quite a critical issue especially for what concerns the human exploration of space beyond Earth orbit where the protection of the Earth magnetic field is not available anymore. The mission envisages the deployment of 6U CubeSats system in one of the Earth-Sun Lagrangian Points, where solar observations for in situ measurements of space weather to provide additional warning time to Earth can be carried out. The proposed mission is also intended as a technology validation mission, giving the chance to test advanced technologies, as for example the solar sail, which is envisaged as propulsion system, and specific radiation dosimeters and advanced materials, foreseen to further investigate the space radiation environment and validate them in view of future implementation in human missions.

One of the objectives of the work is to identify the required subsystems and equipment, needed to accomplish specific mission objectives and to investigate the most suitable configuration, in order to be compatible with the typical CubeSats (multi units) standards.

The paper starts from the definition of the mission, in terms of objectives, requirements and mission analysis. Then it focuses on the CubeSats system, describing its configuration and analysing the subsystems composing it. Finally, the most advanced technologies (e.g. solar sails) implemented in the CubeSats design are discussed.

I. INTRODUCTION

Interplanetary CubeSats could enable small, low-cost missions beyond low Earth orbit (LEO). CubeSat is typically characterized by 10cm x 10cm x 10cm dimensions and a mass not exceeding 1.33 kg; they can also be arranged in double and triple units systems.

Although a large number of CubeSats have already been developed and launched into Earth orbit; none have accomplished an interplanetary mission. Since big missions are usually very costly, relying on CubeSats could be an interesting alternative to accomplish both scientific and technological tasks in deep space, as proved by the growing interest in this kind of application in the scientific community and most of all at NASA.

The CubeSats mission presented in this paper envisages the deployment of a 6U CubeSats system in one of the Earth-Sun Lagrangian Points. It is aimed at supporting measurements of space weather, which is quite a critical issue especially for what concerns the human exploration of space beyond Earth orbit where the protection of the Earth magnetic field is not available anymore. Moreover, the mission is intended as a technology validation mission, with the aim of testing advanced technologies in view of future implementation in larger missions (e.g. solar sails, far distance telecommunications).

Regarding the support to future exploration missions, another issue taken into consideration is the space radiation environment. In this regard, travelling outside the Van Allen belts, the CubeSats system gives the opportunity for further investigations: radiation dosimeters and advanced materials are envisaged to be implemented, in order to test their response to the harsh space environment, even in view of future implementation on manned spacecrafts.

The work has been developed as collaboration between Politecnico di Torino, University "La Sapienza" (Rome), "Osservatorio Astrofisico di Torino" (Astrophysical Observatory of Torino) and DLR (Deutsches Zentrum für Luft- und Raumfahrt) in Bremen.

The AeroSpace Systems Engineering Team, ASSET, at the Department of Mechanical and Aerospace Engineering of Politecnico di Torino has been working for almost a decade at small satellites programs. In February 2012 e-st@r-I was successfully injected into Low Earth Orbit (LEO) by Vega Launch Vehicle during its maiden flight. E-st@r-I was the first Italian cubesat injected into orbit and it has been entirely developed by undergraduate and most of all by graduate and PhD students under the supervision of researchers and professors, with educational and technological/engineering objectives [1]. The e-st@r program followed the PiCPoT program, which was developed at Politecnico di Torino in the 2000s and ended with the unfortunate launch of PiCPoT nano-satellite in 2006, which never reached LEO because of a failure of the launcher [2]. Both PiCPoT and e-st@r programs represent a valuable heritage for the current small satellites activities at Politecnico di Torino [3].

The paper starts from the definition of the mission (see section II), in terms of mission statement, mission objectives, requirements and mission analysis. Then it focuses on the 6U CubeSats system (see section III), describing its configuration and analyzing the subsystems and main equipment composing it, and on its technological challenges (see section IV). Eventually main conclusions are drawn.

