Analysis of Atmosphere-Breathing Electric Propulsion

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To extend lifetime of commercial and scientific satellites in LEO and below (100-250 km of altitude) the recent years showed an increased activity in the field of air-breathing electric propulsion as well as beamed-energy propulsion systems. However, preliminary studies showed that the propellant flow necessary for electrostatic propulsion exceeds the mass intake possible within reasonable limits, and that electrode erosion due to oxygen flow might limit the lifetime of eventual thruster systems. Pulsed plasma thruster can be successfully operated with smaller mass intake, and operate at relatively small power demands which makes them an interesting candidate for air-breathing application in LEO, and their feasibility is investigated within this study. Further, to avoid electrode erosion, inductive plasma generator technology is discussed to derive a possible propulsion system that can handle gaseous propellant with no harmful effects.

Nomenclature

E = discharge energy per pulse
F_{D} = drag force imposed on satellite
f = discharge frequency
h = orbital altitude
m_{bit} = mass shot per pulse
n = number density
t = orbital lifetime

I. Introduction

Carrying propellant for attitude and orbit control on satellites in an Earth orbit results in an increased satellite mass, and, therefore, yields higher costs for manufacturing and launch of the spacecraft. Electric space propulsion helped in the past to reduce the mass requirements compared to standard chemical propulsion as a result of the superior specific impulse, but limitations remain with regards to lifetime and lower

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orbits.

A reduction of the on-board propellant is necessary to go beyond these limitations. The idea for using the residual atmosphere of Earth as propellant is not new,[1] but the question remains whether existing technology can be suitably applied to fulfill this purpose. This applies not only to the capability of using the mixture of nitrogen and oxygen and components thereof in the thruster, but also relates to the feasible operation at low mass flow rates and low power, and reasonable lifetime of the propulsion system in comparison to the lifetime of the satellite.

Several concepts were proposed in recent years for air-breathing electric propulsion on satellites, hence, this study summarizes briefly the conclusions so far, and derives design aspects that need to be kept in mind. To do so, it is further necessary to look at the actual environmental conditions of the residual atmosphere, its composition, and the impact on a possible satellite. This is to be discussed with regard to the various altitudes.

The information are then used to derive a design proposal for an air-breathing pulsed plasma thruster. A performance estimation is included as well as a proposal for mission scenarios.

Further, inductive technology can increase thruster lifetime significantly if used in an electrodeless design. Performance estimations and discussions of such a device are discussed in this study.

II. Application and Requirements

A. Need for air-breathing propulsion

Earths atmosphere does not have a sharp border, but rather gradually extends into space. This means that the closer a satellite or spacecraft is orbiting Earth, the more drag force it will be exposed to. This drag force will eventually slow down the satellite, and the orbit altitude will decrease, thereby further increasing the drag. This effect is typically rather small for orbits above 400 km with the resulting orbit decay to remain acceptable within the lifetime of the satellite (see Figure 1).

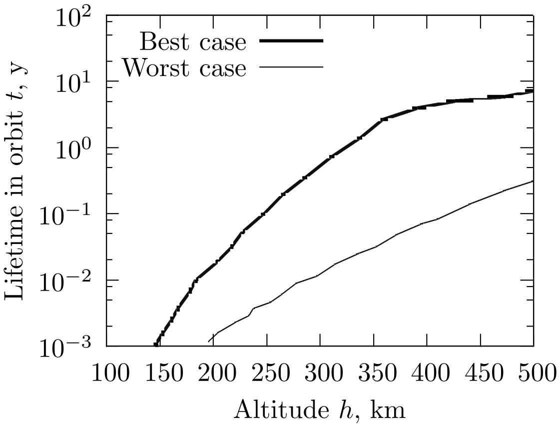


Figure 1: Lifetime for propulsion-less satellites. Best case refers to a ballistic coefficient of 200 kg/m^2 starting at minimum solar activity, worst case to a ballistic coefficient of 20 kg/m^2 at maximum solar activity. Reproduced from Larson and Wertz.[2]

For satellites below 400 km of altitude, the requirement for Δv to keep the satellite in orbit during the envisaged lifetime can be substantial. For orbits around 200 km and below, the atmospheric drag is significantly stronger, limiting the lifetime of a satellite without propulsion to a few days. Even with a propulsion system, a huge fraction of satellite mass is required for the propellant to compensate the drag, and to stay in orbit. Making use of the residual atmosphere as a potential propellant supply would, thus, enable many new missions.

