https://doi.org/10.1007/s44205-022-00024-9

Journal of Electric Propulsion

REVIEW **Open Access**

A review of air-breathing electric propulsion: from mission studies to technology verification

Tommaso Andreussi[1*], Eugenio Ferrato[2†] and Vittorio Giannetti[1†]

†Eugenio Ferrato and Vittorio Giannetti contributed equally to this work.

*Correspondence: tommaso.andreussi@ing.unipi.it

[1] Department of Civil and Industrial Engineering, University of Pisa, Via Girolamo Caruso 8, 56122 Pisa, Italy
[2] Department of Physics, University of Pisa, Largo Bruno Pontecorvo 3, 56127 Pisa, Italy

Abstract

Air-breathing electric propulsion (ABEP) allows for lowering the altitude of spacecraft operations below 250 km, in the so-called Very Low Earth Orbits (VLEOs). Operations in VLEOs will give radical advantages in terms of orbit accessibility, payload performance, protection from radiations, and end-of-life disposal. ABEP combines an intake to collect the residual atmosphere in front of the spacecraft and an electric thruster to ionize and accelerate the atmospheric particles. Such residual gas can be exploited as a renewable resource not only to keep the spacecraft on a VLEO, but also to remove the main limiting factor of spacecraft lifetime, i.e., the amount of stored propellant. Several realizations of the ABEP concept have been proposed, but the few end-to-end experimental campaigns highlighted the need to improve the concept functional design and the representativeness of simulated atmospheric flows. The difficulty in recreating the VLEO environment in a laboratory limits the data available to validate scaling laws and modelling efforts. This paper presents a comprehensive review of the main research and development efforts on the ABEP technology.

Keywords: Air-breathing, Electric propulsion, Very low Earth orbit, Air intake, Atmosphere

Introduction

The first artificial Earth satellite, Sputnik 1, was launched in 1957 on an elliptical orbit with a perigee altitude of 215 km. At these altitudes the Earth atmosphere is dense enough to make an unpropelled satellite like the Sputnik fall in a few weeks. Several years passed since that first launch and the development of space propulsion produced significant changes in the way satellites operate in space. Nonetheless, operating close to the Earth still represents a challenge. Space propulsion relies on propellant stored on board to produce thrust and this ties the platform lifetime to the amount of stored propellant. Lowering the operative altitude implies an increase of drag, and a consequent increase of propellant. But since also the platform size and drag are affected by the propellant mass, this poses severe requirements on the system.

Space missions require to find new and effective ways to use existing resources. The concept of air-breathing electric propulsion (ABEP) relies on an inlet in front of the spacecraft to collect the atmosphere that generates the drag. Using electric power, e.g., gathered from solar arrays, the thruster then ionizes and accelerates the atmospheric particles, generating thrust. By exploiting these scarce and yet renewable resources, it is possible not only to lower the operating altitude but also to untie the link between lifetime and stored propellant, keeping a spacecraft in a very low Earth orbit (VLEO) for an extended period of time.

A first review of ABEP studies was performed by Singh and Walker in 2015 [1]. The authors clearly distinguish two different phases in the ABEP development. After the emergence of the first concepts during the Cold War, when nuclear power was considered a feasible option to balance challenging power requirements, the interest toward the concept was renewed in the late 1990s and 2000s, when novel studies started to shift the focus on solar power architectures [2–5]. The maturity of solar electric propulsion systems and the interest toward VLEOs, demonstrated by the presentation in the same years of the GOCE mission [6], drove the investigation of ABEP systems, balancing the power needed to compensate the drag with that provided by solar arrays. As described in [5], critical technological advancements were deemed necessary, including the development of VLEO environment simulators and the experimental characterization of air collection. Activities then moved from system studies toward the demonstration of the concept feasibility.

The proof of different ABEP concepts was first pursued relying on numerical simulations of atmosphere collection and stand-alone testing of electric propulsion systems operated with air. Few actors then attempted the full end-to-end testing of an ABEP system. When the work of Singh and Walker was published, the only known attempt to characterize air-breathing operation was that of Tagawa et al. [7]. Together with promising indications on the concept feasibility, the review highlighted technological gaps to realize air-breathing electric propulsion.

Even if only few years passed since the publication of [1], much has been done to fill the gaps of ABEP technology. Thanks to the emergence of a new-space economy and to a renewed understanding of the untapped potential of very low altitude orbits, the exploitation of VLEO gained a strong interest among the space community [8, 9]. The attractiveness of these orbits is not only related to the significant improvement of communication and observation performance. Reducing the spacecraft altitude also improves space accessibility, reducing launch costs and offering a more effective utilization of space. Moreover, thanks to the presence of the high atmosphere, in VLEO the spacecraft faces lower radiation levels. Finally, as the orbit of an unpropelled spacecraft at altitudes below 250 km would decay within a few weeks, the same happens to debris population. Operating in VLEO would then offers a measure to cope with increasing space debris and the advantage of an automatic platform disposal.

Electric propulsion is a key enabling technology to operate in VLEO. Air-breathing operation can allow to lower the operative altitude, increase the platform lifetime and reduce the costs associated with propellant management. From sporadic efforts of few institutions, in recent years activities on ABEP concepts started to involve researchers and institutions from several countries (see, e.g., [10]). As an example, in Europe two different consortia between research institutions and industries were established to tackle the challenges of VLEO missions and ABEP technology, respectively the DISCOVERER and AETHER consortia.

This paper aims to review the efforts dedicated to the realization of ABEP-based missions and to analyse the perspectives of the technology. With respect to the previous review of Singh and Walker [1], we here focus on system and mission studies and on concept verification strategies, as well as on recent technological developments. More in detail, in System Design section we first introduce the environment of VLEOs, which defines the drag acting on the spacecraft as well as the input conditions for the ABEP design. ABEP system studies section presents an overview of ABEP system studies, showing the delicate trade-offs that determine the feasibility of the concept. Next, in ABEP Mission studies section, we analyse the advantages of VLEOs and we review mission studies based on the adoption of air-breathing electric propulsion. Intake section covers activities on the air intake, describing the different approaches proposed to collect and compress the residual atmosphere. Investigations focused on purely passive compression strategies are summarized in Passive compression section, whereas in Active compression section we present the research and development activities focused on active compression. Thruster section begins with a review of the efforts to characterize the operation of conventional electric propulsion systems with atmospheric propellant. In ABEP thrusters section, air-breathing electric propulsion concepts are analysed, comparing the different solutions proposed to ionize and accelerate the collected atmospheric flows. The analysis is completed by Neutralization section, where we summarize investigations on air-fed neutralizers and cathodes. The remaining part of the paper contains, in Concept verification section, the review of activities performed to demonstrate the feasibility of the ABEP concept. First, in VLEO flow simulators section we analyse the different approaches adopted to simulate on-ground the VLEO environment, describing the representativeness and limitations of VLEO flow simulators. End-to-end test campaigns of ABEP systems are described in details in End-to-end testing section. Finally, Modeling and simulations section summarizes modeling and simulation efforts that supported the investigation of ABEP concept, aiming either at showing a concept feasibility or at the optimization of a design solution. The paper concludes with Conclusions section, where we briefly summarize the open criticalities of air-breathing electric propulsion and we present a personal perspective on the activities necessary toward the in space application of the technology.

System Design

This section provides a review of past ABEP mission and system studies, focusing on the technical complexities associated with ABEP systems and also presenting their commercial potential. The following general considerations highlight the main critical features of ABEP based scenarios in VLEO:

Propulsion System: at high level, ABEP operation is quantified by power consumption, thrust produced, and collection efficiency. The collection efficiency quantifies the ratio between the flow exhausted by the thruster and the atmospheric flow incident to the intake inlet area. Below a certain particle density value, no thruster ignition can occur, while power levels exceeding the available platform power are reached at high densities. In the intermediate operating envelope, flow properties can significantly impact the thruster efficiency, which determines the amount of thrust produced at a given power level and mass flow rate. For realistic spacecraft geometries and intake performance, to fully compensate the overall platform drag the thruster must be able to accelerate the processed mass flow rate up to exhaust velocities several times larger than the spacecraft orbital velocity (in the order of 7.8 km/s for VLEO missions).

Platform: design figures at platform level include the amount of power available from solar arrays, the platform aerodynamic coefficients, and the volume available for integration of the spacecraft payload and subsystems. In case of low-drag integration of the main subsystems behind the intake wake, a trade-off exists between the amount of propellant that the intake can collect and the volume available for the platform components. Increasing propellant collection relaxes the requirement on minimum specific impulse the thruster should provide to compensate drag, reducing the ABEP power consumption. A lower platform drag relaxes the requirement on electrical power required for drag compensation. Aerodynamic momentum coefficients impact, also in terms of power consumption, the attitude control system used to guarantee nominal attitude and ensure proper payload data acquisition, communications, and alignment between intake axis and atmospheric flow direction. In general, the platform power budget is strictly related to the system capability of drag compensation and long-life operation, as a larger solar arrays area increases both the amount of aerodynamic drag and disturbance torques the thruster and attitude control system should compensate for.

Mission: time of mission and selected orbit parameters have a direct impact on atmosphere properties and thus environmental condition in which the propulsion system will operate. Thus, feasibility regions for mission design are bounded by the operating envelope of the propulsion system. Specific thrust strategies are required to ensure the maintenance of the targeted orbit or mission profile, which is particularly difficult given the high uncertainty associated with atmospheric flow properties and propulsive performance determination. As continuous or quasi-continuous thrust profiles are likely required for orbit maintenance, ABEP will represent a significant power load for the platform. This, together with the advantage of having fixed solar arrays aligned with the incoming flow to reduce drag, may limit the ABEP applicability to low eclipse duration and favorable illumination condition orbits, such as the dawn-dusk SSO.

In the following subsections, we discuss the main models used to assess VLEO atmosphere properties in terms of temperature, density, and composition, and review the main air-breathing system and mission concepts proposed so far.

VLEO Environment

Very Low Earth Orbits (VLEO) are usually defined as orbits with a mean altitude between 100 km and 450 km, though different definitions may be used by different authors. As VLEO is closer to the Earth's surface, near-space missions would greatly benefit from improved reconnaissance conditions, which for the same set of payload performance requirements could result in smaller platforms and lower costs [8]. However, apart from recent JAXA's SLATS (Fujita and Noda [11]), ESA's GOCE (Drinkwater et al. [6]) and SOAR (Crisp et al. [12]) missions, Fig. 1, satellites do not usually operate in VLEO. This is mainly due to the presence of residual atmospheric gases, which causes satellites to experience drag levels in the order of 1-100 mN/m^2, translating into a severe requirement on the on-board propellant mass needed to fulfill the targeted mission

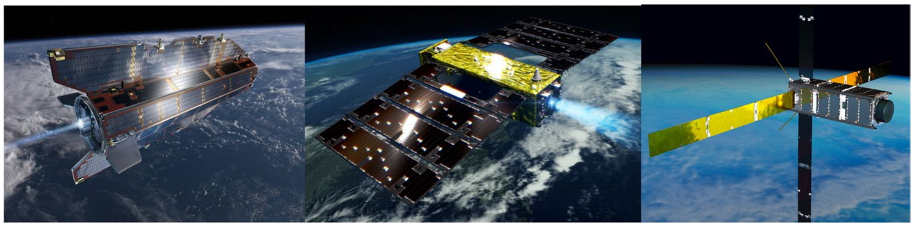


Fig. 1 Left: GOCE spacecraft, credits: ESA/AOES Medialab. Center: SLATS spacecraft, credits: JAXA. Right: SOAR platform, credits: H2020 Discoverer Space

profile. Moreover, the interaction of VLEO atmospheric flow with spacecraft surfaces, including aerodynamic forces, torques and atomic oxygen erosion, adds great uncertainties in the design process, as these phenomena are difficult to predict and still require further research.

VLEO is indeed a complex environment, in which neutral composition and ion concentration undergo spatial and temporal variations as function of solar and magnetic activity, latitude, time of day and solar zenith angle. These changes are actually different at different altitudes due to varying relative ionization, loss and transport phenomena. Studies and modelling techniques of Earth's upper atmosphere started in the 1960s, and were developed based on the empirical data acquired from satellites, sounding rockets and atmospheric probes. The first standard model to describe ionosphere mass density as a function of altitude was the 1976 International Standard Atmosphere (Jacchia [13]). Thanks to the increased number of successful space missions and data available, the first NASA Goddard SFC Mass Spectrometer and Incoherent Scatter (MSIS) model was developed in the 1970s to also include temperature and partial densities of N_2, O_2, O, He, Ar, and H species in the 120 km to 150 km altitude range (Hedin et al. [14]). In the following decade, more data from satellites and scatter radars were included to extend the modelled altitude range from 80 km to 220 km (MSIS-83, Hedin [15]), to include atomic nitrogen among the computable partial densities (MSIS-86, Hedin [16]), and to extend the model output to the lower thermosphere and mesosphere ($<$ 80 km) regions (MSISE-90, Hedin [17]). In the early 2000s, the Naval Research Laboratory (NRL) released the NRLMSISE-00 model, currently among the most advanced tools for assessing atmosphere properties up to 1000 km of altitude (Picone et al. [18, 19]), Fig. 2.

In parallel, Jacchia and Bowman developed a second model, the JB-2006 (Bowman et al. [22]), which was then updated to the JB-2008 version (Bowman et al. [23]). The JB model is deemed to provide more accurate estimations of temperature and total densities variation in the atmosphere, but lacks the capability of computing the partial densities of the constituent species. Both MSIS and JB models are constantly updated, and currently represent the reference for atmosphere assessment according to the European Cooperation for Space Standardization. Other models of use include the Drag Temperature Model DTM2013 (Bruinsma [24]), the International Geomagnetic Reference Field IGRF13 (Alken et al. [25]), and the Horizontal Wind Model HWM14 (Drob et al. [26]). The DTM2013 is a semi-empirical model tuned with the data obtained from CHAMP, GRACE, and GOCE missions, providing Earth's thermosphere temperature, density, and composition in the 200 km to 900 km altitude range. The IGRF13 is instead the reference

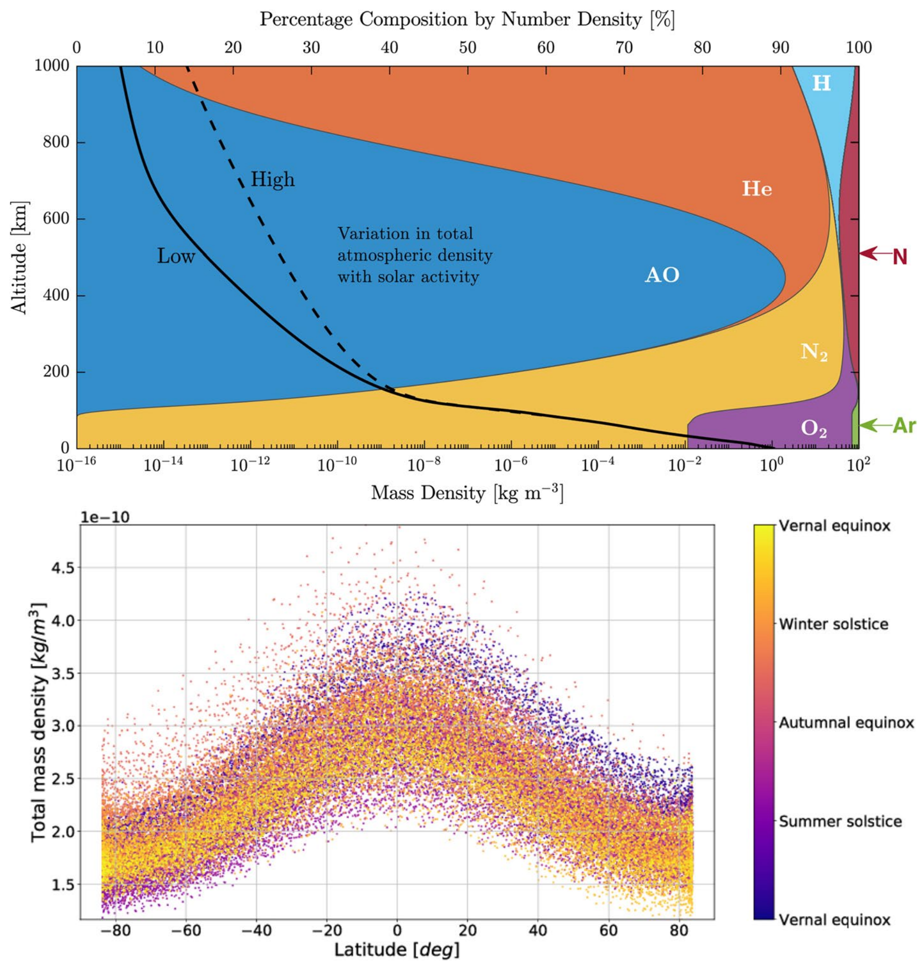


Fig. 2 Top: Representative variation of atmospheric density and composition with solar activity level and altitude in LEO, calculated using the NRLMSISE-00 model. Extracted from [20]. Bottom: Yearly total mass density variation over a dawn duck SSO orbit at 180 km of altitude, computed using the NRLMSISE-00 model in the March 2003 to March 2004 period. Extracted from [21]

standard for Earth's magnetic field intensity at a given time and position, used to assess magnetic disturbance torques and requirement for magnetorquer-based attitude control systems, while the HWM14 is an empirical model for horizontal winds in the upper atmosphere depending on position and solar and geomagnetic activities.

Based on these models and flight data available, the aerodynamic behavior of VLEO platforms was investigated by several authors. Above altitudes of approximately 150 km and for characteristic spacecraft lengths in the order of 1 m, VLEO atmosphere is highly rarefied (having a mean free path in the order of 10-1000 m [21]) and aerodynamics forces are directly dependent on gas-surface interactions (GSI). These interactions, well reviewed by Mostaza-Prieto et al. [27] and Livadiotti et al. [28], are sensitive to several parameters, including energy and momentum accommodation, properties of the incoming particles (mass, velocity, angle, temperature), and surface characteristics (roughness, contamination, composition, and morphology). Direct Simulation Monte Carlo (DSMC) codes and panel-method codes, such as SPARTA (Gallis et al. [29]), ADBSat (Sinpetru et al. [30]), and the Response Surface Modelling Toolkit (Sheridan et al. [31]), implement

GSI models and are often used to estimate the aerodynamic coefficients of a given platform geometry. In this regard, Walsh et al. [32] employed SPARTA to assess optimal aero shell shapes reducing the spacecraft drag in VLEO. For a given constraint on the spacecraft internal volume, the optimization algorithm resulted in biconical shapes allowing for drag reduction up to between 21% and 35%. Crisp et al. [12] used ADBSat to estimate the drag and lift coefficients of the SOAR platform depending on its aerodynamic surfaces orientation and material and flow properties uncertainty. The same tool was used by Livadiotti et al. [33], who employed SOAR's aerodynamic coefficients to model a quaternion-feedback attitude control system based on momentum wheels, magnetorquers, and four independently controlled aerodynamic surfaces. The results of a Monte Carlo analysis showed that coarse pointing within $\pm 5^{\circ}$ could easily be achieved, ensuring robustness against uncertainties, inaccurate environmental modelling, and attitude hardware limitations. The more complex electrostatic particle-in-cell code STARFISH (Brieda and Keidar [34]) was instead used by Andrews and Berthoud [35] to assess the impact of ion thruster plume interaction with VLEO atmosphere on spacecraft aerodynamics. Even if the detailed phenomena involved in ion-surface interaction still remains largely uncertain (Mehta et al. [36], Capon et al. [37]), the analysis showed that the resulting drag is affected by several collisional and electrostatic mechanisms, suggesting that such interaction is often not negligible and should be properly accounted for in future drag models of spacecraft equipped with electric propulsion.

While the studies mentioned above focused on the on-orbit behavior of standard platforms in VLEO environment, several more studies were performed to assess the practicability, at VLEO altitudes below 250 km, of air-breathing electric propulsion systems, often considered as a high potential enabling technology for long life, low altitude missions.

ABEP System Studies

ABEP-based system studies date back to the beginning of modern space era. As reported by Singh and Walker [1], air collection from an orbiting S/C was firstly proposed in 1959 by Demetriades [38], and consisted of an 11 tons vehicle, referred to as the Propulsive Fluid Accumulator (PROFAC), orbiting at 100 km of altitude. The PROFAC was conceived to daily collect, liquefy and store 400 kg of air by means of a 100 m^2 collection system, and to counteract the aerodynamic drag by using a magneto-hydrodynamic (MHD) thruster powered via a 6 MW nuclear reactor. The first work to also analyze solar power in addition to nuclear was conducted in 1961 by Berner and Camac [39], Fig. 3, while Cann [40] was the first, in 1975, to consider exclusively solar power for air-breathing propulsion, suggesting technological feasibility for operating altitudes higher than 160 km. During the 1970s and 1980s, Minovitch [41] developed the concept of a self-refueling rocket, an air-breathing spacecraft powered by a 10 GW microwave energy network of solar ground stations or by an on-board nuclear power reactor. Moreover, in 1983 he proposed to use such a vehicle as interplanetary transport, thanks to the possibility of refueling by aerocapture maneuvers upon arrival to the target planet [42]. This was the first direct mention of utilizing air-breathing technology, or better atmosphere-breathing, around other planets.

Table 1 Summary of ABEP platforms concepts in recent literature, including targeted mission orbit and altitude h, together with spacecraft mass $M_{S/C}$, power available from solar arrays $P_{S/C}$, frontal area A_f, and drag coefficient C_D

Concept	Orbit	h[km]	$M_{S/C}$[kg]	$P_{S/C}$[kW]	A_f[m^2]	C_D
Nishiyama [5]	SSO 6am	140-160	-	0.5-3.3	1.5	2
Di Cara [43]	SSO 10:30	200-250	1000	2.9	1	2
Hruby [44]	-	150	-	2-3	-	2.2
Diamant [45]	-	200	400	1	0.5	2.2
Shabelowitz [46]	-	180-200	325	0.3	0.36	-
Schonherr [47]	-	200	-	-	0.3	2.2
Romano [48]	-	150-250	<1050	3.5	1	2.2
Andreussi [49]	SSO 6am	190-240	500-750	2.5-3	0.7	-
Tisaev [50]	SSO 6am	170-200	200	0.6	0.1	3.2-4.2
Ovchinnikov [51]	SSO 6am	175	120	0.2-0.4	0.2	-
Vaidya [52]	-	160-250	1000	1.6	1	3.7
Crandall [53]	0°-90° inclination	160-250	7.5-10.5	0.01-0.1	0.01	3-6

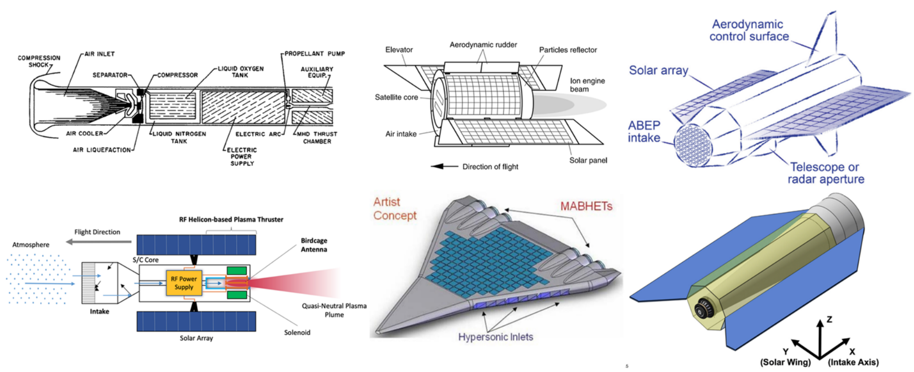


Fig. 3 Examples of ABEP system concepts. Top-left: air-scooping vehicle concept, extracted from [39]; Top-center: ABIE concept, extracted from [54]; Top-right: Crisp et al. concept, extracted from [20]; Bottom-left: Romano et al. concept, extracted from [58]; Bottom-center: MABHET concept, extracted from [44]; Bottom-right: AETHER concept, extracted from [21]

The conclusion of the Cold War marked the end of concepts relying upon nuclear power energy. During the 1990s, the interest in air-breathing spacecrafts was renewed by the work of Conley [2]. This study, which invalidated the simplistic assumptions made during the previous decades, lead to the development of modern ABEP concepts and intake designs. As summarized in Table 1, these concepts mainly account for operations in sun-synchronous orbits at altitude below 250 km, for years-long lifetime, with mass in the 100 kg to 1000 kg range, and available power from solar arrays in the 0.3 kW to 3 kW range.

