Research and Development on Electric and Advanced Propulsion at IRS

IEPC-2017-480

Presented at the 35th International Electric Propulsion Conference
Georgia Institute of Technology • Atlanta, Georgia • USA
October 8 – 12, 2017

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Abstract: More than 30 years of experience have been gained in electric propulsion at IRS. Recent developments within the field of electric propulsion are summarized and foremost results are highlighted. This includes the current arcjet developments at IRS as well as the moderate to high power steady state self-field and applied-field MPD thrusters. Here, significantly relevant results were achieved for the AF MPDT SX3. An inductive system currently still named IPG6-S is under investigation as air breathing propulsion system within the European Union project DISCOVERER. The hybridization of both high power arcjet in series with a high power inductively heated source leads to the advanced thruster TIHTUS, a system that has flexibility in propellant and, additionally, a flexibility in throttability of thrust and specific impulse. An IEC based thruster concludes the incomplete list of electric propulsion systems that are under investigation at IRS.

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I.Introduction

IRS has gathered several decades of experience in the development, operation, characterization and qualification of various plasma sources. Among them are steady state self-field and applied field magnetoplasmadynamic (MPD) sources, thermal arcjet devices, inductively heated plasma sources and hybrid plasma systems[1-3]. These plasma systems are in application for aerothermodynamic testing, heat shield material characterization[4-8], electric space propulsion[9-16] and terrestrial plasma technology (i.e. technology transfer)[17-19].

Inductively heated plasma generators have originally been developed to cope with chemically aggressive working gases for the IRS plasma wind tunnel PWK3. Such gases are CO_2, O_2 and H_2O vapour. The electrodeless design enables additionally a pure plasma which engages the potential for aerothermochemical investigations in the field of heat shield material catalysis[6, 8, 20], nitridation and oxidation[21].

Moreover, the high power inductively heated plasma sources developed at IRS were respectively characterized and modelled in order to provide both an increased understanding and an experimental database for the applications[2, 3, 22].

In addition, air breathing electric propulsion concepts are currently under investigation within the European Union project DISCOVERER on basis of inductive plasma coupling. The respective designs benefit from the electrodeless principle enabling the operation with chemically reactive atmospheric gases[23].

Both aerothermodynamics and electric propulsion are flanked by development of modelling activities aiming at the respective simulation of atmospheric entries under non-equilibrium conditions[21, 24] and, in addition, the behaviour of both plasma sources for plasma wind tunnels and electric propulsion and their respective flow conditions[25].

Magnetohydrodynamic plasma configurations are investigated for two reasons: One is to be seen in the fact that the understanding and optimization of magnetoplasmadynamic plasma sources necessitates both experimental characterization[2] and modelling. The same applies to the influence of plasma flows and boundary layers by magnetic fields e.g. in order to mitigate heat fluxes as they are experienced by surfaces at stagnation point configuration[26, 27].

The paper provides an overview of the current activities in electric propulsion at IRS. For the lunar mission BW1 reference two electric propulsion systems have been developed: TALOS a 1 kW arcjet for fast branches of the trajectory such as crossing of the Van-Allen belt and ADD SIMP-LEX a pulsed PTFE-driven MPD thruster system. The TALOS development has been amended by a low power arcjet called VELARC (Very Low Power Arcjet) and the PPT developments were extended into thermal PPTs operating at low bank energies. Reasons for this are manifold: While TALOS and VELARC have new applications within ESA's Clean Space initiative, the latter has become a reference thruster for a standardization activity in cooperation with ESA. Within this project IRS is systematically investigating the impact and significance of test facilities with respect to the operation of one and the same thruster. A standardized Langmuir probe system is used in order to perform the comparison. While relevant activities were finalized, the investigations at ESA-ESTEC are planned for spring next year.

Within the context of ESA's Clean Space initiative IRS could approve arcjets as competitive systems for deorbit at EoL and orbit raising feed backing on launcher size and/or applied payload[28].

The current PPT developments are of concern for a wider range of capacitor bank energy. The ADD SIMP_LEX system was qualified at a TRL between 6 and 7 due to the final test campaigns in the DLR facility STG. Here, the system was successfully tested at the whole i.e. both thruster and proto-flight PPU were tested simultaneously[12, 29, 30]

For CubeSat developments such as CAPE (CubeSat Atmospheric Probe for Education)[31] the miniaturized PPT PETRUS is the reference for the 2U deorbit module of the capsule MIRKA2. This thruster is currently under development in collaboration with ESA[32, 33].

A further development is related to a Thermal-Inductive Hybrid Thruster of the University of Stuttgart (TIHTUS). This consists of arcjet and ICP stage. While the arcjet generates plasmas with steep radial gradients in the plasma's variables, the ICP stage is used to heat the relatively cold gas layer at the plume's edge. The arcjet used is HIPARC-W (High-Power Arc Jet- Water-Cooled). This is a 100 kW thruster with segmented anode such that nozzle length can be varied. TIHTUS' ICP stage is represented by the IRS' IPG7 (Inductively Heated Plasma-Generator). It is a continuous IPG with frequencies between 0.5 and 1.5 MHz at max. plate power of 180 kW[2, 3, 16, 34]. Experimental and numerical investigations have been extended to assess TIHTUS' behaviour better, to characterize the plasma and to assess its function in presence of the two discharges. New measurement techniques have been set in operation: Mach-Zender-Interferometry and LAS[35, 36].

Investigations have been carried out on self-field magnetoplasmadynamic (SF-MPD) thrusters (T) between 100 kW and 1 MW[15] and steady state applied field MPDT at maximum 100 kW. These thrusters are candidates for interplanetary missions as they achieve high exhaust velocity with likewise high thrust density. All thrusters have been operated in steady state mode with run times of up to several hours. The work is accompanied by numerical codes, allowing the calculation of the MPD thrusters and a comparison with experimental data. The current scenario for SF

MPDT looks at the comparison of steady state systems with and without water-cooling (as preparation for flight capability) as the impact to the fall regions at the thrusters' electrodes, in particular the anode, may be of positive nature concerning the thrust efficiency. The steady state AF-MPDT ZT1 and SX3 have been characterized at powers between 6 and 10 kW (ZT1) and some and more than 100 kW (SX3). The latter led to breakthrough results referring to the achieved thruster relevant parameters such as thrust efficiency, thrust and specific impulse. A summary of the outcome is presented. Detailed analysis is depicted in references.[37, 38]

Low altitude orbits are accompanied by drag to the S/C enforcing an early decay of the orbit. For drag compensation, propulsion systems are needed, requiring propellant on-board. An atmosphere-breathing electric propulsion system ingests the residual atmosphere's particles via an intake and uses them as propellant for an electric thruster. Applicable to any planet with atmosphere, the system might allow a S/C to orbit for an un-limited period without propellant on board. A new range of altitudes (130-350 km, Earth) for continuous operation would become accessible, enabling new scientific missions at reduced costs. IRS has decades of heritage on development of inductively heated plasma generators (IPG). They are electrode-less- issues of electrode erosion are eliminated. Characterizations of IPG using O_2 and CO_2 as propellants exist and show significant electric-to-thermal coupling efficiencies. This summarizes intake analyses using the IRS code PicLas and highlights the IRS IPG-based thruster.[39-41].

