Radio Frequency Atmosphere Breathing Ion Engine development

IEPC-2024-499

Presented at the 38th International Electric Propulsion Conference
Pierre Baudis Convention Center • Toulouse, France
June 23-28, 2024

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Abstract: TransMIT GmbH is working on the Atmosphere-breathing propulsion concept for over 15 years. The number of theoretical and experimental campaigns that has been performed in this time cover various thruster sizes and power ranges, compatible neutraliser designs, operation at Earth and Mars low orbit environmental gas compositions, investigations on relevance of the Atmosphere-breathing propulsion on-ground testing as well as trials of the orbital particle flow reproduction, finally system level global relationships to envelope design requirements and challenges were always amongst the topics of interest.

This paper is the summary of company experience and lessons learnt together with current understating on challenges ABEP development introduces, requirements it sets and best at authors opinion ways to approach ABEP thruster technology selection and design.

Nomenclature

ABEP = Atmosphere Breathing Electric Propulsion
AETHER = Air-breathing Electric THrustER
LEO = Low Earth Orbit
MABHET = Martian atmosphere breathing Hall effect thruster
MAVEN = Mars Atmosphere and Volatile EvolutioN
RAM-EP = Air-breathing Electric Propulsion
S/C = Space Craft
VLEO = Very Low Earth Orbit

I.Introduction

TRANSMIT initial contribution to the RAM-EP topic dates 15 years back and includes participations in the very first European activity on that topic - initiated by ESA contract „EPN Preliminary

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Characterisation Test Campaign Of Ep Technology With Non-Conventional Propellants“ (Prime: ALTA S.p.A, contract 22946/09/NL/CO, 2010-2013) where amongst other theoretical and experimental study with RIT-10 thruster was performed.[1] The RIT-10 was tested in Jumbo facility of the University of Giessen with O2, N2, O2/N2 and O2/N2/Xe mixtures at different power levels. Also initial 10-hours-long test of the thruster showed no signs of erosion operating with N2, whilst noticeable O2 erosion was identified in the acceleration grid. However, in a subsequent study, it was demonstrated that with the appropriate choice of materials, the lifetime can still be in the tens- to hundred of thousand hours range. After using different grid materials, the 500 h endurance test and the complementary calculations demonstrated for the project target thrust profile a lifetime of at least 60 000 hours that would be higher than that for Xenon for the same profile; this proves the usual claim of alternative concepts developers on lifetime limiting factors for classic technologies to be just a misconception.

Parallel internal activities were dealing with testing of the RIT-10-EBB also with CO2 to obtain initial data in order to evaluate possibility to use the Radio Frequency Ion Engine based Atmosphere-Breathing Electric Propulsion system for drag compensation in the Martian Environment. For the maximum reached extracted currents around 30% higher RF power for CO2 than for air mixture was required. The system level considerations allowed to conclude that Atmosphere-Breathing Propulsion could potentially enable drag compensation in the range of altitudes between 140 and 190 km. It was also concluded that alternative to solar power sources such as nuclear reactor could make the technology much more appealing for the implementation.

Further internal activities were related to the development of an air-fed and Oxygen-fed Radio Frequency Neutraliser and small size Radio-Frequency Ion Engine operating with air composites. Both small size systems were extensively characterised and the results fitted within TransMIT internal models.

The recent project „Air-breathing Electric THrustER” - AETHER (Coordinated by SITAEL, Horizon 2020 Research and Innovation Programme under grant agreement No. 870436) allowed TransMIT to investigate performances of a larger system.[2] 22 cm Radio-Frequency Gridded Ion Engine was characterised within the 0,12 to 0,58 mg/s mass flow rates at 4 different proportions between N2 and O2 which corresponded to the altitude range from ~190 km to ~250 km on various beam current level.

The drag compensation conditions for the project target could be achieved in the full investigated range from 185 to 250 km in case of no dissociation in discharge assumed and from around 220 km with full dissociation assumption. Considering the assessed dissociation ranges, compensation between 195 and 250 km should have been possible with the available power for the project target mission scenario.

Particularly interesting results were achieved in investigating environmental conditions in the vacuum chamber at different operating points that allowed to sense variation in system operation while using different air mixtures and power levels.

