D 4.1– Literature Review of ABEP Systems.

Project acronym:	DISCOVERER
Grant Agreement:	737183
Project Duration:	1 January 2017 – 31 March 2021 (51 months)
Issue	1
Version:	1
Revision	1
Date:	30/06/2017
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Due date of deliverable	30/06/2017
Actual submission date	11/07/2017

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1 Executive Summary

This report is realised within the context of the European-funded project DISCOVERER which aims to revolutionise Earth-observation (EO) satellites for sustained operation at very low Earth orbits (VLEO), much lower altitudes than the current state of the art. The report aims to support the development of system models for VLEO platforms by identifying the state of the art and trends in technology developments for atmosphere-breathing electric propulsion (ABEP) development. A literature review of ABEP system studies is presented, beginning with the concept of ABEP, its advantages and intended application. A general overview of each ABEP-related study is then given, supported by a representative mission in VLEO and corresponding spacecraft parameters. Furthermore, operating principles and pros and cons of intake designs are presented. Each study is presented in detail, providing most of the available data. The different electric propulsion technologies contained in ABEP studies are also presented individually with detailed data on thruster, tests, and performance. Finally, conclusions are drawn based on the overview, data from the studies, and the detailed analysis of intake and propulsion system performance. The main intake features needed for high performance are highlighted. Main features and issues of the different propulsion system are discussed as well as proposed future developments for the design of an efficient and long-lifetime ABEP system.

2 Introduction

2.1 Motivation

Very Low Earth Orbit (VLEO) platforms, defined as those that orbits at altitudes between 100 and 450 km, benefit from improved payload performance and geospatial accuracy, lower launch mass, simple End-of-Life disposal, and have reduced space-debris collision risk [1]. Earth based observation missions can benefit from the use of such orbits, for example in environmental monitoring; maritime surveillance; land management, precision agriculture and food security; disaster/emergency support; civil protection and damage assessment; and intelligence and security. For optical instruments, higher image resolution and accuracy can be achieved at lower altitudes. Furthermore, the power required for the transmission of data to Earth is lower compared to that of higher orbits. As a result, overall performance can be increased using the same equipment, or, alternatively, payload costs and launch mass can be reduced without sacrificing performance. However, spacecraft operating in VLEO are have challenged by increased atmospheric drag (Fig. 1), atomic oxygen erosion, spacecraft charging, and shorter communication windows with ground stations. Concerning aerodynamic drag, without further countermeasures, the orbit decays in a short period of time, ultimately resulting in an uncontrolled re-entry and loss of the spacecraft. To cope with the atmospheric drag and increase the total mission lifetime a propulsion system is required [2]. Because the propulsion system and its propellant add additional mass and, thus, limit performance regarding payload capacity or launch mass, it should be as light and efficient as possible, yet able to produce enough thrust to compensate the drag force. Electric propulsion is widely used for propulsion of satellites due to its high specific impulse and therefore efficient propellant utilisation. However, the amount of propellant carried on-board still limits the lifetime, as the propellant generally depletes before the deterioration of other main subsystems, such as solar arrays, thermal control subsystem, and power subsystem. A recent example of a VLEO mission, which

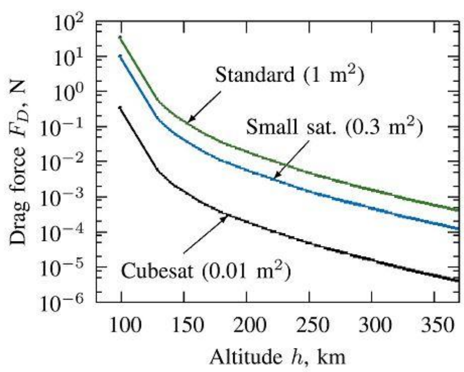


Fig. 1: Drag force over altitude for typical satellite sizes (cross sectional area) [17]

required drag compensation, was ESA's GOCE mission to map and analyse the Earth's surface and gravity field. GOCE required a very low Earth orbit to ensure sufficient precision and accuracy. The satellite had a launch mass of 1090 kg, a length of 5.3 m and a cross sectional area of 1.1 m^2. GOCE orbited at an altitude of between 240 km and 260 km and was equipped with two QinetiQ T5 Kaufman type ion thrusters (the second as a redundant backup), using Xe propellant and producing a variable thrust in the range of 1.5-20 mN [3]. Launched in March 2009, the satellite's propellant reserves were depleted in November 2013, causing it to deorbit and begin its uncontrolled re-entry, followed by a complete burn-up in the atmosphere. The total lifetime of GOCE was 56 months [4]. While lasting substantially longer than the designated lifetime of 20-30 months, depletion of propellant was the limiting factor, ultimately leaving the satellite unable to compensate drag.

2.2 Working Principle of ABEP

Atmosphere-Breathing Electric Propulsion (ABEP) is a promising strategy to efficiently compensate drag in orbit, enabling longer mission lifetime in VLEO and reducing propellant mass requirement. An ABEP system provides thrust to the spacecraft by collecting the incoming residual atmosphere at low altitude orbits using a specially designed intake, and utilising it as propellant for an electric thruster. The concept is to extend the spacecraft's lifetime by eliminating the need to carry propellant on-board. The concept is depicted in Fig. 2. Providing a virtually unlimited amount of propellant, the lifetime-limiting factor is no longer the amount of propellant carried on-board, but the durability of the other satellite's subsystems. Eliminating the need of carry propellant into orbit brings the benefit of reduced launch mass as long as the mass of the ABEP system is less than that of a conventional propulsion system (including the propellant). However, the intake mass is yet to be determined, but might increase as the design altitude reduces. As a result, it is possible to use less powerful launch vehicles, reducing costs. Alternatively, the payload mass can be increased. However, the primary driver is the mission requirement in terms of lifetime. For example, a 1-year mission might have lower mass with a conventional EP system (including tank and propellant) than with an ABEP system that is better suited to longer missions. An ABEP mission of 10-year lifetime could have reduced total cost and weight compared to three launches of 2-years missions each using conventional EP. Nevertheless, the ABEP system should be capable of drag compensation of a spacecraft at a specific altitude with the collected amount of propellant. Therefore, the design of intake and electric propulsion will define the effectiveness of an ABEP system.

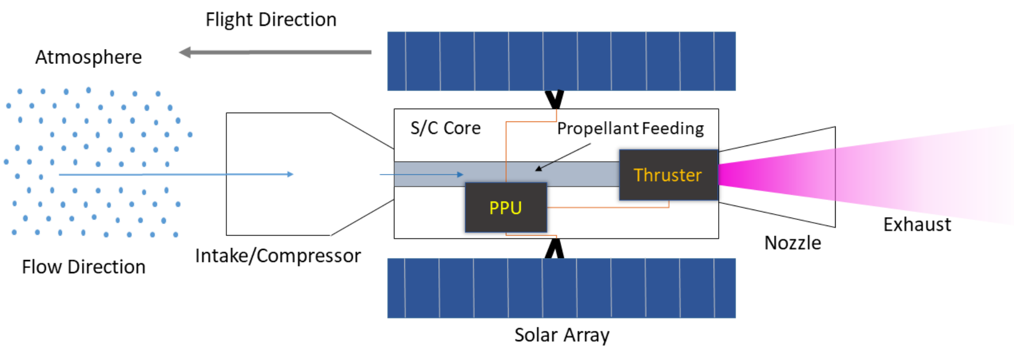


Fig. 2 ABEP concept

3 Literature Review of Existing ABEP Concepts

This chapter will present an overview of the ABEP-relevant studies. Most of the studies involve the use of current EP technology, such as Hall-effect thrusters (HETs), radiofrequency ion thrusters (RITs), gridded ion thrusters (GITs), and pulsed plasma thrusters (PPTs), and newer concepts such radiofrequency plasma thrusters (RPTs), Helicon Hall-effect thrusters (HHTs), and inductive plasma thrusters (IPTs). Before introducing the ABEP studies, a brief explanation of such propulsion systems is presented.

