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Arcjet Development for Amateur Radio Satellite

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The radio amateur satellite P3-D, weighing 400 kg, will be launched into a highly elliptical 16-h orbit in 1996. Its final orbit control will be performed by a 750-W ammonia arcjet system. This work describes the arcjet system and its components as well as the qualification and operation philosophy. The implications of the satellite, the planned orbit, and the orbit control strategy for the arcjet propulsion system are also explained. The thruster system has passed two lifetime qualification tests lasting 670 and 1010 cycles with one hour of arcjet operation per cycle, which are briefly discussed.

Introduction

ARCJET systems were investigated as early as the 1960s and showed promising performance because of their high specific impulses compared to chemical thrusters with storable propellant. Nevertheless, the lack of electrical power available on satellites prevented their application until the late 1980s. Now the improved performance of solar arrays and their steadily increasing size allow advantageous arcjet application for certain orbit and attitude control purposes. The first arcjet application on an operational satellite is for the north–south stationkeeping hydrazine arcjet thrusters[1] on the GE-Astro Telstar 4 satellite. It uses a dual-mode hydrazine system, i.e., the hydrazine is used for orbit control as well as for attitude thrusters.

Two European proposals[2,3] called for arcjet flight experiments on the EURECA platform in 1996–1997 that would allow the postflight inspection of the thruster systems because of the retrieval of the platform. However, these projects were put on hold when future reflights of the EURECA platform were postponed. Consequently, the offer from the German branch of the amateur radio satellite organization (AMSAT-DL) to the Institut für Raumfahrtsysteme (IRS—Space Systems Institute) at Stuttgart University to fly an arcjet system on the P3-D amateur radio satellite was welcomed to prove the advanced development status of arcjet technology in Europe.

The Arcjet Thruster on Oscar Satellite (ATOS) project will be the first application for the control of a highly inclined, highly elliptical satellite orbit. In general, it may not be compared directly with commercial arcjet applications for geostationary communication satellites. However, the amateur radio satellite operators' philosophy and approach towards experimental technology makes it an ideal application to demonstrate the flight readiness of low-power arcjet technology. Future mobile-communication satellites are likely to be placed on highly inclined, highly elliptical orbits. An arcjet thruster for the purpose of orbit control, in view of its high thrust compared to electrostatic thrusters and high specific impulse compared to conventional chemical thrusters, appears to be a good choice.[4] Thus, P3-D may also be considered a prototype application of future satellite systems and hence for arcjet thruster applications.[5] In contrast with the ELITE arcjet flight experiment,[6] which will provide a demonstration of a 26-kW ammonia arcjet with 15 h in orbit, ATOS will explore the semioperational application of low-power ammonia arcjets on a satellite typical of the new generation of small satellites.

AMSAT P3-D Satellite

All AMSAT satellites are designed and built by ham radio operators on a volunteer basis and with support from universities and industry; most participants have a professional background in radio communication or space technology.[7] By the mid-seventies, the communications performance of these Orbiting Satellites Carrying Amateur Radio (OSCAR) satellites was mostly limited by the characteristics of their low orbits. Thus, AMSAT introduced the new phase three (P3) satellite family. They are placed in highly elliptical Molniya-type orbits to optimize their communication coverage. The P3 satellites use bipropellant thrusters to change the geo-transfer orbit (GTO) into which they are launched to the desired orbit, their launch weight being some 75 kg. A total of three P3 satellites have been built so far by AMSAT and have been launched with ARIANE rockets.

Based on the good flight experience with the P3 satellites, the new P3-D class satellite (design series D) in the 400-kg range will extend the communication performance even further. This spacecraft could replace OSCAR 13, launched in 1988, whose orbit is decaying, and which is expected to re-enter the atmosphere in late 1996. The P3-D satellite concept again introduces novel technologies, and its bus structure could well become useful for cost-effective commercial mobile communications of the future[8] or a class of low-cost technology satellites. The satellite is under development as a joint project of the German and U.S.–AMSAT branches[9] with amateur radio payloads from other countries (e.g., Japan and Hungary), and is considered another milestone in size and operational capability for the AMSAT organization. It is planned to be launched as a piggyback payload on the second ARIANE 5 prototype launch (A 502); however, the overall design of the launcher interface would also allow a launch on an ARIANE 4 without modifications to the satellite structure. Although, in view of the previous OSCAR experience, the payload lifetime is expected to be some 8–10 years for most payloads, the onboard propellant will allow active control of the orbit for only about 5 years. After that period, the satellite will slowly drift into a less useful orbit.

