Analysis of Hall-Effect Thrusters application to formation flying and drag compensation

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Abstract: **Hall effect thrusters are generally regarded as fairly high power devices reserved to North South station keeping or orbit transfer. However, the design of small HET is possible and has been demonstrated at ground in several countries. The first flown HET (EOL 1 in 1971) had a thrust of 20 mN.**

A system study has been performed to compare the relative merits of gridded ion thrusters, FEEP (5 to 15 mN range), resistojets and HET for drag compensation, formation flying (drag free) and coarse pointing of formation flying astronomy satellites. The study showed that small HET is the best compromise for the overall set of mission.

I. Introduction

Drag compensation and fine positioning are considered up to now as the domain of choice for cold gas, small gridded ion engines and FEEP.

However, small Hall-Effect Thruster (HET) are very good for drag compensation especially for the low input power required. For a given solar panel power, the available thrust is nearly two times the one of gridded ion thrusters and four times the one of FEEP: this enables a lower flying altitude for the benefit of Earth observation.

Drag compensation requires a wide dynamic thrust range. Existing Hall thrusters are already able to provide a wide range, e. g. 25 – 100 mN for PPS 1350 and 50 – 350 mN for PPS X000 despite the fact they are used in a narrow range for NSSK. It is anticipated to increase this range for small Hall thrusters (e. g. from 10 % to 100 % of nominal thrust) by simultaneous change of mass flow, discharge voltage and magnetic field. The implications on PPU and xenon supply are presented.

This wide dynamic range is very useful also for relative positioning combined with attitude control.

II. Mission Scenario Overview

A. Drag Compensation

Recent ESA-sponsored studies[1,2] on remote sensing applications have clearly demonstrated that the possibility to operate the satellite on a lower altitude orbit would give a net advantage in terms of:

- reduced wet mass at launch, thanks to a reduced payload mass for the same performance or
- enhanced payload performance

This possibility is brought about by the electric propulsion (EP), which allows to compensate orbit and attitude disturbance due to drag force with a little propellant expenditure with respect to chemical engines, thanks to the high

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specific impulse of EP, typically one order of magnitude higher than that of the most advanced Bi-prop engines. In fact the high drag force experienced by any satellite operating at an orbit altitude in the 300-350 km range (as considered in the aforementioned studies) would imply a high Delta-V overall the entire mission, such that conventional (chemical) propulsion would require a prohibitive amount of propellant.

Generally speaking, missions that would result to get some advantage from the EP inclusion, allowing the orbit decrease, belong to the following mission scenarios:

- LIDAR payload mission
- optical payload mission
- SAR (Synthetic Aperture Radar) mission

Among LIDAR scenario, two particular missions have been considered as reference in order to show the advantages obtained with the orbit reduction. These are Earth Care mission and Wales mission, both foreseen in the frame of the Earth Explorer programme. Table 1 provides some characteristic data for both mission baselines.

Table 1. Some mission data for Wales and Earth Care mission, using chemical propulsion for drag compensation according to baseline

Mission	Orbit	Payload	Wet mass
Wales	Dawn-dusk circular 450 km altitude	1 High power LIDAR	1438 kg
Earth Care	450 km altitude	Active : - LIDAR - Cloud Radar Passive: - Multi-Spectra Imager -Broad-band Radiometer	–

Wales mission results to benefit particularly from EP application for drag compensation. With a reduced payload aperture, in order to maintain the same performance, the payload requires more power than for the baseline aperture at any altitude, but because of the reduced drag area, the power required by the EP system decreases. A minimum is found around 300 km. This is particularly attractive since it will allow the use of the Rockot launcher instead of the Soyuz one, reducing appreciably the overall mission cost.

Table 2 provides propulsion parameters and some orbit characteristics at different altitudes for the second study case with an EP system based on T5 ion engine[1] or on a small Hall-effect thruster (mini-HET, mini-Hall), having 1500 s of specific impulse and 17 W/mN of specific power. The two systems appear very similar from the point of view of the final mass budget, the slightly higher xenon mass being compensated by a lower EP dry mass and a lower solar panel PDCU mass; as a matter of fact the difference in weight is basically negligible with respect to the satellite size (> 1400 kg).

