THE UK-10 ION PROPULSION SYSTEM – A TECHNOLOGY FOR IMPROVING THE COST-EFFECTIVENESS OF COMMUNICATIONS SPACECRAFT

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ABSTRACT

The overall launch mass of a communications satellite has a major influence on its economic performance. If this mass can be reduced substantially, the launch cost will be lower, or more revenue can be generated by carrying a larger payload or by extending the operational lifetime. Since an ion propulsion system (IPS) provides an exhaust velocity at least an order of magnitude higher than that of the equivalent chemical thrusters, the propellant mass required for north-south station-keeping of a typical spacecraft can be reduced by several hundred kilogrammes. This application of ion propulsion is described and quantified in the paper, with reference to the UK-10 IPS and the Intelsat VII spacecraft. The paper also considers the need to optimise the operating parameters of the thruster for each mission.

1 INTRODUCTION

Since their introduction more than 25 years ago, commercial communications satellites in geosynchronous orbit have gradually revolutionised the world’s long distance communications. They have provided an ever-increasing capacity at a price per circuit that has been hard to match, together with excellent reliability. Despite the recent competition from optical fibre cable links, these satellites now carry a large proportion of all types of long distance communications, particularly telephone services, and dominate international television transmissions. They are especially competitive for "thin routes", for which normal terrestrial links are too costly, and are now also of increasing importance for mobile services of all types. In parallel, military communications in some countries have become dependent upon communications satellites, particularly for wide area coverage and mobile services.

Table 1

CHARACTERISTICS OF INTELSAT COMMUNICATIONS SATELLITES

<table>
<thead>
<tr>
<th>Designation</th>
<th>I</th>
<th>II</th>
<th>III</th>
<th>IV</th>
<th>IV-A</th>
<th>V</th>
<th>V-A/B</th>
<th>VI</th>
<th>VII</th>
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<td>750</td>
<td>1300</td>
<td>2525</td>
<td>525</td>
<td>1290</td>
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<tr>
<td>Mass at launch (kg)**</td>
<td>88</td>
<td>182</td>
<td>294</td>
<td>1418</td>
<td>1518</td>
<td>1932</td>
<td>1980</td>
<td>4170</td>
<td>3810</td>
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<td>Thor Delta</td>
<td>Atlas Centaur</td>
<td>Atlas Centaur</td>
<td>Atlas Centaur</td>
<td>Ariane 1,2</td>
<td>Ariane 1,2</td>
<td>Ariane 4</td>
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<td>Design lifetime (year)</td>
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<td>3.0</td>
<td>5.0</td>
<td>7.0</td>
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<td>500</td>
<td>800</td>
<td>2300</td>
<td>2180</td>
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<td>1500</td>
<td>4000</td>
<td>6000</td>
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<td>Capacity – TV channels</td>
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<td>2</td>
<td>2</td>
<td>2</td>
<td>3</td>
<td>††</td>
</tr>
</tbody>
</table>

C = cylindrical
E = extended cylindrical array
* with appendages (eg solar arrays) deployed
† dependent on mode of operation
** data from Ref 2
†† data not available in this form

* Now the Aerospace Division of the Defence Research Agency

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The growth of this communications market can best be illustrated by reference to the Intelsat series of spacecraft\textsuperscript{1,2}. Intelsat, the International Telecommunications Satellite Organisation, is a non-profit cooperative of about 117 countries which owns and operates the global communications satellite system used by countries around the world for international communications, and also for domestic services by many of them. Intelsat grew out of the USA’s Communications Satellite Act of 1962, which aimed to provide telecommunications services via satellites to all areas of the world. With a network expected to consist of about 16 active spacecraft at the end of 1991 and approaching 1000 ground stations, Intelsat links together more than 170 countries, providing for many of them services not available by other means. Eight generations of Intelsat spacecraft have now been designed, built and launched, the latest being Intelsat VI\textsuperscript{1,2}, first deployed by an Ariane 4 rocket on 27 October 1989. Their characteristics are summarised in Table 1. Also shown in this Table are the equivalent data for the next generation, Intelsat VII, currently under development by Ford Aerospace\textsuperscript{2,3}.

