Electric Propulsion Systems for Small Satellites: The LEO Mission Perseus

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Abstract

The Institute of Space Systems, Universität Stuttgart, launched a "Small Satellite Program" in 2002. The first two of the four planed small satellites, Flying Laptop and PERSEUS, are both low earth orbit missions. The third mission Cermit is a re-entry satellite and the last of the small satellites – LUNAR MISSION Bw1 – is a mission to the moon. For this purpose, propulsion systems are mandatory. The propulsion system for LUNAR MISSION Bw1 will consist of two different types of thruster systems: a cluster of pulsed magneto-plasmadynamic thrusters (SIMP-LEX) using solid PTFE as propellant and a thermal arcjet thruster (TALOS) using gaseous ammonia as propellant. Both thruster systems are currently under development at IRS. They are going to be tested on board the small satellite mission PERSEUS, one of the precursor missions of LUNAR MISSION Bw1. The thruster systems have been investigated intensely in the past and, furthermore, optimization of the thrusters with respect to the mission requirements of LUNAR MISSION Bw1 has been started. The test procedures for the technology demonstration on the PERSEUS satellite are under development at present.

1. Introduction

Within the "Small Satellite Program" at the Institute of Space Systems (IRS), Universität Stuttgart, consisting of four small satellite missions two low-cost electric thruster systems are developed – the pulsed plasma thruster (PPT) SIMP-LEX and the thermal arcjet thruster TALOS.[1,2] These systems are going to be used as primary propulsion for the IRS small satellite mission LUNAR MISSION Bw1, which will orbit the moon in a polar orbit with an altitude of about 100 km.[3] The flight to the moon is expected to take about two years and is divided into four different mission phases:
Phase one begins after insertion into geostationary transfer orbit (GTO) and ends after the spacecraft's perigee is raised above the outer van-Allen-belt. The second flight phase lasts until the moon's sphere of influence is reached. The third phase ends with the insertion into a stable elliptical lunar capture orbit and during the final mission phase the satellite is injected into his final orbit around the moon.
The thrust demands during mission phases one and three are higher than during the other mission phases. Therefore, the arcjet thruster as the higher thrust unit will be used during these phases. During the other mission phases solely the iMPD cluster will be used for propulsion. The in-orbit validation of those thruster systems is accomplished with the second of the four small satellites, PERSEUS.

The PERSEUS satellite will have a mass of about 150 kg and provide about 1 kW electrical power. The dimensions of the cube shaped satellite will be $50 \times 70 \times 85$ cm. It is scheduled to be launched into a polar, sun-synchronous, low earth orbit below 1000 km. Its mission is planned to be divided into two phases: first the test and validation of the two propulsion systems and second UV-astronomy using a telescope and a spectrometer with a CCD sensor. The sensor will be sensitive in the spectral band of 120 nm - 180 nm. In this part of the spectral band many astronomical phenomena like supernovae can be observed by PERSEUS in a way that is not possible from earth. The UV-astronomy is going to be in cooperation with the *Institut für Astronomie und Astrophysik Tübingen*.

This paper focuses on the current state of the development of the electric thrusters SIMP-LEX and TALOS at IRS. Furthermore, it will give a short overview of the PERSEUS satellite and the test and validation procedure of the thruster systems as planned during the PERSEUS mission.

2. Pulsed Plasma Thruster

A Pulsed Plasma Thruster (PPT) is a magnetoplasmadynamic electric propulsion system. This means, it uses the electric power provided by a spacecraft to accelerate propellant by means of the electromagnetic Lorentz force. As only charged particles can be accelerated, the ablated propellant is turned into ionized plasma. To maximize the power that is provided to the plasma but keep thermal loads at a tolerable level, the plasma is created by a high current discharge pulse. As a result the thruster assembly represents an electric circuit. The parts and working principle are presented in Fig. 1.

