PULSED PLASMA THRUSTER OPTION FOR MYRIADE DEORBITING

N. Pillet [1], B. Pouilloux, P. Pelipenko, J. Folliard, F. Darnon, E. Chesta, P. Bousquet, F. Alby,
CNES - Centre National d'Etudes Spatiales
Toulouse, France

N.N. Antropov, G.A. Diakonov, V.N. Jakovlev, M.M. Orlov, P.M. Trubnikov, V.K. Tyutin
RIAME – Research Institute of Applied Mechanics and Electrodynamics
Moscow, Russia

Summary

At the end of their life, satellites have to be properly deorbited. For satellites in Low Earth Orbit, the requirement is to have an end-of-life controlled atmosphere re-entry in less than 25 years. For satellites equipped with an on-board propulsion system for orbit control, the deorbiting budget is generally taken into account in the initial sizing of the propellant budget. For satellites without propulsion, or with a propulsion system sized for attitude control only, among different options (propulsive or not) which are currently being studied at CNES, a solution based on small ablative Pulsed Plasma Thrusters (PPTs) seems to be quite attractive for its simplicity and performance. This paper briefly presents the development status and performances of the new generation of small PPTs currently under development at RIAME, and compares the PPT option (in terms of mass, complexity, deorbiting scenario etc...) to other propulsive options for the deorbiting of a typical CNES Myriade satellite.

Introduction

In 1996 CNES decided to develop a product line for micro satellites, mainly dedicated to scientific missions but offering also flight opportunities for technological demonstrations. The objective of this product line is to offer frequent flight opportunities with a low cost. This program is called MYRIADE. The in flight satellites are today DEMETER (launched mid 2004), PARASOL and ESSAIM (x4) (launched end 2004), and the next ones will be PICARD, MICROSCOPE and SPIRALE (x2).

These satellites do not all need a propulsion system to fulfil their scientific mission; in order to comply with recent international recommendations on space debris reduction, CNES is currently investigating the easiest way for deorbiting them at the end of their life.

One attractive option is to use the PPT technology, due to its relative simplicity and high performance (Isp up to 2000 s [1]) compared to chemical propulsion options. Therefore CNES, in collaboration with the Russian institute RIAME, is currently evaluating the feasibility and interest to use one PPT thruster for deorbiting the MYRIADE satellites.

[1]: Corresponding author: CNES - Propulsion and Pyrotechnics Department, 18 av. E Belin, Toulouse (F), nicolas.pillet@cnes.fr

The PPT technology status will be presented, including the most recent developments which are currently performed at RIAME and the future work which is planned on PPT at RIAME under CNES contract.

The PPT deorbiting option will be then analysed for the specific case of the Microscope satellite and compared to other chemical options; the results will be finally extended to the whole MYRIADE micro satellite family.

PPT technology development status

Pulsed plasma thrusters for attitude control have been used for the first time in orbit in 1964 on the Russian satellite Zond-2. Since this date, considerable progress has been performed on PPT technology in the US and in Russia.

As shown in fig.1, the basic operation of the APPT consists of repeated discharge pulses across a solid propellant surface. Thrust is produced by gas-dynamic and electromagnetic acceleration of the ablated and partially ionized mass. Since the propellant is solid (typically a bar of Teflon), there is no need of complex pressurized tanks and no risk of leakage, which is an attractive feature for a propulsion system.

The necessary energy for the discharge under high voltage (typically about 1000V) is accumulated into capacitors which are included in the PPT PPU (Power Processing Unit), to be interfaced with the SC power and TC/TM bus.

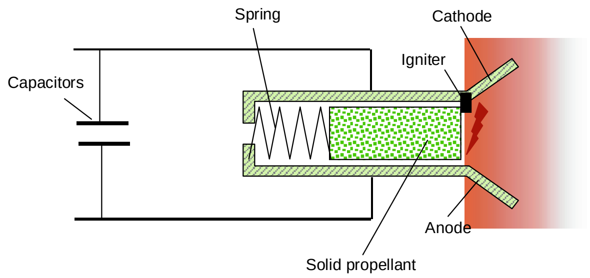

***Fig. 1** – PPT principle*

PPT performances make this propulsion option very suitable for small spacecraft attitude control or fine positioning (application to formation flying missions), but also for orbit control with high total impulse requirements.