B. Possible missions

From the above-mentioned possibility to use the residual atmosphere, several mission scenarios can be derived:

1. For satellites orbiting in a LEO between 250 and 400 km of altitude, the atmospheric gases can replace the on-board propellant. The mission profile in terms of lifetime of the satellite will remain the same, but due to the reduced mass for the propulsion system on the satellite, more commercial/scientific payload can be installed.

2. For satellites between 250 and 400 km, atmospheric gases can be partially added to the on-board propellant, and thereby stretch the supply to extend the lifetime of the satellite.

3. For orbits below 250 km, partial or complete supply of atmospheric gases to the propulsion system can reduce the on-board propellant requirement to a point where certain space missions become feasible. Extensions of lifetime far beyond the few days to weeks can be realized.

4. For very small satellites where on-board propellant is usually a too strong mass constraint, air-breathing propulsion might enable the mission at all.

5. For atmospheres below 160 km (the very low Earth orbit - VLEO), air-breathing technology might enable an extension of the lifetime of a satellite far enough, so that a scientific mission can be considered.

In an ESA study,[3] it was concluded that missions with air-breathing technology in an orbit above 250 km might not be competitive enough with standard electric propulsion, hence, the following focus lies in the stretch between 100 and 250 km of altitude.

III. Review of previous work

A. Atmospheric environmental conditions

For a consideration of air-breathing technology, an estimation of the composition and actual available mass flow of the residual atmosphere in VLEO is necessary. At these levels of altitude, the atmospheric conditions including density, temperature, and conductivity change significantly with altitude, and, thus, the mixing ratio of the residual species - O2, N2, and atomic oxygen varies as well. Studies on the determination of the atmospheric conditions of Earth started with the dawn of spacefaring and has been an ongoing process ever since. Concurrently, modeling techniques of the atmospheric conditions with respect to temperature, total density, partial density, etc. were developed based on the empirical data derived from satellites and atmospheric probes. The 1976 International Standard Atmosphere was a first standard to describe the atmosphere as a function of altitude, based on studies that started as early as the 1960s.[4] With the increasing availability of atmospheric data collected in space, NASA Goddard SFC created the MSIS model (Mass Spectrometer and Incoherent Scatter) to include the composition of the residual atmosphere in the modeling output.[5,6] This first model enabled modeling of global temperature and partial densities of N_2, O_2, O, He, Ar, and H for orbital altitudes of 120 to 150 km.

Subsequent expansions and revisions of the model included more data from satellites and scatter radars. The MSIS-83 model[7] extended the altitude range to 80 to 220 km, and the MSIS-86 version included atomic nitrogen in the computable species.[8] Revisions of the lower thermosphere (120 km and below) were included in the MSISE-90 model as well as an extension into the mesosphere and lower atmosphere.[9]

Further improvements of the NASA MSISE model were conducted at the Naval Research Laboratory (NRL) resulting in the NRLMSISE-00 version.[10,11] New features like anomalous oxygen were added, and further verification by in-space measurements resulted in an improved model. The 2000 version is still in use and currently the most advanced basis for an estimation of orbital conditions.

