Development of a Low Power Radiatively Cooled Thermal Arcjet Thruster

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Abstract

The laboratory model of a modular thermal arcjet for the 1 - 2 kW power range was designed and tested. Until now the tests had to be limited to the 1 - 1.5 kW level due to an insufficient power supply. Thus, a more sophisticated power supply is being built. It is the intention of this program, to develop a complete hydrazine arcjet system, therefore all experiments used simulated hydrazine ($N_2 + 2H_2$ mixture) as propellant. With this configuration, specific impulses of about 430 s and thrust efficiencies of 34 - 35 % were achieved with a mass flow of 40 - 55 mg/s gas mixture. To evaluate the performance changes expected in later experiments with hydrazine, an electric gas heater was positioned in the gas feed line close to the thruster. With approximately 110 W of electric power the gas fed to the thruster was preheated to 870 K, a temperature level comparable to the outlet temperature of a catalytic hydrazine decomposer. During these experiments the specific impulse increased by an average of more than 50 s, indicating that cold gas tests with simulated hydrazine have only limited validity for later hydrazine experiments.

Nomenclature

$I_{sp}$ specific impulse [s]
$F$ thrust [N]
$m$ mass flow [mg/s]
$g$ gravitational acceleration
$= 9.81 \text{ m/s}^2$
$\eta$ thrust efficiency [%]
$P_{el}$ electric power [W]
$H$ enthalpy [J]
$c_p$ specific heat [J/kgK]
$\Delta T$ temperature difference [K]
$c_e$ mean exit velocity of a nozzle [m/s]

Introduction

With the availability of sufficient electrical power for propulsion purposes recent years have seen an increased interest in the application of thermal arcjet concepts for satellite missions. While NASA and the Rocket Research Company have already achieved a flight readiness status for their 1.8 kW thermal arcjet system [1], several other laboratories around the world are developing comparable arcjet systems for the 1 to 2 kW power range. Among them are several laboratories in Japan [2], BPD together with Controspace in Italy [3] and the Institut für Raumfahrtsysteme (IRS) at the Stuttgart University in Germany in a cooperation with MBB-ERNO.

This paper will describe the results recently achieved with a low power arcjet thruster developed under a contract from the German space agency (DARA - Deutsche Agentur für Raumfahrt Angelegenheiten).

This program for the development of a 1 kW hydrazine thermal arcjet was initiated in October 1990. The two contractors involved in this project are the IRS, responsible for thruster design and diagnostic experiments and the MBB-ERNO Raumfahrttechnik GmbH, part of the Deutsche Aerospace (DASA), which is responsible for the power electronics, the necessary hydrazine decomposer and later hydrazine tests. MBB-ERNO became involved in this arcjet project because of their vast experience with hydrazine...
thrusters and their development of the PACT (power augmented catalytic thruster), a resistojet concept already under qualification for satellite missions.

Work on low power arcjets at the IRS started in 1989 when a first laboratory version of a 1 kW thermal arcjet thruster was developed (Glogowski et al. [4]). The main objective of this thruster was to evaluate the concept of a radiation cooled nozzle versus a radiatively as well as regeneratively cooled one. The regeneratively cooled nozzle showed a small increase in thrust and efficiency compared to the radiatively cooled for NH3 as propellant. However, problems with the design of the thruster rendered further experiments impractical. Namely difficulties with material processing and sealing problems prohibited more detailed investigations.

During the first year of the DARA project, a laboratory version of the thruster was performance tested with simulated hydrazine (a mixture of \( \text{N}_2 + 2\text{H}_2 \)) at the IRS to learn more about the operating characteristics of the thruster. During August an electric gas heater, developed by MBB-ERNO, was installed to heat the gas mixture to temperatures of up to 870 K to simulate the influence of a hydrazine decomposer. The second MBB-ERNO contribution to the project is the engineering version of a pulse width modulated power supply that will replace the current DC-power supply which shows an insufficient capability to operate at current levels below 12 A and above 14 A.