Table 1 clearly indicates how the size, mass and complexity of these satellites has increased with time, due to the need to provide greater capacity and a much wider range of services. This has been made possible by continuing technological development, in both spacecraft platform and communications areas. For example, the new Intelsat VI spacecraft have a nominal capacity of 24000 simultaneous two-way telephone circuits, twice that of the preceding Intelsat V, plus three TV channels. However, when digital circuit multiplication equipment is employed, its effective capacity can be as much as 120000 circuits. There has been a corresponding reduction in charges to users from $32000 per half-circuit per year in 1965 to $2520 in 1989\textsuperscript{2}.

Greatly enhanced flexibility is provided in the Intelsat VI design by reconfigurable and steerable transmit beams. For C-band traffic, two large dish antennas give both hemispheric coverage and four zonal beams that can be reconfigured in orbit by selecting an appropriate pattern from among the 149 feed horns. The two Ku-band spot beams can be steered to cover any selected area. There is also a complex on-board switching system, including a static switching matrix which permits a large number of interconnection patterns to be set up between all 48 on-board transponders. In addition, satellite-switched time-division multiple access capability is provided by an on-board microwave switch matrix, which allows dynamic demand-driven interconnections to be made between the separate beams. These innovations provide vastly enhanced flexibility, since capacity no longer has to be pre-assigned to specific beams or coverage areas on the basis of a possibly dubious pre-launch prediction of traffic distribution.

Certain platform advances, such as the provision of higher power levels, are necessary to service the new communications payload. Others, such as the high-performance integrated bi-propellant liquid propulsion system and the use of nickel-hydrogen batteries, are included to reduce total mass and increase lifetime. Since a very large proportion of the launch mass of a typical communications satellite is propellant, any improvements in propulsion system efficiency provide immediate benefits, as do other techniques for reducing mass.

It can be seen from the above discussion that the current trend is towards large, more complex, long-life spacecraft, which can provide enormous capacity in an extremely flexible way. However, the investment required to design, develop and produce these satellites is very great, and the launch costs are also high. Consequently, numerous studies and associated research programmes have been undertaken to find methods of reducing the overall cost of the system, and the cost per circuit\textsuperscript{4,5}. The latter aim is partly addressed by increasing overall capacity, mainly by introducing advanced high bandwidth systems, frequency re-use, spot beams, and so on. The overall cost depends on the procurement and launch costs of an individual satellite, and on the lifetime of each of them once they are in orbit. Consequently, methods of increasing lifetime are very significant and must be considered in any studies of costs.

The major factor in determining overall cost is satellite launch mass, since this determines the launch cost and, within an imposed mass budget, the payload that can be carried. As already mentioned, an often dominant contributor is the propulsion system and, in particular, the amount of propellant that must be carried. With the present day need for accurate station-keeping in the geostationary orbit, the propellant mass required can be very large; as an example, it can reach 2160 kg for Intelsat VII\textsuperscript{3}, although this also includes the amount necessary to circularise the initial geostationary transfer orbit (GTO). This mass increases approximately linearly with lifetime, which may extend to 20 years in the future\textsuperscript{4}.

In order to reduce the propellant mass required for a given mission, the exhaust velocity, v, of the thrusters employed must be increased. Since the velocity attainable by a chemical motor is limited by the energy released in the chemical reactions involved, only small improvements can now be realised using these devices. However, electric propulsion (EP) systems\textsuperscript{6} do not suffer from such limitations, because they employ an ionised or electrically charged propellant which can be accelerated to extremely high velocities by electromagnetic or electrostatic forces. Values of v more than as order of magnitude greater than those provided by the best chemical systems are readily available, allowing rates of propellant consumption to be reduced by the same ratios.

Although many different EP systems have been developed\textsuperscript{7}, for a very wide range of Earth orbit and interplanetary missions, this paper concentrates on one
of the most advanced of these, the ion propulsion system (IPS), and its near-term application to the north-south station-keeping (NSSK) of large geostationary communications satellites. For clarity, the UK-10 IPS is taken as an example, although in-orbit demonstrations will soon be undertaken of the German RIT-10 and Japanese MELCO thrusters as well as of the UK-10.