Currently, TransMIT is working on execution of the “Mars Atmosphere-Breathing Electric Propulsion Thruster” (ESA contract 4000144034/24/NL/RK/cb). The activity aims at design, development and characterization of the Radio Frequency Ion Engine capable of drag compensation in the low earth orbit of Mars. The design phase is almost concluded by now given all the preparatory work done in close collaboration with Bundeswehr University of Munich (UniBw) on the state-of-the-art review of both Mars environment and propulsion systems that were ever trialed against that environment. After Design Review, thruster will be manufactured, integrated and tested in UniBw M laboratories in the same way as AETHER experimental campaign has already been conducted there. Since the last year closer collaboration approach was taken between TransMIT and UniBw M scientists to cover full range of interdisciplinary scientific disciplines needed for the ABEP system study and development.

Further some of authors common believes on various aspects of RAM-EP development will be presented along with selected results of our previous works on this topic.

II.Orbital environment

Understanding of the conditions under which propulsion system is going to be functioning are especially critical for atmosphere breathing propulsion where the amount and the type of propellant available are directly dependent on the characteristic of operating environment. Therefore, as first, a vast and detailed review of the target atmosphere shall be carried out with the aim to assess the variation of the gas properties in the range of altitudes where ABEP is planned to be operated.[3]

A. Earth atmosphere: general aspects

The Earth atmosphere contains nitrogen oxygen (molecular and atomic), argon, carbon dioxide and some further gases including water vapor. It is dominated by nitrogen up until around 185 km where atomic oxygen starts to prevail and then dominate. Air composition, temperature, and atmospheric pressure vary with altitude. The atmospheric condition can vary significantly due to external forces such as solar activity which can heat the atmosphere and ionize atoms and molecules and lead to particles diffusion. The Atmosphere Breathing Propulsion System operation shall take these effects into account and needs to be compatible with them.

The practical information on particle densities and temperature usually utilised from NRLMSISE-00 model.

B. Martian atmosphere: general aspects

The Mars atmosphere is far less familiar than Earth atmosphere. It is a thin CO_2-dominated atmosphere comprising only a few mbar of pressure at low altitudes towards the Mars surface. It is less dense than the Earth's atmosphere, nevertheless it shares multiple similarities: atmosphere is hold by gravity; has complex chemical effects as well as fluid and thermodynamics; it is not uniform and changes with time and place which translates into Mars weather. The Mars atmosphere can be divided into lower, middle (or Mesosphere), upper (or Thermosphere) atmosphere and atmospheric escape part (Exosphere). With more observations done, scientists conclude nowadays that the Martian atmosphere is an interconnected complex system, where the processes near the Mars surface and in lower Atmosphere can also have an impact on the upper Atmosphere. Therefore, global dust storms need to be revised with respect to potential impact on target altitudes and as conclusion to the Electric Propulsion System operation.

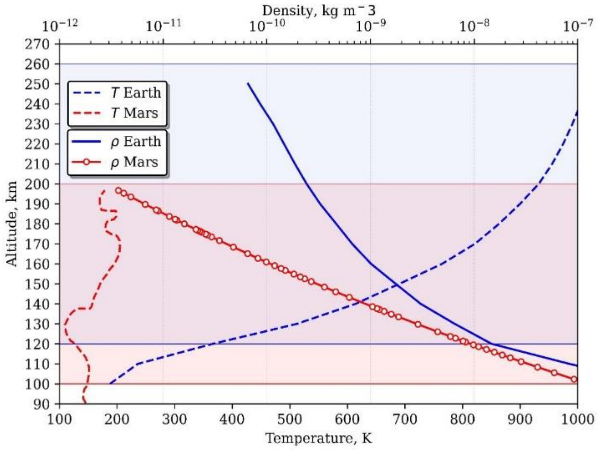


Figure 1. Temperature and density variation with the altitude of Earth and Mars orbits. [3]