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Satellite Bus Structure and Orbit

The outer dimensions of the satellite were prescribed by the ARIANE payload fairing. It will have a hexagonal main structure (Fig. 1) with 2.3-m diameter, a height of 0.75 m, and a total mass (with full tanks) of 400 kg. Its solar array will initially produce 650 W of electrical power. The hexagonal shape was chosen for thermal design reasons to reduce the effort necessary to cool the ten transponders, the main amateur radio payload. Figure 2 shows the integration of the satellite with its payload adapter on top of the ARIANE 5 launcher between the third rocket stage and the main payload of mission A 502.

The P3-D satellite will be three-axis stabilized to optimize the communications performance and to assure sufficient power for payload and arcjet operation. The satellite attitude is controlled with flywheels and magnetic torque coils to orient the spacecraft in the intended direction. Excessive angular momentum of the flywheels will be dumped with the help of the magnetic coils during the perigee passage of the satellite.

To reduce spacecraft complexity, the solar panels are body-mounted to the satellite, i.e., the two wing panels cannot be moved relative to the main structure. A numerical simulation showed that the power loss due to the fixed solar arrays is not more than 19%, after temperature correction, with respect to movable arrays pointed towards the sun. This loss was compensated by a size increase of 20–25%.

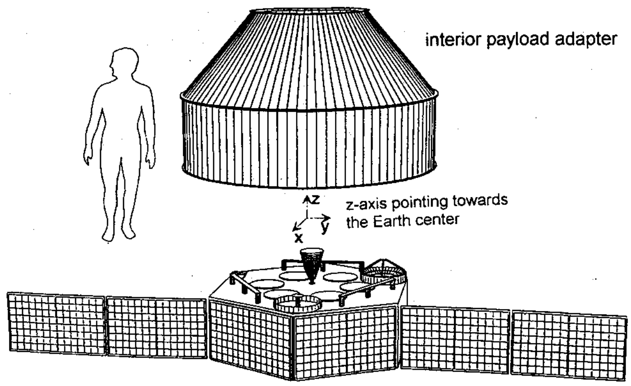

Fig. 1 Present P3-D satellite configuration with payload adapter for ARIANE 502 launch.

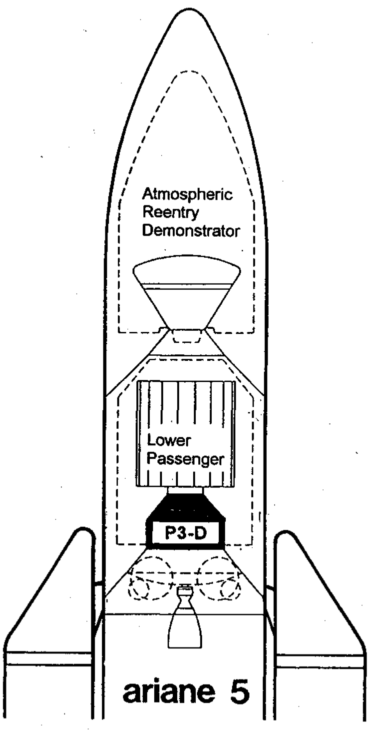

Fig. 2 P3-D integration on ARIANE 5.

Table 1 P3-D primary user time zones

Europe	UTC + 1 h
Americas	UTC − 6 − 9 h
Far East and Australia	UTC + 10 h

Table 2 P3-D orbit parameters

Period	960 min = 16 h
Argument of the perigee	225–270 deg
Eccentricity	0.644–0.723
Perigee height	2500–5000 km
Mean anomaly	0 deg at epoch 0900 h UTC

The P3-D orbit evolution (Fig. 3) has to be discussed in detail to understand the arcjet duty-cycle requirements. The optimal orbit[10] is one that offers the ham radio users extended satellite visibility during local evening hours and in the early morning hours. Obviously, this problem has no general solution, because at any time there is local evening somewhere. One good solution is found by noting that on Earth there are three main population centers in the northern hemisphere, occupying three zones almost equidistantly separated in time (Table 1).