Additionally, a second model called the JB2006 model was developed based on the early Jacchia approach and modeling efforts, and it has an improved representation of thermospheric density variations. The total mean density approximation is assumed to exceed the ability even of the most recent NRL model, but it cannot provide data on the composition of the residual atmosphere, and might not be suitable for further considerations of the thrust performance. Further improvements of the model resulted in the most recent JB2008 version with new temperature equations and reduced errors.[12]

The two models are refined continually, and the current de facto standards are summarized in the European ECSS standard ECSS-E-ST-10-04C (issued 2008).[13]

Essentially, two models exist currently with different modeling baselines. Therefore, the accuracy and the output values differ between them. The NRLMSISE-00 model is the currently most advanced estimation for atmospheric conditions with regard to composition, i.e., partial densities. The second model, JB2006, shows a better representation of the total particle density as function of altitude, but can, however, not predict well the composition of the residual atmosphere. It should therefore only be used for an estimation of the drag force, whereas the former model can be used for an estimation of the composition (N_2, N, O_2, O, anomalous O, He, Ar, and H).

The atmosphere is not only the supply for the necessary propellant for air-breathing propulsion systems, but also the cause for the drag force on the entire satellite. For a 1 m^2-surface satellite, this drag force was estimated previously in a Hall thruster study.[14] The drag coefficient varies with altitude and shape of the satellite between 2.1 and 2.7.[15] For the altitudes of concern, a mean value of 2.2 can be assumed. Using the data for the total mean density from the JB2006 model for different sizes of satellite surfaces at mean solar activity, Figure 2 shows the total drag imposed on the satellite. For a full drag compensation, the mean thrust force needs to be at least equal to the mean drag to justify the application of air-breathing technology. The considered satellite sizes include the standard-sized cubesat of (10 cm)^3, a typical size of a propulsion-equipped small satellites 0.3 m^2, and a standard size of 1 m^2.

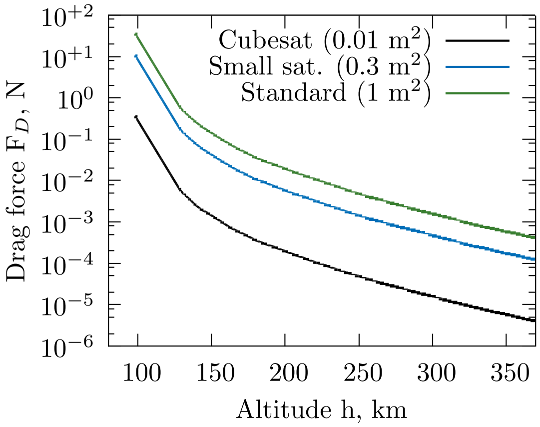


Figure 2: Drag force as a function of altitude for mean atmospheric conditions

The composition of the atmosphere at each altitude changes and can affect the performance of the eventual propulsion system. Derived from the NRLMSISE-00 model, the number densities of the individual species are plotted in Figure 3. For the altitudes of interest (lower than 250 km), the number density of anomalous oxygen is negligible and, therefore, no concern for the further discussion of the propulsion system. With the noble gases (He, Ar) usually less harmful to thruster systems and the number density of hydrogen rather small to the other species, the most dominant and, thus, design-driving species are N_2, N, O_2 and O. An eventual thruster system would need to be evaluated with a gas mixture pertinent to its target altitude.

B. Air intake

An intake is necessary to collect the residual atmosphere. Several studies were conducted in the past regarding a mechanical collector.[3,16] Electromagnetic concepts were discarded due to the relatively small degree of ionization in lower Earth orbits as can be seen in Figure 3. A collimator is also considered in the studies, as pressure and mass flow rate necessary for a stable operation of electric propulsion systems need to be higher than the values of the residual atmosphere. It was, however, previously concluded that the compression will depend on many geometrical aspects, and a proper discussion can only be made on a pre-existent thruster design.