3.1 EP Systems

Electric propulsion (EP) systems can be divided into three subcategories, according to the acceleration strategy:

- Electrothermal: the electric power is used to heat up the propellant which expands through a nozzle,
- Electrostatic: the electric power is used to ionise the propellant and an electrostatic field is used to accelerate it,
- Electromagnetic: the electric power is used to ionise the propellant and an electromagnetic field is used to accelerate it

HETs use electrostatic potential to accelerate the ions. Electrons are provided by an externally located cathode, and trapped into the ring-shaped discharge channel by electromagnetic fields. Propellant injection happens via an annular manifold at the anode. The magnetic field slows down the electron axial mean velocity and forces them to drift around into the discharge channel. Ions are weakly influenced by magnetic fields and, therefore, are accelerated by the electrostatic field only.

A GIT is a design of ion thruster, where the propellant is injected into the discharge chamber and ionised by electron bombardment. Ions are extracted at high velocity by accelerating grids, and the flow is neutralised by a neutraliser mounted outside the thruster. The energetic electrons are produced in several ways:

- Emitted from a hollow cathode and accelerated to the anode (Kaufman type ion thruster)
- Propellant is ionised by electromagnetic fields generated by an RF-coil, without any cathode, and afterwards accelerated by the grids (radio frequency ion thruster RIT)
- Microwave heating

PPTs are devices in which the ionisation and acceleration of the propellant is done with pulses ignited by a spark plug. The ablated and ionised propellant forms a conductive plasma sheet between the anode and cathode. This circuit forms a current loop that creates a magnetic field. The magnetic field perpendicular to the plasma current leads to a Lorentz force accelerating the plasma. Hence, an impulse is achieved by the thruster. One pulse ends when the entire energy stored in the capacitor is discharged.

In an IPT the gas is injected into a discharge channel. A RF-fed coil is wrapped around the discharge channel. The gas is ionised by the oscillating E and B-field generated by the coil. As the ionisation is done electrodeless any gas can be used, even the most aggressive ones. Moreover, the plasma leaving the thruster is neutral eliminating the need of a neutraliser. The RPT is a variation of the IPT concept, equipped with an external magnetic field, and it is especially designed to excite helicon waves within the plasma.

A HHT is a variation of the HET, design to improve its total efficiency and to cope with atmospheric propellant [5]. It is a two-stage HET design to utilise the efficient ionisation of a helicon source with the acceleration mechanism of a HET. Helicon waves are excited in the upstream section by a RF-fed antenna and an axial magnetic field, to increase ionisation efficiency and propellant utilisation efficiency. The second stage operates exactly as a HET explained above.

Considering applications of ABEP, the requirements for the propulsion system are to cope with low mass flow and/or low pressure for ignition. Otherwise, the intake shall be able to provide enough compression and/or mass flow for the thruster operation. Earth atmosphere in VLEO is composed of a high presence of atomic oxygen, known to be very aggressive towards spacecraft surface materials.

This means that the materials of the intake have to cope with possible erosion, and the components of the thruster that are in direct contact with the plasma, such as electrodes and/or grids have to be resistant enough to guarantee the lifetime of the system. Density in the atmosphere varies with the solar activity and the position in the orbit. The thruster therefore has to cope with variation in propellant composition, pressure, and mass flow. While devices such as RITs, HETs, and GITs have a strong background of technology development, they might suffer from the previously mentioned issues. These could be eliminated or, at least reduced by the use of electrodeless devices such as IPTs and RPTs.

3.2 ABIE (JAXA)

One of the most advanced ABEP concepts is the Air-Breathing Ion Engine (ABIE), developed by Nishiyama, Fujita, and Hisamoto of the Japan Aerospace Exploration Agency (JAXA) [6]. The concept consists of an ECR (Electron Cyclotron Resonance) ion thruster. This type of thruster uses a microwave emitting antenna to generate an electric field and magnets to produce a strong magnetic field. The electric and magnetic fields ionise the propellant, which is then accelerated by a set of grids and finally neutralised after leaving the thruster. A ring-shaped intake, surrounding the spacecraft

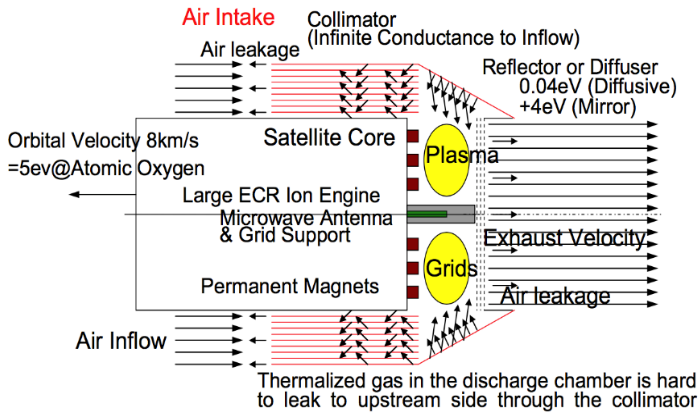


Fig. 3 ABIE concept [21]

core, is designed to provide a sufficient pressure to the thruster, estimated to be around 0.5 Pa. Reflectors and a honeycomb structure of small ducts array at the intake entrance prevent the particles from exiting the intake in upstream direction, see Fig. 3. Designed for altitudes between 150 and 200 km, it achieves a thrust density (thrust-to-power-ratio) of 10-13.7 mN/kW. With the assumption of a frontal cross-sectional area of 1.5 m² and a drag coefficient of C_d=2, the power requirement for full drag compensation at an altitude of 170 km is between 4-5.59 kW. Higher power requirements need bigger solar panels area therefore an increased aerodynamic drag. The feasibility of very large solar arrays in VLEO need to be assessed. Nishiyama [6] and Hisamoto [7],[8] propose the improvement of the ion source and intake by increasing its resistance to corrosion by atomic oxygen, and the improvement of the intake design, based on DSMC (Direct Simulation Monte Carlo) and experimental results. Implementation of these measures would increase lifetime and thrust density, thus improving overall performance significantly.

3.3 ESA RAM-EP

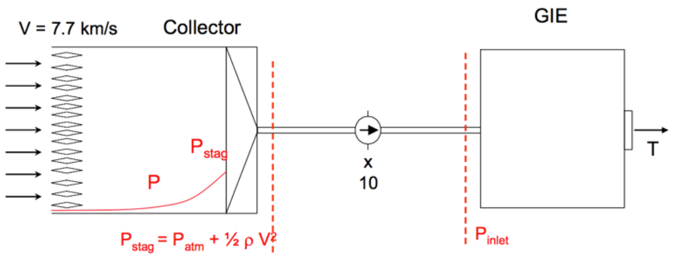


Fig. 4 ESA RAM-EP concept [9]

A different approach was taken by ESA, presenting the RAM-EP Concept [9]. A medium-sized satellite was proposed with an estimated drag coefficient of C_d=2, a cross sectional area of 1 m² and a mass of 1000 kg. This concept differs from ABIE in that the intake and thruster are physically separated. The RAM-EP is equipped with four RIT-10 Radio-Frequency Ion Thrusters to provide a combined thrust in the range of 2-20 mN whilst the maximum power availability for propulsion is set to 1000 W. The spacecraft is designed for a sun-synchronous orbit (SSO) at an altitude range of 180-250 km and has an estimated lifetime of 3-8 years. According to the study, at altitudes of more than 250 km, conventional electric propulsion systems become more competitive, because of the lower drag and therefore lower propellant requirement, and because of the even more rarefied atmosphere that would deliver less mass flow to the RIT propulsion system. The intake is composed of a cylindrical array of diffusor baffles, see Fig. 4, rather than a honeycomb structure of small ducts to increase pressure and prevent backflow. Calculations showed that maximum performance, i.e. an inlet pressure of 10^{-3} Pa, can be achieved with a length of 1 m and a frontal area of 0.6 m² at a non-specified altitude of between 200 and 250 km.