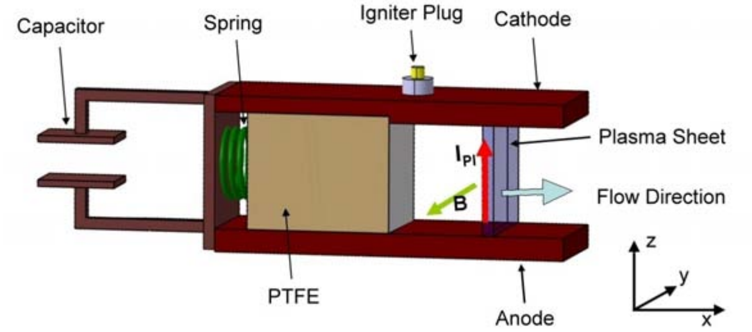


Figure 1: Working principle of a PPT

A PPT consists of four main parts: capacitor, electrodes, propellant feed system and igniter plug. The capacitor is needed to store electric energy for a discharge pulse. The two electrodes, the anode and the cathode, are connected to the capacitor terminals to carry their electric potential. A block of solid PTFE propellant is placed between the electrodes and fed by a mechanic feed system, for example as shown in Fig. 1 by a spring. At the beginning of a discharge pulse, a voltage is applied to the capacitor. A pulse is started by firing the ignition plug. The discharge of the capacitor ablates a small amounts of PTFE at the propellant surface what forms a plasma sheet. The current loop is now closed by the capacitor, the electrodes and the plasma sheet. The current through the plasma sheet creates a self induced magnetic field. Interaction of the ionized particles of the current with this field causes the electromagnetic Lorentz-force and expels the propellant particles to provide thrust.

In the past decades, numerous PPTs have successfully been operated for purposes of technology demonstration, N-S stationkeeping, high precision target pointing as well as drag compensation.[4,5] Their robust and simple design is a clear advantage in terms of safety and reliability while the pulsed operation allows for a very flexible power management.

At IRS the PPT SIMP-LEX is developed as the primary propulsion system of the LUNAR MISSION BW1 and for technology demonstration on the PERSEUS satellite. The thruster has been theoretically and experimentally investigated at IRS as part of a project funded by the Deutsches Zentrum für Luft- und Raumfahrt (DLR) in the past. Recently, extensive development efforts by Nawaz et. al. in the field of thruster optimization in combination with close cooperation with the russian partner institute RIAME MAI at Moscow have led to the introduction of the thruster ADD SIMP-LEX.[6,7] The design was optimized to improve thrust efficiency and to further allow for a closer study of the dependence of the SIMP-LEX thruster on its electrical parameters. A picture of the thruster is given in Fig. 2.

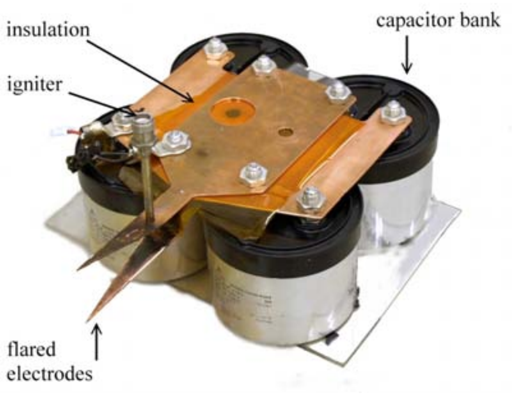


Figure 2: The thruster ADD SIMP-LEX

Currently development efforts are ongoing at IRS to characterize performance and electrical parts as well as life testing of the thruster for optimization and eventually integration of the SIMP-LEX flight model. In the next step, the propellant feeder, the electronic control and the telemetry for on-orbit testing will be integrated for subsequent acceptance testing of the PERSEUS protoflight unit. A second testbed for development of the thruster has been established at the University of Tokyo in Japan to provide data for the simulation of the discharge pulse and for plasma diagnostics.

3. Thermal Arcjet Thruster

A thermal arcjet thruster is an electric propulsion system where the propellant is heated up by an electric arc and accelerating through a Laval-nozzle. The working principle and main parts of the arcjet TALOS are shown in Fig. 3. The thruster consists of an annular anode, which is the nozzle of the thruster, and a central cathode as well as the injector, which is furthermore the electrical insulation between the electrodes. For ignition the propellant flow is opened and a high discharge voltage in the range of 2000 V is applied to cathode and anode. The ignition is accomplished via a Paschen-discharge. Once ignited, the arc between cathode and anode heats up the propellant causing dissociation

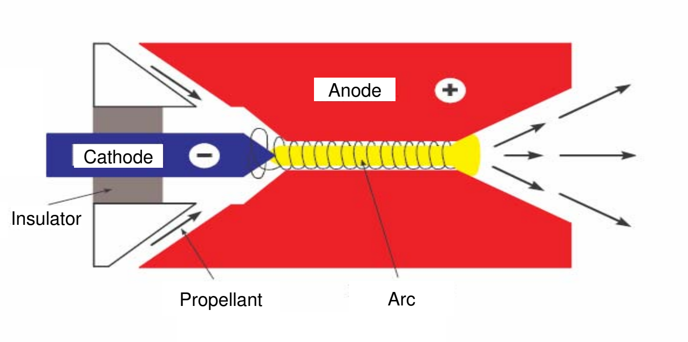
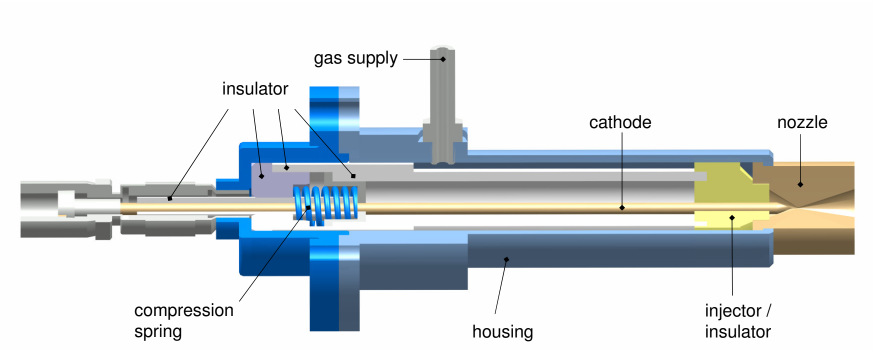