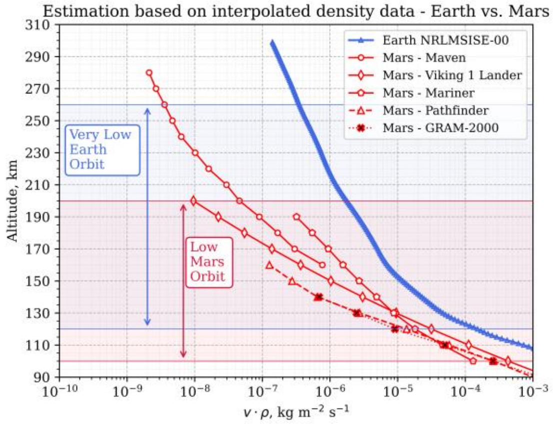
Our first assessment of the the range for Mars Atmosphere Breathing Propulsion Systems targets altitudes between 100 and 150 km mostly belonging to Thermosphere with the slight coverage of upper Mesosphere too. The atmospheric condition there can vary significantly due to external forces such as solar extreme ultraviolet radiation, solar wind, and high-energy particles from the sun, which can heat the atmosphere and ionize atoms and molecules. These effects also need to be assessed for the compatibility with Atmosphere Breathing Propulsion System operation.

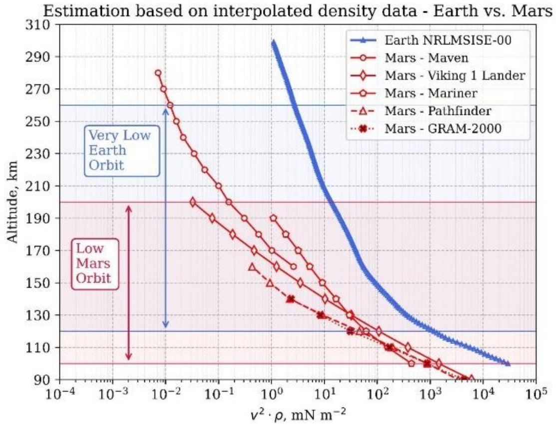
The practical information on particle densities is coming from well-established models and probe measurements. Most renown is Mars-GRAM 2000 model of NASA (and its more recent updates) that is based on numerous measurements data of Viking 1 and Mariner missions of 70s last century and its recent update – Mars-GRAM 2010 – to add user-controlled dust case and better match Mars Global Surveyor,1 Mars Odyssey, and Mars Reconnaissance Orbiter data. Number of more recent data have been complimenting our survey, such as data measured by Pathfinder, and most of all Mars Atmosphere and Volatile Evolution mission (MAVEN). There has been further collected data by Mars Exploration Rovers (Opportunity and Spirit), Mars Express, ExoMars Trace Gas Orbiter and others that could be used for future expansion of the input data to the understanding of the Mars Atmosphere Breathing Propulsion System potential operating environment.

The details on the methodology laying behind the data collection, handling and preparation to set environment boundary conditions can be found in Ref. 3. While figure 1 shows temperature and density, the following figures depict composition variation with the altitude of Earth and Mars orbits.

C. Earth and Mars Atmosphere implications to RAM-EP

The range of altitude typically assessed for Earth ABEP applications is laying between 160 and 250 km while for the Mars 100 to 200 km range is considered in different studies. In this ranges already looking at general mass density plots one can observe around an order in available density at Earth with even much more rapidly density variation at Mars leading to over 4 times order of magnitude in density over there. However, typical electric propulsion systems are not designed to cope with very broad ranges of the mass flows as it is practically impossible to make them operate efficiently through that kind of input parameters variation.





It is fairly easy to transfer from mass density to available for consumption mass flow by multiplying the density plots by orbital velocity or to the function of drag that is going to be generated per satellite frontal area by multiplying the density plot by square of the orbital velocity. In case of the mass flow, the plot will represent maximum available flow, while in case of drag – it will be minimum expected drag with the satellite drag coefficient of 2. Of course, one can correct those further introducing various corrections to those initial approximations and taking into account variation of the orbital conditions, but principally those plots will represent the best case scenario for the ABEP system and that's why, it is suggested to initially trial potential for the drag compensation against these plots.

