RAM Electric Propulsion for Low Earth Orbit Operation: an ESA study.

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Abstract: This paper summarizes the results of the RAM-EP system concept study. The study involved the investigation of the feasibility of using electric propulsion together with gas collected from the atmosphere to provide thrust to counteract the S/C altitude decay caused by drag. This is in order to allow orbit altitude control with a defined thrust profile and within the typical budgets of an Earth Observation type of mission. The final objective was to enable low altitude missions (below at least 250 km) and / or long lifetime missions above 250 km. Moreover the study aimed to apply the concept to a reference technology demonstration mission that could be of interest for Earth Observation.

I. Introduction

Low Earth orbit satellites are subject to atmospheric drag and thus their lifetimes are limited with current propulsion technologies by the amount of propellant they can carry to compensate for it. High specific impulse electric propulsion has been recognised during the past years as a key technology for low orbiting EO missions, with the capability of compensating drag during low altitude operation for an extended period of time (GOCE mission). However even high specific impulse Electric Propulsion thrusters have limitations in their capability, and the present

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state of the art would in most circumstances not allow for more than 2 years of drag-compensating operation at an operative orbit altitude below 250km. The lifetime of the SC would in theory not be limited by the amount of fuel if the same atmospheric constituents producing the drag were collected by an appropriate inlet and used as propellant by the propulsion system. This is the idea behind the RAM-Electric Propulsion system concept.

To investigate the theoretical feasibility of the RAM-EP concept the ESA Concurrent Design Facility (CDF), financed by the ESA General Studies Program, was requested to analyse this innovative electric propulsion drag compensation system, to assess its operational range and to provide a concept outline for a technology demonstration mission. An interdisciplinary team of ESA specialists took part on the study with support provided by Earth Observation specialists and electric propulsion specialists (from industry and university). As main result, the study highlighted that the RAM-EP concept could provide a promising innovative solution to very low altitude (below 250km) and/or long lifetime LEO missions (not over a defined altitude). The study also provided an overview on required technology development activities for such application.

II. RAM-EP Concept

The RAM-EP concept consists of a collection system (Collector) for capturing the rarefied flow and an Electric Propulsion Thruster for generating the required thrust. The following feasibility issues have been investigated during the CDF study:

- Capability to collect the mass flow rate needed to generate the required thrust with an inlet aperture of size compatible with the dimensions of a reference S/C
- Capability to collect the particles flow with properties as required by the thruster
- Choice of an Electric Propulsion thruster that guarantees the required thrust to perform altitude compensation
- Capability of the thruster to work with the encountered flow mixture composition or with selected species
- Capability of the thruster to work with variable inlet conditions caused by different altitudes, solar activities, local time
- Capability of the thruster to operate within the power and mass constraints of a typical Earth observation mission.

A number of trade-offs have been considered during the study. For the collection system it has been investigated the possibility of using an electromagnetic scoop, an aerodynamic intake or a hybrid device. For the thruster the use of a Gridded Ion Engine (GIE) or a Hall Effect Thruster (HET) with modified design has been assessed. Moreover, in order to verify the capability to perform altitude compensation, a thrust strategy has been identified. In principle, to maintain a given altitude there are two approaches: continuous thrust along the orbit to compensate exactly the drag; arcs without thrusting with consequent loss of altitude followed by thrust arcs to recover altitude above the nominal value. In the latter case, the thrust needs to be higher than the drag and is a function of the ratio between thrusting time and orbit period. The altitude decay associated to a non thrusting period increases dramatically for low altitudes. The decay in km can be extrapolated from Fig.1 by multiplying the decay by the S/C mass (kg), the wetted area (m^2) and the drag coefficient.

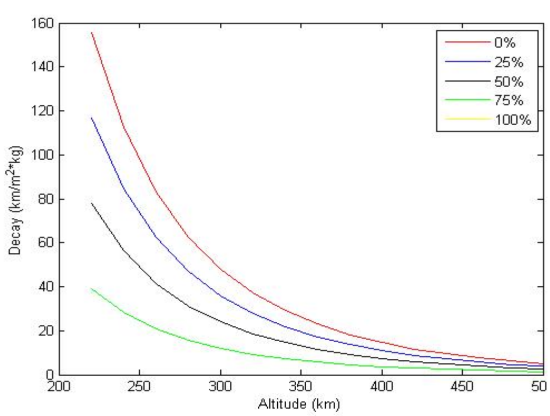

Fig. 1 Decay per orbit for different altitudes depending upon percentage of the orbit where the drag is compensated by an equal thrust.