3.4 Air-Breathing Cylindrical Hall-effect Thruster (ABCHT)

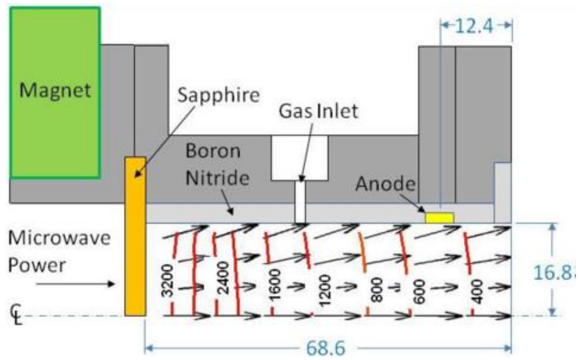


Fig. 5 Air-Breathing Cylindrical Hall-effect Thruster (ABCHT) [10]

The ABCHT (Air-Breathing Cylindrical Hall Thruster) concept proposed by Diamant [10], [11] is a 2-stage design consisting of an ECR ionisation stage, which ionises the atmospheric particles using electric and magnetic fields, and a cylindrical Hall-effect thruster as shown in Fig. 5. Although capable of using air as propellant, tests were only conducted with Xe. The concept satellite used in these studies has a frontal area of 0.5 m², a drag coefficient of C_d=2.2, and an estimated collection efficiency, the ratio between amount of particles delivered to the thruster to that of the encountered ones, of 0.35. With an approximate power availability of 1 kW, altitudes of 220 km and higher are shown to become accessible in terms of full drag compensation. To achieve the required pressure of 0.01 Pa at the thruster, an inlet with a compression ratio of 500:1 is necessary, however, this was not covered in the reports. Another anticipated problem is oxidation, mainly of the cathode, which is addressed in [11]. It is concluded that the use of a high-temperature cathode exposed to oxidising substances may restrict lifetime. A further issue is the inability to compress the incoming air to the degree necessary for operation of the cathode. Utilisation of an ECR cathode running on Ar or Xe is therefore proposed, which ionises the gas. This means that a small amount of propellant has to be carried on board. In case of Xe usage, 8 kg of gas are required for a lifetime of 5 years.

3.5 Martian Atmosphere Hall-Effect Thruster (MAHBET), BUSEK

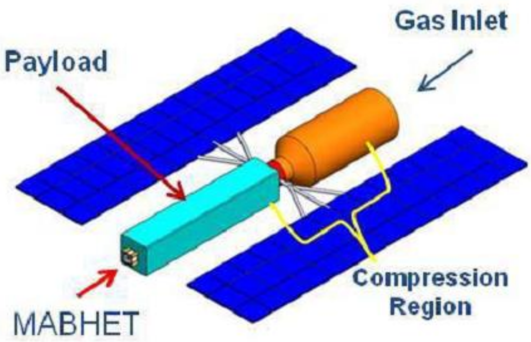


Fig. 6 MAHBET BUSEK concept [12]

The MABHET (Martian Atmosphere-Breathing Hall-Effect Thruster) concept by BUSEK [12] is an ABEP system designed for small spacecraft orbiting in the Martian atmosphere. This concept consists of an intake, a Hall-effect thruster, and solar arrays as main power source, supported by batteries to cope with eclipses and power peaks. The concept is shown in Fig. 6. The thruster has been operated with a gas mixture similar to the Martian atmosphere (95.7% CO_2, 2.7% N_2 and 1.6% Ar), achieving a thrust density of 19-33 mN/kW, depending on the propellant mass flow. The intake is a cylinder measuring 3.7 m in length and 0.6 m in diameter. The intake allows a compression ratio of up to 100 and provides a collection efficiency of η_c = 35%. Assuming a drag coefficient of C_d=3, 1.2 kW of power is estimated to be sufficient for full drag compensation at altitudes between 150-180 km, which proves the general feasibility of this concept. According to [13], a maiden flight is planned for 2025.

3.6 RF Plasma Thruster (Shabshelowitz)

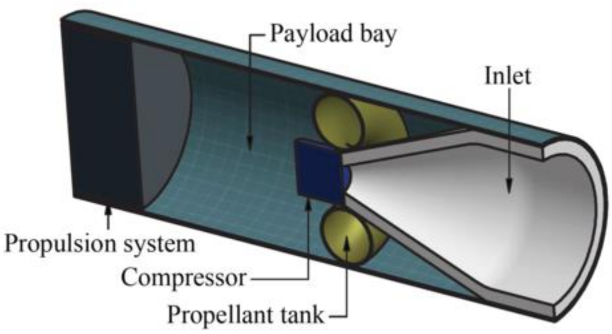


Fig. 7 RF Plasma Thruster Concept [5]

Shabshelowitz [18] examines the feasibility and efficiency of RF plasma technology regarding ABEP. The comparison of two thruster types, an RF plasma thruster (RPT) and a Helicon Hall Thruster (HHT), revealed that the utilisation of atmospheric gases, such as N_2 and O_2 leads to a lower efficiency of the HHT, in addition to the vulnerability regarding oxidation. The RPT produces less thrust operating with air and N_2 than with Ar. The calculations and experiments are based on a concept spacecraft with a mass of 325 kg, an outer diameter of 0.7 m and an inlet area ratio (A_{drag}/A_{inlet}) of 0.5, which can be seen in Fig. 7. η_c is assumed to be 90%. The required thrust at an orbit altitude of 200 km is estimated at 8.8 mNto achieve a mission lifetime of 3 years. To produce this amount of thrust using Ar, the RPT requires 1000-1500 W of power depending on mass flow. With air, however, no thrust could be measured. The HHT requires 306 W of power, and was tested with N_2. It is concluded that further work is needed, mainly to improve the design of several components, such as antennas, and developing a microwave cathode, which is capable of operating with air. A further issue is the need to improve understanding of plasma acceleration to increase thruster performance.

3.7 Air-Breathing Hall-Effect Thruster

Instead of experimental testing, Garrigues [13] took a theoretical approach. Proposing the use of an air-breathing Hall-effect thruster, a numerical simulation of the motion and ionisation of the propellant's particles was performed. The concept spacecraft has an orbit at an altitude of 250 km and has a frontal area of 1 m² with a drag coefficient of C_d=2. The maximum required thrust of 20 mN was determined to fully compensate the drag. The ideal discharge channel length for operation with air is determined to be 5 cm. Based on these results, a minimum mass flow of 2.5-3.5 mg/s was calculated to fully compensate the drag at the proposed altitude. In this case, the required power is 1 kW. Because of the already extensive coverage in other literature, the study disregards several problems, such as cathode oxidation and the required temporary storage of propellant. Apart from these uncertainties, operation of the thruster with air seems possible. However, in another study [14] it is concluded that the use of a Hall-effect thruster is questionable due to the low mass flow and insufficient density achievable by the proposed intake. This would require storing large amounts of ambient air which is currently not feasible.