II. CUBESATS MISSION

II.I Mission Objectives

According to the typical conceptual design process in Systems Engineering, the mission statement, which is reported hereafter, has been firstly established:

To perform solar observation and in-situ space weather measurements from an Earth-Sun Lagrangian point region, pursuing a low-cost approach relying on interplanetary CubeSats and providing a platform for advanced technologies test.

Starting from the mission statement, the mission objectives have been derived. Mission objectives can be split into two different groups:

1. Scientific objectives:
   - *to observe the Sun*
   - *to perform plasma measurements*
   - *to perform radiation measurements*
2. Technological objectives:
   - *to develop a low-cost CubeSats platform*
   - *to implement solar sail propulsion*
   - *to communicate to Earth from very distant region (Earth-Sun L1)*
   - *to collect, store, manage and send to Earth large quantity of scientific data.*

II.II Mission Requirements

Once the broad goals of the system, represented by the mission objectives, had been identified, the system requirements have been defined. On the basis of the system requirements, the conceptual design process of the 6U CubeSats system has evolved through the mission analysis and the system architecture, which consists of two main tasks: Functional Analysis and System Sizing, which is currently under way.

In order to proceed with the sizing of the system the top-level requirements had to be assessed. Hereafter, a summary of the most significant ones is reported.

- Functional requirements
  - The system shall perform an interplanetary mission to the first Earth Sun Lagrangian point.
  - The system shall be provided with interfaces with the launcher.
  - The system shall withstand the launch loads.
  - The system shall withstand the deep space environment.
  - The system shall perform plasma measurement.
  - The system shall take pictures of the Sun.
  - The system shall perform radiations measurements (total ionizing dose).
  - The system shall allow communications with Earth.
    - command data (uplink)
    - telemetry data (downlink)
    - scientific data (downlink)
- Performance requirements
  - The system shall be compliant with 6U CubeSats standards
    - maximum envelope: 20mm x 30mm x 10mm
    - maximum total mass: 6kg
  - The total required power shall not exceed 50W.
  - The max required data rate shall not exceed 500kbps*.

* This value could require to be updated if, after a first sizing of the communications subsystem, it does

II.III Mission Analysis

The 6U CubeSats system motion is modeled as a circular restricted three-body problem (CR3BP), in which Sun and Earth are the massive bodies moving in circular orbits around their center of mass. The CubeSats system has instead negligible mass, thus it is supposed to move in the resulting force field without affecting the motion of the primaries [4, 5, 6]. The solution of the CR3BP is characterized by the presence of 5 points in which the acting forces are balanced canceling each other and allowing the third body to keep the position without requiring any corrective maneuver. Unfortunately only 2 of these 5 equilibrium points are stable thus, given a small body occupying an unstable point or orbiting around it, even a small perturbation can cause its departure making the motion unbounded. To bound the motion in the vicinity of an unstable point, corrective maneuvers are required [7, 8, 9, 10, 11]. In this paper the motion around the L_1 unstable point is considered envisaging the third body, i.e. the 6U CubeSats system, equipped with an ideal solar sail (an ideal solar sail reflects all the incoming radiation and is not interested by deformation).

The motion can be described in a Cartesian reference frame *Oxyz* with the origin fixed in the system barycenter, with the *xy*-plane coinciding with the plane of primaries motion and with the *x*-axis oriented along the Sun-Earth direction. Assuming as unity the distance between the primaries, the mean angular rate of the system and the sum of the primaries masses, the motion of the CubeSat can be described in non-dimensional units through the following system of differential equations

$$\begin{pmatrix} \dot{\mathbf{r}} \\ \dot{\mathbf{v}} \end{pmatrix} = \begin{pmatrix} \mathbf{v} \\ -2\mathbf{\Omega} \wedge \mathbf{v} + \nabla^T U(\mathbf{r}) + a_c \left(\dfrac{\mathbf{r}_1}{\|\mathbf{r}_1\|} \cdot \mathbf{n} \right)^2 \mathbf{n} \end{pmatrix}$$

where:

- $\mathbf{r} = (x, y, z)$ denotes the position
- $\mathbf{v} = (v_x, v_y, v_z)$ denotes the velocity
- $\mathbf{\Omega}$ denotes the system angular rate
- $U = \dfrac{1-\mu}{\|\mathbf{r}_1\|} + \dfrac{\mu}{\|\mathbf{r}_2\|} + \dfrac{1}{2}(x^2 + y^2)$ denotes the potential function
- $\mu = \dfrac{M_{Earth}}{M_{Sun} + M_{Earth}}$ denotes the mass ratio

not result compatible with the required 6U CubeSats system overall dimensions.