A. Atmospheric Model and Thrust Strategy Selection

The first step carried out during the study was to compute a density figure of the upper atmosphere for all solar activities. The composition has been identified as the one presented in Fig.2 based on [1]. The density has then been used to calculate the drag together with the geometrical and aerodynamic characteristics of a reference spacecraft. It has been assumed a reference wetted area of $1m^2$ and a drag coefficient of 2. The variation of density with altitude and solar activity was computed (Fig.3).

Concerning the thrust strategy, two reference ratios of thrusting time over orbital period were considered, the 5/6 and the 2/3. The 2/3 ratio represents a Sun-Synchronous Orbit (SSO) with a local time of around 10:30 am, whereas the 5/6 ratio represents a dawn-dusk SSO with no thrust during small eclipses. The thrust levels required for these different strategies are shown in Fig.4. Given the large increase in thrust necessary for a non-continuous strategy a continuous thrusting strategy would be preferable. However, this requires the spacecraft to thrust during all eclipses, putting a large strain on the power subsystem. The optimum was found to be thrusting for a fraction of the time of each eclipse, resulting in a strategy closer to that of the 5/6 ratio.

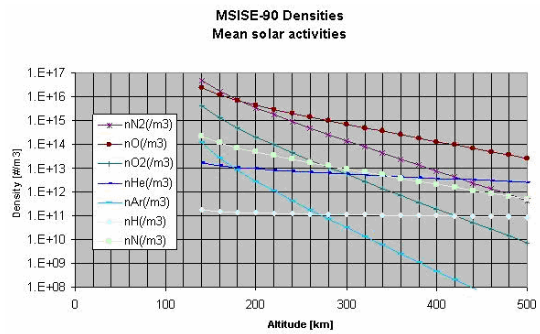

Fig.2 Mean atmosphere composition

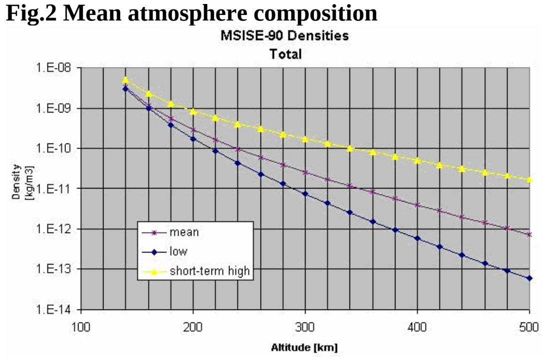

Fig. 3 Atmosphere density – variation with solar activity (total)

