# Atmosphere-Breathing Electric Propulsion in Earth and Mars Orbits: Basic Feasibility Studies and the Case for Radio-Frequency Ion Thrusters

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**Abstract: Atmosphere-Breathing Electric Propulsion (ABEP) has long been proposed as an enabling technology for very low Earth orbits and extended Mars operations. Over more than 15 years, TransMIT GmbH has conducted a broad range of theoretical studies and experimental campaigns, encompassing various thruster sizes, power levels, and compatible neutralisation concepts. These efforts addressed operations in both Earth and Mars low orbits, taking into account the distinct atmospheric environments. Key research activities have included the feasibility and methodology studies of ABEP ground testing, approaches to reproduce orbital particle flows, and system-level analyses to define design envelopes and identify critical challenges. This paper presents the most recent theoretical and experimental results from TransMIT's work on ABEP based on Radio-Frequency Ion Engine technology for application in Earth and Mars orbits. Building on long-term project experience, we discuss the practical potential of ABEP for future missions, outline current understanding of its unique development challenges, and describe methodologies for selecting and designing suitable thruster technologies.**

## Nomenclature

| | | |
|---|---|---|
| *ABEP* | = | Atmosphere Breathing Electric Propulsion |
| *AETHER* | = | Air-breathing Electric THrustER |
| *BB* | = | Breadboard |
| *LEO* | = | Low Earth Orbit |
| *MABHET* | = | Martian atmosphere breathing Hall effect thruster |
| *MAVEN* | = | Mars Atmosphere and Volatile EvolutioN |
| *RAM-EP* | = | Air-breathing Electric Propulsion |

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*RIT* = Radio Frequency Ion Thruster
*S/C* = Space Craft
*VLEO* = Very Low Earth Orbit

## I.Introduction

Atmosphere-Breathing Electric Propulsion (ABEP) has emerged as a key enabling technology for sustained operations in very low Earth orbits (VLEO) and for drag compensation in low Mars orbits. By directly utilising ambient atmospheric particles as propellant, ABEP offers the potential for indefinite orbital maintenance without the need to carry onboard propellants. However, the concept faces unique challenges: the low particle flux in orbital environments, the strong dependence on atmospheric composition, and the need for efficient ionisation and neutralisation under rarefied conditions. These factors impose stringent requirements on thruster design and system integration, and they have made the practical implementation of ABEP a long-standing open question.

TransMIT GmbH has contributed to ABEP research for more than 15 years, performing both theoretical and experimental studies in the context of European and internal projects. Early work, such as the ESA-funded *“EPN Preliminary Characterisation Test Campaign of EP Technology with Non-Conventional Propellants”* (Prime contractor: ALTA S.p.A, ESA Contract No. 22946/09/NL/CO, 2010–2013) campaign, included testing of the RIT-10 breadboard with various atmospheric mixtures and demonstrated lifetimes exceeding 60,000 hours under air-breathing conditions—challenging the assumption that non-noble propellants inevitably shorten engine life[1]. Subsequent studies extended to Martian $CO_2$ operation, highlighted 30% higher RF power requirements but confirmed the feasibility of drag compensation in the 140–190 km altitude range. Furthermore, the use of alternative energy sources—such as nuclear power—was identified as a significant enabler for broader technology adoption and mission feasibility.

More recently, TransMIT contributed to the Horizon 2020 AETHER project (*Air-breathing Electric THrustER*), investigating a 22 cm Radio-Frequency Ion Engine with $N_2/O_2$ inflows representing altitudes between 185 and 250 km[2]. These tests established the drag compensation under various dissociation scenarios and provided valuable insights into how system behaviour changes with gas composition and power input, further advancing the understanding of ABEP operation. Current activities include two ESA projects: the “Mars Atmosphere-Breathing Electric Propulsion Thruster,” developing and testing an RIT optimised for Martian atmospheric conditions, and the “Cathodeless Electric Propulsion Thruster,” exploring beam-switching configurations for combined ion and electron extraction.

Building on this continuous body of work, this paper summarises the main outcomes of TransMIT’s feasibility studies, identifies the dominant limitations of ABEP technologies, and explains why Radio-Frequency Ion Thrusters emerge as the most practical solution for both Earth and Mars applications. We then present recent developments and testing campaigns of RIT-based ABEP systems, outlining lessons learned and future directions for system implementation.

## II.Orbital environment

Understanding the conditions under which the propulsion system will function is especially critical for atmosphere-breathing propulsion, where the amount and type of propellant are directly determined by the operating environment's characteristics. A detailed review of the atmospheric properties in the relevant altitude ranges is therefore essential to assess feasibility and to define requirements for ABEP thruster design.[3]

### A. Earth Atmosphere: General Aspects

The Earth's atmosphere contains nitrogen, oxygen (molecular and atomic), argon, carbon dioxide and some further gases, including water vapour. It is dominated by nitrogen up until around 185 km, where atomic oxygen starts to prevail and then dominate. The composition, density, and temperature vary strongly with altitude and are influenced by external drivers such as solar activity, geomagnetic storms, and diurnal cycles. These variations affect both the total particle flux available for ingestion and the degree of ionisation within the background plasma. The Atmosphere Breathing Propulsion System operation shall take these effects into account and needs to be compatible with them.

For engineering purposes, particle densities and temperatures are typically derived from the NRLMSISE-00 model, which provides altitude-dependent values under different solar conditions.

### B. Martian atmosphere: general aspects

The Mars atmosphere is far less investigated than the Earth's atmosphere. It is a thin $CO_2$-dominated atmosphere comprising only a few mbar of pressure at the Martian low altitudes. While less dense overall, the Martian atmosphere exhibits similar properties: it is held by gravity, has complex chemical effects as well as fluid and thermodynamic effects, and is not uniform, changing with time and place, which translates into Mars' weather. The Mars atmosphere can be divided into the lower, middle (or Mesosphere), upper (or Thermosphere), 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 Martian surface and in the 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 a conclusion to the Electric Propulsion System operation.

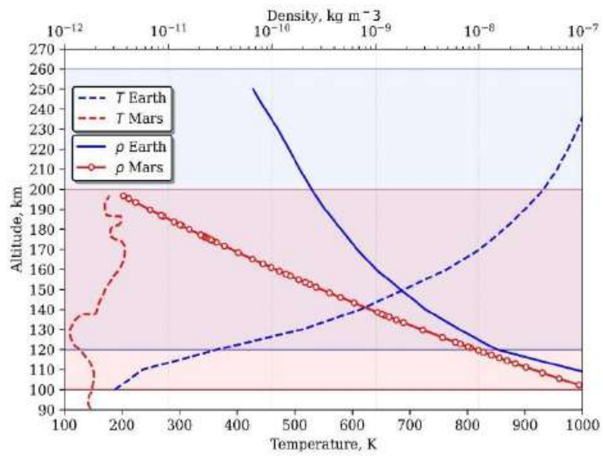



**Figure 1. Temperature and density variation with the altitude of the orbits of Earth and Mars.** [3]

Our first assessment of the range for Mars Atmosphere Breathing Propulsion Systems targets altitudes between 130 and 180 km, mostly belonging to the Thermosphere, with the slight coverage of the upper Mesosphere too. The atmospheric conditions 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 ionise atoms and molecules. These effects also need to be assessed for compatibility with the Atmosphere Breathing Propulsion System operation.

The practical information on particle densities is coming from well-established models and probe measurements. Most renowned is the Mars-GRAM 2000 model of NASA (and its more recent updates) that is based on numerous measurement data from Viking 1 and Mariner missions of 70s last century and its recent update – Mars-GRAM 2010 – to add a user-controlled dust case and better match Mars Global Surveyor,1 Mars Odyssey, and Mars Reconnaissance Orbiter data. Recent data, including measurements from Pathfinder and, most notably, the Mars Atmosphere and Volatile Evolution mission (MAVEN), have complemented our survey. Further data has been collected 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's potential operating environment.

The details of the methodology 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. Implications for RAM-EP at Earth and Mars

The operational altitude range typically considered for Earth ABEP applications is 160–250 km, while for Martian RAM-EP it is 100–200 km. Within these intervals, already from general number density profiles (Fig. 2), one can observe about an order of magnitude variation in atmospheric density at Earth, and up to four orders of magnitude at Mars. Conventional electric propulsion systems are not designed to accommodate such broad variations in mass flow, since optimisation across this entire range is practically impossible.