Figure 3: Working principle of thermal arcjet thruster

and ionization. The propellant is then expanded through the nozzle converting the thermal energy introduced into the propellant by heating into directed kinetic energy. Doing so, the thrust is generated.
As the exhaust velocity of a thermal arcjet is inversely proportional to the molecular weight of the propellant, light propellants are preferred to achieve a high exhaust velocity and, hence, thrust. This includes hydrogen and propellants with a high amount of hydrogen like hydrazine and ammonia. For reasons of storability, hydrogen is only used for ground testing, whereas hydrazine is used in flight applications like north-south station keeping for geostationary satellites.[9] First on-orbit experiments of a 26 kW thermal arcjet thruster using ammonia as propellant have been successfully accomplished by Bromaghim et. al..[8] For the thermal arcjet thruster of the LUNAR MISSION BW1 ammonia is going to be used as propellant as well.

Figure 4 (*left*) shows a sectional drawing of the thruster laboratory model and the thruster during operation with ammonia fig. 4 (*right*), respectively. During the development process of the thermal arcjet thruster the cathode gap, the distance between cathode and anode, can be adjusted stepless and the nozzle geometry can be changed by keeping the outer nozzle geometry constant and changing the inner geometry like the nozzle throat diameter, the nozzle throat length and the expansion angle.



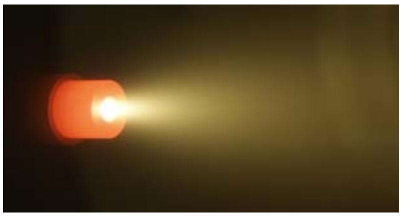


Figure 4: Sectional drawing of thruster (*left*) and thruster during operation (*right*)

The cathode is made of 2% thoriated tungsten and is 3 mm in diameter. The cathode tip is a cone with 22.5° half cone angle. During an optimization process of the thruster to meet the requirements of the LUNAR MISSION Bw1 different nozzle geometries as well as different nozzle materials are investigated. These materials are tungsten alloys with thorium oxide, lanthanum oxide and rhenium, respectively. The geometry of the nozzle is varied at the nozzle throat between 0.4 mm and 0.6 mm, the nozzle throat length between 0.4 mm and 1 mm and the expansion half angle between 19° and 20°, the converging half cone angle of all investigated nozzle geometries is 35°.[2] The thruster is radiation cooled.

In general, the performance characteristics of arcjet thrusters depend on the total energy input, mass flow rate and propellant used. For a nozzle geometry with a nozzle throat diameter of 0.6 mm, a nozzle throat length of 0.7 mm and an expansion angle of 40° the mean performance characteristics for 30 hours of operation with ammonia are as follows: 870 W electrical input power and a mean mass flow rate of 26 mg/s. The thrust level achieved is about 100 mN for this operating conditions at stationary operation. The thruster was operated as planned for the LUNAR MISSION Bw1: one hour stationary operation followed by one hour battery charge mode with no thruster operation. The nozzle material used for this investigation was tungsten-thorium oxide.

The propellant feed system of the thermal arcjet thruster supplies gaseous ammonia at constant mass flow and pressure to the thruster. The mass flow is regulated by the pressure drop over the flow aperture. Therefore, the pressure in front of the flow aperture is set by the pressure regulator.[10] As the ammonia is stored inside the propellant tank in liquid phase a heat exchanger is used to heat up and vaporize the ammonia prior to injection into the thruster. Main parts of the propellant feed system are the tank, a check valve to open and close propellant flow, a vaporizer, a pressure regulator and a flow aperture. Impurities are filtered out by two T-type filters with a pore size of 15 μm and 7 μm, respectively.

Currently, investigations under vacuum conditions for characterization of a co-current micro channel heat exchanger are conducted. The micro channel heat exchanger is developed by the *Institute of Micro Process Engineering* of the *Forschungszentrum Karlsruhe in der Helmholzgesellschaft*.[11]

4. Small Satellite Mission Perseus

The PERSEUS satellite is one of the precursor missions for LUNAR MISSION Bw1. It will be a cube shaped satellite with a mass of about 150 kg and about 1 kW electrical peak power. The dimensions of the satellite will be $50 \times 70 \times 85$ cm. It will be launched into a polar, sun-synchronous low earth orbit below 1000 km. Its mission is divided into two phases:

- the test and validation of the propulsion systems and
- UV-astronomy using a telescope and a spectrometer with a CCD sensor, which is sensitive in the spectral band of 120 nm - 180 nm.