If the local times 0500–0800 h and 1800–2400 h are defined as prime time and plotted for the three zones, a plot of the superposition of prime times gives an indication of relative importance of satellite availability on the UTC scale (Fig. 4). The result shows that there are three daily peaks and three daily zeros, which form almost a sine wave with an 8-h period. This suggests orbits that have perigees during the zeros of this function and apogees during the peaks. The least complicated orbit meeting this requirement has an 8-h period. Unfortunately, this orbit is not desirable from a radiation-environment and therefore altitude point of view.

Another orbit meeting this constraint has a period of 16 h. This orbit results in a periodic pattern where each perigee is aligned with every other zero of the availability function. In a highly elliptical 16-h orbit the duration of high altitude is at least 10 h; thus two adjacent peaks of the prime function can be covered with one orbit. The resulting pattern repeats every two days (three orbits in two days). Furthermore, consecutive apogee locations are separated by 120 deg in longitude, which is ideal from a total-coverage point of view. An orbit with the parameters listed in Table 2 was simulated and found to be superior to 12-h orbits investigated earlier.

Further studies of this orbit—fine-tuning the apogee, eccentricity, and epoch time–offer opportunities to optimize the southern hemisphere coverage. The final result is a perigee of 4800 km and an apogee of 47,000 km, both measured from the Earth surface. The orbit inclination will be 63.4 deg with a 225-deg argument of the perigee (see Fig. 3). The argument of the perigee will hardly change, because interference from the geoid shape, which would otherwise slowly shift this parameter, cancels out at the 63.4-deg inclination.

The 16-h orbit has also some advantages for the spacecraft design: 1) the satellite spends less time in the radiation environment than in orbits with shorter periods, and 2) to achieve the 16-h orbit takes a velocity increment Δv that is 200 m/s less than to achieve a similar 12-h orbit.

The satellite will be placed in a geosynchronous transfer orbit (labelled *A* in Fig. 3) by the ARIANE 5 upper stage. A series of thruster firings (I–III) with a 400-N thruster will bring the satellite into orbit *D*, which has the required perigee and apogee, but an inclination of only 60 deg. This 400-N thruster is a leftover but space-qualified spare thruster from a previous MBB (now Daimler-Benz Aerospace—DASA) satellite project, operating on nitrogen tetroxide (N_2O_4) and aerozine AZ50 [1:1 N_2H_4 and N_2H_2-$(CH_3)_2$, i.e., UDMH mixture]. Approximately 180 kg of oxidizer and fuel are available for these maneuvers; they are planned to be completed two weeks after the launch.

The final orbit will then be reached in a combination of arcjet operations during the nodal passage to change the inclination and the drift process, which gradually changes the argument of the perigee from 180 deg to the desired 225 deg. This maneuver is scheduled to be completed after one year. Then, the arcjet will be used to control

