A Thermal Pulsed Plasma Truster for Microsatellite Propulsion

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The laboratory model of an electrothermal pulsed plasma thruster (PET) in the 3J range has been developed at IRS and a new vacuum facility has been set-up for testing. The goal is to provide a compact and cheap device for propulsion of university micro-satellite technology testbeds and for hands on education of students at the University of Stuttgart. It is without question, that PPT's are considered leading candidates to extend life and mission capabilities of micro-satellites. Further, the simplicity of PET's over other types of electric propulsion (EP) for practical education has remarkably long been left untouched, despite the gap between theoretical lectures and laboratory practice for the next generation of EP scientists and engineers. The presented PET features a simple, easy to handle, modular design for changing of its electrical parameters and geometry. It will be incorporated in a mandatory "student laboratory for electric propulsion" (LEA) at IRS. A short overview on micro-propulsion in literature and the LEA-project is presented. The design and working principle of the PET developed at IRS are described. The PET was tested at IRS for the temporal discharge voltage, I_{bit} and m_{bit}. Preliminary results are shown, briefly discussed and compared to literature.

Nomenclature

$A_{circuit}$	= Effective area surrounded by the discharge current
c_e	= Effective exhaust velocity
E_0	= Initial energy per discharge pulse
I	= Averaged discharge current
I_{bit}	= Change of impulse per discharge pulse
I_{sp}	= Specific impulse
k	= Empiric correlation factor
m_{bit}	= Total propellant mass per discharge pulse
r_a	= Cathode radius
r_c	= Anode radius
τ_c	= Characteristic discharge time
Δv	= Maximum change of speed
x_{gap}	= Electrode gap between thruster anode and cathode
RLC	= Circuit consisting of resistor, inductor, and capacitor

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I. Introduction and Backround

The ablative pulsed plasma thruster (PPT) came up in the 1960's, when ideas about the concept arose fueled by limitations of the satellite's available power output and the desire to increase the specific impulse I_{sp} of commonly used chemical thrusters.[1,2] The feasibility of pulsed micro-thrusters has successfully been demonstrated in the same decade by the ZOND 2 and LES 6 satellites[2,3] without any reported advers effects on the spacecraft. While numerous PPT-systems have been qualified for space flight in the past, three have actually been flown in the last decade.[4−6] The many advantages of PPT's have been pointed out by numerous researchers in literature,[1,7−9] including high I_{sp} together with other electric thrusters and unmatched simplicity. Commonly achieved thrust efficiencies of less than 20% are low in comparison. However, experimental and numerical studies have looked at the electric circuit, thruster geometry, propellant utilization and frozen flow limitations in the discharge pulse for improvement.[10−16]

General cuts in development budgets and low availability of launchers have kept up the recent strong interest in the development of micro-satellites to maintain access to space for the scientific community. This goes hand in hand with the miniaturization of technology and the corresponding benefits of easier handling und much lower procurement costs for propulsion systems. Affordable, standardized satellite testbeds and regularly available space flight opportunities are required for feasible, sustained development and research in space. The mission capabilities of drifting, propulsion-less micro-satellites often fall short to requirements, lending themselves to very limited capabilities by offering no means of positioning and only short orbit lifes due to safety regulations. This could generally be improved by an on-board micro-propulsion system, for example for research, attitude control, station keeping, orbit transfers, de-orbiting at end-of-life (EOL), de-tumbling and formation flights to name a few. Even though orbit maneuvers like a Hohmann transfer require short, high thrust phases, rather than constant operation of an electric thruster, Wong et al.[17] pointed out, that the thrust demands for micro-satellites may be less stringent, allowing for low-thrust orbit-raising.

