1. Abstract and Introduction
Following the successful outcome of the 1992 Granada Ministerial Conference, European Telecommunication activities embodied in the Agency's Data Relay and Technology Mission (DRTM) Programme are being pursued at full speed, with Italy's Alenia Spazio in the role of ESA's Prime Contractor for both elements of the DRTM Programme: ARTEMIS and DRS.

ARTEMIS will act as a pre-cursor on the DRS element, with data relay functions in optical communications as well as in more classic frequency bands, like S / Ka / Ku. In addition, ARTEMIS will carry land mobile payload in L-band that will complement the European Mobile System payload carried on Italy's ITALSAT F-2, thus constituting an operational system to handle land mobile communication traffic. To carry such a large and diverse payload complement and still fit on a platform of attractive sizing for a shared launch on a commercial vehicle like ARIANE is a significant achievement, noting that at least 10 years of station keeping is a must to allow development of the market of small dish users.

A key enabler to this achievement is Ion Propulsion, and the paper addresses the comprehensive cooperation presently pursued under Alenia leadership with full support of Deutsche Aerospace AG (DASA) in close cooperation with Matra Marconi Space (MMS) that is required to raise a new propulsion system to maturity and full operational status commensurate with its use on ARTEMIS spacecraft for launch in 1996.

This effort is driven by the recognition of the substantial increase in revenue that a commercial operator could potentially reap when applying ion propulsion to spacecraft in the class of a shared launch.

The first DRS spacecraft will be launched 3 years after the ARTEMIS launch. Consequently, DRS will be able to benefit from the qualification and life test data gathered during the ARTEMIS flight programme, and is envisaged to close in on a more commercial configuration of the ion propulsion package than presently conceived on ARTEMIS. The paper addresses the main issues in selecting this configuration in relation to the design of the next generation spacecraft.

2. The ARTEMIS Satellite

2.1 General
Today's communication satellites have demonstrated their competitiveness against terrestrial systems in providing cost effective long distance and thin route traffic, fast verification of expanding networks with new communication services and will also become increasingly important for mobile communication.

In order to retain and increase competitiveness new technologies have been developed and are now available for a new satellite generation.

ARTEMIS is a communication technology demonstration satellite, for advanced data relay and land mobile applications. Fig. 2.1-1 shows the on-orbit configuration as presently defined.

The ARTEMIS Baseline Program element consists of the development, launch and in-orbit operation of a single geostationary satellite.

The payload elements are:
- a laser optical data communication experiment (ODR), providing a high data rate link with LEO satellites;
- a Data Relay element in S and Ka band (SKDR) for demonstration and operation of data relay services in 2 GHz and 23/26 GHz bands;
- an L-Band land mobile services payload (LLM), utilizing a large reflector to provide spot beams and exercise frequency reuse.

Besides, ARTEMIS will fly advanced platform technologies. As an outstanding example of the latter, the satellite will use Ion Propulsion for 10 years North/South stationkeeping to demonstrate its operational capability for a commercial application in other future missions.

Although the advantage of S/C mass saving by Ion Propulsion due to its very high specific impulse is evident, there are also some problems for the system to be solved concerning the
availability of mounting area, thrust vector alignment through
the varying center of gravity of the satellite, availability of
electrical power as well as the structural-thermal interfaces
with the spacecraft.

By taking these physical problems already in the design stage
of the satellite into account and by carefully planning the
North/South station keeping manoeuvre strategy it will be
possible to exploit and demonstrate the benefits of Ion Propul-
sion for ARTEMIS first, and for future communication satelli-
tes in general.

2.2 General Benefits of IPP

It is well known that the benefit of Ion Propulsion stems from
the extremely high exhaust velocity of the ions resulting in a
specific impulse \( I_{sp} \geq 3000 \text{ s} \) compared to some 300 s of bi-
propellant engines.

On the other hand the thrust force generated by Ion Thrusters
at a comparable mass is a factor thousand lower than that of a
bipropellant engine (see Table 2.2-1).