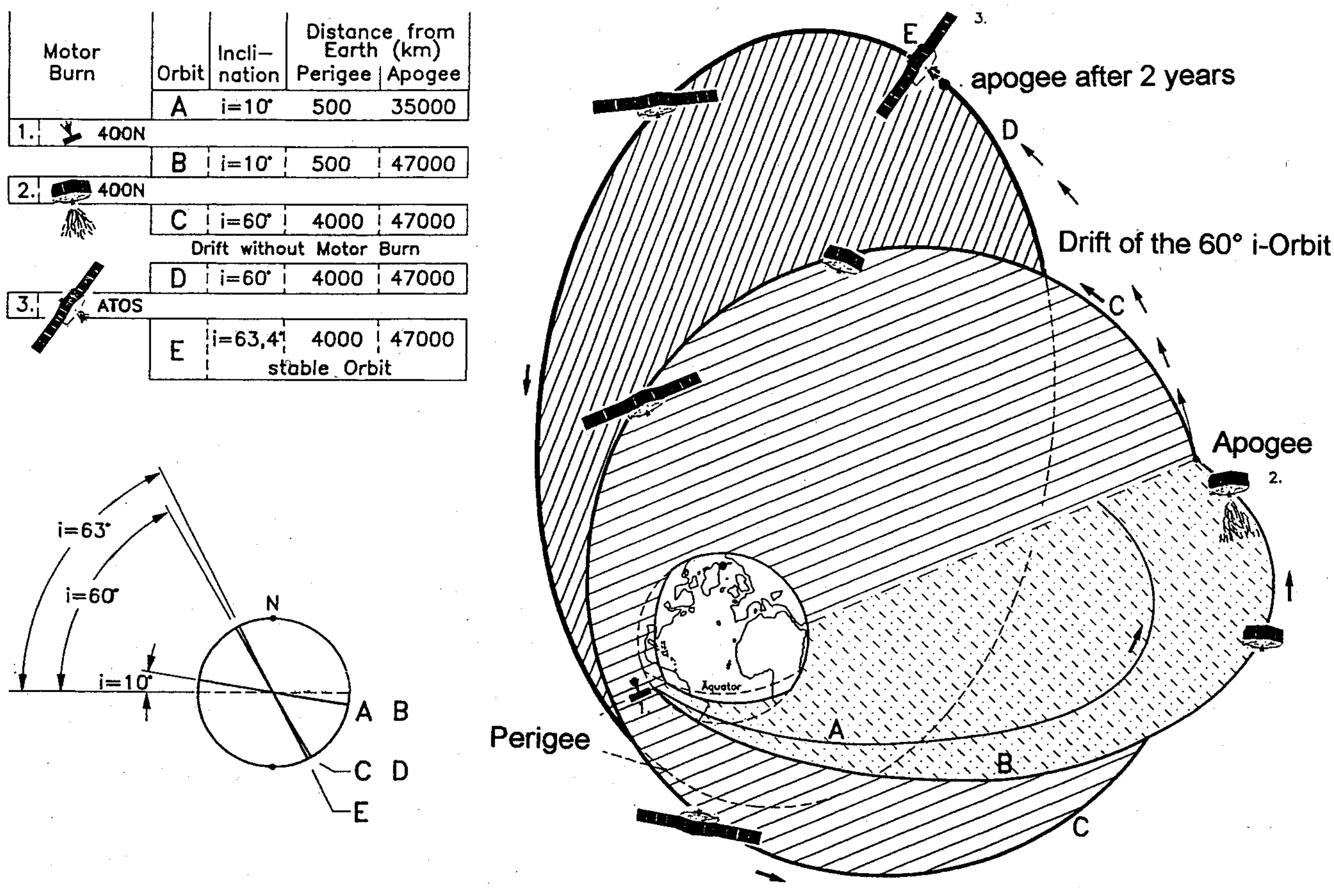

Fig. 3 P3-D orbit evolution.

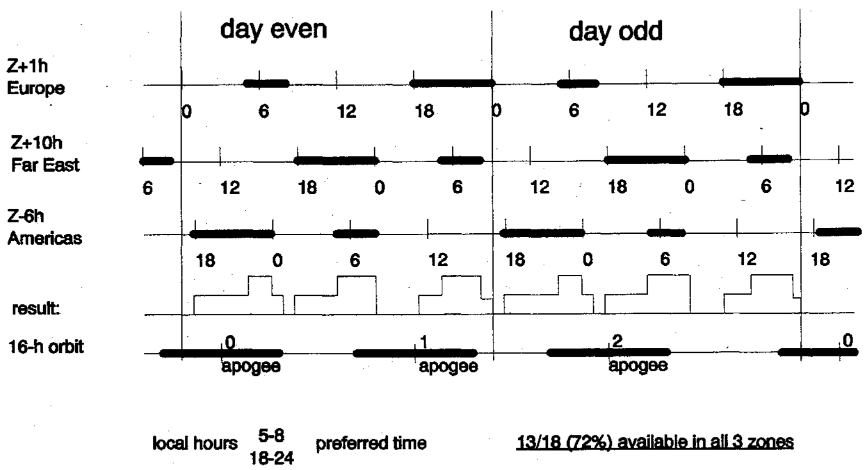

Fig. 4 Satellite visibility vs prime time for the three main P3-D user time zones.

the apogee and perigee to guarantee the 16-h orbit period (orbits E and F, thruster firing IV) until its fuel is spent. The arcjet will not be used for attitude control; as mentioned above, this will be provided by flywheels and magnetic torque coils.