Table 2. Comparison from T5 and mini-Hall application to Wales mission

T5				Small HET			
Mission characteristics				**Mission characteristics**			
Altitude (km)	320	300	280	Altitude (km)	320	300	280
Eclipse period	0.293	0.298	0.303	Eclipse period	0.293	0.298	0.303
Max drag force (mN)	5.8	8.7	13	Max drag force (mN)	5.8	8.7	13
Propulsive requirements				**Propulsive requirements**			
Total Impulse (10^5Ns)	4.9	7.3	11	Total Impulse (10^5Ns)	4,9	7,3	11
Specific Impulse Isp (s)	2000	2000	2000	Specific Impulse Isp (s)	1500	1500	1500
Propellant Mass (kg)	25.0	37.2	56.1	Propellant Mass (kg)	33,3	49,6	74,8
Max Thrust (mN)	6.7	10.0	15.0	Max Thrust (mN)	6,7	10	15
Specific Power (W/mN)	32	31	31	Specific Power (W/mN)	17	17	17
Power (W)	214	311	465	Power (W)	113,9	170	255

For the Earth Care mission, EP implementation has to be carefully evaluated on the base of a trade-off between cost increase and performance worsening of the passive payloads, as main drawbacks, and simplification and mass saving or performance improvement for the LIDAR payload, as main advantages. In any case the expected drag force is still in the **4-12 mN** range, depending on the chosen altitude.

In the frame of the optical mission scenario, as for the LIDAR scenario, the EP application would enable the orbit altitude reduction with more performing payloads or optics reduced size and mass. The reference mission considered for this scenario is the LOEW mission, proposed in the frame of the Living Planet programme.

Table 3. Main results from SAR mission study, using the PPS-1350 HET[1]

Parameter	Ref. Mission	L-band	C-band
Mission			
Orbit altitude (km)	629	340	350
Spacecraft			
S/C wet mass (kg)	2326	1526	1395
S/C dry mass (kg)	2152	1445	1330
Power cons. (daylight/eclipse) (W)	1563/1493	1294/1215	674/610
S/C drag area (m2)	-	9.1	8.5
Payload			
P/L mass (kg)	746	472	472
P/L power (max) (W)	2224	1309	1309
SAR frequency (MHz)	1258	1258	6000
Swath (km)	395	205	211
Propulsion System	Chemical	HET	HET
Isp (s)	230	2300	2300
Specific power (W/mN)	-	20.1	20.1
Thrust (mN)	-	18.8	15.1
Fuel mass (kg)	174	81	65

SAR payload missions would benefit from the possibility of a lower altitude orbit, enabled by the use of EP for drag compensation, analogously to the previous mission scenario. SAR missions are fostered for instance by the ESA's Earth Watch programme and by the GMES sentinel-1 satellite series. Table 3 provides a comparison between the chosen reference mission, the TerraSAR mission for SAR measurement in L-band, and two study cases, one still for L-band measurement, the other for C-band measurement, both using a HET with 2300 s of specific impulse and 20.1 W/mN of specific power, obtained as extrapolation of the PPS1350-G operational map.

B. Formation Flying Maintenance

In the FF mission scenario, the Darwin mission for the detection of Earth-like planets via infrared signal interferometry is among the most challenging ones. It comprises six telescope flyer (TF) satellites, one hub for TF signal collection and processing and one master satellite for ground communication and FF control. The mass and on-board power budgets are reported in Table 4. According to baseline the FF should enter a Lissajous orbit around the second Sun-Earth libration point, L2 and operate for 5 years, with a possible extension up to 10 years. Launch date is scheduled in 2015.

Table 4. Mass & power budget for DARWIN mission[3]

Satellite	Mass (kg)	Power (W)
TF	493	146
Hub	396	491
Master	179	283

Figure 1 reports a summary of the thrust requirements for the two main maneuver classes: during the observation time estimated to be 50% of the overall mission duration, the EP system is required to deliver low and highly accuracy thrust level on a continuous base for fine pointing, while coarse maneuvers (FF reconfiguration and rotation) and recovery maneuvers require thrust levels as high as possible for maximizing the observation time.