In their ABIE (Air-Breathing Ion Engine) concept, Nishiyama [5] and Fujita [54] investigated the use of an air-breathing electron cyclotron resonance (ECR) ion engine, Fig. 3. Their studies considered a spacecraft operating in SSO for at least 2 years, with a frontal area of 1.5 m^2, an intake inlet area of 0.48 m^2, an intake transmission of 0.46, and a minimum thruster performance requirement in the 10-14 mN/kW range. Depending on the operating altitude, in the 140 km to 180 km range, a power requirement between

470 W and 3.3 kW was estimated. Di Cara et al. [43] focused instead on SSO dawn-dusk and 10:30 mission scenarios at 200 km of altitude. Different thrusting strategies were assessed, and quasi-continuous thrust firing for at least 5/6 of the orbital period was recommended. The technological assessment focused on air-breathing concepts based on Hall effect thrusters (HET) and radio-frequency ion thrusters (RIT) and on various atmospheric propellant collection systems, including electromagnetic scooping, aerodynamic intakes, active intakes encompassing turbo-molecular pumps and internal tanks for storing the collected propellant, and hybrid device concepts. A 1000 kg spacecraft with a 1 m^2 frontal area and equipped with a 0.6 m^2 intake was selected as reference design. Power generation was based on 19.7 m^2 solar arrays capable of generating up to 2.9 kW EOL power combined with a 612 Wh Li-ion battery. The use of 4 RIT-10 thrusters adapted to operate in air-breathing mode was considered, resulting in a power requirement of 1 kW and thrust levels in the 2 to 20 mN range for a targeted operating life of 7 years.

In the same years, Busek (Hruby et al. [55]) proposed a Hall thruster-based ABEP system, referred to as MABHET, operating in Mars atmosphere, Fig. 3. Flow capture techniques based on passive intakes, active intakes equipped with pump for compressing or storing the collected propellant were investigated, leading to the selection of a passive intake configuration. Based on experimental data obtained from HET characterization with CO_2, a spacecraft with a 0.3 m^2 frontal area and 0.15 m^2 intake inlet area was proposed, and mission feasibility was assessed for intake transmission larger than 0.35 and HET thrust to power ratios in the 19 mN/kW to 30mN/kW range. A more recent work by Hruby et al. [44] presented the activities carried out to apply the same concept, the ABHET, to VLEO orbits in the 100 to 150 km altitude range. Pekker and Keidar [56] considered instead an air-breathing Hall thruster operating around 95 km of altitude, capable of generating thrust levels up to 9.1 N but resulting into a power requirement up to 800 kW, making nuclear the only viable option. A 2-stage cylindrical Hall thruster ABEP system was instead proposed by Diamant [45], which considered a 200 km operating altitude, a 0.35 intake transmission, and a 0.5 m^2 spacecraft frontal area, and resulted into a power requirements in the order of 1kW. A smaller spacecraft with a 0.36 m^2 frontal area, length of 2.1 m and mass of 325 kg was considered by Shabshelowitz [57], who proposed an air-breathing Hall thruster based on RF ionization operating at 200 km of altitude with an average power consumption of 306 W and for a mission duration of 3 years.

More recently, Romano et al. [48, 59] analyzed an ABEP system based on the IPG6-S inductive plasma thruster, an intake inlet area of 1 m^2, and fully illuminated solar arrays aligned with the incoming flow, Fig. 3. They considered intake transmission and compression ratios in the 0.35 to 0.9 range and 100 to 200 range, respectively, while assuming a drag coefficient of 2.2 and 3.5 kW of power available from solar arrays. By relating the expected inlet flow properties to the operating altitude via the NRLMSISE-00 atmospheric model, the total mass flow accepted into the IPG6-S thruster was assessed and used to estimate, by means of a simplified performance model, the thrust produced and power required. For intake transmissions of 0.35 and 0.9, the modelled thruster was capable of achieving full drag compensation at power levels less than 3.5 kW and for altitudes higher than 200 km and 180 km, respectively. A later study by Ferrato et al.

[60] refined some of the simplifying assumptions used in previous studies, considering the Schaaf and Chambré [61] model for drag coefficient calculation, the impact of intake geometry on the achievable transmission and compression, and the impact of intake flow compression on the thruster capability to ionize the incoming flow. By using a simplified model for thrust and discharge power of an air-breathing electrostatic thruster operating at 600 V of accelerating voltage, an optimization was performed in order to maximize the achievable thrust to drag ratio. The optimization resulted in compact satellite geometries with lengths in the order of 1.5 m, solar arrays area between 2.3 m^2 to 5.8 m^2, operating altitudes between 180 km and 210 km, and intake inlet areas between 0.25 m^2 and 0.4 m^2. While the maximized T/D ranged between 1.1 and 1.4, the results suggested that even larger T/D values could be achieved for acceleration voltages larger than 600 V. The design presented by Andreussi et al. [49] considered instead a larger platform, with a total mass in the 500 kg to 750 kg range, available power from solar arrays in the 2.5 kW to 3 kW range, a 0.7 m^2 intake inlet area, and an operating altitude between 190 km and 220 km, Fig. 3. In the same work, simplified performance models for both Hall-thruster like and Gridded-ion engine like air-breathing thrusters are presented and discussed. Crisp et al. [20] obtained consistent design values by developing a trade-off design algorithm for VLEO platforms, encompassing the preliminary sizing of all the main platform subsystems (including ABEP as an option) and optical and SAR-based payloads, with the goal of finding the operating altitude which minimizes the estimated platform mass and cost. Depending on the selected application, the system trade-off resulted in optimum platforms for optical Earth observation missions having operating altitude between 220 km and 330 km, total mass between 225 kg and 618 kg, and total power requirement in the order of 2 kW, Fig. 4.

For the case of ABEP-based platforms, a circular dependence among the platform design variables was observed. This is due to the fact that the power requirement needed

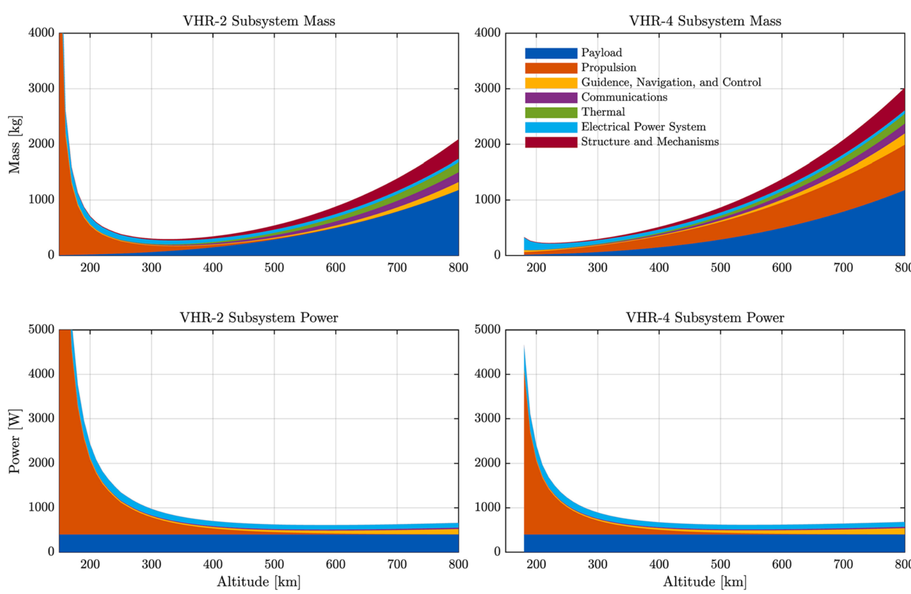


Fig. 4 Variation of subsystem mass and power with altitude for two different optical VHR satellite designs at a given, fixed set of payload performance requirements. The VHR-2 case is equipped with standard electric propulsion, while the VHR-4 case implements ABEP technology. Picture extracted from [20]

for drag compensation increases at low altitudes due to the increased drag experienced, which in turn increases the solar arrays area required for power generation, which in turn keeps increasing both drag and power requirement. To reduce the power requirement, larger intake area could be selected to increase the mass flow rate to the thruster, which, once again, increases both drag and power requirement. In some cases, this circular dependence was enough to prevent the platform design to converge. Tisaev et al. [50] found similar results, obtaining design convergence in narrow altitude ranges depending on the achievable ABEP specific impulse and thrust to power ratio, with minimum required values of 3000 s and 8 mN/kW, respectively. While the minimum altitude was associated to the ABEP power requirement exceeding the one available from solar arrays, the maximum altitude was due to a minimum set flow neutral density at which the thruster could efficiently operate. A reference 200 kg spacecraft with a RIT-based ABEP system capable of producing specific impulses up to 5500 s, thrust to power ratios up to 14 mN/kW, and power consumption less than 600 W was used for further analysis at mission level.

Crandall et al. [53, 62, 63] extended the analysis to more general spacecraft geometries (including cubesats) and orbits, as the previous studies were mainly focused on dawn-dusk SSO orbits due to its favorable illumination conditions. Feasibility regions and comparison with standard EP VLEO platforms were assessed by developing a model depending on the ABEP achievable specific impulse, thrust to power ratio, orbit inclination, and spacecraft geometry in terms of length to width ratio and solar array thickness to spacecraft width ratio. In general, it was showed how the minimum ABEP operating altitude is decreased for larger thrust to power ratios, and that larger ABEP specific impulses and aspect ratios are favorable. Due to illumination constraints, a minimum orbit inclination of 30° seemed feasible depending on ABEP performance. For mission durations above 3 years and altitudes below 240 km, the study also shows how ABEP would offer substantial volume savings for payload allocation compared to traditional EP with on-board propellant. Similarly, Golikov and Filatyev [64] investigated optimal ABEP spacecraft trajectories and layout for fixed spacecraft internal volumes. They assumed a minimum operating density for the thruster, a numerical relation among altitude and intake collection and compression depending on the ratio between intake inlet area and thruster inlet area, and a fixed thruster performance in terms of thrust efficiency. Optimal ABEP spacecraft layouts were then assessed via an optimization algorithm minimizing the power required for a given spacecraft internal volume, set as an optimization constraint. Interestingly, they found that the problem could be solved inside a relatively large range of altitudes, and that an optimum altitude exists in which the ABEP power consumption is minimized, as at high altitude the power increases due to a larger specific impulse required for full drag compensation while at lower altitude the power required again increases due to the larger processed mass flow rate. Depending on the solar activity, this optimum altitude for ABEP operation was found to lie in the 180 km to 230 km range. Associated optimal ABEP spacecraft shapes in terms of length, diameter, and solar array geometries were investigated and discussed, and their dependence on both altitude and solar activity was reported.

ABEP Mission studies

The vast majority of system studies discussed previously pointed out ABEP desirable capabilities at mission altitudes below 250 km. In parallel, several studies are consistently showing how a reduced operating altitude could bring significant benefits for a plethora of mission applications. By considering fixed performance requirement in terms of payload resolution at nadir, coverage, and lifetime, the results presented by Shao et al. [65] indicate that flying multiple small satellites at lower altitudes would lead to lower mission costs for Earth Observation missions compared to traditional large systems. Among 27 different mission scenarios analyzed, 23 resulted into an optimum altitude range for overall mission cost minimization between 200 km and 450 km of altitude. Similar results were obtained by Crisp et al. [20], which presented the impact of operating altitude on overall system mass and cost for optical VHR (Very High Resolution) satellites and SAR (Synthetic Aperture Radar) satellites case studies. The results showed how up to 70% mass and cost savings could be achieved for the optical satellite by operating in the 200 km to 300 km altitude range with the same or improved performance compared to a 500 km of altitude mission. In contrast, cost savings of the same magnitude were not observed for the SAR satellite, though this result was attributed to the use of outdated cost estimates. McGrath et al. [66] obtained similar results, showing how also lidar-based missions in VLEO could lead to a significant reduction in the overall mission costs, as a lidar spacecraft at lower altitude would have both greater coverage and performance than a similar instrument at a higher altitude. For a spacecraft with a defined set of lidar and platform parameters, the 200 km to 300 km altitude range was identified to result in a minimum mission cost, being the turning point at 200 km due to an increased in propellant and solar array mass needed to compensate for the increased drag. Despite the great challenges of VLEO in terms of drag, AO material compatibility, and reduced communication windows, in the study performed by Berthoud et al. [67] considerable benefits were also outlined for telecom-based mission. These include improved link budget, lower latency, lower required transmit power, lower environmental radiation levels, and improved launcher uplift capability. The analyzed case studies included high-speed internet access for fixed services and aggregate mobile stations, mobile 5G high-speed internet access, quantum key distribution, and coverage extension for aeronautical communication services. For all cases, the analysis resulted into an estimated return of investment maximized at altitude between 210 km and 300 km, being a 5G mobile internet access VLEO mega-constellation deemed as the most interesting financial case.

In this context, ABEP represents one of the most promising technology for integration into next-generation VLEO platforms, as the overall mission return of investment could be further increased by simultaneously allowing for propellant savings and longer operation lifetimes in very low altitudes. In this regard, Di Cara et al. [43] firstly proposed a mission based on LIDAR and direct atmosphere sensing payloads consisting of constant operation at 200 km for 3 years, constant climb from 200 km to 250 km for 3 years, and a final descent of 1-year duration. However, the use of ABEP clearly represents a significant challenge as the thruster power requirement is simultaneously dependent on intake efficiency, thrust efficiency, and available mass flow rate, which in turn also depend on the external environment conditions. As atmospheric flow density variation in the order of 50% are typical throughout a single orbit in VLEO, similar variations should be expected for the thruster power consumption, posing a serious challenge to the platform power management capabilities. In this regard, Andreussi et al. [49] proposed the implementation of a discharge controller regulating the voltage applied to the air-breathing thruster electrodes to target a constant power consumption with variable inlet propellant flow properties.

Another critical aspect of ABEP implementation is its capability to ensure orbital stability and correct maintenance of the targeted mission profile. By considering a spacecraft mass of 200 kg, a frontal area of 0.1 m^2, and ABEP performance equivalent to the one reported by Cifali et al. [68] for the RIT-10 thruster operated with N_2/O_2 mixtures, orbital simulation performed by Tisaev et al. [50] showed a divergent altitude behavior leading to a premature spacecraft re-entry within 90 days. The reason was attributed to the impact of Earth's oblateness, which implies an altitude excursion in the order of 30 km along the simulated SSO orbit. Due to the set constraint on minimum operating number density and mass flow rate, the thruster could only be operated in the lower part of the orbit while only experiencing drag in the upper altitude regions. This implied a continuous drift in the orbital eccentricity, leading to a simultaneous increase of the orbit apoapsis and decrease of the periapsis, up to the point in which the drag experienced at lower altitudes was high enough to initiate spacecraft re-entry. This problem was approached by introducing a thrust control variable CT, defined as the ratio between local spacecraft position to mean periapsis distance and mean apoapsis to mean periapsis distance. For CT values less than 0.1, the thruster was constrained to be switched OFF. Such an ON/OFF control was effective in obtaining long-term orbital stability in the 170 km to 200 km altitude range, as the thruster was not allowed to operate near the orbit periapsis, which was sufficient to correct the eccentricity drift associated to the uncontrolled thrust strategy, Fig. 5.

Together with the issue of discharge control and orbital stability, the mutual interaction between spacecraft attitude and propulsion system performance is also an important aspect, as even small misalignments between intake axis and flow direction could significantly degrade the intake capability of collecting and transmitting the atmospheric particles to the thruster. The attitude dynamics of a VLEO spacecraft equipped

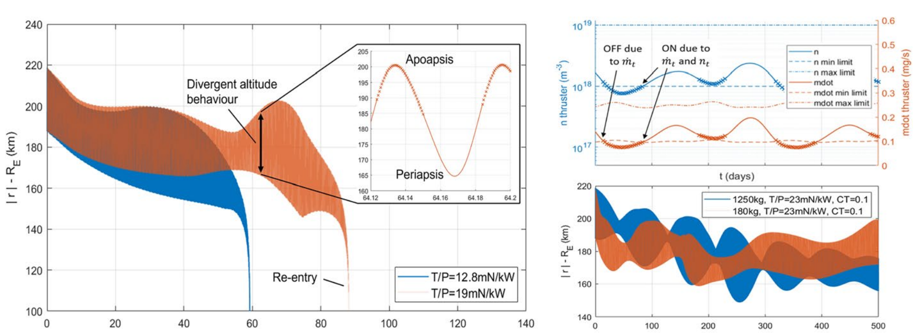


Fig. 5 Left: Propagated altitude profile of uncontrolled thruster for 200 kg s/c mass, showing effect of increased thrust to power ratio from 12.8 *mN/kW* to 19 *mN/kW*. Right: Impact of thruster ON-OFF control on long-term orbital stability. The ABEP thruster is switched off for CT control variable values less than 0.1. Extracted from [50]

with ABEP was investigated by Ovchinnikov et al. [51], which however did not directly account for intake performance sensitivity to spacecraft attitude. Nonetheless, the study presents interesting results in terms of estimating accuracy and power consumption of magnetorquers and flywheels-based attitude control systems, also including the effects of gravity gradient and aerodynamic torques, atmospheric winds, ABEP thrust vector misalignment with respect to the spacecraft center of mass, and residual dipole moments. At the simulated altitude of 175 km and for a reference spacecraft mass of 120 kg and frontal area of 0.2 m^2, magnetorquers and flywheels resulted into a stabilization accuracy in the order of 2.5°-3° and 0.3°-0.7°, respectively. The required attitude control system power consumption was in the order of 20 W for the flywheels and less than 5 W for the magnetorquers. The estimated mean ABEP power consumption was in the order of 200 W to 400 W. As opposed to flywheels, the electromagnetic based control system was shown to be particularly sensitive to center of mass displacements relative to the thrust axis, concluding that the use of magnetorquers would only be recommended for very precise spacecraft layouts, with displacements not exceeding 1 cm.

While the vast majority of ABEP-based mission studies focused on VLEO circular orbits, more advanced mission scenarios were also investigated. For a reference GOCE-like spacecraft having a 1 m^2 intake frontal area, a drag coefficient of 3.7, and 1.6 kW of power available from solar arrays, Vaidya et al. [52] investigated the capability of ABEP to provide full drag compensation in Earth circular orbits at altitudes between 180 km to 250 km, Earth elliptical orbits with perigee altitudes in the 206 km to 277 km range and eccentricities in the 0.2 to 0.8 range, and Mars circular orbits at altitudes between 129 km and 195 km. For all the investigated scenarios full drag compensation was achievable for minimum values of collection efficiencies ranging between 0.43 and 0.77 and thrust efficiencies ranging between 0.2 and 0.5. The same study also considered the possibility to store the propellant collected around Earth and Mars orbits, which could then be also used for advanced space tugs and refueling maneuvers. At the expense of an increased power consumption, up to 40-95 kg/year of collected propellant could be store for Earth missions compared to 190-475 kg/year storable propellant for Mars missions. Similarly, the use of a propellant accumulator for improving ABEP capabilities was investigated by Filatyev et al. [69], which formulated an optimal control problem defining the optimal thrust control vector of an ABEP system, with the goal of maximizing variation of the orbit apogee radius or, alternatively, of maximizing variation in orbit inclination. For both cases, the provided results included optimum thrust angle vs orbit true anomaly and optimal battery and gas accumulator management strategy. However, this study made use of quite strong simplifications, as it considered constant power available from solar arrays (in the order of 1kW), assumed a spacecraft attitude always aligned with the incoming flow direction, and did not implement any upper constraint to the maximum discharge power and mass flow rate which could be provided by the batteries and the gas accumulator.

Intake

The ABEP concept relies on the collection of atmospheric particles to provide the propellant for a plasma thruster. As described more in detail in Thruster section, effective thruster operation typically requires a minimum propellant pressure, which can differ from thruster to thruster but is in general above 10^{-5} mbar. In Earth's atmosphere these pressure levels correspond to altitudes below 150 km. To operate above the minimum altitude compatible with thruster operation, a compression system has to be implemented. The intake is the ABEP element in charge of particle collection and compression.

Even if concepts that do not rely on particle compression have been proposed, a relatively high power is required to operate below 150 km. Consequently, the intake represents a fundamental element of most ABEP systems. An intake performance is often evaluated in terms of collection efficiency and compression factor, which are defined respectively as the fraction of incoming particles that can reach the thruster and the ratio of particle density downstream and upstream the intake. Due to the intrinsic simplicity of purely passive compression, several research teams focused on this approach. Passive intakes exploit rarefied gas-dynamic processes to compress the highly collimated incoming particle flux, allowing to increase the pressure at the thruster inlet by up to two orders of magnitude with respect to ambient conditions. However, in a passive intake the compression factor is strongly related to the collection efficiency. As shown by the simple theoretical model of [70], increasing the collection in general reduces the compression, and vice versa. The inclusion of an active compression stage allows to increase the propellant compression factor, while keeping a high collection efficiency. Moreover, an active compression allows to store the collected propellant and to significantly decouple the thruster operation from the propellant collection. These advantages come at the cost of additional complexity of the ABEP system, mainly related to the operation of fast-rotating components. A summary of the research efforts on the air intake is presented in Table 2.

Passive compression

In 2003, Nishiyama presented for the first time the ABIE concept [5]. A notable feature of the concept was an intake composed by several ducts aligned with the incoming atmospheric flow. Particles moving toward the spacecraft with orbital velocity would pass through the duct as if it is transparent. On the contrary, particles thermalized by

Table 2 Air intake studies available in the literature

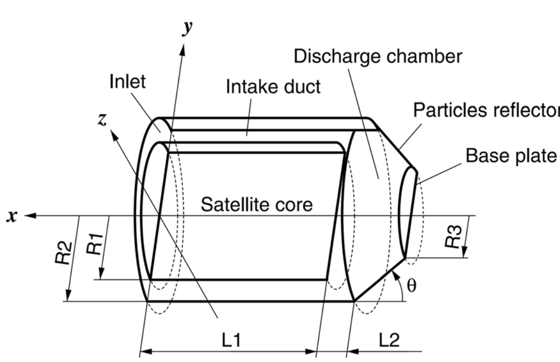
Institution	*Design*	*Reflection*	*Ducts*	*Collection*	*Compr.*	*Refs.*
JAXA	Passive	Maxwell	Yes	0.25-0.45	100-50	[5, 7, 54, 71–74]
Busek	Pas.\Act.	Diffuse	No	0.36-0.60	-	[44, 75]
SITAEL ESA, VKI	Passive	Diffuse	Yes	0.28-0.32	140-95	[43, 49, 70]
		Maxwell	No	0.23-0.25	138-92	[21, 76–79]
LIP	Active	Diffuse	Yes	0.42-0.58	-	[80]
IRS	Passive	Diffuse	Yes	0.31-0.45	-	[81–85]
		Specular	No	0.59-0.94	-	
TsAGI RIAME	Passive	Diffuse	Yes	0.33-0.34	> 100	[69, 86–88]
U. Colorado	Passive	Diffuse	No	0.31-0.35	-	[89]
		Specular	No	> 0.9	-	
Skolkovo Inst.	Active	Diffuse	Yes	Up to 0.98	> 4000	[90]
NUDT	Passive	Spe.\Max.	Both	0.65-0.81	210-100	[91–93]

the collision with the intake or thruster surfaces would be trapped by the small ducts. The author estimated that, using this approach to collect air, the ABIE could reach 70% of propellant utilization. In the paper conclusions, as a first recommendation for future activities on the concept, Nishiyama suggested the design of an air-intake with DSMC methods and its experimental investigation on-ground and in space. As demonstrated by the available literature on the topic, many researchers followed this suggestion, at least for the first part.

There are strong similarities among the investigations performed by different research groups. All the proposed passive intakes either have a long aspect ratio, defined as the ratio between the length and a characteristic transversal dimension, or feature an inlet composed by ducts with long aspect ratio.

For highly rarefied flows, the interactions between particles and intake walls determine the intake performance. In general, gas-surface interactions can be considered a combination of diffuse and specular reflection. However, most surfaces exposed to atmospheric flows show a diffuse reflection behavior, in which particles are thermalized at the surface and scattered in random directions. The idea at the base of the intake of Nishiyama, which was proposed also by Kuznetsov et al. [86] in 1997, is to exploit the different behaviors of rarefied particles entering a duct. Indeed, the probability that a gas particle reaches the end of a duct, called transmission probability or Clausing factor, can change significantly for a population of thermalized particles and a population of particles with a high drift in the duct direction. While in the latter case particles can cross the duct almost without colliding with it, the transmission probability of thermalized particles is the result of a series of successive collisions and decreases with the duct aspect ratio. Approximate expressions of the Clausing factor are available for simple cases (see, e.g., [83]), but no closed-form exact formula exists and, for complex geometries, the transmission probability can only be calculated with Monte-Carlo simulations.

A first study based on DSMC simulations was performed by McGuire on a simple convergent intake positioned in front of the spacecraft [3, 4]. The research, carried out before the work of Nishiyama, highlighted how the intake performance increases by reducing the inlet radius, hence reducing the back-flow Clausing factor. For the considered platform configuration, with the main elements hidden in the intake wake, the drag was determined by the interaction between the atmosphere and the intake.