A number of pulsed plasma thrusters have actually been flown on American satellites [2] in the late 60s (LES 6, 8 and 9), 70s (TIP 2 and 3) and 80s (Nova 1, 2 and 3). In the 90s, a research and technology program has been initiated at NASA GRC, involving universities and industries (Aerojet, former Primex Aerospace Company), and leading to the EO-1 successful flight experience in 2000 [4, 5].

The performances of these flight models range between 300s and 1150s of specific impulse for a maximum thrust of 30µN to 1.5mN and an input power below 100W (1Hz pulse frequency).

The PPT models initially developed in Russia by the Kurchatov Atomic Energy Institute could provide a higher maximum thrust level (up to 20mN), but were not operated for long periods. The very low thrust efficiency (below 10% due to low ionization rate of the ablated propellant) was the main limitation for the maximum achievable specific impulse (the Lorentz body force perpendicular to the discharge current and to the self-induced magnetic field affects only charged particles, while the evaporated propellant is accelerated by thermal expansion in the nozzle) and lifetime (the electrodes being heavily eroded).

The RIAME institute [3, 4, 7] has focused its efforts since 1992 on the development of a new generation of thrusters with improved efficiency. In particular, studies performed in the frame of INTAS project 97-1137 (dedicated to ablative PPT efficiency improvement) have shown that an appropriate optimization of the electric circuit can allow a discharge transition from an "oscillating" regime to a "quasi-aperiodic" mode (fig.3). The two main consequences are better energy conversion efficiency (higher ionization rate and lower late-time ablation phenomena and capacitor heat dissipations) and a reduced solicitation on the electrodes (no voltage sign change at capacitor bank terminals). In these conditions, up to 80% of the propellant can be ionized and accelerated by the $\vec{j} \times \vec{B}$ force.

RIAME has developed and tested a number of prototypes with stored energy varying from 20J to 150J.

In 2000 the geometry and dimensions of APPT discharge channel were conformed with the parameters of discharge circuit (capacitance, inductance and ohmic resistance). As a consequence its thrust efficiency was increased from 8-9 % to 15 % at stored energy of about 50 J and from 12-14 % to 25 % at energy of about 100 J. Such thrusters were defined as high-efficiency APPT of the next generation [4].

In 2003 the APPT was upgraded when the discharge channel was adapted to the energy storage with low ohmic resistance. The ohmic losses of stored bank energy were decreased from 15-20% to 1-2%. So, the efficiency was increased to ~25% at 50 J and ~40% at 100 J.

The laboratory prototype of upgraded APPT is shown in Figure 4. The prototype has the “traditional” for RIAME rail configuration with the plane electrodes and the side feeding of propellant . The thruster is based on two plane electrodes made of copper or steel, and a single or double Teflon bar (back-feeding and side feeding case respectively). In the case of lateral propellant feeding, the discharge chamber is completed by a rear insulating wall, usually made in alundum (Al_2O_3).

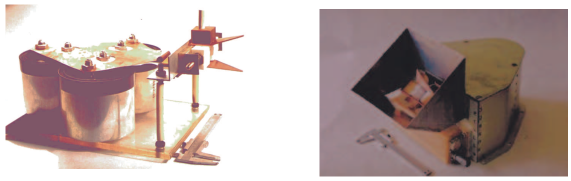

Fig 3 *RIAME APPT-50 Laboratory prototype (left) and engineering model (right).*

As shown in fig 4, the capacitive energy storage system, consisting of several sections connected in parallel, is interfaced to the thruster electrodes with thin copper connections. The different sections can be housed inside a common body or separately. The capacitors are charged by the PPU, having at its output the nominal voltage also needed to operate the discharge initiation unit (DIU), used for producing initial plasma in the discharge gap. The DIU is connected to the auxiliary electrode system fixed in the cathode, can be duplicated to provide redundancy and raise overall system reliability and consumes about 0.5-1% of the main discharge energy.