Arcjet System

Propellant System

Although the 400-N thruster for the main orbit insertion uses already hydrazine (AZ50) as fuel, our choice for the arcjet propellant was ammonia (NH_3) to simplify the thruster development. The qualification tests for this project are conducted in university facilities where hydrazine tests are prohibited. Additionally, a hydrazine decomposer would not have been available. Although a 1.8-kW hydrazine arcjet system under development requires a hydrazine decomposer,[11] it could not be utilized for this project, because it is not suitable for the low mass-flow levels of the ATOS project. Furthermore, the additional decomposer development and operational requirements would make the arcjet system too complicated for AMSAT's design philosophy.

The arcjet propulsion unit of the P3-D satellite will accommodate two tanks holding a total of 90 liters of ammonia; at the maximum ground handling temperature (313 K = 40°C, a safety limit prescribed by the Arianespace ground handling specification) this will be equivalent to 52 kg. During its orbit, thermal control with surface coatings and heat pipes will keep the temperature inside the satellite at approximately 10°C (283 K); thus the tank pressure will be above 0.6 MPa (= 6 bar) because of the ammonia phase conditions.

The propellant-system layout with its components is shown schematically in Fig. 5. Additional thermal and flow analysis will show whether the tanks have to be equipped with electrical heating elements to assure a feed-line pressure above 0.5 MPa under all satellite temperature conditions. This pressure will be necessary to

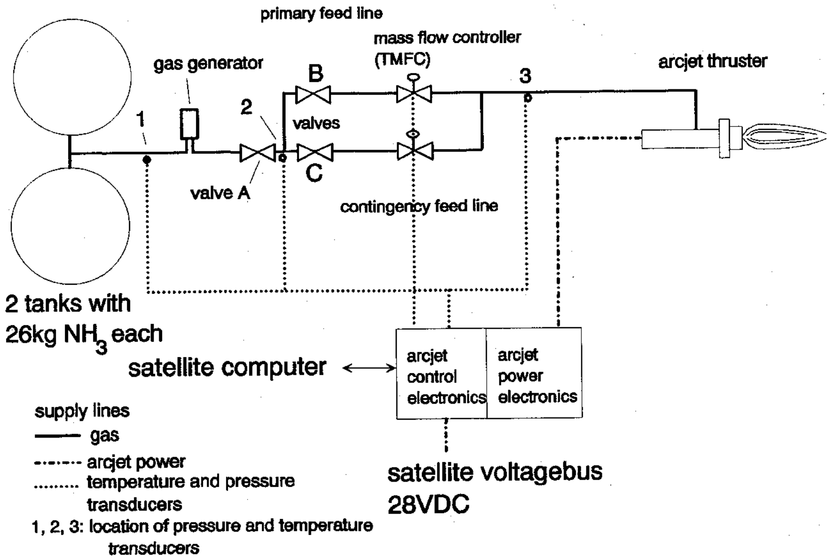

Fig. 5 ATOS system layout.

guarantee a pressure of 0.3 MPa at the arcjet inlet. The ammonia gas generator or vaporizer prevents liquid ammonia from reaching the mass-flow controllers (MFCs). Although ammonia MFCs of this type are not yet space-qualified, commercially available MFCs (measurement principle based on the thermal conductivity of the gas) will be used, because the variable tank pressure complicates the use of a simple choking orifice to control the mass flow.

The major concern in the use of the MFCs is their stability and performance repeatability over a long time period. Thus, MFC performance will be checked with the arcjet voltage–current characteristic during thruster operation. Should these data suggest a problem with the MFC, the use of the contingency feed line with a backup MFC will provide flow regulation to the thruster.

