Performance evaluation of an EO constellation equipped with the HT100 Hall effect thruster

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Abstract: The technology readiness level reached by low power, high performance electric thrusters, like SITAEL's HT100, in combination with a growing number of ongoing technological advances (in particular high efficiency solar cells), makes it possible to equip a small satellite platform with an electric propulsion system. The HT100 can provide a long lifetime for small satellites at low altitudes (300 km to 400 km) and therefore high Ground Sample Distance (GSD) performance. Moreover, this combination enables an entire small satellite constellation to be placed in orbit with a single launch, even as secondary payloads, and then each constellation element can autonomously achieve its target operational orbit with limited propellant mass consumption due to the high specific impulse of the HT100, which could result in a larger payload mass fraction. Thus a high-performing and responsive constellation capable of coping with a wide range of spatial and temporal resolution user requirements can be realized. This paper presents a tool developed at SITAEL to compute a figure of merit which combines various mass parameters related to the propulsion subsystem and allows for rapid evaluation of the mass savings benefits enabled by the use of HT100.

Nomenclature

a_0	=	Nominal semi-major axis
A	=	Ram area
A_b	=	Conjunction box area
C_{bat}	=	Battery capacity
degr	=	Solar array degradation per year
DOD	=	Depth of Discharge
EO	=	Earth Observation
EPS	=	Electric Propulsion System
FOV	=	Field of View
I_d	=	Inherent degradation
Isp	=	Specific impulse
LEO	=	Low Earth Orbit
$Lifetime$	=	Mission lifetime
M_{sa}	=	Solar array mass
M_{bat}	=	Battery mass
n_{trans}	=	Transmission efficiency between battery and load

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N_{bat} = Number of batteries
P_{col} = Collision probability
P_0 = Ideal solar cell output
P_{BOL} = Solar array beginning of life power output
P_{EOL} = Solar array end of life power output
P_{sa} = Power that must be produced by the solar array
PMA = Propellant Management Assembly
PPU = Power Processing Unit
PTA = Propellant Tank Assembly
P_e, P_d = Power required during eclipse and sunlight respectively
RC = Repeat Cycle
SE_{bat} = Specific energy density
$SSRGTO$ = Sun Synchronous Repeating Ground Track Orbits
t_{tr} = time needed to perform each leg of the phase change transfer
t_d = Time spent at loiter altitude
TU = Thruster Unit
T_e, T_d = Eclipse and sunlit period respectively
X_e, X_d = Path efficiencies for eclipse and sunlight respectively
β = Ballistic coefficient
ΔTA = Phase change
Δa_{rev} = Semi-major axis decay per revolution
ΔV_{tr} = Velocity increment for each of phase change transfer
ρ_{deb} = Debris spatial density
ρ = Atmospheric density
τ_0 = Orbital period of nominal orbit
τ_1 = Orbital period of loiter orbit

I. Introduction

During the last few years, Earth observation (EO) missions have become increasingly more demanding. In particular, nowadays the general trend is to use small satellite constellations for missions so far enabled only by 1 ton-class platforms, with relevant overall mission cost savings. This is allowed by the constant technology improvement and the trend in space component miniaturization. One of the main challenges for these small satellites is the inclusion of an autonomous propulsion subsystem, which conforms to the small dimensions, low mass and low power available on-board small satellites. It would significantly enhance the performance of these satellites broadening the possible applications and/or extending their lifetime. The significantly lower propellant mass requirements, at the expense of only slight increase of required power, makes electric propulsion more suitable than chemical propulsion to equip small satellites with maneuver capabilities. In addition to orbit maintenance (drag make-up), the HT100 could also fulfill the need which could arise for repositioning (rephase and/or altitude change) of one or multiple satellites in the constellation in order to maintain or recover the constellation performance, in case of failure of any satellite in the constellation. Owing to the low propellant requirements enabled by electric propulsion, an effective solution is to use the HT100 to perform end-of-life deorbiting maneuvers. In this paper, we present a tool developed at SITAEL, which allows for visualization of the significant mass savings enabled by the HT100 for typical LEO EO constellations. This is done by computing the propellant mass required for typical EO missions, and adding this to the mass of other resources required to support the HT100 propulsive subsystem, namely the power subsystem and propellant storage subsystem, and combining these results into a single figure of merit, which relates the total mass of the resources required to support the propulsion subsystem to the total mass at launch. The procedure followed for the computation of this parameter is described, and a reference EO mission scenario is discussed to demonstrate the applicability of this tool.

II. Earth Observation Constellations

The applications of EO can help nations and organizations reap benefits supporting economic development and supports informed policy and decision making. EO has a wide range of applications, and its advantages cannot be overstated. Such applications include:

- Agriculture & farming- planning, crop health monitoring.
- Forestry- conservation activities, evaluation and survey of growth, deforestation monitoring.
- Disaster monitoring.
- Railway and maritime safety.
- Defense and security.
- Natural resource management.

A large number of LEO satellites are needed to provide continuous coverage over an area. Low Earth Orbits exploit the benefits of shorter distance to the Earth's surface. The key advantages of using LEO constellations are:

- Shorter signal propagation periods (low latency): The minimum theoretical latency for LEO satellite is 20-40 milliseconds whereas the latency for GEO satellite is 250 milliseconds[1].
- Lower power needed for data transmission and instrumentation: The high velocities of LEO satellites relative to the surface imply short contact periods to ground stations and short observation periods of specific surface areas by a single satellite. Hence several satellites in appropriate complementary orbits are necessary to provide continuous coverage, or coverage according to user-desired temporal resolution. Different applications call for different spatial and temporal resolutions.

A select few constellation design parameters and their impacts on the mission are listed below:

- Number of satellites: affects the coverage and the principal costs.
- Number of orbital planes and plane spacing: varies based on coverage needs. It is highly advantageous to have minimum number of orbital planes as transfer between the orbits increases the launch and transfer costs.
- Altitude: Increasing the altitude increases the coverage, but may reduce the GSD.
- Inclination: determines the latitude distribution of coverage.