**Design Description**

**ARTUS 01 Thermal Arcjet**

The laboratory version (Fig. 1) of the thermal arcjet (dubbed ARTUS for Arcjet Thruster, University of Stuttgart) is derived from the laboratory version of the arcjet developed at the NASA Lewis Research Center in Cleveland.

The long cylindrical housing of the ARTUS thruster allows an easy disassembly of the thruster to exchange components like the nozzle or the injector disk. A flange type nozzle was chosen to accommodate the use of different nozzle inserts without lengthy adaptations of tapered fits.

Compared to the NASA design two major changes were incorporated. First, the length of the thruster was reduced because the propellant feed line could be connected directly to the thruster housing, the feed line did not have to be isolated electrically from the anode potential.

The second, more important modification was the mechanism to adjust the gap between anode nozzle and cathode tip. While it needs some experience to adjust the NASA design with a
swage-lok type seal within the required accuracy, the ARTUS design uses a micrometer type assembly. The cathode gap can now easily be adjusted with an accuracy of 0.02 mm. Furthermore, it can be rechecked for changes after each experiment. The criteria for a closed cathode gap is the electric resistance between cathode and anode measured with an ohmmeter. Once contact has been established, eccentricities between anode and cathode sometimes allow the cathode to be moved up to another 0.07 mm inwards. However, no precise reproducible criteria for the gap could be established using only mechanical contact between anode and cathode. Sealing along the cathode feed through is provided by a gland seal squeezed by a gland stud.

Thoriated tungsten (2% ThO$_2$) was used as material for the anode nozzle insert, due to the high temperatures expected at the nozzle throat and to reduce erosion near the throat edges. The nozzle was specified with a 0.60 mm throat diameter, a 0.25 mm throat length and an exit plane diameter of 12.00 mm (expansion ratio 1:400). These specifications were kept within 0.02 mm. The same tungsten material was used for the cathode. Cathode diameter is 3 mm, the cathode tip angle measures 20° - 30°.

TZM (a molybdenum based alloy) was chosen for the thruster housing. TZM withstands high temperatures (2850 K), but can be machined easier than tungsten. The injector disk is also made of TZM. Its two injection boreholes provide the usual tangential propellant injection as well as a small forward component obtained by a 20° forward tilt.

The main insulator, the rear flange and the insulation cap, the components providing electrical insulation, are manufactured from boron nitride. The remaining parts like propellant feed line and cathode adjustment mechanism are manufactured from stainless steel, because their temperature loads were estimated to remain below 920 K. The spiral spring that presses the main insulator and the nozzle against the forward housing flange to provide the sealing between nozzle and housing is manufactured from a high temperature Inconel alloy. All seals were punched from 0.35 mm thick graphite foil.

The assembled thruster has a mass of 1.3 kg, mainly due to its massive housing, dictated by the need for an interchangeable nozzle and injector disk. Thus, future design changes for the flight orientated thruster will be able to reduce the thruster weight below 1 kg.

**Hot Gas Source**

Initial experiments with the arcjet have utilized room temperature N$_2$+2H$_2$ mixtures as a substitute for the products of a hydrazine decomposer. However, to simulate the decomposer more realistically a warm gas source has been manufactured to heat the gas mixture to the temperature of that emanating from a decomposer.

The warm gas source has been tested at varying flow rates between 30 to 50 mg/s nitrogen and can maintain a temperature of 870 K using 120 W heater power.

The hydrazine decomposer that will be used for later real hydrazine experiments is already under development at MBB-ERNO. A preliminary version of it is shown in Fig. 3. It is based on a proven design for the ERNO power augmented catalytic thruster (PACT).

**Arcjet Power Electronics (APE)**

To overcome the problems with the limited operating range of the current power supply, MBB-ERNO developed a new arcjet power processing unit for the thruster system. Power is supplied to the thruster from a pulse width modulated power processing unit with constant current regulation. The unit is capable of delivering up to 20 A and 135 V to a resistive load or an arcjet. The input voltage is limited to 80 V DC.
Starting the arcjet is accomplished by an ignition circuit delivering an ignition voltage of up to 4 kV, adjustable by an external control voltage.