Table 2.2-1: Thruster Comparison Ion vs. Chemical (Biprop)

<table>
<thead>
<tr>
<th></th>
<th>ISP</th>
<th>TRUST F</th>
<th>DC POWER REQUIRED</th>
</tr>
</thead>
<tbody>
<tr>
<td>Ion Thruster</td>
<td>( \geq 3000 \text{ s} )</td>
<td>10 - 25 mN</td>
<td>400 - 800 W</td>
</tr>
<tr>
<td>Chem. Thruster (Biprop.)</td>
<td>300 s</td>
<td>1 - 25 N</td>
<td>4 - 8 W (Short term)</td>
</tr>
</tbody>
</table>

Therefore, for a mission requirement of a determined impulse
the required burn time of an ion thruster is by the factor thou-
sand higher than for a bipropellant engine.

Because of this fact ion propulsion is not suitable for injec-
tion manoeuvres of geostationary satellites which must be on
station within one month after launch, ready for commis-
ioning.

For lower impulse levels as required for attitude control, E/W
stationkeeping and also station change the required dry mass
for the IPP and additional thrusters (or gimbal mechanisms)
needed to cope with the different thrust directions would not
provide significant advantages.

North/South station keeping manoeuvres, however, allow for a
rather simple thruster arrangement on the satellite. Fig. 2.2-1
shows the thruster arrangement on ARTEMIS in principle.

Two (redundant) pairs of ion thrusters are mounted at the rear
end of the S/C North and South panel (near the anti-Earth pa-
nel) in such a way that their thrust vector is acting through the
S/C center of gravity. Thus the disturbance torques are mini-
mized as well as the number of thrusters required.

Each thruster pair is mounted on an Ion Thruster Alignment
Mechanism ITAM, which is also able to control even small
ion beam inaccuracies and keeps the thrust vector aligned
through the varying C.O.G. of the spacecraft during its 10 ye-
ars mission lifetime.

The canting of the thrust vector against the y-axis causes a
loss in thrust efficiency according to the cosine of the mean
vector declination angle of RT and EIT of 45°.

The Figure also shows that thruster firing in both, the ascend-
ing and descending node equalize the in plane disturbing
component.

Together with the small losses occurring during the approx. 3
hours firing time the efficiency decreases to an equivalent \( I_{sp} \)
of 2000 s. These losses can also be expressed in terms of a
corresponding higher velocity increment.

Considering the degradation Fig. 2.2-2 compares the propel-
lant mass ratio of a bipropellant system with that necessary by
applying ion propulsion for North/South station keeping.

The diagram generally visualizes the benefit of ion propulsion
in terms of propellant mass saving taking into account the
constraints described above.

In accordance with the ARTEMIS mission analysis and the
velocity increments required for the sequential mission phases
a mass comparison is presented in TABLE 2.2-11 between a
NSSK manoeuvre performed by the IPP and by the chemical
bipropellant system.

The mission analysis has derived a delta \( v \) figure for NSSK in
this Table from the declined thrust vector of the IPP as well as
from a "constrained firing strategy" which does not allow
thrusting during:

- battery charging periods within the 40 days eclipse
- those hours of the day at which parts of the solar array are
shadowed by the large antennas of the satellite.

By these means the additional solar array power required for
the IPP is only 180 W (about half of a solar panel) compared
to 600 W consumed during thrust operation.

Due to this constraint firing strategy the effective \( v \) for
10 years NNSK increases from nominal 427 m/s to 662 m/s at
an effective specific impulse of 3000 s.
The benefit for ARTEMIS is summarized in Table 2.2-III in terms of a payload mass gain of 46 kg corresponding to a 11% payload increase.