III. HT100 suitability for small satellite platforms

Small satellites have evolved into a new practical business tool that non-commercial and private enterprises can afford owing to the reduced launch cost for piggyback accommodations, and lower lead times. Electric propulsion for spacecraft provides thrust by emitting electrically-charged particles extremely high speeds. Such thrusters do not burn fuel, they eject it. The lower specific propellant consumption of electric propulsion compared to chemical propulsion offers considerable propellant savings. However, the real challenge is to cope with the limited power availability onboard small satellites. Therefore, the need arises to test and qualify low power, low cost electric propulsion systems which can easily be installed on small satellite platforms and which can readily respond to the demands of both commercial and non-commercial organizations. Designing and testing electric thrusters with fair performance and lifetime is the key to open a new market niche for electric propulsion and enter the sector of small satellites which could, in turn, get a tremendous benefit from the mass savings offered by electric thrusters. The HT100 low power thruster produced by SITAEL is an ideal candidate for small satellites, owing to the low power consumption and low mass. The HT100 has a nominal operation point of 175 W and great flexibility, in order to satisfy the performance requirements of a multi-mission scenario. It can be operated in a power range between 100 and 300 W, with a peak efficiency higher than 35%. The performance is very interesting for this class of power: thrust range between 6 mN and 15 mN (Fig. 2) with an anodic specific impulse up to 1600 s (Fig. 3). The HT100 has been coupled with a new cathode developed by SITAEL named HC1 (Fig. 1) capable of sustaining the main discharge with a mass flow rate of 0.1- 0.2 mg/s giving total specific impulse values between 900 s and 1400 s (Fig. 4). The main characteristics of HT100 are summarized in Table 1.

Table 1. HT100 main specifications

Propellant	xenon or krypton
Power	120-300 W
Thrust	6-15 mN
Isp	Up to 1600 s
Efficiency	Up to 40 %
Operating voltage	150-400 V

Thruster mass	460 g (without I/F and cathode support)
Thruster envelope	⌀ 72 ×100 mm (I/F included, cathode excluded)
Lifetime	>2200 hours [demonstrated, @215 W]

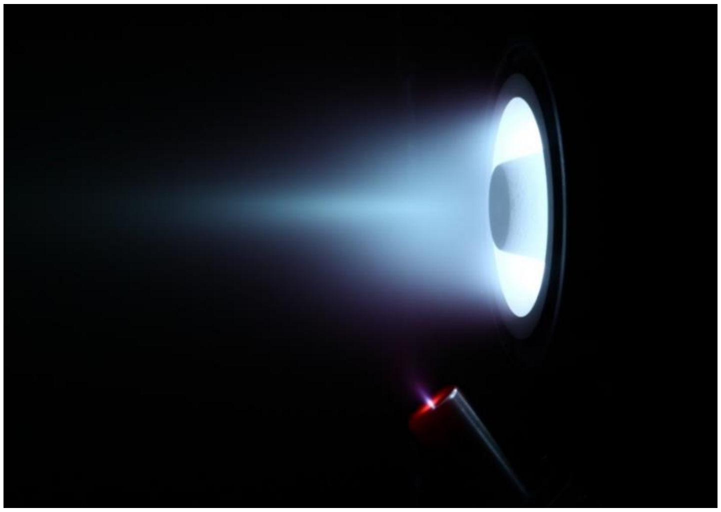

Figure 1. HT100 coupled with HC1 firing in SITAEL's facility

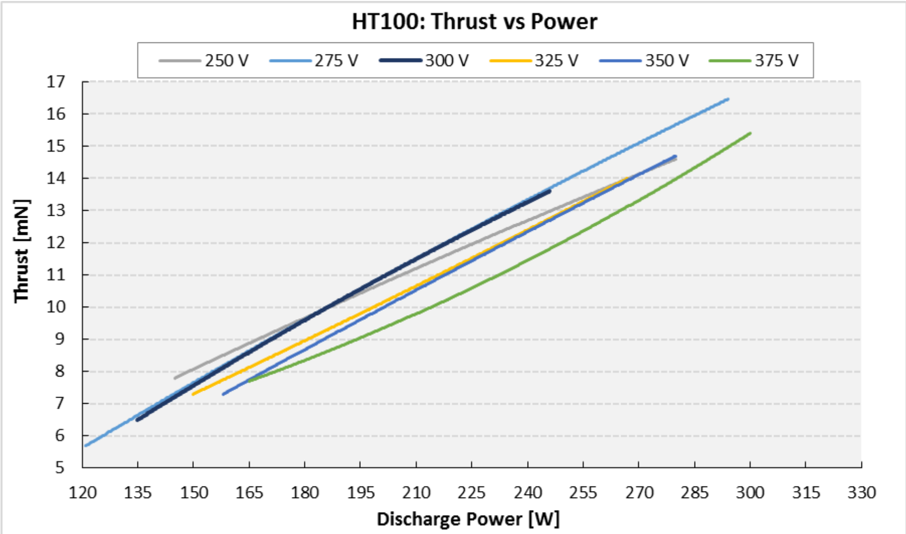

Figure 2. HT100 thrust as a function of discharge power for different discharge voltages

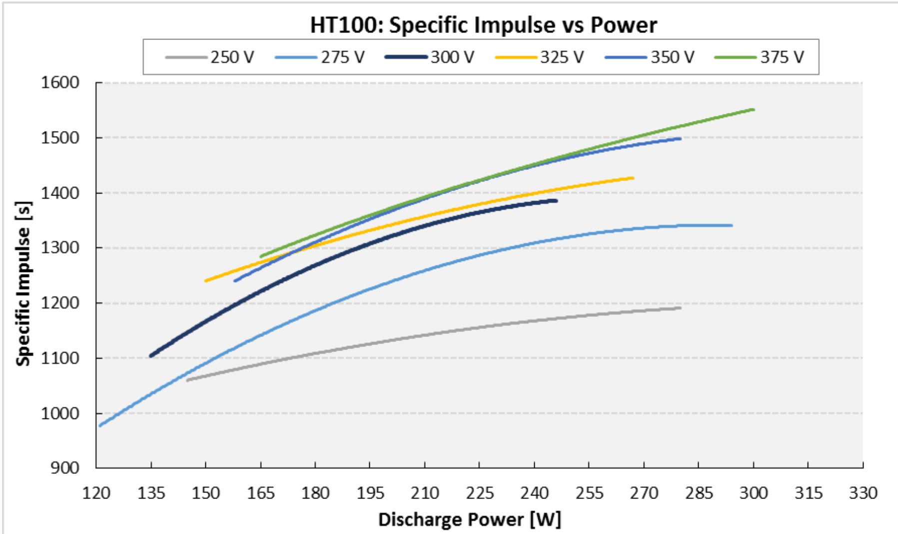

Figure 3. HT100 specific impulse as a function of discharge power for different discharge voltages