The Inertial Electrostatic Confinement (IEC) is a plasma confinement principle with different operation modes. The jet mode is investigated, which occurs with jet extraction relevant for space propulsion applications. Two modes were observed, the tight jet mode and the spray jet mode[10]. Depending on propellant, discharge pressure and power these modes occur at different conditions and characteristics. Emission spectroscopic investigations show a different plasma species composition of confinement and extraction plasma and the transition between those modes has been examined with a high-speed camera. The applicability of IEC plasma sources for space propulsion systems is discussed with respect to the experimental results for the jet energies assessed by an advanced Faraday probe system[42, 43]. The works are concluded by numerical analyses using PicLas and by a development of linear IEC based thrusters in cooperation with Gradel Srl, Luxemburg.

The list is almost completed by a water propulsion system where the propellants (H_2 and O_2) are produced in orbit via electrolysis. In fact, this system is rather combustion-based, however, from the point of view of the propellant generation it can also be considered as a secondary electric propulsion system. The strengths of such a system are evident: Besides the fact that a green propellant is in use, which significantly reduces needed safety efforts, the system offers a maximum flexibility with respect to the tank distribution and with respect to the storage aspects concerning hydrogen[44].

At last IRS is currently assessing a high precision radiation based propulsion system which is planned as a high precision attitude control actuator. This system is foreseen for purposes such as formation (orientation) and extremely small impulse bits for applications such as satellite interferometers[45].

II. Pulsed Plasma Thrusters at IRS

Pulsed Plasma Thrusters (PPT) have been developed at IRS since 15 years. In addition, this work was strongly flanked by the continuous collaboration with the Komurasaki lab. at the University of Tokyo[9, 12, 15, 23, 29, 30, 32, 33, 46-51]. I conjunction with a strong cooperation with RIAME MAI, Moscow, and Kurtschatov Institute, Moscow, these activities led to the advanced pulsed ablative thruster ADD SIMP-LEX. Here, the strong interlink allowed for a significant optimization of ADD SIMP-LEX referring to its thruster relevant properties i.e. thrust efficiency (ca. 30 %), impulse bit and specific impulse (see the table below). The already existing data bases at RIAME, additional thrust measurement campaigns with thrusters from the University of Southampton and Surrey additionally flanked the two research and development contracts from the DLR agency with a strong level of verification. The same applies to the successful reproduction of the thruster relevant parameters of ADD SIMP-LEX at the University of Tokyo, an item that represents a strong element for standardization e.g. referring to inter laboratory comparisons. Finally, this led to the successful verification of the operational behavior of the thruster system by the successful comparison of results from optical diagnostic tools at the University of Tokyo and results from integral measurements but also local measurements of the magnetic flux at IRS. In its final configuration the ADD SIMP-LEX thruster system including both thruster head and proto-flight PPU achieved TRL 6 by the successful tests of the system in DLR's STG facility in Göttingen, Germany.

Due to the significant number of advantages of PPTs such as the absence of toxic and dangerous materials, the absence of pressurized regimes and mechanically active parts, the low costs for the thruster, its throttability, its compact design, robustness and reliability it was decided to choose a miniaturized PPT for the IRS mission CAPE (CubeSat Atmospheric Probe for Education)[31]. An analysis of both MHD-based and thermal thrust fraction clearly shows that miniaturized PPTs are dominated by thermal acceleration effects rather than MHD-based acceleration effects. This is also confirmed by the magnetic Reynolds number which just will become too small if the thruster size is respectively small.

In general, a PPT consists of capacitor bank, a pair of electrodes, an isolator, propellant and an ignitor, whereas, the propellant can be solid, gaseous or fluid. Due to characteristics of being non-toxic, temperature resistant and easy to handle Polytetrafluoroethylene (PTFE) is one of the most propellant used in PPTs. Moreover, when using PTFE additional tank structures are unnecessary; it has low out-gassing rates and has been used in space missions already. PPTs can be divided into two classes: the parallel plate and the coaxial design. Parallel plate PPTs are majorly driven by electromagnetic effects, whereas, coaxial design also can be utilized to use electro thermal effects[52]. Coaxial PPTs, which make use of thermal effects utilizing a hollow propellant cylinder if operated with a solid propellant.
Nevertheless, coaxial PPTs also can be designed to use electromagnetic effects. If a solid propellant is used a breech fed design is very common[53]. As discussed in references [52] and [54] low energy PPTs (E < 20 J) are dominated by electro thermal effects and build as coaxial thrusters. In comparison to that high energy PPTs (E ≥ 20 J) are dominated by electromagnetic effects and, therefore, use the design of parallel plates.
In total three thrusters - ADD SIMPLEX, PET1 and PET2 - already have been successfully developed as well as tested and characterized at IRS[55]. A fourth PPT, called PETRUS, is currently under development. The PPTs at IRS cover an energy range from 2.72 J, which is the coaxial PET2, to 67.6 J being the parallel plate accelerator ADD SIMP-LEX (**SOURCE**). The following table 1 gives an overview of performance data of the PPTs at IRS.

	ADD SIMP-LEX	**PET1**	**PETRA (PET2)**	**PETRUS2.0**
Type	iMPD	Electro thermal	Electro thermal	Electro thermal / iMPD
Design	Parallel plate accel.	Coaxial	Coaxial	Coaxial
Dimensions	370 x 240 x 120 mm	Ø 32 x 55 mm	Ø 17 x 29.1 mm	Ø 12 x 50 mm
Mass	6.5 kg	489 g	180.72 g	≤ 500 g (incl. PPU)
Propellant	PTFE	PTFE	PTFE	PTFE
Propellant mass	Bis zu 43 kg	4 g	1.825 g	1.77 g – 2.7 g
Capacity	80 µF	1.5 µF	1.36 µF	4 µF – 6 µF
Voltage	1300 V	2500 V	2000 V	1200 V - 1600 V
Energy	67.6 J	3 J	2.72 J	2.88 J – 7.68 J
Pulse frequency	1 Hz	1 Hz	1 Hz	0.25 - 1 Hz
Mass bit	53.38 µg	43.4 µg	47.76 µg	1.77 µg - 2.7 µg (theor.)
Impulse bit	1373 µNs	61.7 µNs	72 µNs	15 µNs - 41µNs (theor.)
Amount of pulses	More than 2 million	100000	38211 (theor.)	1 million (theor.)
Specific impulse	≤ 2718 s	140 s	154 s (theor.)	882 s - 1567 s (theor.)
Thruster per pulse	1.373 mN	0.0617 mN	0.072 mN	15 µN - 41.46 µN (theor.)
Input power	< 100 W	< 4 W	< 4 W	5 - 8 W

Table 1: Overview of performance data of PPTs at IRS.

Figure 1: PPTs at IRS. From left to right; ADD SIMP-LEX, PET1, PETRA (PET2) and PETRUS.

Figure 1 depicts the respective thrusters. The pulsed magnetoplasmadynamic thruster ADD SIMP-LEX developed and characterized at IRS is a side-fed and PTFE-fueled parallel plate accelerator for small satellite platforms[56]. The thruster operates in the energy range of 67.6 J and is designed to work as a primary or secondary propulsion system. Furthermore, ADD SIMP-LEX has already repeatedly demonstrated promising results during thermal vacuum tests and extensive operational lifespan testes exceeding one million pulses[57]. Moreover, the thruster reaches a thrust efficiency of ≤ 32.2 %. Due to the auspicious performance results and a technical readiness level of six (TRL 6) this thruster operates at the top of solid fueled PPTs. Subsequently, ADD SIMP-LEX builds the baseline for further thruster design investigations and developments at IRS, bringing the research of scalability of PPTs and their power supply units decisively forward. Within the development of ADD SIMP-LEX different electrode materials and shapes, various types of capacitors and their arrangement, propellants as well as propellant feeding systems was successfully investigated. Especially, polytetrafluoroethylene (PTFE) - also known as Teflon® - was extensively examined. Furthermore, liquid propellants like water have been used and investigated[48]. In order to obtain a self-controlled propellant flow into the discharge region of the thruster a permeable WHIPOX-fibre ceramic was used[48]. Moreover, a proto-flight model Power Processing Unit (PPU) already is developed and ready to use.