The atmosphere intake needs to obtain as much gas as possible while keeping the particle velocity high for the eventual gas feed. Again, geometry plays a key role, but a slender spacecraft might be favorable for a increased capability of intake. It was concluded that collection efficiencies of 40 % at compression factors

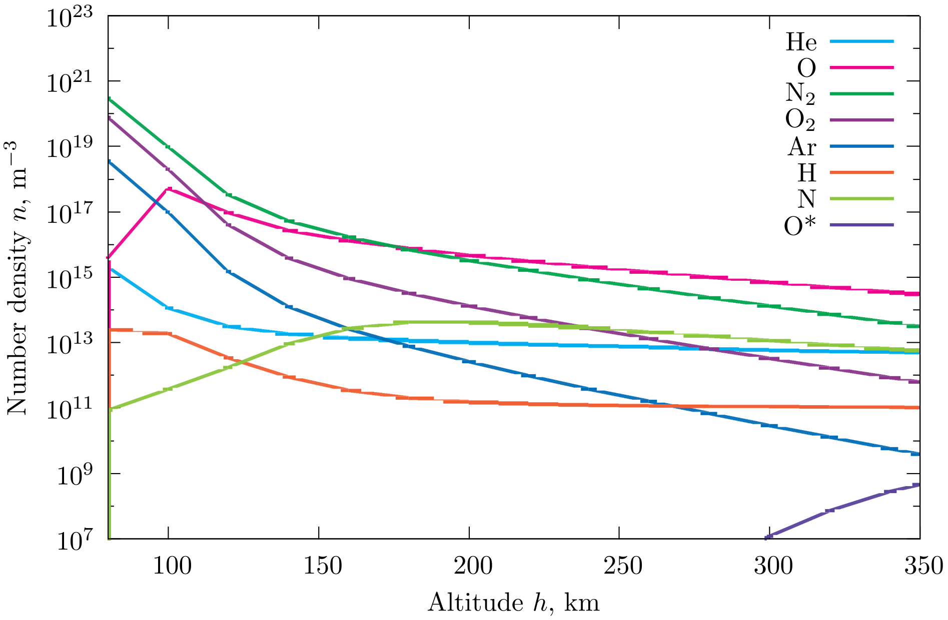


Figure 3: Atmospheric composition for mean solar and geomagnetic activities (NRLMSISE-00 model - F10.7 = $F10.7_{avg}$ = 140, A_p = 15); O* refers to anomalous oxygen.

of 100-200 can be achieved with typical designs.[3, 16, 17] Recent experimental work confirmed the theoretical results.[18]

Due to the impact of the particles on the intake wall, the wall temperature will increase and thermal balance design of the spacecraft needs to include the heat flux[16] as well as possible erosion and corrosion of the intake walls. As the intake needs to be sufficiently large, the spacecraft needs to allow such tremendous geometric components. The choice for air-breathing propulsion is, thus, a design-driving criterion. By and large, all previous studies showed that a pressure of roughly 1 mPa is achievable at the point of injection into the thruster head.

C. Electric propulsion concepts

1. Electrothermal propulsion

Resistojets and arcjets have played a substantial role in electric propulsion for long time, and are the most successful commercial application thereof. However, when considering the usage of residual atmosphere, the question of lifetime becomes dominant. Most thermal arcjet developments rely on oxygen-free propellants like hydrogen, hydrazine or ammonia.[19] Erosion of the nozzle throat in particular due to a high-temperature oxygen flow limits the general lifetime. The same reasoning applies to the resistive body in resistojets, and it is therefore unlikely to think of a concept of electrothermal propulsion system using air. As all atmospheres of the inner planets include oxygen or oxygen compounds, application in other planets orbits seems comparably unlikely.

2. Electrostatic propulsion

Electrostatic propulsion including ion thrusters (GIE, RIT, etc.), Hall effect thrusters (SPT, TAL), FEEP, etc., are current research focus, and therefore most existing studies on air-breathing, or rather atmosphere-breathing, electric propulsion is within this field.

