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ABSTRACT

The recent explosion in proposed microsatellite missions is based on the possibility to mass-produce cheap platforms capable to deliver acceptable performance over a limited lifetime. The assumption behind such scheme is that individual microsatellites are expected/allowed to fail in reasonable numbers, the resulting degradation of constellation performance being limited due to the large population of active spacecraft. We argue that cheap platforms do not necessarily need to be seen as disposable assets, so that low cost constellations featuring a low number of microsatellites may nevertheless be capable of remarkable performance. The key technology needed to enable such feat is low power electric propulsion, whereby microsatellites are allowed to acquire and maintain precisely tuned orbital locations, compensate atmospheric drag to fly longer, and de-orbit safely at end of life. A number of such microsatellites may be fitted with an instrument each from a suite of different sensors operating in various spectral bands. The constellation would operate as an actively controlled system, with the individual instruments providing well coordinated raw data that may be processed using data fusion techniques to yield the final product. Starting from the proven performance of a currently available low power Hall thruster, we present general design criteria for constellations based on a 50 kg-class microsatellite bus. The potential benefits of such technology are outlined with respect to applications such as precision farming, urban area monitoring, and dual use land surveillance.

KEYWORDS: Electric Propulsion, Small satellite, Constellation, Distributed payload, Earth Observation

INTRODUCTION

Aimed at enhancing worldwide government, commercial, industrial, civilian, military and educational communities capabilities to support Earth Observation (EO) missions (e.g. to manage natural resources, to support agricultural practices, to provide climatic assessments, to detect and monitor natural disasters), small satellite technology offers unique opportunities to obtain high performance reducing the mission cost. Such platforms provide the opportunity to carry out Earth observation missions using small, low cost satellites, and correspondingly to reduce the cost of launch, ground stations, data distribution structures, and space system management approaches. In additional small satellites provide unique opportunities to setup affordable constellations[1]. In this respect, small satellites can realize tasks that are not practical with large satellites. In addition, the present technological readiness level reached by low power, high performance electric thrusters, like SITAEL's HT100[2], in combination with a growing number of ongoing technological advances (in particular high efficiency solar cells), make it now possible to equip a microsatellite platform with an electric propulsion system[3].

Microsatellites equipped with low power electric thrusters can enable new kinds of missions in support of Earth observation, e.g. by providing extended lifetime at low orbital altitude and therefore achieving better Ground Sample Distance (GSD) performance with small optical instruments. In particular, the use of electric propulsion makes it possible for an entire microsat constellation to be placed in orbit with a single launch, possibly as secondary payloads. Each constellation element, based on a common microsatellite bus, can be equipped with a payload chosen among a number of different instruments. Once released in the initial orbit, each individual microsatellite can maneuver autonomously to achieve its own operational orbit.

This constellation architecture is based on the implementation of a distributed array of different instruments, acting in a cooperative way to achieve optimal data collection performance. Each satellite, however, occupies an orbital location that can differ in altitude (and if necessary also in inclination or eccentricity) from the other members of the constellation, so to let each instrument operate in the most appropriate orbital conditions. If desired, different microsatellites could be given different revisit periods of the target area, in order to observe the scene in different, co-ordinated moments in time. Such schemes, which can obviously change in the degree of complexity and in the associated cost of operations, are made possible by the combination of low launch mass (enabling multiple platforms to be orbited by a single launch) and propulsion capability featured by last generation, electrically propelled microsatellites.

This study aims at identifying some design criteria for Remote Sensing Constellations of small satellites based of the distributed payload/multiple orbits concept, in order to offer the best compromise in terms of spatial, spectral and temporal resolution performance, ground coverage (from regional to global accessibility) and satellites number. As an additional output of this study, we outline the design of a microsatellite bus equipped with the SITAEL HT100 thruster and compatible with existing small Earth observation optical instruments to cover the whole range of potential remote sensing applications.

MISSION REQUIREMENTS

Orbit class selection

The orbit design starts with the selection of orbital altitude. Only Low Earth Orbits (LEOs) have been considered with an altitude lower than 1000 km. This upper limitation is imposed due to the difficulty to maintain both high resolution, limited instrument dimensions and power demand at increasing orbital altitude.

Traditionally [4,5] the lower bound on the altitude of Earth Observation missions is set at around 500 km, due to the action of drag that limits the spacecraft lifetime and would imposes severe requirements on any propulsion system that should compensate for it. The use of a low power electric thruster enables microsatellite to counteract atmospheric drag even at very low altitude; therefore in our analysis the limitation on the altitude is set at 300 km [6].