Atmospheric density data at a given altitude provides the foundation for feasibility analysis:

- Multiplied by orbital velocity, it yields the collectable mass flow per unit intake area (Fig. 3, left).
- Multiplied by the square of orbital velocity, it yields the drag force per unit frontal area (Fig. 3, right).

This simple scaling already represent a best-case scenario, even before introducing corrections for orbital conditions or environmental variability. Thus, the curves in Figs. 3 can be considered the reference baseline against which any candidate RAM-EP system must be evaluated.

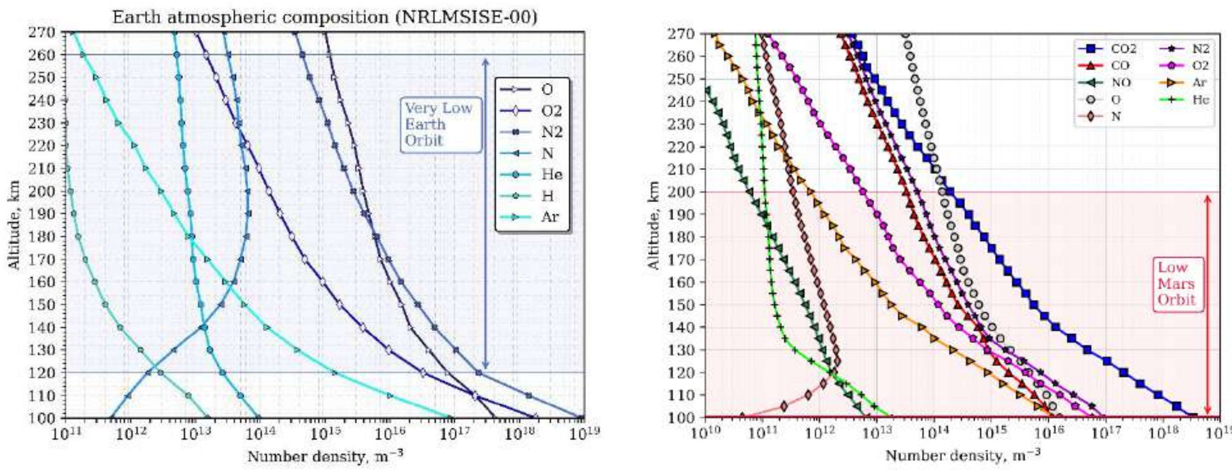



**Figure 2. Composition variation with the altitude of Earth (middle) and Mars (right).** [3]

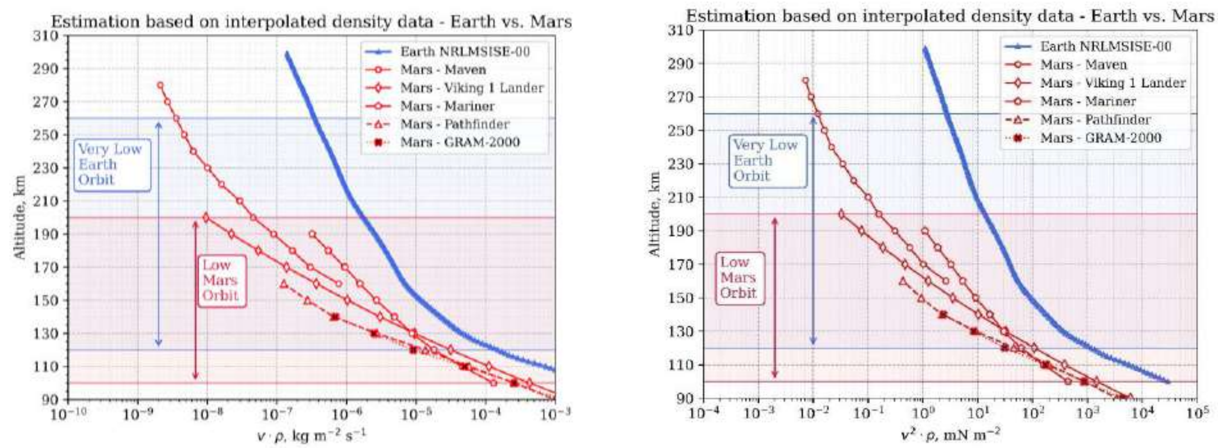



**Figure 3. Mass flow rate available (left) and drag generated (right) per surface unit as a function of altitude.**

Furthermore, several key observations can be made:

- For the same collected mass flow, drag at Earth is nearly twice that at Mars (for similar drag coefficients). However, the ~2.5× lower available solar power density at Mars still limits the achievable thrust–drag balance compared to Earth.
- Mars poses the larger challenge despite its lower absolute drag, because the mass flow and drag curves are far steeper. This implies that any practical Martian ABEP design would be constrained to a very narrow operational altitude window, leaving little flexibility for orbit selection. And that is not touching power availability topic.
- In equivalent terms, 160–250 km on Earth corresponds to ~135–185 km on Mars in mass flow, and ~125–160 km in drag. Below ~130 km, Martian dust storm effects must also be considered, along with extreme space weather variability.
- Crucially, the collectable mass flow is finite—contrary to common assumptions in some RAM-EP concepts. Each collected particle generates not only potential thrust but also drag. Therefore, maximising exhaust velocity becomes the top priority, though the available power budget remains a decisive factor.

## III.Basic Feasibility

The Atmosphere-Breathing Electric Propulsion concept imposes a set of critical and tightly interrelated requirements on the propulsion system, encompassing both the intake and the thruster head. These requirements are inherently

interdependent, necessitating a concurrent engineering approach for the successful development of both components. Therefore, over the past several years, TransMIT and the Bundeswehr University Munich have embraced a more integrated and collaborative development model, aimed at ensuring comprehensive coverage of all interdisciplinary scientific and engineering domains essential to ABEP system design.
Nevertheless, to assess the fundamental feasibility of a thruster technology for ABEP drag compensation, a set of simplified, high-level analyses can—and should—be performed prior to engaging in detailed simulations or complex modelling. These initial assessments rely on classical propulsion equations and broad assumptions. If a technology proves unfeasible at this preliminary stage, it is a strong indication that further investment into its development for ABEP applications would not be justified.

### A. Minimum required exhaust velocity

One of the most critical criteria for ABEP feasibility is the minimum required exhaust velocity. In a first-order approximation, this requirement can be expressed independently of the total mass flow rate, as a function of the satellite's collection efficiency and drag coefficient. It is important to note, however, that this simplified model may not fully account for the contribution of lateral satellite structures to the overall drag, making the resulting value a best-case scenario—or, at most, a lower bound for the exhaust velocity requirement.

Furthermore, the interpretation of this minimum exhaust velocity must be adapted based on the specific propulsion technology under consideration. In particular, different technologies will be affected in varying ways by the dissociation of atmospheric species, which can significantly influence the effective exhaust velocity. This factor must therefore be carefully considered when assessing compliance with the minimum exhaust velocity requirement.

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 at 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 $\vartheta_i$ – energetic particles exhaust velocity and $\dot{m}_i$ – total mass flow rate generating the thrust. Ion mass flow rate, as always, is given by the multiplication of the total thruster mass flow rate ($\dot{m}_{thr}$) and the mass utilisation efficiency of the thruster ($\eta_m$). And in its turn, the mass flow rate of the thruster can be expressed as the total mass flow rate available at 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 equations together, the following equation for the minimum required exhaust velocity of energetic particles can be put together:

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

Or for the full exhausted plume, average velocity:

$$\vartheta_{exh} \geq \frac{\vartheta_{orbital}C_d}{2C_{eff}}, \qquad \text{Eq.6}$$

It is also possible to estimate the exhaust velocity achievable by a given electric propulsion technology for a specific propellant—such as a nitrogen-oxygen mixture in the case of Earth, or carbon dioxide in the case of Mars. For electrostatic thrusters in particular, these calculations are relatively straightforward, as the exhaust velocity of the accelerated particles can be expressed by the following relation:

$$\vartheta_i = \sqrt{\frac{2qU}{m_i}}, \quad \text{Eq.7}$$

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

$$\vartheta_{exh} \propto \sqrt{\frac{3kT}{m}}, \text{ Eq.8}$$

- where k is the Boltzman constant and T is the effective temperature that is limited for Electrothermal propulsion by thermomechanical properties of the thruster materials exposed. Assuming that the best performance achieved by the current technologies already represents the maximum attainable temperature, limited by the properties of existing materials, the conversion for the exhaust velocity of a plume using one propellant to that using another can be expressed as follows:

$$\vartheta_{exh\ atm} = \vartheta_{exh\ state-of-the-art}\sqrt{\frac{m_{state-of-the-art}}{m_{atm}}} \quad \text{Eq.8}$$

This provides a simple way to estimate the possible exhaust velocity of thermal propulsion for atmospheric propellants, assuming identical operating temperatures and similar expansion conditions.