Table 2.2-III: Benefit of Ion Propulsion for ARTEMIS

<table>
<thead>
<tr>
<th>Mo = 2600 kg</th>
<th>UPS only</th>
<th>+ IPP</th>
</tr>
</thead>
<tbody>
<tr>
<td>S/C Dry Mass</td>
<td>1265</td>
<td>1408</td>
</tr>
<tr>
<td>- He-Pressurants</td>
<td>4.2</td>
<td>3.5</td>
</tr>
<tr>
<td>- Balance Mass</td>
<td>13.8</td>
<td>16.0</td>
</tr>
<tr>
<td>S/C Useful Mass</td>
<td>1247</td>
<td>1388.5</td>
</tr>
<tr>
<td>- IPP</td>
<td>-</td>
<td>86.0</td>
</tr>
<tr>
<td>- Margin 10%</td>
<td>-125</td>
<td>-130</td>
</tr>
<tr>
<td>- Platform</td>
<td>-703</td>
<td>-707.5</td>
</tr>
<tr>
<td>Available Payload</td>
<td>419</td>
<td>465</td>
</tr>
<tr>
<td>Payload Gain</td>
<td>46 kg or 11%</td>
<td></td>
</tr>
</tbody>
</table>

2.3 The ARTEMIS Platform Propulsion Concept

The propulsion concept of the ARTEMIS platform is based upon the unified bipropellant system UPS experienced on ITALSAT, uprated to a maximum propellant loading capacity of 1400 kg using two integrated tanks with newly developed propellant management device. Presurization is provided by 3 Helium gas pressure tanks mounted within the lower section of the central cylinder.

Fig. 2.3-1 shows the propulsion system configuration together with the central structure.
The fourth pressure tank is used for storage of the Xenon gas which is the propellant for the Ion Propulsion Package (IPP). The tank is designed for 40 kg Xenon, stored at 120 bar, more than sufficient for at least 10 years of NSSK operation with the redundant ion thruster pairs.

The bipropellant tanks will be filled to the extent necessary to cover apogee injection, East/West station keeping and attitude control, station change and final de-orbiting.

Fig. 2.3-2 shows the UPS block diagram of ARTEMIS.

### 3. THE ION PROPULSION PACKAGE (IPP) DESIGN

#### 3.1 Requirements on the IPP

On the basis of the propulsion concept described in the chapter above the following requirements shall be applied:

An average operation time per node and day for RITA of average 3 hours at 15 mN thrust level will be necessary to compensate the N/S disturbance of each day. This leads to an operation time for each thruster of about 10,000 hours per thruster in 3650 cycles.

The other requirement are:

- Thrust level RITA: 15 mN
- Thrust level EITA: 18 mN
- Thrust repeatability: ± 5%
- Specific impulse: >3000 s
- IPP total impulse: 1.2 x 10^6 Ns
- IPP-Power consumption: < 626 W
- IPP-Dry mass: < 83 kg
- Propellant loading capability: 40 kg

### 3.2 Composition of the IPP

The IPP consists of the following subassemblies:

- 2 Radiofrequency Ion Thruster Assemblies (RITA)
- 2 Electronbombardement Ion Thruster Assemblies (EITA)
- 1 Propellant Storage and Distribution Assembly (PSDA)
- 1 Ion Thruster Alignment Assembly (ITAA)

The propellant Xenon is stored in a tank within the PSDA at high pressure at the beginning, which decreases during propellant consumption. A pressure regulator in the path to the thrusters regulates the pressure down to 2 bars.

2 complete RITA's and 2 EITA's are prepared to generate thrust by acceleration of ionized Xenon in an electrostatic field. 1 RITA and one EITA are mounted on a gimbaled platform of the ITAA, one on the North- and one on the South panel of the satellite, to adjust the alignment of the thrust vector with respect to the actual position of the centre of mass of the satellite to minimize the secondary disturbances.

Fig. 3.2-1 shows the blockdiagram of the Ion Propulsion Package.
3.3 Radiofrequency Ion Thruster Assembly (RITA)

3.3.1 Composition of RITA

The RITA is developed and fabricated under the responsibility of DASA.

