LOW THRUST ION PROPULSION FOR ADVANCED APPLICATIONS ON SMALL SATELLITE CONSTELLATIONS

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Abstract

The paper presents an overview of the main possible applications of low thrust ion propulsion (ref. thrust level in the 2-12 mN range) on small satellites (300-900 kg) LEO constellations, namely: drag-compensation on very low orbit (300-500 km) remote sensing satellites, co-planar orbit raising (up to an altitude of 2000 km), achievement of a separation in Ω, orbit maintenance and gap filling, end-of-life orbit disposal.
Besides, the topic of autonomous navigation based on a GPS system, used to continuously monitor the position of each spacecraft, on a navigation computer (programmed to assess how and when corrections are needed) and on low thrust ion propulsion, used as an actuator of the closed loop control, is addressed.
Finally an outlook on the RMT ion propulsion technology, developed by LABEN Proel Tecnologie Division, is provided, with particular reference to the thruster controllability features.

Nomenclature list

T (N) = Thrust level.

τ_f (s) = Thruster Total firing time.

τ_m (s) = Maneuver time

M_{pr} (kg) = Mass of propellant for the mission.

M_o (kg) = Satellite BOL (Begin of life) mass or mass at the start of the maneuver.

M_f (kg) = M_o - M_{pr} = Final mass of the satellite after the maneuver.

I_s (s) = Thruster specific impulse.

ρ (kg/m^3) = Atmospheric density.

D_F(N) = Average drag force acting on the satellite

ΔV(m/s) = Mission Velocity increment.

g = (m/s^2) 9.807 (gravity accel. at sea level).

C_D = Mean drag coefficient (assumed = 2.2 in the frame of the present simplified analysis)

S (m^2) = Average satellite drag cross-section

v (km/s) = Satellite orbital velocity.

μ (km^3/s^2) = 3.986 10^5 km^3/s^2 (gravitational constant)

r (km) = Orbit radius.

RMT = Radiofrequency with Magnetic field ion thruster.

SPT = Stationary Plasma Thruster.

Ω = Right Ascension of the Ascending Note.

Introduction

By the year 2000, LEO small satellite (300-900 kg) constellations will play a major role in many application fields of the space commercial exploitment. The largest near term market is communication, TV broadcasting, voice and data. Small satellite constellations can provide uniform worldwide communications and can increment capacity as demand grows. Beside they are able to guarantee increased reliability through the redundancy concept and to favour the introduction of new and emerging technologies through constellation replenishment.

After telecommunication, the second emerging market for satellite constellations is space based environmental monitoring. As the satellites of a constellation are spatially distributed, they are ideally suited to observe the global environment. Small satellite constellations are, as well, of great interest also for tactical military applications, in particular for monitoring the operation theatre.

Quite a number of small satellite LEO constellations (e.g. Globalstar, 48 satellites, 500 kg mass, at 1400 km; Skybridge, 64 satellites, 800 kg mass, at 1457 altitude; Teledesic, 288 satellites, 800 kg mass, at 1350 km; M-Star, 72 satellites; Celestri, 63 satellites) are now under development or have reached an advanced stage of mission definition.

In the field of environmental monitoring a small constellation, named COSMO [1], for the Mediterranean area monitoring, has been recently proposed and studied by Alenia Aerospazio.

Electric Propulsion and in particular Ion Propulsion (IP) can provide significant benefits if used on board satellites of a constellation, rendering feasible and cost-effective new and more challenging missions.

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Remote sensing constellations and autonomous navigation based on ion propulsion

At present there are strong motivations to lower the operating altitudes of earth observation / remote sensing satellite constellations, equipped with optical sensors or SAR instrumentation, from the range of 600-900 km to a range of about 300-500 km, due to the following advantages [2] [3]:

- Possibility to relax the requirements (in terms of angular resolutions) and then the costs of the optical payload, still maintaining the desired linear resolution on ground.
- Reduction of the transmitting power and the receiving sensitivity of active systems based on SAR's.
- The accomodation of a reduced size, mass, power demanding instrumentation on small satellites (e.g. 300-900 kg) produce the reduction of mission costs (both the launch and the satellite bus costs), or alternatively, can increase the number of satellites accomodated in a single launcher.
- The lowering of the operating altitude can significantly reduce the launcher requirements (as well impacting on costs) and/or the requirements on the satellite propulsion system for orbit acquisition.