The most recent thruster at IRS is the breech fed PPT PETRUS. The development also is based on experiences gathered with ADD SIMP-LEX as well as the PETs. PETRUS is designed to be a low weight as well as space saving low energy (in range of E < 8 J) thruster. The total volume including thruster and PPU shall be limited to 0.5 U and a mass of about 500 g. Compared to the PETs the ablation area of PETRUS is an annulus and not a cavity.

Due to the cavity inside the PTFE cylinder of the PETs the ablation area changes during thruster operation. Furthermore, in a lifetime test it was investigated that the ablation surface inside the cavity heavily is engaged by debris. Both are responsible for a strong performance decay over time. Subsequently, the approach of an annulus ablation surface, where the PTFE cylinder is pulled forward by springs, shall be investigated and tested. Main area of application of PETRUS is the operation as a primary propulsion system for CubeSat based missions. One application for PETRUS is the "CubeSat Atmospheric Probe for Education" (CAPE). CAPE is a Nanosatellite mission which shall perform a controlled deorbit maneuver in order to accomplish atmospheric measurements and to investigate high performance ablative heat shield materials[56]. Within the mission analysis, it is assumed that the starting orbit of CAPE is the ISS orbit at an altitude of about 400 km. In order to perform a controlled deorbit a cluster of PETRUS thrusters shall be used. Due to the compact and scalable design of PETRUS, the thruster also can be used as a secondary propulsion system, e.g. AOCS for CubeSat and other satellite platforms. Moreover, a software tool has been implemented within the development phase in order to receive design-driving parameters for the PETRUS type thruster. This software tool outputs parameters like impulse bit, mass bit, and specific impulse regarding to mission specific input parameters, e.g. Δv or provided energy. Here, it has to be mentioned that the software is implemented for low energy ranges ($E \leq 10$ J)[32].

During the investigation of first PETRUS designs it was determined that only a maximum of 1/5th of the PTFE surface is ablated per pulse. Moreover, the centered inboard igniter electrode does not create the initial breakdowns at various positions along the igniter head surface. Combining both effects, this leads to heavy charring at the regions where the initial breakdown does not occur (Figure 2)[58]. Even though multiple cathode and igniter designs were tested only minor improvements regarding the reduction of charring and the increase of the effective ablation area could be reached. Correspondingly, PETRUS was redesigned leading to an even smaller and lighter thruster (Table 2).

Table 2: Dimensions and masses of PETRUS and PETRUS2.0.

	PETRUS	**PETRUS2.0**
Anode	$Ø_{inside}$ = 15 mm	$Ø_{inside}$ = 8 mm
Cathode	$Ø_{outside}$ = 8 mm	$Ø_{outside}$ = 1.5 mm
Igniter	Inside the cathode	Surrounding the cathode
Propellant	$Ø_{outside}$ = 15 mm, $Ø_{inside}$ = 8 mm	$Ø_{outside}$ = 8 mm, $Ø_{inside}$ = 2 mm
Thruster dimensions (w/o capacitors)	Ø 24 mm x 50 mm	Ø 12 mm x 50 mm
Thruster mass (w/o capacitors)	74 g	11 g

The results of the redesign can be seen in the upper of Figure 2. Both thruster configurations (PETRUS and PETRUS2.0) were operated at a capacitor bank energy of 5.12 J and 1600 V. Moreover, the camera settings, i.e. ISO, aperture, focal length and exposure time are the same.

Comparing the plasma distribution during one discharge of both thrusters it can be seen that the PTFE surface of PETRUS2.0 is completely and homogenous covered in plasma. As already described before the design of the original

PETRUS suffered from a non-uniform PTFE ablation and plasma creation during operation. In order to draw a conclusion about the charring behavior of both thrusters performed 2072 pulses at a pulse energy of 5.12 J. The difference of the charring can be seen in the lower of Figure 2.

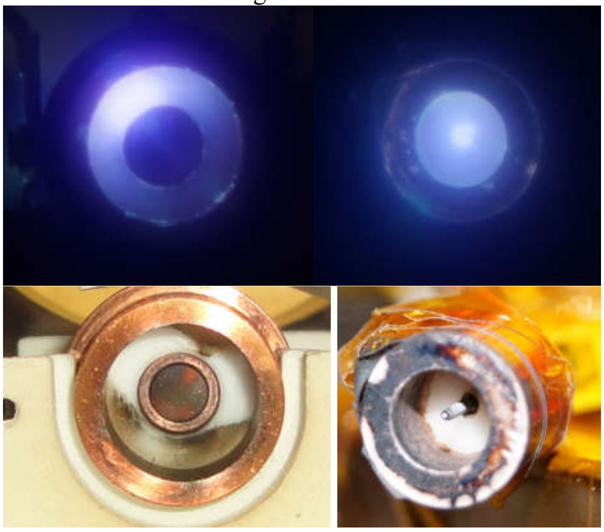

Figure 2: Upper left, PETRUS, upper right PETRUS2.0. Both operated at an energy of 5.12 J and at a voltage of 1600 V. Lower left and lower right both thrusters after 2072 pulses.

In the lower left of Figure 2, the front of the original PETRUS after the lifetime test can be seen. During the test campaign, it was recognized that the initial breakdown from the igniter electrode was alternating between the top left and top right. There, about 2/5th of the PTFE surface were not affected by charring. By performing the same test campaign, but using PETRUS2.0 the results of the lifetime test significantly improved. Expect of some minor sections at the surrounding of the PTFE surface no charring is observed (bottom left in Figure 2).

To sum up, four PPTs at IRS currently exist and being under development, respectively. These thrusters cover an energy range from 2.72 J (coaxial) to 67.6 J (parallel plate), with ADD SIMP-LEX having a TRL of six. In order to provide a PPT, which merges the requirements of the CubeSat mission “CAPE” the thruster PETRUS/PETRUS2.0 is currently is under development. Since the original thruster design of PETRUS suffered from heavy charring and a malfunction of the ignition distribution the thruster had to be redesigned towards an even smaller as well as lighter version, called PETRUS2.0. First test campaigns already showed a very homogenous plasma creation and charring seems not to be an issue anymore. The next step is to further investigate PETRUS2.0 regarding electrode length, voltage discharge curve, impulse bit and mass bit. Additionally, the electrode shape, i.e. length and diameter shall be examined. Finally, yet important, a Power Processing Unit for PPT CubeSat applications is currently under development at IRS.

III.Steady State MPD Thrusters at IRS

Steady state self-field MPD Thrusters have been investigated at IRS since decades[15, 59]. In fact, these propulsion devices benefit from a respectively high thrust density while offering a comparably significant specific impulse. Several designs were investigated systematically: Cylindrical systems, nozzle equipped systems and, in addition, also approaches where the anode, a sub structure to be significantly cooled due to the electrode losses, was designed in a radiatively cooled manner. The activity was strongly flanked by modelling[60].