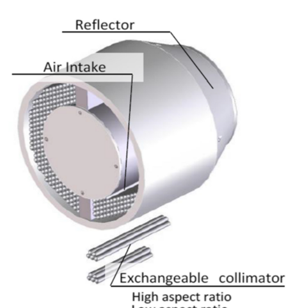


Fig. 6 Schematic of the ABIE air intake (left), extracted from [54], and rendering of the ABIE intake used for the stand-alone testing (right), extracted from [72]

Japanese researchers focused on a different approach to integrate the intake in the platform. As illustrated in Fig. 6, the ABIE intake features an annular inlet, composed by ducts in a honeycomb structure around the satellite core, and a reflector to channel the particles into the thruster. Following the presentation of the ABIE concept, activities on the air intake design proceeded with DSMC simulations and on-ground experiments. In [54, 71], simulations of the intake were performed using the RARAC-3D DSMC code. Gas-surface interactions were modeled with Maxwell's approach, i.e., introducing an accommodation factor to mediate between specular and diffuse reflection. The Variable Hard Sphere (VHS) model was chosen to model particle collisions. The ABIE development led to the realization and testing of the intake design in two different experimental campaigns (see End-to-end testing section). The tests indicated the thermalization of particles inside the intake and the possibility to reach pressure levels of about 10^{-5} mbar [7, 72]. After these experimental activities and the review of ABIE design, novel simulations of the air intake were performed [73, 74].

In 2007, in the first ESA study on the RAM-EP concept [43], the authors presented an intake design consisting of a cylindrical duct placed in front of a GOCE-like platform. A grid system was used to stop the particles at the inlet. Simulations of the intake performance were carried out with the DSMC SMILE code for different intake lengths and shapes. Following this study, activities on the RAM-EP intake were performed by Barral et al. [70]. A passive intake with long aspect ratio ducts was considered and the Q-MoST simulation tool was used to calculate the Clausing factor of different duct shapes. Design optimization led to the realization of a split-ring intake [78], shown in Fig. 7. As described in End-to-end testing section, experiments of the RAM-EP system were carried out by SITAEL in 2017. Although no dedicated measurements were performed to verify the intake performance, the test showed the intake capability to provide the required propellant pressure for thruster ignition. After the RAM-EP test, development of the air-intake design was carried on in strong collaboration by researchers at SITAEL and at the von Karman Institute [49].

In parallel with JAXA and ESA activities, the development of the ABHET concept at Busek led to the investigation of different intake configurations. As described in [44], the intake integration in the platform follows an approach similar to that of [3, 43], with a straight duct that channels the collected particles inside the thruster. During the ABHET concept testing, air collection was demonstrated but detailed measurements were not performed (see End-to-end testing section). In 2012, numerical

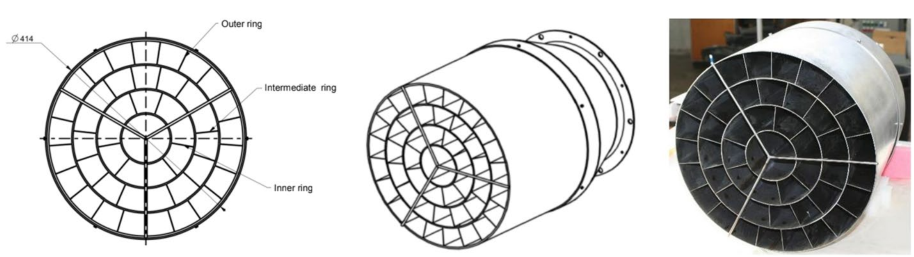


Fig. 7 Schematic of the RAM-EP air intake (left), extracted from [78] (Credits 3AF - Space Propulsion Conference 2016), and picture of the RAM-EP intake used for the end-to-end testing (right), extracted from [79]

investigations with a DSMC code were performed and the impact of a honeycomb inlet composed by small ducts was investigated.

Since 2015, significant work on air intakes for ABEP concepts, combining reduced order modeling and DSMC simulations, has been performed at IRS (see Romano et al. [84] and references therein). The research group provided contributions on the topic validating the results of the first concepts and investigating advanced technological solutions, e.g., based on specularly reflecting materials.

In 2017, Erofeev et al. [87] presented the activities performed at TsAGI on air-breathing electric propulsion. The investigated concept was based on a gas intake, with a honeycomb inlet structure, and a thermalization chamber. Simulations were performed using the Test Particle Monte-Carlo (TPMC) method for free molecular flows and the DSMC method. Based on the experimental characterization of air intakes carried out in TsAGI's wind tunnel VAT-103, a local diffuse reflection of particles was considered [88]. The analysis of intake performance at different Knudsen numbers, from free molecular conditions to transient regimes, was carried out.

Finally, a novel intake concept was recently presented by Prochnow et al. [94]. The research group started the experimental investigation of passive ionization strategies, which rely on the high energy of incoming particles and specific surface coatings. The integration of the flow ionization in the intake would allow for the reduction of ABEP power requirements and the use of electromagnetic fields to guide and collect the atmospheric particles. However, the feasibility of the concept has still to be demonstrated.

The tests carried out at JAXA, SITAEL, and Busek, as well as those performed at TsAGI in the 1960s, represent to date the only experimental efforts dedicated to the verification of intake modeling. Nonetheless, modeling and simulations were used to investigate several critical features of intake design and operation:

Aspect ratio. An important design feature is the aspect ratio of the collecting duct. If we consider the incoming particles to move with a perfectly aligned velocity, increasing the duct aspect ratio helps to block particles inside the intake and hence increases the flow collection. However, the atmosphere at the altitudes of interest has a temperature in the range 700-1000 K. Even if the orbital velocity is one order of magnitude higher then the mean thermal speed, for ducts with a long aspect ratio a fraction of the particles is reflected before reaching the end of the duct. As described in [70, 83], given the thruster transmission probability and the required collection efficiency (or, vice versa, the required compression ratio), an optimal duct length can be deduced.

Ducts shape. As shown by several authors, in the case of diffuse particle reflection the duct shape has only a minor impact on the collection and compression of the incoming flow. In [70], the investigation of single cylindrical ducts with circular, square, and hexagonal cross-sections showed a minor performance improvement for the circular shape. For multiple ducts, different groups adopted different strategies. ABIE intake features circular straws, SITAEL adopted a split-ring configuration, whereas Erofeev et al. [87] proposed square ducts. The work performed at U. Stuttgart IRS concluded that, for diffuse reflection, hexagonal geometries guarantees lower Clausing factors for the thermalized particles [84]. This is confirmed in [93], which compared an hexagonal honeycomb structure and a split-ring configuration, showing similar collection performance and a slight improvement of about 5% of compression for the hexagonal solution. In [84], an iterative procedure for the improvement of an hexagonal intake is presented. Three different design solutions are investigated, considering also multiple duct aspect ratios. This latter solution, with larger ducts in the central region, was able to provide the highest collection efficiency (45.8%) and mass flow rate to the thruster.

Single duct. The addition of a multi-ducts inlet represents an effective strategy to lower the back-flow transmission probability. However, the multi-ducts structure reduces the inlet area and adds manufacturing complexities. For intakes with a large aspect ratio, the presence of a multi-ducts inlet may increase the reflection of incoming particles, producing a degradation of the performance and making the intake more sensitive to flow misalignment [84]. As illustrated also by [44], for high incoming particle densities it is possible to obtain a higher collection efficiency without honeycomb inlet ducts. Considering that the intake performance are optimized by a specific duct aspect ratio, the need to split the inlet area in multiple ducts is basically determined by trade-offs performed at platform level. As an example, a constrained optimization of the intake geometry has been performed in the framework of the AETHER project [21, 49]. The need to accommodate the internal platform subsystems and payload in the intake wake produced an optimal configuration with an elongated shape, for which a multi-ducts inlet does not improve the performance. Elongated intake shapes without inlet ducts were investigated also by Jackson [89] for a CubeSat application and in [84, 91, 93].

Gas-surface interaction. A simple approach to model gas-surface interactions is to consider a combination of diffuse and specular reflection. Specular reflection implies an elastic collision in which the particle energy remains the same, whereas diffuse reflection corresponds to an exchange of transversal momentum and energy, determined by the surface temperature. Numerical simulations typically adopt Maxwell's model and accommodation coefficients to describe the contribution of these two mechanisms (see [28] for a comprehensive review gas-surface interaction models). Although experimental data shows that typical materials produce almost fully diffuse reflections, it is often difficult to have a precise assessment of the accommodation factor of specific materials. Simulations have thus been used to investigate the influence of this factor on the intake performance. As shown in [54], while the wall temperature has only a minor influence, a high specularity of the intake walls reduces the compression. The idea to exploit specular reflections to increase the intake performance was first investigated by Jackson [89], who studied three intake shapes (pyramidal, conical, and parabolic) for either diffuse or specular reflection. While in the first case only minor performance variations were observed, for specular reflection the differences were significant, with the parabolic shape capable to produce an almost 100% of collection. The possibility to have materials or surface treatments to increase the specularity of intake surfaces was actively investigated by the EU-funded DISCOVERER consortium. In this framework, Romano et al. [84, 85] performed numerical investigations of parabolic reflective shapes, and specular multistage intakes. Partial specularity was also analysed, showing an apparent linear behavior. Since particles maintain high orbital velocities, a high collection of specular parabolic intakes is accompanied by a very low compression. To improve the compression ratio, Zheng et al. [91–93] analysed a parabolic reflecting intake coupled with an additional diffusing duct, showing that in this way it is possible to reach compression ratios > 200 with $> 65\%$ of collection efficiency. In a recent research effort of Shoda et al. [74], the effect of surface micro-structures was experimentally characterized. DSMC simulations based on the measured anisotropic scattering properties of these surfaces showed a 20% increase of compression performance with respect to diffuse reflection.

Flow misalignment. The intake alignment with respect to the incoming flow has a significant impact on the performance. Due to horizontal winds, the misalignment of the flow and the orbital velocity can be up to 4 degrees for a typical SSO [76]. The performance degradation associated with such misalignment depends on the intake aspect ratio. As shown by several authors, e.g., [21, 76, 84], for an aspect ratio of 8 small variations (up to 15%) of the intake performance are obtained for a misalignment up to 5 degrees. Above this angle, a rapid degradation of the performance is observed (more than 40% of collection reduction at 10 degrees for the cases of [84] and [76]).

Operating altitude. The variation of intake performance with altitude mainly depends on the variation of atmosphere density and temperature, as well as composition. Notice that, while the particle density decreases with altitude, in the VLEO altitude range the temperature of the atmosphere increases with the altitude. Hence, the effect of the temperature on the collection becomes less significant at lower altitudes. Moreover, due to the different mass of N_2 and O and the fact that the particle thermal velocity depends on $1/\sqrt{M}$, the influence of the atmospheric temperature changes for the two species. This implies a change of collected particle composition. The flow rarefaction, measured by the Knudsen number, increases with altitude. As shown in [76] and [88], the variation of the intake performance with the degree of rarefaction is not monotonic.

Active compression

As part of the first trade-off studies on ABEP systems, the possibility to have an active compression of the collected flow was considered [43, 44]. Even if the concept described in [43] suggests the possibility to have active compression and storage of the collected propellant, the following trade-off activities of Barral et al. [70] evaluated that a high pump area and mass would be required. Due to the low maturity of turbomolecular pumping technology for in-space operation and the maintenance required by such systems, the feasibility of active intakes was considered low and the solution was abandoned. Similar considerations lead Busek to first focus on passive compression strategies. Nonetheless, active compression was investigated in 2012 as a viable option to increase the propellant pressure in the thruster [44]. The work focused on the coupling between a passive intake and an off-the-shelf turbomolecular pump. DSMC simulations were performed considering a columnar incoming flow and a straight tube intake. All particles reaching the apertures at the end of the tube, at the interface with the turbomolecular pump, were considered consumed by the pump. The effect of specular and diffuse reflection was analysed. Test activities were performed to quantify the intake collection efficiency (see End-to-end testing section). Simulations were carried out also considering a cold flow at 300 m/s, representative of the thermal flow ejected by the sources with the discharge off. Collection efficiencies of about $36-37\%$ for specular reflection and up to 60% for diffuse reflection were calculated. However, experimental results showed collection efficiencies below 17%, probably due to the divergence of the flow sources used to simulate VLEO conditions. In 2015, theoretical and experimental activities on an active intake were performed at the Lanzhou Institute of Physics [80]. The research group considered an inlet diameter of 500 mm and a compression system composed of a multi-hole plate and a main turbo pump, followed by a small turbomolecular pump and a miniature scroll pump. The first two stages, i.e., the multi-hole plate and the main turbo pump, determine the collection efficiency of the system. These stages were investigated with numerical Monte-Carlo simulations, whereas the rest of the system was characterized in laboratory. The authors assume that all atoms recombine into molecules and the wall reflections are completely diffuse. The effects of the small turbo is accounted for by a pumping factor. Simulations were performed with both the Test Particle Monte-Carlo (TPMC) approach and a 3D DSMC code, using the variable hard sphere (VHS) model for particle collisions. A low-speed and a high-speed turbo pump configurations were analysed, showing collection efficiencies of respectively 0.42 and 0.57. The corresponding power requirements at 150 km are lower than 12.3 W (low speed) and 52.6 W (high speed). The collection reduces with altitude due to the increase of atmospheric temperature, which affects the collimation of the flow. Experiments on the second part of the compression system, i.e., the small turbomolecular pump and miniature scroll pump, were carried out with off-the-shelf equipment, respectively the Agilent Twistorr 304 FS and the SVF-E2M-5 from Advanced Scroll Technologies Inc. The power consumption of the two pumps was 27.1-150.3 W, but the authors evaluated that for an optimized design the maximum power can be reduced below 100 W.

As shown by [80], the turbomolecular pump technology has a relatively low power consumption and significant heritage coming from on-ground applications. However, using active compression implies additional complexities, related to additional torque, reduced reliability, additional mass, and the novelty of the technology for space environment. In [90], these complexities of active intakes were critically reviewed. A fast rotating compressor implies high friction and low reliability. However, as also suggested by [80], magnetic levitation can remove these issues and has heritage of in-space applications on reaction wheels. Based on these considerations, the authors presented an ABEP concept combining a passive intake, a compressor, and an ion engine. The authors investigated the concept with a simple balancing model of the system and adopted the model of [80] to compute the compression and transmission probability of the compressor blades. After assessing the transmission for a single stage, they composed the result for multiple stages. The influence of blade rotation speed, angle, and aspect ratio was evaluated. From the balancing model at 220 km, a particle density above $10^{19}\, m^{-3}$ can be reached inside the thruster, which allows for effective operation in VLEO. However, as commented by the authors, experimental verification of these results is necessary.

Thruster

As described in previous sections, the main constituents of the atmosphere are nitrogen and oxygen, in both molecular and atomic form and with concentrations that change significantly with altitude and solar activity. With respect to xenon, the typical propellant of plasma thrusters, nitrogen and oxygen have higher first ionization energy (13.62 eV for O and 14.53 eV for N, versus 12.13 eV of Xe), lower electron ionization cross section [95] and lower mass. Additionally, due to their molecular form, energy losses related to dissociative reactions and excitation of vibrational modes are also present. This makes the efficient operation of EP thrusters with VLEO atmospheric flows exceptionally challenging. Additionally, the same atmospheric mass flow generating the drag is the one that needs to be expelled to produce thrust. Since the VLEO ABEP spacecraft is moving at orbital speed, the desired balance between drag and thrust sets a strict requirement on the minimum exhaust velocity (or specific impulse) that must be achieved by the thruster for concept feasibility.

Conventional thrusters

As a first step in the assessment of the feasibility of ABEPs, several research groups have investigated the performance and behavior of traditional electric thrusters fed with atmospheric propellants. A summary of the main actors in this research is presented in Table 3 and relevant experimental results of conventional electrostatic, electrothermal, and electromagnetic thrusters fed with nitrogen and oxygen mixtures are discussed below.

Electrostatic thrusters, such as Hall thrusters and gridded ion engines, rely on electric fields for ion acceleration. The great advantage of these devices is the possibility of obtaining high exhaust velocities of the outbound ion beam. Nevertheless, they require electrodes immersed in the plasma, including a cathode/neutralizer which is typically sensitive to oxidation and contamination.

Among the earliest experimental efforts in this direction, Semenkin et al. in [96] tested anode layer thrusters with different mixtures of xenon and light gases, specifically argon and air. The authors identify that the critical issue in achieving high performance with light gases (including air) as propellant resides in the difficulties in obtaining a sufficiently high ionization efficiency. The conclusion is that the addition of a fraction of Xe

Table 3 Electric propulsion devices tested with atmospheric propellants. Rounded core performance metrics are included when available in the relevant literature*

Institution	*Thr. type*	*Propellant*	*P* [kW]	*T/P* [mN/kW]	$I_{sp}[\times 10^2 s]$	*Ref.*
TsNIIMASH	HET	Air/Xe	-	-	-	[96]
SAFRAN, Giessen Uni., SITAEL	RIT HET	N_2/O_2	0.2-0.6 0.5-1.4	7-14 -	5-64 -	[68, 97]
SITAEL	HET	N_2/O_2	1.2-5.2	~25	6-16	[79, 95, 98]
Busek	HET	$N_2/O_2/Ar$	1-5.5	15-25	10-27	[44]
Stanford Uni. Uni. of Surrey	HET	$Air/N_2/Xe$	0.4-0.7	-	-	[99]
Stanford Uni. Poli. di Torino	HET	N_2	0.5-0.8	27-44	10-11	[100, 101]
Moscow state Tech. Uni.	HET	N_2/O_2	0.3-1.4	16-21	-	[102]
Beijing Inst. of Control Eng.	MCFT	N_2	0.4-1.3	15-18	6-13	[103]
Uni. of Michigan	HHT	N_2	~1.75	21	8-9	[46, 57]
TsAGI	RIT	N_2/O_2	< 0.8	< 26	-	[87]
IRS	IPT	N_2/O_2	-	-	-	[58, 104–106]
Australian National Uni.	Helicon	N_2	0.25	-	-	[107]
Pennsylvania State Uni.	MET	N_2, O_2	1-1.6	-	2-4	[108, 109]
Uni. of Tokyo IRS	PPT	Air	-	-	1-7	[110–113]
Russian Academy of Sciences	MPD	Air	-	-	-	[114]

*Different research teams report performance measurements in different ways. Performance of [46, 57, 68, 97, 102, 103] are partially extracted from graphs, that of [108, 109] are estimated from intensive measurements, whereas for [87, 107] only the expected ion velocity is reported. The values in the table should only be considered as an indication of the operating ranges of the presented devices to propellants with small atomic masses allows to improve the ionization of the light propellants and to reach intense ion beams.

In a series of research efforts between 2011 and 2012, under an ESA-funded activity, Alta (now SITAEL) in collaboration with Snecma (now Safran) and Giessen university, performed an extensive test campaign where state-of-the-art Hall thrusters and radiofrequency ion engines were fed with atmospheric propellants [68, 97, 115]. In this study, Snecma's PPS1350-TSD Hall thruster and a RIT10 EBB ion engine developed at the University of Giessen were tested in Alta's IV10 vacuum chamber and in Giessen's Jumbo facility with different mixtures of molecular nitrogen and oxygen. Observe that throughout all activities detailed in the following the cathode/neutralizer was always supplied with xenon. The test campaign performed on the PPS1350 started by exploring the thruster behavior with pure nitrogen in the 2.3-2.85 mg/s mass flow rate range and between 240 and 350 V of discharge voltage. The comparison of these first results with the xenon benchmark tests highlighted how the computed ionization efficiency is lower for the nitrogen propellant and that, in this case, the current keeps increasing upon increases in discharge voltage at a fixed mass flow rate. A 305 V and 3 A pure nitrogen operating point (neighboring 1kW of discharge power) was then selected for a 10 hours firing test. The thruster operation remained stable throughout the test with a measured thrust between 19 and 21 mN. In a more representative test for VLEO applications, the PPS1350 was also characterized with a nitrogen/oxygen mixture (see Fig. 8). The results highlight how, with the mixture, the ionization efficiency is equal or slightly superior to the pure nitrogen case, especially at low voltages, albeit still inferior to the xenon benchmark. With the addition of oxygen, the thruster exhibited better behaviour with respect to pure nitrogen, especially at low mass flow rates, as low as 2.1 mg/s. Also in the N_2 /O_2 mixture, a 10 hours firing test was performed and a thrust of 24 mN was measured, slightly higher than with pure nitrogen. A post-test inspection revealed oxidation of the anode surface and signs of oxygen operation were also visible on the ceramics, posing concerns for long-duration testing of the device. The thruster was also characterized with N_2/O_2 mixture but with the addition of a 10% mass flow rate of xenon. The addition of xenon has a nonlinear effect on the thruster parameters, depending on the operating condition the discharge current increased between 17% and 40% and the thrust correspondingly increased between 4% and 40%. The authors ascribe this result to the enhanced ionization efficiency of the mixture when xenon is added (similarly to the

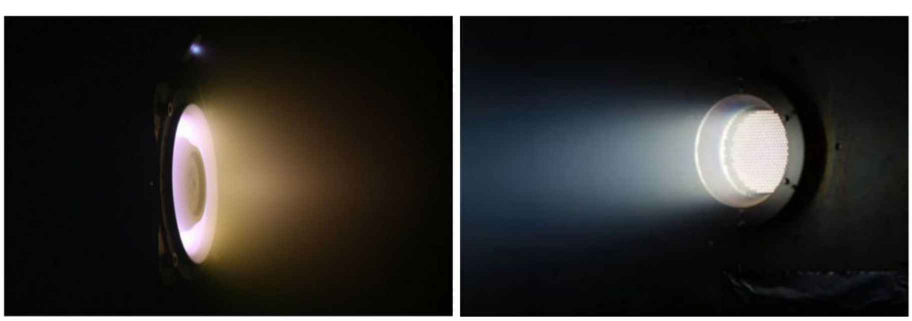


Fig. 8 (Left) PPS1350-TSD operating with N_2/O_2 mixture (305 V, 2.75 mg/s); (right) RIT10 thruster operating with O_2. Extracted from [115]

results of [96]). A similar test sequence was also performed with the RIT10 EBB but only pure nitrogen and pure oxygen propellants were characterized. The pure N_2 test showed that, with respect to pure xenon, the thruster needs a higher amount of gas and total power to operate and delivers an overall lower thrust. Specifically, at a power level of 450 W (including RF power) the thruster achieved 5.25 mN of thrust with the optimal gas flow of 8.5 sccm. The 10 hours test was performed at 150 mA of beam current and 9.89 sccm of mass flow rate. After the test, the erosion of the grid holes, if present, was below the precision of the instrument and, thus, no degradation of the device was observed. The RIT10 EBB was also tested with pure oxygen, with similar general conclusions to nitrogen in the comparison with xenon. With optimal mass flow at a power level of 450 W the thruster achieved a thrust of about 6 mN. A discrepancy between the calculated and measured thrust at high RF power levels was observed. The authors attribute this discrepancy to the varying degree of oxygen molecules' dissociation into oxygen atoms, which is more evident in oxygen than in nitrogen gas due to the lower dissociation energy. Finally, the 10 hours test with pure oxygen was performed in the same 150 mA, 9.89 sccm operating condition. In this case, the grid erosion resulted noticeably higher than after the test with nitrogen, likely due to chemical reactions taking place between oxygen and the graphite constituting the thruster grids.

In the second study [97], Cifali et al. investigated the capabilities of the HET and RIT under investigation of withstanding long operation with atmospheric propellants in a 500-hour endurance testing. The PPS1350 was operated at 305 V of discharge voltage and 2.75 mg/s N_2/O_2 mixture mass flow rate with a 10% addition of xenon mass flow. The thruster operated steadily for about 314 hours with a discharge current between 3.8A and 4A. After this first part of the test, severe oxidation of the anode produced anomalous discharge behavior and a spontaneous flame-out and required a refurbishment of the thruster by cleaning the component. In the subsequent 75 hrs of firing several flame-outs occurred likely linked to the oxidation of the anode surface and the test was prematurely stopped. The authors suggest that alternative, oxidation-resistant anode materials could solve the observed lifetime issues, as the ceramic erosion was still compatible with a 7000-9500 hrs lifetime. The RIT10 was also tested for a 500 hrs endurance firing but the grid material was exchanged with respect to the characterization tests from graphite to titanium. The thruster was operated for the full 500 hrs at a beam current of 150 mA with a flow rate of 6.5 sccm without any degradation of the performance over the test duration. An investigation of the titanium grid holes' erosion also indicated a much lower erosion rate with respect to the graphite grid used in the characterization tests.