In this part of the spectral band many astronomical phenomena like supernovae can be observed by PERSEUS in a way that is not possible from earth. The preliminary design of the small satellite PERSEUS is shown in Fig. 5. Both thrusters are located at the bottom of the satellite, the thermal arcjet thruster in the center of the bottom part and the pulsed plasma thruster at the left.

The LUNAR MISSION Bw1 will feature a cluster of at minimum four SIMP-LEX thruster units to provide the required thrust and total impulse needed to reach and orbit the moon. However, the PERSEUS satellite is only able to host one PPT thruster together with the propellant feeding system needed to store the amounts of solid PTFE inside the LUNAR MISSION Bw1 satellite. Hence, all verification tests on PERSEUS will be performed using a single SIMP-LEX unit. As for all PPT's with solid PTFE propellant, the pulse frequency of the thruster is limited due to thermal self-depolymerization effects of the PTFE, also limiting the fuel consumption rate. This constraint inevitably results in a long mission duration for the LUNAR MISSION Bw1 which extends the overall operation time of the PERSEUS thruster testbed. Nevertheless, testing of an entire SIMP-LEX life cycle of two to three years of continuous operation remains a challenging requirement and is not going to be carried out with the PERSEUS mission but inside the ground test facilities at IRS. The on-orbit test program of the SIMP-LEX flight demonstration mission is planned to be carried out in three phases: A test phase for the study of the thruster operation and health, a verification phase for assessing spacecraft compatibility and finally a phase for spacecraft maneuvering.

The flight system of the thermal arcjet thruster TALOS on PERSEUS will consist of the thruster system as developed for LUNAR MISSION Bw1 but with less propellant and thus a smaller propellant tank. For the verification of the thruster operating cycles of one hour of arcjet firing mode and one hour of battery charging mode as planned for the LUNAR MISSION Bw1 are going to be carried out. Again, full life time tests are not accomplished on the PERSEUS mission but

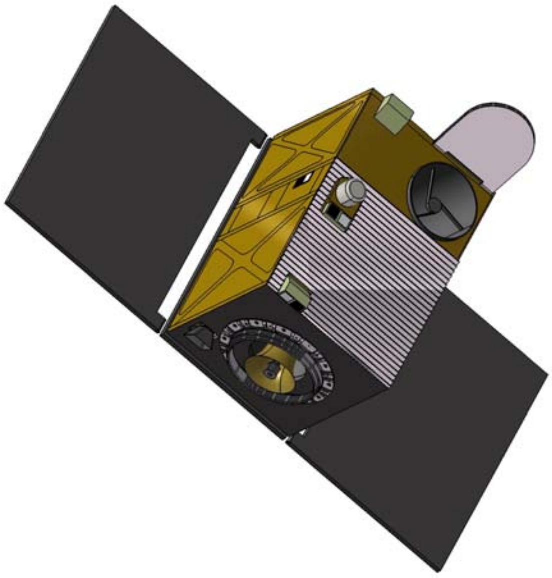


Figure 5: Perseus satellite

inside the test facilities at IRS. During the PERSEUS mission a total test time of 30 hours of arcjet thruster operation is scheduled to be carried out.

5. Future Work

The small satellite mission PERSEUS will be a technology demonstration testbed for the two electric propulsion systems, which are going to be used for LUNAR MISSION BW1, the pulsed magnetoplasmadynamic thruster SIMP-LEX and the thermal arcjet thruster TALOS. The development of these propulsion systems is accomplished at the Institute of Space Systems at present. The optimization process of both thruster systems has been started and for the thermal arcjet thruster a 30 hour subscale lifecycle test has been conducted on ground.
A detailed test program for the in-orbit verification as well as a diagnostic system to gather the necessary information on thruster operation on board the satellite PERSEUS is under development. During the verification procedure orbit maneuvers as foreseen for the LUNAR MISSION BW1 are going to be executed with both thruster systems separately. Prior to these maneuvers the check-out of the thruster systems and sensor systems is accomplished to inspect for malfunctions. After the thruster tests are completed the satellite PERSEUS is going to accomplish UV-astronomy in the spectral range of 120 nm - 180 nm in cooperation with *Institut für Astronomie und Astrophysik Tübingen*.

6. Acknowledgments

The founding of the PhD of Birk Wollenhaupt by *Carl-Zeiss-Stiftung* and Tony Schönherr by *Monbukagakusho* is greatly acknowledged.

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