The overall block diagram of the APE is shown in Fig. 4. It is designed as a four transistor bridge switching circuit, a pulse width modulated control including phase shift control of the power bridge and an output current averaging inductor containing the high voltage winding to entrain the ignition pulse.

On command the ignition circuit is started with repeated pulses until the arc is successfully ignited. An internal current start function is started until the current has reached its preset value. The current is kept constant by the pulse width modulation, its value can be changed by manual control. The efficiency of the current switching circuit, a pulse width modulated control bread board model type version is typical 90% at 1.8 kW output power.

The APE was successfully tested on a water cooled helium arcjet and will be tested
subsequently on the ARTUS thruster to study further the effectiveness of the APE concept.

Fig. 4: Blockdiagram Arcjet Power Electronics

Experimental Facilities

All experiments described in this paper were conducted in a 2.0 m long vacuum chamber with 1.0 m diameter. The experiments described by Glogowski [4] were done in the same chamber equipped with two groups of pumps, allowing a back pressure of 35 Pa at a mass flow rate of 30 mg/s ammonia. This pumping capacity was found to be insufficient for thorough performance testing of the new arcjet system, especially having the comparative tests with and without the hot gas source in mind. Thus, the tank was equipped with a new three stage pump group including one rotary vane and two roots pumps. These pumps are now capable of sustaining a vacuum of 2.0 Pa at a mass flow of 50 mg/s simulated hydrazine, an improvement by a factor of 15 to 20 compared to the old system.

Comparison between early tests with the ARTUS thruster while the old pumping system was still mounted showed an improvement in thrust measurements directly coupled to the lower ambient pressure in the tank. Under otherwise identical performance parameters the thrust was increased by the difference in tank pressure multiplied by the exit plane area, a thrust increase of about 0.007 N. This result corresponds to the more detailed investigations of the ambient pressure influence on thrust measurements by Glocker et al. [5].

In contrast to their measurements, a small increase in arc voltage of about 5 V was observed with the new pumping system. This indicates that the arc length in the ARTUS nozzle (expansion ratio 1:400) is influenced by the boundary layer in the nozzle. If the boundary layer is forced upstream by a higher background pressure, the arc is also forced to attach further upstream in the nozzle. This results in a shorter arc with a lower arc voltage.

The PC-type data collection and reduction for the performance measurements was the same as described in [4]. The software for data collection was modified to improve data storing reliability and software handling characteristics. The performance parameter collected include:
- arc voltage
- arc current
- thrust
- mass flow
- ambient tank pressure
- feed line pressure (to monitor changes in thruster behavior)

with the hot gas source operating:
- hot gas source outlet temperature (measured with a thermocouple)
- hot gas source input voltage

The remaining data were used to monitor the facility status. During later experiments the software incorporated the possibility to activate an emergency shut down option that allowed sustained experiment operation without
supervision. Tank ambient pressure, mass flow and arc voltage were used to monitor the status. In the event of threshold violations, the facility was brought into a safe mode by switching off the power supply, terminating the gas supply with a delay of 30 seconds to cool the thruster down with the gas flow and - depending on the cause for the deactivation - a shut down of the pumping system. These monitoring functions have to be seen in the context of planned life time tests and the necessity to establish a data base determining the "thruster health" for a possible satellite application.

The thrust measurement in this facility is done with a pendulum type thrust balance, the pendulum displacement being sensed with a non-contacting linear displacement transducer (NCDT). This pendulum could measure the thrust with an uncertainty of about 0.002 N after being calibrated with two weights of about 0.1 N each.

The hot gas source was placed at the lower end of the pendulum, parallel to the thruster with a horizontal clearance of about 0.25 m, this location being not an ideal one. However, the hot gas source had to be placed as close to the thruster as possible to minimize heat losses in the feed line. Experiments showed that thermal radiation from the thruster did not interfere with the heater, even during extended operation. The additional heater mass (0.6 kg) at the pendulum end increased the thrust balance uncertainty to about 0.004 N. During experiments with the gas heater in operation the temperature of the gas at the end of the heater and the electrical power used by the heater were monitored to prevent damage to the heating element of the hot gas source.