In the framework of the development of SITAEL's air-breathing electric propulsion system (see ABEP thrusters section), a 5 kW-class Hall thruster (SITAEL's HT5k) was tested with a $0.56N_2/0.44O_2$ mixture to act as a particle flow generator (PFG) for ABEP testing [79]. Stable operation of the thruster with the atmospheric propellant was verified at a discharge voltage of 225 V and anode mass flow rates ranging from 4.3 to 4.7 mg/s. At a discharge power of 2.4 kW the thruster produced a thrust of 64 mN. In this first experimental campaign, the cathode was operated with pure xenon. The main characteristics of the generated flow were evaluated from the experimental data gathered from the PFG validation by means of SITAEL's PicPlus3D software. In a following

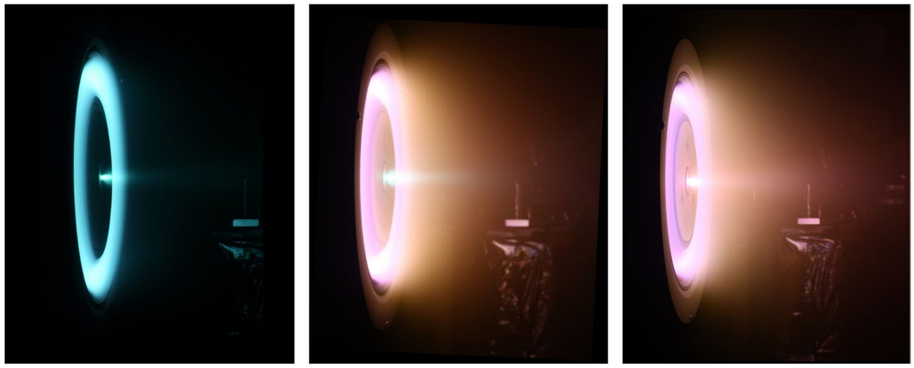


Fig. 9 SITAEL's HT5k (PFG): (Left) operation at 10 mg/s Xe to anode and 1 mg/s Xe to the cathode. (Middle) 6 mg/s 0.56Xe + $0.44O_2$ to anode and 1 mg/s Xe to the cathode. (Right) 6 mg/s $0.56N_2$+$0.44O_2$ to anode and 0.6 mg/s N_2 to the cathode. The discharge voltage is 225 V. Extracted from [116]

research effort, a more advanced model of the HT5k, featuring magnetic shielding of the channel walls, was extensively characterized with atmospheric propellants [95, 98]. The thruster was coupled with a centrally mounted HC20 hollow cathode that was made to transition from xenon to a pure nitrogen flow. The thruster performance and plume properties were characterized with a $0.56N_2/0.44O_2$ mixture as propellant in the 5 to 7 mg/s mass flow rate range and for discharge voltages between 225 V and 300 V (see Fig. 9). In all points tested the discharge current signal was reported to be stable. Over this operating envelope, the produced thrust was measured to be between 30 and 120 mN for discharge powers between 1.2 and 5.2 kW for an anodic efficiency between 8% and 18%.

In a recent report on Busek's activities on air-breathing propulsion [44], Hruby et al. detailed a series of experimental efforts performed in 2005 on a traditional Hall thruster fueled by atmospheric gases. Specifically, a 2 kW-nominal BHT thruster was fed with an "air simulant" propellant, constituted of 68.3% nitrogen, 6.7% oxygen, and 25% argon (based on the similarities of argon with the atomic oxygen present in the higher atmosphere). The thruster was tested between 200 V and 350 V of discharge voltage and for a discharge power between 1 and 5.5 kW. The highest thruster efficiency with the air simulant was achieved in a 350 V, 2.94 mg/s operating condition, with the anodic efficiency reaching about 27%, versus the typical 50% achievable with xenon for this family of thrusters. At the same time, due to the low mass of the accelerated particles, the specific impulse for the air simulant was roughly twice that of the xenon case, with a corresponding reduction of the thrust-to-power ratio by a similar factor. The authors conclude that the performance achievable with conventional thrusters with air as propellant seems to confirm the feasibility of air-breathing propulsion for altitudes higher than 130 km, but the impact of the use of argon instead of oxygen and the comparison between the test gas pressures in the discharge chamber and the pressure achievable in orbit should be critically assessed.

In Gurciullo et al. [99] the authors investigate the plume composition of a Hall thruster fed with variable mixtures of N_2/Xe and air/Xe via a Wien filter. The tested thruster was a 72 mm outer channel diameter Hall thruster, the Z-70, and the ExB probe was positioned 30 cm downstream of the thruster exit section, exploiting a laser to correctly align the probe with the thruster axis. The cathode was an IonTech model HC-252 cathode with a barium oxide emitter always fed with xenon at a mass flow rate of 0.46 mg/s. Following a preliminary benchmark test with xenon propellant, the thruster was operated with various mixtures of xenon/nitrogen and xenon/air at discharge voltages of 270 V and 290 V and power levels between 394 W and 745 W. the xenon mass flow percentage of the tested mixtures ranged from 89% to 10% when mixed with nitrogen and from 96% to 48% with air. The authors studied the shifts in the plume composition upon variations of the propellant mixture ratio and thruster operating condition, highlighting the most probable ion velocity and ion acceleration voltage. One unexpected result of the study was that the lighter ion species (N^+ and O^+) appeared to experience acceleration voltages higher than that of Xe^+, which is counterintuitive since the ionization of lighter species should happen, on average, further down the channel, experiencing a lower fraction of the imposed discharge voltage.

In a subsequent collaboration between Standford University and the Politecnico di Torino, Marchioni et al. [100, 101] built on the previous experience to design and test a Hall thruster specifically conceived to operate with atmospheric propellants. The device, denominated Extended Channel Hall thruster (ECHT), featured a long discharge channel to compensate for the low residence time and ionization cross-section of the lighter neutral particles. The thruster was fed with a constant mass flow rate of 2 mg/s of pure nitrogen and coupled with an Ion Tech HCN-252 cathode operating with Argon. In the described test campaigns direct performance measurements were performed at a discharge voltage ranging from 180 V to 220 V, current levels between 2.6 A and 3.5 A, and with varying magnetic field intensities. The performance trends seem to emphasize a beneficial effect of an extended channel, achieving thrust levels between 20 mN and 23 mN, with specific impulses and anodic efficiencies in the 1009-1116 s and 13-23% respectively, especially for high magnetic fields. Although the performance was increased in comparison with the previous Z-70 thruster model, they were still significantly lower than typical xenon performance figures. The authors comment that this is in any case expected due to the molecular nature of the propellant as several internal modes can sap energy from the plasma. The effect of further increasing the channel length to bolster ionization of the light particles is left uncertain as it would entail diminishing returns by also increasing wall recombination losses.

In a recent work from Dukhopelnikov et al. [102], a 38 mm average channel diameter Hall thruster, originally designed for xenon, was tested with both air and with a 2-to-1 mixture of nitrogen and oxygen. In the experimental campaign, the cathode was always fed with 0.19 mg/s of xenon and the thruster beam was diagnosed through a large Faraday probe placed in front of the thruster. The current efficiency, mass utilization efficiency, and power-to-thrust ratio were tracked as the main performance comparison parameters. While the plume parameters were directly measured with the Faraday probe, the thrust value was estimated assuming a fixed voltage utilization efficiency of 0.75. The thruster was tested with the various propellants for total mass flow rates of 0.8, 0.9, and 1 mg/s and with discharge voltages starting from less than 100 V up to 350 V. The results indicate that with the light propellants the plateau in the discharge current versus voltage curve is found at voltages 50 V to 100 V higher with respect to xenon. Overall, during operation with air, the discharge current was found to be 2.1 - 2.4 times higher than xenon in the 200 V to 350 V voltage range, while the ratio was in the range of 1.9-2.7 for the nitrogen/oxygen mixture case. For what concerns the ion current ratio with respect to the equivalent xenon case with air the results were in the range of 1.7-1.8, while with the N_2/O_2 mixture 1.6-2.2. The reactive, light propellants have shown a mass utilization efficiency on average 2.3 times lower than xenon for equivalent operating conditions. Finally, the current efficiency for air and N_2/O_2 was 1.1-1.2 lower than with xenon for the operating conditions considered. These results highlight the complexities of operating Hall thrusters with light propellants in comparison with xenon. The difficulties in efficient ionization of these gases imply a comparatively low mass utilization efficiency, a large trigger current, not contributing to the thrust, and a consequently low current efficiency, ultimately limiting the total efficiency of the device.

In [103] the operation of a multi-cusped plasma thruster, the MCFT-2139, with nitrogen was investigated experimentally to understand the suitability of the device for an air-breathing propulsion concept. A wide envelope of operating conditions was explored with anode nitrogen mass flow rates ranging from 0.29 mg/s to 1.96 mg/s and discharge voltages between 30 V and 2000 V. A fixed flow rate of 3 sccm Xenon was instead supplied to the cathode throughout the campaign. For each operating condition, a direct thrust measurement was performed and the plume features were diagnosed with a moving Faraday probe. the results highlight the high level of throttleability of the MCFT, operating with anode powers ranging from 0.06 W to 1300 W, which is a desirable feature for air-breathing operation due to the significant changes in atmosphere properties during the mission. As reference performance figures obtained by the device, at 1.06 mg/s anode N_2 mass flow rate and 1000 V of discharge voltage a thrust of 13.3 mN was measured, for a discharge power of 997 W. The availability of the Faraday probe measurements also allowed the authors to derive the propellant utilization efficiency and the current utilization efficiency. These results show that an increase in operating voltage between 400 V and 1000 V for a fixed 1.06 mg/s mass flow rate increases the ion current while decreasing the electron current, improving the current utilization efficiency. Nevertheless, for the flow rate of 1.06 mg/s, the ionization rate seems unaffected by the discharge voltage and the propellant utilization remains around 20%. Conversely, the authors observed a sharp increase of the propellant utilization efficiency up to 46% for a fixed discharge voltage of 400 V when increasing the nitrogen mass flow rate from 1.06 mg/s to 1.96 mg/s.

Shabshelowitz et al. in [46, 57] report on the results obtained by testing a helicon Hall thruster (HHT) with xenon, argon, and, most importantly for the present analysis, nitrogen. The helicon Hall thruster is a two-stage thruster with a first RF stage characterized by the presence of an axial magnetic field and a loop antenna to excite annular helicon waves and a second Hall acceleration stage. The HHT was tested with a 2.6 mg/s mass flow rate of nitrogen, 200 V of discharge voltage, and with an RF power input ranging from 0 to 302 W while measuring the thruster performance via a thrust stand and the plume properties with a Faraday probe, RPA and Langmuir probe. It must be noted that, during the nitrogen test, the cathode was always supplied with 1 mg/s of xenon which represents a 38.4% cathode flow fraction. The test results in single-stage operation (no RF power) highlight that beam divergence efficiency (~60%) and voltage utilization efficiency (~80%) are comparable with xenon tests at an equivalent discharge voltage and particle flow rate. Based on the lower performance figures for nitrogen, the authors deduce that a significantly lower propellant utilization efficiency (~10%) and current utilization efficiency are the two major sources of thruster inefficiency when utilizing nitrogen as propellant. Operating the thruster in 2-stage mode by progressively increasing the RF power produced a minor increase in thrust. However, the thrust-to-power ratio and total anode efficiency both consistently decreased with increasing RF power. The probe data suggest that RF power promotes a slight increase in propellant utilization efficiency but is not sufficient to offset the additional power required.

In a study on the feasibility of air-breathing electric propulsion, Erofeev et al. [87] present some test results of the prototype of the thrust unit of the system, that bears strong resemblances with radio-frequency ion thruster technologies with a beam size of 150 mm. In the limited dataset presented, the thruster was tested for discharge currents in the 0.1-0.4 A range with xenon, molecular nitrogen, and molecular oxygen at 1000 V of accelerator electrode voltage and with a power consumption always below 800 W. The results indicate stable operation with all propellants, except for the pure oxygen tests in the low current range (0.1-0.2 A), with an exhaust speed of 38201 m/s, 116991 m/s and 109435 m/s for Xe, N_2 and O_2 respectively. Taking as an example the highest beam current operating condition of 0.4 A for which all propellants performance are available, the required mass flow rates for Xe, N_2 and O_2 were 0.548 mg/s, 0.0585 mg/s, and 0.0668 mg/s and the corresponding thrust was measured at 20.9 mN, 6.84 mN, and 7.31 mN. As expected for a fixed accelerator voltage, the thrust obtained with atmospheric propellant is significantly lower than the one obtained with xenon but this discrepancy is compensated by a correspondingly higher exhaust velocity.

Electrothermal thrusters, such as arcjets and helicon thrusters with magnetic nozzles, are devices that use electric energy to heat the propellant and then rely on the thermal expansion of the gas for acceleration. Helicon thrusters that rely on magnetic nozzles are often placed in this category, even though electric ambipolar acceleration of the ions plays a role in the expulsion of the propellant. The attractiveness of electrodeless ionization and the utilization of magnetic nozzles for plasma acceleration of atmospheric propellants resides in the absence of immersed electrodes and of the neutralizer, which is found in most other concepts. Electrodeless configurations are particularly appealing for oxygen-rich propellants, to avoid issues with oxidation and contamination of the electrodes and, especially, of the neutralizer. Nevertheless, it is still unclear whether the specific impulse produced by the gas expansion in a magnetic nozzle will be sufficient to counteract the spacecraft drag in real operative scenarios.

In an effort to develop a full air-breathing electric propulsion system, in the framework of the DISCOVERER project, the researchers at the Institut für Raumfahrtsysteme (IRS) of the University of Stuttgart focused on the design and testing of an Inductive Plasma Thruster (IPT) [58, 104–106]. Starting from the successful testing of a laboratory model of the device, the IPG6-S, with 21.95 mg/s N_2 + 4.40 mg/s O_2 [104], the thruster was redesigned to optimize the system performance. Specifically, significant effort was dedicated to the optimization of the RF circuit and to the design of the new thruster birdcage antenna [58, 105]. [106] presents the first ignition and results obtained for the upgraded

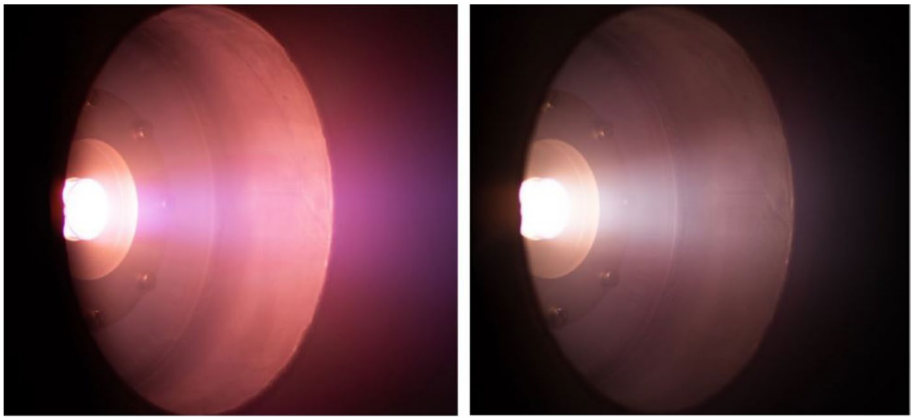


Fig. 10 Inductive Plasma Thruster (IPT) during operation with nitrogen (Left) and oxygen (right). Extracted from [106] (Credits 3AF - Space Propulsion Conference 2020+1)

RF Helicon IPT, based on the cylindrical birdcage antenna and operated with pure nitrogen and pure oxygen (see Fig. 10). The upgraded device showed efficient power coupling at low power and minimized reflected power that is expected to be further tuned by the applied magnetic field. Unfortunately, no direct thrust measurement was reported in the relevant literature and, thus, no thruster performance figures are available.

In [107] the authors investigate the operation of a helicon double layer thruster with alternative propellants, including pure nitrogen. In this experimental study, no direct performance measurements were performed but the ion plume properties were investigated through a retarding field energy analyzer (RFEA) positioned on the thruster axis. The 15 cm diameter device was tested at a nitrogen pressure ranging from 0.34 mTorr to 2.5 mTorr and at a fixed RF power of 250 W and for each condition the ion energy distribution function (IEDF) was derived from the RFEA measurements. The results for low pressures highlighted the presence of two peaks in the IEDF: a low energy peak corresponding to a population of cold ions near the probe (~30 V) and a high energy peak which corresponds to the ion beam (~50 V). The authors describe that the high energy peak should result from ion acceleration in the potential of a double layer spontaneously formed near the end of the source. Assuming an electrostatic acceleration of the ions through the full high energy peak voltage would correspond to an exhaust velocity in the 17-19 km/s range. Observe that these numbers do not directly translate to a high specific impulse of the device and should be considered with caution. For example, only a fraction of the ions is accelerated through the double layer to high speeds, while a significant portion fills the lower energy peak. Direct thrust measurements should be performed to assess the capabilities of the double layer helicon thruster to achieve high performance levels with atmospheric propellant. Direct thrust measurements available for this comparison are found in [117] but for krypton propellant. In the study, a similar analysis is performed with the RFEA measurements indicating an exhaust velocity between 5 and 6 km/s but the maximum effective specific impulse derived from the thrust measurements was 280 s.

Chianese et al. in [108] present the results of an experimental investigation of a kW-class 2.45 GHz microwave electrothermal thruster (MET) fed with nitrogen and oxygen propellants. The MET is an electrothermal thruster where the gaseous propellant is ionized and heated by means of microwave energy, delivered by an antenna, and is successively expanded in a physical nozzle to produce thrust. A spectroscopic system was employed to collect light emitted through a window in the plasma chamber during operation. The measurements were then coupled with emission models for the two propellants to estimate the temperature of the gases and infer the performance of the device. The thruster was tested with both pure nitrogen and pure oxygen and for variable specific absorbed power and chamber pressure (obtained by changing the nozzle size). The results indicate a heavy-particle translational temperature of 2000 K and 5000 K for oxygen and nitrogen respectively, nearly constant for all tested operating conditions. Assuming these temperature values for the gas exhausted through the nozzle and neglecting losses due to thermal conduction and radiation to the walls and non-equilibrium chemistry, the authors estimate an achievable specific impulse of 205 s for oxygen and 395 s for nitrogen. Finally, they conclude that oxygen doesn't heat up sufficiently for effective exploitation in the MET but nitrogen (and nitrogen mixtures with other light gases) could be a promising candidate. A subsequent study from the same group [109] presents a numerical and experimental investigation of two MET models operated with simulants of dissociated propellants typically used in chemical propulsion systems, especially hydrazine and ammonia. The idea behind this research effort was to investigate the capabilities of the MET to be used in a combined chemical and electric propulsion system as an electrodeless substitute for arcjets, often used for similar applications. The thrusters were tested with several mixtures of nitrogen, hydrogen, and ammonia. In addition to the 1kW 2.45 GHz device already discussed, a second 100W 7.5 GHz MET was tested with pure nitrogen with an average microwave forward power of 75 W and a mass flow rate between 1.04 mg/s and 10.41 mg/s gathering similar results to the previous test. The coupling efficiency was found to be higher than 90% for pressures larger than 100 kPa, and the maximum chamber temperature of 2383 K was measured for the high specific power (low mass flow rate) case. Tests with multiple nozzle throat diameters were also performed, showing that for a given mass flow, absorbed power as well as temperature increased with decreasing nozzle throat diameter. However, the absorbed power increase by itself does not fully account for the observed increase in chamber temperature. For example, with a mass flow of 4.16 mg/s, the temperature increased 33% following an 8% increase in absorbed power. This demonstrates an increase in heating efficiency at high pressures.

Electromagnetic thrusters, such as pulsed plasma thrusters (PPTs) and magnetoplasmadynamic thrusters (MPDs), use the Lorentz force and, thus, strong crossed current and magnetic fields to accelerate the propellant. These devices use exposed electrodes, often require very high current levels and are thus operating in pulsed mode (in the case of PPTs) or at very high power levels.

In 2015 the researchers at the University of Tokyo tested a parallel-plate Pulsed Plasma Thruster (PPT) with air as propellant, measuring the current profile of the discharge [110]. To evaluate the feasibility of PPTs as thruster stage for air-breathing electric propulsion systems, in a subsequent collaboration with the University of Stuttgart [111–113], two upgraded versions of the device were experimentally characterized. The two thrusters featured a coaxial geometry and differed in the propellant injection direction which was axial for the device denominated Hagar and radial for the one denominated Asuka. The coaxial design was deemed more suitable than the parallel plate one to better confine the injected gas and consequently increase the thruster performance. Both thrusters were tested with dry air over a wide range of injection pressures and discharge energies (up to 47 J). The test results highlighted that higher performance was obtained by the more compact Asuka engine that managed to achieve higher specific impulse, especially at lower inlet pressures where spectroscopic measurements also indicated a higher ionization degree. Specifically, both the parallel plate prototype and the Hagar thruster produced specific impulses in the 100 s to 200 s range, while the Asuka engine reached more than 700 s at low inlet pressures, albeit strong measurement uncertainty is present in this range due to the accuracy of the employed pressure sensor. Additionally, the two thrusters presented significant differences in terms of scale and overall geometry and, thus, a direct comparison is not immediately conclusive. In [113], employing the lesson learned from the previous designs, a second generation thruster was developed named VIPER (Variable Inlet feeding air-breathing PPT for Electrode eRosion measurements) that allowed both radial and axial injection in a single, flexible device. In the test campaign, the VIPER thruster was fed with air propellant for variable injection pressure levels with both axial and radial injection schemes. The results show that the injection pressure has a significant influence on the discharge oscillations. Higher injection pressures exhibit a stronger damping coefficient while at lower pressure levels a ringing oscillation is observed. Nevertheless, without a simultaneous measurement of the thruster performance, it is not immediately apparent if these oscillations are detrimental to the gas-fed PPT performance. The VIPER test campaign also allowed the authors to investigate the impact of radial and axial injections for a fixed thruster geometry. The observations show that the tendencies and signals look almost identical for an axial and radial injection and, thus, that the injection scheme plays a less significant role than previously believed. Finally, the erosion of the cathode and anode electrodes was measured, indicating a significantly higher erosion rate when compared with typical gas-fed PPTs operating with standard propellants.

As a final note, [114] from 1995 presents some test results gathered for a Magneto-Plasma-Dynamic (MPD) thruster fed with air at ambient pressure. Nevertheless, the focus of the article is only on instabilities in MPD plasmas operated at high pressures and power levels.

All the presented experimental efforts have demonstrated the feasibility of generating and accelerating atmospheric plasmas in electric thrusters. Electrostatic acceleration devices, such as Hall thrusters (HETs) and radio-frequency ion thrusters (RITs), reached levels of thrust efficiency and specific impulse compatible with the requirements of VLEO missions. It is worth mentioning that in most of these test campaigns the cathodes/neutralizers, when present, were fed with xenon to work around the oxygen compatibility issues of typical emitter materials. Although promising results were obtained, especially considering that several of the tested thrusters were not designed for operation with air, these test campaigns did not aim at testing air-breathing operation but simply evaluated the capabilities of existing devices to operate with atmospheric propellant directly injected in the discharge chamber. This often implied the availability of neutral flow densities much higher than what is expected when collecting the atmospheric particles in orbit, which, in turn, greatly enhances the ionization performance.

ABEP thrusters

The efficient ionization and acceleration of the working medium pose a unique challenge in air-breathing electric propulsion systems when compared to traditional EP devices, where the propellant injection properties are regulated with a dedicated flow management system (FMS). In typical FMS-fed systems, once the requirements are fixed, the thruster can be designed to operate with the optimal propellant density ($10^{19} - 10^{20}$ particles/m^3) for efficient ionization in the required power and mass flow range. This is not the case when the propellant is collected via a passive intake. In this scenario, as discussed in Intake section, the mass flow rate ($\sim$ the power consumption for a fixed exhaust velocity) and the achievable pressure in the thruster are strictly related through the intake characteristics. A pressure level in the same range as traditional EP thrusters can only be achieved at relatively low altitudes (<150 km). Nevertheless, operating in these orbits would imply an excessive collection of mass flow rate and, since a sufficiently high specific impulse is required for drag compensation, a prohibitively high power consumption. The high power requirement would then imply a large solar panel area and a consequent high increase in spacecraft drag, producing a divergent loop. Therefore, except for the few instances in the literature where an active intake is foreseen (see Active compression section), the thruster of air-breathing electric propulsion systems is tasked with the problem of producing efficient ionization and acceleration of a very low density atmospheric propellant.

The ionization of atmospheric gases in an air-breathing scenario is investigated by Taploo et al. [118]. Here the authors perform numerical simulations of the detailed plasma chemistry at play in the ionization region of a Hall-like thruster when operated with atmospheric gases. Interestingly, the authors also propose the operation of the thruster without a cathode neutralizer, exploiting the negative ions generated in the discharge for neutralization by operating the thruster alternatively between a low and high electron energy mode. The analysis focuses on atmospheric gas properties found in the 80-110 km altitude range, where no passive compression is needed to have a suitable density for efficient ionization to occur. The results demonstrate that, when factoring only the drag of the thruster cross-section, full drag compensation is possible in the 80-90 km altitude range. For example, for an altitude of 80 km, 1.37 MW of power are required to generate a thrust of 59 N, sufficient to counteract the expected drag. Nevertheless, the authors only consider the drag experienced by the thruster cross section and do not account for the drag induced by the large additional area that will be needed to produce the required power.