3.8 RIT-10 (Cifali et al.)

In addition to the experiments with the PPS 1350, Cifali et al. [15] tested a RIT-10-EBB thruster, see Fig. 8, with O_2 and N_2 as propellant. The maximum achieved thrust operating with O_2 was 6 mN, higher than the thrust measured operating with N_2 (5.25 mN) as propellant. Both had a power consumption of 450 W. After a 10-hour-long test the thruster showed no signs of erosion operating with N_2, whilst O_2 noticeable erosion of the accelerator grid. In both cases performance was lower than with Xe but, nevertheless, future operation with air seemed possible and further testing was announced. In a subsequent study [16], an endurance test of 500 h with a N_2-O_2 mixture was performed with some additional modification. While the performance did not change much, the erosion was reduced. Despite several problems with the testing facility, results showed that continuous operation and higher lifetime would be possible.

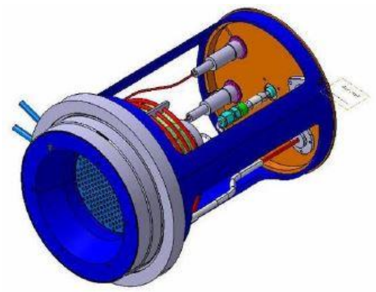


Fig. 8 RIT-10-EBB [15]

3.9 Air-Breathing PPT

A new concept is proposed by Schönherr et al. [17], [18] by performing theoretical simulations on an Air-Breathing Pulsed Plasma Thruster (PPT) and Gaseous Pulsed Plasma Thruster (GPPT). It is designated for a satellite mission at an altitude of 200 km. The satellite has a cross-sectional area of 0.3 m² and a drag coefficient of C_d=2.2, requiring a drag force of approximately 5 mN. Analysing simulations of different types of intakes at this altitude, it is expected that 1 mPa of pressure at the thruster entrance can be provided. Calculations showed an estimated I_{sp} of 5000 s and a power-to-thrust ratio of approximately 30 mN/kW. Furthermore, it is possible to adjust various parameters, such as the pulse frequency, mass injection per shot, and power consumption. This gives the flexibility to operate at different power levels, which can be necessary during eclipses, without altering the performance too much. The required mass flow for drag compensation at 220 km of altitude is < 0.1 mg/s, although this is not experimentally confirmed. However, because of possible fluctuations of mass flow, storage of propellant is required. Furthermore, the thruster's flexibility is limited at high frequencies, as valves used for injection of the gas are restricted in terms of minimum opening times. Full drag compensation could not be guaranteed, but at least partial compensation is expected to be possible in altitudes from 100-250 km.

3.10 SITAEL RAM-HET

SITAEL [19] proposes a concept based on an integrated RAM-EP concept including a Hall effect thruster (HET). The system is designed for altitudes below 200 km and a lifetime of more than 4 years. The specially designed intake, as shown in Fig. 9, is built with a cylindrical array of ducts in a split-ring configuration, and a conical convergent part at the back, which condenses the mass flow. The thruster is a HET with separate stages for ionisation and acceleration. The intake ducts have a length of 0.3 m and a frontal area of 0.12 m². The design altitude is 200 km and tests will be conducted with a corresponding mixture of O_2 and N_2. Ignition will be attempted with Xe or Kr, and in the case of an insufficient mass flow will be assisted by a gas distributor downstream of the intake. The flow generator will be a 5 kW class Hall-effect thruster. However, as of June 2017, experimental results are not yet available. Calculations and simulations regarding the intake were performed in [20]. It was concluded, that, depending on the aspect ratio of the intake, a trade-off between transmission probability and pressure ratio is necessary. The system needs to be adjusted individually to external conditions, such as required mass flow and pressure for the thruster or mission-related aspects, such as altitude of the target orbit or expected drag forces.

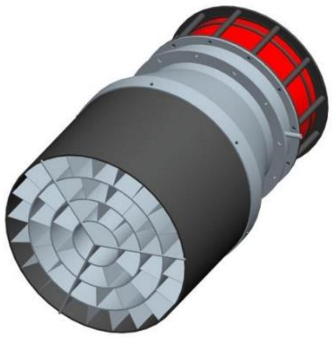


Fig. 9 RAM-HET concept [19]

4 ABEP Intakes

The intake is an essential part of an ABEP system. It has to provide the necessary mass flow and pressure to the electric thruster. However, the design requirements are significantly different to conventional intakes (e.g. jet engines) because ABEP systems operate at altitudes where the density is very low. The flow in VLEO can be assumed as free molecular, which means it behaves as singular particles traveling through free space. Along their trajectory towards the surfaces ("wall") of the intake, they do not interact with each other. During wall reflections, one common assumption is full accommodation, in which they lose their macroscopic velocity and are scattered into a random direction within the half space with solid angle 2π. The main possibility for a reflected particle to reach the thruster is given by a scattering into the solid angle Ω which is much smaller than the whole range of 2π, see Fig. 10. A simple convergent conical intake would be inefficient because the most of the particles entering the thruster would be only those inside the extruded cross section of the thruster. Thus, the aim of designing an intake to provide a sufficient pressure and mass flow for the thruster, results in the necessity to prevent the reflected particles from exiting the intake, but without blocking the incoming particles. Most of the previously presented concepts solved this problem in a similar way, which will be presented in more detail in the following section.

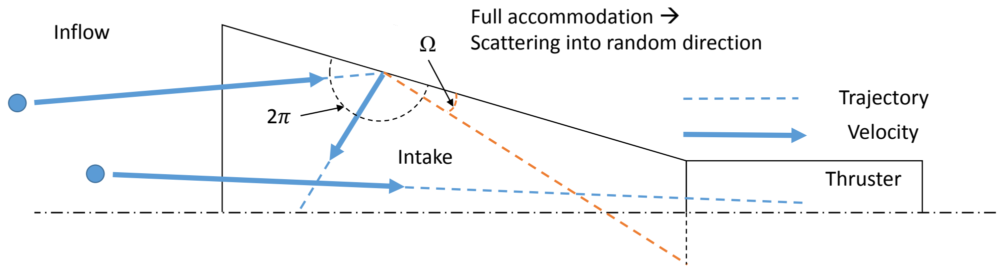
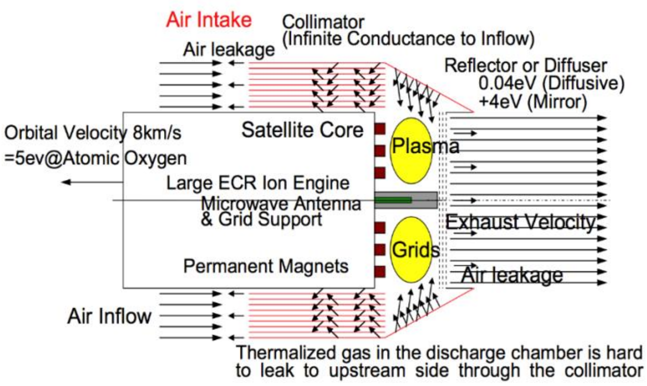


Fig. 10 Simple cone intake in free molecular flow

4.1 Intake Designs

The intake of ABIE [21] is fully integrated with the thruster. The intake section is ring-shaped, around the satellite payload ("core"). To prevent the particles from escaping the intake, an array of long, narrow ducts is mounted at the front, shown in Fig. 11, to minimise the backflow. Particles reach the back of the intake hitting a 45° surface, the diffuser/reflector, and are afterwards scattered on the back of the satellite core, mostly towards the thruster's acceleration grids. An Electron Cyclotron Resonance (ECR) device ionises the particles in the ionisation chamber, which is the region on the back of the satellite core. Acceleration grids extract the plasma and expel it at high velocity to produce thrust. The frontal area, excluding the intake, is 1.5 m² and the intake area is 0.48 m². At the thruster entrance, a pressure of 0.5 Pa is achieved [7], corresponding to a compression ratio of 180. In [21], further simulations (DSMC) regarding the intake performance have been conducted. The simulations show an inverse proportionality between the compression ratio and the intake efficiency, depending on the transmittance of the intake, as shown in Fig. 11.