IP can play a key role for remote sensing missions on small satellites. As an example, assuming the need for a spatial reduction of 1 to 2 m and using a reduced size optical mapping instrument (OMI) [4] as the primary sensor, a mission lasting 5 years can be accomplished at an altitude of 300 km, with a satellite in the 300 kg mass range.

Currently LEO remote small sensing satellite constellations (e.g. COSMO) are baselined in sun-synchronous orbits and have a design lifetime of, at least, 5 years in the above mentioned altitude range.
For such kind of missions, an autonomous orbital control system based on a GPS navigation system and on IP, used as actuator of the controller, can allow the achievement of a very accurate control of the orbital height (of the order few tens meters).

Autonomous navigation, preferably combined with IP, will serve three main purposes:

1. Reducing the ground control costs, the ultimate objective being a large reduction of the ground control stations required for satellites monitoring, and replacing attended stations with unattended ones.
2. Reducing the propellant mass to be carried on-board for orbit-keeping and maneuvering.
3. Achieving very fine, also continuous, orbit control: a tight orbit control, throughout the orbital period, is essential to insure the repeat characteristics of the ground tracks.

The control strategy proposed by Alenia Aerospazio is based on the measurement of the orbital parameters, exploiting a GPS receiver [5] [6] and an estimator and comparing them with a reference orbit derived from an on-board orbital propagator, modelling all known effects. From the comparison of the measurements with the data computed by a reference propagator, the variation of the orbit semiaxis is derived. This variation is used to determine the input to the IP system.

The IP actuator consists of one or two low level thrusters (based for example on the RMT [7] [8], by LABEN or UK10 [9], by MMS/DERA) aligned along the plus-minus roll axis, so that the maneuvers are performed in the orbit plane only. A variable thrust level is adopted with 1 mN steps, and the frequency of thrust changes, to perform the in-plane control, is in the range of 0.01-10 Hz.

The thruster is activated by commands sent by the on-board computer, which performs the calculations concerning actuation instants and thrust levels using GPS-derived position data, the orbital propagator and a mathematical model of the thruster itself.

The results show that long terms semiaxis variations, and in particular those due to drag, can be compensated automatically keeping the maximum deviation between the desired and the achieved orbit within few meters accuracy.

The flow chart of the proposed Closed Loop Orbit Keeping System (CLOKS) is shown in Fig. 1.

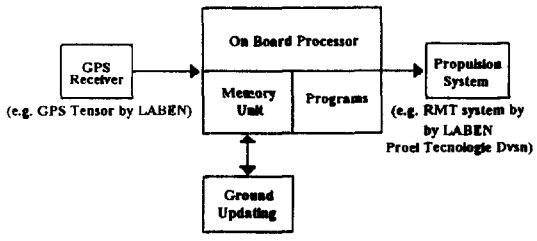

Fig. 1: Flow Chart of the Closed Loop Orbit Keeping System (CLOKS)

In the frame of autonomous navigation [10] concept IP (specific impulse of about 3000 s) with thrust level in the milliNewton range (e.g. 2-12 mN) could provide the best trade-off between the minimization of propellant mass and compatibility with the power generation system (in the range of 1300-1800W) typically associated to a small satellite of the new generation [11].

Within the CLOKS approach, low thrust IP can be used either to provide a continuous low thrust or an on/off thrust.
Demanding missions, where in particular a very tight orbital height control has to be achieved in an extremely variable environment (when, for example, the satellite average drag cross-section, the operating altitude, the solar activity can assume values within a wide range), can take advantage from the adoption of a control strategy based on the concept of acting both on the thrust level (provided by an IP system with real time throttleability of the thrust level) and on the thruster firing duty cycle.

Autonomous navigation with IP, in Molniya orbit

The autonomous navigation concept based on GPS and IP is especially valuable in the control of orbits that are particularly sensitive to perturbing forces, or for constellations for which very precise separations of satellites are mandatory. The Molniya orbit is included in the former category. This orbit is very useful for providing reliable high latitude communications coverage, especially of mobile communications. It is highly elliptical, has an inclination of 63.4 deg, and a period of about 12 hours. The apogee is at geostationary altitude and the perigee at perhaps 1000 km, so that a satellite in this orbit covers a quasi-stationary ground track at high latitudes for about 8 hours out of every 12. Three satellites can provide continuous coverage over a given geographical area, and non-tracking antennas can be used, minimizing cost.