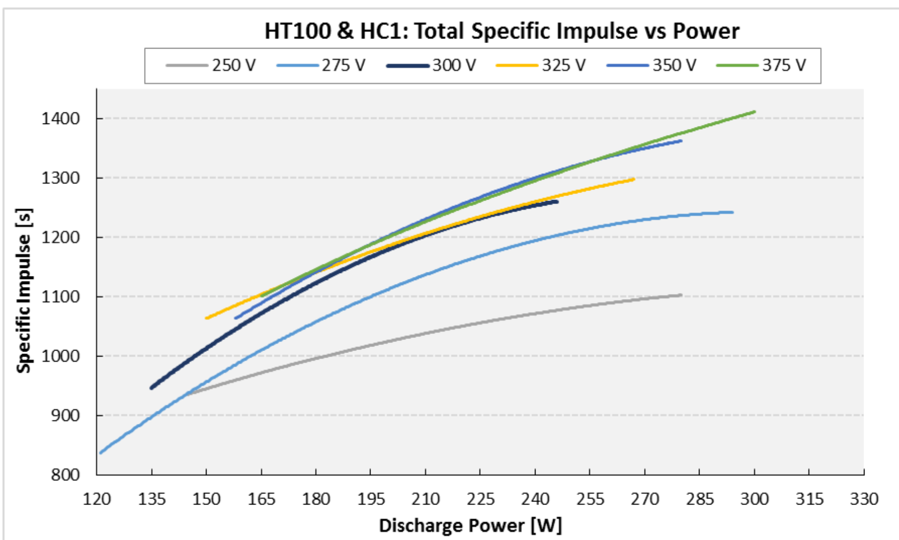

Figure 4. Thruster Unit (HT100+HC1) specific impulse as a function of discharge power for different discharge voltages

In general, the Electric Propulsion System (EPS) is composed of four main units, described below with its functions:

- the Propellant Tank Assembly (PTA) that is in charge of storing the pressurized xenon to feed the thruster throughout the mission;
- the Propellant Management Assembly (PMA) has the task of regulating the propellant pressure and distributing it to the thruster and to the cathodes;
- the Power Processing Unit (PPU) provides the necessary voltages and currents required to operate the TU. It also supplies command signals and electrical power needed to the flow control components (PMA).
- EPS shall contain at least one Thruster Unit (TU) consisting of one HT100 equipped with two hollow cathodes HC1. The TU shall be able to provide the required thrust.

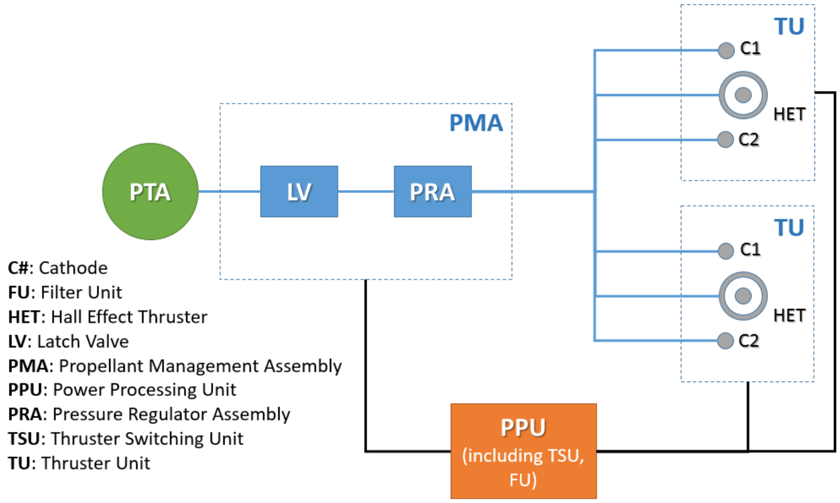

Figure 5. General scheme of the EPS based on HT100

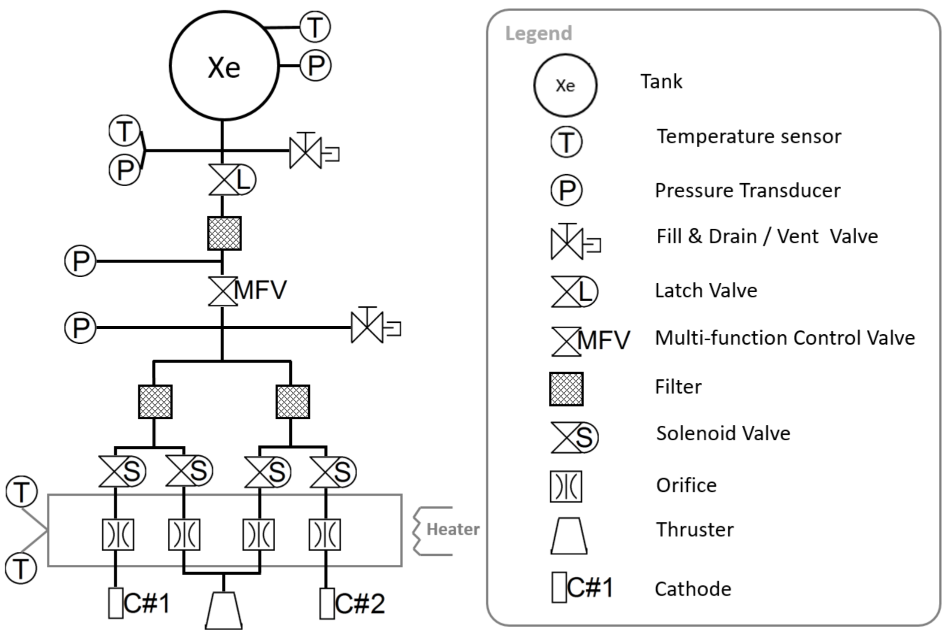

Figure 6. HT100 S/S architecture

The mass and power consumption of the HT100 system will depend on the type of pressure regulation mechanism selected, and the level of redundancy of the system. An estimate of the mass and power consumptions of the HT100 subsystem has been made, considering a redundant cathode branch. According to the architecture proposed in Fig. 6, the PTA is a single branch including a tank, with thermal heaters, high pressure transducers, high pressure fill and drain valve, one latch valve to provide isolation and a mechanical filter to protect against contamination. The PMA high pressure stage is composed of a branch equipped with a multi-function control valve that modulates the pressure, regulates the flow and provides isolation. An additional high pressure transducer completes the PMA high pressure stage. The PMA low pressure stage includes a low pressure transducer, a low pressure fill and vent valve and the distribution module for the thruster. For the thruster unit, the distribution module is composed of two branches (one redundant), each one equipped with a filter and two solenoid valves. Each branch is composed of one cathode, one line for the thruster and two orifices that are designed to cope with the demanded mass flow rate for the thruster and the cathodes. If needed, selection of solenoid valves allows to run the thruster using the cathode of the main (redundant) branch and the redundant (main) line of the thruster. The PPU (not shown in Fig. 6) controls the TU and manages the PMA.