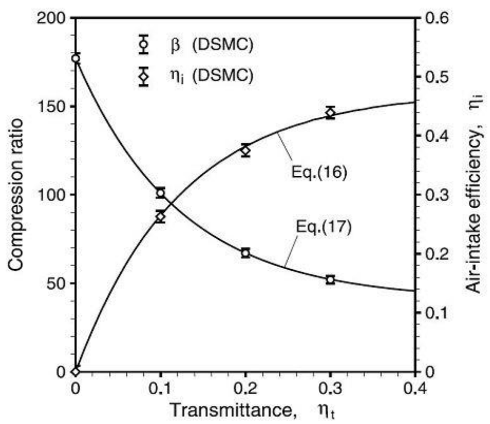


Fig. 11 Left: ABIE concept, Right: compression ratio and intake efficiency of ABIE [21]

The concept by ESA [9] consists of a cylindrical intake with a convergent section at the end. In this case, intake and thruster are two separated devices. Instead of cylindrical ducts ESA proposes an array of diffuser baffles at the front to prevent backflow, see Fig. 12. Based on simulations (DSMC), an intake length of 1 m and a frontal area of 0.6 m² were proposed in order to get the required steady level of pressure at the end of the collection chamber. At the thruster inlet, the intake is designed to provide a pressure of 10^{-3} Pa.

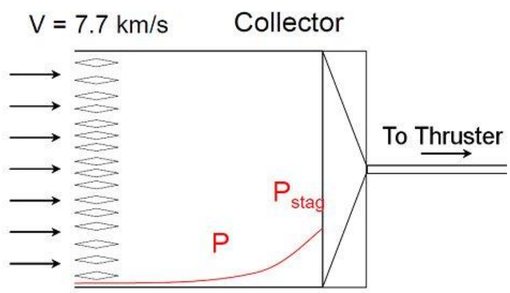
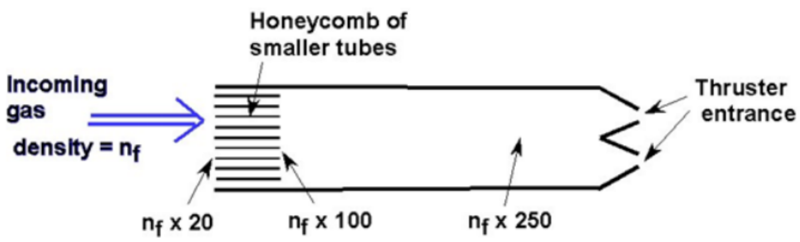


Fig. 12: ESA RAM-EP intake (based on [9])

Fig. 13: MABHET intake concept [24]



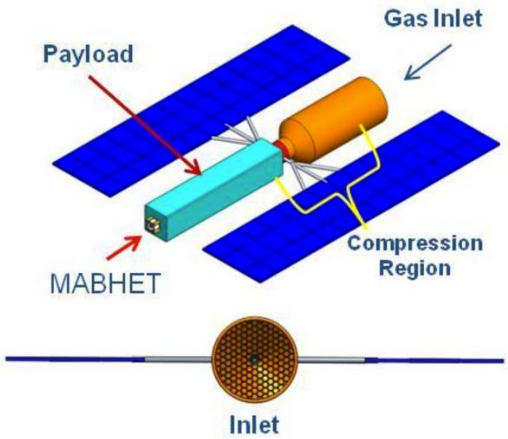


Fig. 14: MABHET, intake detail [12]

Unlike the other concepts, MABHET [12] is designed to operate in the Martian atmosphere instead of the Earth atmosphere, see Figs. 13 and 14. Compared to the Earth atmosphere, Mars has lower pressure and density, as well as a different composition, 95.7 % of the atmosphere is CO_2. The proposed intake consists of a long tube with a conical end. The intake is a cylinder measuring 3.7 m in length and 0.6 m in diameter, with an exit diameter of 0.14 m, resulting in an inlet area of 0.15 m² for a total cross sectional area of 0.3 m². At the front of the intake, a honeycomb structure of small ducts is installed. The long main tube has the same transmittance behavior as multiple small tubes but, due to the higher Knudsen number, compressing collision effects might occur that increase the pressure at the thruster inlet even more (“collision cascade”). A compression ratio of around 100:1 was predicted in simulations (DSMC), while achieving a collection efficiency of 0.35. Its design has better performance in the atmosphere of Mars than Earth.

The intake from SITAEL RAM-HET [19], as shown in Fig. 15, is shaped as a tube with a conical, convergent part connecting it to the thruster. It has an array of ducts as well, however, these are arranged in four concentric rings which are split regularly. This has the advantage of a simply achievable constant cross sectional area over the complete cylindrical part. The ducts are 0.3 m in length, and the frontal area is 0.12 m². In [20], simulations and calculations regarding the performance of the intake were conducted. Calculations show that the collectable fraction of the inflow has to be traded off with compression ratio. This leads to an optimal aspect ratio of the intake for a given effective, total transmission probability, shown in Fig. 17a. Furthermore, simulations were performed for the intake, including the honeycomb duct array in order to validate the calculations. The results clearly show the tradeoff between collection efficiency and compression ratio, which can be seen in Fig. 17b. In conclusion it is proposed, that the collection efficiency should be in the range of 0.25-0.4 with an aspect ratio of 5-10. Further simulations examined the effect of the intake axis and the velocity vector not being parallel. Results showed a negligible effect for an angle of up to 2° for aspect ratios up to 8. Regarding the duct geometry, a multiple-duct intake is recommended, such as the one shown in Fig. 15.

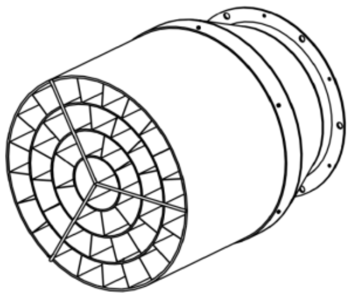


Fig. 15: RAM-HET intake [19]

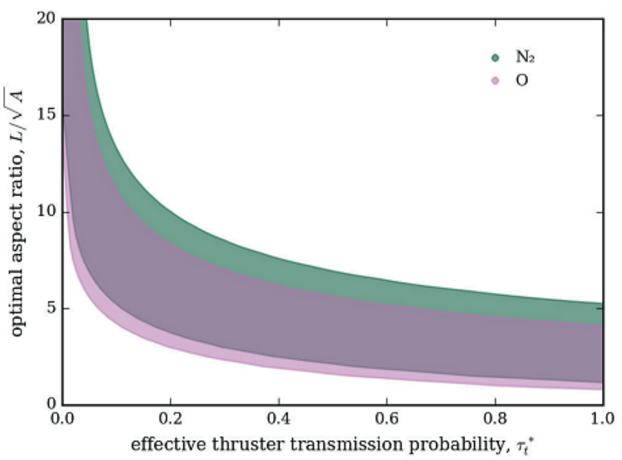


Fig. 17a: Optimal intake aspect ratio over effective, total transmission probability for different gases

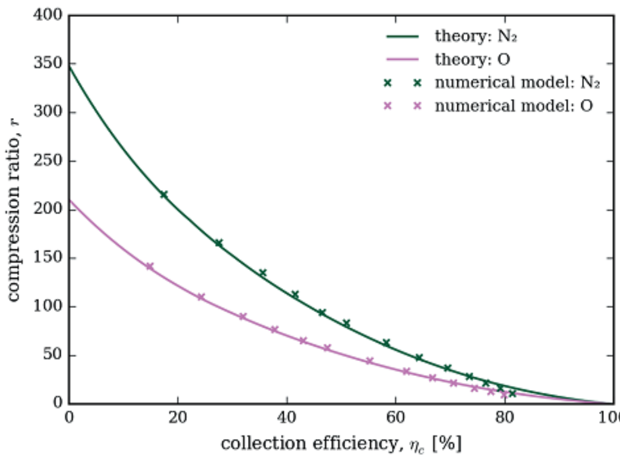


Fig. 17b: Intake compression ratio over collection efficiency for different gases and methods [20]

4.2 Vacuum Air-Intake, Active Compression

Li et al. [22] investigate the possibility of using an active intake, which consists of several vacuum pumps, at altitude ranges of 150-240 km. The proposed structure is shown in Fig. 18.