A previously considered backup thruster was ruled out because of the overall weight limitations of the P3-D satellite, as ESA did not settle early enough on the type and the mass for the primary payload for the launch. Additionally, the results from the 1010-h arcjet lifetime test provided confidence in the arcjet system. Nevertheless, for future projects we still advocate a redundant thruster; if mission success relies on the arcjet operation, the second thruster should reduce the risk of system failure, although it adds to the system complexity. This added complexity would be due only to two additional fuel valves and a duplicated power supply. The second thruster, however, would be positioned less favorably, because the satellite has to be canted with respect to the orbit plane when the thruster is operating. Only the primary thruster has a position that ensures that the thrust line falls through the satellite center of gravity. Thus, the attitude control system requirements would increase when a redundant thruster was used.

Control Electronics

The arcjet electronic module incorporates both the control electronics (ACE) and the arcjet power electronics (APE) and is provided by AMSAT Germany. The prototype power supply has been used without any malfunction during laboratory tests since November 1993 for a total of more than 2000 h.

The APE unit includes a power converter, the ignition circuits, and the thruster control electronics and sequencer. The power converter uses the unregulated bus voltage of $+28$ V and converts nominal 750 W into a negative voltage of approximately -100 V supplied to the cathode of the arcjet motor. This approach allows the motor case to be grounded, simplifying its spacecraft integration and helping to reduce EMI radiated from the system.

A converter of such a power level presents quite a challenge, from the point of view of both efficiency and mass. For the present design a switching rate of 100 kHz was chosen as a compromise between mass and efficiency. The unit has a mass of approximately 2 kg and a measured efficiency of about 93%. This still means that some

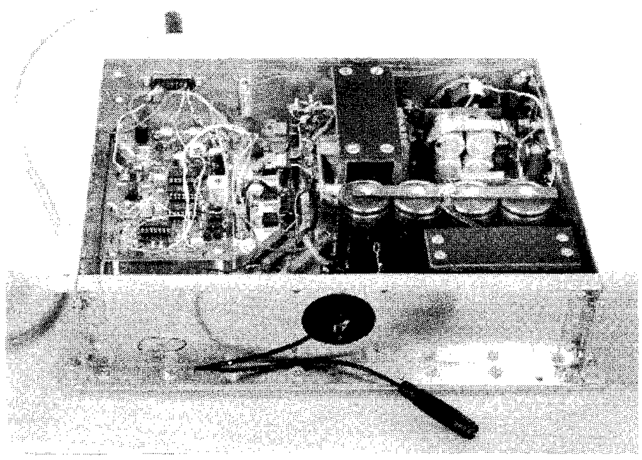

Fig. 6 APE prototype, cover plate removed.

60 W has to be dissipated. The size of the module ($300 \times 200 \times 80$ mm^3) does not provide enough surface area to dissipate this amount by radiation. Thus, the APE module will be mounted on a heat-sink area provided in the P3-D spacecraft for high-dissipation modules. This heat sink consists of several heat pipes inside the satellite that evenly distribute the heat within the structure. The overall satellite temperature is kept at 5–15°C by the position of these heat pipes with respect to the electronic modules that dissipate large amounts of energy and by the application of high-emissivity coatings at appropriate locations.

Once the ammonia flow has stabilized some 10 s after opening the valves, the ignition circuitry discharges an inductor to provide a pulse in the order of -3000 V within 5 μs. After ignition, the power converter takes over and supplies a steady current to the motor. The control loop is well behaved because of the high switching rate of the converter and posed no problem in designing for the negative resistance of the motor.

The present prototype unit (Fig. 6) does not yet include the sequencer for automated motor operation. The final version, however, will additionally include the ACE control electronics for the arcjet system. The system's health status will be determined from the thruster arc current and voltage and the MFC output. All system parameters will be available to ground operators through the telemetry channels at a normal interval of one dataset per minute during nominal arcjet operation. The satellite onboard computer will determine the system status from these data every second; this increased data rate is available to the ground operators only during system abnormalities.

The onboard main computer together with the ACE will provide the sequencing for the thruster startup, the arcjet operation schedule being periodically updated from the amateur radio ground operators. The thrusters' efficiency and thrust level will be determined by evaluating changes in the orbit parameters. The P3-D orbit elements will be determined onboard with an accuracy better than 5 m by an onboard GPS receiver.