Three different LEO geometries have been considered:

- elliptical vs. circular orbits;
- equatorial vs. inclined orbits;
- Sun-Synchronous orbits (SSO).

Elliptical orbits are an attractive solution for Earth observation purposes. They offer a significant potential gain in terms of coverage: when the orbit is elliptical the satellite stays for an extended period at the apogee, so allowing for a major coverage in the corresponding hemisphere. Due to the altitude range restrictions previously set (300 to 1000 km), the maximum value of eccentricity to consider is 0.049. This is a relatively small eccentricity value and therefore such elliptic orbit offers limited coverage advantages, while it is characterized by a wide set of perturbations typical of this kind of orbit. Moreover, due to satellite altitude and velocity variations, adequate instrument performance can not be guaranteed[5]. Accordingly, the use of elliptical orbits appears not convenient in the altitude range chosen, thus only circular orbits are considered in the following analysis.

Equatorial orbits are not suitable for EO missions since these cannot observe high or even mid-latitudes regions. Inclined obits are appropriate if a specific region or latitude belt has to be observed. In these kinds of orbits, the inclination of the orbit itself is determined by the location of the region of interest. The use of inclined orbits has been proposed specially for military applications[5]. Moreover for a generic orbit of this kind the orbit plane rotation induced by the RAAN-rate causes a variation of the illumination conditions of the target sites between consecutive satellite passages; hence the impossibility to observe the same place every time in the same illumination conditions. Furthermore it is very likely that dedicated launches might be required, increasing the overall mission cost.

In conclusion, the use of circular Sun-synchronous orbits is envisaged since they allow for uniform coverage and Sun illumination conditions, high latitude accessibility, limited satellite altitude and velocity variations, high opportunity of launch as piggyback payload.

Sun-synchronous Repeating Ground Track Orbits

A Sun-Synchronous Repeating Ground Track Orbit (SSRGTO) is an orbit which provides simultaneously the capabilities of repeating ground track orbit and Sun-synchronous orbit[7]. SSRGTO orbits are well exploited for example by Landsat, SPOT and RapidEye programs[6].

Sun synchronous orbits are characterized by the combination of inclination (i), eccentricity (e) and semi-major axis (a) that guarantees to have an average regression of the line of nodes due to the Earth oblateness (J_2) equal to the Sun apparent motion around the Earth (1 deg/day).

Repeating ground track orbits are generated by the combination of perturbations on the argument of perigee, mean anomaly and RAAN so to have an integer number of revolutions after a given number of days (accounting also for the Earth natural rotation). In a repeating ground track orbit, the spacecraft returns after a given number of days on the same Earth location, thus the ground trace of the spacecraft repeats itself. The design of such an orbit requires a fixed orbital period; perturbations, however, will cause an orbital period variation. In particular, the rotation of the orbit due to Earth oblateness has to be considered. This results in an iterative process for the design of a RGTO due to the fact that the Earth oblateness effects are a function of altitude.

To design SSRGTO orbits satisfying both requirements a non linear system for a, e, i has to be solved. At first order, however, such orbits can be considered as circular, near polar and with an altitude given by the initial estimate of the altitude of a repeating ground track orbit neglecting Earth oblateness effects, according to relation 1 [7]:

$$h_o = \mu^{1/3}\left(\frac{2\pi j}{\tau_E k}\right)^{-2/3} - R_E \quad (1)$$

where $\tau_E \cong 86164.10035$ s is the sidereal rotation period of the Earth (relative to the fixed stars), R_E is the equatorial radius of Earth, μ is the Earth's gravitational constant, j is the integer number of orbit periods completed in an integer number of k days.

OPTICAL INSTRUMENTS

Following a conservative approach, we restrict our study to state-of-the-art, flight proven instruments, excluding any new developments. To meet the a wide range of EO requirements in terms of spectral and spatial resolutions, the following options have been considered [6,8]:

- Multispectral instrument;
- Hyperspectral instrument;
- Thermal infrared instrument.

Under the assumption to design a small platform with a launch mass of less than 70 kg, only existing instruments with a mass lower than 20 kg and power requirements up to 50 W have been selected for each class. The key parameters considered for each instrument are:

- spatial resolution;
- swath width;
- number, type and width of spectral channels.

Table 1 summarizes the main instruments selected with the associated flight heritage.