Operating Experience and Results

Design Verification

After the completion of the prototype thruster in February 1991 the first tests were aimed at verifying the principal thruster design. The first tests were hampered by leakage problems at the front end of the thruster. A reduction in gasket dimensions together with a change in material for the spiral spring increased the pressure on the front end seal, thus eliminating this problem.
Another difficulty was posed by the boron nitride material. Although the highest available quality was used for the main insulator, the disassembly of the thruster after an experiment was sometimes difficult or even impossible without breaking the insulator. The boron nitride baked together with the TZM housing, probably because the boron nitride still contained a high water portion. Additionally, boron nitride forms an eutectic with molybdenum that would prevent disassembly. Subsequently the design was altered to either prevent direct contact between boron nitride and hot metal without a layer of Grafoil between the parts or by inserting a small tungsten tube between the cathode and the insulator. The cathode tip is still centered by specifying small tolerances for the machining of the parts in question.

Another change required during the first tests was the installation of an insulation cap at the thruster end. The cap became necessary to prevent arcing between the cathode and the washer ring when the old pumping system was not able to sustain low back pressures.

Except for rerouting the propellant feed line at the lower pendulum end when the hot gas source was mounted, no changes had to be incorporated to the thrust stand. When the new pumping system was installed, the thrust balance was thoroughly inspected, but no damage could be detected after 24 months of operation.

### Starting Reliability

The temporary power supply used by Glogowski et al. [4] was used for the performance testing of the thruster reported in this paper. It consists of a conventional, current regulated power supply rated at 160 V and 15 A. A ballast resistor has to be switched in series with the arcjet thruster. The ignition pulse is provided by a high voltage power supply, capable of a 1.6 kV ignition pulse. The pulse is superimposed on the circuit by a high voltage relay, the regular power supply is protected by high speed diodes from this peak. The relay has a 2 ms minimum switching time. Thus, the ignition pulse could not be optimized for the present thruster. Other investigations showed [6], that the ignition voltage is within the necessary range, however, the pulse duration seems to be too long. Therefore the energy delivered in the pulse is too high, probably causing high anode erosion during the start-up phase. This became evident when severe sparking was observed during some experiments. Furthermore, the throat diameter of the nozzle used in the first tests increased from 0.58 mm to 0.75 mm. The throat length was designed with 0.25 mm, probably too short for this destructive ignition process.

A more protective starting procedure is incorporated in the MBB-ERNO power supply, which will enable a variable setting of the pulse length to optimize the start pulse for different mass flow settings and propellants.

Usually, the thruster was successfully ignited with the first ignition pulse when the thruster was cold and the ballast resistor in the power circuit was set to an appropriate level. However, the thruster start became more difficult when it was still hot after a previous experiment or when the hot gas source was operating and delivered the gas with higher temperatures to the thruster. In this case several pulses in connection with a variation of the resistor setting and lower mass flow values were necessary to activate a stable arc.

### Thruster Performance

The first tests with the thruster established a performance data basis for the ARTUS laboratory model design. The results of this tests are listed in table 1 and Fig. 6 to 8. The measured results correspond in principle with data published by other laboratories [3,7]. The tested mass flows include the 40 to 55 mg/s range. This range was chosen with the thruster power in mind. The upper mass flow limit was planned to be about 70 mg/s, however tests with more than 55 mg/s were not possible with the current power supply. The arc voltage added to the voltage loss across the ballast resistor exceeded the 160 V limit of the power supply in these experiments.

Additionally, this 160 V limit already prohibited some current / mass flow settings in the tested range. The second limitation during the first performance evaluation was the current capability of the power supply. Tests with currents higher than 14.5 A were impossible. The lower current

7/12
limit was near 11-12 A, here the power supply control circuit could not cope with the large negative slope of the current - voltage characteristic of the arcjet; and the arc did not burn stable for more than ten seconds. This insufficiency could not be eliminated with the ballast resistor because the voltage drop across the resistor again exceeded the 160 V maximum.