Micropropulsion requirements	μN (precision-pointing)	mN (coarse manoeuvres)
Proportional thrust control	required	preferred
Minimum thrust level	0.5 - 3 μN	~ 0.5 mN*
Maximum thrust level	100 μN	> 1 mN
Thrust resolution	< 3 μN	~ 0.5 mN*
Thrust accuracy	< 0.5 μN	< 0.5 mN
Maximum thrust noise	1.65 μN / √Hz	N/A*

*No stringent requirements have yet been defined

Figure 1. Thrust requirements for precision pointing and coarse manoeuvres for Darwin mission[4]

Total delta-v for coarse manoeuvres (m/s)							
Thrust level	Number of stars						
	100	150	200	250	300	400	500
150 µN	53	60	67	74	81	95	109
0.5 mN	64	77	90	103	115	141	166
1 mN	75	93	111	129	147	183	219
5 mN	119	160	200	240	280	361	441
10 mN	153	210	267	323	380	494	608
50 mN	293	421	548	675	802	1057	1311

Figure 2. Delta –v estimates for coarse manoeuvres[4]

Propellant required for coarse manoeuvres (kg)						
I_{sp} (s)	Coarse manoeuvre thrust level					
	150 µN	0.5 mN	1 mN	5 mN	10 mN	50 mN
6000	0.5	0.7	0.8	1.4	1.8	3.6
3000	1.0	1.3	1.6	2.7	3.6	7.1
1500	2.0	2.6	3.2	5.4	7.1	14.1
300	10.1	13.0	15.6	26.4	34.4	66.6
100	29.6	37.8	45.2	75.1	96.2	174.4
50	57.5	72.8	86.3	139.0	174.0	287.9

Figure 3. Propellant mass for coarse manoeuvres[4]

Concerning the delta-V requirement for coarse maneuvers, Fig. 2 provides some estimates, under certain assumptions, especially regarding the possible geometrical parameters of the FF reconfigurations and rotations. In order to maximize the number of investigated stars and in general the overall observation time, the thrust should be as high as possible. The value of Delta-v will increase with the thrust level but as can be seen in Fig.3, the propellant mass is basically a very low percentage of the TF mass at any specific impulse typical of the EP systems(>1000 s).

Preliminary studies[4] on propulsion technology assessment for Darwin mission conclude with the selection of cold microthruster as the most suitable choice for both maneuver classes. Concerning the EP option, FEEP technology is indicated as the best choice for fine pointing purpose, while it is strongly penalized by the contamination concern for the high-thrust maneuvers, involving well higher mass flow rates; in this case mini-GIT, mini-HET and resistojets are preferred.

Interferometry SAR mission scenario has been envisaged as very interesting in the next future. This kind of mission is enabled by the use of EP, for its unique capability of providing the required low thrust level and high thrust accuracy needed for FF maintenance.

Moreover the adoption of the EP will allow for lower altitude orbits, exploiting the benefits both in scaling of SAR payloads with altitude (reduction of required power) and in the improvement of interferometry resolution.

The need for a closed-loop formation control has been clearly identified in order to maintain the formation baselines behind the required margins, compensating for formation drift due to different ballistic coefficients and formation initialization errors.

Microthruster concepts appear more favorable as control system actuators than minithruster concepts from the point of view of deltaV and fuel consumption. However other aspects as dry mass, accommodation, reliability etc. could orient the choice towards minithrusters. On the other hand the need for a minithruster to be operated in an intermittent mode would place more demanding requirements in terms of on-off cycles; finally the intermittent working regime leads to a general oversizing of most EP subsystem and components.

An important consideration concerns the capability of the EP system of recovering the formation initialization error or unforeseen perturbation event. As an example it has been estimated that depending on the initialization error in terms of Delta –v and on the allowed correction time, the control system is required to provide a thrust in the **0.2-4.2 mN** range. This last aspect seems to prompt the choice of a mini-thruster based control system, otherwise a separate (relatively) high-thrust control system should be considered.

III. Mini-HET state of the art

Small/medium HET have been used in Russia for many years aboard sun synchronous orbit spacecrafts (METEOR, PRIRODA). The table 5 shows the characteristics of the thrusters built by the FAKEL company.

