Overview on Electric Propulsion Developments at IRS

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More than three decades of experience have been gained in the field of electric propulsion at the Institute of Space Systems (Institut für Raumfahrtsysteme = IRS). Recent developments within the field of electric propulsion are summarized and foremost results are highlighted. The various types of electric propulsion systems are not considered as to be competitive. Here, system analysis shows that optimum parameter such as the required exhaust velocity or specific impulse result taking into account both the mission profile and system related sizes such as the power conditioner efficiency, the thrust efficiency and the specific mass of the corresponding power unit. Correspondingly, ion thrusters, Hall thrusters, thermal arcjets, or magnetoplasmadynamics (MPD) thrusters are preferable depending on the mission. In addition, several advanced plasma propulsion designs have been developed and characterized at IRS in the past 10 years. Among them are the hybrid thruster TIHTUS, steady state applied field thrusters and the iMPD IMAX. These concepts have been experimentally and numerically characterized and show promising potential for future missions. The paper will discuss the design and the operational features of the devices. In addition, more advanced systems are under investigation. Here, a focus is in the field of fusion driven systems and M2P2 (magnetic sail systems).

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I. Introduction

There is no doubt that electric propulsion systems can seriously compete with "standard" cold gas or chemical propulsion systems used for primary propulsion purposes.
It astonishes that for many years the point of view was that electric propulsions systems compete with each other. As a result of system analysis this opinion has now changed in the majority of the community. A first insight can be provided by the consideration of simplified propulsion system related parameters. A consideration of these parameters in combination with Tsiolkowsky's rocket equation even allows for optimizations of the concerned electric propulsion systems with respect to the specified mission profile. The degree to which these analyses were performed practically depends on how detailed the aforementioned parameters are considered. The classic approach and the references herein contemplates the specific power of the propulsion system

$$\alpha_F = \eta_{PC} \eta_T \alpha_{PS} \tag{1}$$

which depends on the efficiency of the power conditioner η_{PC}, the thrust efficiency η_T and the specific power of the power system α_{PS} which is the ratio of the electrical power and the sum of power supply mass, power conditioner mass, and thruster mass. The efficiency η_{PC} and the specific power of the power system α_{PS} were considered to be independent from the specific impulse. However, it has to be emphasized that under certain circumstances both η_{PC} and α_{PS} may depend on the specific impulse (e.g. α_{PS} if the thruster mass is in the same order than the power supply mass or η_{PC} if it depends on the electric power. The specific impulse, however, often depends on the power itself as it is e.g. the case for thermal arcjets. This, however, is of importance as most of the known electric thruster systems have a variance of the specific impulse with respect to their operational parameters. The analysis of this is not the task of this document but the thoughts support the statement that, depending on the mission requirements, different electric thruster types such as thermal arcjet, magnetoplasmadynamic (MPD), ion or Hall ion thruster systems have their applicability.

In this work, for the sake of briefness and clarity, only most recent works i.e. e.g. 2011 IEPC papers and the two references [1, 2] are used as reference. Clearly, we have to state that the latter references are extensive peer-reviewed reviews on electric propulsion[1] and advanced propulsion at IRS[2]. Correspondingly, works and results of others that may be used in this work can be traced back as references in the papers [1] and [2] and also [3].

At IRS for the lunar mission BW two electric propulsion systems are developed: The Thermal Arcjet for Lunar Orbiting Satellite (TALOS) a 1 kW arcjet system dedicated for the fast branches of the trajectory such as the crossing of the Van-Allen belt and the Added Stuttgart Impulsing MagnetoPlasmadynamic thruster for Lunar Exploration (ADD SIMP-LEX) a pulsed MPD thruster system (often also named Pulsed Plasma Thruster "PPT") that currently goes through a development and research activity at IRS with the aim to obtain a flight capable thruster system. This project is in cooperation with ASP GmbH and is supported by DLR[4].

The pulsed plasma thruster ADD SIMP-LEX primarily utilizes electromagnetic forces to produce its impulse. Its solid propellant is Polytetrafluorethylene. A parallel-plate geometry was chosen. The design is modular to allow for changes of geometry and components. In the first phase an electronic bread board was provided by ASP GmbH enabling variation of pulse rate and bank energy. To investigate this thruster thoroughly, test facilities and related measurement systems were set-up. Investigations include measurement of mass bit, plasma current, also yielding plasma's acceleration time, thrust and dynamic properties as the behavior and movement of discharges along the electrodes. In the mean the development of the ADD SIMP-LEX thruster system is very advanced with respect to the maximization of specific impulse and thrust efficiency. In a next step the thruster system will be equipped with a proto-flight power unit which is then respectively tested and qualified together with the optimized thruster.

To strengthen the competences in the field of pulsed plasma thrusters an International PPT & iMPD Working Group under the lead of IRS was founded in 2007. Here, the first international workshop was held in Stuttgart as a first attempt to merge the knowledge and experience of specialists from Russia, Japan, Austria, Great Britain and Germany (http://impd.irs.uni-stuttgart.de). The second workshop held in 2011 aimed for a review and categorization of the iMPDs allowing for a simplified identification of spacecraft relevant application of the thruster systems[5].

TALOS has approximately 1 kW of power. For the system this includes design, construction and qualification of an appropriate propellant feed system and an optimization of the ammonia propelled arcjet. The optimization of the thruster includes ground tests and numerical simulations as well as thermal modeling. Beyond this a low power arcjet called VELARC is currently under consideration[6]. This may be an improvement for the design of relevant missions such as BW1 as a reduction in operational power may lead to the elimination of the rather heavy batteries and size and mass of the solar panels that are currently needed aboard of BW1 in order to buffer the 1 kW operation of TALOS.

A further development is related to TIHTUS. The concept consists of a thermal arcjet thruster and an inductively coupled plasma (ICP) stage. While the arcjet thruster generates plasmas with steep radial gradients in the plasma's radial variables, the ICP stage is used to heat the relatively cold gas layer at the plume's edge. The arcjet of use is HIPARC-W (High-Power Arc Jet - Water-Cooled). HIPARC-W, a 100 kW thruster, has a segmented anode, such that nozzle length can be varied. The ICP stage in TIHTUS is represented through the IRS' IPG3 (Inductively Heated Plasma-Generator). It is a continuous inductively coupled plasma generator with operational frequencies between 0.5 and 1.5 MHz at maximum plate power of 180 kW. Each stage's plasma flow is experimentally investigated and currently calculated numerically.

Investigations have been carried out on self-field MPD thrusters between 100 kW and 1 MW. 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 the application of numerical codes, allowing the theoretical calculation of the MPD thrusters and a comparison with experimental data.

Applied field magnetoplasmadynamic (AF-MPD) thrusters are promising devices for orbit control of large satellites and they are suited for interplanetary missions. Their qualification is challenging due to the vacuum quality needed for ground testing (< 0.1 Pa). Also finding an optimized configuration for geometry and applied magnetic field is difficult, because of complex correlations between different acceleration mechanisms. Supported by German Research Foundation DFG a numerical simulation tool is under development and qualification. In parallel a thruster was designed and is now being built up. Its design is based on the X13 thruster developed by DLR in the 1970s. Experimental investigation is foreseen to gain experience by comparing numerical and experimental results.

The implementation of electric propulsion systems virtually requires an understanding of the systems from the scientific point of view. Correspondingly the thruster developments at IRS are accompanied by both experimental investigation using advanced measurement techniques and numerical analysis.

The engineering objective in advancing existing mass ejection propulsion systems can be identified performing analyses and optimization. Relevant considerations show, that there’s an overall benefit in raising the specific impulse, but also that this has to be tuned with the characteristic acceleration of the system, its characteristic masses and last but not least both its efficiency and its mass specific power. The latter is also the driver in the conception of new and even more advanced mass ejection space propulsion.