Similar conclusions were also reached by Pekker and Keidar in [56]. The authors propose a simplified model of a Hall-like thruster operating in air-breathing mode without passive compression of the incoming airflow. Several simplifying assumptions are made, such as the use of only pure nitrogen as working medium, the neglect of the chemistry in the discharge, and the adoption of a reduced-order formulation for the physical phenomena in the thruster ionization and acceleration regions. The analysis focuses on the 90-95 km altitude range and investigates the thruster behavior for different magnetic

Table 4 Classification of ionization and acceleration methods proposed for air-breathing electric propulsion stages

Ionization	*Acceleration*
Radio frequency/Helicon (RF)	Electrostatic grids (EG)
Electron-cyclotron resonance (EC)	Hall-like/ExB (HT)
DC Electron bombardment (DC)	Magnetic nozzle (MN)
	Lorentz force (LF)

Table 5 Air-breathing electric propulsion concepts available in the literature

Institution	*Concept*	*Ion./acc.**	*Status***	*Ref.*
JAXA	ABIE	EC/EG	1	[5, 7, 119–122]
Busek	ABHET	DC/HT	1	[44, 55]
SITAEL	RAM-EP	DC/HT	1	[49, 60, 79, 98]
Uni. of Michigan	HHT	RF+DC/HT	2	[46, 57]
The Aerospace corp.	2-stage CHT	EC/HT	2	[45]
TsAGI	RF & laser IE	RF &DC/EG	2	[87, 123]
IRS	IPT	RF/MN	2	[48, 59, 104, 105]
Beijing Inst. of Control Eng.	MCFT	DC/HT	2	[103]
Uni. of Tokyo IRS	PPT	DC/LF	2	[47, 110]
Masaryk Uni. SpaceLabEU VZLU PlasmaSolve	Gridded HF source	RF∼EC/EG	2	[124]
SITAEL TransMIT	CSAS RAM-EP	DC/EG	3	[49, 98]
NUDT	ICPG	RF/MN	3	[125]
BISEE	ABHT	RF/MN	3	[126]

*Employed ionization and acceleration methods, with reference to Table 4.

** 1- Full or partial end-to-end test, 2- Thruster standalone test, 3- Concept, studies and numerical investigations

field intensities and ionization chamber lengths for a thruster discharge voltage of 3 kV. The results demonstrate that the thruster could work effectively in the investigated altitude range, producing 9.1 to 22 N of thrust, but it would require a supplied power between 700 kW and 2 MW, depending on the altitude. The authors don't investigate the comparison with the expected drag but comment on the very high power requirement, proposing a beamed power transmission to the VLEO spacecraft to limit its cross-sectional area.

Currently, the study and development of full ABEP systems are undertaken by a handful of institutions. All investigated concepts foresee essentially the same overall architecture, with a passive intake that collects and compresses the incoming atmospheric flow to a discharge chamber where the particles are ionized and later accelerated by the electric thruster. In these concepts, different solutions are employed for the ionization and acceleration of the collected propellant. Table 4 presents a classification of the main methods employed for the ionization and acceleration of the propellant in the ABEP concepts found in the literature. Table 5 presents the main ABEP systems investigated, including the classification of the adopted ionization and acceleration methods and their development status. In the following, the main achievements towards full ABEP system demonstration are discussed, as well as thruster

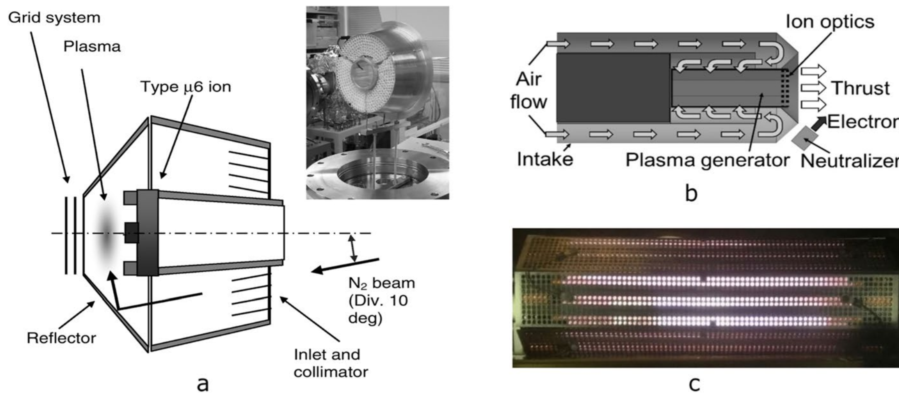


Fig. 11 (a) ABIE prototype tested with the laser beam source, extracted from [7]; (b) schematic of the ABIE and (c) high aspect ratio ECR plasma generator during operation, extracted from [122]

concepts specifically foreseen for air-breathing operation. For a description of the main results concerning thruster standalone testing, also when framed in an ABEP concept, refer to Conventional thrusters section.

JAXA's concept, denominated Air-Breathing Ion Engine (ABIE), focuses on the utilization of an electron cyclotron resonance ionization stage followed by accelerating grids, with a downstream neutralizer to maintain the charge balance of the system. First proposed in 2003 [5], the author's rationale behind the selection of the ABIE ionization method is that, in principle, ECR ionization should be possible at lower pressures than the other options, making it a suitable candidate for the ABEP system. A prototype of the ABIE thruster (including the intake) was assembled and partially verified in a dedicated test campaign using a laser detonation beam source [7], see Fig. 11a. The prototype engine was made of aluminum with a diameter of 155 mm and a length of 150 mm. A microwave antenna surrounded by samarium-cobalt magnets is placed at the center axis of the ABIE, operating at 4.2 GHz and with a maximum microwave power of 60 W. Operation with both propellants showed a maximum ABIE beam current of about 16 mA. At this operating point, with an acceleration voltage of 200 V, the thrust was calculated to be 0.13 mN. Additional details on this verification campaign are provided in End-to-end testing section. Subsequent studies [119, 120] focused on the numerical simulations of the gas dynamics and the ECR plasma formation. The ECR plasma was investigated with the EMSES code and allowed to self-consistently reproduce the electron acceleration and ionization processes in the ABIE discharge chamber. The results suggested that a perpendicular configuration of antenna and magnets should increase the ion generation.

In more recent studies on the ABIE, Miya et al. [121, 122] developed and tested a new high aspect-ratio ECR plasma generator and ion optics, see Fig. 11b and c, intended to operate at very low pressures ($< 10^{-3}$ Pa). The ECR plasma generator has a multi-line-cusp magnetic field to control the drift of electrons and to increase the cross-sectional area of the airflow to prevent neutral particles from escaping to the upstream side. In addition, a negative potential is applied to the shield to block electron inflow to the plasma generator. Through a spectroscopic investigation, it was revealed that the composition of the propellant used in this study differs between 87% and 65% in terms of ionization cross-section with respect to the expected air composition in the 140-270 km altitude range. In the preliminary tests of [121] the relation between magnetic field and ignition performance was explored, identifying the minimum microwave power for ignition. Additionally, notable circumferential asymmetries in the plasma formation were observed. The more recent test results reported in [122] studied the relation between the extracted beam current through the ion optics and the discharge chamber pressure. Current extraction was possible by applying voltage to the electrostatic grids and an optimum pressure was found between 3 and 4×10^{-2} Pa. Nevertheless, a maximum mass utilization efficiency of 0.27% was estimated in the optimal pressure regime.

In 2004 Hruby et al. presented a patent covering an air-breathing electric propulsion system to operate a spacecraft in the 80 to 160 km altitude range [55]. The described device essentially consists of an open-ended Hall thruster, collecting atmospheric particles through an inlet duct. The device also featured an extended channel length to be able to effectively ionize the low density propellant. As an alternative realization of the device, the patent also discusses a multi-stage Hall effect thruster, with multiple electromagnets to establish an extended magnetic/electric field area to increase the propellant ionization and acceleration length in the discharge zone. Only recently Hruby et al. presented the main achievement gathered by Busek on the development of their air-breathing electric propulsion system [44] denominated Air Breathing Hall Effect Thruster (ABHET). On top of the standard HET testing with air simulant as propellant, already discussed in Conventional thrusters section, the literature reports the assembly and testing of an ABHET prototype, as depicted in Fig. 12.

Not many details are shared on the detailed design of the ABHET device. The thruster was operated both with the simulant air (68.3%N_2+6.7%O_2+25%Ar) injected in an inlet duct and with a simulated flow of high-altitude air produced by a RF Hall thruster. Unfortunately, no direct performance measurements of the ABHET during operation are available because of the impingement of the incoming beam on the thrust stand and the excessive facility pressure ($> 1 \times 10^{-4}$ torr) during operation. For additional details on this verification campaign, refer to End-to-end testing section.

The most advanced result in air-breathing electric thruster development was achieved by SITAEL in 2017, with the experimental validation in a representative environment of

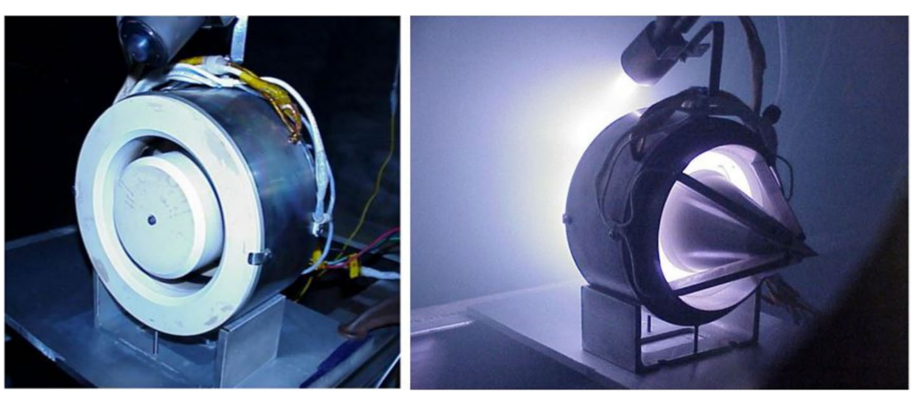


Fig. 12 (Left) ABHET prototype installed in the vacuum chamber; (Right) ABHET thruster during operation. Extracted from [44]

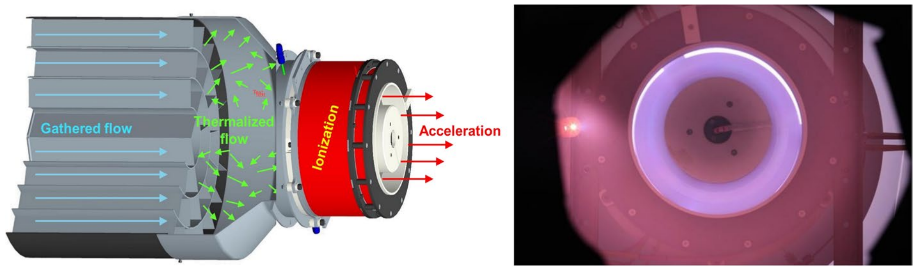


Fig. 13 (Left) RAM-EP system scheme; (Right) RAM-EP exit side during operation in the end-to-end test campaign. Extracted from [79]

its ABEP concept, denominated RAM-EP [79]. Following a split-ring passive intake, the RAM-EP thruster essentially consists of a double-stage Hall thruster, see Fig. 13. The first stage, the ionization stage, is dedicated to the improvement of the ionization of the incoming low density atmospheric propellant while the second stage, the acceleration stage, features the required electromagnetic fields for a Hall-like electrostatic acceleration of the ionized gas.

A prototype of the RAM-EP system was assembled and tested with purely atmospheric propellant with similar properties to the flow expected in orbit while measuring the system performance. The test campaign included an air-fed Hall thruster as particle flow generator. This work was funded by ESA as a technology research programme. Since then, the development of the RAM-EP system has continued under multiple projects [60]. Of particular interest are the results obtained in the framework of the EU-funded AETHER project, that focused on the technological development of the RAM-EP thruster [49, 98]. As part of the project, a second prototype of the thruster was designed to increase the system efficiency with the objective of achieving full drag compensation. In this new thruster model, two different versions of the accelerator stage are proposed, the first one, the Closed Drift Acceleration Stage (CDAS), was developed by SITAEL and features a Hall-like ExB mechanism. The second one, the Charge Separation Acceleration Stage (CSAS), was developed in collaboration with TransMIT and is based on electrostatic grids. One objective of the program will also be to test both options and investigate the optimal acceleration mechanism for ABEP systems. Nevertheless, the two versions of the second prototype are still in development and no experimental results are available at the time of writing.

SITAEL, JAXA, and Busek have all reached a relatively advanced status of development for their ABEP system, performing full or partial end-to-end tests of the technology. Nevertheless, the demonstrated performance have not achieved, so far, full drag compensation and, thus, additional technological development is needed to ascertain the feasibility of the concept. On top of these activities, other groups have proposed novel thruster concepts to cope with the complexities of air-breathing propulsion but have not reached such an advanced stage. Diamant et al. in [45] proposed a 2-stage Air-Breathing Cylindrical Hall thruster (ABCHT) featuring an ECR ionization stage and a cylindrical Hall-like acceleration stage. In principle, the ECR stage should allow for effective ionization at the low pressures encountered during air-breathing operation. A prototype of the thruster was assembled but only tested with xenon propellant while measuring the plume properties with an RPA. Mrozek and colleagues in [124] investigated both numerically and experimentally the operation of a gridded electric propulsion thruster based on magnetized high-frequency plasma operating in the pressure range of 10^{-3} Pa to 1 Pa. The electromagnetic wave used in the ionization stage has a frequency of 435 MHz which is lower than the required ECR frequency. Nevertheless, the authors describe that the plasma source showed resonance behavior with respect to the magnetic field, with the plasma being formed only in a specific magnetic field range, similar to ECR plasmas. The current is then extracted from the plasma through electrostatic grids. The results suggest that the magnetized high-frequency plasma source can only operate reasonably above 5 mPa of internal pressure. Additionally, it was found that the current density scales with the average power absorbed per heavy particle and that larger plasma source diameters are beneficial for the scaling of the concept. Finally, although limited information is available, it is worth mentioning the ABEP concept proposed by the Beijing Institute of Spacecraft Environment Engineering (BISEE) where numerical simulations were performed on a helicon air-breathing thruster, working in the 0.2-2 kW power range to ionize the atmospheric propellant and accelerate it in a magnetic nozzle [126].

Neutralization

Together with the complexities of operating the thruster at low pressures and ensuring ABEP robustness to highly variable environmental conditions, another major challenge is the neutralization of the thruster plume. While in some concepts (such as the IRS inductive plasma thruster developed by Romano et al. [48]) no plume neutralization is needed, a cathode or neutralizer is required for ABEP concepts based on electrostatic acceleration. As discussed before, electrostatic-based ABEPs are among the most promising concepts for achieving full drag compensation in a realistic platform design, thanks to the relatively high specific impulse levels they can provide. However, hollow cathodes (which are among the most mature and flight-proven neutralization technology) are sensitive to oxygen contamination, and scientific literature contains no evidence of experimentally demonstrated cathode/neutralizer operation with atmospheric propellant

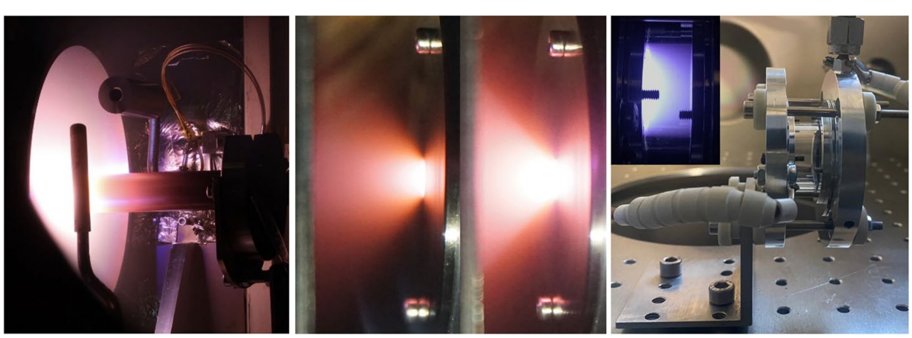


Fig. 14 Left: HC20h hollow cathode operating with pure N_2, extracted from [49]. Center: ECR neutralizer operating with N_2/O_2 mixture at 40 V and 80 V bias voltage, extracted from [128]. Right: diamond-based neutralizer prototype operating with Kr, extracted from [134]

in air-breathing mode. On the other hand, both GOCE and SLATS spacecraft demonstrated correct Xe-fed hollow cathode operation in the atomic oxygen rich environment of VLEO, enabling the option of sustaining the electrostatic ABEP discharge and/or neutralizing its plume by feeding the cathode/neutralizer with a small amount of Xe stored on-board. For current levels not exceeding a few mA, emission filaments or field emitters are a possible neutralization technique. As an example, a 1% thoriated tungsten filament was selected by Diamant [45] for its ABEP concept and prototype testing. For what concerns oxygen-compatible neutralization technologies suitable for larger current levels, the operation of a number of cathode designs with atmospheric propellant provided by a fluidic system (thus, not in air-breathing mode) was recently investigated in the framework of the AETHER programme (Andreussi et al. [49]). In particular, the HC20h hollow cathode (Pedrini et al. [127]), originally designed by SITAEL to operate with Xe and based on a LaB_6 thermionic emitter, was characterized with N_2 and N_2/O_2 mixtures, Fig. 14. The HC20h was capable of emitting current levels in the 8 A to 20 A range with N_2, and a reference performance test with Xe performed after the N_2 characterization showed no evidence of cathode performance degradation. On the other hand, the inspection of the cathode after the test with the atmospheric mixture showed severe erosion and embrittlement of the cathode tube and of some ceramic elements, suggesting the incompatibility of the HC20h design with oxygen propellant. Still in the frame of AETHER, Tisaev et al. [128] developed a second cathode model, the operation of which was based on microwave power deposition into the cathode plasma, see, e.g., Diamant [129, 130]. Similarly to other technologies, such as radio-frequency (Godyak et al. [131]), helicon waves (Kamhawi et al. [132]), and ECR cathodes [133], in a microwave cathode electrons are mainly created in the plasma bulk rather than at wall surface of an emitter (usually operating at a very large, reactive temperature in the order of 1800 K), thus avoiding emitter poisoning phenomena when operating with chemically aggressive propellants such as atomic oxygen. The design of Tisaev et al. [128] features a cylindrical ion collecting electrode, and extraction orifice, and an Al_2O_3 coated molybdenum antenna placed in the plasma bulk and used to deliver microwave energy at a 2.45GHz frequency. At a source microwave power of 60W and 0.1 mg/s inlet N_2/O_2 mass flow rate, this cathode demonstrated emission current levels between 50 mA and 500 mA depending on the ion collecting electrode bias voltage, which ranged between 20 V and 120 V, Fig. 14.

Even if still not verified with atmospheric propellant mixtures, a number of other cathode technologies seem promising for ABEP applications. These include plasma cathodes featuring novel magnetic field topologies to reduce plasma interaction with walls material, such as the Hall-type neutralizer developed by Gurciullo et al. [135], and thermionic cathodes featuring oxygen compatible emitting materials, such as C12A7 carbide hollow cathodes (Reitemeyer et al. [136]), and diamond-based neutralizers (Ahmed et al. [134]). In general, significant progress was made in advancing cathode/neutralizer technological compatibility with oxygen propellant. Nonetheless, experimental demonstration of long-term cathode operation and ON/OFF cycling with oxygen mixtures still needs to be demonstrated. Moreover, whether any cathode technology may be compatible with atmospheric propellant operation in air-breathing mode still remains unknown.

Concept verification

The verification of novel technologies is often the result of a combination of modeling and testing. In the case of plasma thrusters, the complexity of the processes involved in their operation limits the ability of models to fully predict the thruster behavior. Even when specific tests are devised to calibrate a model, the extrapolation of a thruster response outside the experimentally investigated frame is often unreliable. On-ground testing thus represents a significant and still irreplaceable part of electric propulsion development. However, testing requires a representative environment and, even if state of the art testing facility have been developed for conventional EP systems, recreating on-ground the VLEO environment is a major technological challenge, representing one of the main obstacles for the development of air-breathing electric propulsion.

An air-breathing electric propulsion system can be divided in two main parts, the intake and the thruster. A simple approach to reduce the complexity of system verification, and to perform an assessment of the concept feasibility, consists in modeling and testing the two parts separately. With few notable exceptions, this is the approach followed by most research groups, as described in the previous sections. By separating the intake from the thruster, it is possible to investigate the performance of the intake with state of the art simulations of rarefied atmospheric flows, removing most of the model complexities associated with plasma generation and acceleration. Different approaches have been adopted to model the interactions between rarefied atmospheric flows and the intake, showing that simulations are able to provide a clear, in depth description of the intake behavior. The coupling between the intake and the thruster is modeled, in this approach, by introducing in the simulations the thruster transparency (or transmission probability) with respect to the incoming flow, i.e., the probability that a particle entering into the thruster leaves through the thruster exhaust, instead of going back to the intake. Intake simulations allow to derive the conditions, in terms of particle density, flow rate and composition, at the thruster inlet. The thruster behavior can then be characterized through stand-alone testing in conventional EP test facilities, directly feeding the atmospheric particles to the thruster.

Even if this represents an effective approach to preliminary investigate the concept, it has significant limitations. First, simulations of the intake rely on assumptions that need to be experimentally validated. The interactions between atmospheric particles and intake surfaces, which have a central role on the intake performance, are introduced in the simulations by means of an accommodation factor, which gives the probability of a particle to undergo a specular reflection or a diffuse one. This empirical factor may be subject to significant uncertainties, in particular when we consider the evolution in time of the surface properties or the adoption of novel materials and coatings developed to improve the intake performance. Next, modeling the intake as separated from the thruster may be inaccurate for those ABEP configurations where collection and ionization regions overlap (e.g., [44, 79]). Ionized particles are accelerated by the thruster and removed from the intake. The thruster transparency is then determined by the effectiveness of the thruster operation, which can strongly depend on the inlet conditions and thruster operating parameters.

Similarly, limitations apply to the stand-alone testing of a thruster. Directly feeding the atmospheric propellant to the thruster is often not representative in terms of composition, since only molecular oxygen can be supplied. Moreover, in conventional thrusters particles are injected at a relatively low speed, and consequently in conditions not fully representative, either in terms of particle density or mass flow rate. Notice that many of the tests described in Thruster section were conducted with a high particle number density, which can improve the ionization efficiency, making the assessed performance not representative. Finally, since in a pipe-fed thruster all the injected flow rate will go out from the thruster exhaust, the direct feeding of propellant does not allow for the characterization of thruster transmission probability.

Notice that, in general, the interactions between the intake and the thruster are important to determine the effectiveness of an ABEP concept. From this point of view, the adoption of an active intake has the advantage to provide a better decoupling between the thruster and the intake, due to the larger pressure drop and the clear separation between the compression stages and the thruster.

To develop the ABEP technology is fundamental to characterize the particle interactions with the spacecraft, and in particular with the intake walls, to validate modeling and simulations, and to investigate the operation of complete ABEP systems. To perform these activities on-ground, several laboratories invested in the development of atmospheric flow generators. In VLEO flow simulators section, we review the different approaches adopted to recreate the VLEO environment, describing their level of representativeness and their specific limitations. In three different cases, the characterization of flow generators was the preliminary step of end-to-end test campaigns of ABEP systems. These campaigns, focused on Busek's ABHET, JAXA's ABIE, and SITAEL's RAM-EP, represent the first proofs of the ABEP concept and are described in End-to-end testing section.

VLEO flow simulators

The space environment close to the Earth is characterized by the presence of a significant fraction of atomic oxygen (AO), which represents the main reason of spacecraft

Table 6 VLEO flow simulators for ABEP testing

Institution	*Type*	*Velocity*	*Notes*	*Ref.*
Busek	RF heater	3-4 km/s	Free jet, large AO fract., high divergence	[44]
	Arcjet	3-4 km/s	Free jet with stripping, large AO fract., high divergence	[44]
	Hall thruster	7.7 km/s	Large ion fract., small AO fract., velocity spread, Xe cathode	[44]
	RF Hall thr.	-	Large ion fract., Ar added, Xe cathode	[44, 138, 139]
JAXA	Laser detonation	5.8-8.4 km/s	Pulsed O_2 or N_2 operation, large AO fract.	[7, 140]
	RF source + surf. neut.	~8 km/s	Small fract. of accel. particles	[72, 141, 142]
SITAEL	Hall thruster	9.1 km/s	Large ion fract., small AO fract., velocity spread, Xe cathode	[78, 79]
	Hall thruster	6-15 km/s	Large ion fract., small AO fract., velocity spread, ceN2 cathode	[49, 95, 116]
TU Dresden	RF source + surf. neut.	-	Small fract. of accel. particles	[94]
Uni. of Manchester	Ele. stim. desorption	~8 km/s	Low fluxes, large AO fract.	[143]

material degradation. With the increase of space activities in LEO, the need to assess the interaction of the residual atmosphere, and in particular of atomic oxygen, with spacecraft surfaces lead to the development of particle sources and fast atomic oxygen beam facilities (see, e.g., [137]). Different mechanisms have been proposed to generate high fluxes of atomic oxygen, including gas-dynamic expansion, ion acceleration and neutralization, and electron stimulated desorption. These efforts, focused on the characterization of material degradation, served as a base for the development of VLEO flow simulators to verify the ABEP technology.