Some reference components which form part of the PTA and PMA have been chosen in order to estimate the mass and power consumption. The mass of the tank is not included in the mass budget shown below, since it will depend on the amount of propellant. The PPU has been considered internally fully redundant. For the architecture reported in Fig. 6, the preliminary mass and power budget are reported in Table 2.

Table 2. HT100 S/S mass and power budget

	MASS	**POWER**
TU (HT100 + 2 x HC1)	0.70 kg	200 W
PTA (excluding tank)	0.60 kg	5 W
PMA	2.30 kg	20 W
PPU	9 kg	90% efficiency
Total	12.6 kg	250 W

IV. Performance evaluation

Since electric propulsion offers considerable mass savings, a figure of merit will be derived to visualize this fact. A tool was developed at SITAEL, the objective of which is to derive a parameter which combines the total mass of the HT100 S/S, propellant mass , propellant tank mass and supporting power subsystem mass, and relates it to the total mass at launch, which is fixed at 200 kg in this paper. The first step is to calculate the propellant mass required to support a typical EO constellation for the duration of the mission. This involves evaluating the propellant requirements for each mission phase, and then calculating the propellant storage requirements. Then the required power resources to support the HT100 S/S is evaluated, after which the figure of merit is derived.

A. Mission analysis

The typical mission phases for LEO EO missions and the corresponding method of propellant requirement evaluation is now described below:

Phase (a) Orbit lowering to nominal operative altitude: small satellite constellations, typically, rely on piggyback launches, that is, they share the launcher space with other small satellites which are not necessarily of the same constellation. Due to this, the deployment altitude is highly dependent on the launcher, and orbit raising or orbit lowering maneuvers along with an inclination change might be necessary before the beginning of the mission. A survey was carried out of 6 operational large and medium launch vehicles for rideshare launch to find the typical deployment altitude for each of them , and this can be seen in Table 3.

Table 3. Typical deployment altitudes of operational launch vehicles for rideshare launch[2]

Launcher	Deployment altitude [km]
Falcon 9 v1.2	600
Dnepr	600
Rockot	700
PSLV	620
Vega	700
Soyuz ST	820

For the analyses carried out in this paper, the deployment altitude will be fixed at 600 km.

Phase (b) Nominal mission phase: for EO missions in LEO, atmospheric drag greatly influences the semi-major axis, causing a slow altitude decay. This causes a drift of the groundtrack on account of the reduced orbital period, which affects the coverage characteristics. Depending on the type of EO missions, constellation parameters and type of observation payload, different levels of groundtrack maintenance precision can be demanded. This groundtrack maintenance is essential for the duration of the nominal mission phase in order to maintain the desired coverage characteristics. The frequency of thruster of firing and the firing time per ignition is dependent on the precision of groundtrack maintenance i.e. the tolerance band. From the mission analysis point of view, it should be noted that a narrow tolerance band means more frequent firing with lower firing time per ignition, since maintenance of the orbit at a higher altitude means lower atmospheric drag. This will lead to a small but noticeable propellant saving. The process used to evaluate the propellant required for groundtrack maintenance over the nominal mission is now described; given the nominal mission lifetime, fractions of the time for which the levels of solar activity will be high, medium and low are defined. The atmospheric densities corresponding to these solar activity levels are evaluated using the NRLMSISE-00 model. The corresponding decay in semi-major axis per revolution is evaluated using[3]:

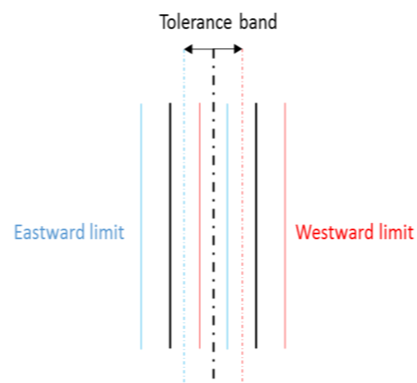

Figure 7. Groundtrack maintenance

$$\Delta a_{rev} = -2\pi\beta\rho a_0^2 \qquad (1)$$

The tolerance band can be specified manually, or it can be related to the swath of the optical instrument by specifying the minimum percentage of overlap that must be maintained between successive ground tracks. The parameters of this transfer are calculated by using the Edelbaum formulation [3].

Acknowledging the considerable lifetime of EO missions, the propellant required for collision avoidance maneuvers also has to be acknowledged. The first step towards calculating the propellant mass required for this operation is to calculate the cumulative probability of conjunction. The spatial density of orbital debris greater than ten cm is taken from NASA's EVOLVE model[4] . The total number of burns for collision avoidance can be estimated using the following formulation [5]:

$$P_{col} = \rho_{deb}(10\ km/s\ )A_b * Lifetime \qquad (2)$$

For the sake of the analyses carried out in this paper, we assume the average relative velocity between objects in orbit to be 10 km/s, and the length of the conjunction risk box to be 1 km on each side. To determine the velocity change required to complete a maneuver, the Clohessy-Wiltshire equations are used, which describe relative positions and velocities between two objects in the orbital radial /in-track /cross-track plane. This is done by finding the initial relative velocity components needed to create a relative position greater than a predetermined value representing acceptable closest approach, in a given amount of time. This length of time corresponds to the maximum continuous firing time of the HT100 allocated for collision avoidance maneuvers. This Δv is then multiplied by P_{col} from Eq. (2) to find the total Δv for collision avoidance maneuvers over the mission lifetime [3].

Phase (c) Constellation performance recovery (rephasing**):** in the event of satellite failures, in order to maintain the coverage performance of the EO constellation, a worst-case rephasing requirement of 180 deg has been assumed. To size this rephasing, transfer by means of electric propulsion, the formulation presented by Janson [6] is used. With this approach, a true anomaly rephasing of a satellite is performed through a low-thrust orbital transfer consisting of a climb/descent to a higher/lower orbit, a period of loiter in this "drift" orbit, and then an inverse transfer back to the desired altitude. At the end of this transfer-drift-transfer maneuver, the satellite is placed at the desired altitude with the required ΔTA with respect to its initial anomaly.