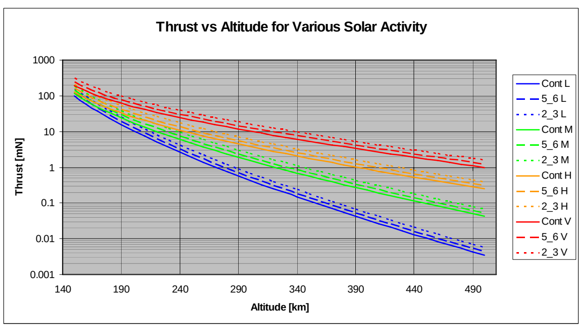

Fig.4 Thrust for different strategies, solar activities, and altitudes

B. Thruster Selection

To meet the goal of defining the boundaries of the RAM-EP concept for low Earth observations mission the following constraints were applied to the propulsion system: thrust range between 2 and 20mN and a maximum available power in the order of 1000W. The main design driver selected was to use an existing thruster adapted to work with the constituents of the upper atmosphere such as atomic oxygen and nitrogen. Two classes of existing Electric Propulsion thrusters were considered, namely HET and GIE. Theoretical models were created to simulate the effect on the thruster of using the selected species as propellant and to compute predicted de-rated performances. The result of the analysis for both HET and GIE is a performance chart, which can be seen in Fig.5 (the chart refers to the mean solar activity case). It is worth noticing that in case long lifetime is needed, due to the 11 year duration of the solar cycle, a period of high solar activity is certainly encountered. If short missions are sought but if low altitude is the requirement, then, the analysis can be based on the low solar activity assuming the mission using RAM-EP is opportunely phased with respect to the solar cycle. The diagonal block line represents the thrust required at a given altitude for the selected quasi-continuous thrusting. The blue 'o', the red '+' and the green 'x' horizontal lines represent the top of the thrust envelope for the thrusters considered. The thrust required and the thruster envelopes are plotted against the logarithmic scale on the primary Y axis in mN. The thruster power requirements for the given thrust ranges have been scaled considering O and N_2 as the propellants and 5% loss for the PCDU. To compute the power demand thruster's polynomial models were provided by the manufacturers. They are represented by the solid green, blue and red curves in the lower left portion of the graph. These curves are plotted against the second Y axis in Watts. A straight line corresponding to the reference power level of 1000W is shown. The reference power level was chosen considering the solar-arrays sizing and the launcher's fairing volume.

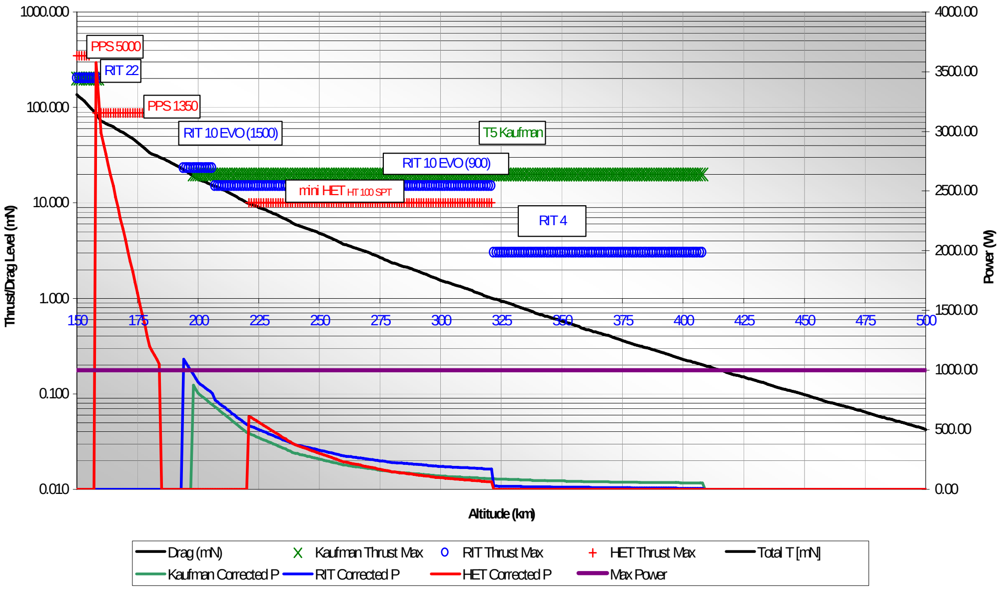

Fig.5 Performance chart (mean solar activity)

From the performance chart of Fig.5 it can also be noted that the mini-HET and the RIT10 comply with the thrust requirements and the availability of power. For T5 the same model applied to the RIT10 was assumed. Amongst them, and based on operational parameters (e.g. required thruster's discharge chamber pressure with respect to collector capabilities) the GIE RIT10 thruster was selected. The study highlighted also that the RAM-EP concept is only interesting for the combinations of altitude-lifetime that would imply high propellant mass penalties in the case of use of conventional EP thrusters. Fig.6 shows the propellant mass as a function of lifetime for various 'conventional' Electric Propulsion thrusters and at different altitudes. On the basis of the propellant mass required by the conventional EP thrusters, it can be noted that above about 250km the RAM-EP concept is not a competitive concept anymore.

C. Minimum Altitude Definition

The selection of the most suitable thruster is a function of the minimum altitude the RAM-EP can be used for. This depends on the power available onboard and the duration of the non-thrusting phase as function of the eclipse time. In case of low solar activity an altitude as low as 180 km seems achievable, but for mean solar activity the minimum altitudes is around 200 km.