Each RITA consists of the following units:

- One Radiofrequency Ion Thruster (RIT) and one Neutralizer
- One Radiofrequency Generator (RFG)
- One Flow Control Unit (FCU)
- One Power Supply and Control Unit (PSCU)

The blockdiagram of the two RITA's for ARTEMIS are shown in Fig. 3.3-1.

In the following the main RITA data are shown:

<table>
<thead>
<tr>
<th>Component</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrust</td>
<td>15 mN</td>
</tr>
<tr>
<td>Anode voltage</td>
<td>+1500 V</td>
</tr>
<tr>
<td>Accel grid voltage</td>
<td>-1500 V</td>
</tr>
<tr>
<td>Deceleration grid voltage</td>
<td>0 V</td>
</tr>
<tr>
<td>Power consumption for 15 mN</td>
<td>585 W</td>
</tr>
<tr>
<td>Specific impulse incl. losses</td>
<td>&gt;3000 s</td>
</tr>
</tbody>
</table>

3.3.2 Design and Function of the RIT Thruster

Fig. 3.3-3 shows the cross-section of the RIT Thruster.

The propellant Xenon enters the discharge chamber of the RIT through the isolator and the extraction anode which also operates as a flow distributor.

To start the ionization of the Xenon the neutralizer is activated first. Electrons generated in the discharge at the neutralizer tip are drawn into the discharge chamber towards the extraction anode when a voltage of +70 V to the accelerator grid and of +1500 V to the anode is applied. These electrons accumulate energy from the high frequency field of the induction coil surrounding the discharge chamber and ionize the neutral propellant atoms by inelastic collisions.

Once the discharge has started it is self sustaining and the voltages on the electrodes can be switched off. The RIT is now in stand by condition, ready for thrusting.

To generate thrust, a voltage of +1500 V is applied to the anode and -1500 V to the accelerator grid. Under the influence of this electrostatic field positively charged atoms (ions) are accelerated towards the thruster outlet.

The ion beam is neutralized by electrons from the discharge at the neutralizer tip. The ion beam will act as a potential wall for free electrons. The current withdrawn from the neutralizer thus will match with the needs for neutralizing the ion beam automatically.

Thrust control is realized by the control of the beam current via the density of the ions in the discharge chamber by variation of the energy of the RF-field via an automatic control loop.
The electrons in the neutralizer are generated by ionization of Xenon in a low voltage arc discharge between cathode and keeper of the neutralizer.

3.3.3 Radiofrequency Generator (RFG)

The objective of the RF-generator is to provide RF-power at the natural frequency of the series resonance circuit set up by a resonance capacitor, the induction coil and the Xenon plasma. Since the natural frequency of this circuit is dependant on the plasma conditions a frequency control logic is used to prevent unintended breakdown of the discharge process.

Fig. 3.3-4 shows the functional blockdiagram of the RF-generator.

The tracking capability of the Phase-locked-loop (PPL) within the RF-generator ranges from 800 to 1200 kHz. The output frequency is continuously varied following the respective natural frequency of the resonant circuit. The lock status of the PPL is controlled by the inlock detector. If PLL lock is lost, the Automatic Frequency Search (AFS) starts a frequency search to reacquire lock status.

The RF-power output is between 10 and 150 W depending on the operation condition of the RIT. The power demand for the nominal operating condition of 15 mN thrust is 120 W.

3.3.4 Flow Control Unit (FCU)

The FCU basically is a double electronic pressure regulator, one for the RIT and one for the neutralizer. Flow control is achieved by adjustment of the pressure in the two plena through the respective chopper valves by opening when the concerned plenum chamber pressure equals or falls short of the a defined lower threshold and by closing when the plenum chamber pressure equals or exceeds a defined upper threshold.

Fig. 3.3-5 shows the functional blockdiagram of the FCU.

The plenum chamber pressures are sensed by means of pressure transducers, the signals of which are processed in the FCU electronics installed in the PSCU.