Arcjet Thruster

The ATOS arcjet thruster (Fig. 7) is a derivative of the 1.6-kW Arcjet Thruster, University of Stuttgart (ARTUS II) hydrazine arcjet thruster.[11] Its design is based on the engineering model from that project, but the nozzle and electrode geometry and size were downscaled to match the lower mass-flow and power levels. Contrary to the hydrazine version, the propellant feed line of the ATOS thruster is positioned at the thruster rear end to increase the regenerative cooling effect from the cool gas flowing through the thruster body.

The nozzle section is machined from thoriated tungsten; it consists of an injector ring to inject the propellant with a tangential component into the arc chamber. The main thruster body is produced from a molybdenum–rhenium alloy; the material for the rear end of the thruster is Inconel. All like-on-like metal connections are welded, whereas metallic joints of different materials are high-temperature vacuum-brazed. The different welding and brazing processes were selected with respect to the material compatibility and the expected temperatures.[12] The insulating material is boron nitride.

The thruster will be mounted on the satellite bottom side opposite the 400-N thruster. The fin-type thruster mounts will reduce the heat load from the thruster body to the satellite structure, and the thruster length will minimize the temperatures at the thruster end by increasing the radiation surface.

Table 3 presents the overall ATOS system performance. The achievable lifetime was calculated with the nominal 24-mg/s mass flow, based on a 52-kg tank filling. It is obvious that the thruster cannot rely on the solar-array power alone. Thus, the thruster will draw most of its energy from the satellite power storage battery with its 40-A-h capacity, which will allow 60 min of continuous arcjet operation. The arcjet system will be able to provide a total Δv of 1150 m/s to the satellite. About 300 m/s will be necessary to perform the inclination change over the first year.[13] Furthermore, a typical 180 (m/s)/year is required to stabilize the 16-h orbit and to correct any disturbances. Thus, P3-D will be able to actively control its orbit for roughly five years.

Lifetime Tests

Two lifetime qualification tests were conducted in a vacuum test chamber. The vacuum tank is equipped with two Roots pumps and one rotary vane pump; they maintain a background pressure of 5 Pa at a mass flow of 30 mg/s. It is equipped with a pendulum-type thrust balance, an improved version of the thrust balance described by Glogowski et al.[14] The ammonia mass flow through the thruster is controlled with an MFC; in addition, the integrated ammonia consumption is monitored with a high-precision scale that allows weighing the ammonia gas cylinders with an accuracy of 1 g. This method of verifying the MFC was found necessary during the performance mapping of the ARTUS I thruster[11] because the MFC does not have the required accuracy to use its output for this purpose.

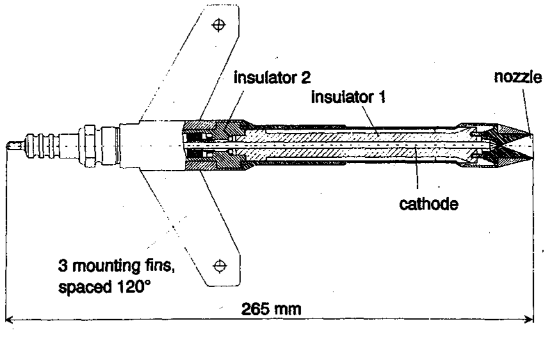

Fig. 7 ATOS thruster design.

Table 3 ATOS system performance

Total power to arcjet system	850 W[a]
Power for heater elements	50 W[a]
APE efficiency	>93%[a]
Power available to thruster	750 W[a]
Thruster current	8 A[a]
Thruster voltage	93 V[a]
Mean mass flow	24 mg/s[a]
Mean thrust	115 mN[a]
Mean specific impulse	480 s[a]
Achievable Δv	1150 m/s[b]
Thruster can be operated (with 52 kg NH_3)	600 h[a]
Demonstrated system lifetime	1010 h,[a] 1010 firings[a]
Thruster mass	0.5 kg[a]
Power-supply mass	2 kg[a]
System mass[c]	5 kg[a]

[a]Measured or demonstrated values.
[b]Estimated values.
[c]Includes one thruster, one APE–ACE module, valves, 1.5 m of fuel lines, but no tanks.