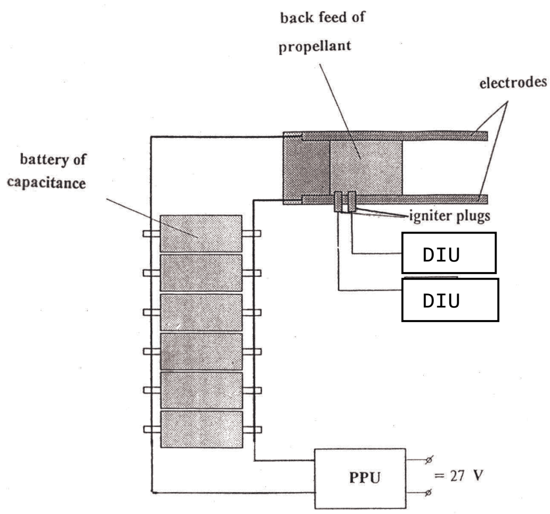

Fig 4 *PPT functional schematic*

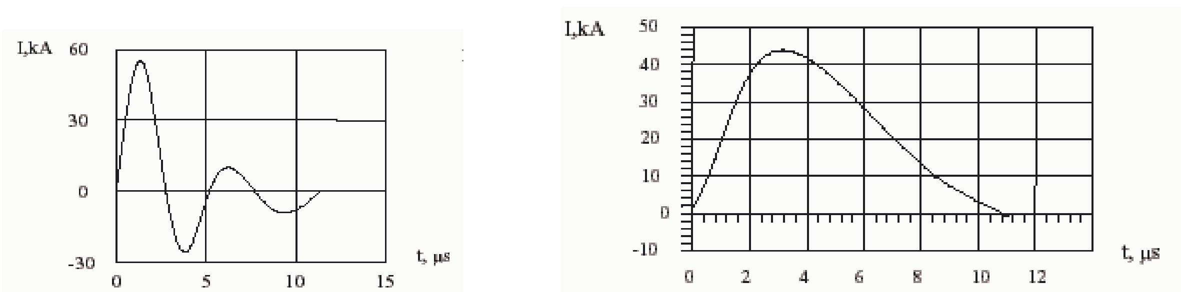

Fig 2 *– Discharge behaviour of APPT belonging to first (left) and second (right) generation*

The dependencies for the thrust and specific characteristics of APPT on the stored energy, parameters of discharge circuit, discharge channel geometry and its dimensions, frequency of pulses were experimentally investigated. The thrust impulse bit P and thrust efficiency η as a depending on the stored energy are presented in *Fig 5*.

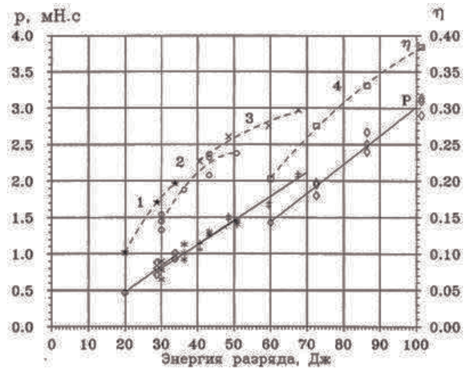

Fig 5. *Thrust and specific characteristics of the upgraded APPT (1 – two capacitors; 2 – three capacitors; 3 – four capacitors; 4 – six capacitors).*

All main characteristics of APPT-50 were verified experimentally during the testing of a laboratory prototype in RIAME MAI. The APPT-50 flight prototype manufacturing is ready now for the demonstration flight test in 2006 - 2007 years.