The investigation of steady state applied field thrusters was engaged at IRS more than 10 years ago[61, 62]. Although self-field MPD thrusters are now proven to fulfill mission requirements such as they were needed for a flexible manned mission to Mars, it has to be stated that they suffer from losing their self-induced field given that the power (i.e. the current) is too low (say somewhere below 200 kW). Steady state applied field MPD thrusters can fill this gap easily as the respective loss of the self-field is quasi compensated by an independent applied field.

A. Steady state self-field MPD Thrusters

Nowadays self-field MPD plasma generators are in use at IRS for the purpose of aerothermodynamic testing[1]. However, one field of research is linked to both MHD and aerothermodynamic testing: Here, the steady state self-field MPD Plasma Generator is in use to produce a significantly ionized Argon flow which is used as incident flow for plasma probes equipped with variable magnetic fluxes[26]. This is achieved by a modular approach where a number of permanent magnets can be staked in a flexible way. In order to be able to adequately rebuild both flow and interaction between magnetic field and boundary layer the condition was very well characterized making use of Langmuir probes (n_e, T_e), Pitot pressure probes (Total pressure, estimate of Mach number) and, in addition, spectroscopic investigation of the boundary layer in front of the probe with and without magnetic field[63, 64]. In addition, similar experiments were performed at Baylor University, Waco, USA. These investigations confirmed the observed charged particle separation in the spectra of reference[27, 63]. Correspondingly, the experimental data base provides a test case for ambipolar movement/acceleration of (Argon) ions.

The outcome of the numerical analyses of variation of boundary layer distance subject to a variation of the flux is reported in reference[65]. For this task the numerical code SAMSA was used signifying that with the successful comparison of the boundary layer thicknesses in experiment and modeling SAMSA can be considered as very well verified. This is flanked by the successful simulations of the IRS steady state MPG Thrusters in reference[60]. A comparison of the experimental data base for the above mentioned Argon condition using the IRS steady state self-field plasma generator RD5 is shown in reference[66]. For the sake of briefness this is not repeated in this paper.

B. Steady state applied-field MPD Thrusters

Steady-state applied-field magnetoplasmadynamic thrusters feature a combination of high exhaust velocities, high thrust density, and power scalability, making them relevant for interplanetary missions, NEO deflection with a power level between several kW up to some MW [67, 68, 69]. However, achievable thrust efficiencies of steady-state AF-MPD thrusters are typically between 20 and 45 %.[67, 68] Basically, two different steady-state AF-MPDT design trends tend to stand out through their operational regimes, high current driven devices such as LiLFA [68], and high magnetic flux density thrusters such as DFVLR's X16 [67]. In contrast to high current driven lithium fed applied-field MPD thrusters the institute for space systems (IRS) approaches mainly a high magnetic flux density and moderate discharge current gas-fed concept [67], such as X16 from DLR with 38 % thrust efficiency at 11.6 kW arc power. This motivates the current research at IRS, which concentrates on integral characterization of laboratory thruster model and increasing of the thrust efficiency via optimal operative conditions.

Therefore, a new 100 kW gas-fed steady-state AF-MPD SX3 thruster was developed at IRS in the frame of ESA and EU projects. The water-cooled laboratory model SX3 is a 100 kW class thruster featuring separate electrode propellant gas injection. In 2014-2015 the MPD testing facility was refurbished and drastically improved towards higher safety requirements and also allowing a steady-state operation up to 100 kW arc power at relatively high current levels up to 1500 and 2500 A and applied magnetic fields up to 400 mT. In 2016 the MPD setup has been put back in operation and several calibration measurements have been performed, including characterization of applied-field coil, tare forces in short circuit operation. The SX3 thruster has been operated in steady state mode up to 115 kW arc power at relatively high applied magnetic field of 400mT (see Figure 3).

The highest thrust of 3.39 N and evaluated specific impulse of ~ 1916 s has been achieved with mass flow rate of 180 mg/s at ca. 115 kW with thrust efficiency of ca. 27.8 % (90+90 mg/s, 0.635 kJ/mg). Most efficient operation of SX3 thruster has been achieved close to onset regime at 65 kW, 400 mT, 428 A, and argon mass flow rate of 12+48 mg/s with the thrust of 2.16 N, specific impulse of 3700 s and thrust efficiency of 59 %. Initial results with a total performance envelope are very promising (see Figure 4) [37]. However, the influence of ambient pressure via re-entrainment effect is likely present and needs to be investigated at top conditions in near future and further review of experimental data and potential verification via numerical tools. In case of material wear the single channel hollow cathode shows normal signs of erosion after ~ 13 hours of operation in 2016 and proves the robustness for steady state regime at high magnetic fields [37].

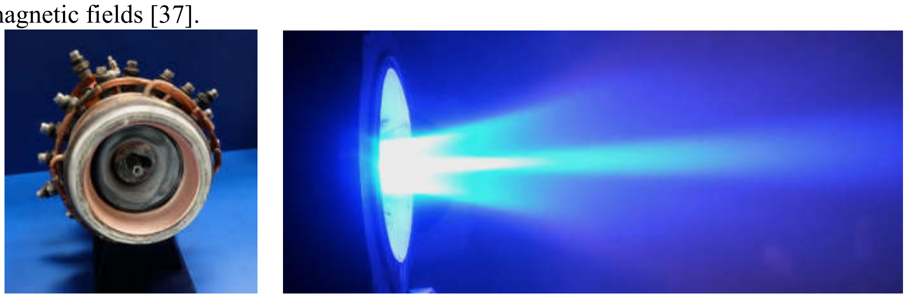

Fig. 3. Steady-state AF-MPD thruster SX3 thruster (left) in operation at 100 kW arc power, 690 A arc current, 400 mT of applied magnetic field and with argon mass flow rate of 120 mg/s.

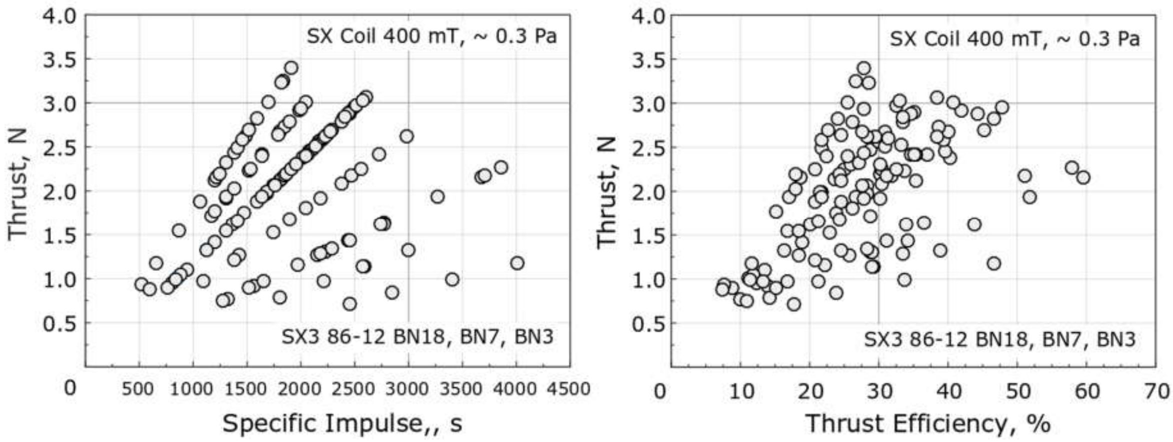

Fig. 4. Performance envelope of SX3 thruster at 400 mT.