Instrument Class	Instrument
Multispectral	HPT (Rising-2), HRMS (Hodoyoshi-4), IRIS (X-Sat), Mx-T (IMS-1), NAOMI (SPOT 6-7) OC (Hodoyoshi-1), SLIM-6-22 (DMC)
Hyperspectral	CHRIS (PROBA-1), COMIS (StSat-3), Phytomapper (-)
Single Thermal IR	CIRC (Alos-2), HSRS (Bird)

Table 1: Reference EO payloads considered

CONSTELLATION DESIGN

With the aim of designing a constellation based on the presence of different optical sensors, the orbital parameters analysis has been conducted separately for each kind of instrument. The entire constellation is conceptually divided in a set of sub-constellations, each based on a single common payload, and with microsatellites conveniently spaced into a given orbit plane.

Obviously, the number of microsatellites is a driver factor for the overall system cost, thus the number of satellites shall be minimized. The number of orbit planes is another design variable open to multiple choices. In terms of constellation growth and degradation, a single-plane constellation has some advantages with respect to constellations with multiple orbit planes: if a satellite fails, it is possible to re-phase the remaining satellites, with a relatively limited propellant consumption, by means of an in-plane maneuver. On the contrary, repositioning a satellite in a multiple plane constellation may be prohibitive for the high maneuver cost[5]. In addition, using a single near-noon plane the optical payloads will acquire the images with the better and the same illumination conditions[5].

Sub-constellations are aimed to independently achieve payload specific requirements exploiting individual, ad-hoc designed orbits. The entire constellation resulting from their combination offers a very high degree of completeness and versatility, aimed at allowing users to exploit images of only one sub-constellation, or to use a logic combination of some or all of them, depending on specific objectives. A convenient and fast access to space-data to many different users is therefore ensured.

SRGTO identification

Our constellation design begins with the definition of the SRGTO orbit altitude (and associated inclination), starting from the required repeat cycle. The requirements of most instruments set the Revisit Time (RT) in between 1 and 3 days[6]. However, taking into account the unique opportunities offered by the microsatellite constellation, a nominal Repeat Cycle (RC) of up to 5 days has been considered. Equation 1 is used to calculate the altitudes corresponding to the desired nominal RCs. Figure 1 shows the recurrence diagram for Sun-synchronous satellites for altitudes between 250 and 1200 km.

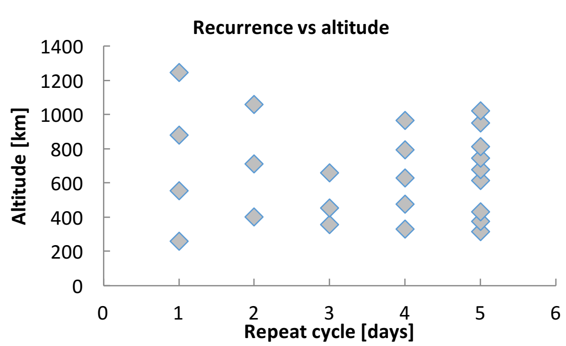

Figure 1. SRGTO altitudes for 1-5 days repeat cycles

Once the possible orbital altitudes are known, the most suitable solutions to cover a given region of interest has to be identified. For this purpose, given the instrument Field of View (FOV), the swath width (S_w) is calculated for each altitude h_o through a spherical Earth assumption[9].

To design the orbit constellation, in addition to the swath width, the dimension of the observation area perpendicular to the satellite motion direction (D_c) shall be taken into consideration. If D_c is equal or lower than the swath width S_w , only one satellite placed in a 1-RC orbit is sufficient to cover every day the area of interest. Otherwise, the area of interest could be divided into a number of strips (N_{strip}) characterized by a dimension equal to the instrument swath width S_w.

Under the assumption to cover the whole area of interest at the same time through a micro-satellite constellation, the number of strips N_{strip} corresponds to the number of orbital planes in which at least one microsatellite has to be placed. These planes are characterized by the same orbital parameters, besides the Right Ascension of the Ascending Node (*RAAN*): they will be separated by an angle $\Delta\Omega$ satisfying the relation:

$$S_w / 2 = R_E \, \Delta\Omega \qquad (2)$$

As an example, Figure 2 shows the number of planes required to cover at once an area of interest for a specific instrument (FOV=26.6°, SLIM-6). The number of planes is plotted as function of altitude h_o for different values D_c of the target size.