- $\mathbf{r}_1 = (x + \mu, y, z)$ denotes the position wrt the Sun
- $\mathbf{r}_2 = (x - (1-\mu), y, z)$ denotes the position wrt the Earth
- a_c denotes the sail characteristic acceleration, that is the acceleration provided at 1 AU from the Sun
- $\mathbf{n} = (n_x, n_y, n_z)$ denotes the unit-vector which is normal to the sail surface.

The attitude of the sail is described through two angles α and β and an orthonormal rotating reference frame $Cx_vy_vz_v$ as shown in figure 1.

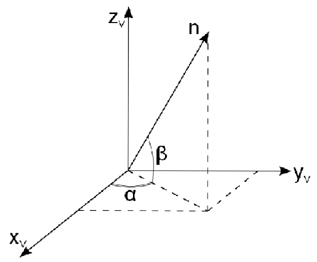

Fig. 1: Solar Sail Attitude

The $Cx_vy_vz_v$ frame has the origin in the centre of the sail surface, while the three axes are defined as follows

$$\mathbf{x}_v = \frac{\mathbf{r}_1}{\|\mathbf{r}_1\|} \quad \mathbf{y}_v = \frac{\mathbf{z}_v \wedge \mathbf{x}_v}{\|\mathbf{z}_v \wedge \mathbf{x}_v\|} \quad \mathbf{z}_v = \frac{\mathbf{r} \wedge \mathbf{x}_v}{\|\mathbf{r} \wedge \mathbf{x}_v\|}.$$

The sail attitude and the satellite path have been obtained solving an optimal control problem with the Direct Collocation with Non Linear Programming (DCNLP) approach [12, 13, 14].

In defining the optimization process, a Halo orbit is used as initial guess for the trajectory. A Halo orbit is an approximated solution of the CR3BP characterized by the equality of the in-plane and out-of-plane motion frequencies and can be computed using the approach shown by Richardson [15]. For the L1 point of the Sun-Earth system, Halo orbits have a period T of approximately 177 days, which is roughly half a year, hence to simulate a one-year CubeSat trajectory tests for 2T have been conducted.

In order to obtain a trajectory as close as possible to a periodic orbit, the optimal control problem has been solved minimizing the following performance index

$$J = \Delta\mathbf{r} + \Delta\mathbf{v}$$

which represents the difference between the initial and the final state, evaluated considering both the difference between the positions and between the velocities.

In addition constraints on the control vector have been imposed to limit the sail attitude rates to 5 degrees per day.

It is finally worth pointing out that no Halo station-keeping has been performed; Halo orbits have only been used as initial guess for the final optimal trajectory.

III. CUBESATS SYSTEM CONFIGURATION

III.I Functional Analysis

The Functional Analysis is a fundamental tool of the design process to explore new concepts and define their architectures. When systems engineers design new products, they perform Functional Analysis to refine the new product's functional requirements, to map its functions to physical components, to guarantee that all necessary components are listed and that no unnecessary components are requested and to understand the relationships among the new product's components [16].

Primary results of Functional Analysis are the functional tree and the product tree: the former identifies the basic functions, which the system has to be able to perform, while the latter individuates all system physical components, which are able to carry out the basic functions. In other words, these components may be the equipment or the subsystems, which make up the whole system.