The low-orbit small unified space platform (USP) "Vulcan" is now developed in Research Institute for Electromechanics (NIIEM). It has 250-350 kg overall mass and 7-10 years lifetime. The platform can be readily adapted for different kinds of payload. The USP "Vulcan" includes the correction propulsion system (CPS) with upgraded APPT has been designed in NIIEM in collaboration with RIAME MAI. CPS of the USP "Vulcan" includes two APPT-50 thrusters and one block of power processing unit (PPU) as it is presented in *Fig 6*.

Two APPT-50 thrusters are placed on opposite satellite edges. The common PPU is placed inside the satellite body. The gyroscopic orientation and attitude stabilization system can turn the satellite of ±45^{o} around its vertical axis. This will allow to use one pair of opposite thrusters for the correction of the orbit altitude and inclination.

The CPS of USP 'Vulcan" will provide the characteristic velocity 85 m/s during the 7-year active lifetime.

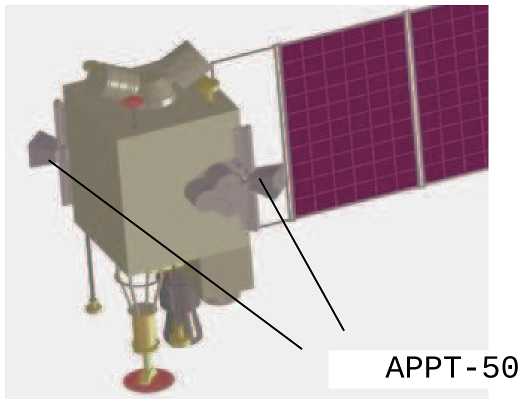

Fig 6. *Unified space platform "Vulcan" with correction propulsion system based on two APPT-50 thrusters*

In the frame of CNES 2004 R&T program, two new thrusters prototypes are being developed by the RIAME institute and will be fully characterized, used to validate theoretical ablation, ionization and energy transfer models, and finally optimized on the base of the achieved results.

[1]: Corresponding author: CNES - Propulsion and Pyrotechnics Department, 18 av. E Belin, Toulouse (F), nicolas.pillet@cnes.fr

Energy (J)	30
MIB (mNs)	1.0
Specific impulse (s)	1150
Frequency (Hz)	2
Thrust (mN) @2Hz	2
Power (W) @ 2Hz	60
Mass (Thruster + PPU) (kg)	7 (without Teflon)
Total Impulse (kN.s)	> 20 (≈1.7 kg Teflon)
Size of the thruster, mm (HxLxD)	100x160x160
Size of the PPU, mm (HxLxD)	< 55x80x160

***Table 1** – APPT 30 (RIAME) characteristics*

Mission analysis

IADC Recommendations

The world's 11 largest space-faring nations have formed the Inter-Agency Space Debris Coordination Committee (IADC). The IADC has recommended to the United Nations several actions in order to minimize the increasing number of debris in space. The main measures recommend de-orbiting the LEO (Low-Earth Orbit, altitude less than 2000 km) satellite to an orbit, which ensure a re-entry within 25 years with a complete depletion of the on board energy sources.

Space mechanics

One main parameter for the propulsion system is the required necessary delta V (DV) for achieving the deorbiting impulse. This DV is obviously depending from the initial and final orbits.

The final orbit is chosen such that the satellite can re-entry in less than 25 years and is therefore depending of the ratio S/M (S spacecraft surface in front of the velocity, M mass of the satellite) of the satellite.

With a typical value of a S/M = 0.8% M2/kg, the circular orbit 600 kM X 600 kM is considered as an acceptable (w.r.t IADC recommendation) final orbit.

For the Microscope spacecraft study, the characteristics of an APPT 30, presented in *Table 1* have been considered.

Results of the calculation for continuous low thrust trajectory from 3 initial circular orbits (700, 750 and 800 km) down to the 600 km circular final orbit are presented in Table 2 for a spacecraft launch mass of 220 kg.