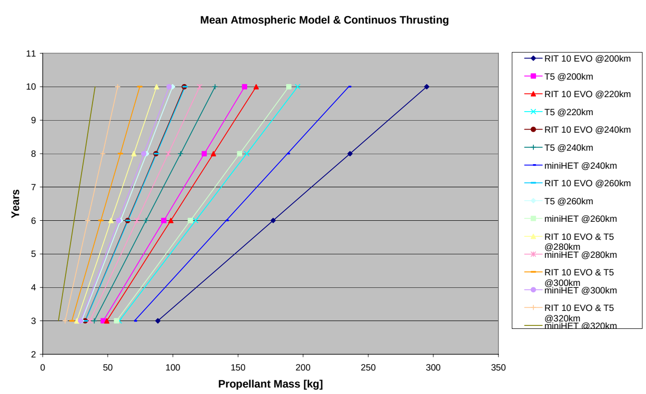

Fig.6 Competitiveness of RAM-EP vs. standard EP

D. Orbit Definition

The power available onboard was found to be the main limiting factor for the RAM-EP concept: decreasing the altitude increases the thrust requirements, leading to increasing power demand. Based on this consideration, an analysis was carried out to select the operational orbit and the spacecraft configuration in order to maximize the available power. The orbit was selected on the base of guaranteed constant power availability throughout the mission duration (typically several years) and to minimize the eclipse durations. The applicability of the concept to Earth observation missions was also considered. As result, a Sun-Synchronous Dawn-Dusk was selected as the most optimised orbit.

E. Collector Design

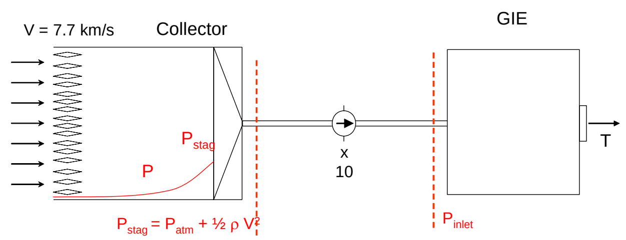

Fig.7 Schematic diagram showing the collector concept

The collector system is positioned in the RAM direction of the S/C and is meant to provide the requested Mass Flow Rate (MFR) directly to the thruster's inlet at a certain pressure. Various concepts of collection systems were considered: an electromagnetic scoop, an aerodynamic intake or a hybrid device. A collection system based on electromagnetic effect would require a sufficient amount of ions in orbit. As most of the atmosphere is neutral at the altitudes of interest no sufficient MFR can be provided by this concept. For the same reason also the hybrid device was discarded. Therefore the aerodynamic intake was selected as collecting concept. According with the calculation performed, an area of 0.15m^2 was judged as sufficient to collect the required MFR for all the altitude ranges considered in the study. Nevertheless, a higher surface area of 0.6m^2 was allocated to the collector intake, to take into account the grid system used to stop the particles at the entrance of the collector, and an extra design margin. The selected area was consistent with the assumption of 1m^2 front area taken for the calculation of the drag. The collector length is fixed in order to get the required steady level of pressure at the end of the collection chamber. Various simulations performed with the DSMC-software SMILE were executed varying the collector length from 0.2 m to 1.0 m. The stagnation pressure at the back of the collector, computed for 3 cases, is shown in Fig.8 and in Fig.9. As showed, the maximum is reached for a length of 1.0m. In this case the collector can provide at the inlet of the thruster a pressure of 10^{-3}Pa. Further simulations have been performed with a collector 1.3 m long and with varying collector configuration at the end: concave, straight and divergent. The first conclusion is that from a certain point the length has no positive effect on the pressure. Changing the collector configuration at the end of the aerodynamic intake with a concave or divergent flow seems not to result in an improvement either.

During the study it was noted that from the spacecraft operations point of view, a tank, located between the collector and the thruster, would help to cope with fluctuations in atmosphere and non-nominal situations. The tank would be filled when excess mass flow is available and emptied when mass flow is scarcer. However, to store a quantity of air significant for the purposes above, a very high pressure would be required in the tank, non-compatible with the current spacecraft configuration. A spherical tank volume would be limited by the spacecraft cross section. Moreover the tank could be filled only during the 11minutes of non-thrusting periods of the long eclipses. To store all the particles collected in such period of time it can be easily estimated from the equation of the perfect gas that the required pressure in the tank would be prohibitively high. The proposed suggestion was to accept the variation in the atmosphere conditions, and verify that the engine could cope with it.