For the purposes of this study, several top-performing hydrazine and hydrogen arcjet systems—representing the current state-of-the-art within their respective technology classes—were used as benchmarks. Their performance was recalculated for operation with a nitrogen-oxygen mixture to evaluate the expected exhaust velocity under similar thermal constraints.

Electromagnetic systems—or more precisely, electric propulsion technologies often categorised as such—present a particularly complex case when it comes to determining exhaust velocity. In practical implementations, these systems frequently exhibit a combination of all three acceleration mechanisms: thermal, electrostatic, and electromagnetic. In some cases, thermal or electrostatic acceleration may even dominate over the idealised E × B electromagnetic mechanism.

However, for a system operating under *pure* electromagnetic acceleration, the exhaust velocity can be described by the following expression:

$$\vartheta_{exh} \propto \sqrt{\frac{2qE\times B}{m_i}} \quad \text{Eq.9}$$

With certain assumptions and limitations—particularly in the case of systems operating under constant E×B acceleration—it is possible to extrapolate from the best-performing current technologies a theoretical upper limit for exhaust velocity when operating with atmospheric propellants.

However, it must be emphasised that molecular propellants, such as those present in planetary atmospheres (e.g., $N_2/O_2$ for Earth, $CO_2/N_2/O_2$ for Mars), are likely to fall short of this theoretical maximum. A significant portion of the input power in such systems is consumed by the dissociation of these molecules into atomic or ionised species before they can be effectively accelerated. This process substantially reduces the mass utilisation efficiency, which in turn results in a notable decrease in achievable plume exhaust velocity compared to the ideal case.

Figure 4 depicts the calculated minimum required exhaust velocity to compensate for 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 for the drag. In the case of Inductively/capacitively coupled plasma thrusters, performances are calculated based on the best reported performance to date, as referenced in Ref. 4.

It should be noted that the efficiency enhancement strategies successfully demonstrated in the referenced work cannot be directly applied to RAM-EP systems. In that study, the initial increase in efficiency—up to 20%—and corresponding rise in the exhaust velocity were achieved by injecting the propellant laterally at the magnetic nozzle exhaust. This improvement was attributed to the increased residence time of neutrals and the redistribution of the neutral density within the discharge channel.[5]

A further increase in efficiency—up to 30%—was reported as a result of employing external magnetic coils. However, the power consumed by these coils (estimated by the authors to be at least 1000 W, or approximately one-sixth of the total system power) was not included in the reported efficiency budget.[4]

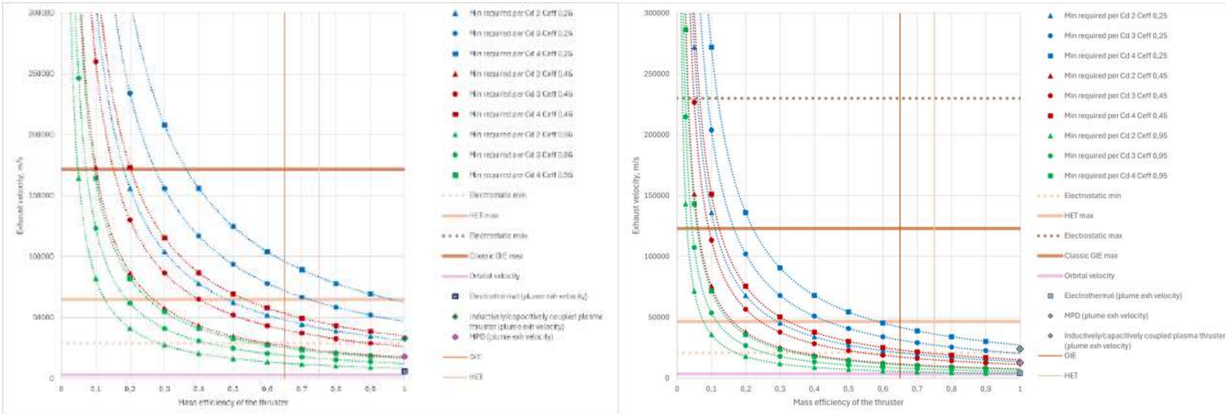



**Figure 4. Maximum theoretical performance possible to reach per technology plotted over the minimum required exhaust velocity per incoming particle for Earth (left) and Mars (right).**

While the use of magnetic coils is theoretically applicable to RAM-EP systems and could positively influence both efficiency and exhaust velocity, the real impact is likely to be more constrained. If active coils are used, the additional power consumption must be considered in the system's overall efficiency. Alternatively, employing permanent magnets would eliminate the power cost but would introduce a different limitation: the magnetic field strength would become fixed, making it difficult to optimize the efficiency across the wide range of input mass flow rates typically encountered in RAM-EP applications. Given the variability and unpredictability of atmospheric conditions, the inability to dynamically adjust the magnetic field would restrict the system's performance to a narrow operating window, therefore reducing its practical applicability.

The injection of atmospheric mass flow from the side of a magnetic nozzle presents several notable challenges. On one hand, for this class of thruster, it is particularly advantageous to receive the incoming atmospheric flow directly from the orbital direction—ideally in the form of atomic oxygen. In such a case, no additional energy would be required for dissociation, thereby improving overall efficiency. On the other hand, the need to inject the flow through the thruster exhaust path introduces complications: the gas must navigate complex internal geometries of the propellant supply system, during which it is likely to become thermalised, and recombination of atomic oxygen into molecular oxygen is probable.

As a result, we believe that the realistic plume exhaust velocity achievable by this class of thrusters will be significantly lower than the theoretical estimates—likely approaching the performance limits of magnetoplasmadynamic thrusters in the best-case scenario.

It is also important to note that in inductively or capacitively coupled plasma thrusters, higher exhaust velocities are typically associated with increased input power. However, this higher power requirement nearly always translates into greater satellite drag coefficients, due to the larger or more exposed power generation systems (e.g., solar arrays). Notably, this relationship tends to be approximately linear and does not scale favorably with system size.[6]

An analysis of Fig 4 allows for a straightforward assessment of the potential of various electric propulsion technologies to achieve drag compensation in both Earth and Mars orbits based on their exhaust velocity.

Electrothermal propulsion is unlikely to be a feasible candidate for RAM-EP systems at Mars, unless an exceptionally high collection efficiency—on the order of 95%—can be achieved at the intake, and the satellite drag coefficient ($C_d$) is strictly maintained at 2 or lower. Even under these optimal conditions, there would be no tolerance for flow misalignment or additional drag contributions from lateral structures. In the Earth case, electrothermal systems are clearly inadequate for RAM-EP applications due to the higher drag environment and insufficient exhaust velocities.

Classic electromagnetic thrusters such as magnetoplasmadynamic (MPD) systems could potentially enable drag compensation at Mars, particularly if the drag coefficient is kept at or below ~3, with a collection efficiency ($C_{eff}$) of approximately 45%. For lower collection efficiencies, the drag coefficient must be reduced to below 2 to ensure feasibility. In Earth orbit, MPD thrusters could theoretically compensate for drag only if an extremely high collection efficiency of 95% were achieved, leaving no operational flexibility.