The faint lines in Figure 4 at 3 N and 3000 s depict the required conditions for a flexible manned Mars mission as derived in reference [69].

IV.Arcjet Developments at IRS

Thermal arcjet thrusters have been investigated at IRS in a wide power range since decades[11, 15]. The current application scenarios are derived from secondary propulsion aspects (e.g. such as NSSK). However, recently additional application scenarios have been identified in the scope of ESA's Clean Space initiative. These applications are mostly linked to EoL mission phases of satellites in order to guarantee that the Code of Conduct is followed[28]. In addition, small launcher developments but also requirements of GTO→GEO transfer options for heavy satellite classes likely motivate the use of arcjets. The former one is related to the aspect of an advantageous scenario given that an arcjet, that is already planned as deorbit system for a later EoL, provides a significant orbit raising in order to enable the use of a launcher that is reduced in size[28]. The latter refers to heavy satellites to be put to a GEO within a limited period of time motivating the high thrust arcjets. Potentially some of the planned satellite swarms may motivate arcjets in order to allow for fast maneuvers e.g. for siding tasks.

Three aspects are currently under investigation at IRS referring to arcjets:

1. Arcjet based deorbit module and orbit raising system study making use of IRS low power arcjets as reference (Clean Space Project)
2. Standardization project with ESA: Here, the IRS arcjet VELARC is in use for a standardized inter laboratory comparison between ESA and IRS making use of Langmuir probes.
3. Within the project IRAS relevant ALM technologies are assessed ad analysed. Here, sub elements and structures are planned to be produced using ALM in order to further improve arcjets (e.g. regeneratively cooled tungsten nozzle)

A. IRS Arcjets for EoL deorbit and BoL orbit raising module

Correspondingly IRS has just investigated an arcjet based deorbit module in combination with an orbit raising system in collaboration with ESA, Noordwijk, The Netherlands, and ASL, Bordeaux, France. A low-power arcjet study for satellites beginning-of-life and end-of life servicing by de-orbit was assessed. Two distinct use cases were analysed while considering the common propellants hydrazine and ammonia, as well as the green alternative ADN based propellants. For the first case allows for a stand-alone arcjet system, while the second case is hybridized with the main (chemical) propulsion system. Both cases come with their individual challenges, which are high number of ignitions, long-time operation time and the necessity to cope with tank pressure blow down. A number of various feeding system approaches are explored, a trade-off for minimal mass is made and an estimation on the technological readiness is given. The thrusters used as reference for the performed analyses have already been developed at IRS (ATOS and ARTUS) in the past leading to a significant increase in confidence for the design and also for the TRL level. Complimentary to these thrusters feeding systems have been designed. The final systems are compliant to the majority of the requirements while few requirements have still to be confirmed in the development phase. However, it is expected that compliance to all requirements can be achieved during development.

Case 1 is a system for an 800 kg satellite, which is independent of the satellite's propulsion system. The system is supposed to provide a total impulse of 200 kNs to the satellite, which can be used for orbit raising and deorbiting of the satellite. For the arcjet power levels of 750 and 1000 W have been considered, where the higher power yields mass savings of over 10 %. Various system designs with propellant hydrazine, ammonia and ADN have been analysed. Generally, all systems have a mass on the order of 55 kg (@1000 W) including propellant. Where variations between the systems are on the order of 10 %. With ammonia as propellant the mass is minimal, which may be attributed to the low complexity of the feeding system and the strong heritage of ammonia thrusters at IRS. Hydrazine, however, promises to yield similar performance, when the system is operated with a mass flow controller, to provide constant feed conditions. Further a flow control by a dual branch system has been proposed, which however, has lower performance due to the variation of feed conditions. In addition, pressure control has been proposed, which, however, results in a high system mass due to the need of a pressurant tank.

Case 2 is a system for a 1500 kg satellite, which needs to provide a total impulse of 670 s. It uses the synergy with an existing chemical propulsion system. Due to this synergy, no additional tank is required, which leads to a simpler system overall. The thruster is required to operate at 1000 W and provide an Isp of 600 s. As this was found not to be feasible, a power increase to 1500 W was proposed. For the feeding system both a dual branch approach and a single branch approach with flow control have been designed. Assuming the current thruster characteristics, a flow control may lead to mass savings on the order of 7 %. Total system mass with current thruster performance (@1000 W) is on the order of 160 kg, while thruster improvements may reduce this to 150 kg. The proposed power increase to 1500 W will reduce the total system wet mass to 130 kg. The study clearly showed the beneficial constellations provided by systems as such[28].

B. Standardization Approach within Collaboration between ESA and IRS: Characterization of Electrostatic Probes and Assessment of a Standardization Procedure for Electric Propulsion

The IRS is currently conducting a project together with ESA-EPL to standardize analysis of electric propulsion devices with Langmuir probes in different tests facilities. In a stepwise approach, IRS and ESA at first decided on the EP device to be used in the measurement campaigns. IRS has a vast experience with pulsed plasma thrusters (PPT), thermal arcjet thrusters and inertial electric confinement thrusters (IEC). The selection of the thruster represents a compromise with respect to its operability, portability (including PPU) and level of understanding of the thruster from the point of view of its operational behavior. Correspondingly the assessment led to the selection of the low power arcjet VELARC. This was also motivated from a technical point of view as the relevant propellant mass flow rates had to fulfill the facility requirements from both sides ESA-ESTEC and IRS. The same applies to the power level, although this has not been critical concerning VELARC. For the sake of simplicity Argon was chosen as propellant.

Besides the pure reproduction of integral parameters such as mass flow rate, current and voltage the challenge of the project is driven by the assessment of local parameters in the plasma jet employing a rather advanced measurement technique as it is the case for Langmuir probes. Therefore, the relevant theories were processed and a standardization approach was developed[70, 71]. In a later step a qualification campaign for both facility and Langmuir probe system was performed making use of verifying data from literature[72].

The low-power arcjet VELARC was chosen to serve as plasma source for the Langmuir probe measurements. It is operated at a power range of 100-400 W with a radiation cooled nozzle using argon as propellant. The thruster, displayed in Figure 5, is a laboratory model allowing fast changes and replacement of defective parts.[11]

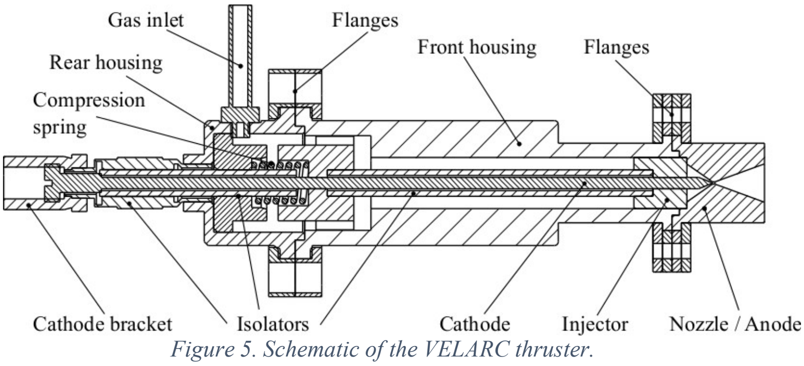

Figure 5. Schematic of the VELARC thruster.