According to the Functional Analysis, once the basic functions had been identified, the components to perform those functions have been selected by means of the so-called functions/components (or functions/devices) matrix. The functions/components matrix has therefore been used to map functions to physical components. Figure 2 illustrates the functions/components matrix for the complete 6U CubeSats system.

DEVICES / FUNCTIONS	Structure S/S				Electrical Power S/S					Thermal Control S/S					Command and Data Handling S/S				Communications S/S				Attitude & Orbit Det. & Control S/S				Mission Observation S/S			
	Radiation shielding	MDPS panels	Primary structure	Secondary structure	Solar arrays	Batteries	Power distribution unit	Regulators	Converters	Radiators	Heaters	Coolers	MLI	Surface finishes	Acquisition unit	On-board computer	Memories	Watchdog timer	Antenna/optical photoreceptor	Data Bus	Receiver/Transmitter	Auto-tracking equipment	Orbit sensors	Thrusters/Solar sails	Attitude sensors	Attitude Actuators	Camera	Spectrometer	Magnetometer	Dosimeter
Radiation protection	X																													
Micro meteoroids and debris protection		X																												
Ground handling loads withstanding			X																											
Flight accelerations withstanding			X																											
Operational loads withstanding			X	X																										
Power generation					X																									
Energy storage						X																								
Power distribution							X																							
Power regulation and control								X	X											X										
Heat rejection										X																				
Local heat sink											X	X																		
Thermal insulation													X	X																
Commands processing															X	X	X			X										
Telemetry and mission data processing															X	X	X			X										
Attitude control functions															X	X	X			X										
Computer health monitoring provision																X		X												
Command reception																			X	X	X									
Telemetry transmission																			X	X	X									
Orbit determination (navigation)																X							X							
Orbit control (guidance)																X								X						
Attitude determination																X									X					
Attitude control																X										X				
Sun/Deep Space observation																											X			
Plasma measurements																												X	X	
Radiation measurements																														X

Fig. 2: Functions/Components Matrix

As result of the Functional Analysis the assessment of the subsystems and components needed to accomplish the mission has been derived. In summary, the following subsystems compose the 6U CubeSats system:

- structure, which supports all other spacecraft subsystems, and includes the mechanical interfaces with the launcher and the ground support equipment interfaces (to be defined);
- electrical power subsystem, which is in charge of providing, storing, distributing and controlling the spacecraft electrical power; it mainly consists of solar cells mounted on the external surfaces of the system as power source, Li-ion batteries for the energy storage and power distribution unit;
- thermal control subsystem, designed to maintain all spacecraft and payload components and subsystems within their required temperature limits for each mission phase; for this mission a passive solution is envisaged;
- command and data handling subsystem, which receives, validates, decodes, and distributes commands to other spacecraft systems and gathers, processes, and formats spacecraft housekeeping and mission data for downlink;
- attitude and orbit determination and control subsystem, needed to stabilize the vehicle and orient it in desired directions during the mission despite the external disturbance torques acting on it; solar sails are exploited for orbit control;
- communications, which provides the interface between the spacecraft and the ground systems, transmitting both payloads mission data and spacecraft housekeeping data; for an interplanetary CubeSats mission optical communication is likely to be implemented, in order to be compliant with mission requirements and constraints (see section "*IV.II Communications*" for more details);
- harness;
- mission observation subsystem, which includes the scientific instruments for Sun observation and plasma measurements (see section "*III.II Mission Payloads*").

Besides the allocation of the subsystems, one of the main issues related to CubeSats is how to fit big science within a small package - namely power, mass, volume, and data limitations. One of the objectives of the work is therefore to identify and size the required subsystems and equipment, needed to accomplish specific mission objectives, and to investigate the most suitable configuration, in order to be compatible with the typical CubeSats (multi units) standards.

A reference system able to fulfill the scientific objectives of the proposed mission may consist of:

- 2U occupied by the scientific payloads;
- 2U for the solar sails;
- 1U devoted to telecommunications;
- 1U for the power subsystem, attitude control system and command and data handling.