There are few observations one can make:

- For the same available mass flow almost twice of the drag would be generated at Earth considering systems with similar drag

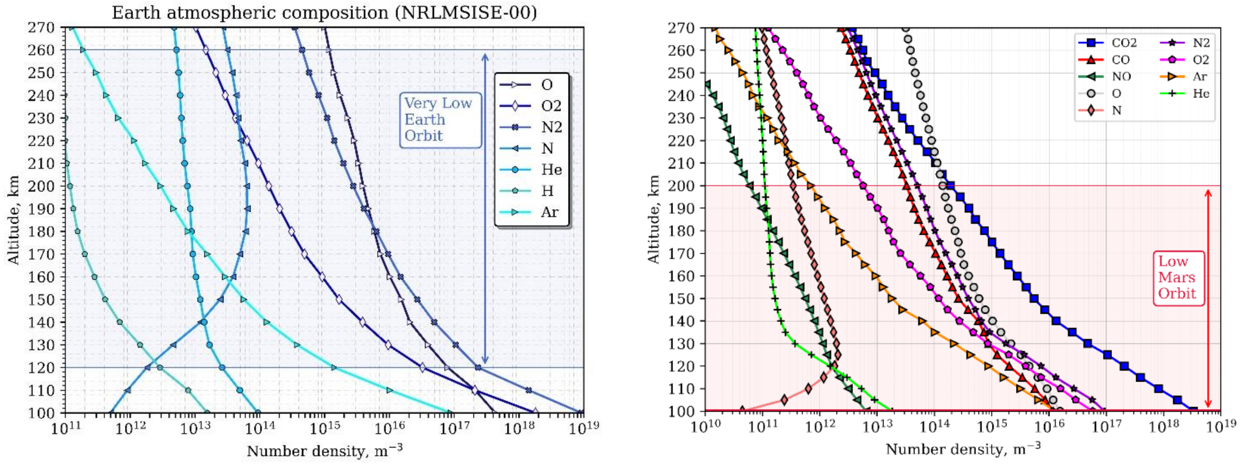


Figure 2. Composition variation with the altitude of Earth (middle) and Mars (right). [3]

coefficients. Of course, given almost 2,5 times lower power density at Mars, there is still a gap in ability to generate proportional thrust by the systems with similar drag coefficients.

- Mars atmosphere probably presents larger challenge despite overall lower drag due to the slope of the flow and drag curves. That means that efficient ABEP design should target much more narrow altitude gap which let small to no very much needed flexibility to the system.
- As it can be seen, 160 to 250 km on Earth are equivalent to around 135 to 185 km on Mars in terms of available flow and around 125 to 160 km in terms of drag expectations. Additionally, for Marsit should be highlighted that it seems that the dust storms influence is major up to at least around 130 km which needs to be accounted for. The variations in the space weather are dramatic over there.
- Another important observation is that mass flow (mass that can be collected per surface unit per time unit) is not unlimited, in contrary to what some of the RAM-EP developers prefer to believe, thus exhaust velocity becomes of highest priority although power budget still matters. In other words, one shouldn't forget that every particle that can be collected to generate thrust does generate drag that should be compensated.

III.Exhaust velocity

The level of minimum required exhaust velocity in first approximation can actually be expressed independently of total mass flow rate as function of collection efficiency and drag coefficient of the satellite. In this case part of the drag corresponding to the lateral structures of the satellite might be not entirely included, thus making it still best case scenario or bare minimum of required exhaust velocity. However, it should be highlighted that interpretation of this exhaust velocity should vary a bit for different propulsion technologies as it will be influenced through different handling of the dissociation impact in the process.

Drag imposed on the satellite can be written as follows:

$$\frac{1}{2}\dot{m}\vartheta_{orbital}C_d = D, \quad \text{Eq. 1}$$

where, $\dot{m}$ – total mass flow rate available on the thruster intake. Assuming energetic particles have much larger exhaust velocity than neutrals going through the system, the thrust of the satellite can be expressed as:

$$\dot{m}_i\vartheta_i = T, \qquad \text{Eq. 2}$$

where ϑ_i – energetic particles exhaust velocity and $\dot{m}_i$ – total mass flow rate generating the thrust. Ion mass flow rate as always is given by multiplication of total thruster mass flow rate ($\dot{m}_{thr}$) by mass utilisation efficiency of the thruster (η_m). And in its turn mass flow rate of the thruster can be expressed as total mass flow rate available on the thruster intake multiplied by the intake collection efficiency (C_{eff}).