If inductively or capacitively coupled plasma thrusters can reach the exhaust velocities shown in Fig 4, they could support drag compensation over a wide range of drag coefficients—provided the intake collection efficiency exceeds 35–45%. With a lower collection efficiency (around 25%), compensation would only be feasible at Mars and limited to satellites with Cd < 3. For Earth, these technologies would require $C_{eff}$ > 45% to handle drag coefficients under 4, or $C_{eff}$ of 25% for Cd values of ~2 or lower.

In electrostatic systems, such as Hall Effect Thrusters (HETs) and Gridded Ion Engines (GIEs), the combination of exhaust velocity and mass utilisation efficiency ultimately determines feasibility.

Hall Effect Thrusters (HETs) could be suitable candidates for Mars-based RAM-EP, albeit with very limited margins. At $C_d = 4$ and $C_{eff} = 25\%$, HETs operate at the edge of feasibility, offering little room for additional losses or design compromises.

Gridded Ion Engines (GIEs), with mass utilisation efficiencies up to 65% and exhaust velocities up to 175% higher than other EP technologies, offer greater flexibility. They can potentially accommodate flow misalignment, increased drag due to power system-induced drag and variations in thrust-to-power ratio.

For Earth-based RAM-EP, HETs are less viable, as they require intake collection efficiencies exceeding 45%, offering minimal operational margin. GIEs, however, remain an attractive candidate, even under more demanding Earth-orbit conditions.

A key limitation for both HETs and GIEs lies in neutralisation. To maintain high plume exhaust velocity, traditional external neutralisers, which consume a portion of the collected atmospheric particles, introduce a major constraint. The same particles used to generate thrust also must be diverted for electron (or other negative charge carrier) generation, effectively increasing the minimum required exhaust velocity for the thrust-producing particles. Although GIEs have a theoretical margin to accommodate this increase, the feasibility is conditional. Moreover, the integration of a neutraliser into a RAM-EP system introduces additional challenges. Unlike classical EP systems, where a separate propellant feed for the neutraliser can be straightforwardly implemented, RAM-EP requires this feed to be connected through the intake itself. Designing such a coupled system is non-trivial. Neutraliser technologies must be compatible with reactive propellants and avoid the need for significantly higher neutral densities, which could compromise efficiency or lead to increased erosion and operational instability.

When considering alternative solutions for neutralisation, it is worth noting that terrestrial plasma applications have employed various approaches for several decades. For example, applying an alternating current (AC) field in the extraction region has proven effective for simultaneous ion and electron extraction, achieving results comparable to those obtained with traditional DC pulsing or beam switching methods.

A few specific configurations and schematics are commonly used in ground-based systems, and some companies are actively working to adapt these techniques for space applications. Notable examples include ThrustMe[7] and Ion-X[8], both of which are exploring compact, alternative neutralisation architectures for space-based electric propulsion.

In the context of RAM-EP, a system capable of utilising the same incoming atmospheric particles to generate both the energetic ions for thrust and the electrons (or negatively charged ions) required for charge compensation would be a transformative advancement. Such an approach would eliminate the need for a dedicated cathode, removing the component that most critically limits lifetime in conventional EP systems. Preliminary studies of grid erosion under atmospheric propellants already indicate lifetimes comparable to, or even exceeding, those achieved with classical xenon propellants. This makes cathode-less GIE configurations a particularly promising candidate for sustained RAM-EP operation.

### B. Minimum required electrical efficiency and thrust-to-power ratio

The minimum required thruster electrical efficiency $\eta_p$ for drag compensation in a RAM-EP system can be estimated using the following expression:

$$1 > \eta_p \geq \frac{T^2}{2*P_{in}*\dot{m}_{thr}} = \frac{D^2}{2*P_{in}*\dot{m}_{thr}} = \frac{\dot{m}^2*v_{orbital}^2*C_d^2}{4*2*P_{in}*\dot{m}_{thr}} = \frac{\dot{m}_{thr}^2*v_{orbital}^2*C_d^2}{C_{eff}^2*8*P_{in}*\dot{m}_{thr}} = \frac{\dot{m}_{thr}*v_{orbital}^2*C_d^2}{C_{eff}^2*8*P_{in}} = \frac{\dot{m}_{thr}*v_{orbital}^2*C_d^2}{8*P_{in}*C_{eff}^2}$$

Eq.10

Where $P_{in}$ is the input electrical power available to the thruster.

In this way, the minimum required thruster electrical efficiency can be expressed as a function of thruster mass flow, orbital velocity, drag coefficient, available electrical power, and collection efficiency. A closer look, however, reveals that these parameters are strongly interdependent:

- **Thruster mass flow** depends on orbit (atmospheric properties), intake cross-sectional area (satellite outer geometry), and collection efficiency (intake design).
- **Orbital velocity** is determined by orbital altitude.
- **Drag coefficient** is set by the satellite's external geometry, particularly the size and shape of the main body and solar arrays.
- **Available electrical power** scales with solar array area and geometry, and is therefore directly linked to the drag coefficient.
- **Collection efficiency** is governed by intake design.

For example, one might assume that scaling down the satellite or its elements would ease efficiency requirements. While this strategy reduces mass flow and drag coefficient, it also decreases the available electrical power due to the smaller solar array area. As a result, the minimum required thruster efficiency often becomes even more stringent.

The minimum required electrical efficiency also sets a bound on the necessary minimum thrust-to-power ratio, which can be expressed as:

$$\frac{T}{P_{in}} \geq \sqrt{\frac{2*\dot{m}_{thr}*\eta_p}{P_{in}}} = \sqrt{\frac{{\dot{m}_{thr}}^2*{v_{orbital}}^2*{C_d}^2}{4*{P_{in}}^2*{C_{eff}}^2}} = \frac{\dot{m}_{thr}*v_{orbital}*C_d}{2*P_{in}*C_{eff}} \qquad \text{Eq.11}$$

As with the Eq. 10, all components of this equation are interdependent and rely on satellite geometry, mission properties, and RAM-EP characteristics. This additional constraint narrows the choice of feasible propulsion technologies even further.

These interdependencies highlight that achieving sufficient propulsion performance in RAM-EP systems is not a matter of optimising each factor independently. Instead, it requires a tightly coupled, multi-parameter optimisation, where intake design, spacecraft geometry, and propulsion technology must be developed in concert.

It is worth noting that during most thruster development activities, the thruster is often designed without well-defined satellite characteristics, making precise use of Eq. 10 and 11 nearly impossible. Nevertheless, even at an early stage, rough estimates—such as typical drag coefficients, state-of-the-art intake efficiencies, expected orbital velocities, and preliminary thruster massflow (given the chosen orbits, and the expected intake size) and power consumption values—can already provide valuable insight into the feasibility of a given propulsion concept for RAM-EP applications.

### C. Further considerations

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 electrical efficiency, thrust-to-power ratio, ability to operate on available flow rates and densities (considering reachable compression ratios) 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, massflow rate per time unit per surface unit is determined solely by orbital conditions and collection efficiency of intake (Fig. 3). 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 the compression ratio of the intake. Obviously, an increase in passively achieved compression ratio or intake collection efficiency would be favourable for any thruster technology, while the introduction of active compression would add to the required propulsion system power and thus further increase the drag coefficient of the satellite.

Last but not least, the topic that should be addressed is operation with reactive propellants in terms of potential discharge losses. Dissociation of oxygen, nitrogen and carbon dioxide happens with quite low electron temperatures and is 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, which usually corresponds to lower discharge powers, of course, but also is technology-specific.

Radio-Frequency Ion Engines are known for their low electron temperature, leading to the lowest number of doubly charged ions in operation with noble gases. In experiments with Oxygen, TransMIT has shown that under a 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 necessitates constant drag compensation. Although perfectly optimal in terms of dissociation conditions, it might not always be reached, as the operation is dictated by different primary requirements.