The ΔTA for a spacecraft performing this kind of transfer can be computed as:

$$\Delta\mathrm{TA} = 2\pi \mathrm{t_d} \frac{\tau_0 - \tau_1}{\tau_0 \tau_1} + 2[3 \frac{\Delta \mathrm{V_{tr}} t_{tr}}{2a_0}] \tag{3}$$

The time of the transfer can be computed as:

$$\mathrm{t_{tr}} = \frac{\mathrm{m_0}[1 - e^{\frac{-\Delta \mathrm{V_{tr}}}{\mathrm{g_0} Isp}}]}{\frac{\mathrm{F}}{\mathrm{g_0} Isp}} \tag{4}$$

By assigning the required phase change (180 deg, corresponding to the worst case), the equations can be inverted, yielding a set of $\{td + 2t_{tr}, \tau 1\}$ combinations (total transfer time vs loiter altitude) that realize that change. The total firing time and propellant consumption is chosen by defining a maximum acceptable 'constellation down time', which corresponds to the sum of the loiter time and the total firing time.

Phase (d) EOL deorbiting: the propellant requirements for re-entry involve sizing a maneuver so that the final altitude allows for a natural reentry within a specific period of time. Due to the typically high ballistic coefficient of EO satellites and low altitudes of such missions, natural reentry is guaranteed at a time frame in the order of months, however, a final decay altitude of 250 km is fixed in the analyses carried out in this paper, to get worst case propellant requirements. This maneuver is sized assuming that during the uncontrolled reentry the spacecraft will keep a constant attitude profile, corresponding to a constant drag area. This is quite the conservative approach since this area is significantly smaller than the area most probable to face the velocity during an uncontrolled reentry.

B. Propellant storage

From the previous computations, the total firing time of the thruster is evaluated. In addition to this, the mass of propellant is then used to calculate the mass of the tank required to carry this amount of propellant. Propellant storage of xenon is done in supercritical conditions in order to guarantee high storage density. This leads to limit in most cases the temperature from 20°C (to be sure to stay above 17°C at any time) to 55°C (to limit the pressure to sustain) for the storage. Typically, the storage pressure is 150 bar with a temperature range between 40°C to 55°C. In this condition, the xenon density is between 1550 kg/m^3 and 1800 kg/m^3. Corresponding hardware to foresee is a tank with an active thermal control (heaters and temperature sensors) and passive cover (multi-layer insulation). Tank temperature control is usually performed at spacecraft level. Aimed at sizing the tank, the first step is to evaluate the xenon density as function of storage conditions (pressure and temperature). For this purpose, the Peng Robinson model has been used[7]. The selected storage pressure is 150 bar. Concerning the storage temperature, in order to guarantee the supercritical condition, the gas temperature shall be taken to a temperature above the critical temperature. At a constant storage pressure, an increase in storage temperature corresponds to a reduction of xenon density and consequentially an increased tank inner volume (for the same propellant mass). Therefore, for safety the tanks have been sized considering the maximum expected temperature. This value of temperature has been set to 55°C. In this storage condition (150 bar, 55°C), the Xe density is about 1586.69 kg/m^3 Since the propellant density and the storage conditions are known, the procedure indicated in [7] has been used to size a spherical tank. Under the assumption of considering mission for which the propellant mass is less than 20 kg, the material selected for the tank is the titanium alloy Ti-6Al-4V grade 5, which represents a good compromise between mechanical properties and fracture.

C. Power subsystem

Power consumption by the electric propulsion subsystem has to be taken into account, especially when it comes to dealing with small satellites. Calculation of power subsystem contributing mass to support the propulsion subsystem is therefore an important parameter to consider. The procedure followed to size the solar array and batteries is as described by Wertz[8], and will be summarized here. The maximum sunlit period and maximum eclipse period is first calculated. The power requirements of the HT100 are assumed as equal in eclipse and sunlight, and is dependent on the chosen operating point of the thruster.

$$\mathrm{P_{sa}} = \frac{[\frac{P_e T_e}{X_e} + \frac{P_d T_d}{X_d}]}{\mathrm{T_d}} \tag{5}$$

Here, P_{sa} is the power that must be produced by the solar array must provide during daylight, and is dependent on the location of the thruster operation arcs along the orbit. The power demanded by the HT100 subsystem is dependent on the chosen operating point of the thruster. Typically, for direct energy transfer (DET) $X_e = 0.65$, X_d =0.85 and for peak power tracking (PPT) $X_e = 0.6$,X_d =0.8. At beginning-of-life, the array's power per unit area is calculated by:

$$\mathrm{P_{BOL}} = \mathrm{P_0} I_d \cos(45) \tag{6}$$

where 45 degrees is assumed to be the fixed Sun angle. In reality, this angle will change along the orbit.

Next, the end-of-life power of the solar array is calculated by considering the degradation of the solar array performance during the mission lifetime.

$$\begin{aligned} L_d &= (1 - degr)^{Lifetime} \\ \mathrm{P_{EOL}} &= \mathrm{P_{BOL}} L_d \end{aligned} \tag{7}$$

P_0 , I_d and the degradation are dependent on the type of solar cells used; the options of either Si or GaAs are made available in the tool, with the corresponding values for ideal solar cell output and degradation taken equal to the average values from commercial manufacturers.

The corresponding solar array and mass, assuming a specific power of 25 W/kg, can be calculated as:

$$\begin{aligned} \mathrm{A_{sa}} &= P_{sa}/P_{EOL} \\ \mathrm{M_{sa}} &= P_{sa} * 0.04 \end{aligned} \tag{8}$$

The battery sizing also has to be taken into consideration. For energy storage during eclipse, the battery capacity can be calculated as:

$$C_{bat} = \frac{(P_e * T_e)}{(DOD) N_{bat} n_{trans}} \tag{9}$$

The total mass of the batteries is then calculated based on typical specific energy densities of different battery types, 4 of which are made available for user selection in the tool, namely Nickel-Cadmium, Nickel Hydrogen, Li-ion and Li- polymer. The specific energy densities have been taken by compiling data from various commercial manufacturers. Adding the two masses from Eq. (8) and Eq. (10) we obtain the total mass of the power subsystem required to support the HT100 subsystem, which is given by Eq. (11).