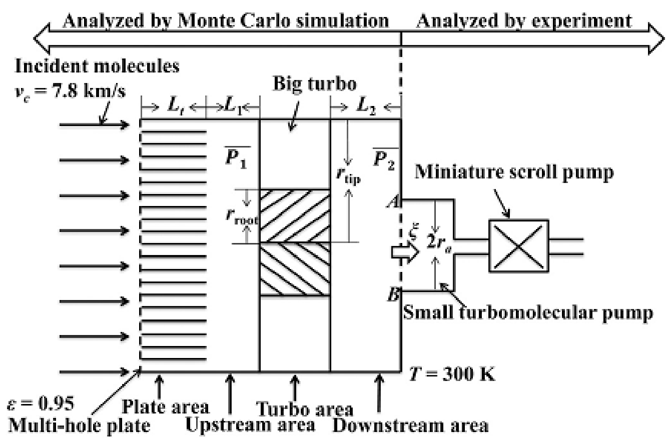


Fig. 18 Active vacuum intake [22]

After the particles pass the multi-hole plate, with a diameter of 0.5 m that prevents backflow of the atmospheric gases, they are compressed by a turbo, a small turbomolecular pump, and a miniature scroll pump. The performance of the first turbo is evaluated using TPMC (Test Particle Monte Carlo) and DSMC (Direct Simulation Monte Carlo) simulation methods, while the other pumps are experimentally tested. Depending on the pumping speeds, collection efficiencies between 0.417-0.579 could be achieved in the simulations. Including the compression effect of the multi-hole plate, the turbo provided compression of the atmospheric gases to a pressure of 30.2-532 mPa, depending on altitude, while consuming 0.848-52.627 W of power. This corresponds to a calculated compression ratioof 2918-3864. The gas flow was 3.23-65.57 sccm of air, assuming a collection efficiency of 0.5. Through the other pumps, the flow was further compressed to atmospheric pressure with a total power of 27.1-150.3 W at gas flows of 0-50 sccm, respectively. These results show that a significant compression is possible with a total power consumption of < 200 W. Obviously, an active system adds complexity and mass. The two smaller pumps have a combined weight of 6.5 kg, excluding the turbo and other required components. Additionally, an active system increases the risk of failure, therefore a failsafe design is required.

4.3 Conclusion and Further Work at IRS

Further analysis of intake designs and their performance was done by IRS [23]–[25]. Intakes from ESA [9], BUSEK MABHET [12] and ABIE (JAXA) [7] were reviewed, and the last two analysed with DSMC for verification. Furthermore, a model based on simple algebraic relations was derived and verified by corresponding simulations. It is assumed that a generic "baseline" ABEP-intake assuming fully diffuse wall reflections can be divided into an inlet section and an adjacent, fully thermalised reservoir condition as outflow to the thruster. By balancing all mass flows with their individual transmission probabilities, the description of the system is fully closed. A transmission refers to a specific direction through a single structure and is the ratio between particles entering the entrance plane and the gas particles leaving the exit plane (besides passing directly they can be scattered by which they might return back to the entrance), similar to the collection efficiency of the whole intake. The common feature of the inlet section of all passive intake designs described above, is the idea of a "molecular trap" based on the transmittance characteristic for a flow of large speed ratio through an effectively long geometry, e.g. a tube or thin ring, with diffuse scattering afterwards. The speed ratio S is defined as macroscopic to thermal velocity and is typically on the order of 10 for free stream flow in the context of ABEP systems. Simulations show that the transmittances depend only on S and the geometry, therefore the geometry can be optimised for the respective inflow conditions [25].

Another study [24] applied this balance model to a geometry based on the IPG6 discharge channel dimensions with $A_{out}=1.075\times10^{-3}$ m². Calculations were done for an altitude of 140 km in Earth atmosphere and 110 km in Mars atmosphere. Again, it was shown that an increasing area ratio A_{in}/A_{out} raises the collected mass flow, but reduces collection efficiency, which is equivalent to the thrust-to-drag-ratio of the spacecraft when considering only the drag on A_{in} and constant inflow conditions and thruster operation. However, the mass flow gain decreases for raising area ratio, leading to the conclusion, that above a value of area ratio in the order of 10 or at most 100, the increase of drag and loss of efficiency outweighs the gain of mass flow. It therefore recommended not to exceed this value. The described baseline intake with a small IPT therefore prefers implementation on small satellites (i.e. on a CubeSat) or as a clustered design.

5 ABEP Thruster

5.1 RPT/HHT (Shabshelowitz)

Shabshelowitz [18] examines two thruster types, a RF plasma thruster (RPT) and a Helicon Hall Thruster (HHT). The RPT is a propulsion device designed to excite a helicon wave in an axial magnetic field as shown in Fig. 19. The discharge channel of the RPT was designed such that in an ABEP configuration it would achieve $p \sim 0.1$ Pa and the frequency of the power supply was $f = 13.56$ MHz. The discharge channel is made of quartz and has an internal diameter of 90 mm and a length of 381 mm. Antenna is a Twisted Nagoya type. The RPT tested with N_2 and air showed very low thrust values, as well as ion currents suggesting that less ions are produced.

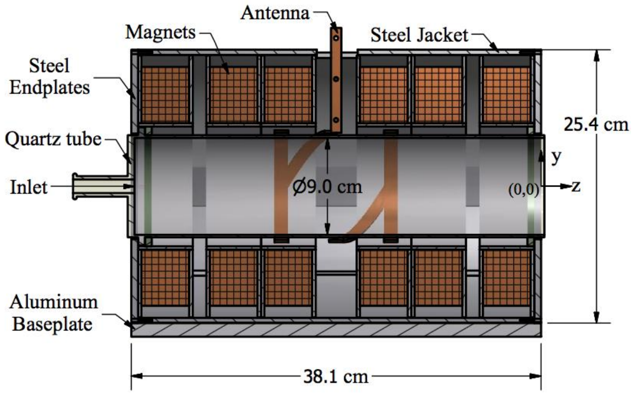
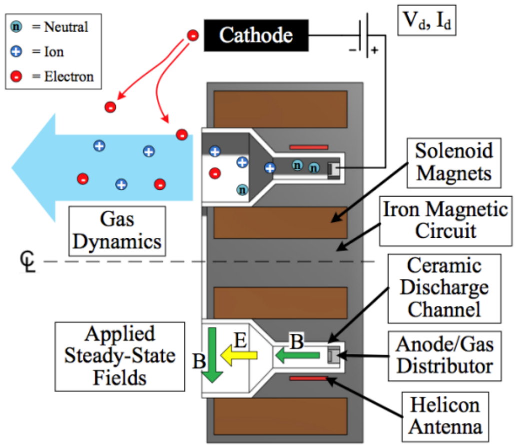