$$\dot{m}_i = \dot{m}_{thr}\eta_m \qquad \text{Eq. 3}$$

$$\dot{m}_{thr} = \dot{m}C_{eff} \qquad \text{Eq. 4}$$

Combining all the above equation together, the following equation for minimum required exhaust velocity can be put together:

$$\vartheta_i \geq \frac{\dot{m}\vartheta_{orbital}C_d}{2\dot{m}_i} = \frac{\dot{m}\vartheta_{orbital}C_d}{2\dot{m}_{thr}\eta_m} = \frac{\dot{m}\vartheta_{orbital}C_d}{2\dot{m}C_{eff}\eta_m} = \frac{\vartheta_{orbital}C_d}{2C_{eff}\eta_m} \qquad \text{Eq.5}$$

One can also estimate what exhaust velocity could be reached by this or other Electric Propulsion technology given particular propellant – nitrogen-oxygen mixture in case of Earth or carbon dioxide in case of Mars. The calculations are pretty straightforward when it comes to Electrostatic Thrusters as particles exhaust velocity could be expressed as:

$$\vartheta_i = \sqrt{\frac{2qU}{m_i}}$$

- where U is accelerating potential difference and q is ion charge. Electrothermal Thrusters generate kinetic energy of the particles through the heating and exhaust velocity can be written as:

$$\vartheta_{exh} = \sqrt{\frac{3kT}{m}}$$

- where k is Boltzman constant and T effective temperature that is limited for Electrothermal propulsion by thermomechanical properties of thruster materials exposed. Assuming, best of technology performances achieved already represent maximum reachable temperature given up-to-date materials properties, conversion factor for plume exhaust velocity of nowadays used propellants into another propellant will be represented by the following:

$$\vartheta_{exh\ atm} = \vartheta_{exh\ state-of-the-art}\sqrt{\frac{m_{state-of-the-art}}{m_{atm}}}$$

For the purpose of this paper couple of technology top performing hydrazine and hydrogen arcjets (as best in technology class) were used to recalculate those into performances of nitrogen-oxygen mixture.

Electromagnetic systems (or rather EP technologies often attributed to that type) are probably most complex when it comes to identification of exhaust velocity since in practice they might use all three acceleration mechanisms with some times thermal or electrostatic mechanisms even prevailing E-cross-B acceleration. However, for pure electromagnetic acceleration exhaust velocity could be described as:

$$\vartheta_{exh} = \sqrt{\frac{2qE \times B}{m_i}}$$

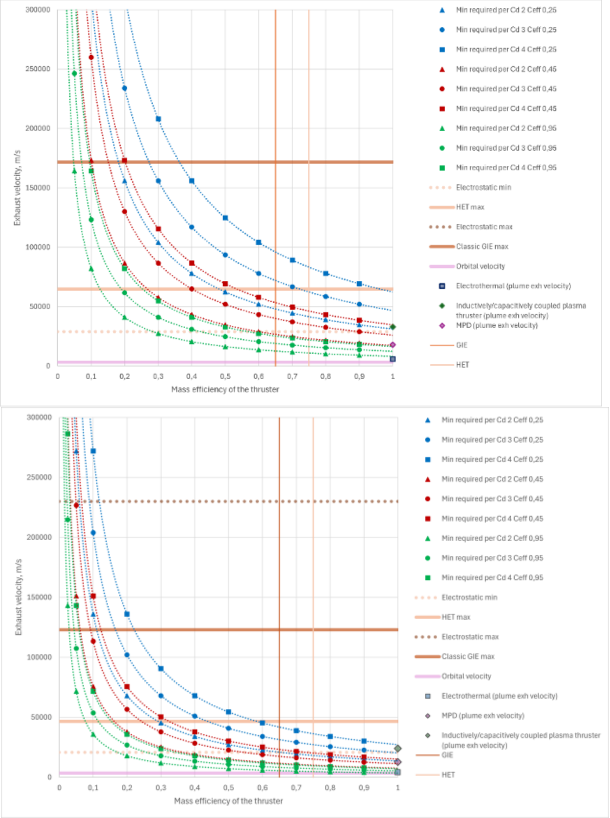


Figure 5. Maximum theoretical performance possible to reach per technology plotted over minimum required exhaust velocity per incoming particle for Earth (above) and Mars (below).