The life requirement was consistent with the mission objective. SPT 70 was used only for station acquisition and EWSK on Russian geostationary satellites, hence the relatively low life requirement. The ultimate life could be much higher. From the most recent qualification tests of SPT 100 (8900 hours) and PPS 1350 (10500 hours) it could be inferred that the life of a new 70 mm thruster should be in excess of 7000 h at full power for a discharge voltage of 300 V and much higher at lower voltages.

For a 50 mm dia thruster, the life could be traded off with maximum operating voltage. A reasonable total impulse objective could be 350000 N.s at voltages below 240 V.

Small HET are intensively developed in the USA, where the BUSEK BHT 200 (200 watt power) has been recently tested in space.

In western Europe, a small HET unit is under development at ALTA since 2003[5,6], following the identification of an interesting application niche for microsatellites[7]. It has been named HT-100 (Fig. 4) and is the smallest and lowest power HET ever developed in Europe; its performance and characteristics represent the state-of-the-art in the segment. The unit has a nominal operating power with xenon of 100 W and a permanent magnets configuration; this last characteristic and other design solutions have been implemented to simplify manufacturing and integration, thus maintaining recurring costs to a minimum.

During 2004 the first prototypes underwent two test campaigns (with different cathode arrangements) which allowed demonstrating the main performance aspects of the selected design. The prototypes showed stable operation in the power range between 10 and 235 W. A relatively flat anode efficiency in the range of 30±3 % was obtained over the 130-230 W power range

Table 5. Characteristics of Small FAKEL SPT thrusters

Thruster	SPT 35	SPT 50	SPT 70
Thrust mN	10	20	40
Power W	200	350	650
Isp s	1200	1250	1450
Efficiency	0.3	0.35	0.48
Life (specification) h	2500	2250	3100
Mass kg	0.4	0.8	1.5
Status	Eng. model	Flight model	Flight model