However, there are also other impulsive space propulsion concepts that do not eject mass and have to be studied differently, such as solar and magnetospheric sail. At IRS the investigation of magnetospheric sails has just been initiated. Here, particular emphasis has to be put on PICLAS a code in which the IRS DSMC code LasVegas, the PIC code of the KIT in Karlsruhe and a Fokker-Planck solver are hybridized with the aim to cover technical systems in higher Knudsen regimes and that are accompanied by the presence of internal and external electromagnetic fields.

II. Thruster Systems ADD SIMP-LEX and TALOS for the IRS Moon Mission BW1[1 and refs herein]

The moon mission BW1 is one of the four small satellite missions within the small satellite program of IRS, Universität Stuttgart. For this all electrical satellite mission two different electric propulsion systems are used, which are under development at IRS at the moment. One propulsion system consists of a cluster of instationary pulsed plasma thrusters, ADD SIMP-LEX, and the other propulsion system is a thermal arcjet thruster system. It consists of the thruster – TALOS –, the propellant feed system, and the power supply and control unit for the thruster and the propellant feed system. The development and qualification of the two thruster systems is completely accomplished at IRS in cooperation with industrial partners.

A. Pulsed MPD Thruster ADD SIMP-LEX[1 and refs herein]

The pulsed MPD thruster ADD SIMP-LEX is being designed as part of the institute's endeavor to place the 200 kg satellite BW1 in a lunar orbit. It is developed to form as a cluster the main propulsion system together with the thermal arcjet system TALOS.

ADD SIMP-LEX is a pulsed MPD thruster using the solid propellant PTFE (Polytetrafluorethylene). The main parts are shown in Fig. 1.

Its capacitor bank stores up to 68 J in 80 µF of total capacitance. Two electrodes are linked to the capacitor carrying the applied electric potential. The propellant necessary is fed in between the electrodes. In order to initiate the pulses, an igniter is installed. This semiconductor spark plug receives 1000 V and triggers the capacitor discharge. This discharge leads to the ablation and acceleration of propellant and, hence, to the thrust. A side view of the thruster's electrodes during discharge is presented in Fig. 2.

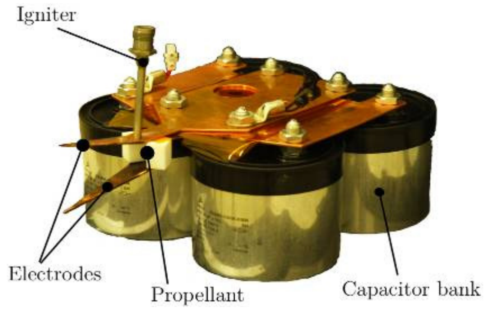

Figure 1. Magnetization as a function of applied field.

The satellite BW1 poses certain demands on its main propulsion system. Apart from being a low weight, robust, easy-to-integrate system, the pulsed MPD cluster will also have to provide BW1 with a □v of about 5 km/s, and, hence, will need a lifetime expectation not precedented for pulsed MPDs so far. This requires investigations of the durability of the components, which are already ongoing. In addition, propellant storage for this mission poses a challenge. Here, it is necessary to feed the propellant from the side using a helix shape.

The measurement systems available at IRS for characterizing and optimizing the thruster allow for current and capacitor voltage measurements, thermal measurements of the electrodes, high speed camera measurements, mass bit measurements, time of flight probe measurements, magnetic probe measurements and impulse bit measurements. Especially the setup and calibration of the thrust balance to measure the impulse bit are very important with respect to the thruster's application on BW1. Electrode shape and geometry were varied in order to find an optimum with respect to the lunar mission. Refinements of these investigations are necessary to assure best performance of the thruster. Measurements of the magnetic field and the plasma velocities during the discharge were done to better understand the acceleration processes in order to estimate the total impulse of the thruster.

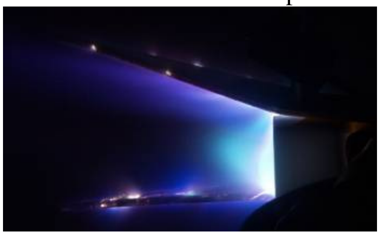

Figure 2. ADD SIMP-LEX in operation.

In addition to the experimental investigation of these values and evaluating their effects with respect to an optimal design of ADD SIMP-LEX, 1D analytical models are used and refined to predict the thruster's behavior. Especially, the discharge circuit and its parameters allow comparing the model and measurements well. For modeling of the self-induced magnetic field between the thruster's electrodes, the Biot-Savart law was applied to the different electrode shapes. Further, a particle-in-cell (PIC) code is under development to simulate the plasma respecting the non-equilibrium conditions of the chemical and thermal processes solving the Boltzmann equation for low particle densities.

Current investigations of the thruster include life time testing of capacitor unit and igniter, as well as ablation behavior of the propellant bar. In a next step, the system domain including the power supply and the on-board diagnostic will be developed and the plasma composition will be investigated using optical methods. After an on-orbit verification aboard the test satellite PERSEUS, a cluster of these pulsed MPDs will be integrated into BW1.

ADD SIMP-LEX is the result of rigorous investigation efforts at IRS to optimize the thruster's overall efficiency in order to support the mission BW. Two important contributions to the overall efficiency depend on the capacitance C and the initial inductance L_0 of the thruster: the thrust efficiency η_T and the electrical efficiency η_e. To find an optimal η_T the thruster's capacitance was varied in steps keeping the initial energy constant while the η_e was increased by significantly reducing L_0. For the latter, a change in construction of the thruster was necessary to minimize the electric circuit's inherent area. Both these modifications lead to a significant improvement in overall efficiency, hence allowing a better mission performance and flexibility as well as higher payloads.

Strong contributions to these developments are made by the Komurasaki Lab. of the University of Tokyo where significant investigations in the field of optical diagnostics with respect of the discharge behavior and resulting thruster relevant parameters such as flow velocity were performed[7].

B. Thermal Arcjet Thruster TALOS[1 and refs herein]

The thermal arcjet thruster TALOS (Thermal Arcjet Thruster for Lunar Orbiting Satellite) is under development at IRS at present. Together with the MPD thruster described above it is used as the main propulsion system.

The propellant used is gaseous ammonia. A sketch of the thruster is shown in Fig. 3.

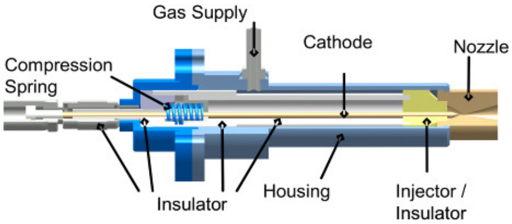

Figure 3. Sectional Drawing of TALOS.

After first optimization investigations had been conducted with respect to the requirements of the lunar mission BW1 (thrust ~ 100 mN, maximum electrical input power for thruster system 1 kW, mass flow between 20 and 30 mg/s and effective exhaust velocity ~ 5000 m/s) and after one possible operation point had been defined in earlier work, investigations of the thruster's performance for operating cycles as foreseen at the lunar mission Bw1 – 1 hour on / 1 hour off – have been conducted. Fig. 4 shows the thruster during operation inside the vacuum chamber.

During this multi-hour test campaign optical investigation and measurement of the nozzle throat diameter is done after 1 hour, 5 hours, 10 hours, 20 hours and 30 hours of operation. By the optical investigation performed during the multi-hour test possible changes in the constrictor area were monitored. This activity was motivated by the results of other researchers where the constrictor closure phenomenon appeared as well and was identified as one of the lifetime limiting factors.

To get a realistic impact of cathode erosion effects on thruster performance in comparison to the operation on board the satellite, the cathode gap had been adjusted to 0.8 mm prior to the experiments.