With certain limitations for the constant E×B one can then as well recalculate from best performing up-to-date technologies theoretical maximum for plume exhaust velocity operating with atmospheric propellants. Though it should be mentioned that molecular propellants would likely to underscore that theoretical limit as substantial amount of total power in those systems would go towards dissociation of the reactive atmospheric propellant lowering dramatically mass utilisation efficiency and in its turn plume exhaust velocity.

Figure 5 depicts calculated minimum required exhaust velocity to compensate drag for 9 different combinations between collection efficiency of intake (0,25; 0,45; 0,95) and drag coefficient (2; 3; 4) as a function of the mass utilisation efficiency of the thruster used. Maximum theoretical performances of various technologies are also plotted over the minimum required exhaust velocity to compensate the drag. In case of Inductively/capacitively coupled plasma thrusters, performances are calculated based on best to date reported performance taken from Ref. 4. It should be noted that it is not straightforward to apply the ideas behind the efficiency increase successfully implemented in that work to the case of RAM-EP systems. The first increase in efficiency and exhaust velocity was reached by injecting the propellant flow from the side of the magnetic nozzle exhaust and attributed to higher neutrals time of residence, increase and redistribution of neutral density inside of discharge channel[5]. Further increase in exhaust velocity and efficiency is due to utilisation of external magnetic coils and their power (estimated to minimum of 1000 W or 1/6th of total power) is not accounted in efficiency budget[4]. While coils indeed could be used in RAM-EP and would impact the efficiency and exhaust velocity, the impact would probably be not that high as either additional power cost need to be accounted or permanent magnets used that will limit the high efficiency region to a very narrow input mass flow range given the magnetic field needs to be adjusted for every specific operating flow rate to achieve the efficiency maximum. Injection of the mass flow from the side of magnetic nozzle represents of course a challenge: on one hand, for this type of thruster is especially important to accept the flow from the orbit uninterrupted and preferably in the form of Atomic oxygen as in this case no additional loses for dissociation will be drawn, on the other hand, need to inject from the thruster exhaust means the flow will have to travel through the complex geometry around the thruster and will be likely thermalised, molecular oxygen will probably be formed from the atomic. Overall, our believe is that realistic plume exhaust velocity for this family of thruster will be lower and towards that of MPDs. One should also be aware of the fact that higher exhaust velocities in inductively or capacitively coupled plasma thrusters are associated normally with higher power applied, which almost inevitably will lead to higher satellite drag coefficients given required power generation. The trend also has close to linear form and seems not to scale with the size[6].

Analysing figure 5 is rather easy to assess potential capabilities of different technologies to compensate the drag for Earth and Marth. Electrothermal propulsion would probably be not feasible candidate for RAM-EP technology at Mars unless collection efficiency of 95 % could be achieved on the system intake and satellite drag coefficient kept strictly at 2 or below. Of course, even in the case where very high collection efficiency could indeed be achieved, no room for flow misalignment or any extension of drag coefficient through unaccounted lateral flow could be accommodated. For the Earth case scenario, these technology couldn't make the job at all.

Classic Electromagnetic propulsion such as MPD could compensate the drag at Mars given C_d of around or under 3 at C_{eff} 0,45. With lower intake collection efficiency satellite drag coefficient should be kept under 2 to enable drag compensation. For Earth, this EP thrusters would compensate the drag when C_{eff} of 0,95 would become feasible but with no flexibility.

Inductively/capacitively coupled plasma thrusters, would they be able to achieve proposed by figure 5 exhaust velocity, could potentially compensate either the drag in broad range of satellite drag coefficients for intake collection efficiencies exceeding 35-45% or with lower collection efficiency of 25% only the drag drawn by a satellite with the drag coefficient of under 3 when at Mars. For Earth, they would need intake collection efficiency to be over 45% to compensate in drag coefficients under 4 or with lower collection efficiency of 25% but for the drag coefficients of around 2 and below.