Further steps within the project are to reach stable operation of the EP device and perform measurements of electron temperatures and densities afterwards. The procedures are first conducted at IRS and later for comparison at ESA-EPL. For the measurements 5 probes of same design were manufactured and then used in successive and repetitive experiments leading to both a statistical data base within one and the same experiment due to repeated probe sweepings and, in addition, a cross-over statistic data base due to the fact that the experiment was performed five times making use of the five probes. Correspondingly, both the visible sensitivity of ne and Te with respect to the dynamic behavior of the thruster and the additionally repeated experiments with the 5 different probes provide one of the most sustainable experimental data bases in this field.

With the successful conclusion of the campaigns at IRS and the current analyses[73, 74] later, in spring 2018, the experiments under the same conditions will be conducted at ESA-EPL.

C. Advanced Manufacturing Processes (e.g. ALM) for the further improvement of Arcjets (DLR-IRS Project IRAS)

Further research activities at IRS aim for the achievement of higher efficiencies of the VELARC thruster via new nozzle designs taking into account the availability of new manufacturing methods such as ALM. The high temperatures occurring at the electric arc attachment points require materials with high melting points. Since conductivity is essential to operate the thruster, only heavy metals and certain ceramics are appropriate. For VELARC, the cast tungsten lanthanum-oxide was chosen, as it provides the necessary temperature resistance and a reduced work function compared to pure tungsten. However, since the thruster is very small, features like regenerative cooling were complicated to be implemented or simply not possible.

With the new prospects of additive layer manufacturing (ALM), the limits in designing a thermal arcjet nozzle dropped significantly. The company Plansee, specialized on the field of refractory metals, promises to gained a manufacturing precision that allows channel structures with a diameter down to 0.5 mm. This precision opens the door to regenerative cooling designs for nozzles of the size of VELARC.

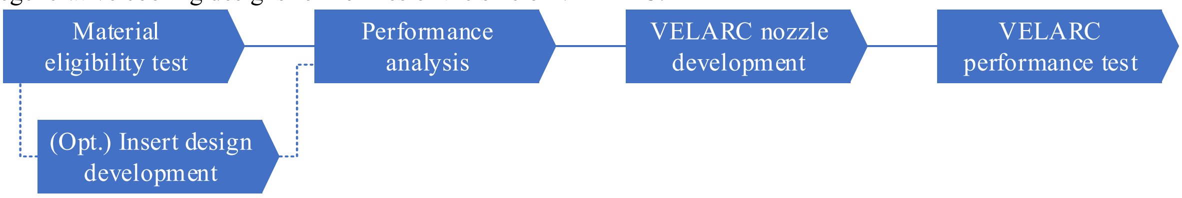

Figure 6. IRS roadmap for ALM nozzle development

Figure 7. Design of a regeneratively cooled arcjet nozzle (ALM-based).

A roadmap was established at IRS to investigate the suitability of ALM arcjet nozzles (Figure 6). Since the material density does not reach the 100% of conventional manufacturing processes, it has yet to be determined, if an electric arc can be properly sustained. Initial experiments will validate the general eligibility of laser-sintered tungsten in electric arc applications. Should the material fail, an alternative design with a conventionally produced insert as constrictor will be manufactured. Another point is the rough surface quality of ALM produced parts, which might require post processing at flow critical locations.

The following performance analysis will be done with the medium power arcjet MARC-6 developed at IRS.[11] Since MARC-6 is a much larger arcjet thruster compared to VELARC, the benefit from ALM optimized designs can be investigated without driving the manufacturing process to the edge of its capabilities. Cooling channels e.g. can be designed with an inner diameter of 2 mm and more, which poses less risk of choking due to impurities. The nozzle design resulting from an exchange with the manufacturer is displayed in Figure 7. Propellant is guided towards the thrusters exit where it enters the three helix shaped cooling channels. Inside these channels the propellant flows back towards the discharge chamber, which it enters with an initial spin. The propellant covers the nozzle with two layers, on the outside and inside the cooling channels, promising to raise the heat transfer resulting in higher thrust efficiency and weight specific impulse. Tapering the nozzle on the outside reduces weight and allows more propellant around the constrictor area where temperatures are at a maximum.

Another reason for a test campaign with MARC-6 is that power levels in the range of 1-5 kW result in a much more stable operation. Especially if performance gains due to ALM designs are in a small order, measurements with stable discharge characteristics are less subject to uncertainties. Moreover, a data base on a conventionally regeneratively cooled MARC-C already exists such that this can be directly compared with the newly designed nozzle.

Once the performance analysis is concluded, a VELARC sized nozzle will be developed based on the experience gained from the previous tests. The follow-up performance analysis shall determine the actual improvement in thrust efficiency and weight specific impulse for low-power arcjets.

V.IEC Developments at IRS

Inertial electrostatic confinement (IEC) is originally a fusion concept proposed in 1950s, which offers both plasma generation and its confinement at the same time. The basic concept of the IEC device is composed by a pair of concentric spherical grids, both of which serve as electrodes as shown in figure 8. By applying high voltage between two grids, plasma can be produced from electron impact which forms a virtual anode at center of IEC.[75] The virtual anode is a non-neutral plasma which results in a spherical double layer within cathode grid and provides better ionization and confinement for plasma.[42]

Development of IEC for space propulsion purpose was initiated at IRS in 2010. The IRS-IEC is operated in non-fusion condition with a stable and constant plasma jet extraction. Investigations of electrical properties, discharge phenomena, and loss mechanism were performed both numerically and experimentally.[10, 43] The plasma extraction is achieved by distortion of electric potential field, which creates a weak point on the confinement envelop. Depending on the operational conditions, the extraction can lead to two different jet modes: tight jet and spray-jet, as Figure 9 shows. The tight jet is confirmed as high energy electron beam (2-3 keV) while the spray jet is a diffused plume composed by ions and electrons. Details for the extraction mechanism for both plasma jets are explained in Ref. 43.

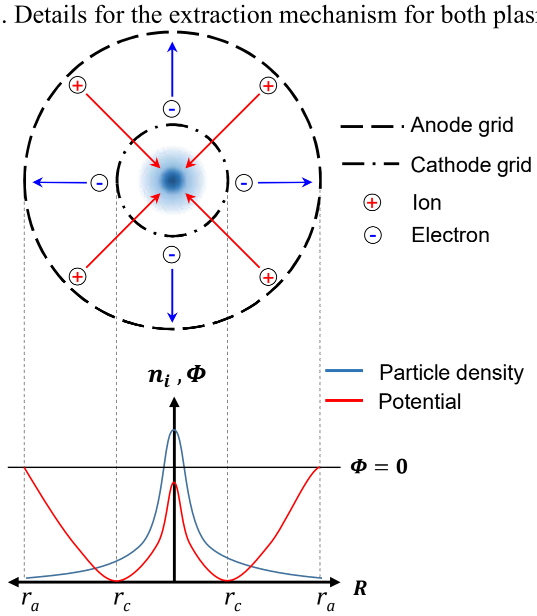

Figure 8. Schematic of IEC principle[42]