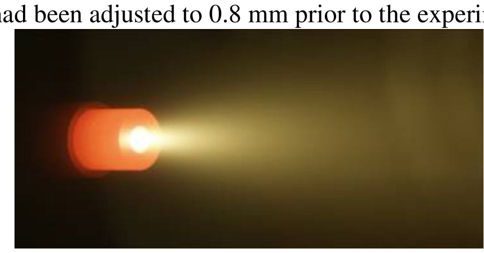

Figure 4. TALOS during operation.

Fig. 5 shows the laboratory model of the propellant feed system. It supplies gaseous ammonia as propellant for the thermal arcjet thruster. The conveyance is accomplished by impressing a pressure difference between the tank, where the ammonia is stored in liquid phase, and the thruster exit plane.

The ammonia is vaporized and heated inside the gas generator by means of electric heating. Variable input pressures, which could be caused by differing phase composition of the ammonia inside the gas generator, are compensated by the pressure reducer. The outlet pressure of the pressure reducer is fixed. Doing so, the pressure in front of the flow aperture is kept constant. This is necessary because the mass flow is regulated by a defined pressure-ratio over the flow aperture. By changing the outlet pressure of the pressure regulator the mass flow can be adjusted. Two T-type filters made of stainless steel with a pore size of 15 μm and 7 μm filter out impurities inside the propellant such as particles. First experiments conducted with the laboratory model of the propellant feed system give the following results:

- the mass flow can be varied between 10 mg/s and 29 mg/s,
- the mass flow is a linear function of the pressure behind the pressure regulator,
- mass flow and pressure are constant behind the flow aperture for an inlet pressure lower than 0.5 MPa (maximum inlet pressure of pressure regulator) and
- a significant temperature drop behind the gas generator is depicted.

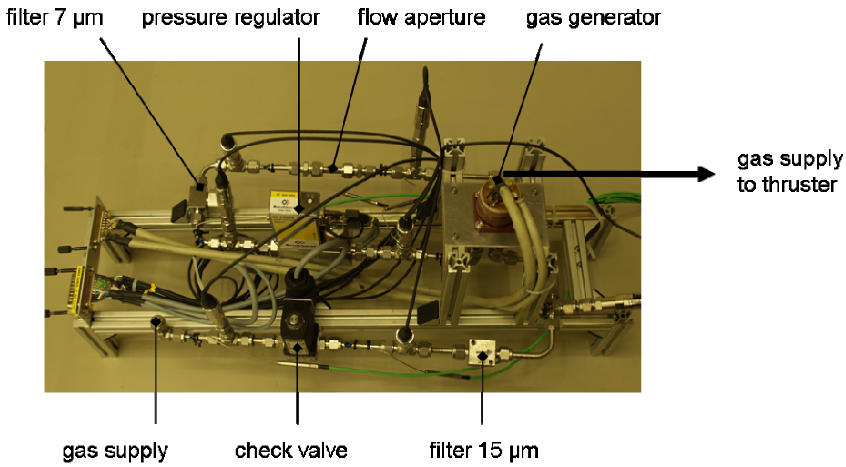

Figure 5. Laboratory model of propellant feed system.

All in all the development activities of IRS in the field of thermal arcjets are significantly more extended than depicted here. A respective review on IRS arcjet thruster developments is given by reference[8].

III. Steady State Magnetoplasmadynamic Thrusters

Steady state magnetoplasmadynamic thrusters have been under investigation at IRS for decades. The overall outcome for the self-field magnetoplasmadynamic thrusters (SF-MPD Thruster) is that they are promising candidates for missions that require both a somewhat high specific impulse in combination with an increased thrust density. One example to mention are manned missions to Mars that include e.g. emergency scenarios i.e. scenarios that include a flexibility. However, SF-MPD Thrusters suffer from the fact that they cannot be operated adequately at power levels below 100 kW as the needed self-induced magnetic fields are not high enough for a production of an adequate magnetohydrodynamic thrust. This in turn motivates the development of steady state applied field thrusters (AF-MPD Thruster) as they can be operated at power levels that are also one to two orders of magnitude lower than 100 kW.

C. Steady State SF-MPD Thrusters[1 and refs herein]

A broad spectrum of self-field magnetoplasmadynamic (MPD) thrusters in the high power range from 100 kW up to 1 MW have been examined in the last decades, all operated in steady state mode.

The ZT-thrusters were investigated at power levels up to 350 kW and currents up to 15 kA. In operation, the limiting factor was not the thruster itself but the insufficient cooling of the vacuum chamber. For a comparison with numerical computations, extensive plasma diagnostics with emission spectroscopy and electro-static probes as well as investigation of the temperature distribution on the cathode surface have been performed. With argon as propellant thrust values up to 25 N with a thrust efficiency of about 10 % were obtained and no instabilities were detected. The nozzle-type MPD thrusters have a coaxial geometry with the cathode upstream on the centre axis and the anode downstream forming the nozzle. Nozzle throat and first part of the nozzle are formed by neutral segments which reduce the potential difference between cathode and anode stepwise. With this geometry (DT and HAT/CAT series), which combines the thermal expansion of pure arc jets with the magnetic acceleration of pure MPD-thrusters, thrust values of 27 N at electrical power levels up to 550 kW, current levels up to 8000 A and thrust efficiencies up to 27 % were obtained. However, the maximal current levels were limited by incipient voltage oscillations which were followed by onset phenomena of the arc at the anode.

The experimental work has been accompanied by the development of numerical codes, allowing for a theoretical calculation of the MPD thrusters and a comparison with experimental data. Within this work, optical diagnostics mainly by means of emission spectroscopic measurements were carried out. Theoretical and numerical investigations on the onset of instabilities were carried out showing that these phenomena are most probably caused by charged particle depletion in the anode boundary layer. Figures 6-8 show the different thruster geometries with the thrusters in operation with argon.

The HAT thruster was designed for an investigation of the influence of the hot anode surface on the instabilities. During the design process of the radiation cooled anode manufacturing needs and the necessity of a large surface for radiation cooling required a modification of nozzle geometry in comparison to the DT thruster series. Indeed, the instabilities were not found at the critical currents for the formerly investigated water-cooled nozzle type thrusters of the DT-series with nozzle type geometry but due to the geometry difference this effect could not be assigned clearly to the anode surface temperature. Therefore, a thruster with the HAT nozzle geometry but with a water-cooled anode (CAT) was built showing that the instabilities were only delayed due to the different nozzle geometry which was confirmed by numerical investigation.

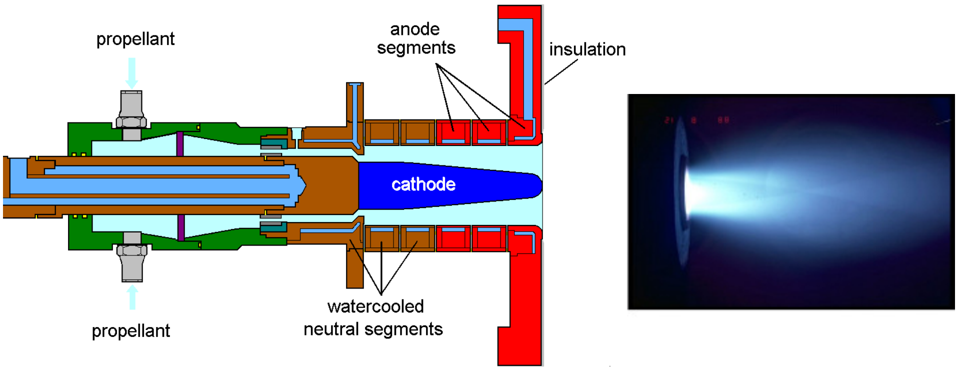

Figure 6. ZT3 geometry and thruster in operation with argon.