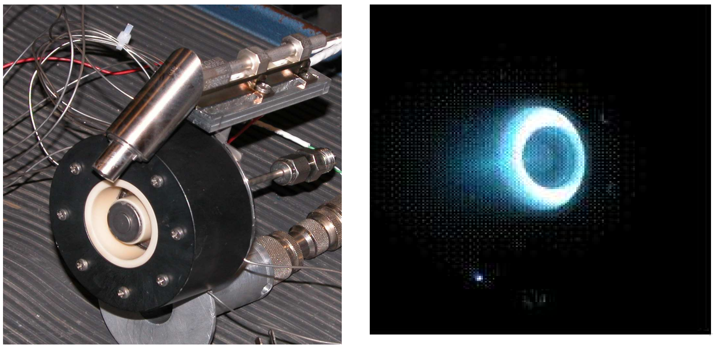

Figure 4. HT-100 thruster development model (DM2),©Alta

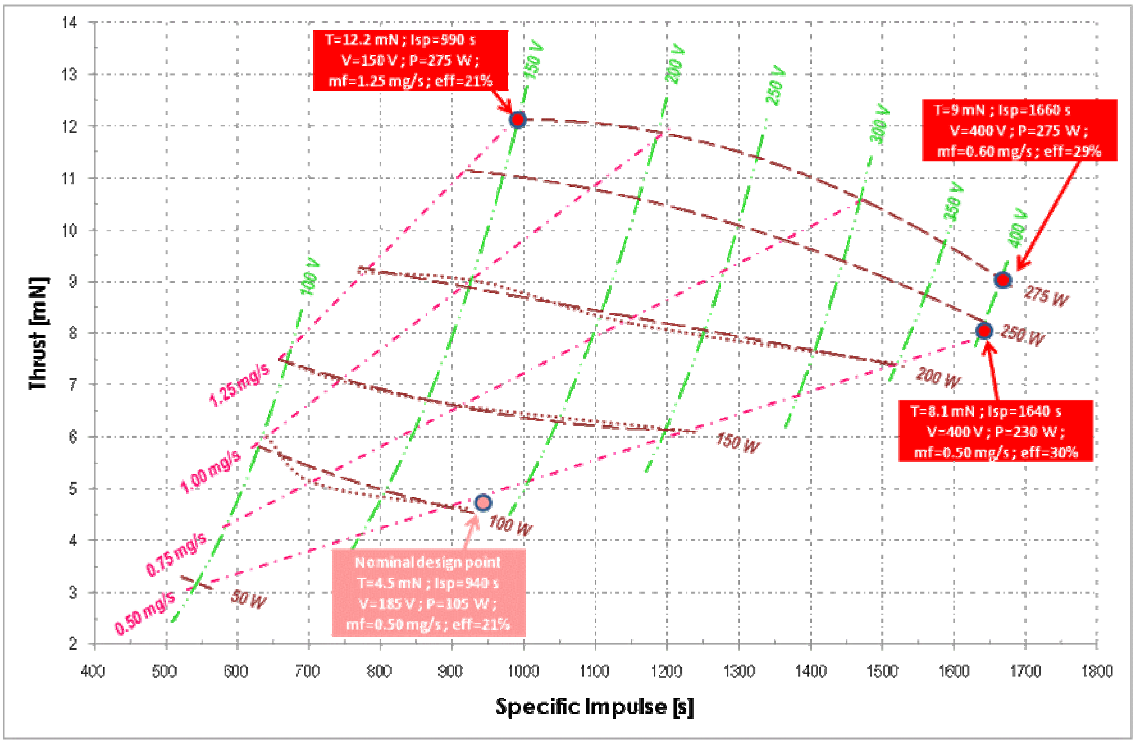

Figure 5. HT-100 experimental performance map from 2007 test campaign

(hollow cathode mode), with thrust level between 5 and 9 mN. Although no detailed optimization was performed, discharge voltage was arranged in order to maximize the measured thrust: the resulting specific impulse proved to be quite constant as well, being around 1500 s between 130 W and 210 W. Stable operation with thermionic cathode was maintained at power as low as 10 W. Over most of the thermionic operation range flat anode efficiency in the order of 18% was achieved; however this datum is affected by a large uncertainty due to the use of a thrust balance not tailored for such a low thrust level.

A second design iteration has been performed and a first EM has been manufactured in 2005, with the objective to improve reliability aspects and reduce the overall size of the unit. In parallel to the development activities on the thruster unit, a breadboard power supply and control unit (PSCU) has been manufactured. A first full subsystem test has been performed in 2006.

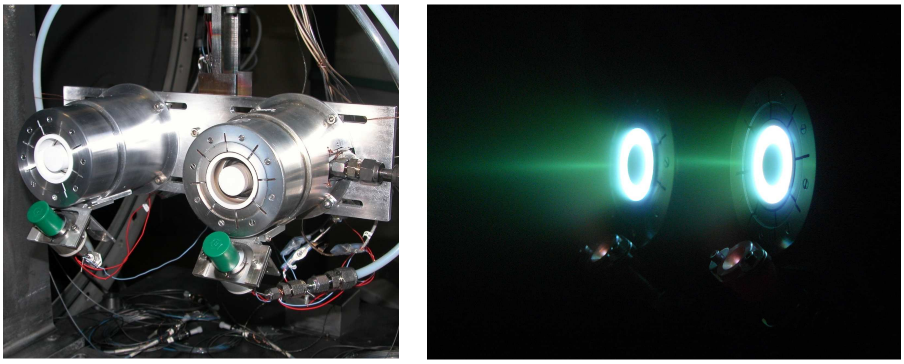

Figure 6. HT-100 demonstration model (DM3) and engineering model (EM1) in cluster configuration for preliminary test

Although the possibility of using thermionic or field emission cathodes is still investigated, the present thruster configuration includes a single cathode-neutralizer as baseline; cathode redundancy has been sacrificed to the need of minimising system mass, complexity and costs. The selected cathode is a flight proven unit manufactured by Alenia Spazio-Proel (NccA/1000 unit, originally developed for the RIT-10 ion engine and presently used on board of the ARTEMIS spacecraft). Figure 5 reports thruster performance mapping over a quite large operational range. So far the thruster has been operated for about 200 h, moreover a two-thruster cluster has undergone a first preliminary test (Fig. 6).

IV. System aspects

To get the maximum thrust range with a given channel size, it is necessary to act simultaneously on the anode flowrate and discharge voltage.