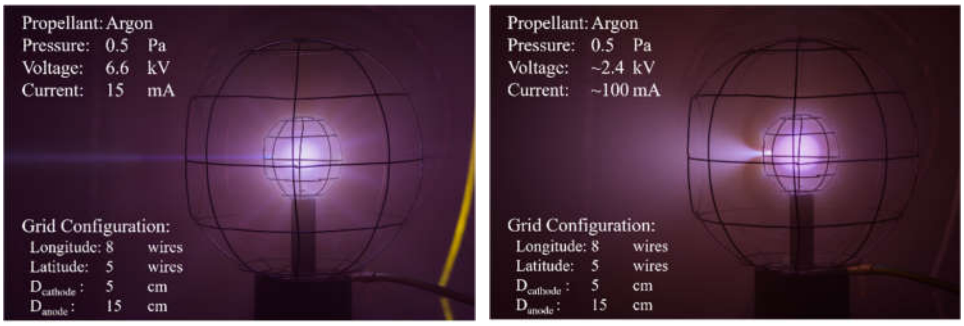

Figure 9. Tight jet (a) and spray jet (b) mode in IRS IEC.[2]

The observation of a spherical double layer (SDL) suggests that IEC has a great potential to overcome the low-particle-density and energy density issue for EP devices. In addition, the application of tight jet and spray jet mode can be further extended to other EP concepts such as IEC-Hall thruster and Atmosphere-Breathing EP.[42] Furthermore, the IEC propulsion concept is adaptable for fusion propulsion in futures space mission.[42] The flexibility enables IEC in all-spectrum space exploration. Moreover, the promising applications such as electron/plasma source, radiation source, and neutron source illustrates the potential of IEC in scientific and engineering development.[42] This core technology will bring a significant innovation not only on Earth but also in space.

VI.Atmosphere-Breathing Electric Propulsion at IRS

DISCOVERER is a €5.7M, 4 1⁄4 years Horizon 2020 funded project which aims to radically redesign Earth observation satellites for sustained operation at significantly lower altitudes. Operating satellites at lower altitudes allows them to be smaller, less massive, and less expensive whilst achieving the same or even better resolution and data products than current platforms. However, at reduced orbital altitude the residual atmosphere produces drag which decreases the orbital lifetime. An Atmosphere-Breathing Electric Propulsion (ABEP) system is a concept (see Fig. 10) in which the residual atmosphere encountered by a satellite orbiting at low altitudes is collected by an intake and used as propellant for an electric thruster, theoretically eliminating the requirement of carrying on-board propellant.

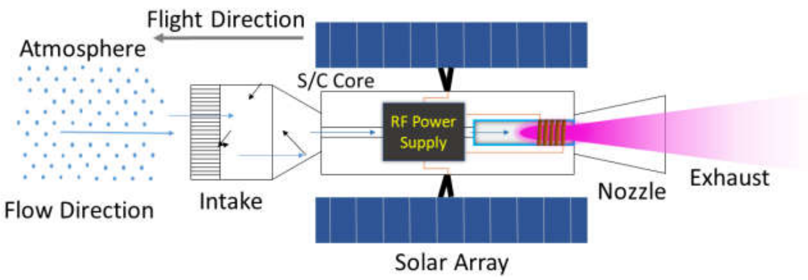

Fig. 10 ABEP Concept

Some concepts have already been investigated[23] but most of these have not developed beyond paper studies. These studies imply the use of an electric thruster, due to its scalability to small satellites and its high specific impulse, however, at present the focus has been on gridded ion thrusters (GIT) or Hall effect thrusters (HET) which suffer from performance degradation over time due to erosion of the accelerating grids or the discharge channels especially if chemically aggressive propellants are used.
Previous work by IRS[76] was based on intake concepts from JAXA[77] and BUSEK[78] and considered an optimised design to be applied to an inductive plasma thruster (IPT) of the size of IPG6-S[39]. Although this thruster concept departs from currently assessed concepts such as HET and GIT, the inductive thruster is strongly justified by several benefits: a) its propellant flexibility (e.g. as needed for dynamic and/or altitude dependent composition changes within the thermosphere; b) it is electrodeless and gridless, eliminating the

lifetime limiting erosion issue; c) it does not need a neutraliser as the plasma leaving the thruster is already neutralised; and d) the electrodeless function of the thruster is already qualified by the extensive application of comparable devices, e.g. for pure high enthalpy material tests using pure oxygen as the working gas[2]. Within DISCOVERER, this IPT-based ABEP concept is explored and developed to establish the feasibility of such a propulsion system with realistic power and mass constraints. First, a consolidated system analysis for an ABEP system, the available and verified models for the intake design, and the innovative approach of using an inductively heated plasma thruster will be carried out. Ultimately, an IPT will be designed, built, and tested, based on the experience and heritage at IRS on inductively-heated plasma generators (IPG) used for re-entry simulation and propulsion. Such a device is composed of a discharge channel, where the gas propellant flows, surrounded by an RF-fed coil which ionises the gas. Preliminary studies have successfully operated a small inductively-heated plasma generator (IPG6-S) with atmospheric propellant (air, N_2, O_2, CO_2) at mass flows derived from an ABEP system analysis[79], see Fig. 11. Further work has to be done for the development of a laboratory model of an IPT based on IPG6-S, with a discharge channel diameter < 40 mm, 0.5-5 kW input RF power, with a complete design of the accelerating stage. This prototype device will be tested in a representative orbital environment to evaluate minimum conditions for ignition and measure the respective produced thrust.

Fig. 11 IPG6-S running on N_2 at 3.5 kW

The intake must efficiently collect the residual atmosphere and feed it to the electric thruster. However, the design requirements are significantly different to conventional intakes (e.g. jet engines) because ABEP systems operate at altitudes where the flow can be assumed as free molecular, which means it behaves as singular particles traveling through free space. Along their trajectory towards the surfaces of the intake, they do not interact with each other and during wall reflections one common assumption is full accommodation by which they are scattered into a random direction. The main possibility for a reflected particle to reach the thruster is given by this scattering into a solid angle Ω which is much smaller than the whole range of the possible half space. Thus, a simple open intake would be inefficient because the most of the particles entering the thruster would be only those inside the extruded cross section of the thruster. This results in the necessity to prevent the reflected particles from exiting the intake, but without blocking the incoming particles. This idea of a "molecular trap" can be realised in a passive design based on the transmittance characteristic for a hyperthermal flow through an effectively long geometry, e.g. one/several tubes or thin ring, with diffuse scattering afterwards. Recent studies at IRS provide numerical verification of previous intake studies and a verified tool for analytical intake evaluation[76]. For DISCOVERER, possible intakes are analyzed using DSMC simulation for the design, and scaled models are built and tested in an atomic oxygen wind tunnel for verification. The first steps will deal with the influence of different kinds of gas-surface-interaction on the intake performance.

VII. Water Propulsion System

The development of a water propulsion system has been engaged some 2 years ago in collaboration with Airbus. This propulsion system is based on water electrolysis. The secondary electric propulsion system uses low pressure water as main propellant, which is decomposed via electrolysis on orbit to generate gaseous hydrogen (H_2) and oxygen (O_2) to propel a chemical thruster.

Compared to pure chemical thruster systems the water propulsion system combines high performance with non-toxicity and therefore easy ground handling. Usually chemical propellants store their energy chemically and are mostly toxic (e.g. hydrazine and its derivatives) and therefore have high risk while handling. With a water electrolysis based propulsion system working on already fueled spacecraft is possible, because energy addition to the propellants happens electrically not until on orbit. The system is totally inert. The propulsion system consists of a low pressure water storage, an electrolyzer as gas generator and one or several chemical thrusters.

Promising experiments with an electrolyzer lab model have been conducted where relative gas humidity as low as 40% has been reached. An optimized flight like prototype is currently being developed. In its basic configuration the modular electrolyzer will have a power of 20 W and will store the generated gases under a pressure of 50 bar with a total system volume of a 1U CubeSat. Due to its modular design additional power can be provided, if required by the mission.

