ABSTRACT.

Artemis will be the first ESA satellite which operationally flies an advanced Ion Propulsion Package (I.P.P.) to prove the viability and reliability of such technology for future European commercial spacecraft.

The paper will, first, present an overview of the mission benefits stemming from the use of ion propulsion for orbit control in general, and to the Artemis mission in particular, taking into account that such technology is also a candidate for the European DRS system.

The satellite, designed for ten years lifetime, carries a mixed chemical-electrical propulsion system. The platform will be based on an uprated Italsat bus, suitably adapted to the Artemis mission. The electrical propulsion section consists of a redundant pair of thruster assemblies: two RITA based on RIT-10 devices by MBB, and two EITA, based on UK-10 devices by MSS. These will be installed on the north and south spacecraft panels, providing minimum interferences to payloads, instruments and other experiments carried on board the satellite.

An overview of the major satellite design and interface requirements, stemming from the use of electric propulsion, will be briefly presented. The entire matter will be dealt with in some detail in the paper 91-053 given at the conference.

The major program objectives will be discussed, with emphasis on the Satellite Design and Development Plan, which considers the different levels of development maturity of the equipments.

An overview of the European Companies participation to this ambitious program will be, also, given.

1. INTRODUCTION.

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 (increase of insufficient telephone and TV capacities) 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. Figure 1-1 shows the on-orbit configuration as defined during the definition phase B2/2.

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

The payload elements are:

- a laser optical data relay communication experiment (ODR), providing a high data rate link with LEO satellites;
- an S-Band multiple access data relay payload (SDR), for demonstrations of the technology needed by future data relay services for medium data rates;
- an L-Band land mobile services payload (LLM), utilizing a large reflector to provide spot beams and exercise frequency re-use.

Besides, ARTEMIS will fly advanced platform technologies. As an outstanding example of the latter, the satellite will apply Ion
propulsion for 10 years North/South station keeping to demonstrate its operational capability for the follow-on Data Relay satellites and 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 solar array 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 propulsion for ARTEMIS first, and for future communication satellites in general.

2. MISSION BENEFITS.

2.1 General Benefits of IPP.

It is well known that the benefit of Ion propulsion stems from the extremely high exhaust velocity of the ionized plasma resulting in a specific impulse Isp ≥ 3000 sec compared to some 300 sec of bipropellant 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 I).

<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>≥ 3000 sec</td>
<td>10 - 25 mN</td>
<td>400 - 800 W</td>
</tr>
<tr>
<td>Chem. Thruster (Bipropell.)</td>
<td>≤ 300 sec</td>
<td>1 - 25 N</td>
<td>4 - 8 W (Short term)</td>
</tr>
<tr>
<td>Ratio Ion/Chem.</td>
<td>10(^1)</td>
<td>10(^{-3})</td>
<td>10(^2)</td>
</tr>
</tbody>
</table>

Table I. Thruster Comparison Ion vs. Chemical (Biprop); (Typical Values)

Therefore, for a mission manoeuvre of a determined impulse requirement \(i = M_p \cdot \Delta v = F \cdot t_b = \text{Const.}\), the required burn time \(t_b\) of an Ion thruster is a factor \(10^3\) higher than for a bipropellant one.

Because of this fact Ion propulsion is not suitable for injection manoeuvres of geostationary satellites which must be on station within one month after launch ready for commissioning.

For lower impulse levels as there are attitude control, E-W station keeping and also a 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 body.

Figure 2-1 shows the Ion thruster arrangement on ARTEMIS in principle:

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

Each thruster pair is mounted on an Ion Thruster Alignment Mechanism ITAM, which is 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 years mission lifetime.

The declination \(\Theta\) 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 RIT and EIT of \(\Theta \sim 46\) deg.

The Figure also shows that thruster firing in both, the ascending and descending node equalize the inplane disturbing component \(T \cdot \sin \Theta\).

Together with the small losses occurring during the approx. 3 hours firing arc the efficiency decreases to an equivalent isp of 2000 sec. These losses can also be expressed in terms of a corresponding higher velocity increment \(\Delta v\).
Considering this degradation Figure 2-2 compares the propellant mass ratio of a unified propulsion system UPS (all manoeuvres with bipropellant) 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 only taking into account the constraint described above.