In this continuous low thrust mode, for these circular orbits transfer, it is interesting to point out that the total DV to be provided by the thruster is not bigger than the Hohman DV requirement which could be produced by impulsionnal boost with chemical propulsion. In addition, the orbit considered here (6H / 18H) allow a continuous thrust mode due to the fact that there is no eclipse constraints.

Initial orbit (km)	**700**	**750**	**800**
DV (m/s)	53	80	106
Transfer time (days)	45	68	89
Propellant need (kg)	1.04	1.43	2.05
Total impulse need (kN.S)	12	16.4	23.5

***Table 2.** Characteristics for a transfer to the 600 km circular orbit*

Lay out of the PPT on the spacecraft

The thruster lay out on the satellite takes into account the following constraints:

-GNC: to be on a face such that the solar panel are permanently in front of the sun (with a certain tolerance) during the continuous thrust phase: this means that the PPT should be parallel to the solar array plan.

- Positioning margin with other equipment items / field of view of antenna etc...

- Alignment of the thrust toward the spacecraft Centre of Gravity

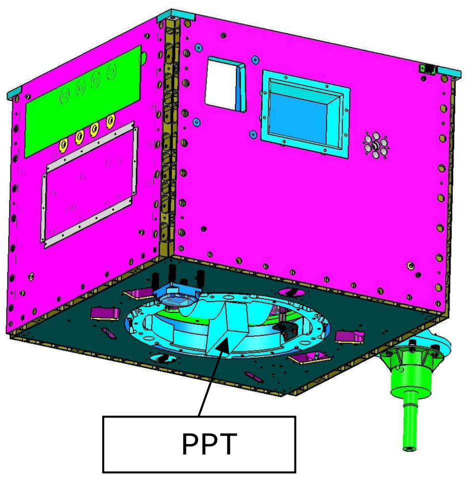

***Fig.** 7 - Lay out of the PPT on a standard Myriade spacecraft*

Attitude control during thrust phase

The AOCS of the satellite should be able to control the satellite attitude during the continuous thrust phase.

For this "one-thruster" configuration, without any possibility to control the thrust direction (no thrust orientation mechanism) it should be investigated that perturbating torque from the PPT itself, are well compensated by the AOCS on board actuators (i.e. reaction wheels, magneto-torquer ...)

The following figures have been taken into account to evaluate the maximum torque possibly induced by the PPT, to be compensated by the AOCS on board actuators (maximum capability is 0.2 mN.M for the magneto-torquer) of the spacecraft:

- Misalignment error of the PPT (mechanical mounting of the PPT on the structure, knowledge of the CoM position ...): 5 mm witch gives 0.01 mN.M with a 2 mN thrust. TBC
- Thrust off axis (max angle between theoretical thrust axis and effective thrust axis): 5°.

The thrust off axis of 5° gives an additional 4.3 cm misalignment w.r.t the center of mass of the spacecraft. If we cumulate both errors, we have approximately 5cm which gives 0.1 mN.M to be compensated by the magneto torquer of the spacecraft. As this value is in the order of magnitude of the on board mageto torquer capability, deeper AOCS studies would have to be performed to confirm the attitude control of the satellite during the thrust phases.

Comparison with other propulsive options

To do the deorbiting of the Myriade satellite, different options for propulsion system are considered and compared to the PPT option. This comparison has been done on the basis of 2 initial spacecraft mass which fully cover the Myriade product line: [100, 200] kg,

The figures (mass / volumes) are included in the table 3.

The capacity of the reference hydrazine propulsion system used on DEMETER and PARASOL satellites (6 liters hydrazine tank) is limited to approximately a DV of 52 m/s for a 200 kg spacecraft mass and it would be necessary to enlarge the tank capacity (up to 9.2 liters) to achieve the 80 m./s required for deorbiting from 750 km to 600 km.