Unfortunately, this is not a truly repeating orbit and uncorrected drifts can be permitted for only a few days. The rapid change in the longitude of apogee with time is caused by the cyclical change with time of the semi-major axis. Correction of these perturbations must be carried out every few days, with thrust applied at both apogee and perigee. The necessary annual velocity increments, amount to about 45 m/s at apogee and about 5 m/s at perigee.

The firing of low thrust ion thrusters (like the RMT or UK 10) both at apogee and perigee, at a reference thrust level of 10 mN, could successfully guarantee the feasibility of a 10 years mission with $M_{pr} \cong 9$ kg (considering a satellite BOL mass M_o = 500 kg).

Low thrust ion propulsion for drag compensation tasks

The use of a low thrust IP system (e.g. the RMT system) on remote sensing satellite constellations has been assessed within a simplified analysis in which a multi-year drag compensation mission is considered at an operating altitude range between 300 and 500 km.

The assumptions that have been introduced, to carry out the assessment, are hereunder summarized:

- Mission centered around a year of maximum solar activity (worst case situation).
- The "cosin" model has been assumed to describe the variation of the average (within the orbit period) air density ρ (depending on the altitude) versus the solar activity.
- Data on ρ value of 300 km have been taken from literature **[12]**. Values of ρ at higher altitudes have been calculated assuming a variation of ρ versus altitude according to the exponential law, based on the gas law **[11]**.
- The model for the drag force D_F acting on the satellite is the traditional one, namely:

$$D_F = \frac{1}{2} \rho v^2 F S C_D \text{ (1)} \qquad (1).$$

- The following constraints have been considered on the thruster operation:
  - Thrust level: 2-12 mN
  - τ_f: up to 15000 hrs
  - I_s: 3000 s.
- Satellite BOL mass M_o = 500 kg.
- Satellite drag cross-section(2):
  minimum cross-section S_{min} = 1.5 m^2
  maximum drag cross-section S_{max} = 4 m^2.

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(1) *F [11] is a coefficient assuming the following value*

$$F(i) = (1 - \frac{r\omega}{v} \cos i)^2 \text{ if } i \leq 90°$$

$$= 2 - (1 + \frac{r\omega}{v} \cos i)^2 \text{ if } i > 90°$$

where: i is the orbit inclination
ω is the atmospheric rotation.

(2) *From the drag compensation task viewpoint the near-noon sun-synchronous orbits (typically associated to the operation of an optical payload) are more demanding than the dawn-dusk sun-synchronous orbits (typically associated to the operation of a SAR payload).*
For dawn-dusk orbits a more or less fixed orientation of the solar panels (whose surface are always oriented orthogonal to the sun vector to maximize the power generation) is compatible with a minimum satellite drag cross-section (being the solar panels oriented parallel to the satellite velocity vector).
In the case of near-noon orbits the mean satellite drag cross-section (averaged along an orbit period) can span from a minimum value (minimum drag cross section), where the solar panels have a fixed orientation resulting always parallel to the satellite velocity vector, to a maximum value.
The maximum value corresponds to the case where the solar array orientation mechanism orients the panels orthogonal to the sun-vector during the illuminated semi-period of the orbit and parallel to the satellite velocity vector during the eclipse semi-period.

The parameters summarized in Tab. 1 have been calculated, on the basis of the above presented assumptions

Altitude (km)		300		350		400		450		500	
Inclination (deg)		96.67		96.85		97.03		97.21		97.4	
Solar Activity level		low	high	low	high	low	high	low	high	low	high
Density ng/m^3		8.00	36.50	2.77	17.27	0.96	8.17	0.33	3.86	0.12	1.83
D_F (mN)	S=1.5	0.80	3.66	0.28	1.72	0.09	0.81	0.03	0.38	0.01	0.18
ΔV_y (m/s)	min.	50.56	230.69	17.41	108.40	5.98	50.86	2.06	23.86	0.71	11.20
D_F (mN)	S=4	2.14	9.75	0.74	4.58	0.25	2.15	0.09	1.01	0.03	0.47
ΔV_y (m/s)	max.	134.84	615.19	46.42	289.07	15.95	135.63	5.48	63.63	1.88	29.85