2.2 Benefits of Ion Propulsion for ARTEMIS.

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

<table>
<thead>
<tr>
<th>TRANSFER ORBIT</th>
<th>ΔV (m/s)</th>
<th>Propellant Mass (Kg)</th>
<th>Spacecraft Mass (Kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Apogee Injection</td>
<td>1495</td>
<td>925.0</td>
<td>Mo = 2368</td>
</tr>
<tr>
<td>LAEF1 + LAEF2</td>
<td>(31)</td>
<td>25.5</td>
<td>1417.5</td>
</tr>
<tr>
<td>Drift Orbit Attil. Contr. + Station Acquisit.</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>GEO ORBIT BOM</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Attil. Control</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>NSSK UPS</td>
<td>432.2</td>
<td>965.0</td>
<td>1403</td>
</tr>
<tr>
<td>IPP</td>
<td>698.1*</td>
<td>13.0</td>
<td>1390</td>
</tr>
<tr>
<td>EWSK UPS</td>
<td>29.5</td>
<td>12.3</td>
<td>14.0</td>
</tr>
<tr>
<td>IPP</td>
<td>29.9 Xe</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Station Change</td>
<td>17.0</td>
<td>7.0</td>
<td>8.0</td>
</tr>
<tr>
<td>Deorbiting (40 Km)</td>
<td>15.0</td>
<td>6.2</td>
<td>7.0</td>
</tr>
<tr>
<td>Residual + Margin</td>
<td>14.5</td>
<td>12+2.1 Xe</td>
<td></td>
</tr>
<tr>
<td>Total Delta Mass</td>
<td>1214.0</td>
<td>1051.0</td>
<td>1154</td>
</tr>
<tr>
<td>Δ Mp = 163</td>
<td>Δ S/C = 163 Kg</td>
<td>Dry Mass Gain</td>
<td></td>
</tr>
</tbody>
</table>

Table II.  Artemis Overall Mass Budget Comparison Nssk with UPS only (Isp = 290 sec) vs. IPP (Isp = 3000 sec)

(1) LAEF = Liquid Apogee Engine Firing

Cosine losses of N/S thrust component considered in ΔV (UK-10).
The mission analysis has derived the ΔV 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 of 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 solar panel) compared to the 600 W consumed during thruster operation.

Due to this constraint firing strategy the effective ΔV for 10 years NNSKK increases from nominal 432 m/s to 608 m/s for RITA and 698 m/s for EITA respectively at an effective isp = 3000 sec. The 698 m/s have been considered in the comparison.

The non-operating IPP periods are selected in such a manner that the satellite is drifting within the allowed inclination error limit cycle of ± 0.07 degree.

The comparison IPP vs. UPS for NSSKK results in a propellant mass saving of 163 Kg or a 7% reduction of S/C launch mass. The additional solar array power of 180 W, or 4 Kg, has been taken into account in the 8 panels design and in the platform mass of 707.5 Kg.

The benefit for ARTEMIS is summarized in Table III in terms of a payload mass gain of 65 Kg corresponding to a 21% PL increase.

<table>
<thead>
<tr>
<th>Mo = 2386 Kg</th>
<th>UPS only</th>
<th>+ IPP</th>
</tr>
</thead>
<tbody>
<tr>
<td>S/C Dry Mass</td>
<td>1154</td>
<td>1317</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>1136.0</td>
<td>1297.5</td>
</tr>
<tr>
<td>- IPP</td>
<td>-114</td>
<td>-121</td>
</tr>
<tr>
<td>- Margin 10%</td>
<td>-703</td>
<td>-707.5</td>
</tr>
<tr>
<td>Available Payload</td>
<td>318</td>
<td>383.0</td>
</tr>
<tr>
<td>Payload Gain with IPP</td>
<td>65 Kg (or 21%)</td>
<td></td>
</tr>
</tbody>
</table>

Table III. Benefit of Ion Propulsion (IPP) for ARTEMIS

2.3 Benefit of IPP for DRS

The payload requirements for the European Data Relay Satellite (DRS) exceeds that of ARTEMIS and amounts to 405 Kg. The according PL-power amounts to 2500 W and increases the total solar array power from 2500 to 2640 W. Hence the platform mass increases too.

A further comparative analysis may now demonstrate the benefit of ion propulsion for the operational DRS mission.

The currently required Masses are:

- PL-Mass 405
- Platform 720 (increased power)

Employing UPS only for 10 y life the total bipropellant mass would be 1324 Kg and the launch mass Mo = 2581 Kg.

With IPP for NSSK this reduces to Mp = 1072, and leads to a launch mass saving of 165.5 Kg.

Since ARTEMIS is the forerunner of DRS and uses the same platform/propulsion capacity we may assume a common satellite launch mass of about Mo = 2400 Kg.

3. THE ARTEMIS PLATFORM PROPULSION CONCEPT

The ARTEMIS or DRTM platform is derived from ITALSAT and uprated in terms of propulsion, structural size and power.