An extensive breakdown of micro-propulsion thrusters for the TechSat21-mission identifies micro-PPT's as leading candidates, rather being mass limited by the PPU and capacitors then power limited.[18] This could even be extended as more recent survey results become available. For the scope and background of the presented work and in accordance with a recent worldwide PPT review and classification at IRS,[19] the definition of micro-spacecraft by the U.S. Air Force is used.[20] Referring to this, a micro-spacecraft has a mass of 100kg or less, which is further broken down into several classes of spacecraft, for example masses of 10kg or less are referred to as nano-spacecraft. Recent international development activities have focussed on coming up with new PPT-thrusters for a new fleet of micro-/ and nano-spacecraft with wet masses ranging from 25kg to as little as few kilograms.[21−29] The micro-satellites have much more stringent limits due to their overall compact system in comparison to common satellites. Some of the derived unique technical requirements for a micro-propulsion-system are:

- Very low power consumption,
- Simple or no propellant feeding mechanism,
- Minimal propulsion system mass,
- Compact size,
- Reduced life-cycle cost, including substancial reductions in procurement costs,
- Mass production technologies,
- Scaleable technologies,
- Spacecraft compatibility (contamination).

PPT's are leading in fulfilling all these requirements, due to their inherent simplicity. The corresponding low number and complexity of components has a positive impact on cost reduction and minimizes technical risks in the development process. Due to the relatively low amounts of solid propellant for a mission, an ablative PPT experiences a change in impulse bit with increasing number of shots as the shape of the solid propellant changes. The total abandonment of a propellant feeding mechanism on a PET was found to even have a positive influence on this by increasing the impulse bit.[25] In addition to the technical requirements, demands on the functionality and mission performance of the thrusters, although varying, can be derived.

Since the interest in micro-propulsion in Europe is still on the raise, reliable studies are years away from being available. To put up bearings, upper thresholds can be found in literature and a recent ESA Tender for "Medium ΔV, Low Power, Low Voltage, μ-thruster Module Breadboarding".[17,30] Endurance tests have demonstrated very long operating lifes of between 0.5 to 3 million pulses.[7,29,31] Of course also missions with lower number of shots are feasible.[25]

Micro-propulsion with power levels below 100 W and its optimization is still in its infancy, requiring much more research, investigation and testing of "scaled-down" concepts with respect to mission and satellite demands as well as available technologies. At present, two low-power micro-thrusters are under investigation at IRS, a very-low-power-arcjet (VELARC)[32] and the PET presented in this paper. To work towards capable micro-propulsion systems it is clear, that optimization always has to be carried out with respect to the resources and goals of the respective mission. However, this is beyond the scope of the presented work. The presented PET also represents a promising opportunity to better get the next generation of researchers involved, building a bridge towards sustained research and investing into high quality education at IRS. This presence and the thruster's easy handling may also serve well in the public promotion of electric propulsion in general.

II. Thruster Description

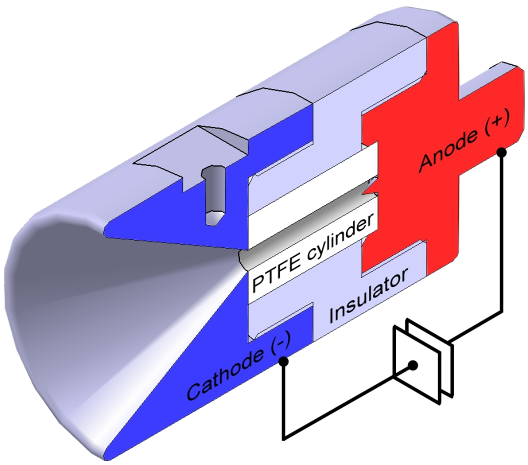
Besides the common geometry of pulsed plasma thrusters with coaxial parallel electrodes described in literature[33,34,35], the Pulsed Electrothermal Thruster, or PET, uses a coaxial serial electrode configuration.[1,24,36] The Coaxial PPT features parallel, aligned electrodes as defined by Jahn.[2] A PET however consists of two electrodes at either end of the thruster, distinct from a pulsed self-field MPD[7] by an internal high-pressure-discharge cavity. Figure1(a) shows a schematic overview of the PET-assembly at IRS.