Consequently, the tested range included mass flow settings of 40 - 55 mg/s and current settings from 12 to 14 A. The possible power settings were limited to 1.1 - 1.4 kW, additional testing will be necessary to reach the scheduled 2 kW level. This will be possible once the more sophisticated power electronics supplied by MBB-ERNO are fully operational.

**Hot Gas Source**

During the tests with the hot gas source the heater element was powered by a constant voltage power supply. Input voltage and gas temperature near the outlet of the heater were monitored to control the heater status. The heater element limited the gas temperature to about 870 K. During the tests it was kept between 850 and 880 K by varying the input voltage manually. The time constant of approximately 15 minutes for the gas heater to reach thermal equilibrium mandated the 30 K temperature span (3.5% of the 870 K) and prevented a more accurate and automated regulation during these experiments.

With a resistance of about 60 Ω the mentioned condition was achieved with about 75 to 80 V and a current of some 1.3 A. The electrical power of about 110 W fed to the heater was not considered when the thrust efficiency for the hot gas tests was calculated. This is justified, because with the use of a hydrazine decomposer the gas would enter the thruster with a temperature of about 1000 K without any influence on the electrical power budget. With the equation

$$\dot{H} = c_p \cdot \dot{m} \cdot \Delta T$$

(No. 3)

the energy needed to preheat the hydrogen portion of 6.25 mg/s to 870 K is calculated to be 51.5 W, for 43.8 mg/s nitrogen 26.4 W are needed for the heating. Thus, the efficiency of the hot gas source is calculated to be 71 %. With the equations for an ideal thermal rocket engine, the 77.9 W of the preheater would yield an additional 0.073 N of thrust (equivalent to 148 s of specific impulse, all data for 50 mg/s gas mixture).
\[ C_o = \sqrt{\frac{2 \Delta H}{m}} \] (eq. 4)

When the gas heater was tested for the first time, the thruster performance was tested without the arc operating. During thirty minutes of testing the thrust increased from 0.042 N to about 0.085 N at a mass flow of 50 mg/s.

The problems with the power supply - especially the need for the potentiometer to adjust the power circuit load (see starting reliability) - prevented studies of the starting reliability with the hot gas flow. The power supply reduced the useful range for performance testing compared to the "cold-gas" tests, because the higher arc voltage in connection with the voltage drop across the ballast resistor reached the 160 V limit already at current - mass flow combinations where a stable operation was still possible during the "cold-gas" tests.

The performance results with the hot gas source operating are presented in Table 1 and Figures 9 to 11. The measured thrust increased considerably compared to the "cold-gas" experiments. It is quite simple to explain this performance increase because the gas enters the thruster already with a higher enthalpy and this enthalpy is recovered during the nozzle expansion. Whether the warm gas also changes the arc attachment and the arc formation has to be subject to further investigations.

Furthermore, the warm gas tests indicate that it is not possible to draw direct conclusions from cold-gas simulated hydrazine tests to actual hydrazine thruster performance, as was shown by Hardy and Curran [7] in their study of different hydrogen - nitrogen mixtures. Cold-gas tests should only be used to evaluate the basic performance of a thruster design (the influence of nozzle geometry effects etc.). However, they cannot substitute real hydrazine tests to characterize the performance of a complete thruster system.

In addition, the results show once more the problems with the efficiency definition. The efficiencies listed in Table 1 for the warm-gas tests should always be judged with equation 2 in mind. The listed efficiency should not be seen as an overall thruster efficiency, because it does not consider any energy already available to the propellant when it enters the thruster. It is obvious that theoretically efficiencies over 100% might be possible with this definition, because equation 2 calculates only the ratio of total thrust power to electric arc power. Thus, the efficiency increase measured with the hot gas source is only a virtual increase.