The purpose of the paper is to discuss the use of an IPS for this application, and to assess the optimum thruster operating conditions. The potential financial benefits of employing this new technology for NSSK are then estimated, assuming that there is no chemical back-up system on the spacecraft.

2 THE UK-10 IPS

The UK-10 IPS is being developed by a team led by Space Department of the Royal Aerospace Establishment, Farnborough. Other major contributors to the programme are Matra Marconi Space UK, the Culham Laboratory of AEA Technology, ERA Ltd, and Philips Components Ltd. Funding is provided by the Ministry of Defence, the Department of Trade and Industry, the Industrial Contractors and Intelsat.

The UK-10 is based on a Kaufman-type electron-bombardment ion thruster employing xenon propellant, although mercury was used in an earlier phase of the development. The thruster, designated T5 Mark III, is shown as an artist's impression in Figure 1 and mounted in an RAE test facility in Figure 2. It was originally designed to produce a thrust of 10 mN, but has since been shown to operate successfully at up to 70 mN. Consequently, it has been decided to qualify it formally for operational use at 25 mN thrust, which is more representative of current missions, such as Intelsat VII. This version, de-rated to 18 mN, is to be flown operationally on ESA's ARTEMIS spacecraft, together with the RIT-10, with a launch scheduled for 1995.

The UK-10 IPS consists of the T5 thruster, a propellant supply and monitoring equipment (PSME) and a power conditioning and control equipment (PCCE), as shown in the schematic in Figure 3. Also necessary on the host spacecraft are a propellant storage equipment (PSE) and a mounting arrangement on the external surface to carry the thruster; on ARTEMIS, the latter incorporates a gimbal system to allow the thrust vector to be directed as necessary to minimise the consumption of attitude control propellant.
The PSME, as indicated in Figure 3, provides three carefully controlled xenon flows to the hollow cathode, discharge chamber and neutraliser of the thruster. Control is achieved by admitting gas, from a 2 bar supply from the PSE, to a plenum chamber in each feed line, in response to signals from the PCCE. This is accomplished by the use of fast-response solenoid valves, which are currently qualified to 10^7 cycles, with a near-term objective of extending this to 5 \times 10^7. The engineering model of the PSME is shown in Figure 4. It has a mass of under 2 kg and dimensions of 215 \times 144 \times 300 \text{ mm}. The flight model will be slightly more compact, and is predicted to have a mass of 1.6 kg.

The PCCE\textsuperscript{16} contains all the power supplies necessary to operate the thruster and PSME, together with a microprocessor-based control system, which starts and stops the thruster automatically, and regulates its operation during steady-state running. It can handle input voltages ranging from 26.5 to 42.5 V with high efficiency, and the overall power consumption reaches 750 W at 25 mN thrust. The efficiency is about 88% and the mass of the flight model is estimated to be between 9 kg and 10 kg. The microprocessor used is the MAS-281 silicon-on-sapphire device, with ADA programming.

At the time of writing, a breadboard version of the PCCE is being integrated with the thruster and PSME, prior to commencing a life-test programme. The development of an engineering qualification model (EQM) is under way, with the aim of achieving full qualification for the ARTEMIS mission by late 1992. A sketch of the EQM is shown in Figure 5, with key dimensions.

As a separate exercise, a modular PCCE is also being developed by RAE\textsuperscript{17}. This is intended
specifically for 10 mN thrust and a 50 V input, with the aim of flying an experimental IPS in the mid-1990s on an RAE satellite.

It has already been mentioned that the T5 thruster is extremely versatile, with the capability of covering a very wide thrust range at consistently high efficiency\textsuperscript{13}. This achievement results from the decision to retain the rather complex control system, originally developed when mercury was employed as the propellant\textsuperscript{12}. This system also permits high propellant utilisation efficiency to be retained over life, despite the degradation which inevitably accompanies long periods of operation. In addition, it is possible to control the rate of this degradation to some extent, at least insofar as the cathode and neutraliser are concerned.