A chemical thruster is being developed at IRS. Main objective of the thruster is to be propelled by the stoichiometric mixture ratio of H2 and O2 delivered by the electrolyzer. Due to the high combustion temperatures of such a mixture ratio with up to 3400 K, certain combustion chamber and nozzle cooling effort is required.

The thruster is designed to have a thrust of one Newton with a specific impulse (Isp) of over 350 s. Ignition is carried out catalytically. Compared to classic electric propulsion systems the thrust to power ratio is high. First ignition tests have been conducted with the injector head. A mixture ratio of 0,5 was set with orifices. This first test proves ignitability by catalyst even with very rich mixture ratios, which are needed to protect the catalyst from high combustion temperatures. For more details, refer to reference [80].

VIII.High Precision Attitude Control System Based on the Emission of Electromagnetic Radiation

From basic physics it is known that the active emission of 1 W of electromagnetic power is associated with a generated force of 3.3 nN. This physical principle has been frequently suggested as a main propulsion system for space missions, despite the huge power demand for significant forces. On the other hand, it has not been thoroughly exploited for attitude and orbit control purposes with ultrahigh precision demands, where very small forces are certainly of interest. General concept and some ideas for the underlying technology and the integrated system components for this idea, named high-Precision Attitude Control system based on the emission of Electromagnetic Radiation (PACER), has been elaborated and described in[81]. PACER is a scalable, fuel-less thrust vectoring system, facilitating the conduction of protracted space mission profiles and providing an exhaust-free environment. By exploiting the thrust and torque generating effects due to the emission of electromagnetic radiation, an unprecedented accuracy in attitude control could be achieved. Due to its inherent scalability, the system could facilitate an adjustable thrust level undercutting the 10 nN order of magnitude, if required. In addition, ideas about a high precision thrust stand, which shall enable performance characterization and thrust measurements of the unit are under investigation. Here, the concept of resonant operation during measurements is currently investigated as one option to achieve the required level of precision. Essential system sub-components required for the electrical power system including power generation, distribution, allocation and storage are outlined in reference [81].

IX. Thermal-Inductive Heated Thruster of the University of Stuttgart (TIHTUS)

The Thermal-Inductive Heated Thruster of the University of Stuttgart (TIHTUS), combines the complimentary operations of arcjet and inductive plasma sources in order to generate a flexible, high-performance propulsion test platform. Since its initial development, the system has been modified with a number of new measurement tools, allowing passive data collection within the thruster whilst plume diagnostics are conducted[34]. Figure 12 depicts a schematic of the thruster and the system in operation using hydrogen as propellant. The flexibility of TIHTUS can be assessed by several aspects. On the one hand TIHTUS' second stage has a maximum flexibility with respect to the used propellant. Here, both the inductively heated high power plasma stage IPG3 and later IPG7, both qualified at maximum anode powers of 180 kW, have already been qualified for various propellants within their use as plasma sources for aerothermodynamic testing[1, 2, 3, 6, 8, 19, 22, 39].

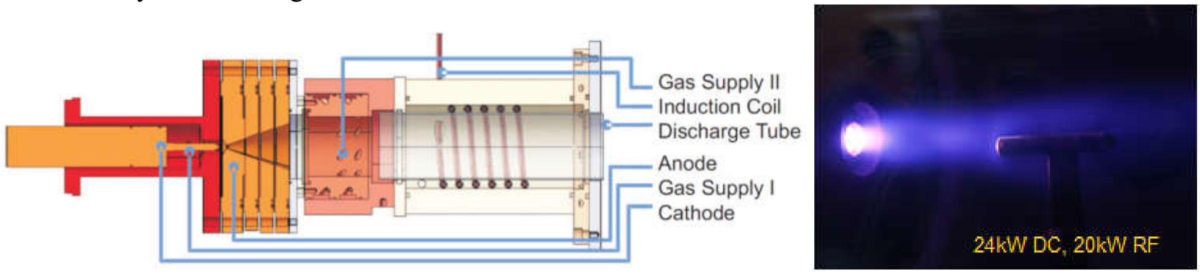

Fig. 12 TIHTUS schematic set-up and TIHTUS in operation (using IPG3 as second stage, hydrogen as propellant)

Due to a needed advancement in the understanding of the functional principles of the 2nd stage extensive investigations varying the propellant configurations and blends have been performed recently. These investigations have led to a further improvement of the understanding of power coupling and structural behavior[16, 82]. Moreover, this aspect is valuable from the systematic point of view as both IPG3 and IPG7 have already been qualified for high mass specific enthalpy CO_2 flow conditions. This raises the potential use of the second stage e.g. for manned missions to Mars, where either ISRU-relevant working gases or waste gases from the LSS can be used as propellant. In the final stage a geometric optimisation of the thruster stages shall be achieved (ongoing work). Additionally, an improved design of the inductive stage is intended to increase the total obtainable data from the system.

X. Conclusion

A synopsis of the electric propulsion systems currently under investigation at IRS is given. For the MHD-based systems such as the high bank energy PPT ADD SIMP-LEX and the steady state AF MPD Thruster SX3 breakthrough results have been achieved e.g. with respect to the overall thruster relevant parameters (thrust, I_{SP} and η_{thrust}). For the previous this may lead to the fact that potential applications of PPTs e.g. for larger CubeSat missions have to be assessed and discussed from the new. This is also flanked by their simplicity and robustness. Moreover, PPT do not require pressurized sub systems and, in addition, they are throttable in a wide range. The latter i.e. the steady state AF MPD thrusters likely find their application in both heavy cargo missions and astronautical missions e.g. to Mars.

Self-field MPD thrusters have been developed and characterized at IRS for almost 2 decades. It is evident that level of understanding concerning these systems is rather high. Nowadays IRS applies self-field MPD mostly for the production of high enthalpy plasma flows. Anyway, for the regime of MHD-based boundary layer influencing the SF MPD RD5 has been used and the corresponding numerical rebuild of both the boundary layer influencing and the operation of RD5 itself have led to a significant increase of the verification and validation level of the numerical tool SAMSA which in turn is a significant and important preparation for the planned simulations of the IRS AF MPD thrusters.

Arcjet development and characterization is currently oriented in three directions: The first direction assesses and confirms new applications of arcjets that are either derived from EoL scenarios for satellites taking into account the code of conduct or from needed fast maneuvers as they are e.g. needed for the orbit transfer of heavy satellites, orbit raising in general or fast maneuvers. The second direction makes use of arcjets as reference thrusters within standardization projects aiming at systematic inter laboratory comparisons. With the third direction it is attempted to further improve arcjet designs by employing new and advanced manufacturing processes such as ALM.

The incomplete list is contributed by even more advanced systems such as IEC based propulsion, offering a significant potential to further extend current I_{SP} levels and the application regime in general, atmosphere breathing propulsion as currently under investigation within the European project DISCOVERER, water-based propulsion bridging combustion and EP via the generation of the propellants using electrolysis, electromagnetic emission based propulsion and TIHTUS. The majority of them is covered by current contributions on the IEPC2017 where they are introduced more in detail.

Acknowledgements

The authors wish to thank ESA for the currently ongoing cooperations and contracts within AF MPD thruster development, standardization and IEC thruster systems. In addition, thanks go to European Union and all our partners involved in the activities.

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