ABIE - air-breathing ion engine - was one of the first air-breathing electric propulsion systems to be proposed.[1] Using a microwave discharge, the thruster with a performance of 10 mN/kW was designed to handle very low input pressures in the order of a few mPa to about 0.5 Pa,[20] still higher than the above-estimated 1 mPa. Mission application was thought to be in altitudes of about 150-200 km, and computation showed that 10^{18} m^{-3} in particle density could be achieved with the proposed intake.[16] The estimated size of the target satellite is 0.9 m^2 resulting in a drag of about 50-100 mN (see Figure 2). Thus 5-10 kW of on-board power supply are necessary under the given performance data for full drag compensation.

In a different study, a RIT-10 thruster was tested on the applicability of nitrogen and oxygen.[21] Using 450 W and 8.5 sccm of either gas, an optimum thrust of 5 mN was achieved. Hence, one derives about the same 10 mN/kW as for the ABIE system. However, the surface area of a RIT-10 is only about 0.02 m^2. The drag at the target orbits exceeds again the maximum achievable thrust, so full drag compensation seems equally unlikely. Grid erosion was studied, but it was concluded that a long lifetime even at operation with oxygen can be achieved.[22]

Alternatively to ion thrusters, Hall thrusters are of higher interest due to the higher thrust density. It is, therefore, possible to achieve the same thrust levels with a much smaller surface area, and thereby reducing the resulting drag. The US-American Busek company was the first to describe an air-breathing Hall thruster.[23] The concept was later refined into a propulsion concept for a glider in the Martian atmosphere,[17] thus, most experiments were conducted with CO_2 or a mixture thereof showing thrust-to-power ratios up to 30 mN/kW at 90-130 sccm. Erosion seemed negligible.

In another study, a French PPS®1350 SPT was operated with nitrogen, and a mixture of N_2/O_2.[21] The application of 2.5 mg/s and 1 kW resulted in about 20 mN/kW of thrust performance. Here, anode erosion was considered a significant problem within the few hundred hours of operation. More suitable electrode materials were recommended. However, this phenomenon could be reduced by application of a magnetic-shielding configuration.[24]

Computational studies on the feasibility of air-breathing Hall effect thrusters were conducted. Regarding VLEO around 80-95 km of altitude, a very high drag can be derived, hence, a necessary thrust-to-power ratio of at least 13 mN/kW was concluded.[25] As the total necessary thrust is in the order of 10 N and even higher, this would require several 100 kW of power input for full drag compensation. Such low orbits seem therefore impossible in the near future.

Another study investigated an orbital altitude of 250 km where the necessary drag compensation is significantly smaller.[14] The model concluded that the necessary mass flow rate of at least 2.5 mg/s would be difficult to be achieved at inlet pressures of only 1 mPa. Therefore, the study concluded that intermediate propellant storage is necessary, resulting in an increased system weight and complexity.

3. Electromagnetic propulsion

Only few electromagnetic propulsion systems were successfully tested in space, and recent research does suggest that steady-state devices require a sufficient amount of power (more than a few kW) for a stable and efficient operation. The thrust density values of steady MPD thrusters are likely the highest among electric propulsion, but the necessary mass flow rates might be too much a constraint for a breathing operation. However, pulsed MPD thrusters (PPT) have much smaller power requirements and can easily operate at a few watts. Several past laboratory models operated with gas instead of the typical solid or liquid propellant.[26] Especially, in recent times, Princeton University conducted an extensive study on GPPT using argon. In the 1960s, PPT using nitrogen and xenon as propellant were studied with a thrust-to-power ratio using N_2 ranging between 10-20 mN/kW, with the highest values at low energy (158 J) and high mass bit (348 µg). In a different study, a 540 J-GPPT was developed that was ignited by injecting about 10^{20} m^{-3} of nitrogen. Further optimization of the design might lead to smaller values necessary for proper ignition. A recent study with argon operated at a few J of discharge energy, but required a very high pulse frequency of up to 4 kHz to operate efficiently without losses of injected propellant. However, if propellant is abundant, the thrust efficiency becomes obsolete, and the above-stated thrust-to-power ratio becomes the dominant design criterion. Here, values of 6 mN/kW were achieved with a few µg of argon per shot. It was further derived that cathode erosion might require further attention.