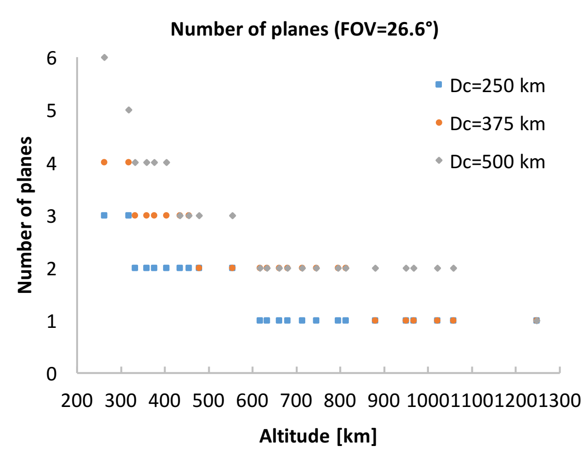

Figure 2. Number of planes for different altitudes at fixed instrument FOV

Figure 3 shows the number of planes required to cover the area of interest (assumed to be D_c = 250km), at once with instruments with different FOV. As a reference, the following values of FOV are presented: 19° (CIRC), 26.6° (SLIM-6), 42° (Phytomapper). The number of planes is plotted as function of altitude h_o .

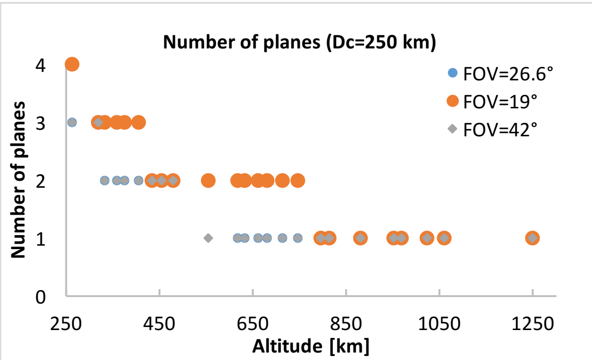

Figure 3. Number of planes for different altitudes at fixed dimension of area of interest (D_c=250 km)

This approach allows for a simultaneous coverage of the area of interest but it does not guarantee the required revisit time, except for altitude with 1-RC. To revisit the region of interest every day in any case, one or more satellites shall be placed along the same orbit. Under the assumption to cover the whole region in quasi-nadir pointing mode, this number of satellites (N_s) is a function of the repeat cycle, and of the swath width resulting by the instrument performance and satellite altitude. Accordingly, for a revisit time of 1 day the maximum number of satellites per orbital plane is:

$$Ns \leq RC \cdot N_{strips} \tag{3}$$

For a revisit time between 1 day and RC, instead, the number of satellites is equal to the number of strips.

Figure 4 summarizes the number of satellites in the same orbital plane required to cover the entire region (D_c=250 km) with a revisit time of less than 5 days. The figure shows both the number of satellites needed to guarantee a revisit time of 1 day and larger, up to 5 days, as a function of RC. The two values coincide for altitude with RC equal to 1.

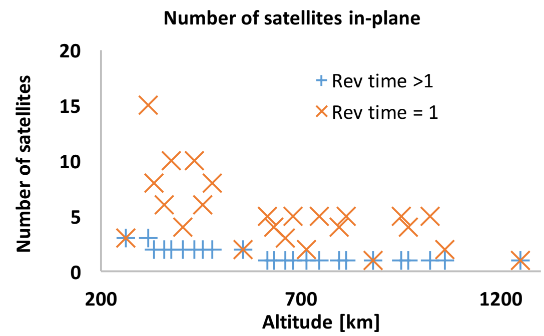

Figure 4. Number of satellites per orbital plane

CASE STUDY: TUSCANY REGION COVERAGE

A possible EO system for continuous monitoring of the agricultural activities in the Tuscany region of Italy is here analyzed as reference case. Table 2 presents the preliminary user requirements, considered as upper level constraints.

Requirements	Value
Product level	Surface reflectance and temperature.
Spectral wavelengths	Wide-band visible (VIS)/near infrared (NIR). Hyperspectral VIS/NIR/short wave infrared (SWIR). Thermal infrared (TIR).
Spatial resolution	Multispectral and hyperspectral images: 100-30 m, 30-10 m, < 5 m. TIR images from 500 m to 100 m.
Revisit time	From 2 months to 1 day.