One further aspect that is particularly applicable to the Earth RAM-EP conditions and is often overlooked is variation in VLEO composition. Until recently, most experimental campaigns were conducted with fixed nitrogen–oxygen ratios, rather than reflecting the altitude-dependent variations actually encountered in orbit. However, plasma-chemical processes in the EP-relevant range of electron temperatures and plasma densities are highly sensitive to the $N_2/O_2$ balance, as additional processes such as dissociative recombination begin to play a significant role. To the authors' knowledge, TransMIT was the first to test a thruster across the full range of oxygen-to-nitrogen compositions expected in VLEO (180–250 km) and Mars Low Orbit (130–180 km). The resulting plume composition was qualitatively characterised by the mass spectrometry, and clear interdependence between internal plasma parameters and the outgoing plume fractions has been observed.[2]

## D. State-of-the-art

For the few thruster technologies where performance characterisation with air or carbon dioxide has been published, experimental data have been extracted and plotted together with the minimum required thrust for drag compensation, shown as a function of the operating mass flow rate. These limits were calculated for various combinations of drag coefficients and intake collection efficiencies, using the corresponding minimum exhaust velocity described above. Although this analysis includes uncertainties, notably due to the manual extraction of data from published graphs, the resulting trends still provide practical bounds for assessing feasibility. Figure 5 summarises all collected data and, as can be seen, validates the theoretical assessment presented in Fig. 4. For clarity, scaled-up versions of Fig. 5 highlighting specific ranges of interest are provided in the Appendix.

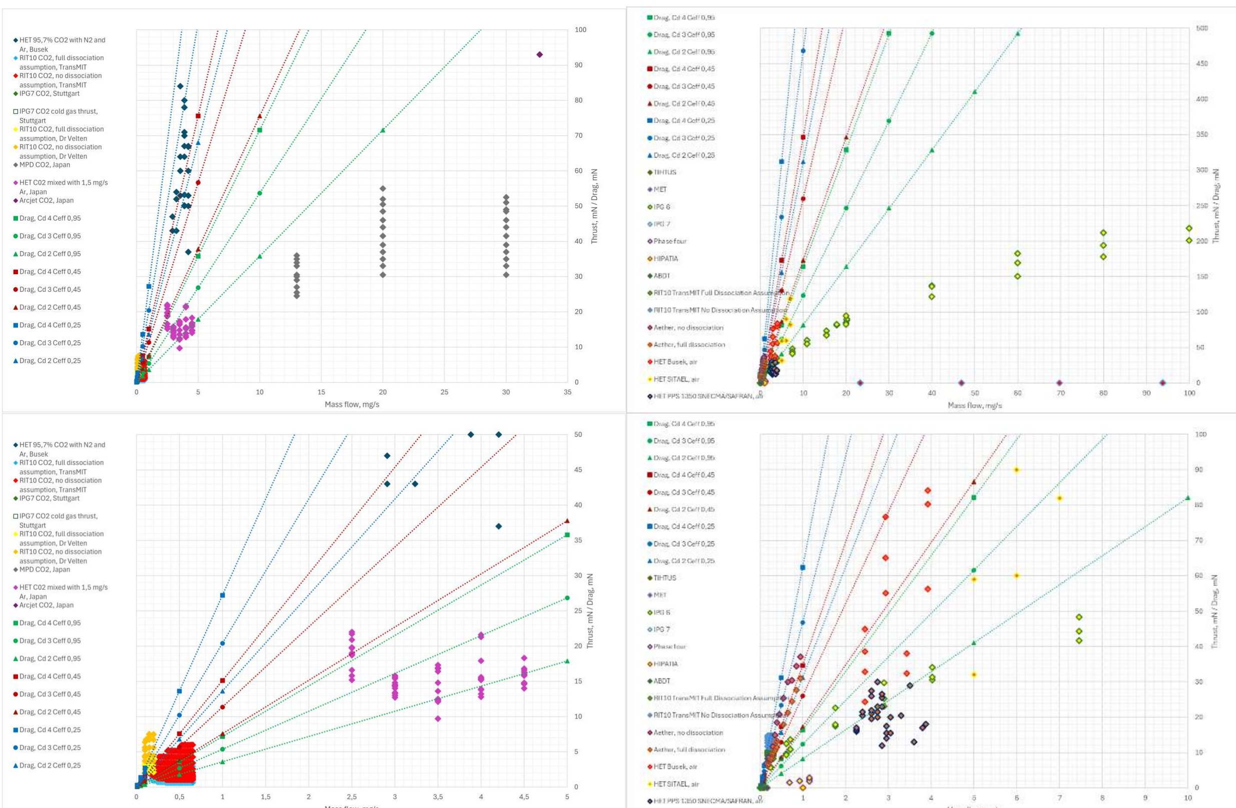



**Figure 5. Experimentally reached performance of different EP groups plotted over minimum required thrust per thruster mass flow for Mars (left) and Earth (right)[9-42].**

# IV. Radio Frequency Atmosphere Breathing Ion Engine for Earth and Mars low orbits

Recent efforts of TransMIT are focused on the design, development, and characterisation of Radio-Frequency Ion Engines for Atmosphere-breathing Electric Propulsion capable of drag compensation in low Earth and Martian environments for smallsats. The design effort, supported by preparatory work carried out in close collaboration with the Bundeswehr University of Munich (UniBwM), included an extensive review of Martian environmental conditions, assessment of high-level intake performance in the Martian environment, a high-level system feasibility study and appropriate thruster optimisation and construction. The thruster was tested with an air mixture back in 2022 and is now going through the characterisation test campaign for the Martian atmosphere mixture in the R2D2 vacuum facility of UniBwM.

### A. RAM-RIT 22/16

The thruster is a prototype of the radiofrequency ion thruster sized to 22 cm, originally developed within the AETHER activity. The thruster employs an electrodeless inductive discharge for plasma generation, based on the standard configuration of an inductively coupled plasma source. A schematic of the operating principle is shown in Fig. 6.

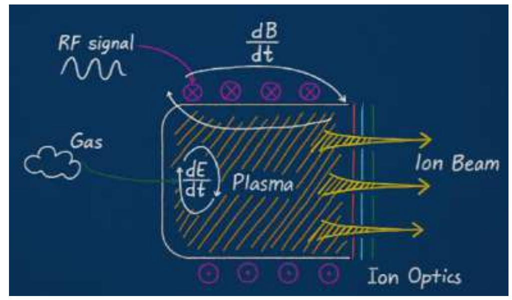


**Figure 6. Operating principle of a Radio Frequency Ion Thruster.**

The RAM-RIT 22/16 consists of four main sub-assemblies:

- **Ioniser:** ceramic discharge chamber with an RF coil for inductive plasma generation. The design is adjustable, allowing for the change of the length of the discharge channel and the geometry of the RF coil as needed.
- **Gas inlet:** feedline with gas isolator to electrically decouple the floating discharge chamber from ground.
- **Grid system:** three-grid extraction system specifically designed to support atmospheric propellant extraction.
- **Thruster housing and interface:** structural support and back plate with interfaces for RF, HV, and propellant connections.

A photograph of the RAM-RIT 22/16 is shown in Fig. 7.

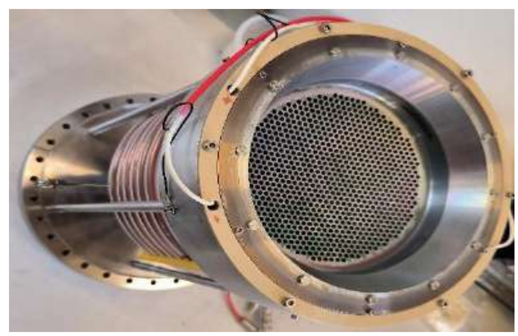


**Figure 7. Engineering model of the RAM-RIT 22/16**

### B. Test Facility

The experiments were conducted in the R2D2 vacuum facility located at the REL laboratory on the UnibwM campus in Munich (depicted at Fig. 8). The facility consists of a main chamber and a hatch, separated by a high-vacuum gate valve, with a total volume of about 4.5 m³. The main chamber is approximately 1.9 m in diameter and 4 m in length, while the hatch provides an additional cylindrical volume with a diameter of 0.75 m.

The pumping system combines turbomolecular and cryogenic stages, providing a total pumping speed of about 24,000 l/s and achieving base pressures as low as $10^{-8}$ mbar without propellant flow. During thruster operation with target flow rates derived during the activities, chamber pressures was kept in the range of $5 \times 10^{-4}$ to $1 \times 10^{-5}$ mbar were maintained.