The xenon flow control device technology will be selected to allow a fast wide range variation (e. .g piezoelectric actuator, digital MEMS valves (discrete flow control)), eventually it could be combined with an electronic pressure reducer with variable pressure output.

Table 6. Thruster characteristics versus discharge voltage and discharge current

	Discharge voltage (V)	240	160	100
minimum thrust	Discharge current (A)	0,24	0,24	0,24
	Thrust (N)	3,89E-03	2,87E-03	1,89E-03
maximum thrust	Discharge current (A)	1,25	1,25	1,25
	Thrust (N)	2,03E-02	1,49E-02	9,84E-03

The hollow cathode will face large discharge current variations. The auto-cathode operation is therefore not always possible. At low discharge current, it will be helped by a keeper current (as in the case of gridded thrusters neutralisers).

In order to retain the basic simplicity of the PPU the voltage variation will be preferably stepwise (e. g. three output voltages: 240, 160 and 100 V.

For a drag compensation mission, the maximum thrust is taken as 20 mN (margin w. r. t. the WALES case). The Tab. 6 provides the thruster characteristics at three voltages and for the minimum and maximum flowrate: With these simple commands the thrust range covers one order of magnitude. Continuous variation down to zero thrust could be obtained by pulse width modulation. 50 ms pulse yields a MIB of 10^{-4} mNs.

This translates into the dynamic thrust range as displayed on Fig. 7.

Figure 8 shows the block diagram of the PPU (which would result further simplified in case of thruster with permanent magnets).

From the thrust level telecommand (digital), the microcontroller selects the discharge voltage, flowrate and coil current combination from a table to obtain the desired thrust. Some attitude control /relative positioning mission require "instant thrust". To obtain this, the hollow cathode could be permanently heated (either by heater or by keeper activation). The PPU could then receive a thrust order at any time and execute it within 50 ms.

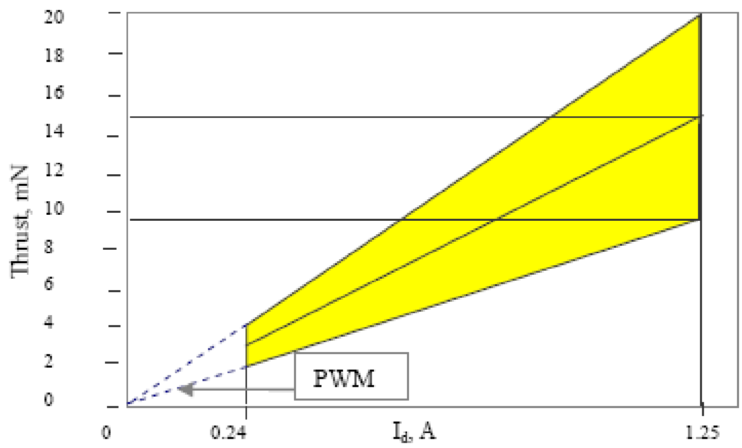

Figure 7. Small HET dynamic thrust range

The anticipated mass for a 20 mN class thruster / XFC assembly would be less than 800 g, while the PPU mass would be lower than 1500 g (300 W power).