Tab. 1: Drag forces and yearly velocity increments (ΔV_y) as function of altitude, drag cross-section and solar activity

The carried out analysis has produced the conclusions summarized in Tab. 2:

	Satellite average cross-section in the orbit period (m^2)					
	1.5	2	2.5	3	3.5	4
300 km	up to 7	up to 4.5	up to 3.5	up to 3	up to 2.5	up to 2
350 km	>15	>15	up to 14	up to 8	up to 6	up to 5

Tab. 2: N° of years for which the drag compensation mission is feasible with the assumed constraints for thruster operation

At altitudes of 400, 450, 500 km, for any considered average satellite drag cross-section in the range 1.5-4 m^2, the thruster operation respectively with a max thrust level of 9 mN, 4 mN, 2 mN can guarantee the feasibility of missions of duration up to 15 years.

The use of low thrust IP is capable of limiting the propellant mass necessary to fulfil the mission to a small fraction (not higher than 5-10%) of the satellite mass

$$M_{pr} = T\tau_f/(gI_s) \qquad (2)$$

For example with T = 12 mN, τ_f = 15000 hrs, I_s = 3000 s, M_{pr} results about 22 kg, in any case <30 kg, also considering 20% of margin.

Orbit acquisition with low thrust ion propulsion

Co-planar orbit raising

One of the major issue, when dealing with propulsion tasks related to the management of a telecom satellite constellation (e.g. Gobalstar, Teledesic, Skybridge) is the orbit acquisition, namely the maneuver to transfer each satellite of the constellation to the nominal operating orbit. IP can definitely play a key role for the above task, thanks to its intrinsic capability to dramatically reduce the propellant demand associated to the mission. This feature can allow a significant reduction in the single satellite launch mass rendering feasible the increase in the number of satellites that can be accomodated in a single launcher.

Early work [13] led to the derivation of the following equation, which gives the mass of propellant, M_{pr}, required for a spiral orbit-raising maneuver between two co-planar circular orbits:

$$M_{pr} = M_f \left[\exp\left\{ \frac{\mu^{1/2}}{g I_s} \left(\frac{1}{r_o^{1/2}} - \frac{1}{r_f^{1/2}} \right) \right\} - 1 \right] \qquad (3)$$

In this expression, M_f is the mass in the final operational orbit, r_o and r_f are the initial and final semi-major axes of the orbit. This equation applies strictly to the drag-free case with continuous thrust, but it has been shown that it provides a good approximation even when interruptions of thrust are experienced during eclipses and when the residual upper atmospheric drag is considered.

Equation 3 (applicable to both circular and elliptical orbits) can be used to relate the mass of a satellite, at any time during an orbit-raising maneuver, to the initial and instantaneous magnitudes of the semi-major axes.

Assuming continuous thrust, the time τ_m taken to execute a given orbit transfer, using total thrust T, is coincident with τ_f and can be derived from (2).

Whether the thrust can continue to be applied through eclipses depends upon battery capacity and recharging power budgets; any interruptions to thrusting will lead to corresponding increases in transfer times.

To illustrate important features of the transfer process, M_{pr} is plotted against I_s in Fig. 2 for the case of a maneuver from an initial altitude of 500 km to an operational orbit, typical of some of the proposed constellations of small communications satellites, of 2000 km. As an example, this altitude has been proposed for the Eco-8 constellation. Exhaust velocities typical of a bi-propellant chemical system, a SPT and an IP are indicated, M_f are taken to be 100, 300, 500 and 800 kg.

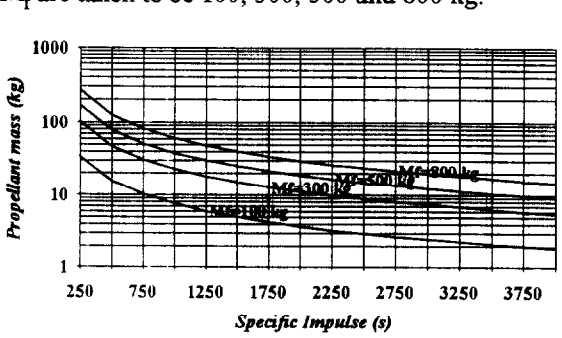

Fig. 2: Orbit raising from 500 km to 2000 km

It can be seen that very large mass savings can be obtained by using a high I_s propulsion system. Of course, other constraints, such as the time allowed for the transfer and the power available, also influence this decision. Indeed, when all factors are taken into account, an optimum value of I_s can usually be derived for any particular mission [8].