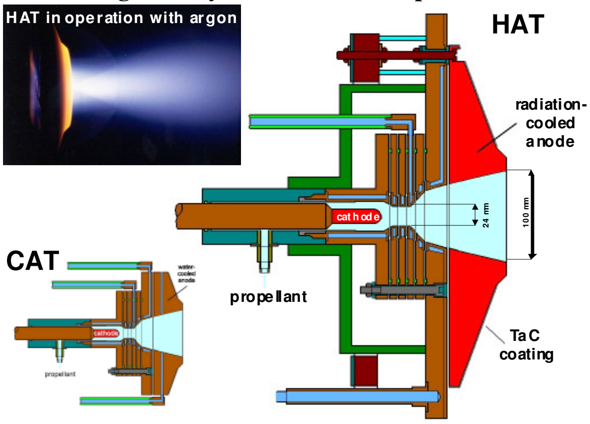

Figure 7. HAT and CAT geometries and HAT in operation (argon).

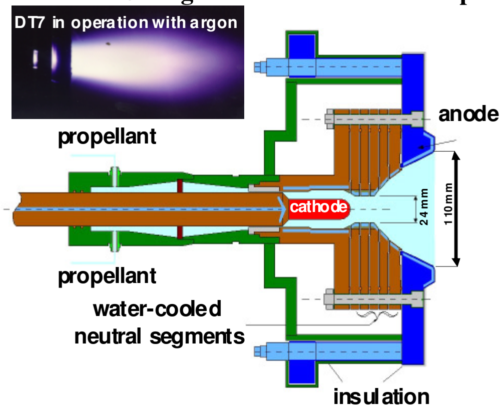

Figure 8. DT thruster geometry and thrusters in operation (argon).

The Fig. 9 and Fig. 10 show the measured thrust values. The numerical simulations agreed with the experimental results showing that a primary reason for plasma in-stabilities at high arc currents is the depletion of density and charge carriers in front of the anode because of the pinch effect.

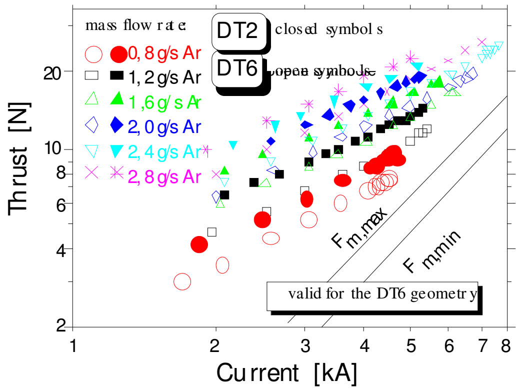

Figure 9. DT thrust performance data.

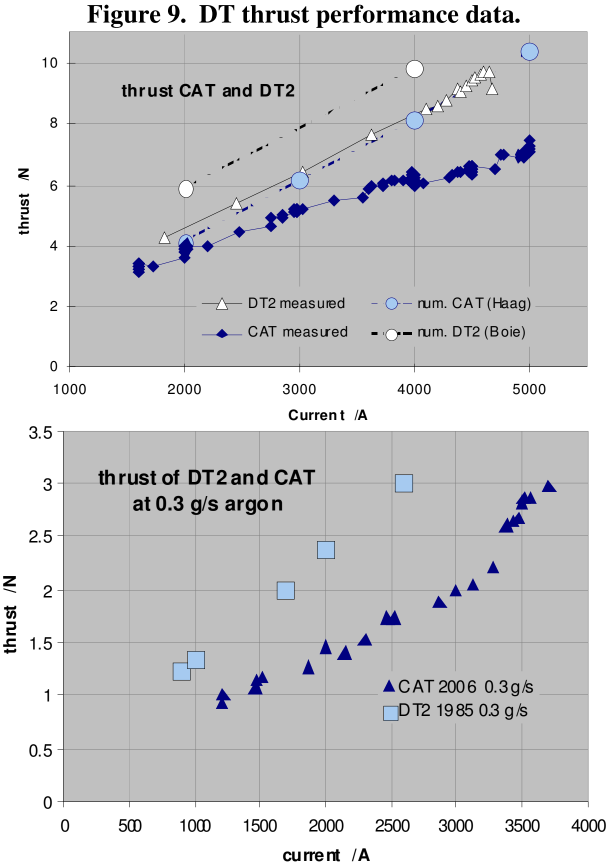

Figure 10. Measured thrust versus current for the DT2 and CAT at a mass flow rate of 0.8 g/s and 0.3 g/s argon in comparison with values from numerical simulation.

Even slight geometric changes can have an important influence on the initiation of instabilities due to slightly changed flow dynamic. This process can be observed in the results for the DT2 and for the HAT thrusters at different current levels. The results for both devices showed that the density close to the anode decreases remarkably stronger with current at the DT2 than at the HAT. For the simulations of the HAT and CAT thrusters different temperatures for the anode surface have been specified and held constant during the computation. The anode temperature for the CAT thruster simulations was set to 500 K and for the HAT thruster to 1500 K.

The comparison of the results of numerical simulations for the CAT and HAT thrusters at current levels ranging from 2000 A to 5000 A shows no significant difference in the plasma flow parameters. The numerical results agree with the assumption that the depletion of density and charge carriers is not mainly affected by the higher anode temperature of the HAT thruster, but by the thruster geometry.

D. Steady State AF-MPD Thrusters at IRS[2 and refs herein]

Stationary applied field magnetoplasmadynamic (AF-MPD) thrusters in the power range 5-100 kW are promising devices for orbit control systems of large satellites, because of their high specific impulse, thrust density and efficiency. Furthermore, AF-MPD thrusters at higher power levels appear to be excellently suited for interplanetary space missions like manned and unmanned Mars missions. The step to flight-qualified AF-MPD thrusters has not been taken yet. Besides the low pressure needed for experimental investigations of AF-MPD thrusters in ground test facilities, the optimization of the devices is difficult, because AF-MPD thrust depends on the distribution of interacting plasma parameters. In addition, the configuration of the applied magnetic field has a significant effect on thrust. The complex acceleration processes are not well understood yet. Therefore, efficient numerical simulation tools are needed to gain experience and to support further development. AF-MPD thrusters use four acceleration mechanisms: expansion through nozzle, interaction between self-induced magnetic field and discharge current resulting an axial and radial acceleration, interaction of discharge current and applied magnetic field leading to plasma rotation and interaction of induced hall current and applied magnetic field producing Lorenz force.

At IRS such a numerical tool is currently under development. The simulation software SAMSA (Self and applied field MPD thruster simulation algorithm) is intended to be used to achieve a better understanding of the basic plasmaphysical processes, which lead to the acceleration of the propellant, and to optimize the thruster and electrode geometry and particularly the configuration of the applied magnetic field of an AF-MPD thruster.

Figure 10. AF-MPD thruster mounted on thrust balance at IRS.

Due to axisymmetric thruster geometries to be investigated, the numerical scheme of SAMSA is based on an axisymmetric finite volume method on unstructured meshes. Considering all relevant acceleration mechanisms of AF-MPD thrusters, azimuthal components for electrodynamic variables and plasma velocity have to be included into the physical model. For this reason, a quasi three dimensional approach is used for the balance equations in cylindrical coordinates with azimuthal derivatives set to zero. Assuming continuum flow, the plasma flow is considered as quasineutral two-fluid plasma in thermal and chemical nonequilibrium. The arc discharge is described by a conservation equation for the azimuthal magnetic flux density or rather the equivalent stream function. The vector potential formulation is used for the description of the applied magnetic field, which has axial and radial components. So, not only the influence of the applied field but also the change of the entire magnetic field as a result of the induced current density is precisely manageable. Moreover, the zero divergence constraint is satisfied for the quasi three dimensional approach.