Adjustability of pressures and hence flow rates is possible since thruster and neutralizer characteristics can vary during life and flow restrictors used in the valves are subject to manufacturing tolerances. At begin of the mission RITA starts with nominal pressures preset as default values in the PSCU.

The neutralizer branch is equipped with an oxygen absorber which will protect the hot cathode, when the neutralizer is in operation, against the poisonous effects of oxygen and humidity.

3.3.5 Power Supply and Control Unit (PSCU)

The PSCU contains all the electronic sections necessary to operate the RITA from the spacecraft mainbus and interface with telemetry- and telecommand subsystem, developed and fabricated by FIAR in Italy.

Fig. 3.3-6 shows the PSCU blockdiagram.

RITA is switched ON and OFF by two high level commands. The normal thruster operation is controlled by the Control Logic Section, which is based on a 80C31 microcontroller, running at 4 MHz.
Operation parameters like thrust level, mass flow and negative high voltage can be adjusted by ground commands. Automatic optimization of the operation parameters has not been foreseen.

3.4 Electronbombardement Ion Thruster Assembly (EITA)

3.4.1 Composition of EITA

EITA is developed and fabricated under the responsibility of Matra Marconi Space.

Each EITA consists of the following units:
- One Electronbombardement Ion Thruster (EIT) and neutralizer
- One Propellant Supply Monitoring Equipment (PSME)
- One Power Conditioning and Control Equipment (PCCE)

The blockdiagram of EITA is shown in Fig. 3.4-1.

In the following the main data of EITA are summarized:
- Thrust level 18 mN
- Anode voltage +1148 V
- Accel grid voltage -350 V
- Decelerator grid voltage 0 V
- Power consumption at 18 mN 600 W
- Specific impulse inc. losses 3200 s

Some of the main data, mainly the decelerator grid voltage, may change in order to minimize the erosion rate in the grid system.

3.4.2 Design and Function of EIT Thruster

The basic construction of EIT is a perforated stainless steel cylinder with a conical rear section. Three gas inlet pipes and two electrical cables pass through it.

The main structural items of the thruster are the Swedish iron backplate and solenoid cores to which the casing is attached by means of ceramic isolators.

The outlet of the body is enclosed by the accelerating grid assembly. The grid assembly consists of three perforated molybdenum grids. The grids are maintained at an accurate separations by means of six isolator/bracket assemblies.

Fig. 3.4-2 shows the cross-section and the function principle of EIT.

Ionization of the Xenon propellant in EIT is achieved by an electron colliding with a neutral propellant atom and having sufficient kinetic energy to remove an electron by collision.

Electrons supplied by the main cathode gain energy by acceleration through the potential difference between cathode and anode. The probability of an ion collision is increased by increasing the travel path length by superimposing an electromagnetic field over the discharge chamber.

The ions generated within the discharge chamber are accelerated in the 3 grid dished acceleration system. The first grid is at +1100 V, the second grid on -350 V and third grid is grounded.

![Fig. 3.4-1: EITA Blockdiagram](image-url)
or will be slightly positive to decrease erosion by charge exchange ions.

To prevent charging of the spacecraft due to the expulsion of the positively charged beam, a plasma bridge neutralizer is mounted on the thruster with its outlet adjacent to the grids.

3.4.3 Propellant Supply and Monitoring Equipment (PSME)

The PSME is designed to provide Xenon propellant to the three thruster inlets: main flow, cathode and neutralizer. Fig. 3.4-3 shows the block diagram of the EITA-PSME.

The mass flow rate to each of the thruster flow components can be adjusted independently by the power conditioning and control equipment which regulates the Xenon supply to each plenum, by varying the regulator valve duty cycle.

Each regulator branch contains a regulator valve which controls the Xenon flow to the plenum. Restrictors are placed in the valve and downstream of the plenum to regulate the pressure of the gas flowing through the system. The cathode and neutralizer regulator have start valves included which bypass the plenum and the downstream restrictors. A start valve is used to generate a short pulse of gas to the cathode or neutralizer to aid the striking of an arc.