The thruster under test was mounted in the hatch and operated downstream into the main chamber toward a carbon beam target located at the chamber end. The target surface was positioned at 45° relative to the thruster axis to reduce back-sputter towards the ion optics while maintaining realistic scattering conditions. Chamber conditions were monitored with six vacuum gauges and a set of calibrated temperature sensors on the cryopanels and target.


**Figure 8. R2D2 Test Facility.**

## C. Test Setup

The RAM-RIT 22/16 thruster was installed in the hatch of the R2D2 facility and operated downstream into the main chamber. The thruster was aligned toward a carbon beam target located at the opposite end of the chamber.

During the test following diagnostics were implemented (see Fig. 9 for a schematic view):

- Residual Gas Analyser (RGA): connected to the main chamber to monitor gas composition and detect any contamination during long-duration operation.
- Optical access: allowed for optical emission spectroscopy (OES), which, together with probe diagnostics, was used to confirm nominal thruster operation after each reassembly and to monitor changes in the thruster characteristics over time.

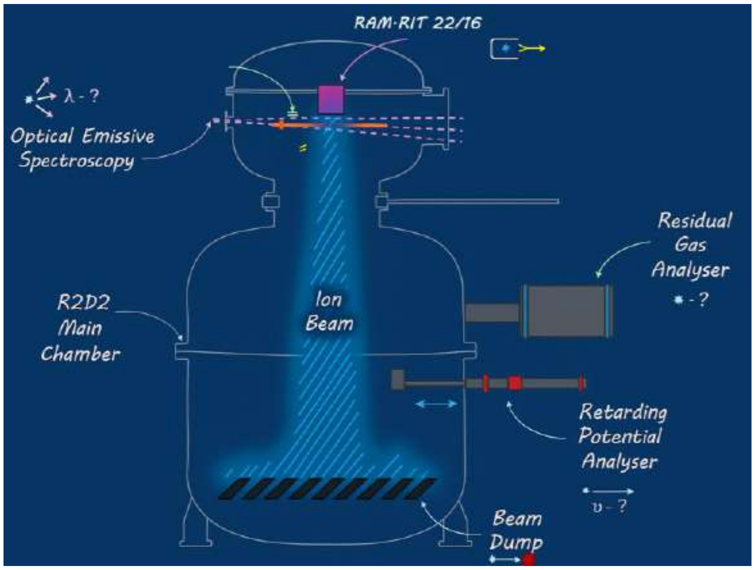



**Figure 9. Test setup schematic.**

- Retarding Potential Analyser (RPA): mounted on a moving arm in the main chamber compartment, enabling measurements of the ion energy distribution function (IEDF). At full extension, the RPA could be positioned in the beam centre, while during endurance operation, it was retracted to avoid influencing sputtering.
- Thermal monitoring: PT100 sensors recorded cryopanel and target temperatures, with data collected via a Keithley DAQ system.

The RF generator used to ignite and sustain the discharge was an industrial SEREN RFG HR-1001 with specially designed matching network. The high-voltage power supplies and mass flow boards were installed externally, adjacent to the hatch. While plasma diagnostics were applied only at specific operating conditions, thruster operating parameters (mass flow rates, voltages, currents, RF power and frequency) as well as facility conditions (vacuum pressures, cryopanel and target temperatures, laboratory environment sensors for air quality, temperature, pressure, and humidity) were continuously monitored and recorded throughout the campaign.

## D. Test Results

Figure 10 shows operation of the RAM-RIT 22/16 with Xe and atmospheric propellants.

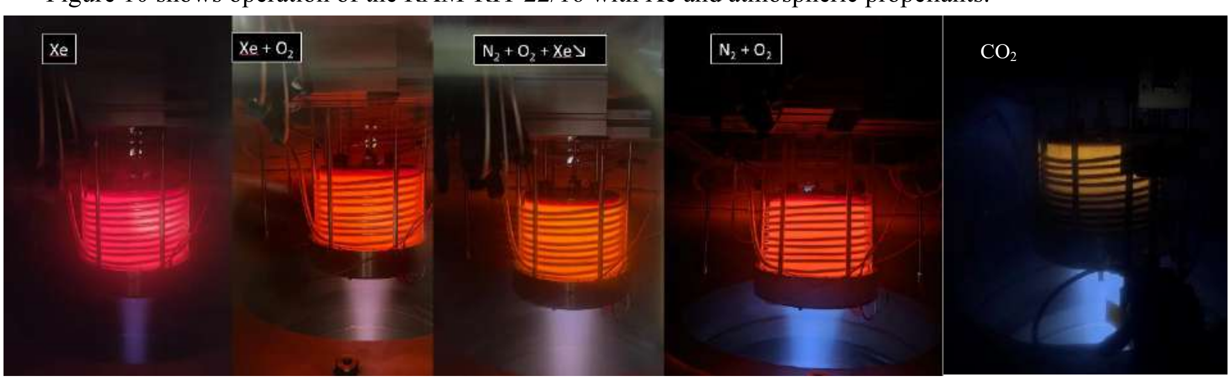



**Figure 10. RAM-RIT 22/16 in operation.**

Thrust of the RAM-RIT 22/16 is calculated based on the electrical parameters measured using classic formula:

$$F = \sum_{m_1..m_n} I_{Beam} \cdot \eta_{div} \cdot \sqrt{2 \frac{m_{ion}}{q_{ion}} (U_+ + V_P)} \quad \text{Eq. 12}$$

As the dissociation rate is not exactly known, thrust interval is calculated for every operating point with maximum thrust based on no dissociation assumption and minimum thrust in assumption of full dissociation.

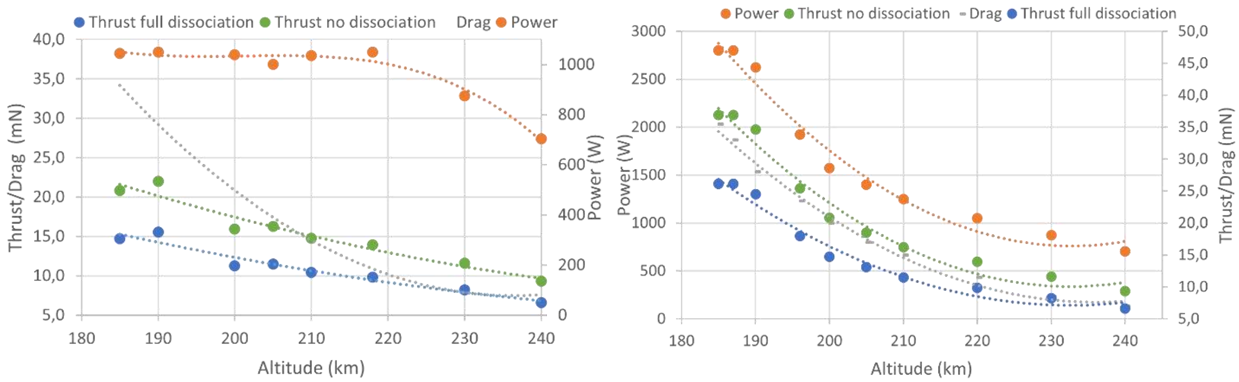



**Figure 11. RAM-RIT 22-16 performance with air mixtures, power limited (left) and total (right)**

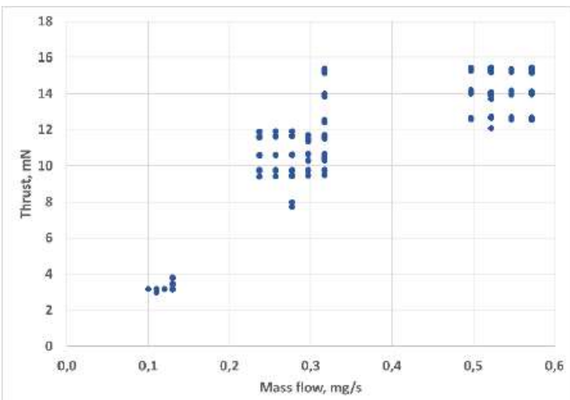



**Figure 12. RAM-RIT 22-16 performance with atmosphere mixture of Low Martian Orbital Environment**

Performances of RAM-RIT 22/16 against AETHER drag requirements are shown in the Fig. 11. The composition of the propellant used was representative of average composition at given altitude for every operating point. As can be seen, under power constraints of AETHER the thruster was capable of drag compensation above around 205 km orbits all the way up to 250 km showing also good compatibility with ultra low flow amounts. With extended power though, thruster was able to demonstrate up to 35 mN thrust required at 185 km orbit.