Fig. 3 provides another example of orbit transfer data to illustrate the thrust and mission times required.

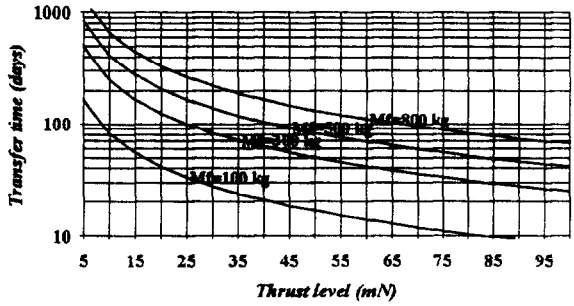

Fig. 3: Orbit raising from 500 to 2000 km using high spec. impulse propulsion

It is clear from this example that the most significant problem with the use of low thrust devices for orbit-raising is the time required, which can become many weeks or months. This time is determined mainly by the power available and the thrust that can be achieved using it. For this reason, a higher thrust, lower I_s thruster as the SPT is attractive for this task, if the shortest transfer times are needed. However, if mass is a prime driver, the high I_s gridded ion thruster is the best choice. It is therefore plausible to suggest that the replenishment of satellites within a constellation, has less constraints than does the initial emplacement of the constellation. It can therefore be advocated that the highest performance thusters should be used for such tasks, namely low thrust gridded ion thrusters.

Achievement of a separation in Ω

Another significant possible benefit of using IP for orbit acquisition maneuvers can be focused considering the case of a satellite constellation (for example of the Globalstar class) operating at a circular orbit of 1400 km, with an orbit inclination of 52°.

The strategy that has been considered foresees the launches of 2 satellites. The first satellite is launched in a circular orbit with altitude lower than the nominal one. While the second satellite is deployed in a circular orbit with a higher orbit altitude. In this way one satellite has a recession with respect to the nodal axis of the nominal orbit, while the second satellite has a precession. After a certain period the two satellites are shifted of 45° (22.5°x 2) in Ω, as desired. This strategy, integrated with IP adoption, is strongly suggested (also if there is a certain waiting time for orbit transfer) due to the significant advantages deriving from mass saving and launch cost minimization.

The data presented in Tab. 3 define the parameters for the feasibility of the mission with low thrust IP in comparison with chemical (bi-propellant) propulsion.

Final Orbit (nominal) 1400 km			Ion Propulsion		Chemical Propulsion	
Parking Orbit	Time to achieve Δ(ΔΩ) = 22.5°	ΔV to achieve the final orbit of 1400 km	Mass of Propellant (including 20% margin)	Time to achieve the final orbit	Mass of Propellant (including 20% margin)	Time to achieve the final orbit
800 km	24 days	293 m/s	6.9 kg	1948 hrs	61 kg	3.86 hrs
1000 km	39 days	191 m/s	4.5 kg	1275 hrs	40 kg	2.57 hrs
1200 km	82 days	94 m/s	2.2 kg	626 hrs	20 kg	1.28 hrs
1600 km	93 days	90 m/s	2.1 kg	602 hrs	19 kg	1.23 hrs
1800 km	49 days	177 m/s	4.2 kg	1180 hrs	38 kg	2.38 hrs
2000 km	34 days	261 m/s	6.1 kg	1736 hrs	55 kg	3.46 hrs

Tab. 3: Orbit acquisition of 2 satellites of the constellation whose orbits are shifted of 45° in Ω

Notes to the table
The considered ion propulsion system is based on a couple of ion thrusters each capable of providing a thrust level up to 12 mN and a I_s of 3000 s, mounted with a canting angle of 30°.
The reference chemical thruster (Bi- propellant) is assumed to provide a thrust level of 10 N, at a I_s of 280 s.
The considered satellite launch mass is 500 kg.