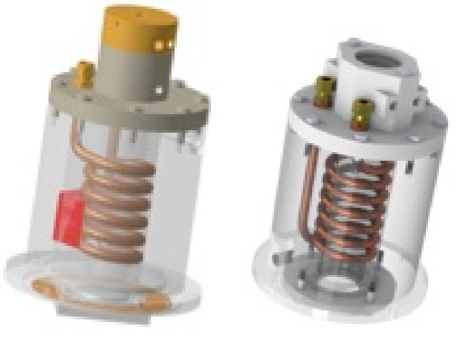
D. Inductive plasma sources

As erosion of the electrodes in electrostatic and electromagnetic propulsion can eventually become a constraint for the lifetime, electrodeless concepts are regarded in this study as well. Especially, inductively heated plasma flows show promising features for an air-breathing application.

Two miniaturized inductively heated plasma sources are currently being developed within a cooperation between Baylor University (Waco, TX, USA) and IRS (Stuttgart, Germany). These sources IPG6-S (Stuttgart) and IPG6-B (Baylor) are set in operation and qualification phases have been performed using working gases such as air, oxygen and carbon dioxide.[27]

For the initial characterization of IPG6-S, a cavity calorimeter has been built to measure the plasma power and the respective efficiencies in order to assess both the facilitys and the plasma sources operational envelope. A complete map of power consumption, power coupling and plasma power has been done for air as working gas at a pressure range between 60 and 410 Pa. This corresponds to several experiments with injected mass flow rates between 20 and 400 mg/s. As a result of the measurements mean specific enthalpy range for IPG6-S has been determined to vary from 1 to 7.5 MJ/kg. The power supply works in both continuous and pulsed modes. First experiments in the latter mode have been conducted by adjusting the pulse and pause times (1-10 ms). Moreover, tests using CO_2 as working gas have been performed. The obtained data are compared with respective conditions using air and they represent the first steps in the characterization of the generator with this gas.

The IPG6 test facilities at Baylor University (IPG6-B) and the University of Stuttgart (IPG6-S) have a very similar general setup, which slightly differs as the development stands in the stepwise improvements in both. The external geometry of IPG6 is of 230 mm length and 130 mm diameter. In Figure 4 the isometric CAD models of the two plasma generators (IPG6-B and IPG6-S) are shown.



(a) IPG6-S (b) IPG6-B

Figure 4: Inductive plasma generators as an example for electrode-less propulsion systems.

The plasma generators currently operate at moderate power levels of up to 20 kW and operating frequencies of 4 MHz in the IPG6-S or 15 kW and 13.56 MHz in the IPG6-B. The electrodeless design clearly favors IPG6 as an propulsion system in an operational environment where chemically aggressive propellants such as carbon dioxide (spacecraft waste, Martian atmosphere) and pure oxygen can be used (drag compensation in LEO).

IV. New concepts

A. Air-breathing PPT

1. Performance estimation

Based on the previous design proposals for air-breathing EP technology, a conceptual design for an air-breathing pulsed plasma thruster can easily be derived. Herein, the intake is likely to provide 1 mPa of gas pressure. Further compression and propellant processing might be necessary before injection into a RAM-GPPT. From past PPT developments, it can be concluded that a higher thrust-to-power is achievable when using a coaxial configuration. Unlike ablative PPT, the amount of injection mass can be varied independently from the energy, so an optimum ratio is to be expected. Previous results with nitrogen suggest that specific impulses of about 1000 s per (J/µg) can be considered.[28] Current state-of-the-art performance of GPPT with roughly 20 mN/kW can be increased by further optimization of the electrode design, discharge optimization, reduction of electric losses, etc. So far, no experimental values for oxygen exist, which increases the difficulties of a performance estimation. However, as propellant is mostly dissociated in higher altitudes, the propellant utilization efficiency will be higher than in terrestrial GPPT, and even higher values for T/P can be expected. For an energy-to-mass ratio of 5 J/g, a specific impulse for nitrogen of 5,000 s can be expected with a thrust-