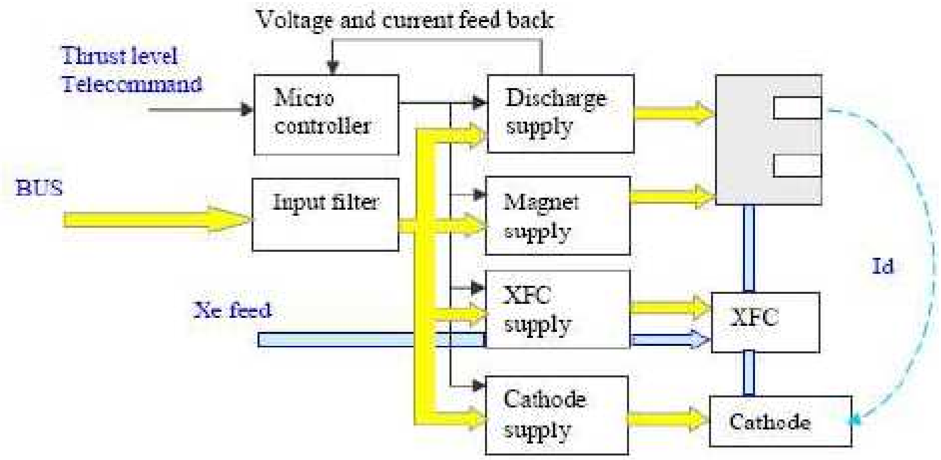

Figure 8. Small HET / PPU layout

For attitude control applications, it makes sense to cluster the thrusters as in the case of chemical propulsion. Several anode blocks can be combined with a single cathode and a single PPU thus leading to a simpler system offering a lower mass and lower cost than separate thrusters.

Figure 9 shows a possible layout of a fully redundant cluster providing three thrust vectors. Two clusters on satellite opposed faces could provide two DOF in translation and one in rotation (e. g. roll control, East West and North South corrections on a small GEO satellite).

The full redundancy is provided by three cold redundant anode blocks and a second hollow cathode. The XFC should be preferably integrated in the cluster in order to decrease the response time.

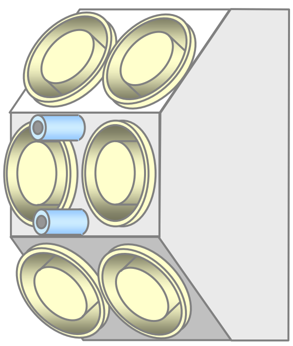

Figure 9. Layout of a fully redundant cluster

V. Comparison with other concepts

For both drag compensation and formation flying applications, the thrust requirement lies in the 5-15 mN range. Any particular requirement is not posed on the specific impulse, but it is intended to have the order of magnitude typical of the EP family, i.e. 1000s. The other important requirement is the total impulse, which depends on the mission duration: for the FF applications, the envisaged missions require a total impulse between 0.5 and 2 10^5 Ns, while for drag compensation it should range from 5 10^5 Ns to 2 10^6 Ns.

Looking at the European technologies with a development perspective by the next 10-15 years, the concepts whose typical thrust level belongs to the range of interest are mini-HET, mini-resistojet, high-thrust FEEP and gridded ion engine (GIE).

From the point of view of the multi-mission compatibility, the mini-resistojets appear penalized with respect to the other concepts, due to their low specific impulse (<100 s) which makes them inappropriate for drag compensation, while they still can been favorably considered for FF applications (preliminary studies on the propulsion assessment for Darwin have selected the cold thruster technology as the optimum system).

Similarly high-thrust FEEP could be unsuitable for drag compensation; Alta-Snecma study has demonstrated that technological feasibility limits the maximum achievable thrust for a single FEEP thruster to 5 mN. This value is really at the lower limit of the range of interest; as a consequence the FEEP thruster could remain a viable option only for a strict subclass of drag compensation applications, those where the chosen orbit and the satellite size (in terms of drag area) result in a perturbation force up to 5 mN.

Through a cross analysis, the propulsive requirements allow for the identification of those systems which are potentially able to satisfy them; in most cases however other non-propulsive requirements can have a strong weight in the selection of the optimum thruster. This is the case for the contamination requirement: the missions we are dealing with have in general payloads very sensitive to the contamination concern.

Xenon is not critical from the contamination side; then GIE and mini-Hall, using Xe as conventional propellant and also Xenon RJ, such as Alta XRJ, do not pose any problem. Other interesting propellants for resistojets are water, butane, nitrogen and other gases; neither of them can be considered a contaminant in space.

On the contrary, alkali metals as caesium can pose severe contamination concerns. Especially, in high-thrust applications, where the exhausted mass is relatively high, and/or with optical payloads, the contamination can be above any tolerable limit. The contamination risk is one of the most important raisons for which the FEEP system has been discarded for the Darwin mission.

Other fundamental aspects to be considered are the costs and the envisaged development risks associated to a particular technology.

Generally speaking, resistojets are very simple systems; the thruster comprises few parts, any of which does not seem particularly critical from the technological side. The overall EP system can be very simple too; most of the existing mini-RJs are designed for operating directly on the satellite power bus, without the need of any PCU. The associated costs are then very limited.