Constellation orbit maintenance and gap filling

Satellites of a LEO constellation dedicated to telecommunication tasks are subjected to perturbing forces. Perturbing forces acting on the satellite orbit are (atmospheric drag is not significant at altitudes above 700 km):

- Earth's oblateness, impacting the uniformity of the gravitational field, has significant influence on the Ω;
- Luni-solar forces (mainly the gravitational attraction of the Sun and the Moon), but also solar radiation pressure, alter mainly the orbit inclination at altitudes higher than 700 km.

The analysis of the orbit perturbations for the Globalstar orbits has been carried out by Alenia Aerospazio. The errors of the orbit parameters (namely the semiaxis, eccentricity, inclination and Ω errors) have been computed by considering as a reference orbit the nominal Globalstar one, with only the perturbations produced by J_2 (2nd" zonal harmonic of the geopotential). The actual orbit has been computed by considering all the external perturbations.

The results of the computations, in terms of overall velocity increments necessary for the constellation orbit maintenance throughout a reference

operational lifetime of 10 years, have provided the following figures:

$\Delta V_R \cong 20$ m/s (due to semiaxis + eccentricity errors)

$\Delta V_i \cong 10$ m/s (due to inclination error)

$\Delta V_\Omega \cong 30$ m/s (due to Ω error)

The total ΔV for the orbit control is about 60 m/s.

The detailed analysis of the impacts of recovering the gap filling between adjacent satellites in a constellation has not been performed in the present work (most likely it will be the subject of future activities). It is however reasonable to assume that the necessary ΔV can be in the range between 70 and 100 m/s for a 10 years mission. If we include also the gap filling maneuver the total max. figure for ΔV necessary for orbit maintenance and gap filling should be of the order of 150 m/s.

With the assumed satellite mass, a single ion thurster operated at 10 mN or, alternatively, a couple of ion thrusters operated in parallel at about 6 mN (considering a canting angle of 30°) could perform the mission being fired for a time of the order of 2000 hours.

The disposal of satellites at the end of life

With the ever-increasing hazard of collision in space with defunct satellites and debris, positive steps are required in order to prevent a worsening of this situation in the future. This is especially true for the orbits that will be populated with large constellations of small and medium-size satellites, where collisions could cause the problem to escalate rapidly.

A simple way of alleviating this difficulty will be to de-orbit future spacecraft so that they decay into the Earth's atmosphere, or to raise them to an altitude where they are not likely to present a problem. IP is ideal for this purpose, because the time constraints, which apply to orbit emplacement, are absent, and the complete power generating capability of the solar arrays can be employed.

For de-orbit of remote sensing and communication satellites, from the moderate and low altitudes suggested for a variety of constellations, the propellant requirements have been calculated assuming a lower altitude of 200 km, from which decay will be very rapid. The results are plotted as a function of initial altitude in Fig. 4 for chemical thrusters, SPT and IP.

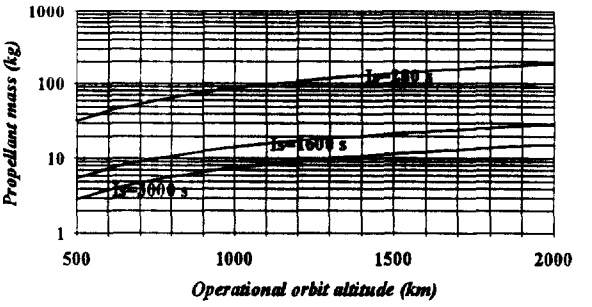

Fig. 4: De-orbiting of a satellite (M_f = 500 kg) from an operational altitude in the range 500-2000 km to a burial altitude of 200 km

It is evident that the propellant requirements for chemical systems are in general prohibitively large, apart for the lowest altitudes. It is also clear that IP will accomplish the task with an expenditure of no more than 3 to 15 kg of propellant.

An alternative approach for the disposal of satellites belonging to telecom LEO constellations is to move them into a higher orbit.

The altitude comprised between 2000 and 5000 km seem suitable because, being it subjected to high energy particles bombardment, it is generally not used for operational satellites. This may suggest to use the above mentioned altitude range as graveyard, for example, for LEO telecom satellites, belonging to constellations, at the end of their operatinal life.