This control system\textsuperscript{18}, shown schematically in Figure 6, makes use of the three separate xenon flows to the thruster and the variable magnetic field. In principle, the magnetic field is used to control the propellant utilisation efficiency, via the energy of the primary electrons in the discharge chamber. The beam current, and therefore the thrust, is controlled by means of the gas flow to the discharge chamber, and the cathode and neutraliser flows serve to control their operating conditions, and thus their rates of degradation.

![Figure 6. Schematic of UK-10 IPS control system](image)

In addition to the development of the components of the IPS, considerable effort has been devoted to examining the interaction between the thruster and a host spacecraft, also to the life-limiting factors\textsuperscript{19}. This work was based on the measurements made with mercury propellant, also the extensive life-testing carried out at that time. Much of this is soon to be repeated with xenon; it has already been shown that the beam divergence is below 10\(^\circ\) half-angle, as was the case with mercury, and that the doubly-charged content of the beam has decreased\textsuperscript{8,19}.

The performance of the thruster has recently been increased significantly by incorporating minor changes to the inner polepiece/baffle region (Figure 1). A typical performance map is shown in Figure 7. In the analysis which follows, the operating point designated "P" has been selected; data appropriate to this point are summarised in Table 2. Although nominally operated at 25 mN, the thrust, corrected for the presence doubly-charged xenon ions in the beam, was 27 mN at point P.

![Figure 7. Typical T5 thruster performance map under nominal 25 mN thrust conditions](image)

**Table 2**

<table>
<thead>
<tr>
<th>PERFORMANCE OF T5 ION THRUSTER AT 25 mN NOMINAL THRUST</th>
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</thead>
<tbody>
<tr>
<td>DATA APPLICABLE TO POINT P IN FIGURE 7</td>
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<tr>
<td>----------------------------------------------------------</td>
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<tr>
<td>Net beam accelerating potential (V)</td>
</tr>
<tr>
<td>Exhaust velocity (km/s)</td>
</tr>
<tr>
<td>Total flow rate to thruster (scc/min)*</td>
</tr>
<tr>
<td>Beam current (mA)</td>
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<tr>
<td>Discharge current (A)</td>
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<tr>
<td>$\Delta V$ (Anode-keeper voltage) (V)</td>
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<tr>
<td>Anode voltage (V)</td>
</tr>
<tr>
<td>Doubly-charged ion content of beam (%)</td>
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<tr>
<td>Data corrected for doubly-charged ions:</td>
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<tr>
<td>Thrust (mN)</td>
</tr>
<tr>
<td>Propellant utilisation efficiency (%)</td>
</tr>
<tr>
<td>Specific impulse (s)</td>
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<tr>
<td>Beam power (W)</td>
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<tr>
<td>Discharge power (including keeper) (W)</td>
</tr>
<tr>
<td>Additional power (W)</td>
</tr>
<tr>
<td>Electrical efficiency (%)</td>
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<tr>
<td>Total efficiency (%)</td>
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<tr>
<td>Power/thrust (W/mN)</td>
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* 6% additional flow required for neutraliser

### 3 OPTIMISATION OF OPERATING CONDITIONS

For any given mission, it is likely that the effectiveness of the IPS can be enhanced by operating under optimised conditions. A process by which those conditions may be selected is described below, with reference to the UK-10 IPS and the Intelsat VII spacecraft\textsuperscript{3}. It is assumed in this that a thrust is
applied in the north-south direction symmetrically about the nodes of the orbit, and that this is done on every day for which there is no eclipse.

A further assumption, in line with current thinking, is that the power for the IPS is provided by the spacecraft batteries. These need to be of substantial capacity to provide full payload operation during eclipse, yet they are employed only infrequently for this purpose. If the IPS is used only in eclipse-free periods of the year, the only penalty in this strategy is an increase in the number of battery charge/discharge cycles, perhaps by a factor of two or three. If this proves to be unacceptable, these additional cycles can be reduced in number very considerably by using excess solar array power to operate the IPS in times of low payload demand, especially near the beginning of life when radiation damage to the solar cells is small.