The components of the ablative PET are similar to conventional parallel plate PPT's, representing an RLC-circuit. They include two electrodes made of copper or tunxten, the cathode and anode. The cathode, acts as exhaust electrode, forming a type of nozzle. The anode is closed. Both of the electrodes are placed on either side of a hollow cylindrical electrical insulator. Often this insulator is made of acryl, or plexiglas®, which allows for optic access to the cavity. Solid polytetrafluoroethylene (PTFE), or Teflon^{TM} us used as a propellant. The propellant can be fed continuously by a mechanic spring or be provided as a single batch. However, Shintani et al. have presented a compromise between both principles, by changing an entire batch after its worn off. For a more detailed description of the disc-feed-mechanism, please refer to the literature.[24] Low molecular weight gases are feasible for operation, though not within the scope of this work.

The PTFE is made into a hollow cylinder to form a long skinny discharge cavity. The cylinder is fit into the hole of the insulator between the two electrodes. Thus, the length of the cylinder represents the electrode gap. Both of the electrodes are connected to a capacitor, that stores the electric energy for a discharge pulse. Further, typically a spark-plug is fit into the cathode nozzle and protrudes into the gap between the electrodes.

Prior to a pulse, a voltage of several kV is applied to the capacitor to charge it. An electric field now exists between the electrodes, with the insulation preventing unwanted breakdown. To initiate a discharge pulse, the spark plug is triggered. This donates electrons into the field and creates a pressure increase in the cavity. The loop is closed and breakdown of the capacitor occurs. A high current pulse of several kA runs through the cavity, creating and heating a plasma. PTFE from the inner cavity wall material is ablated and ionized to a degree. The high pressure plasma is then accelerated through the cathode nozzle by gas-dynamic forces, marking the PET as an electrothermal device. An electromagnetic thrust component can occur depending on the thruster design. It can be estimated by the equation for MPD thrusters by Jahn[2] $T_{magn} = (\mu_0/4 \cdot \pi) \cdot I^2 \cdot \ln{(r_a/r_c)}$.

For the presented PET no magnetic thrust component is expected, as $r_a/r_c = 1$. In this case, the effective exhaust velocity c_e represents the mass averaged thermal velocity over all contributing species in the exhaust flow $\bar{c}$.[16,37] However, assuming a discharge current of 5kA and by changing the design of the nozzle to an electrode radius ratio of $r_a/r_c = 2.5$, an electromagnetic impulse bit of $2.29\mu Ns$ could be achieved in addition to the thermal impulse. At the end of a pulse, the energy inside the capacitor is replenished for the next pulse. Electrothermal thrusters typically suffer from limitations of the frozen flow, since the internal ionization energy is not recoverable. Burton et al.[36] pointed out, that high recombination rates must therefore be achieved by means of design optimization. Working towards equilibrium conditions inside the cavity during discharge would allow to recover additional energy for acceleration of the propellant.



(a) Thruster setup of a PET.

(b) PET and several thruster modules at IRS.

Figure 1. PET thruster setup and modular concept.

A picture of the assembled PET developed at IRS together with several component modules is shown in Fig.1(b). The cylindric thruster has an overall length of 55mm and a diameter of 32mm. The modular design is very simply realized by screwing the desired components together. For variation of the electrode gap x_{gap}, insulators with different lengths can be inserted between the electrodes. The cavity can be changed by using PTFE-cylinders with various inner diameters. Different types of cathode nozzles can be used for variation of the nozzle shape. For variation of the electric parameters, two different capacitors with 1.5μF or 10μF are available. The described characteristics are ideal for parameter variation studies. The major benefits of this simple approach are threefold. First, this allows for a systematic optimization of the design and thruster operation as well as research to improve understanding of the PET principle and simulation codes at IRS. Second, the PET offers a suitable platform for hands on education on scientific methods and laboratory experience as well as for the promotion of electric propulsion in general. Third, this allows for improvements in the understanding of scaling of the PPT-technology as well as for necessary adjustments to the measurement equipment and methods at IRS due to the new demands of micro-thrusters.