to-power ratio of about 30 mN/kW (currently: 10-15 mN/kW).
Assuming a mission where a full drag compensation is required for a small satellite of 0.3 m^2 surface area, the drag derived from Figure 2 immediately defines the required power by the thrust-to-power ratio. PPT have the advantage that the discharge frequency can be regulated to vary the power level without interfering much with the performance per pulse. That is, for a given power, it is possible to ignite the thruster with a high frequency, thereby requiring low energy and small mass per pulse, or to ignite at low frequency with scaled quantities. As an example, Figure 5 shows the results of discharge energy and mass shot per pulse for 100 Hz and 4 kHz.

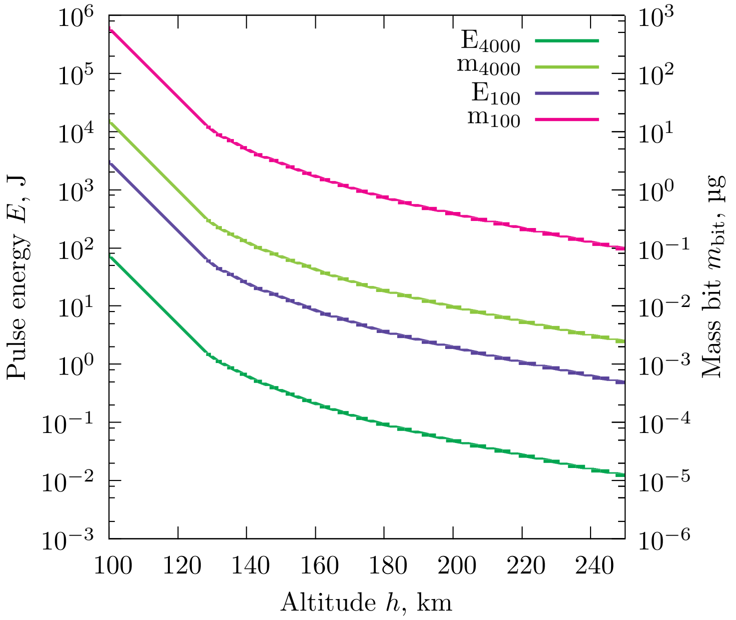


Figure 5: Estimated performance for an air-breathing PPT on a 0.3 m^2-sized satellite at full drag compensation (Index refers to the pulse frequency).

From the estimation, it can be derived that a combination of 1 J of discharge energy at 220 km of altitude requires less than 1 μg per shot. For a pulse frequency of 100 Hz this means less than 0.1 mg/s of mass flow into the thruster. These values seem promising for a realization, but obviously require experimental confirmation.
Storage of the propellant and compression of the inflow is necessary to enable a proper injection, and will increase the system mass. However, as seen before, other electric propulsion will require similar hardware, and this poses therefore no disadvantage. As the PPT can use much less energy and mass to create thrust, the hardware mass might even be significantly smaller than what is necessary for, e.g., electrostatic propulsion. By and large, it is clear that further optimization of GPPT is necessary to find optimum conditions for the combination of energy and mass input, and to increase specific impulse and T/P ratio concurrently.