$$M_{bat} = \left(\frac{C_{bat}}{SE_{bat}}\right) N_{bat} \tag{10}$$

$$M_{power} = M_{sa} + M_{bat} \tag{11}$$

D. Figure of merit

The propellant mass, mass of the power subsystem from Eq. (11), and mass of the HT100 S/S from Table 2 are summed, and the fraction of this 'propulsion subsystem contributing mass' to the total launch mass (as mentioned before, fixed at about 200 kg) is computed, and this is the figure of merit which can be easily used to quantify the mass impact of the HT100 propulsive subsystem and the resources required to support it, on the total launch mass. The mission parameters are highly customizable, and the tool developed will allow calculation of this mass figure of merit, the so called 'propulsion system contributing mass factor' for a wide range of mission scenarios. This tool was combined with a constellation design tool developed at SITAEL and in conjunction, this combined tool can allow for constellation design and performance evaluation of the HT100 subsystem together. The workflow of this tool is shown in Fig. 8, with the 'Constellation Design' module consisting of the aforementioned constellation design tool. The boxes highlighted in yellow signify inputs, whereas the boxes highlighted in green represent outputs. The outputs of the Constellation Design module, in particular, the altitude and inclination of the orbit, are inputs to the tool presented in this paper.

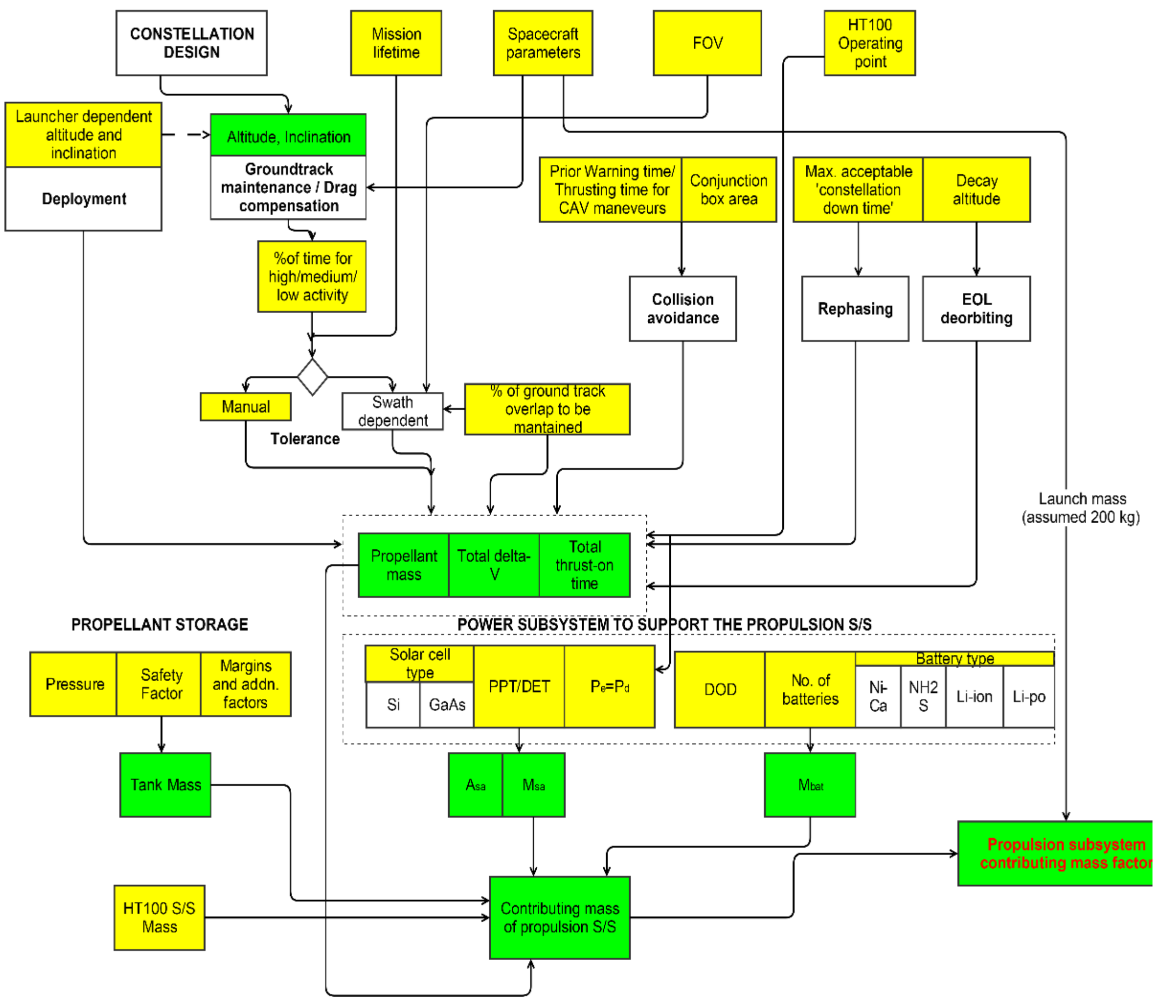

Figure 8. Performance evaluation tool workflow

V. Reference scenario

A reference scenario will be analyzed, by using the constellation design tool to determine the satellite constellation parameters. The figure of merit for the HT100 subsystem is calculated for various thruster operating points. The inputs used for the tool are listed in Table 4.

The reference scenario to be studied is focused on constellations in SSRGTO. A SSRGTO is an orbit which provides simultaneously the capabilities of repeating ground track orbit (RGTO) and Sun-synchronous orbit (SSO). SSRGTO orbits are well exploited by many EO missions, e.g. by Landsat, SPOT and RapidEye. Under the assumption to use SSRGTO, this module needs as input only the instrument field of view (FOV) and reference dimensions of the Area of Interest. For repeat cycles (RC) between 1 and 30 days, the altitudes are iteratively computed along with the number of satellites needed to cover the whole area of interest. Regarding the inputs for the Constellation Design module, the selected Area of Interest for this EO case study is Italy. The instrument selected is a hyperspectral imager, typically used for land observation and detailed vegetation classification. Fig. 9 shows the recurrence diagram for the reference mission, i.e. the number of satellites needed to cover the Area of Interest for each combination of RC and altitude. To show the capabilities of the whole tool, the reference mission scenario highlighted in Fig. 9 was chosen. This reference scenario was selected considering the minimum altitude for which the number of satellites is lower than 8 and the corresponding RC is at the minimum. Table 5 shows the whole output set.