III. LEA - Student Laboratory for Electric Propulsion

Lectures on electric propulsion, from the principles to the design and application of thermal, magneto-plasmadynamic and electrostatic thrusters, are embedded into the specialization of space systems at the IRS. The topics and examples that are part of the lectures mirror latest achievements in science and industry, current developments of electric thrusters and reflect on their impact on space exploration. The lectures of course also act as an open door and showcase for students to evaluate options to get involved in the field, which is anxiously used.

While the lectures can mostly only cover theoretic and factual knowledge, this is often experienced by students to be hindering with respect to internalizing the complex correlations and principles. This becomes especially apparent, when the knowledge and methods are rooted in experience from measurements and laboratory development. Guided laboratory tours are part of the lectures at IRS, although feedback from students indicates the need for a more involving educational instrument. With the introduction of the bachelor and master courses at IRS, theses comprising a work period of only three month require generally experienced students for them to actually be able to focus on developing scientific skills and successfully fulfilling their tasks. The wish for basic practical preparation is also indicated with respect to a thesis in the industry and/or abroad. This, combined with the wide variety of subjects covered at IRS, creates an ever growing demand for a possibility to gather first practical experience on measurement equipment, laboratory processes and practical problem solving before entering written examinations or starting a thesis.

True to the code "Learning by doing", a new mandatory student laboratory lecture for electric propulsion (LEA) has entered the build-up phase, with work about five months in. The goal of the laboratory is to close the gap between theory and basic hands-on experience on electric propulsion systems, to get students involved and not limit them to passive observation. In the past, this step has been prevented by safety conciderations and the mere complexity and fragility of most electric propulsion laboratory applications. Not so with the PET presented in this paper. Its easy to handle design and operation, combined with affordable hardware and minimal spatial requirements still offers the full research capability and numerous measurement methods. For the first time at IRS, a suitable hardware platform and education instrument becomes available for hands-on implementation and internalization of the acquired theoretic knowledge and basic measurement equipment as well as software skills. The laboratory seminar will include:

- Principle and operation of electric propulsion,
- Thruster modeling for prediction and evaluation of performance,
- Measurement methods and equipment for experimental investigation,
- Independent preparation and carrying out of experiments,
- Data processing, analysis and evaluation,
- Writing of a report.

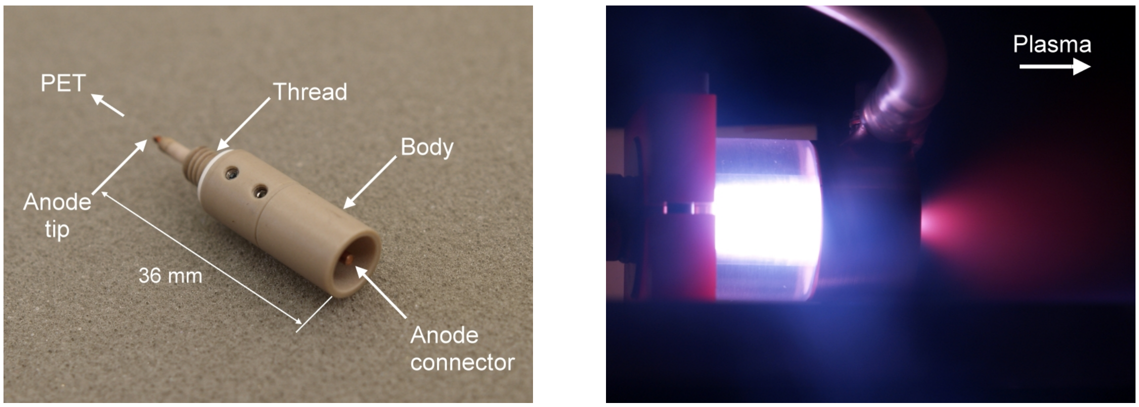
It should be noted, that the PET and the facility was literally designed by students for students. The small test chamber shown in Fig.2 has been set-up and successfully tested at IRS. It has a length of 0.5m and a diameter of 0.3m. The work is funded in equal parts by the University of Stuttgart and the IRS. An evaluation phase of the laboratory is planed for the upcoming winter semester.