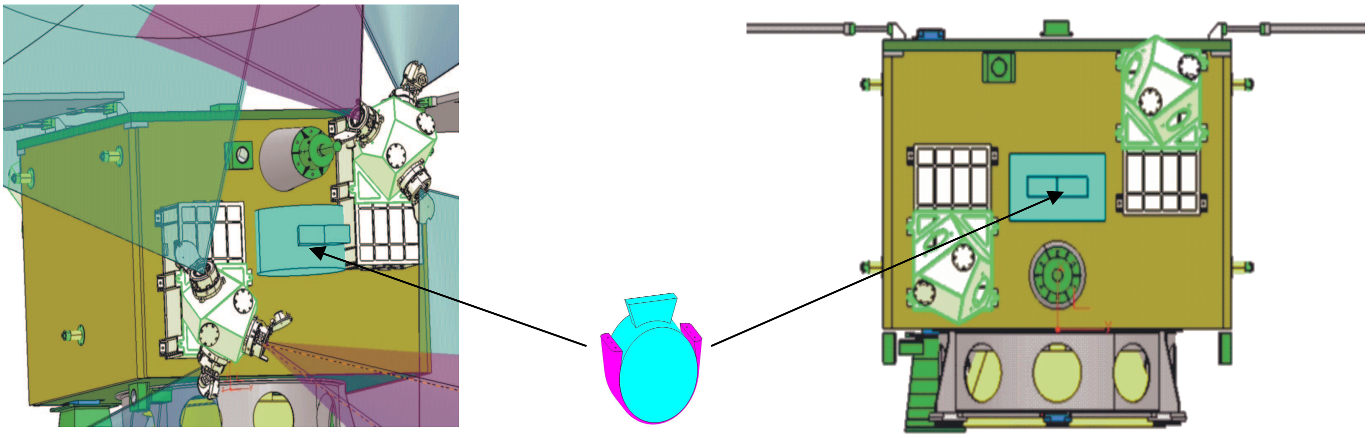

Fig. 8 - Lay out of the PPT on Microscope

		Myriade baseline	Solid pro-pulsion	Cold gas	Resistojet	APPT 30	HET
Propellant		Hydrazine	Butalite	Nitrogen	Nitrous oxide	Teflon	Xenon
SI (s)		220	260	70	110	1150	1000
Propellant mass (kg)	100 kg	3.6	3.1	11	7.3	0.7	0.8
	200 kg	7.3	6.2	22	14.6	1.4	1.6
Tank volume (l)	100 kg	6	NA	38	12	NA	1
	200 kg	9.2	NA	76	24	NA	2
Total mass (kg)	100 kg	8.8	<7	38	20	7.7	12
	200 kg	10	9.8	46	26	8.4	<13

Initial spacecraft mass

Table 3 - Propulsive options trade-off for a transfer from 750 km to 600 km circular orbit (DV = 80 m/)

This table shows that in term of propulsion system mass (including the propellant mass), the PPT solution is the best solution for the 200 kg spacecraft mass. For the 100 kg spacecraft mass, ths PPT propulsion system is compatable to the existing hydrazine propulsion system and the solid propulsion option. However, we can expect a big advantage of the PPT solution due to it's relative simplicity compared to the hydrazine and solid propulsion.

Conclusions

From this preliminary study, it seems that PPT could be an attractive option, for deorbiting the Myriade satellites at the end of their life. From a system point of view, it would be necessary to refine the mission analysis for a dedicated spacecraft, including the potential AOCS issues, and all other points which have not been treated in the present paper, to confirm the interest of the PPT solution.

From the PPT side, the coming CNES /RIAME work will be focused on the following topics:

- Concept and technology assessment of PPT
- Fabrication and test of a laboratory model
- Modelisation of propellant ablation and ionisation, modelisation of the discharge process
- Correlation of the model with experimental results
- Fabrication and test of an improved APPT model.

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[1]: Corresponding author: CNES - Propulsion and Pyrotechnics Department, 18 av. E Belin, Toulouse (F), nicolas.pillet@cnes.fr