The flow rates can be adjusted by the Power Conditioning and Control Equipment (PCCE) to allow for changes in performance during the mission lifetime.

3.4.4 Power Conditioning and Control Equipment (PCCE)

The PCCE provides electrical interfaces between the EITA and the spacecraft and ten power supplies to the thruster. An electrical independent circuit is used to provide power and control of the pressure regulator of the PSDA (Propellant Storage and Distribution Assembly).

Fig. 3.4-4 shows the block diagram of the PCCE.

Fig. 3.4-2: Cross-Section and Function Principle of EIT

Fig. 3.4-3: Block diagram of the EITA-PSME

Fig. 3.4-4: Block diagram of the EITA-PCCE
The Power Conditioning Unit transforms and conditions primary power to provide all the electrical power needs of the thruster and its ancillary equipment.

A total of ten regulated outputs are provided for the thruster, some voltage regulated, most current regulated. Each is enabled, as required, by the controller. Various parameters are monitored for telemetry purposes and for thruster control. Auxiliary power is provided for circuits that are referenced to primary return, to neutralizer return and to discharge return.

The auxiliary inverter is controlled by the spacecraft level telecommands, and provides line-regulated power to the neutralizer regulators and to all circuitry except that referenced to discharge return and the beam supply unit.

Post regulators, when enabled, provide regulated currents to drive the neutralizer heater or keeper, and also high voltage to initiate the neutralizer discharge.

The discharge inverter provides line regulated power to the discharge supplies. These comprise heater and keeper regulators very similar to those of the neutralizer, plus continuously variable regulated current for the anode. The magnet regulator also provides a CV regulated current to the magnet coil. These are both controlled by a digital to analogue signal.

The beam supply provides regulated high voltage bias, at high current between the neutralizer return and the anode. It also provides a low power negative high voltage output which is utilized to bias the accelerator grid and the decelerator grid.

A high voltage interface is provided to pass telemetry and control signals across the high voltage barrier between discharge and neutralizer returns.

A total of ten regulated outputs are provided for the thruster, turn to nominal and the electronic pressure regulator will, on ground command, commence regulating.

Propellant gas is supplied to the inlet of the active isolation valve at high pressure, the pressure being dependant upon the tank temperature and the mission phase.

The purpose of the ITAA is to store the propellant Xenon for RITA and for EITA at high pressure level and to deliver it at low pressure level of 2 bar ± 10% to the FCU's and PSME's with a mass flow capability of maximum 0.6 mg/s.

The individual components of the PSDA are:
- The Xenon storage tank
- The electronic pressure regulator mechanism (EPRM)
- Two isolation valves
- Two regulator valves
- Four low pressure transducers
- One high pressure transducer
- One oxygen absorber
- The high pressure fill- and vent valve
- The two low pressure fill-and vent valves
- The electronic pressure regulator electronics (EPRE), installed in the PCCE box

3.5.2 PSDA Function
The Xenon tank will be filled on ground to its required propellant capacity.

The first actuation of the FCU's and PSME's in space will be a flushing operation, removing any air or oxygen which may have entered the thruster feed lines before or during launch. This flushing operation allows the downstream pressure to return to nominal and the electronic pressure regulator will, on ground command, commence regulating.

Propellant gas is supplied to the inlet of the active isolation valve at high pressure, the pressure being dependant upon the tank temperature and the mission phase.

A heater is fitted on the pipework immediately upstream of the isolation valve to ensure that gaseous propellant only passes through the regulator. The critical point of Xenon is 58 bar and 16.6°C. The isolation valve is driven open continuously and held open at low power dissipation. The active regulator valve is driven open for a short time whenever the downstream pressure, as monitored by one of the low pressure transducers, falls below a set point. A second low pressure transducer acts as a pressure switch and is connected to the isolation valve. Should the regulator valve fail open or leak, or should the pressure transducer linked with the pressure valve fail, such that the pressure exceeds a preset value, the isolation valve is closed via the pressure switch. Also, in the event of power loss, both isolation and regulator valves will close automatically.