Table 2: Preliminary user requirements

The analysis allows for designing a constellation based on the cooperative combination of different optical instruments, like multispectral (MS), hyperspectral (HS), or thermal infrared (TIR) sensors, providing the capability to exploit a large portion of the electromagnetic spectrum in both wide and narrow spectral bands. Moreover, such a combination allows also to produce images at many different levels of spatial resolution, and therefore to respond to various classes of users. In particular, the simultaneous and cooperative presence of these instruments allows for covering the whole range of spatial and spectral resolution levels potentially required in Tuscany agriculture activities. Four optical instruments were selected to provide with high spatial and spectral resolution both surface reflectance and temperature measurements.

The most promising solution is a constellation of four microsatellites each equipped with a specific different optical instrument, which acquires images in the VIS Red (R), Green (G), Blue (B) and NIR channels, and in the SWIR and TIR domains. Considering the limited extension of the Tuscany region (D_c = 210 km), RC values not larger than 3 days have been finally selected. Indeed, in the case of a small geographical coverage and for the same RT, it is convenient to stay with low values of RC, so to limit the number of satellites. Table 3 and Figure 5 summarize the constellation architecture.

Sensor	Altitude, [km]	GSD, [m]	Spectral bands	RT, [days]
MS #1	554	10 – 30	R, NIR	1
MS #2	358	2 – 5	R, G, B, NIR	3
HS	358	30 – 50	Thousands VIS, NIR, SWIR	3
TIR	554	100 – 500	TIR	1

Table 3. Microsatellites constellation architecture characteristics

Figure 5. Satellites passes over the Tuscany region

The space-born data obtained can be used for a large number of applications; e.g. land cover and use mapping, crop classification and health monitoring, soil moisture quantification, timely and located fertilization and irrigation strategies definition (precision agriculture).

The MS #1, HS and TIR instruments are expected to be able to provide a swath large enough to cover the entire region during each pass. The MS #2 instrument has been added with the aim at providing very fine spatial details suitable for add-value applications, and for very targeted observations down to the single-crop level[8]. Observations from at least one VIS/NIR and one TIR instrument are daily provided, and this allow to provide also a marginal service of disaster monitoring (floods, wild fires detection).

Finally, observations from the entire constellation are ensured two times per week, perfectly in line with agriculture and disaster monitoring requirements if also partial cloud coverage is considered.

PLATFORM DESIGN

The standard platform is sized and designed to demonstrate the feasibility of the combination of a microsatellite platform, an electric propulsion system and a set of existing small remote sensing instruments. The platform is designed according to the following requirements:

- use of off-the-shelf components to the larger possible extent;
- the whole system has to be designed to be compatible with the presence of an electric propulsion system on-board;
- overall launch mass <70 kg, including payload and propellant;
- maneuver capabilities to counteract the atmospheric drag at very low altitude and to perform orbital maneuvers.

The design is aimed at exploiting a thrusting module based on the SITAEL's HT100 low power Hall effect thruster[2]. Table 4 and Figure 6 show the main thruster performance and characteristics.

Performance	Value
Power, [W]	120-350
Thrust, [mN]	6-18
Specific Impulse, [s]	1000-1600
Efficiency	Up to 40%
Thruster Unit Mass, [g]	< 440
Thruster Envelope, [mm]	Φ 60 x 41 (I/F and cathode excluded)
Propellant	99.996% Xenon
Technology	Hall Effect Thruster, closed electron drift with extended acceleration zone

Table 4: HT100 main performance

Figure 6. HT100 thruster assembly

According to thruster power and thermal requirements, the entire design has been developed in order to make the platform completely thermally isolated from the electric propulsion device. The platform shall also be compatible with the great majority of the existing small optical instruments, and shall be able to provide in-orbit performance (e.g.: attitude pointing accuracy, payload data rate) suited to EO missions. The standard platform is based on off-the-shelf components, and is aimed at ensuring a sufficient power margin for electric thruster operation[2]. Table 4 summarizes the platform dimensions and Figure 7 shows the platform external layout.

Performance	Value
Platform dimensions, [m3]	0.5 x 0.4 x 0.5
Dry mass w/o payload, [kg]	< 40
Power generation BOL, [W]	250
Battery capacity, [Wh]	252
Payload available volume, [lit]	20
Payload available mass, [kg]	12
Payload available power, [W]	30
Fine pointing accuracy, [°]	0.025
DeltaV capacity, [m/s]	1250
Mission lifetime	Up to 5 years
Launch compatibility	VEGA, DNEPR
Communication	X-band downlink (up to 100 Mbit/s) S-band uplink