A second focus in the investigation of AF-MPD thrusters at IRS is the development and experimental investigation of a laboratory model, see Fig. 10 and Fig. 11. The intention of the experimental investigations is to determine the operation and performance parameters of the laboratory thruster and characterization of its plume. The experimental investigations allow a judgement of the quality of the numerical simulation program SAMSA.

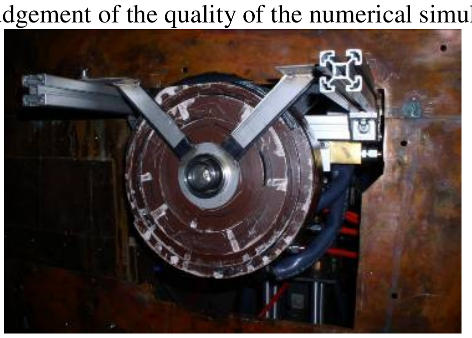

Figure 11. IRS AF-MPD thruster with coaxial magnetic coil.

The characterization of the thruster includes the variation of the applied magnetic field strength up to 0.5 T on axis of symmetry and axial variation of the magnetic coil. The spatial distribution of the magnetic flux density of the applied magnetic field was measured using a Hall probe. Measured magnetic field and numerical results are in good agreement.

Also, the influence of mass flow rate will be investigated and arc current and voltage are measured. To allow the measurement of thrust the laboratory thruster is assembled on a thrust balance. The thrust is evaluated by measuring the restoring force employing a force sensor mounted on a linear positioning system. The characterization of the thruster's plume includes measurements using electrostatic probes.

The IRS thruster mounted on the thrust balance is shown in Fig. 10. The Fig. 11 shows a front view of the AF-MPD thruster surrounded by the coaxial magnetic coil. The main design criterion was a modular assembly of the thruster. This allows changes in electrode geometry and magnetic field configuration as a consequence of numerical or experimental results. The thruster is radiation-cooled, which not only simplifies the design but also allows further development of the thruster with regard to a flight model. The IRS laboratory thruster has just been engaged successfully.

Based on the AF-MPD X16 thruster from DLR a laboratory AF-MPD ZT1 has been built up for a power range of 10 kW to investigate the optimization of the AF-MPD with respect to thrust and efficiency. The AF-MPD ZT1 was successfully operated at 2.7 kW discharge power in steady-state mode with applied magnetic of 0.14 T, arc current of 90 A and Argon mass flow rate of 10 mg/s (fig.12).

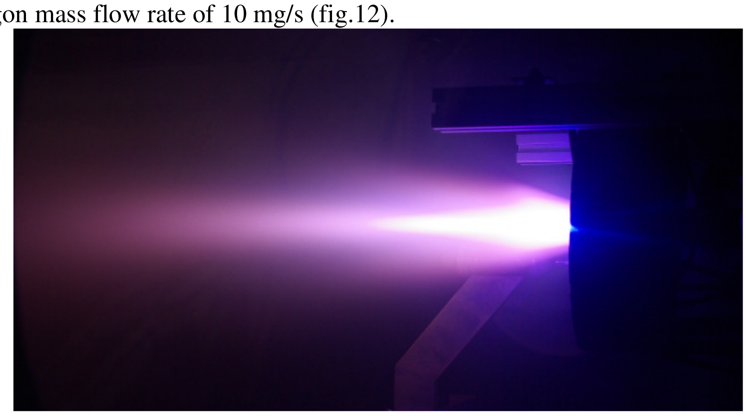

Figure 12. IRS AF-MPD thruster ZT1 in operation.

By comparing of experimental data with advanced scaling model for applied-field MPD thrusters a discrepancy in the scaling slope can be observed (fig.2). However, experimental data and model agree to a reasonable extent. For future work the experimental data base has to be extended in order to allow for a better validation of the used model.

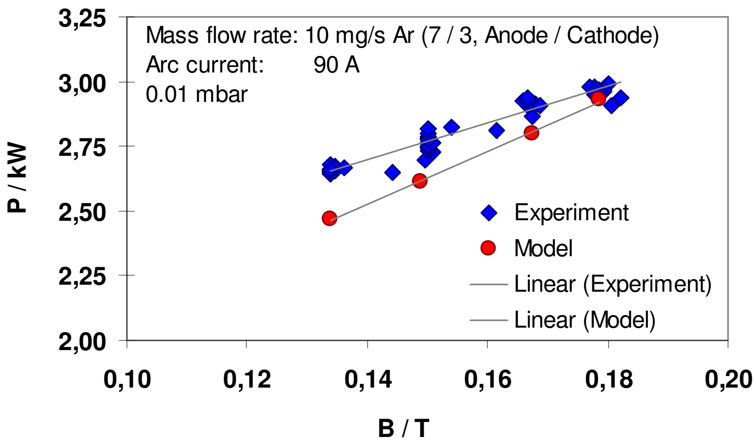

Figure 13. Comparing of experimental results with advanced anode scaling model.

To provide more experimental data for improving of advanced scaling model and for comparing with SAMSA Code a new active water cooled configuration of the thruster was designed to allow steady-state operation of the SX1 thruster in a power range of 10-20 kW.

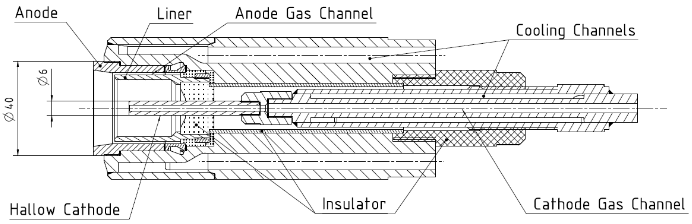

Figure 14. Sectional view of new AF-MPD SX1 thruster (Improved AF-MPD ZT1).

Involved in ESA and EU (HIPER) programs in cooperation with Alta, IRS is aiming for the development of high power electric propulsion systems.

In the framework of these programs, a 100 kW AF-MPD thruster is to be developed and tested within 2011. Primary designed for this project AF-MPD ZT2 was improved to SX3 (fig.4) configuration to guaranty a steady-state operation with respect to sufficient cooling in power level of 100 kW. The new redesigned 100 kW thruster AF-MPD SX3 is being manufactured. The followed experimental investigation of SX3 thruster will round the experimental data of SX1 thruster up with respect to thrust, efficiency and scaling behaviour of AF-MPD thrusters.

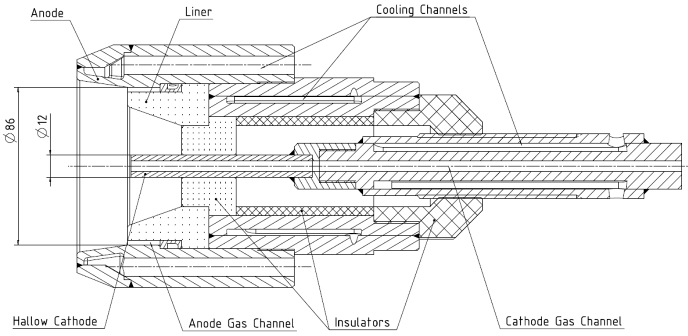

Figure 15. Sectional view of new AF-MPD SX3 (redesigned AF-MPD ZT2).

IV. Advanced electric propulsion systems at IRS

A variety of advanced space propulsion systems is investigated at IRS. Among them are the hybrid thruster TIHTUS, a concept which is investigated experimentally and numerically at IRS. This concept combines a water-cooled 100 kW arcjet as first stage with a high power inductively heated stage (IPG3). In addition, a grid-based inertial electrostatic confinement (IEC) system is currently being designed at IRS in cooperation with EADS and the Advanced Concepts Team of ESA.