2. Possible mission scenarios

As shown in the previous part, RAM-PPT have the potential of enabling drag compensation even for very small satellites, as the performance can be scaled by frequency, and as PPT can be operated at much smaller power levels than other electric propulsion devices. Even if storage and compression of the atmospheric gases seems a technical challenge, partial drag compensation could still be considered. One problem that a propulsion system will face in this scenario is that the satellites orbit is decaying. Hence, the drag, the input mass flow rate, etc. will change, and the thruster must have a very wide range of operation envelope. This is not always given for electrostatic propulsion where optimum performance is sensitive to many parameters.
In PPT, however, the overall performance can be regulated by the frequency as was shown. That means, that if a satellite is decaying from an orbit of 250 down to 100 km of altitude, the frequency can be adjusted in a way that the performance per pulse (mass bit and discharge energy) remains constant. The problem

of the thruster head is shifted to a challenge on the injection control and the propellant handling system. By doing so, the lifetime of the satellite can be extended in a way to enable further scientific analysis or technology demonstration. With the data given in Section 1, and assuming a constant discharge energy of 5 J, the discharge frequency needs to be adjusted to produce the corresponding mean thrust. The results are plotted in Figure 6.

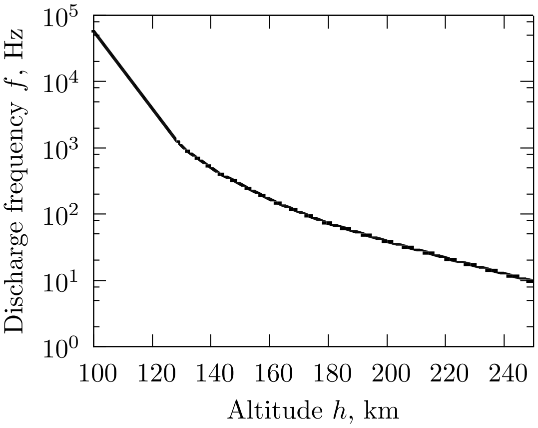


Figure 6: Discharge frequency for a constant discharge energy of 5 J (and $m_{\text{bit}} = 1$ µg) and full drag compensation.

Obviously, if full drag compensation can be achieved then altitude will no decrease, so the data given in Figure 6 rather refer to the best case. In a real case where power might be limited, the orbit will gradually decay, but at a smaller speed than shown in Figure 1. As too many parameters are involved to provide absolute numbers, it is, however, clear that lifetime can be extended substantially if the realizable discharge frequency is not too far off from the necessary one.

B. Inductive propulsion

A well-educated guess for the thruster relevant parameters can be given on basis of the experience with IPG3 and as a derivation from the bulk enthalpies that are known for IPG6. Here, values of even more than 10 MJ/kg were determined. Given this as an upper limit plasma jet kinetic energy a maximum velocity of 5 km/s can be derived. Using the typical mass flow rates (several tens of mg/s) roughly 4.4 mN per 1 mg/s can be estimated. These properties correspond pretty well with the scenarios for altitudes larger than 120 km as shown in Figure 2 if some 10 mg/s propellant mass flow rate is assumed.

Moreover, the IPG designs are already qualified for the oxygen operation for long operational times. In the ongoing procedure IPG6 has to be assessed for this application. System analysis will then have to show whether a pure down scaling is needed and whether this has to be combined with intermittent operation.

V. Conclusion

The necessity for air-breathing electric propulsion is given, and previously-studied concepts show the general feasibility in terms of operation and thruster lifetime. Concerns are nevertheless given for sufficient pressure and mass flow for an in-orbit application, especially with ion thrusters due to their small thrust density.

Orbits below 150 km experience an almost exponential rise in drag force with lower altitudes, and will be even more challenging for real application. As studies showed, orbits below 100 km are unrealistic in their power requirements for an air-breathing system. It is therefore wise to consider an altitude of 150-250 km for a first demonstration. PPT, although not yet experimentally studied with atmospheric gases extensively, show promising performance features, and offer the advantageous capability of scaling the performance by the discharge frequency. Again, experimental verification seems inevitable for the future.

Inductive propulsion technology can help to relieve the lifetime constraints coming from electrode erosion. A scaled-down model for a performance derivation is, thus, recommendable to estimate the feasibility of such a device.

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