The following section focuses on the description of the scientific payloads, which occupy up to two of the CubeSats system units.

III.II Mission Payloads

In this section a brief overview of the scientific instruments to be included in the system, according to the mission objectives, is reported [17].

Specifically, the types of instruments to be considered are:

- Plasma Instruments, for plasma measurements;
- Radiation Dosimeters and Advanced Materials, to investigate the space environment and validate technologies in view of future implementation in human missions;
- Imagers/Cameras, to take pictures of the Sun.

For each instruments class, several options have been considered and among them only the most significant ones have been selected, also according to constraints deriving from the CubeSats standards. In particular, all the scientific payloads shall fit 2U CubeSat sizes (10cm x 10cm x 20cm, 2kg).

Hereafter, the main features of the instruments are discussed and the justification for the selection of specific ones is reported.

Two instruments are envisaged to perform measurement of the plasma environment, a magnetometer and a plasma Spectrometer

The reference magnetometer considered for this mission is a tri-axial magnetometer utilizing Anisotropic Magneto-Resistance (AMR) [18]. It is a low cost magneto-resistive magnetometer designed for use in LEO small satellites and CubeSats, with very low mass and small size. Its main features are listed in table I.

Mass	Volume	Power	Data
Sensor: 15g Electronics: 150g	Sensor: 10x10x5mm Electronics: 90x30x11mm	Power consumpt.: 400mW Power supply: +5V and +15V DC or 28V unregulated option	Measurement range: +50,000nT to -50,000nT Sensitivity: 10nT Update rate: up to 10Hz Data Rate: 140bps

Table I: Magnetometer features

The reference spectrometer is an Ion and Neutral Mass Spectrometer (INMS) [19, 20], that is a miniaturised analyser designed for sampling of low mass ionised and neutral particles in the spacecraft ram direction. The key sensor components consist of a collimator/ion filter, an ioniser and a charged particle spectrometer. Particles enter the aperture into the ion filter region where charged particles can be rejected. This is followed by a series of baffles for collimation and further charged particle suppression. Collimated neutral particles are subsequently ionised in the ionizer by a 50 eV electron beam followed by mass selection in the analyser. The spectrometer can be operated in different modes, optimised for ions or neutral particle analysis. The INMS main features are listed in table II.

Mass	Volume	Power	Data
Mass: 350g	Envelope: 100x100x50mm (½U)	Power consumption: 500mW	Data Rate: ~23bps

Table II: INMS features

As introduced before, the CubeSats mission represents an opportunity to study the deep space environment, and in particular to test the response of specific materials, which can be used to shield the spacecraft.

Radiation micro dosimeters are envisioned [21], which are compact hybrid microcircuits which directly measure the total ionizing dose absorbed by an internal silicon test mass. The test mass simulates silicon die of integrated circuits on-board a host spacecraft in critical mission payloads and subsystems. By accurately measuring the energy absorbed from electrons, protons, and gamma rays, an estimate of the dose absorbed by other electronic devices on the same vehicle can be made. The dosimeters' main features are listed in table III.

The dose of radiation accumulated on a system will depend on the shielding capability of the material used to shield.

The shielding effectiveness depends on the chemical composition of the material (for example hydrogen is very efficient shielding and therefore materials with high hydrogen concentration shall be preferred), and according to this, very different masses of shielding could be needed, to meet the requirements on the maximum absorbed dose, while considering different materials.

Mass	Volume	Power	Data (type/quantity)
Mass: 20 g	Envelope: 35x25x10 mm	Power consumption: 280mW Electric I/F: 10 mA at 13-40 VDC	Measures up to 40 krads Data Rate: 1 Byte/s

Table III: Radiation Micro Dosimeter features

In the CubeSats mission here discussed, two different materials are envisaged to be implemented and tested, through dosimeters' measurements: Kevlar [22] and High Density Polyethylene (HDPE) [23], which indeed have good shielding performances.