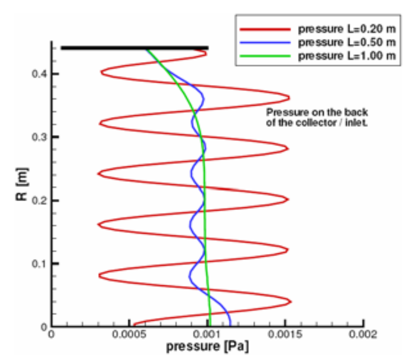

Fig.8 Stagnation pressure levels as a function of the collector length

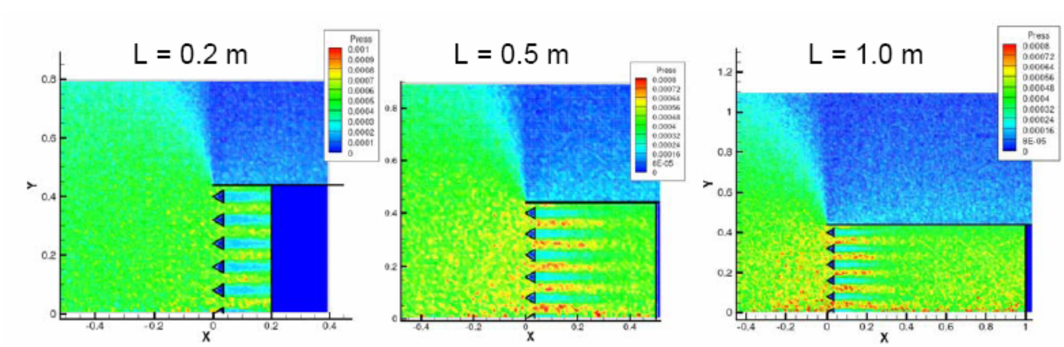

Fig.9 Pressure in the collector

III. RAM-EP Technology Demonstration Mission

In addition to the assessment on the feasibility of the concept, the RAM-EP study aimed to apply the concept to a reference mission that could demonstrate the RAM-EP technology and be at the same time used for Earth Observation purposes. be of interest for Earth observation. This mission was defined in order to show at system level the potentiality and the limitation of the RAM-EP technology.

The objectives of the reference technology demonstration mission for RAM-EP were identified as the following:

- To fly a S/C at orbital ranges consistent with range of performance identified for the RAM-EP concept
- To have a duration sufficient to prove the lifetime increase benefits from the RAM-EP concept
- To carry a technology demonstration propulsion subsystem based on the RAM-EP technology
- To rely only on the RAM-EP propulsion subsystem during the nominal mode of operation when altitude compensation is performed
- To allow the monitoring of a representative set of health and functional parameters of the RAM-EP propulsion sub-system.

Having fixed a target mass of 1000kg for the demonstration S/C, VEGA was selected as reference launcher on the basis of the predicted performance into SSO (200X200km).

In order to prove the applicability of the RAM-EP concept for Earth Observation purposes, the demonstration included the operation of some EO payload at the same time as the RAM-EP thruster. Being the RAM-EP technology demonstration mission not driven by science objectives or by accommodation of Earth Observation payload, the selection of EO instruments on board the SC was done to avoid introducing unnecessary complex requirements which could endanger the feasibility demonstration of the RAM-EP technology. Other criteria taken into consideration for the selection of the most suitable instruments were the altitude range and mission profile, the power consumption, mass, volume, the support to the RAM-EP demonstration, the future EO applications and the low resource requirement on the SC. Instrument selection were, therefore, focused in two specific areas: instruments that benefit from very low altitude operation; instruments used for direct sensing of atmospheric properties. LIDARS were identified as one of the type of EO payloads that most benefit by low altitude operation. Lower mass, simplicity, and less power consumption are the most remarkable improvements that can be achieved when decreasing the operational altitude while maintaining the same performance.