Fig. 19 RF Plasma Thruster (RPT)

The HHT is a two-stage Hall-effect thruster concept designed to utilise the efficient ionisation of a helicon source with the acceleration mechanism of a Hall-effect thruster. It is designed to excite annular helicon wave modes in the upstream part of the HET discharge chamber and can be seen in Fig. 20. In the first stage, RF power is applied to the helicon antenna to ionise the propellant leaving the gas distributor. An axial magnetic field is applied to the antenna to couple the annular helicon



waves into the just ionised propellant. In the second stage, a radial B-field and a voltage between the anode and an external hollow cathode generate the Hall-effect acceleration. Electrons emitted from the hollow cathode neutralise the ion beam leaving the discharge channel. The HHT was tested with N_2 with and without the RF stage in operation. Results without the RF stage showed that N_2 propellant had a lower I_{sp} and thrust to power ratio than operation with Xe. Thrust was about 120 mN at P < 302 W. With the RF stage, no thrust improvement was observed with N_2 as propellant. The study finally suggests that RPT is a promising technology, especially due to lack of a cathode, however, the low efficiency and thrust requires that the physics involved into the thrust production has to be better investigated. As HET operating on atmospheric propellant is known to suffer of lower performance, it was experimentally tested for HHTs whether an RF stage could somehow boost the overall thruster performance. However, this was not the case. Moreover, for long-term ABEP missions it is suggested that investigation of the lifetime of HET when using atmospheric propellant is performed due to the erosion of atomic oxygen.

Fig. 20 HHT Helicon Hall-effect Thruster

5.2 ABIE (JAXA)

In the concepts from [6], [21], [26] an intake integrated with the thruster is proposed. The atmospheric propellant is ionised through microwave ECR in an ionisation chamber. Propellant is extracted and accelerated by accelerating grids. No tests were performed of the thruster only, but the issue of erosion due to atomic oxygen is assessed by suggesting the application of a ceramic coating for the parts that do not require electrical conductivity, and gold coating for those that require it.

5.3 RIT

The presented studies from ESA [9] and Cifali et al. [15] evaluated the use of RIT for ABEP applications. ESA chose the RIT-10 because of the input pressure requirements, which could be theoretically reached by their collector design, but could not assess its performance with the atmospheric propellants O_2 and N_2 as the study was theoretical, utilising simulations. Cifali et al. [15] therefore conducted tests with atmospheric propellant mixtures in 2010. RIT-10 has been operated with N_2/O_2, but ignition difficulties require Ar. The RIT-10-EBB version was used for atmospheric propellant testing. The difference in this thruster from the standard RIT-10-ARTEMIS include water cooling, a movable acceleration system, and an easily interchangeable discharge chamber. Operation with N_2 required higher power compared to Xe operation, but showed no erosion of the grids after 10 hours continuous operation. Operation with O_2 also required higher power compared to Xe operation, whilst erosion of the graphite grid was highly noticeable at the end of the 10-hour test. In 2012 the same group tested the RIT-10-EBBM by applying a mixture of 1.27 N_2+O_2 as representative condition at 200 km altitude [16]. To cope with oxygen, the accelerating grid made of graphite was substituted by one made of titanium; the thruster/RF generator supply was also updated. A thrust between 2.25-7.25 mN was calculated for a total power between 180-520 W. Lower erosion was detected during a 500 h test, however issues such as damaging of the beam target and power outages did not allow the test to be performed continuously [16]. Finally, the RIT-10 could operate with such mixture, however the lifetime was strongly affected by corrosion/erosion phenomena due to the presence of atomic oxygen. It was stated that with appropriate materials, the lifetime could be increased to 1000 – 10000 hour range, corresponding to maximum operations of about 1 year and 52 days [27].

5.4 HET

The use of a Hall-effect thruster has been considered by the studies [10], [13], [14], [12], [15], [16], [19] for Mars application. These would be preferred to RIT because of simpler construction, more compact form, and most surfaces in contact with the plasma are ceramics and therefore resistant to the aggressive oxygen [11].

In [13], [14] calculations were performed for the design of an Hall-effect thruster to work with atmospheric propellant, in particular N_2 and O in Earth orbit. The conclusion is that a modification of the geometry of the discharge channel is needed for efficient ionisation of the atmospheric propellant. At 250 km of altitude, 3 mg/s of total mass flow applied to a HET with a thrust efficiency of 0.1, and a required power of 1 kW, fulfils the requirements for an ABEP mission for full drag compensation.

[10], [11] consider first the use of a microwave cathode as a solution to the problem of ion beam neutralisation on long duration missions using ABEP. Later the team studied a 2-stage cylindrical HET for ABEP applications, the ABCHET. An ECR is used for ionisation for lower operating pressure and greater resistance to aggressive gases (ABEP). Ionisation was achieved but the thrust efficiency was lower.

HETs have been tested by Cifali et al. in [15], [16] with the PPS1350-TSD thruster. The propellant mixture was 1.27 N_2+O_2, comparable to the atmosphere composition at 200 km altitude in Earth orbit. Ignition was performed with 100 % Xe and successively shifted to 100 % N_2 (or to the previously mentioned mixture) at the anode, while the cathode always ran on Xe only. Maximum applied power was of 1.4 kW at 350 V. Tests with pure N_2 showed a higher power requirement than with Xe and provided thrust between 19-21 mN at 1 kW during a 10 hour test. With the mixture, ionisation efficiency is lower than with Xe but equal or higher than pure N_2. Discharge is more efficient with O_2 than with N_2 and can be stable at lower mass flow rates. A higher thrust of 24 mN was achieved. Concerning erosion, the anode appeared rusty after operation, while the ceramics of the discharge channel also showed signs of operation with oxygen. The anode oxidation with an increased electrical resistance was the main concern. In the second study, the test was repeated for 500 h endurance. Before the test the oxidation layer of the anode was removed rather than changing it to a titanium one due to budget constraints. , The gas mixture also included an additional 10 % of Xe. The addition of Xe increased the thrust from 4 % to 40 % and the discharge current from 17 % to 40 %. The total test time was 390 h. It needed a stop and restart at 314.5 h to clean again the anode from the oxidation layer. At 390 h the test had to stop due to the damage of the anode due to severe oxidation. The oxidation also caused flames to appear during test. The thrust was measured at 22-23 mN and at the end of the endurance test, a thrust of 14 mN was detected and this, compatible with the anode damage. Chamber erosion was estimated to have a lifetime between 7000-9500 h.

SITAEL RAM-HET foresee the use of a HET thruster at nominal power of 5 kW (2-8 kW) at an atmospheric mass flow rate of 3-10 mg/s + 1-4 mg/s of Xe at the cathode, to produce thrust between 35-130 mN, exhaust velocity of 7-10 km/s, and an anode inlet pressure of 10^3 Pa. Due to lower neutral density that can be provided by the intake, an additional ionisation stage is needed to improve the propellant utilisation efficiency. Within this stage, the central coil (named "mixine") produces an intense magnetic field. This magnetic field magnetises the particles in this region moving them with a cyclotron motion around the central coil. This will increase the plasma density and the residence time of the particles in this area. The chamber walls and the central coils are biased at higher voltage than the second stage anode located at the beginning of the thruster. The resulting E-field increases plasma energy and contributes to the propellant ionisation. The thruster has three sections: collection, ionisation, and acceleration, shown in Fig. 21.