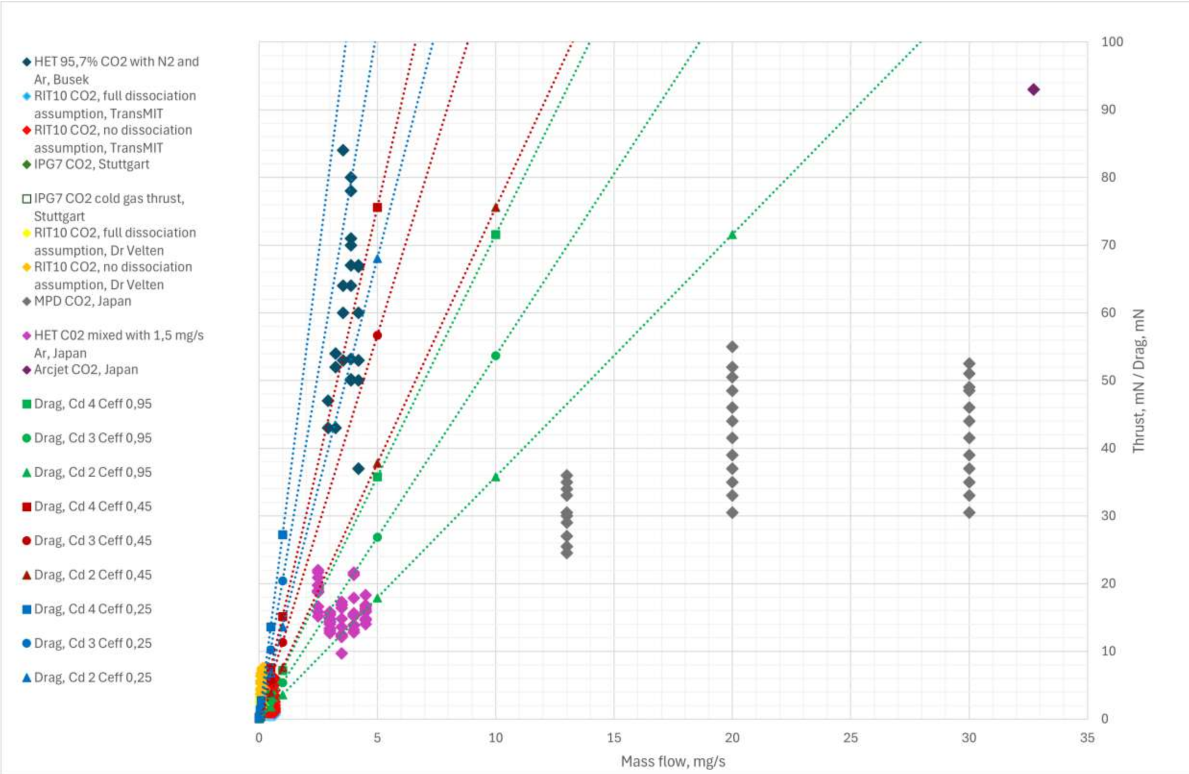
Test for martian environment is still on-going therefore only preliminary evaluation of the performance collected up to date is available. Firure 12 depicts thrust per incoming mass flow performance of RAM-RIT 22/16 operating on carbon dioxide mixtures with nitrogen and oxygen corresponding to low martian environment.

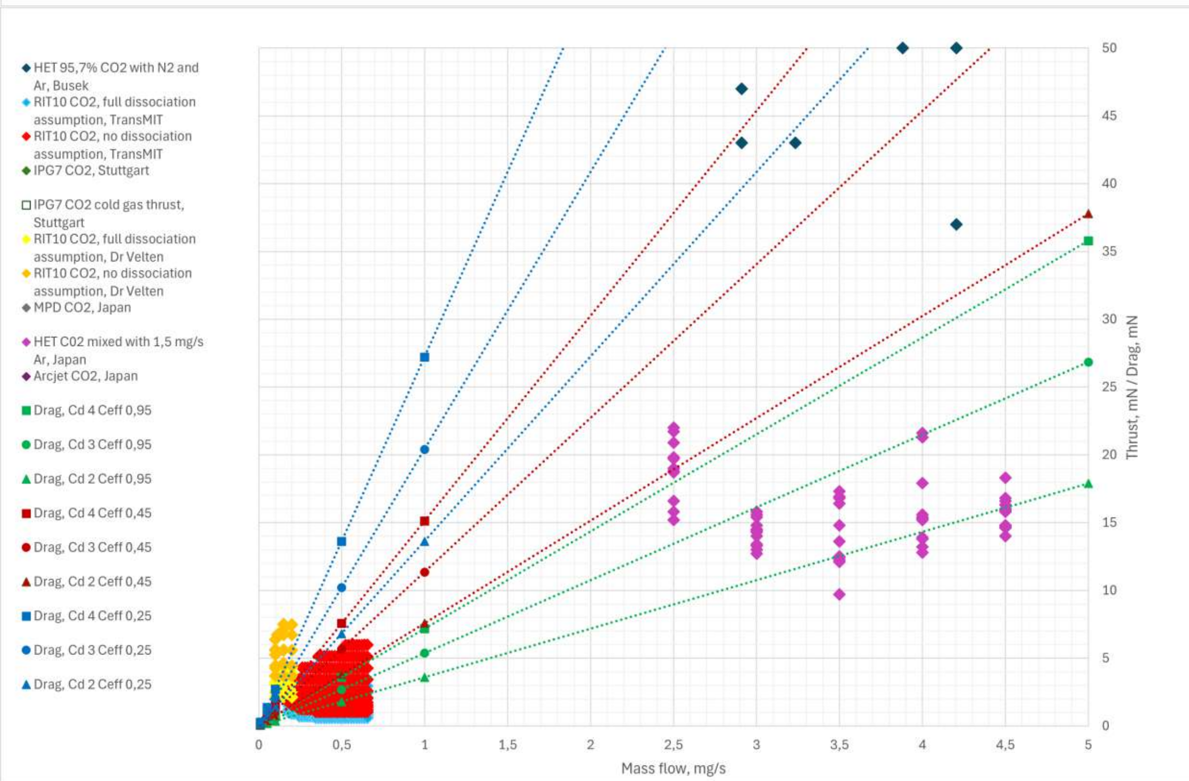
## Conclusion

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, electrical efficiency and thrust-to-power ratio 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 should be made.

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.

# Appendix


HET 95,7% CO2 with N2 and Ar, Busek
RIT10 CO2, full dissociation assumption, TransMIT
RIT10 CO2, no dissociation assumption, TransMIT
IPG7 CO2, Stuttgart
IPG7 CO2 cold gas thrust, Stuttgart
RIT10 CO2, full dissociation assumption, Dr Velten
RIT10 CO2, no dissociation assumption, Dr Velten
MPD CO2, Japan
HET C02 mixed with 1,5 mg/s Ar, Japan
Arcjet CO2, Japan
Drag, Cd 4 Ceff 0,95
Drag, Cd 3 Ceff 0,95
Drag, Cd 2 Ceff 0,95
Drag, Cd 4 Ceff 0,45
Drag, Cd 3 Ceff 0,45
Drag, Cd 2 Ceff 0,45
Drag, Cd 4 Ceff 0,25
Drag, Cd 3 Ceff 0,25
Drag, Cd 2 Ceff 0,25
Thrust, mN / Drag, mN
Mass flow, mg/s