The isolation and regulator valves, plus both pressure transducers and electronics are duplicated to provide a redundant set of regulator equipments.

A blockdiagram of the PSDA is contained in Fig. 3.2-1 of this paper.

3.6 Ion Thruster Alignement Assembly (ITAA)

The task of the ITAA is to allow for on-orbit alignment of the thrust vector of the operating ion thruster with the S/C center of gravity via ground station control. For this purpose 2 thrusters are mounted on a stiff platform which allows for platform rotation around 2 perpendicular axes.

The industrial approach of DASA for a compliant ITAA is based on a new mechanism offered by the Austrian company ORS.

The ITAA is constituted of the following subunits:
- The two Ion Thruster Alignement Mechanisms (ITAM’s)
- The Ion Thruster Alignement Electronics (ITAE).

The blockdiagram of the ITAA is shown in Fig. 3.6-1.

Fig. 3.6-2 shows the principle of the mechanical part of the ITAA, the ITAM. It basically consists of a mounting platform carrying one RIT thruster and one EIT thruster on its top side and being suspended on three points on its bottom side.

Whilst one suspension point of the platform is supported by a fixed spherical joint the two other suspension points are linked to hinged points of struts. The respective opposite hinge points of two of these struts are movable by means of nuts driven by threaded spindles whilst the opposite hinge point of the third strut is again supported by a fixed spherical joint.
The ITAE is equipped with position counters which determine the respective motor position by counting the pulses in positive and negative sense. The position counters are volatile and position information is lost when power is switched off, unless position information is additionally kept on ground. Following definite loss of position information the motors will be commanded to the drive into an end position where an end switch will be actuated and the counters be reset.

During the launch the ITAM platform is kept in the nominal neutral position by a Hold Down and Release Mechanism which will be released by a definite command before the ion thrusters are set in operation. Two redundant thermal cutters will be used for this purpose. If both fail the ion thrusters can be operated in the neutral nominal position.

3.7 Conclusion on the Artemis IPP-Configuration

The close cooperation between DASA, Alenia and Matra Marconi Space has resulted in the definition of an Ion Propulsion Package that can be entrusted to keep Artemis in geosynchronous orbit for 10 years as an operational subsystem. Its reliability will be high (0.98) and adequate function and technology will be provided for this first opportunity in an operational mode.

4. ION PROPULSION FOR FUTURE GENERATION OF COMMUNICATION SATELLITES

4.1 Technology Evolution for future Applications

The EURECA experiment and the Artemis qualification efforts conducted to-date have revealed that ion engine technology has matured and has become ready for efficient integration of ion propulsion on telecommunication spacecraft.

Nevertheless, a better understanding of ground test facility effects and more refined and accurate diagnostics would considerably help the spacecraft design office to establish simpler spacecraft interfaces and ease operating procedures that are now driven by inherently limited ground based data.

There is thus considerable interest in improving test facilities, in particular regarding thrust vector alignment, including alignment stability. Although the laws of physics imply that a highly stable thrust vector is more likely to exist, uncertainties arising from limitations in measurement results obtained with to-days beam diagnostics determine the definition of spacecraft interfaces.

A similar problem can be noted on the issue of contamination, where the mass collected on deposition sensors mainly arrives from backsputtering off the target in the vacuum chamber.

A highly interesting but not yet proven development that might assist in the removal of test facility effects could be the decelerator concept proposed by the Russian Institute MAI/RIAME. This concept essentially utilizes the reverse principle of the SPT. Practical applications have so far been limited to coating of small objects, hence further development seems necessary before application to ion engine test facilities can be considered.

Surely a reduction in thruster erosion rates could help relax spacecraft configuration constraints and hence the costs of accommodation ion engines.

Starting from the Artemis configuration, other areas for cost reduction are:

- Reduction of dry mass by cross-strapping of power supplies and thrusters
- Simplified Xenon flow control, using either long-life analogue or hybrid, microswitch operated flow controllers.