The mini-magnetospheric plasma propulsion system, M2P2, is a concept with a low demand of propellant. This is a consequence of using solar wind energy to create a distinguished thrust. The main idea of M2P2 is based on works of Zubrin, the magnetic sail concept. Here, IRS uses the hybrid code system PICLAS where the non-collisional long term interactions, they are describing the plasma behaviour dominated by collective plasma phenomena and neglecting the coulomb collisions, described mathematically by the Vlasov equation. A widely used approach for solving this equation is the Particle-In-Cell (PIC) is used. The second part is the collisional long-range interactions, the Coulomb collisions which cannot necessarily be neglected for small spatial scales, i.e. at the plasma injection region. In such a case one would have to solve the Fokker-Planck equation. For both M2P2 and Coupled Electrodynamic Tether / Electrostatic Propulsion Systems (CETEP) such an approach is used to numerically describe function and operational properties of these advanced concepts.

E. TIHTUS[2 and refs herein]

A hybrid plasma thruster is under experimental and numerical investigation. TIHTUS (Thermal-Inductive Heated Thruster of the Universität Stuttgart) is two-staged and consists of an arcjet as the first stage and an inductively heated plasma generator (IPG) as the second stage. A photograph together with a schematic and the device in operation is shown in Fig. 16 the plasma plume downstream the arcjet has high temperature and velocity at the core and lower energies in the surrounding flow. To increase the energy in the plasma flow, the second stage inductively couples energy into the colder region of the plasma near the wall. For a better understanding of the physical coherences and for a comparison of the experimental results numerical investigations of the thruster are needed. After the validation of the numerical code, efforts in optimizing thrust and efficiency of TIHTUS can be performed.

The first stage of TIHTUS is a high power arcjet of the 100 kW class, called HIPARC that was developed at the IRS in cooperation with NASA. The plasma flow of the first stage is expanded in a nozzle before entering the second stage. The outflow of the first stage has a fast and hot core with steep gradients to the outer regions, as already mentioned. Between the two stages an additional mass flow can be added. This may be erosive gases, which would destroy the cathode of the first stage. The plasma now enters the second stage, an IPG. Due to the skin effect, mostly the colder regions of the flow near the wall are heated by the inductive discharge. This increases the energy in the plasma and hence the specific impulse and thrust. On the other hand the radial profile of the plasma flow is made more homogeneous by the second stage, which may be interesting for plasma technology processes.

For optimization of the thruster there are several interesting possibilities. One is the electric input power of the single stages and their ratio to each other. The geometry of the thruster and its single stages will also be of interest as well as the frequency of the inductive discharge, which is of course connected to the geometry of the coil. This leads to a large amount of experiments for optimization and some of the variations, like changes in geometry, are rather expensive both in money and time. This effort is intended to be reduced by numerical simulations.

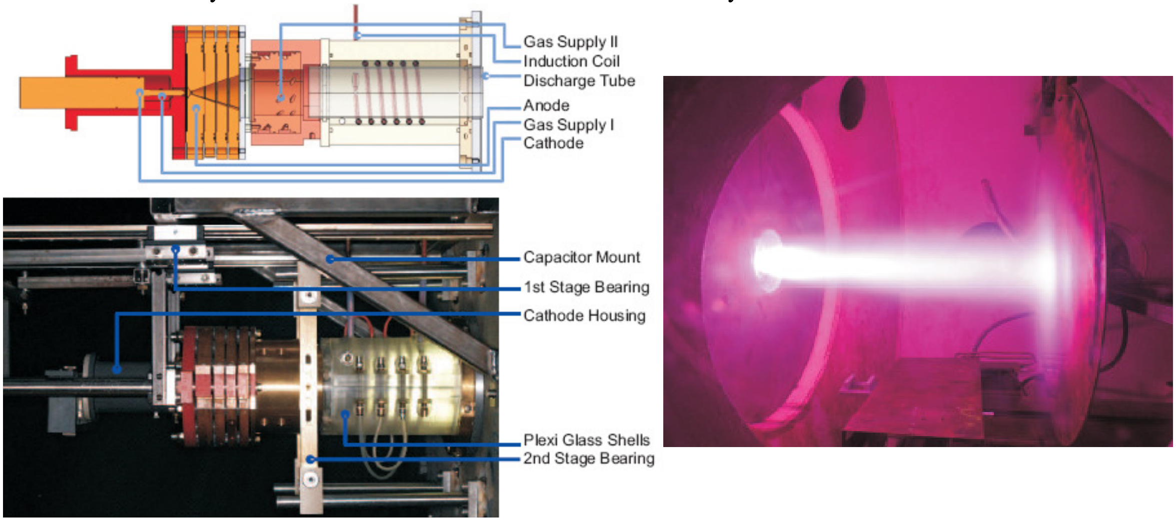

Figure 16. Schematic of TIHTUS (left) and in operation (right).

For the numerical simulation of TIHTUS the program system SINA (Sequential Iterative Non-equilibrium Algorithm) will be used and improved. SINA was developed in order to numerically simulate the complex thermal and chemical phenomena in the plasma wind tunnel facilities at the IRS that are used for the testing of heat shield materials. It can also be applied for axisymmetric plasma sources or thrusters with an electric arc. SINA consists of three different semi-implicit and explicit independent solvers which are loosely coupled, as shown in Fig. 17. It is also possible to additionally utilize the radiation solver HERTA, which computes the radiation loss in the flow field.

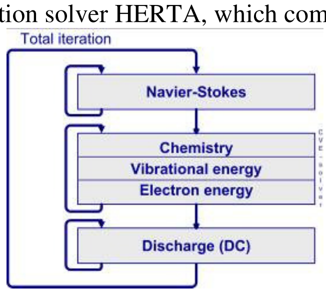

Figure 17. Structure of the code SINA for simulation of TIHTUS.

The flow solver (Navier-Stokes solver) accounts for mass, momentum and total energy conservation, by using an AUSM scheme. The usage of multi-block grids enables SINA to deal with complex geometries although structured grids are used. The time reduction for the numerical simulation is possible by the use of the program version, which runs on parallel computers.

The second solver (CVE solver) employs complex thermo-chemical relaxation models for different gases. In order to cope with the conditions in plasma wind tunnels, the transport properties are calculated for partially ionized air in chemical and thermal non-equilibrium employing an 11-component air model. The verification of the modeling of the complex thermal and chemical phenomena was performed by comparison with experimental data for air as flow medium. Since TIHTUS uses hydrogen as propellant, the CVE-solver has to be further developed to deal with this propellant.

The third solver (discharge solver) finally is used to solve the discharge equation in order to account for ohmic heating and magnetic acceleration in magnetoplasmadynamic plasma generators. At this time only a DC discharge of arcjets can be treated by the discharge solver. Thus, this part of the code has to be further developed, to deal additionally with the RF discharge of the second stage of TIHTUS or single IPGs.

F. Mini-magnetospheric plasma propulsion system[2, 3 and refs herein]

M2P2 is a concept with a low demand of propellant. This is a consequence of using solar wind energy to create a distinguished thrust. The main idea of M2P2 is based on works of Zubrin, the magnetic sail concept. Here, a coil is used to generate a magnetic field around a space craft. By the encounter of charged solar wind particles with the magnetosphere, there is an interaction according to the Lorentz force

$$\vec{F}_{Lor} = q\vec{v} \times \vec{B}\,, \tag{2}$$

where q is the particle charge, v is the particle velocity and the coil produced magnetic induction. Thus, the charged particles are deflected and produce a momentum transfer to the magnetosphere and finally to the spacecraft. This final momentum transfer to the magnetosphere depends on the magnitude of the interaction cross section between the magnetic field and the solar wind particles. The biggest problem of this concept is that a coil with a very large diameter of several km and a current of several kA are essential to produce a non-negligible momentum transfer to the magnetosphere.