HET 95,7% CO2 with N2 and Ar, Busek
RIT10 CO2, full dissociation assumption, TransMIT
RIT10 CO2, no dissociation assumption, TransMIT
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Drag, Cd 4 Ceff 0,95
Drag, Cd 3 Ceff 0,95
Drag, Cd 2 Ceff 0,95
Drag, Cd 4 Ceff 0,45
Drag, Cd 3 Ceff 0,45
Drag, Cd 2 Ceff 0,45
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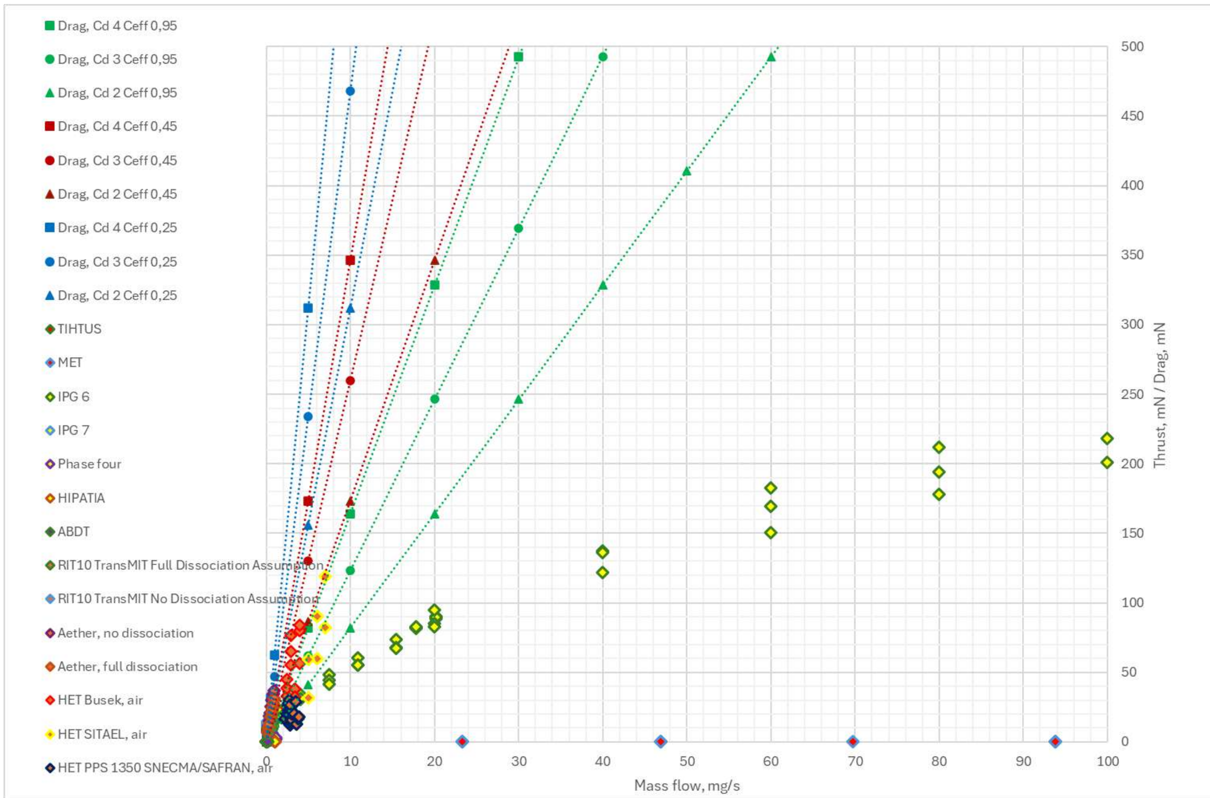
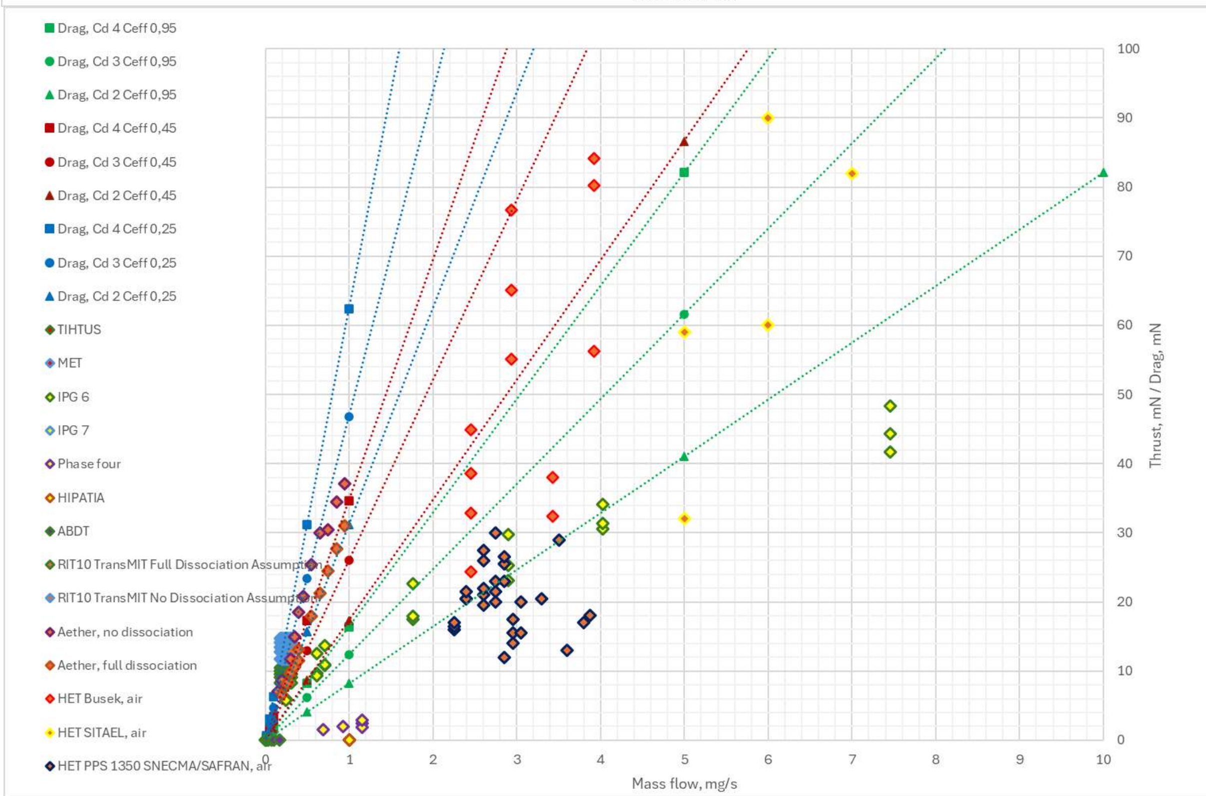
Drag, Cd 3 Ceff 0,25
Drag, Cd 2 Ceff 0,25
Thrust, mN / Drag, mN
Mass flow, mg/s



Drag, Cd 4 Ceff 0,95
Drag, Cd 3 Ceff 0,95
Drag, Cd 2 Ceff 0,95
Drag, Cd 4 Ceff 0,45
Drag, Cd 3 Ceff 0,45
Drag, Cd 2 Ceff 0,45
Drag, Cd 4 Ceff 0,25
Drag, Cd 3 Ceff 0,25
Drag, Cd 2 Ceff 0,25
TIHTUS
MET
IPG 6
IPG 7
Phase four
HIPATIA
ABDT
RIT10 TransMIT Full Dissociation Assumption
RIT10 TransMIT No Dissociation Assumption
Aether, no dissociation
Aether, full dissociation
HET Busek, air
HET SITAEL, air
HET PPS 1350 SNECMA/SAFRAN, air
Thrust, mN / Drag, mN
Mass flow, mg/s

Drag, Cd 4 Ceff 0,95
Drag, Cd 3 Ceff 0,95
Drag, Cd 2 Ceff 0,95
Drag, Cd 4 Ceff 0,45
Drag, Cd 3 Ceff 0,45
Drag, Cd 2 Ceff 0,45
Drag, Cd 4 Ceff 0,25
Drag, Cd 3 Ceff 0,25
Drag, Cd 2 Ceff 0,25
TIHTUS
MET
IPG 6
IPG 7
Phase four
HIPATIA
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RIT10 TransMIT Full Dissociation Assumption
RIT10 TransMIT No Dissociation Assumption
Aether, no dissociation
Aether, full dissociation
HET Busek, air
HET SITAEL, air
HET PPS 1350 SNECMA/SAFRAN, air
Thrust, mN / Drag, mN
Mass flow, mg/s

## Acknowledgments

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