Small HET propulsion systems, such as that under development at Alta, can be simpler than higher power HET systems. When very high total impulse is required (1 to 2 MNs), a larger (existing) HET engine, operating at low power will provide the required impulse, PPU and XFC will remain the same since the thrust requirement (power and flowrate) are the same than for lower total impulse.

Ion engines are much more complex and then expensive devices than the previous technologies; at thruster level the most critical part is the grid system, which requires special manufacturing processes. At system level the PCU design is very critical, since it must provide different discharge parameters for the ionisation (EB or RF discharge) and the acceleration stages, other than for neutraliser operation (heating, stand-by mode), bombardment cathode operation and electromagnets in EB GIEs. The costs associated to further development activities required for adapting a T5-based system to a possible drag compensation application are expected to be not negligible. Moreover recurring costs for a complete system probably remain high.

By far FEEP technology has required development costs one order of magnitude higher than for the mini-HET. The recurring costs are also expected to be high, both at thruster and subsystem level. The thruster unit requires some manufacturing processes at the limit of the available technology.

Concerning the development risks, they are strictly related to the existing heritage for each of the potential technology.

EURECA & ARTEMIS experiences suggest a medium risk for GIE technology.

The mini-HET derives from the HET technology. HETs have a long flight experience especially in Russia; their use on western satellites is still quite limited, but is expected to increase in the next years on the wave of some successful missions, like the most recent SMART 1. Then the mini-HET technology can surely benefit from the enormous know-how on Hall thrusters. However it is a new development, then the associated risk can not be perceived as negligible, but, more realistically as medium.

Concerning the mini resistojets, starting from UoSat mission (1999) several RJ systems have flown in European missions. Although the need for the design scaling-down, due to its inherent simplicity, the perceived development risk is low.

No flight heritage exists for FEEP propulsion, neither as demonstration mission; although most of the scientific and technical know-how about low-thrust FEEP could be exploited for high-thrust concept, some major criticality, especially concerning the manufacturing process and the attainable total impulse still remain. Honestly the risk associated to this technology seems high.

In conclusion, with respect to other EP concepts potentially of interest for applications such as drag compensation and FF coarse maneuvers, the mini-HET seems preferable. Other than fully satisfying the thrust requirement for both types of application, the mini-HET does not pose any particular problem about the contamination aspect. The contained development/recurring costs and the acceptable risks, both aspects connected to the relatively low system complexity, definitively push the authors to select mini-HET as the optimum propulsive concept for those applications which are the focus of this paper.

VI. Conclusions and Perspective

The trade-off study showed that small HET can cover a wide range of missions in the 1–20 mN range involving large thrust variations. This includes drag compensation, relative positioning in LEO (requiring simultaneously drag compensation and formation flying) and fast slew of relatively positioned spacecraft (astronomy, interferometry). The large total impulse provided by small HET is especially interesting for Earth observation. A flying altitude of 350÷400 km could be easily maintained, thus allowing a drastic reduction of optical aperture or – even more important- synthetic aperture radar size and power.

The system analysis, backed by experimental results on larger thrusters, shows that it is possible to control the HET thrust within an order of magnitude and probably more (zero to maximum thrust) if PWM is used for very low values.

The next step would be to analyze in more depth the large thrust variation by test and after this first step to develop a small HET with long life.

It is proposed to explore the thrust variation possibility offered by a small HET acting on all parameters (discharge voltage, mass flow and coil current).

The wide thrust range measurement may pose problems: the lower thrust value is very low (mN range) so it may be necessary to use two different thrust balances, in order to get a sufficient precision at low thrust values with a dedicated one. Conversely, the thrust measurements could be combined with beam diagnostics (ion energy and density) enabling to refine the thrust measurements at low values.

Once the operating parameters are determined, the next step will be to define an Engineering Model providing a long life (by design) and to test it. Then the rest of the system (PPU XFC hollow cathode) could be developed in view of a potential application.

The small HET could be tailored to the application: when highly variable thrust is not required, the PPU and XFC could be simplified (cost saving). It could be used for attitude control with constant thrust but pulsed mode when control requirements are not too severe. If very high total impulse is required, a de-rated larger thruster will be used with same PPU.

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