Winglee et al. therefore proposed an idea to avoid the problem of the Zubrin concept injecting a plasma into the magnetic field. This causes an inflation of the magnetic bubble, explainable with the MHD induction eq. (10). So a magnetosphere around the spacecraft is created (fig. 11). The MHD induction equation

$$\frac{\partial \vec{B}}{\partial t} - \nabla \times \left(\vec{v} \times \vec{B}\right) = \frac{1}{\sigma\mu_0} \Delta\vec{B}\,, \tag{3}$$

(σ: electric conductivity, μ_0 magnetic constant) is valid in the vicinity of the plasma source, this means the consequence according equation (3) is a change in the magnetic induction $\vec{B}$ resulting in the discussed enlargement of the magnetosphere. Thus, the proportionality of the magnetic field changes from $B \propto 1/r^3$ (r: distance) to $B \propto 1/r^k \mid k < 3$ with the decrease parameter k.

The typical geometrical size of the enlarged magnetosphere corresponding on the recent theoretical studies is 20 (Winglee et al., see refs in 3) to 80 km (Khazanov et al., see refs in 3) for different estimations. This huge diameter is also the problem for experimental studies of the M2P2 system such that there are only a few experimental investigations which are focused on the change of a small magnetic field after the injection of the plasma (see refs in 3). The consequence is, that numerical simulations are essential for studies about the M2P2 system.

There are currently two main concepts for simulating the M2P2 system. The first is a MHD approach used by Winglee et al., see refs in 3, the second is an hybrid approach which is combining the MHD with the kinetic equations proposed by Khazanov et al., see refs in 3. The results of size of the magnetosphere and of the resulting force between the two approaches differ of six orders of magnitude which makes a statement about the possibility of a practical use of a M2P2 system extremely difficult. The reason for the large difference of the results are the different spatial scales for the underlying model approach spanning from a few centimeters up to approximately 50 km which is a great numerical difficulty. On the one hand the MHD approach is not valid for all required scales on the other hand is the coupling in a hybrid approach difficult and the kinetic approach is a great computational effort.

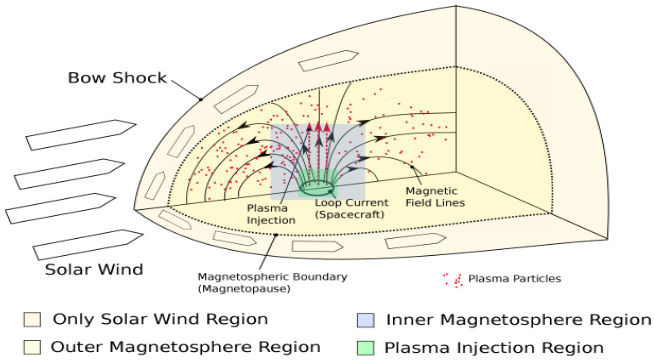

Figure 18. Schematics of the magnetic sail with plasma injection and enlarged magnetosphere.

Despite the computational effort we want to discuss about the application of a fully kinetic approach in order to obtain data on M2P2. The fundamental equation for the kinetic approach is the gas kinetic Boltzmann equation:

$$\left(\frac{\partial}{\partial t}+\vec{v}\,\frac{\partial}{\partial \vec{x}}+\frac{\vec{F}}{m}\frac{\partial}{\partial \vec{v}}\right)f\left(\vec{x},\vec{v},t\right)=\left.\frac{\partial f\left(\vec{x},\vec{v},t\right)}{\partial t}\right|_{coll}, \tag{4}$$

Here, $f\left(\vec{x},\vec{v},t\right)$ is the single particle distribution function at location $\vec{x}$, at time t, with velocity $\vec{v}$. Furthermore, $\vec{F}$ is an external force and m is the mass of the particles. The term on the right-hand side is called the collision term to describe the collision effects between particles. This term is the reason for the great mathematical effort in solving the Boltzmann equation, see refs in3. For the M2P2 problem the equation can be divided into two parts. The first part is the non-collisional long term interactions, they are describing the plasma behaviour dominated by collective plasma phenomena and neglecting the coulomb collisions, described mathematically by the Vlasov equation. A widely used approach for solving this equation is the Particle-In-Cell (PIC) method which should be used. The second part is the collisional long-range interactions, the Coulomb collisions which cannot necessarily be neglected for small spatial scales, i.e. at the plasma injection region. In such a case one would have to solve the Fokker-Planck equation. An example for a highly efficient method for solving this equation was developed by Nanbu, see refs in 3.

A brief discussion about the solvers and the numerical requirements for fully kinetic approach is given by ref 41 in ref 3, which should be realized within a project at IRS.

G. Further Investigation[2 and refs herein]

A set of advanced concepts is investigated at IRS. These experimental and theoretic assessments are described more in detail in reference 2.

Advanced water-fed iMPD

Pulsed Plasma Thrusters have been under investigation for several decades leading to successful satellite missions, most recently on FalconSatIII. They feature a robust and low cost design, providing high c_e, flexible and low power consumption as well as thrust control without losses in total Δv. A common configuration uses a solid block of Polytetrafluorethylene (Teflon™) as the propellant. This represents the sole type of PPT operated in space so far. However, researchers have pointed out the possible benefits of using liquid propellants.

The self regulated ablation from a solid block of Polytetrafluorethylene (PTFE, or Teflon™) is limiting the thruster performance, reason being that only a fraction of the ablated mass can actually be ionized and electromagnetically accelerated. The rest evaporates at relatively low velocities, a process, which is also known as late time ablation. The supply of a pre-determined liquid mass would improve the acceleration and hence thrust performance. A laboratory thruster, ne-glecting late time ablation could also provide valuable insights into efficiency margins of the PPT technology. Further, a liquid system could avoid layer depositions of carbon and fluoride on surfaces of the thruster and ad-jacent parts of the satellite altogether. Very small satellites with close by solar panels can be sensitive to any kind of contamination, which is not acceptable regarding the already scarce power budget in earth orbit. Also a liquid PPT system could be used with available waste liquids or in combination with other liquid propulsion systems.

To be able to compete with other propulsion systems, it is crucial for any pulsed liquid system design to keep as close to the simplicity of a solid propellant PPT as possible. A highly complex system would degrade the feasibility

in terms of cost, reliability, size and performance. This means, the normally occurring complexity of handling a liquid propellant and the system mass, i.e. from the tank structure, pumps, feed lines, valves and controllers, must be reduced to a minimum effort. A possible approach to achieve this is to literally stick to the block, meaning to renew the area of the surface layer for ablation after each pulse. This can be imagined quite similar to a burning candle, using a wick to draw from the available liquid. The liquid has to be fed to the wick-element in a self regulated and passive manner, for the reasons given above.
The following section will give an overview of the proposed system design and working principle for pulsed liquid operation. Fig. 19 shows a schematic overview of the respective system components, with the arrows pointing in the flow direction.

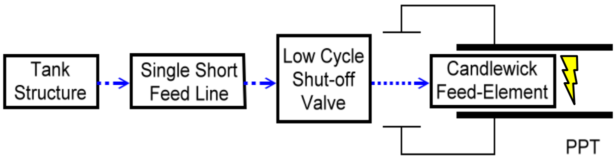

Figure 19. Schematic Setup of Liquid PPT.