![Fig. 9: Voltage Current Characteristic, Gas Heated to 870 K](image)

![Fig. 10: Specific Impulse versus Specific Energy, Gas Preheated to 870 K](image)
91-042

Radio satellite Amsat P3-D. This OSCAR satellite will be launched in 1995 on the second flight of an ARIANE 5 launcher. A letter of intent between the IRS and the satellite operators states the application of the arcjet on this satellite if certain performance data are guaranteed by the thruster system. The power available on this satellite would limit the thruster to 700 to 800 W of electric power. Thus, this application would need a tailored power supply for this power range. To reduce the risk for the ground crew, hydrazine was not accepted as a propellant. Instead, ammonia will be used for this mission. The thruster will be used as an additional attitude and orbit control of the satellite. The satellite will use the arcjet system not as its primary attitude control system due to the thruster’s experimental status. However, the thruster could support the mission by increasing the possible velocity increment for orbit raising. The primary mission of the satellite, to serve as a telecommunication relay for radio amateurs, would support the application of the arcjet system as a payload despite any problems that could originate during thruster operation. To monitor the thruster status, a small onboard computer will register and regulate thruster current and voltage, the thrust level will be determined by observing the satellite orbit and its changes.

Flight Opportunities

At the moment there are two flight opportunities envisaged for this arcjet system. The first one is the application of the arcjet system as a payload for the EURECA III mission, planned for 1997. The proposal for this mission [8] was submitted to a call for proposals and ideas by ESA and was judged positively in a first evaluation by ESA. It calls for the application of a complete arcjet system including a battery pack, power supply, two redundant thrusters and a diagnostic package with 55 kg of hydrazine propellant to the six-month EURECA mission (Fig. 12). A total of 800 hours operation time on 1 kW electrical power, divided into at least 1000 firings of the two thrusters is planned. The diagnostic package on this mission would include telemetry experiments to determine any EMI problems that might occur during the thruster operation, a CCD-camera to monitor the thruster plume behavior and thruster temperature. Furthermore, the experiment would use a small computer to register the thruster status by monitoring mass flow, arc current and arc voltage. The great advantage of this experiment would be the possibility to examine the thruster after the six-month mission because the EURECA carrier will be retrieved and brought back with its launch vehicle, the Space Shuttle. Currently, the proposal for this experiment is evaluated by ESA in detail and a decision is expected by the end of 1991.

The second flight opportunity would be the application of the arcjet thruster on the amateur radio satellite Amsat P3-D. This OSCAR satellite will be launched in 1995 on the second flight of an ARIANE 5 launcher. A letter of intent between the IRS and the satellite operators states the application of the arcjet on this satellite if certain performance data are guaranteed by the thruster system. The power available on this satellite would limit the thruster to 700 to 800 W of electric power. Thus, this application would need a tailored power supply for this power range. To reduce the risk for the ground crew, hydrazine was not accepted as a propellant. Instead, ammonia will be used for this mission. The thruster will be used as an additional attitude and orbit control of the satellite. The satellite will use the arcjet system not as its primary attitude control system due to the thruster’s experimental status. However, the thruster could support the mission by increasing the possible velocity increment for orbit raising. The primary mission of the satellite, to serve as a telecommunication relay for radio amateurs, would support the detection of any EMI problems that could originate during thruster operation. To monitor the thruster status, a small onboard computer will register and regulate thruster current and voltage, the thrust level will be determined by observing the satellite orbit and its changes.

Conclusions

The current ARTUS thruster design allows sustained operations in the 1 - 1.5 kW range with simulated hydrazine as propellant. During the tests valuable information was gained on the operation and performance of this thruster type. Especially the need for a variable power supply capable of adjustments for the different arcjet operation regimes became obvious. Nevertheless, the thruster was tested extensively and showed thrust and efficiency performance as known from other arcjet systems tested in the same power range in other laboratories.

The implementation of a hot gas source to simulate better the influence of a hydrazine decomposer clearly showed that cold gas experiments cannot be used to describe the performance of the thruster when real hydrazine is tested. A performance increase should be expected for hydrazine tests because a significant amount of energy is already stored in the gas due
Fig. 12: Experiment Arrangement for the Proposed EURECA Arcjet Test Mission

to the high temperature generated by the catalytic dissociation processes.

Acknowledgement

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References


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* with respect to arc power

Table 1: ARTUS Thruster Performance (cathode gap kept constant at 0.6 mm)