RAM-EP concept could also provide the means for direct sensing of the lower layers of the atmosphere. Properties such as density or chemical composition could be measured at low altitude using appropriate instruments. The electromagnetic or the gravity field of the Earth could also be measured at low altitudes but payloads would impose many constraints of the mission and compromise the RAM-EP demonstration. Consequently, the following instrument package was proposed to be implemented on board the demonstration mission: a LIDAR, an Accelerometer, a Mass Spectrometer and a Radio Occultation package. Additionally, in order to monitor the performance of the RAM-EP electric propulsion subsystem, it was decided to carry on board an Electric Propulsion Diagnostic Package as additional payload.

F. Mission Profile

The technology demonstration mission was divided into three phases:

a) Phase 1, constant altitude 200 km for 3 years. The launch is phased so that at the beginning of Phase 1 the solar activity is at its minimum. During this Phase the S/C stays in a 6:00am Sun-Synchronous Orbit (SSO) at constant altitude of 200km with an inclination of 96.33^{o}. The drag to be compensated is in the order of 10.6mN; however at this altitude there would also be short eclipses of up to 16 minutes and long eclipses of up to 29 minutes depending on the time of year. It was agreed that the design of the power subsystem shall be capable of providing enough power for thrusting for up to 17 minutes of eclipse time. This is equivalent to a required delta-thrust of 1.7mN. During this Phase the LIDAR payload is active.

b) Phase 2, constant climb from 200 km to 250km in 3 years. This Phase starts while the solar cycle is at an average value and progress up to and through solar maximum. During this Phase the S/C raises its altitude from 200km to 250km over 3 years at a constant rate of climb resulting in a small Delta Thrust required. The required thrust is ranging between 17.8mN and 5mN. The eclipse strategy applied is the same as that for Phase 1. Measurements will be performed by the environmental instruments to sample the atmosphere at different altitudes.

c) Phase 3, descent from 250km down for at least 1 year. This Phase begins just after solar maximum. During this Phase the S/C decreases its altitude from 250 km to 200km over at least a year. To maintain this constant rate of decay, the thrust required is 3.6mN at the beginning of Phase 3 and up to 16.6mN at the end of it. The eclipse strategy applied is the same as that for Phase 1 and Phase 2.

At the end of the mission, the S/C will be at 200km. The available thrust is too low to provide a change in the eccentricity of the orbit sufficient to change the flight path angle and having a controlled trajectory for disposal. However, due to the low mass of the spacecraft, the natural decay will inevitably cause the S/C to perform an atmospheric entry which is predicted to completely burn up the satellite during re-entry.
A summary of the subsystem design is presented in Table1.

Subsystem Design	Operation Duration	At least 7 Years
	Attitude Control	Three-axis stabilized, 6 DOF, with 4 reaction wheels and 3 magneto-torquers. 0.5 kg back-up propellant (Xe) for contingency
	Power	GAGET 2-ID/160-8040 Solar Cells with 100um cover glass. Total 19.7417.1 m^2 (Consisting of 4 deployed wings of 2 panels each, and three body mounted panels on sun-pointing faces of the spacecraft)
		3297 W BOL
		2900 W EOL
		28V unregulated bus
		Li-Ion Battery, 612 Wh
	Data Handling	1 CDMU Computer, 1 solid-state memory unit, 2 TM/TC Modules & 7 Micro RTUs
	Propulsion	4 ASTRIUM RIT-10 GIE, 4 RFG, 2 PSCU

Table1. RAM-EP Technology Demonstrator Mission Summary of Subsystem Design

IV. Conclusion

The RAM-EP feasibility study has shown that the RAM-EP concept is potentially interesting for very low altitude and/or long lifetime missions in the range 200 to 250 km and for lifetimes of 3 to 8 years.

Nevertheless, the results of the study are based on prediction of the thrusters' performances with the given gas mixture using available theoretical models. An experimental characterisation of the thruster with O, O_2, mixture of O and N_2 and Xe is recommended to confirm performance predictions. An assessment on the operation of the thruster at different discharge pressure levels and of the neutraliser is required. Moreover a dedicated study on design of the inlet shall be performed. All these activities shall be part of a technology pre-development plan prior to further system studies on the RAM-EP concept.

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