Table 4. Reference mission scenarios input parameters

CLASSIFICATION LEVEL 1	CLASSIFICATION LEVEL 2	PARAMETER	INPUT
CONSTELLATION DESIGN	Area of Interest (Italy)	Minimum latitude	36.4 deg
		Maximum distance	900 km
		Area	301338 km^2
	Instrument	FOV	2.63 deg
		Off-nadir angle	0 deg
THRUSTER PARAMETERS		Thrust	[9; 12.5; 7.5; 10.5;5.8; 7.8; 14; 7.7] mN
		Isp	[1100; 1150; 900; 1250 ; 840; 720; 1290 ;1120] s
		Discharge Power	[175; 225; 145; 230; 120; 170; 265; 165] W
MISSION ANALYSIS	Mission lifetime		5 years
	Deployment	Deployment altitude	670 km
	Groundtrack maintenance	High solar activity duration	10 % of lifetime
		Medium solar activity duration	40 % of lifetime
		Low solar activity duration	50 % of lifetime
		Cd	2.2
		A	$1.2m^2$
		Tolerance band	1 km
		Constellation down time	7 days

		Rephasing /Constellation maintenance	Phase change	180 deg (worst case)
		Collision avoidance	Acceptable closest approach	500 m
		Collision avoidance	Max. firing time for CAV	3600 sec
PROPELLANT STORAGE		Tank sizing	Density	4.4 kg/dm^3
PROPELLANT STORAGE		Tank sizing	MEOP	15 MPa
PROPELLANT STORAGE		Tank sizing	Safety factor	3
PROPELLANT STORAGE		Tank sizing	Yield strength	800 MPa
POWER SUBSYSTEM	Solar array sizing	Solar cell type	GaAs	
POWER SUBSYSTEM	Solar array sizing	Xe, Xd	0.6, 0.8 (PPT)	
POWER SUBSYSTEM	Battery sizing	Battery Type	Ni-Cad	
POWER SUBSYSTEM	Battery sizing	DOD	38	
POWER SUBSYSTEM	Battery sizing	N_{bat}	2	
POWER SUBSYSTEM	Battery sizing	n_{trans}	90	

Table 5. Constellation Design module output set

Repeat cycle	16 days
Altitude	320.69 km
Inclination	96.745°
Number of satellite	8
Satellite phasing	2.591°

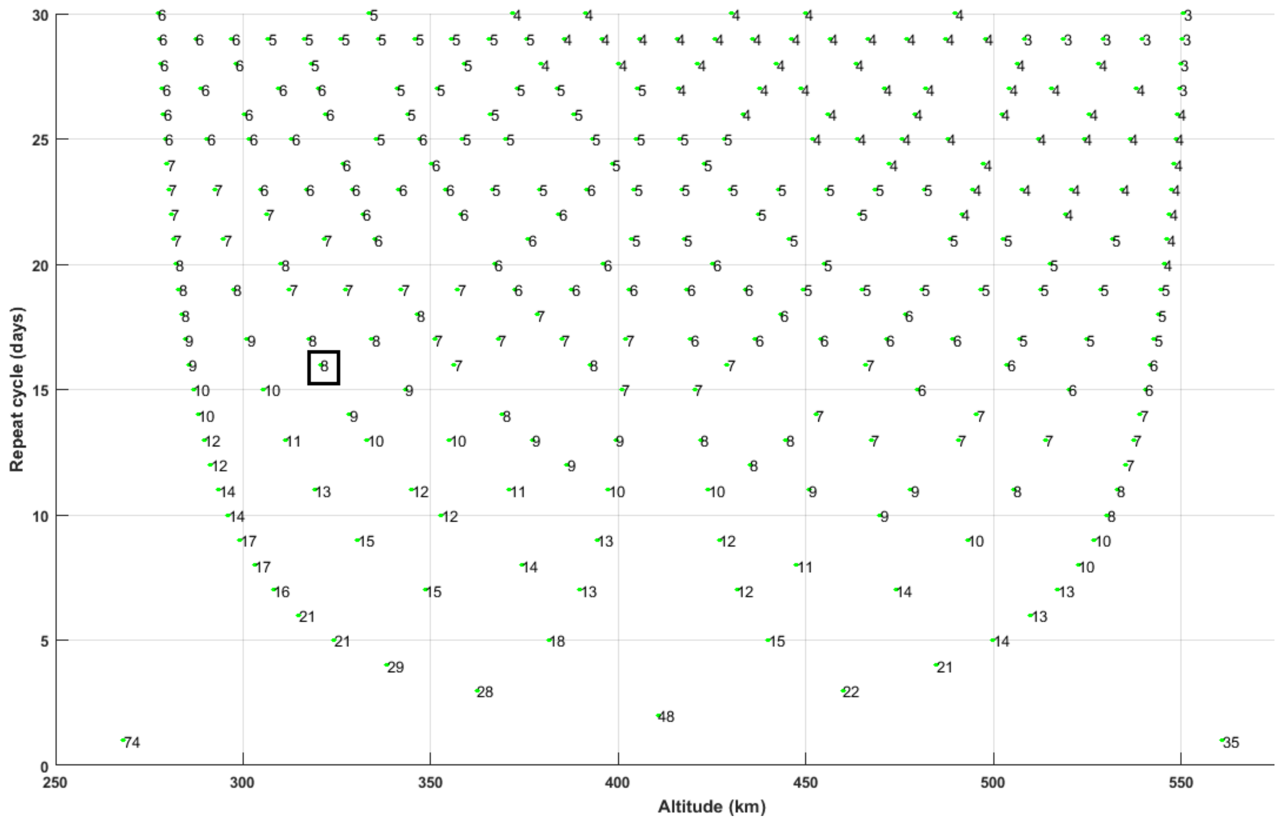

Figure 9. Number of satellites as a function of RC and altitude [FOV=2.63 deg]

From Fig. 10 the mass benefits of equipping a small satellite with the HT100 is apparent. Mass savings ultimately translate into cost savings owing to the fact that launch is a major contributor to the total mission cost. This can lead to more satellites being launched together, and as a result, can lead to lower RC's , as can be seen from Fig. 9. In order to absolutely quantify the performance of the HT100 subsystem, the mission requirements assuming thruster operation at a fixed point 9 mN (1100 s)@175 W (indicated by the black box in Figure 10), the total output set of the tool is presented in Table 6.

Due to the mission total impulse requirements demanded by long duration missions (such as the presented reference scenario), one TU alone may not be able to satisfy the total firing time/ total impulse requirements. To satisfy the total impulse requirements, this will involve either operation at higher thrust or mounting multiple TU's on the satellite platforms. In the case of the presented reference scenario, operation at the higher thrust of 14 mN will add about 12 kg to the power subsystem mass, with respect to the power subsystem mass considering operation at 9 mN. This results in a propulsion subsystem contributing mass factor of >38%, which is not efficient from the point of view of mass savings. Moreover, the erosion rate at higher discharge powers can contribute to reduction of thruster lifetime.