As final configuration, three dosimeters are envisioned, positioned in three different spots. Two of them are coupled with Kevlar and HDPE covers, in order to measure the shielding capabilities of the two materials.

In particular, it is assumed to have two equal tiles having a thickness of 20mm for both materials (each tile is 50x50x20mm, which corresponds to 72g for Kevlar and 48g for polyethylene).

A NanoCam C1U [24] is finally envisaged to take pictures of the Sun. It is a high performing camera system fitting a single unit cubesat, based on a CMOS technology. Its main features are listed in table IV.

Mass	Volume	Power	Data (type/quantity)
Mass: 170 g	Envelope: 96x90x58 mm	Power consumption: Idle: 360mW Image acquisition: 634mW Image processing 660mW Supply voltage: 3.3V	CMOS camera Data Rate: 400kbps

Table IV: NanoCam C1U features

IV. TECHNOLOGICAL CHALLENGES

The enabling technologies for this kind of mission mainly regard the solar sail control and navigation, deep space tracking and telecommunications

IV.I Solar Sails

In the last decade the possibility to execute maneuvers without requiring propellant, but exploiting an unlimited source like the solar radiation pressure, aroused more and more interest in the field of solar sails. A solar sail cancels the dependency of the mission duration from the amount of propellant stored on board and has the further advantage of providing a continuous thrust. Unfortunately solar radiation pressure represents at the same time the advantage and the drawback of this propulsion system, since it limits the available thrust to very small ranges.

The real challenge for the CubeSats mission is not just using a solar sail, but a small solar sail, since the provided thrust depends on the sail surface area and the mission restrictions on sizes and volumes limit considerably the sail dimension. For this paper solar sails with characteristic acceleration a_c = 0.01 mm/s^2 and a_c = 0.05 mm/s^2 have been taken into consideration. As stated above, the optimal trajectory has been found for a timeframe of 2T, where T denotes the period of the Halo orbit used as initial guess. For each value of the characteristic acceleration, tests have been conducted using Halos with z-axis amplitude A_z = 250000 km and A_z = 350000 km as initial guess. An optimal trajectory obtained with a_c = 0.01 mm/s^2 and A_z = 250000 km is shown in Figure 3. The reference frame *Oxyz* is used, but for easy viewing the origin O and the Sun are not included in the figure.

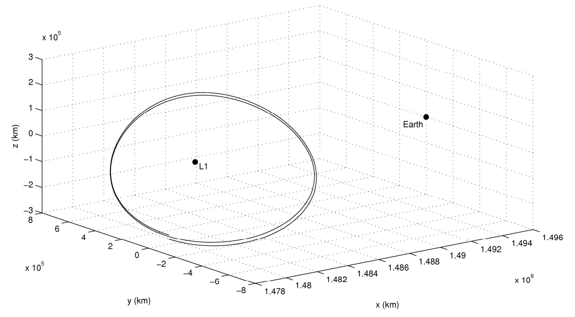

Fig. 3: Optimal trajectory obtained with a_c = 0.01 mm/s^2 and with a A_z = 250000 km Halo

IV.II Communications

As demands on space communication systems become greater, both in terms of data to be transmitted and distance from Earth, it becomes more and more important to pay close attention to the selection of the best communication technology.

There are significant differences between RF and laser communication systems, and much of it results directly from the several orders of magnitude difference in wavelength, which actually results in very different antenna sizes.

RF communications systems provide wide-area coverage, multicasting service, and easy point-to-point wireless communications. Optical communications systems have no regulatory restrictions on the use of frequencies and bandwidths and are immune to jamming and interception by adverse parties.

In order to select the most suitable configuration for the CubeSats mission, a trade-off has been performed to compare the RF solution and the optical one.

The first step of the analysis consisted on the evaluation of the link budget: a summary of the obtained results is shown in tables V and VI for RF and optical system, respectively.

The computations have been performed considering a required data rate of 400kbps and a link range of 1.5x10^6 km (distance between Earth and the first Earth-Sun Lagrangian point).