Since the time required to deorbit a satellite at the end of the mission is not of paramount importance, low thrust (2-12 mN) ion thrusters can indeed be used for accomplishing the orbit raising maneuver. Considering for example a Globalstar class satellite (nominal operating altitude of 1400 km), the maneuver to reach a graveyard orbit of 2500 km ($\Delta V \cong 460$ m/s), performed with a 10 mN ion thruster fired approximately for half of the orbit period (total firing time about 6400 hrs), could be fulfilled in about 1.5 years with a propellant consumption of about 8 kg.

Once the burial orbit has been achieved, the satellites tend to stay there for a very long time, since the effect of drag is negligible and the risk of having their orbital parameters changed by the disturbance forces, in such a way to cross the orbit of other operational LEO satellites in lower orbit, is practically negligible.

RMT Thruster description

The RMT thruster uses a radiofrequency plasma discharge excitation. A relatively small magnetic field (~100 G) is implemented for the optimization

of the thruster performance in every working regime.

A schematic of a first RMT engineered version is provided in Fig. 5.

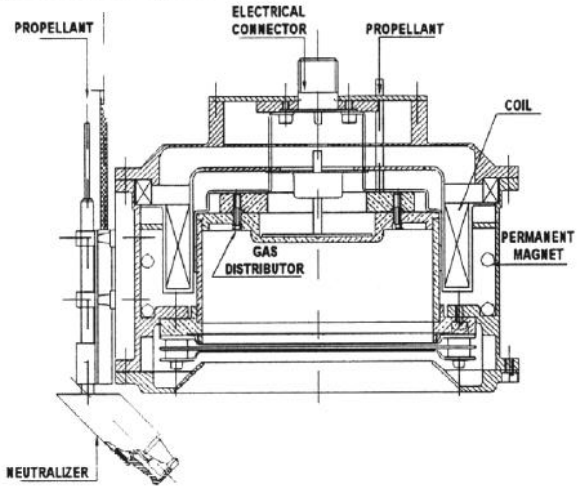

Fig. 5: Sectional schematic of the RMT ion thruster

The discharge chamber is made of Al_2O_3 and contains one component only, acting both as Xenon distributor and electrons collecting electrode. The RF electrodes consists of two metal pieces brazed on the outside walls of the discharge chamber rear side. SaCo permanent magnet rings and a coil provide an axial magnetic field ranging in the order of 100 G, in order to enhance the efficiency of the ionization process as well as the RMT thrust throttling capabilities.

Three main input parameters, namely the Xenon mass flow $\dot{m}$, the RF power W_{RF} and the Magnetic Coil current I_c, allow to control the thruster operation, providing the required throttleability (see Fig. 6):

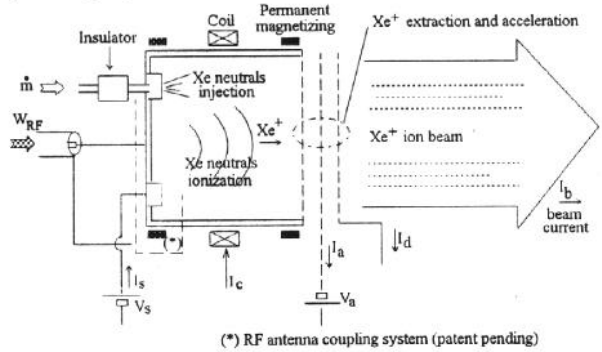

Fig. 6: RMT input parameters

For every ($\dot{m}$, W_{RF}) combination, the beam current I_b (therefore, the thrust) depends on I_c.

RMT controllability

The control of I_c (magnetic field) only can provide fast (a bandwidth in the order of 100 Hz has been demonstrated) and wide thrust variations, as well as a thrust modulation around a fixed operating point.

If max efficiency η_t is desired, a two level control strategy is conceived:

- a main controller setting W_{RF}, $\dot{m}$ [3] for coarse thrust level (gross changes);
- a closed loop fine controller of the beam current, acting on I_c for optimization of thruster operation, main controller errors compensation and fast small thrust variation.