The propulsion concept is based upon the unified bipropellant system UPS uprated to a max. propellant loading capacity of ~ 1400 Kg using 2 ingreasable tanks (640 dm³ each, 1150 mm DIA) with newly developed propellant management device. Pressurization is provided by 3 He-pressure gas tanks (p = 270 bar) mounted within the lower section of the central cylinder.

Figure 3-1 shows the propulsion system configuration together with the central structure.

The fourth pressure tank is used for storage of Xenon gas (p = 120 bar) which is the propellant for the ion Propulsion Package IPP.

The tank is designed for 40 Kg Xenon, more than sufficient for at least 10 years North-South station keeping 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.

Figure 3-2 depicts the UPS block diagram of ARTEMIS.

4. THE ION PROPULSION PACKAGE (IPP)

4.1 Description

The IPP consists of four major assemblies as shown in the block diagram of Figure 4-1:

- The propellant supply and distribution system, PSDA, stores Xenon under high pressure and distributes it to the Ion thruster assemblies RITA and EITA.

- Two RITA, based on RITA-10 devices by MBB, and two EITA, based on UK-10 devices by MSS, form the redundant pairs of thruster assemblies.

Each thruster has its own power supply and control unit/equipment (PSCU respectively PCCE), a low pressure flow control unit or propellant supply equipment (FCU respectively PSME).

The radiofrequency thruster RIT requires a radiofrequency generator RFG.
Figure 3-1. ARTEMIS Propulsion System Configuration

Figure 3-2. UPS Block Diagram
The redundant thruster pairs are mounted outside the spacecraft on two alignment mechanisms, ITAM, controlled by one common electronics, ITAE.

The probability of success of the IPP for 10 years lifetime is assessed to be $R = 0.97$, achieved by means of full (cold) redundancy of RITA and EITA equipment in one package. This means a 3% failure probability compared with 30% for the total satellite ($R = 0.7$ required).

The requirements generated by this new technology have to be matched/optimized with the other subsystem requirements, mainly those of structure, thermal control, power supply, and AOC subsystem.

The mounting provisions as described in chapter 2.1 meet the operational and AOC requirements in an optimised way.

The required thrusting through the spacecraft C.O.G. of each operating thruster is provided by the alignment assembly, ITAA. The alignment range is ± 10°.

Beam accuracy measurements, performed with one RITA breadboard at the University of Giessen test facility, have shown considerable smaller thrust vector deviations of only 0.2 degrees during the warm-up phase, (whilst the initially estimated value was ± 3 degrees). After 15 min. warm up this value stabilizes during one firing period.

This improvement allows preadjustment by telecommand of the ITAM prior to firing periods and avoids frequent closed loop commands either on-board (via AOCs) or via ground processing.

With 583 W DC power (one thruster in operation) the IPP takes about 25% of the total S/C power. This is, however, not simply added to the solar array output, but is only taken into account for the Solstice periods.

The thrust level generated by RITA is 15 mN at a DC power of 578 W, whilst that of EITA is 18 mN at a DC power of 583 W total.

The specific impulse amounts to $\geq 3000$ sec and the total impulse capability $\geq 600.000$ Ns.

The IPP dry mass will be 76.5 Kg plus 40 Kg of Xenon propellant.

During the 42 days eclipse periods about the Equinoxes when the batteries are charged, and during those days where the solar arrays are shadowed by the large antenna reflectors the IPP is in an "Off" mode and no NSSK is provided.

During Equinox some 400 W power are needed for battery charging including thermal control which are made available to the IPP during Solstice periods.

Hence only 180 W additional Solar array power are charged to the IPP.

The Power distribution diagram Figure 4-2 of one IPP shows that the major part is radiated outside the S/C to the space environment.

The RF radiation of 254 W (RIT) respectively 369 W (EIT) occurs within a beam angle of ± 12° which causes no impact on the satellite. The thermal radiation of 106 W respectively 115 W may require thermal shielding to protect the solar array and the S/C body from excessive heating.

The electronics components radiating thermally about 90 W are mounted on a special section at the rear North and South radiator panels of the platform.

The PSDA equipment, especially its temperature sensible high pressure regulator, is placed on the lower (-z) side of the central horizontal platform which is the least exposed to thermal variations.

Figure 4-3 outlines the IPP equipment distribution on the ARTEMIS spacecraft.
Figure 4-2. IPP Power Distribution Block Diagram

Figure 4-3. IPP Equipment Distribution on ARTEMIS Spacecraft
4.3 Possible Ion Beam RF Interference.