In general, it appears that a second design iteration with ion

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Fig. 3.6-1: Blockdiagram of ITAA

Fig. 3.6-2: Principle of ITAM

Angular displacement of the thruster platform is achieved by changing the inclination angle of the struts via the movement of the nuts on the spindles. The struts with the non-movable hinge point is required to yield a statically determined fixation of the platform at any position.

The three suspension points of the platform allow angular movement by minimum 5° around the theoretical center position. The relationship between spindle revolutions and platform angular displacement is non-linear.

The two spindles are driven by stepper motors. Acquisition of a particular angular orientation of the platform is achieved by application of a defined number of pulses with reference to a known position. The pulses are generated through the ITAE according to the position correction commands received via memory load commands.
propulsion as an integral part of the spacecraft configuration is necessary to further future commercial applications. For instance, a more stringent control of the spacecraft centre of mass combined with a higher integration of the propellant supplies for ion propulsion with the pressurant system for chemical propulsion would lead to eliminate the need for an orbit alignment system and hence reduce mass, cost and criticality of this subsystem.

Another aspect is the feasibility to reduce the engine cant angle, which on Artemis is some 45 degrees. Enhanced knowledge on thrust vector stability and reduced contamination rates could help achieving this goal. Obviously, the reduced level of propellant consumption, of ground station operations, of operational life requirements would all contribute to a more competitive spacecraft system.

4.2 Implementation of the Ion Propulsion System on DRS

Following the decision taken at the Munich Ministerial Council, ESA has modified the data relay content of Artemis and DRS to better match the foreseen user requirements.

In this scenario, after the initial period of service over Europe, Artemis will be moved to a DRS orbital position and will provide an operational data relay service to the Polar Platform(s) together with the first DRS satellite, planned for launch in 1999.

Studies are on-going to define the strategy to implement the first DRS satellite making the best use of the Artemis heritage and possibly re-using the same hardware developed for Artemis.

For the Ion Propulsion Package (IPP) the strategy is based upon the following cornerstones:

- Hi-rel parts for a second flight subsystem are being procured by the IPP contractors as a part of the baseline Artemis Phase C/D programme.

- An option identified as “option B”, has been requested to the IPP subcontractors, including manufacture, integration and test of a second flight standard system, built using those hi-rel parts procured within the before mentioned baseline.

DRS is currently carrying out system and subsystem level analyses aimed to:

- Identify whether there are any specific DRS requirements that should be fed back into the Artemis subsystem specifications.

- Confirm the applicability of Artemis “option B” hardware for the DRS satellite.

Under the assumption that the contribution of the various countries on DRS will be comparable to Artemis, it is expected that the studies will confirm the possibility to take up “option B” for DRS and therefore the baseline design of the IPP subsystem on DRS should be just a recurring model from Artemis, with identical performance and characteristics as described in section 3 of this paper.

Should however in particular the UK-contribution not be available to support DRS then two options could be pursued:

- One is a four RITA option with improvements on the ion propulsion subsystem as indicated in the paragraph before. This would require additional investment, modification of boxes and of the configuration. The advantage of this solution is that it would bring RITA closer to a commercial application.

- The second option is that the UK thrusters are replaced by the SPT Mk II thrusters on the assumption that these thrusters respect the interfaces as defined on Artemis in order to contain costs on system level. The SPT thrusters would be designed, developed and tested to the same requirements as for the ion thrusters and ALS/DASA would have more degrees of freedom when proposals for commercial satellites are to be made in the future.

ALS/DASA want to point out that on the basis of current knowledge and experience in electric propulsion ion thrusters have proven characteristics, advantageous for the application to orbit control of geostationary satellites.

However, SPT thrusters appear to have the potential of competitive design and operation which still need confirmation through comprehensive ground qualification and characterization process. Until this process is brought to completion successfully, there is no doubt that ion engines will prevail in the selection for commercial applications.