RMT modelization

A static modelization **[14]** of the thruster has been performed in order to identify the optimum path model (on the basis of a chosen criterium like, for instance, the max η_t) and a look-up table with the necessary inputs for variable thrust commanding. Interpolating functions (indicated with the acronymous "int" in the formulas) were defined, spanning the overall ($\dot{m}$, W_{RF}) domain on the basis of experimental points:

$T = T^{int}(\dot{m}, W_{RF})$

$I_b = I_b^{int}(\dot{m}, W_{RF})$

$I_c^{opt} = I_c^{int}(\dot{m}, W_{RF})$

Then the numerical solution of the equations:

$$\begin{cases} T^{int}(\dot{m}, W_{RF}) = T_r \\ \eta_t(\dot{m}, W_{RF}) = MAX \end{cases}$$

was found to determine the optimum thruster input parameters versus required thrust (T_r).

As an example, Fig. 7 shows the optimum path vs. thrust, with η_t as parameter.

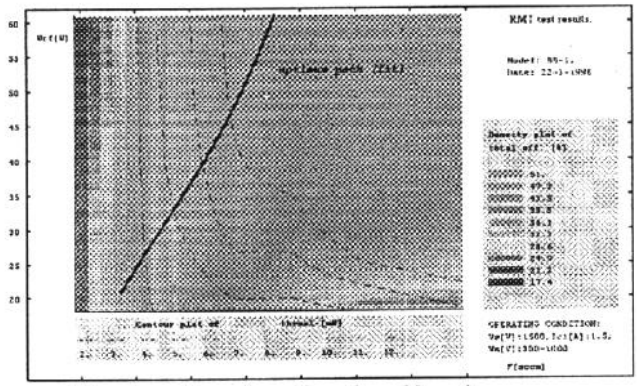

Fig. 7: Static modelization results. Map of total efficiency ($\dot{m}$, W_{RF}) domain

[3] *Fast variations of $\dot{m}$ could require new technologies for the realization of "analog" flow controller with response time of the order of few seconds (instead of minutes or hours typically associated to standard technologies.*

RMTA (RMT Assembly) Overview

Fig. 8 shows the configuration of the RMTA in a dual (redunded) version. The thrusting philosophy could be the following:

- both thrusters are on in normal operation and provide half thrust each
- one thruster must provide the full thrust in case of failure of the other branch.

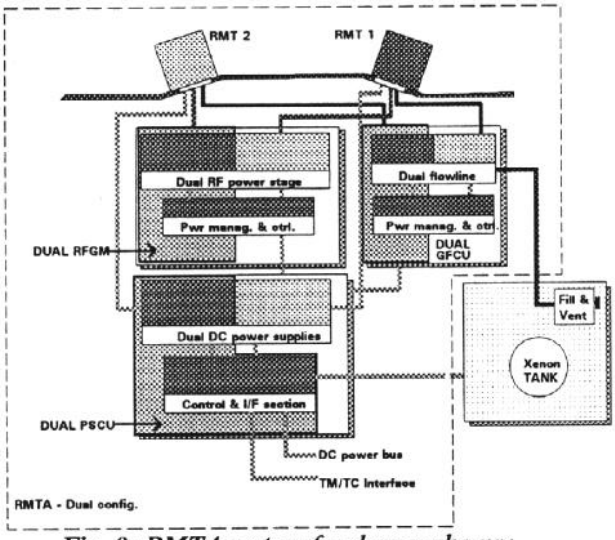

Fig. 8: RMTA system for drag make-up: dual (redunded) configuration

A preliminary information on mass and size of an RMTA dual system is presented in Tab. 4.

Unit	Mass Kg	Size w, mm	l, mm	h, mm
RMT1	1.6	∅ 150		130
RMT2	1.6	∅ 150		130
Dual RFGM	2.9	150	200	100
Dual PSCU	7	370	250	185
Dual GFCU (***)	2.2	100	100	100
Harness & piping	2	N/A		N/A
Subtot.	**17.3**			
Xe prop. (*)	11.5	N/A		N/A
Xe tank (**)	1.7	∅ 350		N/A
Fill/vent valves	0.8			
Total	**31.3**			

(*) Considering a 10000h operation time of two thrusters both working at 5 mN.
(**) Considering a tankage fraction of 15%.
(***) Including main valve redundancy. The dimensions are referred to innovative GFCU relying on a solid state design.

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