Figure 2. PET vacuum test facility at IRS.

IV. Experimental Setup

A first test series, including measurement of mass bit and Impulse bit has been carried out at IRS to determine the PET characteristics. In order to do this, the electrode gap x_{gap} and pulse energy E_0 have been varied. Prior to the thruster characterization, functional tests were run, focussing on thruster operability, i.e. thruster ignition and discharge behavior. First functional tests were conducted using a unison semiconductor ignitor. The type of igniter was adapted from pulsed magnetoplasmadynamic thrusters (iMPD) at IRS. Due to the compact design of the PET, the igniter was placed downstream of the exhaust nozzle and facing towards the cavity.

First tests showed satisfying results with respect to ignition, yet proved difficulties with respect to the discharge. With the rather bulky igniter size, breakdown outside of the thruster was observed on a few occasions which was solved by increasing the insulation. Further, sputtering of the igniter surface occurred, which was directly exposed to the accelerated plasma. As a result, a new custom high-voltage (HV) igniter was built and tested. The final HV-igniter design was compact enough to be incorporated into the cathode nozzle. When triggered, the igniter creates a spark between its own anode and the thruster cathode. Since a ceramic was used for insulation, the igniter trigger voltage increased accordingly. An ASP laboratory breadboard capable of up to 20kV was sufficient for operation. A picture of the HV-igniter and the PET during operation with an electrode gap of $x_{gap} = 20$mm are presented in Fig.3.



(a) High-voltage-igniter for PET operation.

(b) PET with 20mm electrode gap during vacuum operation.

Figure 3. HV-igniter and PET operation.

The temporal discharge voltage was measured with a high-voltage-differential-probe. For the impulse bit testing, the PET was mounted on an impulse pendulum in test chamber 16 at IRS. The pendulum allows for measurement of the I_{bit} via a very sensitive optical sensor, that monitors the pendulum deflection over time. The magnitude of the deflection can be correlated to the impulse energy that was transferred from the thruster to the pendulum. For a more detailed description of the pendulum setup and calibration, please refer to literature.[16,38] Since the igniter electronic is directly mounted on the thrust balance, I_{bit} tests were carried out using the semiconductor igniter. For each configuration, 9 signals where recorded and the average value was taken. The experimental setup on the impulse pendulum is shown in Fig.4.

The pressure inside the chamber 16 was monitored by a Pfeiffer Full-Range vacuum gauge (PKR 251) and kept below 0.05Pa. All other tests were carried out with the HV-igniter in the new PET-facility. The small facility reaches ambient pressures of 10^{-4}Pa during thruster operation. For measurement of the m_{bit} the thruster was continuously operated for 500 pulses with a constant pulse frequency of 1Hz. The mass of the PTFE-cylinder before and after the test was measured and the difference divided by the number of pulses to get the total ablated mass per pulse. The same capacitor has been used in all tests. It has a capacitance of 1.5μF and was charged up to 2kV. For the thruster cathode a diverging nozzle with a length of 22mm was used. The throat diameter of the nozzle matches the cavity diameter, which was kept constant for all tests at 4mm. Consistency of thruster operation with the two different igniters was evaluated prior to testing by means of comparing the respective discharge voltage signals and produced mass bits. An overview of the test parameters is given in Table 1.

Table 1. Overview of test parameters.

E_0 / J	3.0			2.5			2.0			1.5	1.0	0.5
x_{gap} / mm	20	32	50	20	32	50	20	32	50	32	32	32
I_{bit}	x	x	x	x	x	x	x	x	x	x	x	x
m_{bit}	x	x	x	-	x	-	-	x	-	-	-	-