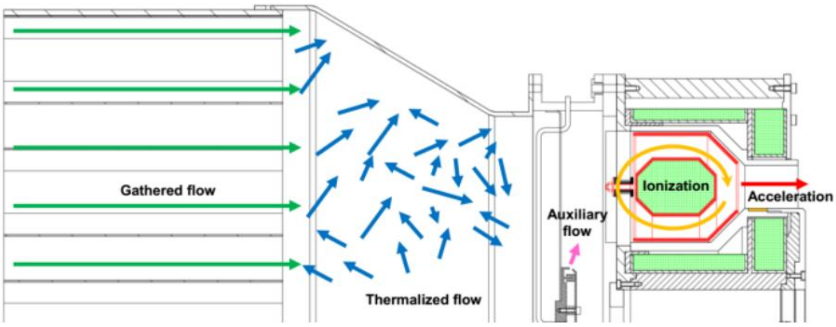


Fig. 21 SITAEL RAM-HET system scheme [19]

BUSEK Co. Inc., tested a standard Xe Hall-effect thruster for Martian operation for the MABHET concept. The thruster was without any modification and was operated with a mixture representing the Martian atmosphere: 95.7% CO_2, 2.7% N_2, and 1.6% Ar. Over 10 hours testing did not show any kind of anode erosion. The thrust to power ratio is just over 30 mN/kW. A change in the discharge chamber volume is suggested to improve the thruster characteristics. An ECR microwave cathode is also suggested to cope with the dissociated O_2.

6 Synthesis of the literature

A brief summary of the results of the literature review is presented in Tab. 1.

The presented literature review of ABEP included various intake designs that were supported by simulation and testing. This provides a basis on which the research should focus for improving the intake performance. Each design depends strongly on thruster typology, size and requirements, but also on the mission. This means, as for now, that the intake has to be specifically designed for the mission. A passive structure in the front, such as a honeycomb composed by many small ducts, will help to increase the collection efficiency of a baseline design that assumes fully diffuse wall reflections and results in a “molecular trap”. The next steps will be include (numerical) investigation of detailed gas-surface interactions including heat loads, erosion and influence of molecular recombination, which will clarify the importance of the actual intake material which has not yet been considered. However, the modelling approach still needs additional experimental validation. Erosion issues are expected on the intake, due to continuous impact of atomic oxygen at about 7.8 km/s in Earth orbit. Approaches such as ceramic coating of the materials, or the use of advanced 2D-materials would increase its durability over time. Alternative designs for improving efficiency need also to be discussed. Utilisation of specularly reflecting materials might constitute a great benefit. Moreover, discontinuous operation modes with active components could be discussed. However, due to the system adaptation time (µs-ms), this would require very high operation frequencies for a significant gain in total efficiency.

The presented studies considered mostly RIT and HET to operate on atmospheric propellant. Each study acknowledged the issue of reduced performance while operating on atmospheric propellant. More energy is necessary to ionise the propellant, and less thrust is produced due to the lower molecular mass of N_2 and O, compared to that of Xe and Ar. Finally, this results in a lower thrust density. Moreover, the operation of the thruster with aggressive gases such as O_2 has been linked, in each study, to an issue of erosion of thruster components. RIT are affected in particular by the erosion of the accelerating grids resulting in a much lower lifetime of the thruster. Degraded performance in thrust and I_{sp} linked to this erosion will also reduce the mission lifetime. JAXA studies did not test the thruster, but, in order to cope with atomic oxygen, suggested to coat the thruster parts that do not require electrical conductivity, with ceramics, and to cover those that requires electrical conductivity with gold. [16] stated that a substitution of the accelerating grids with one made by titanium would increase the thruster lifetime. However the maximum lifetime is estimated at 10 000 h, about 1 year and 51 days. Considering full drag compensation and a constantly active thruster, this is significantly shorter than the lifetime of a typical ABEP missions amongst the covered studies. HETs have been finally considered to be a more advantageous candidate by [16], [19] because of being of simpler construction and of having less parts in contact with the plasma. However, while atmospheric propellant still requires more power and provides less thrust, a change in the discharge channel volume would lead to more optimised performance of the thruster. The oxidation of the anode was observed, forming an oxidised layer. After less than 400 h of operation with atmospheric propellant, the anode presented damage leading to degrading performance. Here, the substitution of the anode with a titanium one would increase the thruster lifetime providing longer lifetime but still in the range up to a maximum of 9500 h. Notably, the use of HETs presented issue at ignition, and Xe was used to ignite the thruster which was then switched to the desired propellant. Moreover, the cathode was continuously fed during operation with Xe. A microwave cathode has been suggested for operation with atmospheric propellant. For application in the atmosphere of Mars, the situation may be improved in terms of oxidation, but still requires deeper analysis. Finally, the use of RF-inductive/helicon thruster has been analysed. This technology has a high potential, as the erosion issues would be removed. Moreover, low-pressure ignition is possible, contrary to the minimum pressure requirement of RIT and of ignition issues with HETs. Furthermore, the plasma leaving the discharge chamber is already neutral, so no neutraliser is needed. Plasma could be generated, however low thrust and efficiencies were achieved. Nevertheless, the physics behind plasma acceleration in such devices needs deeper investigation. In this way, higher thrust values may be able to be produced.

Tab. 1 Estimation of TRL for the reviewed ABEP concepts

ABEP Concept	*Intake TRL*	*Thruster TRL*	*System TRL (if available)*
ABIE [7], [21]	3	2	2
ESA RAM-EP [9]	2	4	2
ABCHT [11]	2	3	2
MABHET [12]	2	4	2
RPT/HHT [5]	1	2-3	1
ABHET [13], [14]	-	3	-
RIT-10/HET [15], [16]	-	4	-
AB-PPT [18]	-	3	-
RAM-HET [19], [20]	3	4	3
Active Intake [22]	3-4	-	-
IRS [17], [23], [24]	3	3-4	3

Tab. 2: Synthesis of ABEP Concepts Database

		ABIE	ESA RAM-EP	ABCHT	MABHET	PPS 1350	RIT-10	Garrigues	Shabshelowitz	GPPT	IRS-IPG6-S	RAM-HET
Thruster	Thruster	ECR	RIT-10	2.St.HET	HET	HET	RIT-10 EBB	HET	HHT/RPT	PPT	IPT	HET
	T, mN	-	2-20	-	-	24	6	20	8.8	5	5-250 [3]	-
	T/P, mN/kW	10-13.7	-	-	19-33	51-54 [2]	13.3 [2]	-	29-59	30	11.6-144.4 [3]	-
	P, kW	0.47-3.3	1	1	1.2	1.4	0.45	1	1-1.5	-	0.5-3.5[4]	-
	I_{sp}, s	3000	-	-	-	900 [1]	5000 [1]	-	>800	5000	-	-
Intake	η_c -	-	-	-	0.2-0.4	-	-	-	0.90	-	0.35	-
	A_{in}, m²	0.48	0.6	-	0.15	-	-	-	0.13 [2]	-	0.3-1	-
	$p_{@thruster}$, Pa	0.5	1E-3	0.01	-	-	-	-	-	1E-3	-	-
Spacecraft	m, kg	-	1000	-	-	-	-	-	325	-	-	-
	A_{front}, m²	1.5	1	0.5	0.3	-	-	1	0.39	0.3	0.3-1	0.12
	L, m	-	-	-	-	-	-	-	2.2	-	-	-
	C_d, -	2	2	2.2	3	-	-	2	2-4	2.2	2.2	-
	h, km	150-200	180-250	>220	≥150	200	200	250	200	100-250	110-240	200
	Lifetime, y	>2	3-8	5	-	-	-	-	≥3	-	-	-

[1] according to Singh [26]
[2] own calculations based on paper data
[3] estimated values based on tests
[4] anode power

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