If thrusting takes place over an angle ±β about each node, the effective thrust $T_{ns}$ needed in the north-south direction and velocity increment $ΔV$ required per year can be accurately calculated. However, if it is assumed that the lunar-solar perturbing force is a constant, $T_{ns}$ may be found with sufficient accuracy for most purposes from the approximate relationship:

$$T_{ns} = \frac{πKMv_0}{2 \sin β},$$

where $K$ is the rate of drift of inclination in rad/s, $v_0 = 3.074$ km/s is the orbital velocity, and $M$ is the mass of the spacecraft. The factor $\sin β$ in this equation reflects the decrease of thrusting efficiency $\eta_T$ as distance from the nodes increases, where $\eta_T$ is defined as the ratio $\sin β/β$. Figure 8 shows the variation of $\eta_T$ with $β$, from which it will be seen that the degradation of efficiency is not greater than 5%, provided that $β$ does not exceed about $32°$. The minimum value of 63.7% is reached at 90%.

$$\Delta V_{MN} = T_{ns}τ,$$ (2)

where $N$ is the mission duration in years.

In most envisaged thruster installations, an additional inefficiency occurs, because the geometry of the majority of three-axis stabilised spacecraft prevents thrusting along the N-S axis. To avoid direct ion impingement on the solar arrays, the thrusters are mounted at an angle $φ$ to the N-S direction, reducing the effective thrust by the factor $\cos φ$. Thus the overall thrust efficiency becomes

$$\eta_T' = \cos φ \frac{\sin β}{β}.$$ (3)

Figure 8 includes a plot of $\eta_T'$ against $β$ for typical values of $φ$, $30°$ and $40°$.

3.1 Maximum thrust level

Although the thrusting inefficiency discussed above implies that a high value of $T_{ns}$ should be selected, other criteria must also be taken into account. For example, the use of a large thrust necessitates the provision of increased power and, for short thrusting periods, proportionally more propellant is wasted.

Detailed theory shows that this parameter varies significantly with the year of launch. Figure 9 depicts $K$ as a function of time for the years 1990 to 2008, and also shows the equivalent values of $ΔV$ for small $β (τT ≈ 1)$. It will be seen that $ΔV$ varies from 41 to 51 m/s in a cyclic manner; in all subsequent analyses, values near the upper limit have been assumed.

![Figure 9. Annual inclination drift and velocity increment as a function of time](image-url)
during start-up and shut-down. With regard to the latter problem, it is likely that the shortest times for these operational phases will be about 5 and 3 minutes respectively, so it is reasonable to assume that the equivalent of 2 to 3 minutes of propellant flow will be wasted on each thrusting cycle; thermal and PSME time constants will not allow this loss to be significantly reduced. This represents a 3 to 5% reduction of mass utilisation efficiency over a 1 hour cycle, or 1.5 to 2.5% over 2 hours. The former is probably acceptable and effectively fixes the upper usable thrust level for operation at fixed intervals of time or, alternatively, defines those intervals. However, the importance of these factors cannot be fully established until mission parameters are more precisely defined. For a cycle of 1 hour, $\beta = 7.5^\circ$ and, for $M = 1425$ kg, $T_{ns} = 27$ mN if thrusting occurs on every possible occasion. Values are higher with less frequent operation.

3.2 Minimum thrust level

The theoretical minimum thrust level $T_{ns,m}$ can be derived by substituting $\beta = 90^\circ$ into equation (1), thus assuming that thrusting takes place continuously around each orbit. For $K = 0.932$ ($\Delta V = 50$ m/s), $T_{ns,m} = 3.55$ mN for $M = 1425$ kg. If thrusting has to be at an angle $\phi$ to the north-south direction, this value will be increased to 4.1 mN for $\phi = 30^\circ$.