The working principle is to supply the liquid, i.e. the purified water, through a very short feed line to the wick-element. The wick will saturate with water from the tank by means of the pressure gradient between the tank and ambient space. The tank structure is commonly a heavy component due to high pressure propellant storage and losses from outgassing into space. However, the wick-element can work on very low supply pressures as it soaks up with the liquid and the PPT operation does only require very low mass flow rates in the µg-range. This allows for utilization of a very lightweight tank design, which can further be shaped with regard to the storage volume required and satellite demands. The very short supply line as well as only one shut-off valve required both help to further keep the system dry mass low. The valve ensures safety from unintended mass flow during transition into orbit or in case of PPT idle times. Therefore, it features very low operation cycles and is the only moving part of the system. No pumps or controllers are included whatsoever, as the mass flow is self regulated by the candlewick. Due to the low ambient pressure and low temperature in space, a layer of solid ice is created on the surface of the wick between the two PPT electrodes drafted in fig. 19. The operation of the PPT is considered similar between liquid and PTFE feeding. After initiation of the PPT pulse, a discharge arc forming between the thruster electrodes and across the surface of the wick-element will transfer energy to the wick surface, thus ablating the propellant similar to its PTFE-pendant. After the propellant from the surface has been ablated, no late time ablation is likely to occur. In order to avoid ablation from the wick element itself, ceramic materials are considered for implementation.
The limitation in overall size of the liquid PPT system suggests a possible application on Cubesat-type satellites, prolonging lifetime of the satellite as compared to operation without any thrusters. This demand is supported by recent ESA studies and an increasing interest in easy to launch and cheap flight opportunities for research and technology demonstration.
First steps towards a liquid PPT design are on the way at IRS by means of building up of an experimental laboratory setup for testing of handling, feeding and pulsed operation with purified water. The feasibility of this type of system has been successfully demonstrated by operating a liquid fed PPT-thruster inside IRS test facilities with a porous feeding element. The thruster allowed for safe and controlled operation and showed no misbehaviour. The characterization, analysis of the preliminary results and design changes are subject to further investigation at IRS. A photograph of the pulsed discharge during thruster operation with purified water inside the vacuum chamber is shown in fig. 20.

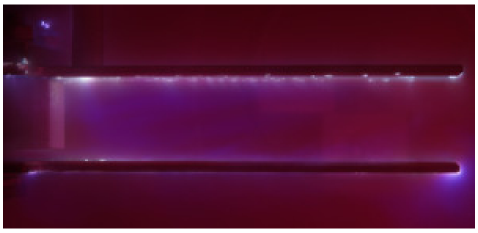

Figure 20. Liquid PPT during Operation at IRS.

Inertial Electrostatic Confinement: Status

The most common fusion based setups, TOKAMAK based reactors as ITER or inertial confinement fusion as in NIF, require huge, heavy and complex structures. A more applicable solution due to their simplicity of setup is

Inertial Electrostatic Confinement (IEC). This technology offers high energy densities in small and light reactors. Inertial electrostatic confinement IEC, originally proposed by Farnsworth, the inventor of electronic television, was first studied experimentally by Hirsch in the 1970s. However little was done to study this concept until R.W. Bussard and G.H. Miley renewed studies in the early 1990s. There are several design types of IECs, e.g. single-grid devices, multiple-grid designs or a hybrid magnetic-electrostatic version. All these experiments are fusion power concepts, yet no breakeven is expected. But due to the design simplicity and scalability, development was still pursued and research done over decades aimed to increase the fusion rates as much as possible, so that they can be used as commercial neutron sources as it was planned by Daimler-Chrysler Aerospace (IEC star-mode fusion neutron source, R&D). Today the NSD Fusion GmbH pushes the enhancement of commercial IEC devices amongst other things for luggage inspections or neutron radiography in Germany. The development activities are in strong cooperation with EADS and the advanced concepts team of ESA.

Setup and Principle

The simplest setup, explained in this chapter, was thought up by Farnsworth. A spherical, concentric and strong negatively biased grid is placed in a grounded, evacuated, spherical chamber (Figure 21). The grid is the cathode, the chamber wall the anode. The chamber will be flooded with a fuel. By a glow discharge, ions can be generated which will then be accelerated to the grid center. If the ion energy is sufficient, occasionally fusion processes can be recognized. But the majority of ions doesn't take part in a fusion process and leaves the grid center. Due to the potential gradient between cathode and anode they become slower and get accelerated into the center again.

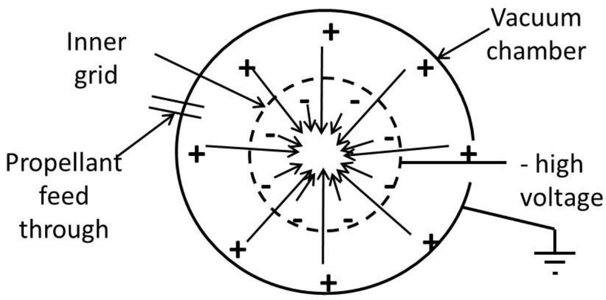

Figure 21: Scheme IEC device.

Outlook

The potential of IEC for space applications was discovered in the USA and Japan decades ago. The reason for starting investigations in the field of IEC is that the technique belongs to the most promising concepts for the next generation of electric propulsion with high specific impulses. At the University of Illinois IEC was tested for several applications, inter alia ion thrusters. An IEC setup can also be operated in ranges where no fusion processes occur. Ions will be accelerated into the center of the cathode grid and will be trapped there until a hole in the potential surface of the cathode grid allows them to escape. This was tested for the first time by the G.H. Miley group. In these studies the IEC thruster had a thrust of about 30mN and an I_{sp} of 3000s and was compared to conventional ion thrusters. But it also seems promising technology for specific impulses in the range of e.g. the DS4G with 21400s or higher. Since it is feasible to accelerate ions to an energy that is enough to allow for particles to fuse, it may also be possible to neglect these fusion processes and to use the high kinetic energy of ions to generate thrust. Since an IEC thruster doesn't consists of the conventional grid configuration, but uses spherical wire grids with an effective transparency of 98%, which can be increased under certain operating conditions as in the *star mode*, the severest life-time limiting factor of conventional ion thrusters can be reduced in IEC thrusters. Additionally the alternative wire grid system is very light weighted. In this concept, it may be thinkable to have a two-stage system that allows operating in ion thruster and fusion thruster mode.

Moreover the extracted ion beam shall be a reference case for the PICLAS code which is under development at IRS (University of Stuttgart), IAG (University of Stuttgart) and IHM (Forschungszentrum Karlsruhe). PICLAS is supposed to model and simulate highly rarefied plasma flows, e.g. as in Pulsed Plasma Thrusters, by combining Particle-In-Cell (PIC) with Direct-Simulation-Monte-Carlo (DSMC) and Fokker-Planck (FP).

If a plasma beam with very high ion velocities can be extracted, a third interesting aspect is the potential application for simulation of high energetic radiation. IRS is following a cooperation with the Center for Astrophysics, Space Physics and Engineering Research of the Baylor University so as to develop a facility for environmental simulation of complex dusty plasmas in space using a inductively- heated plasma generator. The IEC test stand could possibly

provide ions with energies that are in the range of natural plasmas and can be simulated by the above mentioned facility only insufficiently.
In order to gain the knowledge necessary for the development of a new space propulsion system, the fundamentals of confinement and plasma beam extraction have to be understood. At IRS, there are studies, which deal with these topics. A test stand will be build up in the IRS Laboratory consisting of a two-grid-system. For this a gird design study has been conducted in order to obtain the most adequate setup for a test campaign that is supposed to deliver knowledge about the confinement process within the cathode grid and certain operation conditions e.g. the *star mode*. In a second test sequence a plasma beam extraction shall be established by allowing for a through in the potential surface of the cathode grid. If the plasma coupling out is successful the beam shall be examined with Langmuir probes and LIF measurements in order to obtain information about electron and ion properties.

V. Conclusion

The research and development activities at IRS in the field of electric space propulsion have been presented. However, this work is a mixture between a description of the status quo at IRS and some scientific results. Therefore, the authors emphasize that an evaluation to detail knowledge and information on the theme can be achieved by assessing the references of this paper.

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