The test stand is controlled by a personal computer that is monitored by watchdog electronics to allow a 24-h hands-off (i.e., unsupervised) operation. All safety-related components are designed in such a way that, e.g., a power malfunction leaves the test stand in a save mode (valves closed, power supply off, etc.).

The first life test lasted for two months and covered 670 cycles (1h on, 1 h off). All system components, such as the APE, valves, gas generator, and MFC, completed the test without problems and were thus considered to be ready for flight.

However, the thruster itself posed some problems; thus the test was cut short before the originally intended 1000 cycles were completed. Nevertheless, the test was considered a success, even for the thruster, by the AMSAT organization, as the thruster was operational for a time period longer than the anticipated lifetime on the satellite.

The problems with the thruster started after some 50 cycles, when constrictor closure[15] was observed that led to rather high inlet pressures. The constrictor closure reversed after about 100 cycles, when severe erosion occurred. After about 200 cycles the thruster stayed in the low-mode phase after ignition for typically 5 min. This phase is characterized by a rather unstable arc with a voltage typically below 60 V; most erosion in the arc chamber is usually attributed to this mode. The remaining 55 min of normal high-mode operation during each cycle were characterized by frequent instabilities, when the thruster fell back into low-mode operation roughly every minute for less than 2 s. Because of these apparent problems the life test was cut short when 670 cycles were reached, since no new results would have been gained by a continuation of the test. It became evident that some redesign of the nozzle region would be necessary. Nevertheless, the system was accepted as flight-ready on the condition that additional lifetests would show that the constrictor closure problem could be solved. This decision was eased in that the thruster still had a specific impulse of slightly over 400 s with mass flows of 25 mg/s and a power level of 780 W averaged over a 1-h cycle at the end of the life test.

Posttest inspection of the nozzle showed severe erosion near the constrictor and significant deformation at the cathode tip, typical signs of prolonged operations in the low mode. All evidence suggested that the constrictor–nozzle region of the thruster became too hot during the experiments. Consequently, the second prototype thruster was modified to reduce the constrictor temperature by increasing the constrictor diameter from 0.6 to 0.7 mm and by providing a larger radiation surface for better cooling. Both the small nozzle size and the stringent time schedule precluded experiments with additional regenerative cooling or other measures.

This modified thruster was used in a second life test, which achieved the planned 1000-cycle duration. Although the inlet pressure and constrictor diameter measurements after the test showed that the constrictor closure phenomenon occurred again, its extent was such as to pose no problem to the arcjet operation on the satellite.[16] Additionally, the thruster displayed an specific impulse of 480 s at the end of the life test, thus surpassing the 400 s prescribed by the satellite designers by a comfortable 20%.

Consequently, three identical thrusters incorporating the design changes from the second life test were fabricated and tested for 50 h each. The thruster with the best performance with regard to constrictor closure, ignition reliability, and arc stability was selected as the flight thruster. Together with the propellant feed system, it has since been installed in the satellite structure; the remaining two thrusters will be available for additional ground tests, including thruster startup under extreme temperature conditions (−50°C) and vibration shaker tests.

Pending Tests and Résumé

The flight-type power supply currently being built by the amateur radio organization will be used for a final series of end-to-end tests of the whole arcjet system to examine the correct sequencing of the system operation on the satellite. Additionally, the completed satellite will have to undergo thermal vacuum and shaker tests. The thermal vacuum test will be conducted in a space simulation chamber that unfortunately does not allow arcjet operation because of the high mass flow of the arcjet thruster. Finally, the arcjet control software on the satellite will have to undergo robustness tests to check the system for various error conditions, both from the thruster system and the satellite environment, to verify the save arcjet operation even under extreme conditions. The flexible software architecture of the P3-D satellite allows this test to be conducted when the satellite is already integrated into the launcher.

If successful, the project should be suited to demonstrate once more the mature status of arcjet systems. Although the AMSAT qualification standards are more entrepreneurlike than industry standards can be, and the ATOS thruster might not be applied without changes to a commercial project, ATOS should be able to pave the way for arcjet applications beyond north–south stationkeeping, making them a prime candidate for attitude and orbit control of nongeostationary communication satellites.

Acknowledgments

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