As an alternative, high mission total impulse requirements can be satisfied by mounting an additional TU. In the case of the presented reference scenario, the additional mass needed for this will have to take into account only the mass of the additional TU, the mass of the distribution module and the mass of the thruster switching unit, which accounts to about 2.5 kg, which will add only about 1 % to the propulsion subsystem contributing mass factor. This results in a propulsion subsystem contributing mass factor of only 33.14 % for a 5-year mission at the selected thruster operating point.

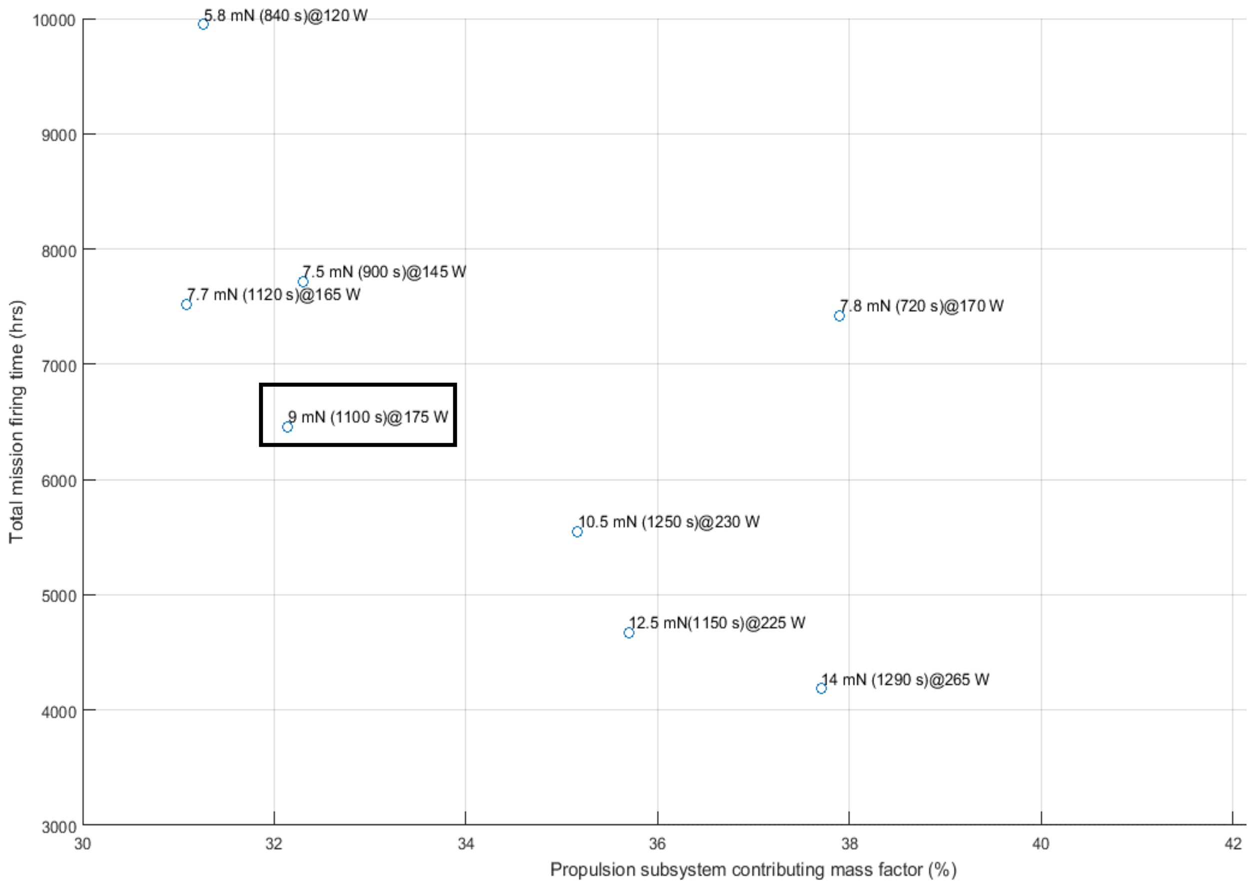

Figure 10. Propulsion system contributing mass factor versus total thruster firing time for the reference 5-year mission scenario

Table 6. Performance evaluation tool output set

	Phase	Propellant consumption [kg]	Firing time [hours]	delta-v [m/s]
HT100 S/S PERFORMANCE REQUIREMENTS	Deployment	2.91	969.67	155.95
	Orbit maintenance	15.1	5043	829.08
	Rephasing	0.64	215.8	35.53
	EOL deorbiting	0.69	230.86	41.025
	Collision avoidance	0.24	32	13.36
	Total	19.69	6459	1075
PROPELLANT STORAGE REQUIREMENTS	Tank Mass		3.99 kg	
	Tank volume		0.87 dm^3	
POWER SUBSYSTEM REQUIREMENTS	Solar array area		3.7364 m^2	
	Solar array mass		16.5638 kg	

	Battery mass	11.5213 kg
	Total power S/S mass	28.0851 kg
PROPULSION SUBSYSTEM CONTRIBUTING MASS FACTOR	**32.14 % (with 1 TU)** **33.14 % (with 2 TU's)**	

VI. Conclusion

The important operations pertaining to typical LEO EO satellite constellation missions were analyzed, and the mission requirements required to be satisfied by the HT100 propulsive subsystem were analyzed. It was found that a wide range of mission scenarios can be accomplished with low propellant mass carried on-board, even if the launch vehicle releases the satellites at a higher altitude. This makes small satellites equipped with the HT100 ideal for a rideshare launch, and thereby contributing to considerable cost savings. A figure of merit, called the propulsion subsystem contributing mass factor was derived, which quantifies the mass impact of the HT100 S/S and the resources required to support it (power, propellant storage) on the total launch mass. Owing to the TU mass of only 700 grams, and the small envelope, the HT100 is found to be ideal for long missions owing to the possibility of mounting multiple TU's on the satellite platforms to satisfy high total impulse requirements, while having a minimum impact on the total launch mass. Due to the modular architecture of this propulsion system, it can be easily installed on small satellites and can respond rapidly to market needs. Therefore, SITAEL's HT100 is found to be an effective propulsion system to realize a highly performant and responsive constellation capable of coping with a wide range of spatial and temporal resolution user requirements.

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