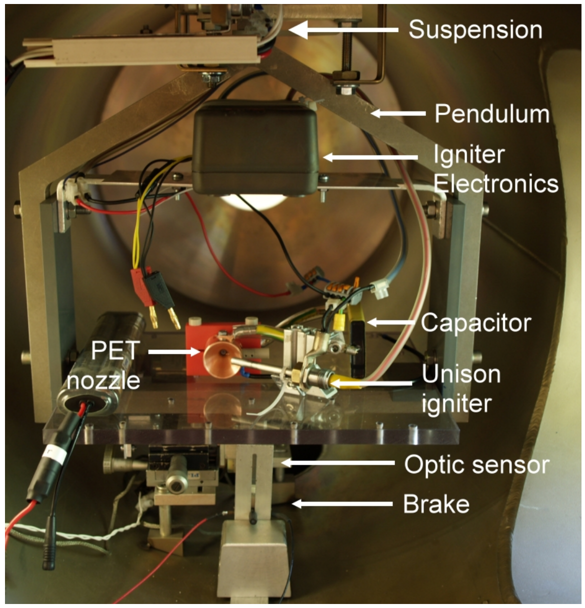


Figure 4. Experimental setup on the impulse pendulum at IRS.

V. Preliminary Results

The goal of the measurements was to confirm flawless operation and gain first insights into the performance and discharge behavior of the presented PET. In general the PET demonstrated very stable operation and a repeatable discharge behavior at energy levels of 2.0J and above. A relatively abrupt threshold for successful ignition with respect to the ignition spark energy was observed, suggesting a possible sensitivity of the igniter to carbon depositions. Although no misfire had been observed, cleaning of the HV-igniter before each test was performed to be on the safe side. Endurance testing of the PET will be necessary for closer investigation. Measurement of the m_{bit} with both types of igniters revealed no impact of the utilized igniter type on thruster operation and performance. The temporal discharge voltage was recorded for indication of ignition success and evaluation of the thruster operation. Figure 5 shows the discharge voltage over time for the three utilized electrode gaps at a common discharge energy of 3J, which corresponds to a capacitor charge voltage of 2000V.

The high frequency noise on the signals around the 0μs mark is caused by the HV-igniter. Overall discharge duration stays below 8μs. For estimation of the electric parameters, the PET can be modeled by a damped RLC-circuit.[2] A decrease in amplitude for the first half-cycle as well as in overall discharge duration can be noted with increasing x_{gap}. Both of these observations show increased damping, which is defined as $\delta = -R/(2 \cdot L)$. This indicates higher resistance and undesired energy losses. The current rise is assumed lowest for the 50mm signal, indicated by a lesser voltage drop following the 0μs mark.

To estimate the influence of the PET's electric circuit inductance L, the characteristic discharge time $\tau_c = \sqrt{L \cdot C} = T/2\pi$ can be used, wherein T is the signal period. It should be noted that, since the discharge current arc is not subject to significant propagation like in iMPD-thrusters, the RLC-circuit-parameters capacitance C and inductance L are assumed to be constant over time and only dependent on the geometry and the capacitor. However the circuit resistance R depends on the changing conditions in the discharge arc. With reference to the 2μs mark in Fig.5, the fist half-cycle's amplitude is delayed with increasing x_{gap}. The increase in the period T also means an increase of L. This can be expected and explained by the increase in effective area $A_{circuit}$ surrounded by discharge current.

To evaluate the aforementioned observations, the electric parameters were estimated from the signals in Fig.5. This was done by fitting a sinodial function to each of the voltage signals and applying the RLC-model by Jahn.[2] The circuit's resistance $\bar{R}$ was further assumed to be constant over time. The results are summarized in Table 2.

An effective gas-dynamic acceleration favors high pressure raises combined with high recombination rates, which implies a high discharge current magnitude and fast raise times. The estimated values show the