From the comparison between the two link budgets it results that the laser communications system needs a much smaller antenna, which will correspond to lower mass and easier integration requirements. Moreover, the required power is less for optical system.

RF system – Ka band	
Transmit power	1.8 W 2.55 dBW
Frequency	32 GHz
Atmosphere loss	-4 dB
Antenna pointing loss	-2 dB
Free space loss	-246 dB
BER	10^{-6}
RX antenna diameter	34 m
RX antenna gain	78 dB
System noise temperature	196 K
Link margin	10 dB
TX antenna gain	42 dB
TX antenna diameter	**52 cm**

Table V: RF system link budget

Optical System	
Transmit power	500mW 27dBm
Wavelength	1.55μm
Frequency	193THz
Pointing loss	-6dB
Free space loss	-322dB
RX antenna diameter	5m
RX antenna gain	139 dB
RX loss	-3dB
Sensitivity	90 photons/bit
Link margin	10 dB
TX antenna gain	94 dB
TX loss	-3 dB
TX antenna diameter	**3 cm**

Table VI: Optical system link budget

Besides the link budget considerations, to conduct a realistic trade study of RF versus laser communications, other important characteristics or factors must be identified and included in the trade [25].

In the present work the following parameters have been considered for the trade-off (Please note that, some of them are only qualitatively evaluated):

- mass
- power
- cost: the lifecycle cost includes two contributions, that are development, or non recurrent cost, and recurring costs; the development cost would be higher for laser communications, but recurring costs would be lower (overall RF are preferable).
- integration impact: it includes several factors that denote the overall effect of integrating a communications system.

- volume needed to allocate the system (related to size)
- field of view: the requirement to provide a clear view throughout a range of angle is more stringent for RF systems due to larger antennas;
- need to stow and deploy the antenna
- dynamic reaction effect (related to deployment operations)

- technical risk: it includes parts availability and level of space qualification, development and testing.

The results of the comparison are shown in table VII.

As overall result of the trade-off, the optical communications turned out to be the best solution.

	Mass	**Power**	**Cost**	**Integration impact**	**Technical Risk**	TOT.
Weight [%]	23	10	25	20	22	100
RF	-1	-1	1	-1	1	-0,06
Optical	1	1	-1	1	-1	0,06

Table VII: RF vs Optical communications trade off

It is also worth underlining that one of the main objectives of the proposed CubeSats mission is to provide a platform for test and validation of advanced technologies. According to this the choice of implementing laser communications is even more significant.

V. CONCLUSIONS

The paper describes a 6U CubeSats system interplanetary mission to one of the Earth-Sun Lagrangian point.

The problem of cost reduction is a significant driving factor in advancing space technologies, and it mainly involves two main points, that are the miniaturization or mass and power reduction of platform and instruments, and the implementation of new launch strategies, mission planning and use of ground network to reduce the cost. These issues are important not only for extremely small satellites, but are significant for any bigger spacecraft, as a reduction of the mission cost is always desirable.

According to this, the interest in small satellites, and in particular CubeSats, is growing up, as they can represent valuable platforms both for scientific and technological scopes, with lower costs than big satellites.

In particular a mission like that discussed in the paper would represent a good opportunity to improve the national interest and capabilities in the exploration of the solar systems, pursuing both scientific and technological objectives, foreseeing sun observation and plasma measurements, as well as advanced technologies demonstration (e.g. optical communications, solar sails), in view of their future implementation on larger spacecraft. Moreover, it would give the chance to expand the academic presence in developing systems needed for future missions, including human expeditions.

VI. ACRONYMS

AMR – Anisotropic Magneto-Resistance
BER – Bit Error Rate
HDPE – High Density Polyethylene
INMS – Ion and Neutral Mass Spectrometer
LEO – Low Earth Orbit
MDPS – Micrometeoroids and orbital Debris Protection System
MLI – Multi Layer Insulation
RF – Radio Frequency
RX – Receiver
S/S – Subsystem
TX – Transmitter

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