In VLEO, the residual atmosphere impacts on the spacecraft with a velocity of approximately 7.8 km/s. As described in VLEO Environment section, the conditions encountered in terms of particle density and composition can change significantly with altitude, as well as with the solar and geomagnetic activity and with the geographical position.

Table 6 presents a summary of VLEO flow simulators for testing air-breathing electric propulsion. The first attempts to simulate the VLEO environment for this purpose were carried out at Busek Co. Inc. [44] using gas-dynamic sources. Flow generators based on gas-dynamic expansion rely on a nozzle or a free-jet to convert the gas internal energy into directional kinetic energy. The maximum velocity of the generated particle beam is then directly proportional to $\sqrt{T/M}$, where T is the gas temperature in the nozzle chamber and M is the average mass of the particles. The gas particles can be heated with different approaches, e.g., using radio-frequency (RF), DC, or microwave sources. The Busek team developed a first source based on a DC arcjet operating on N_2, with O_2 added downstream the arc, and a second source based on a RF heater coupled with a free-jet expansion.

Even if gas-dynamic expansion allows reaching high densities of neutral particles, energies are typically below 3 eV for continuous operations, due to the temperature limit of nozzle materials. This was also the case for the first two sources developed by Busek, which could only produce particle velocities around 3-4 km/s. Several atomic oxygen sources overcome this limit by relying on the pulsed operation of a laser that heats up the particles injected into the nozzle [144]. This strategy has been adopted by Tagawa et al. [7] to perform the end-to-end test campaign of the ABIE concept. The selected laser detonation source, which was developed at Kobe University to study oxygen-induced material degradation, uses a pulsed supersonic valve to inject the gas inside a nozzle and a carbon dioxide laser to provide energy for gas dissociation and acceleration. The laser light is focused by a concave Au mirror on the gas inside the nozzle. The source operation was characterized separately with oxygen [140] and with nitrogen [7] using time-of-flight measurements, with a quadrupole mass spectrometer and a scintillation detector installed in the beam line. These measurements allowed to verify that the source is able to provide pulses of N_2 with a velocity up to 8.4 km/s. Less than 20% of N_2 molecules are dissociated in the nozzle, due to the higher inter-atomic bonding strength of N–N. The generated beam is divergent, with a divergence angle of 10 degrees. However, the pulsed operation hindered the full characterization of the ABIE concept and the research group then focused on the development of a source based on ion reflection [141].

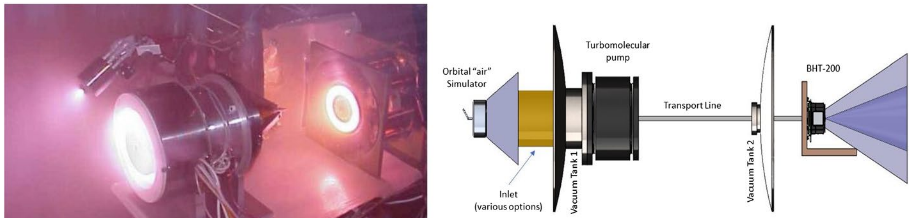
As shown by test results of air-fed electric thrusters (see Conventional thrusters section), ion sources are able to provide continuous fluxes of particles with directed kinetic energies above 5 eV. The generated ions can be then neutralized by means of different techniques, e.g., by resonant charge-exchange in a gas target or by surface neutralization [137, 145]. To ionize and accelerate air particles it is possible to use the same technological solutions adopted by plasma thrusters. To perform the ABHET proof-of-concept, the Busek team tested as a third flow source an air-fed Hall thruster. Thanks to the absence of space-charge effects, the thruster allowed for generating a high-speed and high-density flux of particles, overcoming the limitation of gas-dynamic sources in terms of particle velocities. To improve the ionization of injected particles and reduce the ion acceleration, the team then developed an RF Hall thruster [44, 138], which was used in the end-to-end test campaign of the ABHET concept. The ion source relied on a conventional hollow cathode, fed with xenon, to neutralize the ion beam. Most of the tests with the RF sources were performed replacing the atomic oxygen with argon to reduce the complexity associated with the production of atomic oxygen. The choice of Ar as a surrogate relied on the observation that its ionization rate is almost identical to that of atomic oxygen over a broad range of electron energies. From the ABEP thruster point of view, this allowed to demonstrate the effectiveness of the incoming flow ionization. Since argon metastable state could slightly enhance the ionization, neon replaced argon in some experiments, without any significant change of the ABEP thrust performance. The Busek team further developed the RF Hall thruster to realize an atomic oxygen flow source [139]. The source was operated with a 10% volumetric fraction of argon to enhance the ionization and it was characterized using Faraday cups and an RPA. To reduce the particle energy and obtain a neutral flow of oxygen atoms, surface neutralization was adopted. The ion deflection and energy loss depends on the incident beam angle and the particle mass of the surface. The authors characterized the neutral flow scattered by a hand-polished stainless steel plate set at different inclinations with respect to the incoming ions. The AO flux was measured using kapton erosion and assessing the flow energy through time of flight measurements. At the optimal angle of about 15 degrees the source produced a flux of $10^{16}\,cm^{-2}s^{-1}$ with an energy distribution centered close to 5 eV.

Based also on the work of Homan et al. [139], the JAXA team designed a continuous flow source that combined an ECR ion source and surface neutralization [72, 141, 142]. The ions generated by the source are accelerated by the plasma sheath of a metal plate. By colliding with the solid surface, the ions are neutralized and scattered back. The source has a compact design, with a transverse magnetic field and a thick grid in front to filter only the particles with an axial velocity. The team performed an experimental campaign of the source, using a polyimide covered QCM and time of flight measurements to estimate the AO flux. Even if a significant fraction of injected molecular oxygen is not ionized, the obtained flux results compatible with VLEO conditions at about 220-240 km. This source was used to characterize the ABIE intake [72].

Independently from Busek, SITAEL focused on the adaptation of a Hall thruster to simulate the VLEO environment [49, 78, 79, 95, 116]. The first so-called particle flow generator (PFG) was developed and characterized in the framework of an ESA funded study. The PFG was based on the first development model of the HT5k, SITAEL's 5 kW-class Hall thruster. As discussed in Conventional thrusters section, the thruster was fed with a $0.56N_2/0.44O_2$ mixture, simulating the conditions at 200 km of altitude, while the external hollow cathode was operated using xenon propellant. A stable PFG operating condition was identified and characterized using a thrust stand, Faraday cups, and a drag balance, able to measure the force produced by the PFG plume on a circular plate simulating the intake [79]. An half-angle divergence of 52 degrees was assessed. Dedicated plume simulations were performed to assess the properties of the generated flow as a function of the distance from the PFG. Even if the flow comprises high-speed ions and low-speed neutrals, the characterization proved that the generator is able to provide a flow that is representative, in terms of average properties, of the selected flight scenario. A second PFG was developed and characterized by SITAEL in the framework of the EU-funded AETHER project [49], see Fig. 9. A main modification involved the inclusion of a N_2-fed hollow cathode, to avoid the contamination of the test environment with xenon. Furthermore, the second PFG, based on the magnetically-shielded HT5k DM2, was characterized with an advanced diagnostic system, including a thrust balance, a set of Faraday cups, a RPA, a fast-diving Langmuir probe, and a spectroscopy system [116]. The use of a single Hall thruster source does not allow for the tuning of flow composition and for the differential acceleration of the different species. Nonetheless, based on the collected measurements and on a dedicated discharge model [95], it was possible to identify a range of conditions representative of average flow density and velocity.

Activities on a VLEO flow simulator are on-going also at the Technische Universität Dresden [94]. For ion sources, the significant fraction of charged particles may interact with the ABEP intake and thruster, modifying the flow ionization and introducing additional complexities on the electrical setup. In the case of [94], since the aim is to test an ABEP concept based on the surface ionization of the incoming atmospheric flow, the production of a neutral flow is critical. The team focused on a RF ion source coupled with surface neutralization, obtained with a stainless-steel grid with small, long holes. Preliminary test were performed, also replacing the grid with a molybdenum plate. The flow properties will be characterized through a combination of a force measurement and a catalytic probe.

Finally, it is worth reporting here also the activities of the DISCOVERER consortium focused on the development of a ground facility for orbital aerodynamics research [143]. With respect to conventional sources, which focused on producing a high flux of atomic oxygen at 5 eV, to perform an accelerated characterization of material erosion, the facility developed at the University of Manchester aims to create a controlled and representative environment to characterize the aerodynamic properties of materials. The facility includes a complex pumping system, to guarantee high mean free paths, an



a b

Fig. 15 (a) ABHET prototype during operation with a second Hall thruster acting as VLEO flow simulator and (b) schematic of the end-to-end test setup for the active compression concept, extracted from [44] atomic oxygen flow source, and an ion-neutral mass spectrometer. The atomic oxygen source is based on electron stimulated desorption. The detection system is composed of four stages: an ion filter used to separate charged particles from the neutral ones, followed by an ioniser, a time-of-flight and an analyser. Moreover, the samples and the mass spectrometers are both mounted on moving mechanisms, which allow for measurements of scattering patterns.

End-to-end testing

Despite the potential advantages associated with the development of the ABEP concept, to date only three institutions, i.e., Busek, JAXA, and SITAEL, performed an end-to-end testing of the technology.

The first proof-of-concept was performed at Busek before the end of 2006, with a major part of the work performed under the US Department of Defense SBIR funding. Notice that information about most of the activities performed at Busek were disclosed only in 2022 [44]. After the characterization of a conventional Hall thruster operated with an air simulant (for more details, see Conventional thrusters section), two air-breathing Hall-effect thruster (ABHET) prototypes were designed and built. Figure 15 (a) shows the ABHET LX2 without the intake. The concept was tested in two configurations, with an inlet duct supplied with air simulant and by collecting the VLEO flow simulated by a RF Hall thruster. The latter experiments were performed also without the intake mounted on the thruster inlet. The tests showed that the ABHET was able to operate with the simulated VLEO flow. However, the full ABHET performance was not characterized, since major modifications were needed to adapt the thrust balance and to reduce the facility back-pressure during the thruster operation.

A second important campaign was carried out at Busek in 2012 as a proof-of-concept of an active-intake ABEP configuration, see Fig. 15 (b). The test campaign investigated for the first time the intake operation and the active compression of the collected flow. The team tested various intake configurations (a straight pipe, a straight pipe with a honeycomb structure at the inlet, and a conical pipe) coupled with an off-the-shelf turbomolecular pump. The tests were carried out using two different VLEO flow sources, the RF Hall thruster and a conventional Hall thruster, and assessing the collected flow for nominal source operation (hot flow) and with the discharge off (cold flow). The effects of the flow source distance from the intake and of the intake surface finishing were also characterized. Since the adopted flow sources produced a divergent flow, the collection showed a strong dependence on the distance of the intake. The test results were then

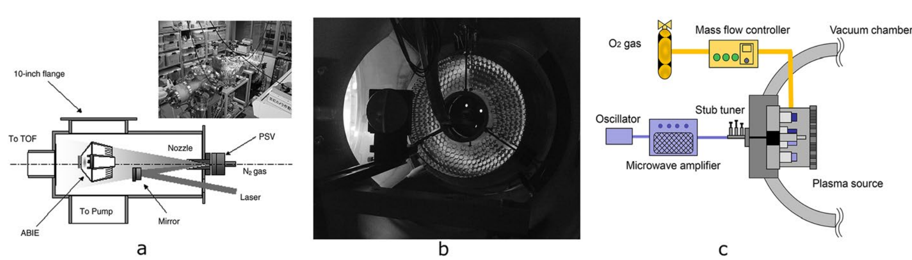


Fig. 16 (a) Schematic of the ABIE end-to-end test setup and (b) oxygen plasma observed from ABIE front side, extracted from [7]; (c) schematic of the ABIE intake test setup, extracted from [72]

compared with the numerical predictions for a columnar VLEO flow, showing a significant discrepancy of the performance. For the cold-air test with the RF source a $11 - 12\%$ of collection efficiency was measured with respect to the 27% of the simulated optimal condition.

In the same years, between 2009 and 2012, JAXA carried out the experimental investigation of the ABIE concept, performing an end-to-end test campaign [7] and the verification of the air intake [72], see Fig. 16. The ABIE test was performed using the laser detonation beam source of Kobe University. The ABIE configuration included the air intake and a central space reserved for either a pressure gauge or the ion engine. Based on the source characterization, the intake was optimized for the recreated flow, adapting its shape to the divergence of the particle flux. The thruster used for the test was a microwave plasma source with the same configuration of JAXA's $\mu6$ ion engine. The team first characterized the intake compression using the pressure gauge. By comparing the rise of the pressure during the source operation, with and without the laser heating, the authors deduced that the incoming molecules are thermalized at the reflector by diffuse reflection. Furthermore, two collimators with different lengths were characterized, showing that the longer duct length does not affect the incoming flow but allows for maintaining the particles inside the ABIE for a longer time. The tests with the thruster were performed separately with O_2 and N_2 at approximately 5.8 km/s. The team was able to ignite the thruster during the flow source pulse, requiring 30-60 W for the N_2 flow and 16 W for the O_2 flow. By observing the plasma generation and the pressure variation during the flow pulse, it was possible to deduce a pressure limit for N_2 ionization of 10^{-4} torr. The team characterized the thruster performance by measuring the maximum current extracted during a flow pulse as a function of the input microwave power. For both N_2 and O flow pulses, the extracted current saturated at 16 mA, corresponding to an estimated thrust of 0.13 mN at 200 V. Based on this observation, the authors hypothesize that oxygen recombines at the walls, also due to the high pressure inside the ABIE. As described by Hisamoto et al. [72], additional test activities on the air intake were performed using a continuous flow source. The campaign was performed with the pressure gauge inside the intake, measuring the compression for the two different collimator lengths and for two atomic oxygen fluxes. Even if the pressure was affected by the oxygen molecules leaked from the simulator, the longer collector allowed for reaching a pressure above 10^{-6} torr.

In 2017, the end-to-end testing of the RAM-EP concept was performed at SITAEL [79, 146]. The RAM-EP system comprised an air intake and a dual-stage Hall thruster. The air intake featured a split-ring collimator and a conical collection region at the interface with the thruster. The intake design, optimized through numerical simulations, was adapted to the results of the flow source characterization, to obtain a representative particle density and mass flow rate at the thruster inlet. The RAM-EP thruster implements a dedicated ionization stage and an electrostatic acceleration stage, based on the ExB ion source design (see Thruster section). During testing, a conventional xenon-fed hollow cathode was used to neutralize the ion beam. After the stand-alone thruster characterization, the intake was integrated and the full RAM-EP system was tested with the particle flow generator (Fig.17). The RAM-EP system was installed on a mechanical support equipped with a thrust

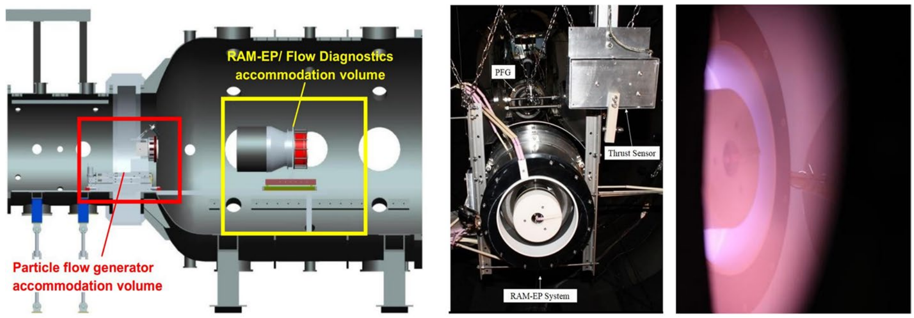


Fig. 17 (a) Schematic of the RAM-EP end-to-end test setup, (b) picture of the setup showing the RAM-EP, the particle flow generator, and the thrust sensor, and (c) RAM-EP operation with collected atmospheric propellant, extracted from [79]

sensor, placed on the rear side with respect to the simulated VLEO flow. No direct measurements of the particle collection and compression in the intake were performed. The operation of the RAM-EP system was verified with a set of firing tests, first feeding the PFG with xenon and then with a $0.56N_2/0.44O_2$ flow. In both cases, the RAM-EP thruster showed a good ionization capability. However, the acceleration stage showed performance below expectations, limiting the thrust produced by the system. The thrust sensor allowed to assess the drag acting on the system when the thruster was off, 26 ± 1 mN, as well as the thrust generated by the discharge ignition, 6 ± 1 mN. Even if the system was not able to fully compensate the drag, the test proved the concept feasibility. Furthermore, a good agreement between drag measurements and simulations was assessed.

Modeling and simulations

The complexity associated with on-ground testing is often mitigated by using models and simulations to evaluate the performance of a novel concept, or at least part of it. Of course, the scarcity of experimental data affects the modeling efforts. This is the case also for the ABEP concept, for which few theoretical investigations are currently available. Nonetheless, despite the uncertainties associated with the lack of experimental data, modeling and simulations represent a fundamental aid for the technology development.

First of all, simulations proved essential in the analysis of the flow collection from the intake, allowing to overcome, at least in part, the main obstacle toward the concept verification. Due to the relatively large collisional mean free path with respect to the intake length, interactions between particles are often negligible with respect to particle-surface interactions and a free molecular flow can be assumed. As described in Intake section, the simulation of the atmospheric flow collection typically relies on a TPMC (for free molecular flow) or DSMC approach [147]. A first exception is the work of U. Stuttgart IRS [81, 83]. Based on analytical approximations of the transmission probability, the authors derived a simple balance model to assess the performance of an intake with a ducted-inlet. The model results were compared to DSMC simulations, showing a good agreement, even if the collection efficiency resulted underestimated. A similar approach was used by Parodi et al. [76]. The authors developed a lumped parameter model, considering the transmission of simple ducts, a hypersonic inlet flux, thermalization at wall and diffuse reflection. The model allowed to perform a sensitivity analysis of the intake performance with respect to orbital conditions. The analysis was refined using SPARTA, an open source DSMC software. Finally, in [77], a view-factor based panel method was developed to reduce the computational cost of DSMC simulations. The modeling approach relies on the similarity of the equations that describe the radiative heat flux and the free molecular fluxes with diffuse reflection. The developed code, called SMARTA, was then adopted in the framework of the AETHER project to perform the intake optimization [49].

In the simulation of air intakes, the surface behavior is often modeled in terms of an accommodation coefficient, which depends on surface materials and finishing, and it may change in time, e.g., due to the interaction of the surface with atomic oxygen. A detailed review of the different particle-surface interaction models is presented in [28]. Thanks to the heritage of several space missions, experimental data are available for common materials, which typically shows diffuse reflection [148]. Even if the intake performance can change significantly with the accommodation coefficient, the uncertainty associated with this parameter can be tackled by performing simulations that encompass the whole range of possible values [21, 54, 84], or by performing dedicated on-ground or in-orbit experiments [12].

Although the lack of experimental validation remains the major criticality of intake development, a preliminary verification of the simulations can be performed by comparing the results of different numerical codes. This cross-model validation was performed in [81] and, recently, in [149], in which the open-source solver dsmcFoam+ of OpenFoam was used to compute the performance of the ABIE intake. Simulations performed with and without inter-particle collisions showed that the free molecular flow approximation represents well the investigated configuration. The comparison of the results with those of RARAC-3D DSMC [54] shows a great agreement between the two codes.

Another important function of modeling is to assess a concept feasibility and identify possible technological solutions. A critical process for ABEP operation is the ionization of atmospheric particles, which has a higher energy cost with respect to noble gases typically used in EP systems. An effective propellant ionization strongly depends on the pressure that can be reached inside the thruster. Given the compression ratio of the intake, the capability to ionize the collected particles determines the upper operational limit of the concept. However, modeling of the ABEP ionization requires taking into account several reactions between particles of different species and with different charge and excitation states. Reduced order models of air-breathing plasma thrusters have been presented by several research groups (see, e.g., [150, 151] for gridded-type acceleration and [56, 60, 101, 152] for Hall-type acceleration).

In [152], simplified analytical scaling laws and a two-dimensional hybrid model are used to analyse atmospheric Hall thruster discharges. Simulations are performed at a fixed discharge voltage of 300 V, for different mass flow rates, and for two different channel lengths. The anomalous electron transport, which represents the main assumption of the model, is described with a Bohm approach and two fixed coefficients, applied inside and outside of the channel. The work highlighted the major differences between xenon operation and air operation for a conventional Hall thruster. Due to the higher ionization cost and the relatively low mass flow rates, the propellant utilization efficiency of the thruster is lower for air. The decrease of magnetic field intensity allows to increase the propellant utilization, but it also affects the thruster efficiency. The investigation of thruster operation with air at 250 km is presented. Even if a longer channel results beneficial, the study concludes that a compression and storage system may be necessary to sustain the thruster discharge.

In [120], the Electromagnetic Spacecraft Environment Simulator (EMSES) was used to investigate the plasma generation processes in the ABIE chamber. In [153], a dual-stage Hall thruster, similar to SITAEL's RAM-EP concept, is simulated with a fully kinetic 2D-3V particle-in-cell Monte-Carlo collision code. Simulations were performed considering xenon propellant and different mixtures of atmospheric species.

Theoretical investigations on the ABEP ionization processes were performed by Taploo et al. [118]. This study focuses on an ABEP concept similar to that proposed by [44, 56], based on Hall-type acceleration. An electron source placed before the acceleration region provides electrons to ionize the air flow. Simulations of the complex plasma chemistry in the ionization region are performed for different electron energies and for altitudes in the range 80-110 km, where the relatively high atmospheric pressure facilitates the air ionization. Based on the observation that negative ions can be generated by tuning the electron energy, the authors propose to operate the thruster alternating the production of positive and negative ions, thus removing the need of a hollow cathode.

Sarwar et al. [154] present a global and particle-in-cell modeling of a dual-stage ABEP concept operating at 150 km. No compression of the incoming flow is considered, neutral particles are simulated entering the ionization region with orbital velocity. The ionization stage relies on electro-magnetic confinement and a 200 eV electron beam. The balance of neutral and charged species is evaluated with a set of volume-averaged continuity equations. The effect of generalized electron energy distribution functions was investigated. Preliminary studies of kinetic effects and of plasma acceleration were performed using EDIPIC, a parallelized 2D-3V particle-in-cell code.

Modeling of air-fed plasma thrusters is typically performed by considering given propellant conditions at the thruster inlet. However, for ABEP concepts based on a passive intake there could be a wide region of overlap between the particle collection and ionization and the interactions between the intake and the plasma thruster become important.

A 0D-hybrid model of a complete ABEP concept is presented in [21]. The model is applicable to systems composed of a passive intake, an ionization stage, and an electrostatic acceleration stage. For each stage, the proposed physical description consists of a system of continuity equations for the different species coupled with the electron energy balance. The dynamics of heavy species is introduced in the model by using 3D Monte-Carlo methods, which allow to compute in the control volumes the particle transmission, the mean residence time, and the mean velocity of each species. Inter-particle collisions are neglected, whereas the CLL reflection model is adopted to describe gas-surface interactions.

Modeling of innovative concepts often relies on several assumptions and calibration parameters, which need to be validated by means of experimental measurements. Modeling efforts that rely on experimental data are presented by few authors. A calibrated model of an air-fed radio-frequency ion thruster is described in [150, 155]. The model couples discharge equations, used to simulate the balance of different species in the ionization chamber, with a model of the ion extraction, which depends on the grids geometry and voltage. The model was calibrated with experimental data of the air-fed RIT-10 [97]. The same dataset has been recently adopted by Lopez-Uricoechea et al. [156] to validate a 0-D steady-state model of radio-frequency ion thrusters operating with atmospheric propellant. The proposed model follows the approach of 0-D global models developed for conventional propellants, introducing neutral molecule dissociation and molecular ions.

In [124], the authors present a global model of a magnetized high-frequency plasma source intended as the ionization stage of an ABEP concept. The model considers the balance of charged and neutral particles and of electron energy in the control volume, where the plasma is confined by a magnetic field. The power is deposited into the plasma by means of an alternating electric field operating in the RF range. Experimental measurements were used to calibrate the model, using the input power (assumed as a constant) as a fitting parameter. To perform the comparison, the analysis focused on molecular propellant. Simulations were performed over a wide range of pressure levels, showing a good agreement of the model with the experiments and confirming that a lower pressure limit of 5 mPa exists for the proposed source design.

In [95], the model presented in [21] was calibrated on the experimental data available for SITAEL's HT5k Hall thruster. By tuning an anomalous electron diffusion coefficient and an effective ionization cost parameter, the model showed a good agreement with the experimental measurements in a wide range of operating conditions, providing insight of the thruster operation and on the discharge chemistry.

Finally, modeling efforts are fundamental to support the on-ground testing. As described in VLEO flow simulators section, simulations of VLEO flow sources and the flow-intake interactions allowed to adapt the intake design for on-ground testing. Plasma plume models were used in [79] to verify the representativeness of the particle flow generator and determine the sensitivity of the intake collection on the PFG-intake distance. As described in [49], thermo-chemical plasma modeling can be used to investigate the composition of the flow generated by a VLEO simulator.