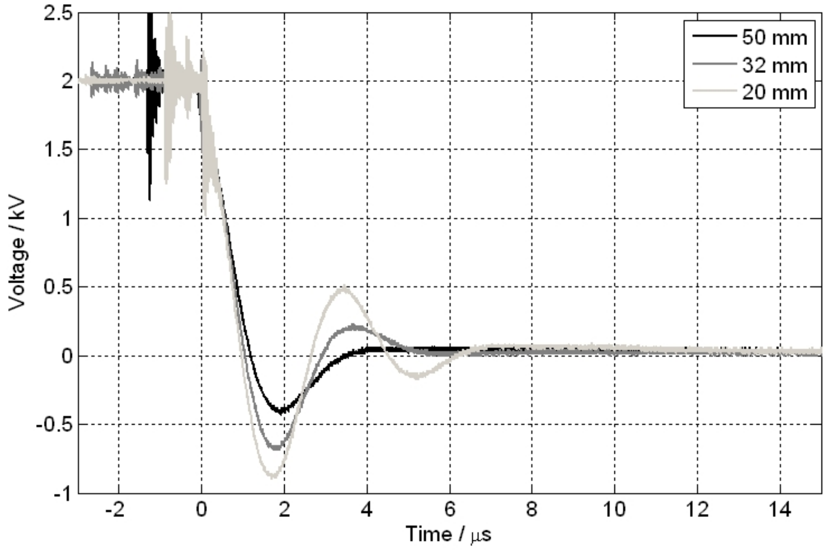


Figure 5. Influence of x_{gap}-variation on PET temporal discharge voltage for 3J pulse energy.

Table 2. Estimated mean resistance and inductance.

x_{gap} / mm	20	32	50
C / μF	1.5	1.5	1.5
$\bar{R}$ / mΩ	745	782	871
L / nH	118	166	232
τ_c / μs	0.71	0.62	0.56

predicted increase in $\bar{R}$ and L with increasing x_{gap}. In general they are relatively high, although it should be noted, that the numbers represent an unoptimized system. Model predictions indicate higher thrust efficiency for τ_c values between $1.5 - 2$.[8]

Figure6 shows the impulse bit over the pulse energy for an electrode gap of 20mm, 32mm and 50mm.

For energy levels of 2J and below, the 50mm configuration shows the lowest impulse, which could be due to the relatively low energy per propellant surface area as well as an issue with HV-igniter triggering and igniter energy, which has only been solved recently. More conclusive data is subject to future investigation. At 2J and above however, reliable operation was possible for all configurations. The error bars represent the standard deviation. From 2J on, the 50mm PET has the highest slope and increases its impulse above both of the other configurations. The difference in impulse of the 20mm and 32mm is very low in general and only starts to become more apparent at 3J. It is remarkable to see though, that for an energy of 2.5J, all three configurations seem to show the same impulse bit of 46μg. It suggests, that at this level of energy, the electrode gap could have almost no effect on the impulse bit, but only on the mass bit. This would make this point very interesting for investigations to improve understanding of the thruster operation. However, the behavior is yet to be confirmed in further testing.

The impulse bit trends shown in Fig.6 correspond well with the data from other researchers.[24,31] Comparison of the effective exhaust velocity, calculated from the equation $c_e = I_{bit}/m_{bit}$ show rather low values in comparison. For example, Williams et al. found a c_e of 190s at 1.85J and slightly lower impulse bit compared to the PET. The exhaust velocity was found to be higher for a short electrode gap, which also corresponds well to the literature. A linear trend for the mass bit and the energy according to the relation $m_{bit} \sim k \cdot E_0$ has been noted in literature [8]. A good correlation with this has been found regarding the mass bit data at the three different energies 3.0J, 2.5J and 2.0J, for a constant x_{gap} of 32mm. The k-factor was calculated to 17.8μg/J for this, showing an error of only 1.6% in this energy range.

The nozzle efficiency has been estimated based on a method by Burton et al.[36]. It assumes full recom-