For Electrostatic technology, the combination of exhaust velocity and mass utilization efficiency will also determine whether or not drag compensation will be possible on different combinations of drag coefficient and collection efficiencies. Hall Effect Thruster could be potential good candidate for the Mars Atmosphere Breathing Electric propulsion, but having low to no margin at maximum drag coefficient of 4 and collection efficiency of 25%. Gridded Ion Engine with around 65% on mass efficiency and around 175% on exhaust velocity could even accommodate some margin on flow misalignment, higher drag coefficient related to power demand due to lower thrust to power ratio etc. For the Earth RAM-EP, HETs become worse at compensation of the drag as they require higher collection efficiencies at intake (of at least 45%) while GIE would still be attractive technology providing some margins. However, neutralisation problem stands for those technologies and in order to keep plume exhaust velocity as high, alternative solutions should be found. This is due to the fact that when external neutraliser is used, the very same particles that generate the drag shall be deducted to participate in process of electron generation. That basically means, that the minimum exhaust velocity of energetic particles generating thrust has to be increased further. Principally, that only would still keep GIE feasible for the task in terms of exhaust velocity given the existing margin if neutraliser technologies were not that sensitive to reactive propellants and would not require majorly higher neutral density. Still, potentially that option can not be discarded only by exhaust velocity capability.

When speaking of alternative solutions with respect to the neutralisation, terrestrial applications are utilising various options since some decades. For example, AC field applied to the extraction region allows for ion-electron extraction as good as DC pulsing commonly known as beam switching. There are multiple particular schematic that are being used in terrestrial application and couple of companies that are trying to extend those to the space applications, e.g. ThrustMe[7] or Ion-X[8]. Basically, for RAM-EP the mean to utilise same incoming particles to generate both energetic ions producing thrust and electrons (or negatively charged ions) for compensation of the charge would convert GIE into invisible technology for ABEP applications.

IV. What is next

When the thruster technology is assessed for its capability to principally generate needed exhaust velocity for given expected system performance – drag coefficient and collection efficiency – global performance can be assessed, such as power to thrust ratio, ability to operate on available flow densities or any other mechanisms that could potentially lead to the power losses and as a result to the increase of the drag coefficient through the need of larger solar arrays to compensate for that power loss.

As it has already been determined, mass flow rate per time unit per surface unit is determined solely by orbital conditions and collection efficiency of intake. There is no potential to increase it further. Particle density on the contrary could be increased from the intake to the thruster discharge chamber passively or actively which is defined by compression ratio of intake. Obviously, increase in passively achieved compression ratio would be favourable for any thruster technology while introduction of active compression would add into required propulsion system power and thus further increase the drag coefficient of the satellite.

Last but not least topic that should be addressed is operation with reactive propellants in terms of potential discharge loses. Dissociation of oxygen, nitrogen and carbon dioxide happens with quite low electron temperatures and practically unavoidable within any propulsion system. The rate of dissociation and other attached plasma-chemical reactions will be lower for technologies with lower reachable electron temperatures that usually corresponds to lower discharge powers of course but also technology specific. Radio-Frequency Ion Engines are known for its low electron temperature leading to lowest number of doubly charged ions in operation with noble gases. In experiments with Oxygen, TransMIT has shown that under certain proportionality between injected power and available mass flow, dissociation could be avoided almost entirely. Though, it should be noted that operating in the broad range of orbits in the need of constant drag compensation perfectly optimal in terms of dissociation conditions might not always be reached as the operation is dictated by different primarily requirements.

One further aspect that is particularly applicable to the Earth RAM-EP conditions and is often overlooked, is variation in VLEO composition. Usually, experimental campaigns are performed using some preset proportionality between nitrogen and oxygen. The plasma-chemical process in EP range of electron temperatures and plasma densities, however, is very much dependent on proportionality between nitrogen and oxygen in the plasma as further processes start kicking in, such as for example dissociative recombination. To authors knowledge, TransMIT has been first company to test their thruster in the range of the oxygen-to-nitrogen compositions expected in VLEO between 180 to 250 km. The resulting plume composition was qualitatively characterized by the mass spectrometry and clear interdependence between internal plasma parameters and the outcoming plume fractions has been observed.[2]

Conclusion

Whenever the thruster technology shall be considered for RAM-EP, the research team should first be assessing whether or not that technology is able to deliver high enough exhaust velocity given the team expectations on the other RAM-EP system and satellite performance. Further, the required neutral density estimations, power demands expectancies and realistic evaluation of what the consequences of that power demands would be to the system global parameters, such as drag coefficient.

In the best of TransMIT believes, if any thruster technology will be able to perform the RAM-EP concept demonstration in space, meaning the full drag compensation utilising only the propellant that has been collected by the Spacecraft in orbit, then it will be Radio Frequency Ion Engine.

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