Conclusions

Air-breathing electric propulsion (ABEP) systems have the potential to revolutionize space assets by allowing spacecraft to operate in very low Earth orbits for an extended period of time. The ABEP concept relies on a delicate balance that involves platform and mission design. Several factors shall be considered in the design trade-offs, including, e.g., the highly variable flow density and composition, the area available to collect the atmospheric flow, the solar arrays surface and orientation, the need to keep the spacecraft alignment, the constraints of payload accommodation and operation, the requirement imposed by the power management and the firing strategy. In this balance, however, the performance of the ABEP system plays a major role. Consequently, a central part of this review paper has been dedicated to analyse the performance of the two main stages of ABEP concepts. For the same reason, a section of the review has been dedicated to the verification of the intake and thruster performance, alone and integrated together.

In recent years the development of air-breathing electric propulsion gathered the efforts of an increasing number of research institutions and space industries. As the advantages of VLEO operation are becoming clearer, much has been done to develop the ABEP technology. Several concepts have been proposed and analysed. An improved understanding of the VLEO environment and of the influence of design parameters on the system performance has been achieved. This has led to a gradual consolidation of the system architecture. At the same time, the numerical investigation of air intakes has matured through the efforts of multiple research groups, which addressed almost independently several design solutions, reaching nonetheless similar results. Furthermore, different approaches to ionize, accelerate, and neutralize atmospheric propellants have been characterized. Thanks to the development of VLEO flow simulators, end-to-end testing of air-breathing systems have been performed. However, to date no ABEP concept has demonstrated the capability to perform full drag compensation.

In the authors opinion, further technology development towards real operative scenarios will require significant efforts in two main directions: (i) to improve the ABEP performance, both of the intake and of the thruster, and (ii) to improve the feasibility and representativeness of the technology verification strategies. As nuclear power architectures or laser beam methods for power transmission seem long term options, our analysis focuses solely on spacecrafts with solar electric generators.

Considering a conventional electric propulsion device and the compression levels currently achievable with passive intakes, effective ionization of the propellant would require to operate at low altitudes. Nevertheless, the corresponding high external atmospheric density would result in drag levels exceeding the thruster capabilities. If a thruster was capable of efficiently ionizing the incoming propellant at a density roughly one to two orders of magnitude lower than the one in conventional devices, full drag compensation could be achieved. As discussed in the review, it appears that a technological breakthrough related to intake compression or a novel thruster design compatible with very low density operation could be feasible in the near future, making ABEP a viable solution.

If the intake collection efficiency is low, either because of particle compression or because the intake inlet corresponds to a small fraction of the platform frontal area, high specific impulses are necessary, which makes electrostatic acceleration a preferred solution. In this case, one or several cathodes/neutralizers may be needed. Traditional hollow cathodes seem to be incompatible with atmospheric propellant due to the rapid contamination of the emitter material. Advanced cathode concepts more resilient to contamination are being investigated. However, these concepts usually require a non-negligible energy cost per electron, which may significantly increase the overall power consumption of the propulsion system. To date, cathode operation in air-breathing mode was never verified experimentally.

In general, significant work is being performed on ABEP mission analysis by several research groups. The consolidation of system design and the refinement of mission studies indicate a strong interest of institutional and commercial stakeholders on the application of the ABEP technology for the long-term exploitation of VLEOs. Nonetheless, a truly comprehensive mission study of a flight representative ABEP-based platform, including orbital propagation, spacecraft attitude, ABEP performance sensitivity to environmental conditions, discharge control, thrust strategy for orbital stability, eclipse and battery management, payload and link budget performance, and expected platform operating lifetime in comparison with traditional platforms still seems to be missing from the available literature. Moreover, the power distribution and processing systems of an ABEP platform should likely implement a control logic aimed at regulating and stabilizing the operation of the propulsion system, given the expected highly variable inlet flow properties.

Testing of air-breathing electric propulsion systems represents the main obstacle toward the development of the technology. Intake simulations typically neglect the interaction between the intake and the thruster. The creation of dedicated facilities, provided with representative flow generators, is necessary to perform the development and optimization of ABEP systems performance, and to investigate their operation in different conditions. The development of flow generation technologies and of ad-hoc diagnostics could support the ABEP development, while allowing for the investigation of particle-wall interactions and surface degradation phenomena. A possible strategy to verify the integrated ABEP technology is to adapt the tested intake design to compensate for the limitations of the recreated atmospheric flows.

Finally, The limits of on-ground testing can be overcome through in orbit experiments. In this regard, reducing the scale of the propulsion system could be a sound approach to minimize the cost and complexity of the demonstrator missions. Since traditional electric propulsion systems show a clear degradation of efficiency with the reduction of size and power, reducing the size of the thruster may represent a conservative benchmark for the ABEP concept.

Acknowledgements
The authors wish to acknowledge the longstanding mentoring of Mariano Andrenucci and his pioneering role in the field.

Authors' contributions
Conceptualization: Tommaso Andreussi; literature search and data analysis: Tommaso Andreussi, Eugenio Ferrato, Vittorio Giannetti; Writing - original draft preparation: Tommaso Andreussi, Eugenio Ferrato, Vittorio Giannetti; Writing - review and editing: Tommaso Andreussi, Eugenio Ferrato, Vittorio Giannetti. The author(s) read and approved the final manuscript.

Funding
Not applicable.

Availability of data and materials
Authors can confirm that all relevant data are included in the article and/or in the cited literature.

Code availability
Not applicable.

Declarations

Competing interests
The authors declare that they have no competing interests.

Received: 1 September 2022 Accepted: 19 October 2022
Published online: 15 December 2022

References

1. Singh LA, Walker ML (2015) A review of research in low earth orbit propellant collection. 75:15–25. https://doi.org/10.1016/j.paerosci.2015.03.001
2. Conley BR (1995) Utilization of ambient gas as a propellant for low earth orbit electric propulsion. Master thesis, Massachusetts Institute of Technology. http://hdl.handle.net/1721.1/31061. Accessed 15 Nov 2022.

3. McGuire TJ (1999) Aero-Assisted Orbital Transfer Vehicles Utilizing Atmosphere Ingestion. Master thesis, Massachusetts Institute of Technology
4. McGuire TJ, Sedwick RJ (2001) Aero-assisted orbital transfer vehicles utilizing atmosphere ingestion. 39th Aerospace Sciences Meeting and Exhibit. https://doi.org/10.2514/6.2001-840
5. Nishiyama K (2003) Air breathing ion engine concept. 54th International Astronautical Congress 3(October):383–390. https://doi.org/10.2514/6.iac-03-s.4.02
6. Drinkwater MR, Floberghagen R, Haagmans R, Muzi D, Popescu A (2003) GOCE: ESA's First Earth Explorer Core Mission. Springer Netherlands, Dordrecht, pp 419–432. https://doi.org/10.1007/978-94-017-1333-7_36
7. Tagawa M, Yokota K, Nishiyama K, Kuninaka H, Yoshizawa Y, Yamamoto D, Tsuboi T (2013) Experimental study of air breathing ion engine using laser detonation beam source. J Propuls Power 29(3):501–506. https://doi.org/10.2514/1.B34530
8. Crisp NH, Roberts PC, Livadiotti S, Oiko VT, Edmondson S, Haigh SJ, Huyton C, Sinpetru LA, Smith KL, Worrall SD, Becedas J, Domínguez RM, González D, Hanessian V, Mølgaard A, Nielsen J, Bisgaard M, Chan YA, Fasoulas S, Herdrich GH, Romano F, Traub C, García-Almiñana D, Rodríguez-Donaire S, Sureda M, Kataria D, Outlaw R, Belkouchi B, Conte A, Perez JS, Villain R, Heißerer B, Schwalber A (2020) The benefits of very low earth orbit for earth observation missions. Prog Aerosp Sci 117(September). arXiv: 2007.07699. https://doi.org/10.1016/j.paerosci.2020.100619
9. Spektor R, Jones KL (2021) A breath of fresh air: air-scooping electric propulsion in very low earth orbit. Cent Space Policy Strateg. https://csps.aerospace.org/sites/default/files/2021-08/Spektor-Jones_AirBreathing_20210318.pdf. Accessed 15 Nov 2022.
10. Zheng P, Wu J, Zhang Y, Wu B (2020) A comprehensive review of atmosphere-breathing electric propulsion systems. Int J Aerosp Eng 2020(4). https://doi.org/10.1155/2020/8811847
11. Fujita K, Noda A (2009) Rarefied aerodynamics of a Super Low Altitude Test Satellite. 41st AIAA Thermophysics Conference (June):22–25. https://doi.org/10.2514/6.2009-3606
12. Crisp N, Roberts P, Hanessian V, Sulliotti-Linner V, Herdrichd G, Garcia-Almiñana D, Kataria D (2022) A method for the experimental characterisation of novel drag-reducing materials for very low earth orbits using the satellite for orbital aerodynamics research (soar) mission. CEAS Space J 1–20. https://doi.org/10.1007/s12567-022-00434-3
13. Jacchia LG (1974) Variations in thermospheric composition: A model based on mass spectrometer and satellite drag data. J Geophys Res 79. https://doi.org/10.1029/ja079i013p01923
14. Hedin AE, Reber CA, Newton GP, Spencer NW, Brinton HC, Mayr HG, Potter WE (1977) A global thermospheric model based on mass spectrometer and incoherent scatter data msis, 2. composition. J Geophys Res (1896-1977) 82(16):2148–2156. https://doi.org/10.1029/JA082i016p02148
15. Hedin AE (1983) A revised thermospheric model based on mass spectrometer and incoherent scatter data: Msis-83. J Geophys Res Space Phys 88(A12):10170–10188. https://doi.org/10.1029/JA088iA12p10170
16. Hedin AE (1987) Msis-86 thermospheric model. J Geophys Res Space Phys 92(A5):4649–4662. https://doi.org/10.1029/JA092iA05p04649
17. Hedin AE (1991) Extension of the msis thermosphere model into the middle and lower atmosphere. J Geophys Res Space Phys 96(A2):1159–1172. https://doi.org/10.1029/90JA02125
18. Picone J, Hedin A, Drob D, Meier R, Lean J, Nicholas A, Thonnard S (2000) Enhanced empirical models of the thermosphere. Phys Chem Earth C Solar Terr Planet Sci 25(5):537–542. https://doi.org/10.1016/S1464-1917(00)00072-6
19. Picone J, Hedin AE, Drob DP, Aikin AC (2002) NRLMSISE-00 empirical model of the atmosphere: Statistical comparisons and scientific issues. J Geophys Res 107(A12):1468. https://doi.org/10.1029/2002JA009430
20. Crisp NH, Roberts PC, Romano F, Smith KL, Oiko VT, Sulliotti-Linner V, Hanessian V, Herdrich GH, García-Almiñana D, Kataria D, Seminari S (2021) System modelling of very low Earth orbit satellites for Earth observation. Acta Astronaut 187(June):475–491. https://doi.org/10.1016/j.actaastro.2021.07.004
21. Ferrato E, Giannetti V, Tisaev M, Lucca Fabris A, Califano F, Andreussi T (2022) Rarefied Flow Simulation of Conical Intake and Plasma Thruster for Very Low Earth Orbit Spaceflight. Front Phys 10(March):1–17. https://doi.org/10.3389/fphy.2022.823098
22. Bowman BR, Kent Tobiska W, Marcos FA, Valladares C (2008) The jb2006 empirical thermospheric density model. J Atmos Solar-Terr Phys 70(5):774–793. https://doi.org/10.1016/j.jastp.2007.10.002
23. Bowman B, Tobiska WK, Marcos F, Huang C, Lin C, Burke W (2008) A New Empirical Thermospheric Density Model JB2008 Using New Solar and Geomagnetic Indices. https://doi.org/10.2514/6.2008-6438. https://arc.aiaa.org/doi/abs/10.2514/6.2008-6438
24. Bruinsma S (2015) The dtm-2013 thermosphere model. J Space Weather Space Clim 5. https://doi.org/10.1051/swsc/2015001
25. Alken P, Thébault E, Beggan C, Amit H, Aubert J, Baerenzung J, Bondar T, Brown W, Califf S, Chambodut A, Chulliat A, Cox G, Finlay C, Fournier A, Gillet N, Grayver A, Hammer M, Holschneider M, Huder L, Hulot G, Jager T, Kloss C, Korte M, Kuang W, Kuvshinov A, Langlais B, Léger JM, Lesur V, Livermore P, Lowes F, Macmillan S, Magnes W, Mandea M, Marsal S, Matzka J, Metman M, Minami T, Morschhauser A, Mound J, Nair M, Nakano S, Olsen N, Pavón-Carrasco F, Petrov V, Ropp G, Rother M, Sabaka T, Sanchez S, Saturnino D, Schnepf N, Shen X, Stolle C, Tangborn A, Tøffner-Clausen L, Toh H, Torta J, Varner J, Vervelidou F, Vigneron P, Wardinski I, Wicht J, Woods A, Yang Y, Zeren Z, Zhou B (2021) International geomagnetic reference field: the thirteenth generation. Earth Planets Space 73(1). https://doi.org/10.1186/s40623-020-01288-x
26. Drob DP, Emmert JT, Meriwether JW, Makela JJ, Doornbos E, Conde M, Hernandez G, Noto J, Zawdie KA, Mcdonald SE, Huba JD, Klenzing JH (2015) An update to the horizontal wind model (hwm): The quiet time thermosphere. Earth Space Sci 2(7):301–319. https://doi.org/10.1002/2014EA000089
27. Mostaza Prieto D, Graziano BP, Roberts PC (2014) Spacecraft drag modelling. Prog Aerosp Sci 64:56–65. https://doi.org/10.1016/j.paerosci.2013.09.001

28. Livadiotti S, Crisp NH, Roberts PC, Worrall SD, Oiko VT, Edmondson S, Haigh SJ, Huyton C, Smith KL, Sinpetru LA, Holmes BE, Becedas J, Domínguez RM, Cañas V, Christensen S, Mølgaard A, Nielsen J, Bisgaard M, Chan YA, Herdrich GH, Romano F, Fasoulas S, Traub C, Garcia-Almiñana D, Rodriguez-Donaire S, Sureda M, Kataria D, Belkouchi B, Conte A, Perez JS, Villain R, Outlaw R (2020) A review of gas-surface interaction models for orbital aerodynamics applications. Prog Aerosp Sci 119(June). arXiv: 2010.00489. https://doi.org/10.1016/j.paerosci.2020.100675
29. Gallis MA, Torczynski JR, Plimpton SJ, Rader DJ, Koehler T (2014) Direct simulation monte carlo: The quest for speed. AIP Conf Proc 1628(1):27–36. https://doi.org/10.1063/1.4902571
30. Sinpetru LA, Crisp NH, Mostaza-Prieto D, Livadiotti S, Roberts PC (2022) Adbsat: Methodology of a novel panel method tool for aerodynamic analysis of satellites. Comput Phys Commun 275. https://doi.org/10.1016/j.cpc.2022.108326
31. Sheridan PL, Paul SN, Avendaño-Franco G, Mehta PM (2022) Updates and improvements to the satellite drag coefficient response surface modeling toolkit. Adv Space Res 69(10):3828–3846. https://doi.org/10.1016/j.asr.2022.02.044
32. Walsh J, Berthoud L, Allen C (2021) Drag reduction through shape optimisation for satellites in very low earth orbit. Acta Astronaut 179:105–121. https://doi.org/10.1016/j.actaastro.2020.09.018
33. Livadiotti S, Crisp NH, Roberts PCE, Oiko VTA, Christensen S, Maria Domínguez R, Herdrich GH (2022) Uncertainties and design of active aerodynamic attitude control in very low earth orbit. J Guid Control Dyn 45(5):859–874. https://doi.org/10.2514/1.G005999
34. Brieda L, Keidar M (2012). Development of the Starfish Plasma Simulation Code and Update on Multiscale Modeling of Hall Thrusters. https://doi.org/10.2514/6.2012-4015. https://arc.aiaa.org/doi/abs/10.2514/6.2012-4015
35. Andrews S, Berthoud L (2020) Characterising satellite aerodynamics in very low earth orbit inclusive of ion thruster plume-thermosphere/ionosphere interactions. Acta Astronaut 170:386–396. https://doi.org/10.1016/j.actaastro.2019.12.034
36. Mehta PM, Walker A, Mclaughlin CA, Koller J (2014) Comparing physical drag coefficients computed using different gas-surface interaction models. J Spacecr Rocket 51(3):873–883. https://doi.org/10.2514/1.A32566
37. Capon C, Smith B, Brown M, Abay R, Boyce R (2019) Effect of ionospheric drag on atmospheric density estimation and orbit prediction. Adv Space Res 63(8):2495–2505. https://doi.org/10.1016/j.asr.2019.01.013
38. Demetriades S (1962) Preliminary study of propulsive fluid accumulator. J Br Interplanet Soc 18:392–402
39. Berner F, Camac M (1961) Air scooping vehicle. Planet Space Sci 4:159–183. https://doi.org/10.1016/0032-0633(61)90130-1
40. Cann G (1975). A Space Electric Ramjet. https://doi.org/10.2514/6.1975-377. https://arc.aiaa.org/doi/abs/10.2514/6.1975-377
41. Minovitch M (1983) Solar powered, self-refueling, microwave propelled interorbital transportation system. 18th Thermophysics Conference. https://doi.org/10.2514/6.1983-1446. https://arc.aiaa.org/doi/pdf/10.2514/6.1983-1446
42. Minovitch M 1985) Self-refueling rocket propulsion for future space travel. 21st Joint Propulsion Conference. https://doi.org/10.2514/6.1985-1370. https://arc.aiaa.org/doi/pdf/10.2514/6.1985-1370
43. Di Cara D, Gonzalez del Amo J, Santovicenzo A, Carnicero Dominguez B, Arcioni M, Caldwell A, Roma I (2007) RAM Electric Propulsion for Low Earth Orbit Operation: an ESA study. 30th International Electric Propulsion Conference, Florence, Italy, IEPC-2007-162. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
44. Hruby V, Hohman K, Szabo J (2022) Air Breathing Hall Effect Thruster Design Studies and Experiments. 37th International Electric Propulsion Conference, Boston, Massachusetts, IEPC-2022-446. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
45. Diamant K (2010) A 2-stage cylindrical hall thruster for air breathing electric propulsion. 46th AIAA/ASME/SAE/ASEE Joint Propulsion Conference. https://doi.org/10.2514/6.2010-6522
46. Shabshelowitz A (2013) Study of RF Plasma Technology Applied to Air-Breathing Electric Propulsion. Phd thesis, University of Michigan
47. Schonherr T, Komurasaki K, Romano F, Massuti-Ballester B, Herdrich G (2015) Analysis of atmosphere-breathing electric propulsion. IEEE Trans Plasma Sci 43(1):287–294. https://doi.org/10.1109/TPS.2014.2364053
48. Romano F, Massuti-Ballester B, Binder T, Herdrich G, Fasoulas S (2017) Schönherr T (2018) System analysis and testbed for an atmosphere-breathing electric propulsion system using an inductive plasma thruster. Acta Astronaut 147:114–126. arXiv: 2103.02328. https://doi.org/10.1016/j.actaastro.2018.03.031
49. Andreussi T, Ferrato E, Paissoni CA, Kitaeva A, Giannetti V, Piragino A, Schäff S, Katsonis K, Berenguer C, Kovacova Z, Neubauer E, Tisaev M, Karadag B, Lucca Fabris A, Smirnova M, Mingo A, Le Quang D, Alsalihi Z, Bariselli F, Parodi P, Jorge P, Magin TE (2022) The AETHER project: development of air-breathing electric propulsion for VLEO missions. CEAS Space Journal. https://doi.org/10.1007/s12567-022-00442-3
50. Tisaev M, Ferrato E, Giannetti V, Paissoni C, Baresi N, Lucca Fabris A, Andreussi T (2021) Air-breathing electric propulsion: Flight envelope identification and development of control for long-term orbital stability. Acta Astronautica 191(2021):374–393. https://doi.org/10.1016/j.actaastro.2021.11.011
51. Ovchinnikov MY, Mashtakov YV, Roldugin DS (2022) Mathematical Modeling of the Dynamics of a Low-Flying Spacecraft with a Ramjet Electric Propulsion Engine. Math Model Comput Simul 14:452–465. https://doi.org/10.1134/S2070048222030139
52. Vaidya S, Traub C, Romano F, Rodriguez-Donaire S, Garcia-Almiñana D, Sureda M, Garcia-Berenguer M (2022) Development and analysis of novel mission scenarios based on atmosphere-breathing electric propulsion (abep). CEAS Space J 1–18. https://doi.org/10.1007/s12567-022-00436-1
53. Crandall P, Wirz R (2022) Air-breathing electric propulsion: mission characterization and design analysis. J Electr Propuls 1(12):931–937. https://doi.org/10.1007/s44205-022-00009-8
54. Fujita K (2004) Air Intake Performance of Air Breathing Ion Engines. J Jpn Soc Aeronaut Space Sci 52(610):514–521. https://doi.org/10.2322/jjsass.52.514

55. Hruby V, Pote B, Brogan T, Hohman K, Szabo J, Rostler P (2004) Air breathing electrically powered hall effect thruster. United States Patent US 6834492:B2
56. Pekker L, Keidar M (2012) Analysis of airbreathing hall-effect thrusters. J Propuls Power 28(6):1399–1405. https://doi.org/10.2514/1.B34441
57. Shabshelowitz A, Gallimore AD, Peterson PY (2014) Performance of a helicon Hall thruster operating with xenon, argon, and nitrogen. J Propuls Power 30(3):664–671. https://doi.org/10.2514/1.B35041
58. Romano F, Chan YA, Herdrich G, Traub C, Fasoulas S, Roberts PC, Smith K, Edmondson S, Haigh S, Crisp NH, Oiko VT, Worrall SD, Livadiotti S, Huyton C, Sinpetru LA, Straker A, Becedas J, Domínguez RM, González D, Cañas V, Sulliotti-Linner V, Hanessian V, Mølgaard A, Nielsen J, Bisgaard M, Garcia-Almiñana D, Rodriguez-Donaire S, Sureda M, Kataria D, Outlaw R, Villain R, Perez JS, Conte A, Belkouchi B, Schwalber A, Heißerer B (2020) RF Helicon-based Inductive Plasma Thruster (IPT) Design for an Atmosphere-Breathing Electric Propulsion system (ABEP). Acta Astronautica 176:476–483. arXiv: 2007.06397. https://doi.org/10.1016/j.actaastro.2020.07.008
59. Romano F, Herdrich G, Chan YA, Crisp NH, Roberts PCE, Holmes BEA, Edmondson S, Haigh S, Macario-Rojas A, Oiko VTA, Sinpetru LA, Smith K, Becedas J, Sulliotti-Linner V, Bisgaard M, Christensen S, Hanessian V, Jensen TK, Nielsen J, Fasoulas S, Traub C, García-Almiñana D, Rodríguez-Donaire S, Sureda M, Kataria D, Belkouchi B, Conte A, Seminari S, Villain R (2022) Design of an intake and a thruster for an atmosphere-breathing electric propulsion system. CEAS Space J. https://doi.org/10.1007/s12567-022-00452-1
60. Ferrato E, Giannetti V, Piragino A, Andrenucci M, Andreussi T (2019) Development roadmap of sitael's ram-ep system. 36th International Electric Propulsion Conference, Vienna, Austria, IEPC-2019-886. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
61. Schaaf SA, Chambre PL (1958) Flow of Rarefied Gases. Princeton University Press, Princeton. http://www.jstor.org/stable/j.ctt1m32248. Accessed 15 Nov 2022.
62. Crandall P, Wirz RE (2022). Air-Breathing Electric Propulsion Mission Characterization and Design Analysis. https://doi.org/10.2514/6.2022-2189. https://arc.aiaa.org/doi/abs/10.2514/6.2022-2189
63. Crandall P, Wirz RE (2022) Air-Breathing Electric Propulsion Spacecraft Performance and Aerodynamic Maneuverability. 37th International Electric Propulsion Conference, Boston, Massachusetts, IEPC-2022-444. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
64. Golikov A, Filatyev A (2022) Integrated optimization of trajectories and layout parameters of spacecraft with air-breathing electric propulsion. Acta Astronautica 193:644–652. https://doi.org/10.1016/j.actaastro.2021.06.052
65. Shao A, Madni AM, Wertz JR (2016). Quantifying the Effect of Orbit Altitude on Mission Cost for Earth Observation Satellites. https://doi.org/10.2514/6.2016-0974. https://arc.aiaa.org/doi/abs/10.2514/6.2016-0974
66. McGrath C, Lowe C, Macdonald M, Hancock S (2022) Investigation of very low Earth orbits (VLEOs) for global spaceborne lidar. CEAS Space J (0123456789). https://doi.org/10.1007/s12567-022-00427-2
67. Berthoud L, Hills R, Bacon A, Havouzaris-Waller M, Hayward K, Gayrard JD, Arnal F, Combelles L (2022) Are very low earth orbit (vleo) satellites a solution for tomorrow's telecommunication needs? CEAS Space J. https://doi.org/10.1007/s12567-022-00437-0
68. Cifali G, Misuri T, Rossetti P, Andrenucci M, Valentian D, Feili D (2011) Preliminary characterization test of HET and RIT with nitrogen and oxygen. 47th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit 2011. https://doi.org/10.2514/6.2011-6073
69. Filatyev A, Yanova O (2017) On the optimal use of electric ramjet for low-orbit spacecraft. Procedia Eng 185:173–181. https://doi.org/10.1016/j.proeng.2017.03.299
70. Barral S, Cifali G, Albertoni R, Andrenucci M, Walpot L (2015) Conceptual design of an air-breathing electric propulsion system. 34th International Electric Propulsion Conference, Kobe, Japan, IEPC-2015-271. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
71. Fujita K (2004) Air-intake performance estimation of air-breathing ion engines. Nihon Kikai Gakkai Ronbunshu, B Hen/Transactions of the Japan Society of Mechanical Engineers, Part B 70(700):3038–3044. https://doi.org/10.1299/kikaib.70.3038
72. Hisamoto Y, Nishiyama K, Kuninaka H (2012) Design of air intake for air breathing ion engine. 63rd International Astronautical Congress. Proceedings of the International Astronautical Congress, IAC: International Astronautical Federation, IAF. 9:7630–7634. ISSN 00741795.
73. Ozawa T, Imamura S, Nishiyama K, Tagawa M, Fujita K (2018) Numerical analyses of air intake for air breathing ion engine. JAXA Special Publication: Proceedings of the 50th Fluid Dynamics Conference/the 36th Aerospace Numerical Simulation Symposium, JAXA-SP-18-005
74. Shoda K, Kano N, Jotaki Y, Ezaki K, Itatani K, Ozawa T, Yamashita Y, Nishiyama K, Yokota K, Tagawa M (2022) Anisotropic molecular scattering at microstructured surface for rarefied gas compression inside air breathing ion engine. CEAS Space J. https://doi.org/10.1007/s12567-022-00430-7
75. Hohman K (2012) Atmospheric Breathing Electric Thruster for Planetary Exploration. NIAC Spring Symposium. https://www.nasa.gov/pdf/636899main_Hohman_Presentation.pdf. Accessed 15 Nov 2022.
76. Parodi P, Le Quang D, Bariselli F, Boccelli S, Alsalihi Z, Magin T (2019) Study of a collector-intake system for VLEO air-breathing platforms. International Conference on Flight vehicles, Aerothermodynamics and Re-entry Missions and Engineering (FAR). https://www.researchgate.net/publication/337060417_Study_of_a_collectorintake_system_for_VLEO_air-breathing_platforms
77. Parodi P (2019) Analysis and Simulation of an Intake for Air-Breathing Electric Propulsion Systems. https://doi.org/10.13140/RG.2.2.12966.34889. https://www.researchgate.net/publication/342159445
78. Cifali G, Andreussi T, Giannetti V, Leporini A, Rossodivita A, Andrenucci M, Barral S, Longo J, Walpot L (2016) Experimental Validation of a Ram-EP Concept Based on Hall Effect Thruster Technology. Space Propuls Conf 2016, SP2016-3125202
79. Andreussi T, Cifali G, Giannetti V, Piragino A, Ferrato E, Rossodivita A, Andrenucci M, Longo J, Walpot L (2017) Development and Experimental Validation of a Hall Effect Thruster RAM-EP Concept. 35th International Electric Propulsion Conference, Atlanta, Georgia, IEPC-2017-377. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.