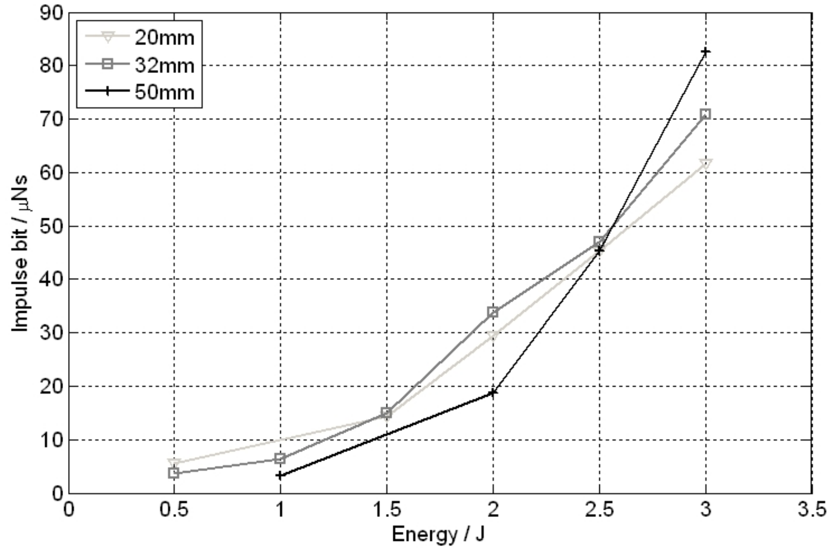


Figure 6. Impulse bit over pulse energy for variation of the electrode gap.

Table 3. Results for variation of electrode gap at 3J pulse energy.

E_0 / J	3.0		
x_{gap} / mm	20	32	50
I_{bit} / μNs	61.7	70.8	82.5
m_{bit} / μg	43.4	54.2	61.4
η_N	0.55	0.51	0.53

bination of ions and an adiabatic and equilibrium flow in the nozzle. However, the nozzle efficiency is to be considered an upper threshold to show the possible potential of an optimized PET. For more detailed information, please refer to the literature. The changes in performance of the PET due to changes of the more and more ablated cavity where tested to have had no impact on the presented results. This was done by measuring the mass bit for one PET-configuration before and after the conclusion of the tests, which where found to be identical. No significant signs of electrode erosion were found on the PET during the testing.

VI. Conclusion and Outlook

A new pulsed electrothermal thruster development at IRS has been presented. The design's simplicity and easy handling recommend it not only for possible future applications on micro- and nano-satellites, but also facilitate research efforts on thruster scaling, technology development and evaluation of numerical simulations. It has so far been the basis for development of a new HV-igniter as well as a small power supply laboratory unit (PPU), which are successfully operated at IRS. The PPU has completely been developed at IRS in cooperation with the newly founded Institute for Electric Energy Conversion (IEW) at the University of Stuttgart. It is used to operate the PET and its igniter. It will now be subject to miniaturization at IRS to have a fully operational PPU available in the near future for application on a micro-satellite. The flight of a PET is feasible, which is demonstrated by the unoptimized laboratory model. A future optimization will always have to be carried out with respect to the mission application. The PET is for the first time, a resourceful modular instrument for practical education of the next generation of electric propulsion engineers and researchers. Further, it can also be used in a wide field to help promoting space science and propulsion to the public. Presentation of the PET operation at schools or exhibitions seems within range.

A new small and mobile facility has been set up at IRS. It will allow for safe and fast testing of the PET in the future, avoiding long pumping down idle times. The facility is able to handle voltages up to 20kV if necessary. Fully independent operation by students of the facility is planned as part of the LEA student laboratory for electric propulsion. The laboratory will be mandatory for all students of electric propulsion and will close the gap between practical experience and theoretical knowledge. Preliminary results on the thruster functional and performance testing have been presented. Expectations on the thruster behavior and performance at IRS have been fully met. Although the results are in relatively good agreement with literature, a lot of work has to be carried out with respect to the thruster scaling and component development. Further investigation of the discharge is necessary to overcome limitations of the frozen flow. A possible supporting method under consideration at IRS is to further involve students by creating new research data as part of the LEA. A further increase of I_{bit} and a parameter sensitivity study of the system could be carried out in the future by means of:

- Increase of pulse rate,
- Decrease of the cavity diameter,
- Testing of compound PTFE,
- Supersonic nozzle,
- Reduction of parasitic Inductance.

The PET is yet another endeavor at IRS to allow for sustained research and development as well as improved understanding of pulsed plasma thrusters.

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