A diagnostic of possible unwanted radiative emissions and contamination generated by the ion beam is desirable for this first operational mission.

A diagnostic package has been conceived for this purpose and will include:
- Contamination sensors measuring surface absorptance/emittance changes and solar cell power losses.
- RF signal monitors searching for RF emissions in the L, S and Ku frequency bands.

These Sensors are mounted on distinct parts of the S/C surface controlled by an electronics box mounted inside the S/C body.

5. PROGRAM OBJECTIVES/DESIGN AND DEVELOPMENT PLAN.

ARTEMIS is the first European spacecraft implementing operationally full North/South station keeping up to 10 years using Ion Propulsion only.

This experimental spacecraft offers a unique opportunity to demonstrate the maturity of this new propulsion technology for which developments in Europe are continuing since more than 20 years.

Once this is flight proven by ARTEMIS, the advantage of N-S station keeping by means of ion propulsion may be fully exploited in future communication satellite designs. Such satellites may get rid of large propellant tanks and consequently of propellant sloshing and C.O.G. shifting.

A thrust alignment mechanism can then be avoided.

In order to maximize the probability of success of ARTEMIS, redundant technologies are pursued to minimize risks. This combination of redundant technologies means the introduction of both the MBB (D) and MSS (UK) designs and hardware on-board the Advanced Relay Technology Mission spacecraft.

The different development status of the assemblies and equipment leads to a special model philosophy outlined in the design and development flow diagram of Figure 5-1 and 5-2.

The Ion Thruster Assemblies RITA and EITA have reached a rather mature development status achieved during foregoing developments using Mercury and Xenon propellants: a Rita flight experiment was already delivered to the EURECA platform. After successful completion of the breadboarding activities the developments will follow a protoflight philosophy in which EQM’s (engineering qual. models) are qualification tested followed by one protoflight model (PFM) and one flight model for integration in the PFM spacecraft.

The same philosophy is applied for the PSDA equipment and the alignment mechanism electronics, ITAE.

The Ion thruster alignment mechanism ITAM is a newly developed equipment and needs, therefore, an engineering model followed by a qualification model and two flight models. After qualification testing the QM is exposed to life cycle tests in thermal vacuum.

After unit qualification the Ion thruster units are assembled and functionally checked together with the ITAA engineering model at subsystem level and then delivered to the EM spacecraft for compatibility tests.

These tests shall confirm the Ion Propulsion subsystem performance and finally define the effort required for the IPP Flight Model procurement.
6. INDUSTRIAL TEAM AND RESPONSIBILITY.

More than 10 companies are involved in the IPP development for the ARTEMIS spacecraft. Figure 6-1 outlines the industrial team and their responsibilities.

The IPP subsystem depends upon the strong responsibility delegated at assembly level where the various required functions have to be implemented and verified. The companies responsible for the three main assemblies are:

- MBB for the RITA (based on RIT-10 thruster) (for phase B2/2, under negotiation for phase C/D).
- MSS for the EITA (based on UK-10 thruster).
- ORS/T.B.D. for the Ion thruster alignment assembly ITAA.

The diagram shows the suppliers (or potential suppliers in phase C/D) for the main subassemblies/units.

The subsystem management and the integration and test activity becomes a rather complex task for the prime contractor ALENIA Spazio and its supporting companies MBB/MSS and requires a careful monitoring and control of both assembly and unit level tasks performed by all the involved companies.

The prime contractor is responsible for the accommodation of the IPP subsystem on to the spacecraft and its mission success. On the other hand he is also dependent on the know how of the companies developing the IPP assemblies RITA and EITA since several years. A good cooperation of all participating companies is, therefore, achieved to guarantee a successful breakthrough of this new technology.

Figure 5-2. IPP Flight Models AIT

Figure 6-1. IPP Industrial Team and Responsibilities
7. CONCLUSIONS

Ion propulsion on ARTEMIS will save 163 Kg chemical propellant mass during 10 years North-South station keeping manoeuvres, although some 80 Kg additional hardware is required. The payload mass gain compared with UPS operation only is 65.5 Kg or 21%. Additional thermal radiation area is required for 90 W heat dissipation of the electronics equipment mounted inside the North and South panels of the satellite. The additional DC power to be generated in the solar array is limited to 180 W due to a special thrashing strategy.

The cold redundancy of power supply and control electronics as well as the thruster gimbal mechanism ITAM may become unnecessary once the IPP has proven its reliability and thrust vector stability during this first operational mission.

Furthermore a more straightforward development model philosophy and a reduced complexity in the industrial management can be considered for future missions.

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