80. Li Y, Chen X, Li D, Xiao Y, Dai P, Gong C (2015) Design and analysis of vacuum air-intake device used in air-breathing electric propulsion. Vacuum 120(PA):89–95. https://doi.org/10.1016/j.vacuum.2015.06.011
81. Romano F, Binder T, Herdrich G, Fasoulas S (2015) Air-intake design investigation for an air-breathing electric propulsion system. 34th International Electric Propulsion Conference. https://doi.org/10.13140/RG.2.1.2500.4646
82. Romano F, Binder T, Herdrich G, Fasoulas S (2016) Intake design for an atmosphere-breathing electric propulsion system. Space Propuls Conf 2016:SP2016-3124981
83. Binder T, Boldini PC, Romano F, Herdrich G, Fasoulas S (2016) Transmission probabilities of rarefied flows in the application of atmosphere-breathing electric propulsion 1786. https://doi.org/10.1063/1.4967689
84. Romano F, Espinosa-Orozco J, Pfeiffer M, Herdrich G, Crisp NH, Roberts PC, Holmes BE, Edmondson S, Haigh S, Livadiotti S, Macario-Rojas A, Oiko VT, Sinpetru LA, Smith K, Becedas J, Sulliotti-Linner V, Bisgaard M, Christensen S, Hanessian V, Jensen TK, Nielsen J, Chan YA, Fasoulas S, Traub C, García-Almiñana D, Rodríguez-Donaire S, Sureda M, Kataria D, Belkouchi B, Conte A, Seminari S, Villain R (2021) Intake design for an Atmosphere-Breathing Electric Propulsion System (ABEP). Acta Astronautica 187(May):225–235. arXiv: 2106.15912. https://doi.org/10.1016/j.actaastro.2021.06.033
85. Espinosa-Orozco J (2020) Intake Design, and Optimization for an Atmosphere-Breathing Electric Propulsion System. Master thesis, KTH Royal Institute of Technology
86. Kuznetsov Y, Flaksman Y (1977) Influence of the cross-sectional shape of the channel on the characteristics of free-molecular air intake vents. Trudy TsAGI 1863:25–29
87. Erofeev AI, Nikiforov AP, Popov GA, Suvorov MO, Syrin SA, Khartov SA (2017) Air-Breathing Ramjet Electric Propulsion for Controlling Low-Orbit Spacecraft Motion to Compensate for Aerodynamic Drag. Sol Syst Res 51(7):639–645. https://doi.org/10.1134/S0038094617070048
88. Erofeev AI (2018) Air intake in the transient regime of rarefied gas flow. TsAGI Sci J 49:137–148
89. Jackson SW (2017) Design of an Air-Breathing Electric Thruster for CubeSat Applications. PhD thesis, University of Colorado
90. Papulov A, Uzhinsky I (2020) Assessment and design of the active intake for vleo operation. Proceedings of the International Astronautical Congress, IAC. pp 12–14
91. Zheng P, Wu J, Wu B, Zhang Y (2021) Design and numerical investigation on the intake of atmosphere-breathing electric propulsion. Acta Astronautica 188:215–228. https://doi.org/10.1016/j.actaastro.2021.07.036
92. Zheng P, Wu J, Zhang Y, Wu B (2021) Optimization investigation of vacuum air-intake for atmosphere-breathing electric propulsion system. Proc Inst Mech Eng G J Aerosp Eng. https://doi.org/10.1177/09544100211029829
93. Zheng P, Wu J, Zhang Y, Zhao Y (2022) Design and Optimization of vacuum Intake for Atmosphere-Breathing electric propulsion (ABEP) system. Vacuum 195. https://doi.org/10.1016/j.vacuum.2021.110652
94. Prochnow F, Peter C, Wulfkühler JP, Drobny C, Tajmar M, Cook J, Sparkes M, O'neill W, Lockwood N (2022) Development of a novel passively ionizing air-breathing electric propulsion concept for low earth orbits. 37th International Electric Propulsion Conference, Boston, Massachusetts, IEPC-2022-420
95. Ferrato E, Giannetti V, Califano F, Andreussi T (2022) Atmospheric propellant fed hall thruster discharges: 0d-hybrid model and experimental results. Plasma Sources Sci Technol 31(7):075,003. https://doi.org/10.1088/1361-6595/ac7904
96. Semenkin AV, Chislov GO (1995) Study of anode layer thruster operation with gas mixtures. 24th International Electric Propulsion Conference, Moscow, Russia, IEPC-95-78. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
97. Cifali G, Dignani D, Misuri T, Rossetti P, Andrenucci M, Valentian D, Marchandise F, Feili D, Lotz B (2012) Completion of HET and RIT characterization with atmospheric propellants. Space Propulsion Conference 2012, Bordeaux, France, SP2012-2355386. 3AF – Association Aéronautique et Astronautique de France.
98. Andreussi T, Ferrato E, Paissoni CA, Kitaeva A, Giannetti V, Piragino A, Rossodivita A, Schaeff S, Katsonis K, Berenguer C, Kovacova Z, Neubauer E, Tisaev M, Karadag B, Fabris AL, Smirnova M, Mingo A, Le Quang D, Alsalihi Z, Bariselli F, Parodi P, Jorge P, Magin TE (2021) AETHER Air Breathing Electric THrustER: Towards Very Low Earth Orbit missions. Proceedings of the 72nd International Astronautical Congress, IAC, IAC-21,C4,5,x66453. InternationalAstronautical Federation, IAF. ISSN 00741795.
99. Gurciullo A, Fabris AL, Cappelli MA (2019) Ion plume investigation of a Hall effect thruster operating with Xe/N2 and Xe/air mixtures. J Phys D Appl Phys 52(46). https://doi.org/10.1088/1361-6463/ab36c5
100. Marchioni F (2020) Design and Performance Measurements of a Long Channel Hall Thruster for Air-Breathing Electric Propulsion. Master thesis, Politecnico di Torino
101. Marchioni F, Cappelli MA (2021) Extended channel Hall thruster for air-breathing electric propulsion. J Appl Phys 130(5). https://doi.org/10.1063/5.0048283
102. Dukhopelnikov DV, Riazanov VA, Shilov SO, Manegin DS, Sokolov RA (2021) Investigation of the laboratory model of a thruster with anode layer operating with air and nitrogen-oxygen mixture. XLIV Acad Space Conf 2318. https://doi.org/10.1063/5.0036251
103. Hu P, Shen Y, Yao Z, Mao W, Hu Y, Liu X (2021) Study of multi-cusped plasma thruster applied to Air-Breathing Electric Propulsion. Vacuum 190(March):110,275. https://doi.org/10.1016/j.vacuum.2021.110275
104. Romano F, Herdrich GH, Fasoulas S, Schönherr T, Crisp NH, Edmondson S, Haigh SJ, Lyons RE, Oiko VTA, Roberts PC, Smith KL, Becedas J, González G, Vázquez I, Braña Á, Antonini K, Bay K, Ghizoni L, Jungnell V, Morsbøl J, Binder T, Boxberger A, Garcia-Almiñana D, Rodriguez-Donaire S, Kataria D, Davidson M, Outlaw R, Belkouchi B, Conte A, Perez JS, Villain R, Heißerer B, Schwalber A (2017) Performance Evaluation of a Novel Inductive Atmosphere-Breathing EP system. 35th International Electric Propulsion Conference, Atlanta, Georgia, IEPC-2017-184. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
105. Romano F, Herdrich GH, Roberts PC, Boxberger A, Chan YA, Traub C, Fasoulas S, Smith KL, Edmondson S, Haigh SJ, Crisp NH, Oiko VTA, Lyons RE, Worrall SD, Livadiotti S, Huyton C, Sinpetru LA, Outlaw R, Becedas J, Dominguez RM, González D, Hanessian V, Mølgaard A, Nielsen J, Bisgaard M, Garcia-Almiñana D, Rodriguez-Donaire S, Sureda M, Kataria D, Villain R, Perez JS, Conte A, Belkouchi B, Schwalber A, Heißerer B, Magarotto M, Pavarin D (2019) Inductive Plasma Thruster (IPT) for an Atmosphere- Breathing Electric Propulsion System: Design and Set in Operation.

36th International Electric Propulsion Conference, Vienna, Austria, IEPC-2019-488. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.

106. Romano F, Chan YA, Herdrich G, Traub C, Fasoulas S, Roberts PC, Crisp N, Holmes BE, Edmondson S, Haigh S, Livadiotti S, Macario-Rojas A, Oiko VTA, Sinpetru LA, Smith K, Becedas J, Sulliotti-Linner V, Bisgaard M, Christensen S, Hanessian V, Jensen TK, Nielsen J, Garcia-Almiñana D, Garcia-Berenguer M, Rodriguez-Donaire S, Sureda M, Kataria D, Belkouchi B, Conte A, Seminari S (2021) Design , Set-Up , and First Ignition of the RF Helicon-based Plasma Thruster. Space Propulsion Conference 2020+1, SP2020-00247. 3AF – Association Aéronautique et Astronautique de France.
107. Charles C, Boswell RW, Laine R, MacLellan P (2008) An experimental investigation of alternative propellants for the helicon double layer thruster. J Phys D Appl Phys 41(17). https://doi.org/10.1088/0022-3727/41/17/175213
108. Chianese SG, Micci MM (2006) Microwave Electrothermal Thruster Chamber Temperature Measurements and Performance Calculations. J Propuls Power 22(1). https://doi.org/10.2514/1.15337
109. Clemens DE (2008) Performance Evalulation of the Microwave Electrothermal Thruster Using Nitrogen, Simulated Hydrazine, and Ammonia. Phd thesis, Pennsylvania State University
110. Schönherr T, Han G, Gürbüz C, Koizumi H, Komurasaki K (2015) First Experiments Towards an Atmosphere-Breathing PPT. 34th International Electric Propulsion Conference, Kobe, Japan, IEPC-2015-272. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
111. Skalden J, Ilic D, Herdrich G, Schonherr T (2017) Characterization and Optimization of Liquid-Ablative and Air-Breathing PPT , Part I : Thrust and Discharge Performance. 35th International Electric Propulsion Conference, Atlanta, Georgia, IEPC-2017-174. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
112. Ilic D, Skalden J, Herdrich G, Schonherr T, Komurasaki K, Koizumi H (2017) Characterization and Optimization of Liquid-Ablative and Air-Breathing PPT , Part II : Spectroscopic Investigation. 35th International Electric Propulsion Conference, Atlanta, Georgia, IEPC-2017-175. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
113. Schönherr T, Cepeda JLG, Skalden J, Ilic D, Herdrich G, Komurasaki K (2018) Coaxial Air-Fed Pulsed Plasma Thruster Research and Development for RAM- EP Application. Space Propuls Conf 2018: SP2018-116
114. Kirdyashev KP, Kubarfv YV (1995) Electrostatic instabilities of a magnetoplasmadynamic thruster in the inhomogeneous medium. 24th International Electric Propulsion Conference, Moscow, Russia, IEPC-95-10. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
115. Cifali G, Misuri T, Rossetti P, Andrenucci M, Valentian D, Feili D, Lotz B (2011) Experimental characterization of HET and RIT with atmospheric propellants. 32nd International Electric Propulsion Conference, Wiesbaden, Germany, IEPC–2011–224. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
116. Andreussi T, Ferrato E, Kitaeva A, Paissoni CA, Giannetti V, Piragino A, Pedrini D, Rossodivita A, SpA S, Schaeff IS, Berenguer C, Katsonis K, Kovakova Z, Neubauer E, Tisaev M, Karadag B, Fabris AL, Mingo A, Smirnova M, Lequang D, Alsahili Z, Bariselli F, Parodi P, Jorge P, Magin TE (2022) Characterization of an atmospheric propellant-fed hall thruster as a vleo simulator. 37th International Electric Propulsion Conference, Boston, Massachusetts, IEPC-2022-435. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
117. Pottinger S, Lappas V, Charles C, Boswell R (2011) Performance characterization of a helicon double layer thruster using direct thrust measurements. J Phys D Appl Phys 44(23). https://doi.org/10.1088/0022-3727/44/23/235201
118. Taploo A, Lin L, Keidar M (2021) Analysis of ionization in air-breathing plasma thruster. Phys Plasmas 28(9). https://doi.org/10.1063/5.0059896
119. Yasukochi T, Usui H, Miyake Y, Kawaguchi S, Yokota K, Tagawa M (2015) Numerical Simulation of ECR Plasma in Air Breathing Ion engine (ABIE): Neutral Gas Density Distribution and ECR Plasma Formation. 34th International Electric Propulsion Conference, Kobe, Japan, IEPC-2015-454 pp 1–7. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
120. Shirakawa R, Usui H, Miyake Y, Tagawa M, Nishiyama K (2019) Particle Simulations of ECR Plasma Generation in Air Breathing Ion Engine. 32nd International Symposium on Space Technology and Science and 9th Nano Satellite Symposium. pp 1–7. https://jglobal.jst.go.jp/en/detail?JGLOBAL_ID=202002274168897443. Accessed 15 Nov 2022.
121. Miya Y, Koda D, Nishiyama K (2019) Ignition performance evaluation of the ion source of the air breathing ion engine changing the magnetic field. Space Transportation Symposium FY2019, STEP-2019-002
122. Miya Y, Nishiyama K (2022) Performance evaluation of a plasma generator and ion optics for air-breathing ion engine. CEAS Space J. https://doi.org/10.1007/s12567-022-00422-7
123. Chernyshev SL, Loktionov EY, Sagalakov AE, Skvortsov VV, Filatyev AS, Uspensky AA (2021) Prospects of Infrared Lasers in Air-Breathing Electric Thrusters. Dokl Phys 66(11):307–310. https://doi.org/10.1134/S1028335821110045
124. Mrózek K, Dytrych T, Moliš P, Dániel V, Obrusník A (2021) Global plasma modeling of a magnetized high-frequency plasma source in low-pressure nitrogen and oxygen for air-breathing electric propulsion applications. Plasma Sources Sci Technol 30(12). https://doi.org/10.1088/1361-6595/ac36ac
125. Zheng P, Wu J, Zhang Y, Che B, Li J (2021) Simulation investigation of inductively coupled plasma generator for atmosphere-breathing electric propulsion system. Acta Astronautica 187:236–247. https://doi.org/10.1016/j.actaastro.2021.06.044
126. Qiongying Rea (2020) The concept study of air-breathing helicon thruster used in ultra-low orbit flight. Spacecr Environ Eng 37(1):17–24. https://doi.org/10.12126/see.2020.01.003
127. Pedrini D, Misuri T, Paganucci F, Andrenucci M (2017) Development of hollow cathodes for space electric propulsion at sitael. Aerospace 4(2). https://doi.org/10.3390/aerospace4020026
128. Tisaev M, Karadag B, LuccaFabris A, Ferrato E, Kitaeva A, Andreussi T (2022) Development of an air-breathing neutraliser as part of the AETHER system. 37th International Electric Propulsion Conference, Boston, Massachusetts, IEPC-2022-105. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
129. Diamant K (2012) Resonant cavity hollow cathode. 41st AIAA/ASME/SAE/ASEE Joint Propulsion Conference. https://doi.org/10.2514/6.2005-3662. https://arc.aiaa.org/doi/pdf/10.2514/6.2005-3662

130. Diamant KD (2009) Microwave cathode for air breathing electric propulsion. 31st International Electric Propulsion Conference, Ann Arbor, Michigan, IEPC-2009-015. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
131. Godyak V, Raitses Y, Fisch NJ (2007) RF Plasma Cathode-Neutralizer for Space Applications. 30th International Electric Propulsion Conference, Florence, Italy, IEPC-2007-266. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
132. Kamhawi H, Foster J, Patterson M (2012). Operation of a microwave electron cyclotron resonance cathode. https://doi.org/10.2514/6.2004-3819. https://arc.aiaa.org/doi/pdf/10.2514/6.2004-3819
133. Funaki I, Kuninaka H (2001) Overdense plasma production in a low-power microwave discharge electron source. Jpn J Appl Phys 40(Part 1, No. 4A):2495–2500. https://doi.org/10.1143/jjap.40.2495
134. Ahmed M, Karadag B, Fabris AL, Leone S, Cicala G, Velardi L (2022) A Novel Diamond-Based Plasma Neutraliser. 37th International Electric Propulsion Conference, Boston, Massachusetts, IEPC-2022-109. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
135. Gurciullo A, LuccaFabris A, Aaron K (2017) Alternative Neutralization Technologies Enabling the Use of Exotic Propellants in Electric Propulsion. 35th International Electric Propulsion Conference, Atlanta, Georgia, IEPC-2017-251. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
136. Reitemeyer M, Zschatzsch D, Holste K, Chen L, Klar PJ (2019) Applicability of electride materials for hollow cathodes. 36th International Electric Propulsion Conference, Vienna, Austria, IEPC-2019-604. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
137. Kleiman J, Iskanderova Z, Gudimenko Y, Horodetsky S (2003) Atomic oxygen beam sources: A critical overview. Eur Space Agency (Spec Publ) ESA SP (540):313–324
138. Hruby V, Homan K, Brogan T (2006) Combined Radio Frequency and Hall Effect Ion Source and Plasma Accelerator System. United States Patent US 0284562 AI
139. Hohman K, Olson L, Brogan T, Prebola J, Stuckey J (2008). Development of an Atomic Oxygen Source for Space Simulation Applications. https://doi.org/10.2514/6.2008-461. https://arc.aiaa.org/doi/abs/10.2514/6.2008-461.
140. Tagawa M, Yokota K, Kishida K, Okamoto A, Ishizawa J, Minton TK (2010) Effect of ultraviolet radiation from an oxygen plasma on the atomic oxygen-induced etching of fluorinated polymer. High Perform Polym 22:213–224. https://doi.org/10.1177/0954008309335941
141. Hisamoto Y, Nishiyama K, Kuninaka H (2011) Development Statue of Atomic Oxygen Simulator for Air Breathing Ion Engine. 32nd International Electric Propulsion Conference, Wiesbaden, Germany, IEPC-2011-294. pp 1–7. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
142. Hisamoto Y, Koizumi H, Nishiyama K (2010) An upperatmosphere simuratore using ecr discherge for air breathing ion engine. 61st International Astronautical Congress. IAC 8:6207–6211
143. Oiko VT, Roberts PC, Macario-Rojas A, Edmondson S, Haigh SJ, Holmes BE, Livadiotti S, Crisp NH, Smith KL, Sinpetru LA, Becedas J, Domínguez RM, Sulliotti-Linner V, Christensen S, Jensen TK, Nielsen J, Bisgaard M, Chan YA, Herdrich GH, Romano F, Fasoulas S, Traub C, Garcia-Almiñana D, Garcia-Berenguer M, Rodriguez-Donaire S, Sureda M, Kataria D, Belkouchi B, Conte A, Seminari S, Villain R (2020) Ground-based experimental facility for orbital aerodynamics research: Design, construction and characterisation. pp 12–14
144. Caledonia GE, Krech RH, Green BD (1987) A high flux source of energetic oxygen atoms for material degradation studies. AIAA J 25(1):59–63. https://doi.org/10.2514/3.9580
145. Cuthbertson JW, Langer WD, Motley RW (1992) Reflection of low energy plasma ions from metal surfaces. J Nucl Mater 196-198(C):113–128. https://doi.org/10.1016/S0022-3115(06)80017-6
146. Andreussi T, Ferrato E, Giannetti V, Piragino A, Cifali G, Andrenucci M, Paissoni CA (2019) Development status and way forward of sitael's air-breathing electric propulsion engine. AIAA Propuls Energy Forum Exposition. https://doi.org/10.2514/6.2019-3995
147. Bird GA (1994) Molecular Gas Dynamics and the Direct Simulation of Gas Flows. Oxford University Press, New York
148. Moe K, Moe MM (2005) Gas-surface interactions and satellite drag coefficients. Planet Space Sci 53(8):793–801. https://doi.org/10.1016/j.pss.2005.03.005
149. Rapisarda C (2021) Modelling and simulation of atmosphere-breathing electric propulsion intakes via direct simulation Monte Carlo: A study of the air-breathing ion engine. CEAS Space J. https://doi.org/10.1007/s12567-021-00414-z
150. Lotz B (2013) Plasma physical and material physical aspects of the application of atmospheric gases as a propellant for ion-thruster of the rit-type. Phd thesis, Justus-Liebig-University of Giessen
151. King ST, Walker MLR, Chianese SG (2014) Atmospheric electric propulsion mission performance tool. J Spacecr Rocket 51(3):931–937. https://doi.org/10.2514/1.A32235
152. Garrigues L (2012) Computational study of hall-effect thruster with ambient atmospheric gas as propellant. J Propuls Power 28:344–354. https://doi.org/10.2514/1.B34307
153. Marano MF (2021) Kinetic particle-based plasma model of air-breathing Hall thruster. Master thesis, Politecnico di Torino
154. Sarwar S, Khrabrov A, Kaganovich I, Sydorenko D (2022) Global and particle-in-cell modeling of air-breathing plasma engines. 37th International Electric Propulsion Conference, Boston, Massachusetts, IEPC-2022-445. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
155. Feili D, Lotz B, Meyer B, Loeb H, Collingwood C, Smirnova M, Khartov S, Cara DD, Bulit A (2013) Testing and comprehensive modeling of a gie utilizing atmospheric propellants. 33rd International Electric Propulsion Conference, Washington, DC, IEPC-2013-354. https://www.electricrocket.org/index.php?page=iepc-article-repository. Accessed 15 Nov 2022.
156. Lopez-Uricoechea J, Lev D, Walker MLR (2022) 0-d composition and performance analysis of an air-breathing radiofrequency ion thruster. J Electr Propuls 1(1). https://doi.org/10.1007/s44205-022-00013-y