Table 5: Platform performance

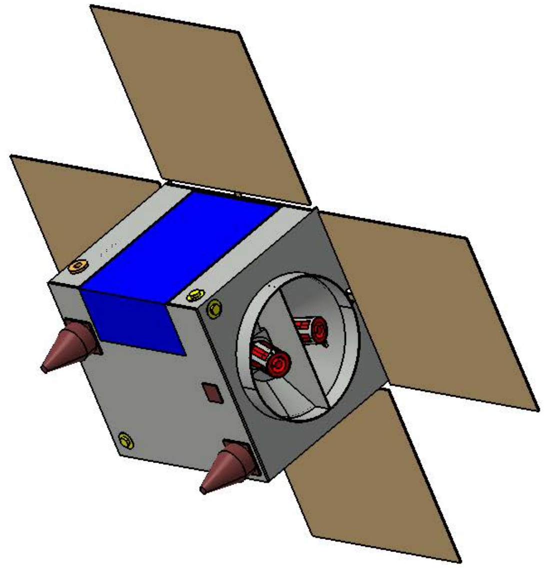

Figure 7. Platform design (*payload vane* in blue)

Two HT100 thrusters in cold redundancy are considered. The thrusting module is completed by appropriate plume shields and by an internal monolithic titanium tank. Four deployable solar panels and an additional body mounted solar array allow for generating up to about 250 W at satellite begin of life (with a reduction of about 10% at the end of life). This power level, rather high for a microsatellite and enabled by the recent technological developments in terms of high efficiency solar cells, is fundamental to operate the HT100. The attitude determination and control system relies on four redundant reaction wheels coupled to a pair of star trackers to provide a very fine attitude pointing accuracy during thrusting, target acquisition or data transmission.

The design proposes the exploitation of coarse sun sensors and magnetic torques to perform coarse attitude control during acquisition or safety mode phases. Magnetic torques take care of momentum dumping too. The platform design is completed by two redundant X-band antennas, and by two Li-Ion secondary batteries aimed to provide a total storage capability of 252 Wh[3,7]. This storage capability is aimed at providing the possibility to perform altitude maintenance ignitions also during eclipse periods. This eclipse thruster ignition capability is aimed to perform very fine station keeping maneuvers, and to allow for, limited electric thruster ignitions in favor of platform thermal control.

The preliminary design resulted from this analysis offers high versatility to the payload in terms of available volume, mass and power, and high performance in terms of data transmission, and pointing accuracy.

Figure 8 illustrates the overall platform logic architecture.

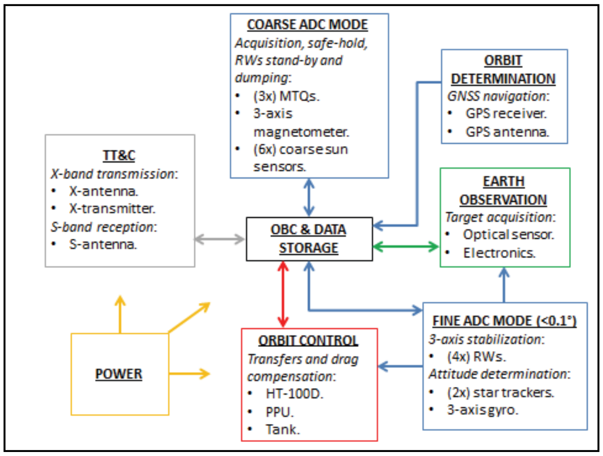

Figure 8. Platform logic architecture

CONCLUSIONS

The paper presents a simple analytic approach for the preliminary design of a small satellite constellation. The approach requires the definition of the size of the region to be observed, of the frequency required and of the payload characteristics and returns the number of satellites and of orbital planes required for a complete every day coverage.

As an applicative case, the design of a Tuscany region agriculture support mission is presented. Starting from upper level user requirements and considering the performance of existing small optical instruments, the analysis results in a constellation based on four microsatellites, each equipped with a different optical instrument (multispectral, hyperspectral and thermal infrared) responding to specific spatial and spectral performance.

In order to guarantee very frequent revisit, microsatellites are placed in SSRGT orbits from 358 km to 554 km. Each microsatellite is equipped with two low power Hall effect thrusters, to provide orbital maneuvering capability and drag compensation for station keeping.

The versatility ensured by the presence of the electric thruster, the consequent capability of optimally and simultaneously exploiting different optical sensors, and the large compatibility of the platform with the great majority of existing small optical sensors, make the proposed constellation able to easily respond to requirements coming from a variety of different users.

Finally, the total mass and the overall dimensions of the microsatellites are such that the entire constellation can be launched in a single shot with any of several low cost launchers.

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