Although the above thrusts represent absolute minimum values, in practice other considerations will dictate the choice of higher levels. In particular, the total operating time of any thruster must be kept within reasonable limits, so that its proven lifetime is not exceeded. Owing to the need to restrict development and space qualification costs, this proven lifetime probably cannot be as great as would be dictated solely by physical degradation. Thus, although life-tests have shown thatthrusters can easily surpass the 5000 hour point and, indeed, that lifetimes considerably in excess of 10000 hours are feasible, it is likely that considerations of cost, at least in initial applications, will limit the qualified life-time to around 5000 hours. At this level, multiple life-tests are possible, without requiring excessive resources or time.

It is often accepted that a suitable thruster installation for station-keeping consists of two pairs of thrusters, one on the north-facing side of the spacecraft, the other on the south-facing side. This scheme provides a good measure of redundancy, together with flexibility of operation. Under normal circumstances, the south-pointing thrusters operate together about one node, whilst the north-pointing ones operate about the other. Consequently, $T_{ns}$ is produced by two thrusters, and each of the four installed thrusters has to operate for a total time $\tau/2$. However, if one thruster of a pair fails, that pair can no longer be used if serious east/west drifts are to be avoided. Thus the qualification level of the IPS must be equated to the operating time required if such a failure occurred at the beginning of the mission. Therefore, with a feasible qualified lifetime of about 5000 hours, the minimum acceptable thrust level should be such that $\tau$ is also 5000 hours.

It should be pointed out that an alternative mounting scheme, with the thrusters in the plane containing the spacecraft-Earth radius vector (Figure 10), does not suffer from this disadvantage to the same extent, and has therefore been adopted for the ARTEMIS mission. In this case, the failure of one thruster of a pair does not lead to unacceptable orbital perturbations, so the degree of redundancy is much enhanced. In fact, for ARTEMIS, both thrusters of a pair are mounted together on the anti-Earth face of the spacecraft, with the perturbations caused by firing at one node being cancelled by those at the opposite node.

![Figure 10. Alternative thrusting configurations](image-url)
system mass is not a linear function of thrust level, especially if the mass of the power source must be included. At low thrust, the inefficiency due to large values of $\beta$ increases the amount of propellant that must be carried, and the fixed mass components, such as valves, mounting structure and insulators, become of increasing importance. Conversely, at high thrust, the mass of the power conditioner and of the power source tends to dominate.

PCCE $M_{pc}$ can be divided into a constant part $b_1$, together with a variable part dependent on power output and thus on thrust and exhaust velocity. If $P_h$ is the power consumed by those components not changing appreciably with $F$ or $v$, such as the PSME and solenoids,

$$M_{pc} = b_1 + b_2 \left[ F \left( \frac{Y}{2} + \frac{c}{v} \right) + P_h \right],$$

where $b_2$ is the variable mass per unit power output and $c$ is the product of discharge chamber efficiency in W/A and the ionic charge to mass ratio.

From conservation of momentum, the propellant used by one thruster during the mission is $F t / \tau_{lm} v$, assuming that two south-pointing and two north-pointing thruster pairs are operated alternately (Figure 10). Here, $\tau_{lm}$ is the overall mass utilization efficiency, corrected for doubly-charged ions and propellant flow through the neutraliser. If $\tau_M$ is the total mission time, $\tau = 2 \tau_M / \pi$ and, using equation (1), the mass of the complete propulsion system, excluding the PSE and PSMEs, becomes

$$M_{ps} = \frac{2 (1 + d) \tau M_{ns}}{\pi \tau_{lm} v} \sin^{-1} \left( \frac{T_{nsm}}{T_{ns}} \right),$$

excluding any correction to account for the mounting angle $\phi$ of the thrusters. For simplicity it is assumed that thrusting takes place each day of the year and that the tank mass can be represented by a fraction $d$ of the propellant mass.

Combining the above results,

$$M_S = \frac{4 M_t + 4 M_{pc} + M_e + M_{ps}}{\cos \phi}, \quad (4)$$

where $M_e$ is the total mass of the PSE and all four PSMEs. Recalling that $T_{ns} = 2 F \cos \phi$, equation (4) can be expanded to give:

$$M_S = M_e + 4 (a_1 + b_1 + b_2 P_h)$$

$$+ \frac{4 F}{v} \left[ a_2 + \frac{b_2 v^2}{2} + b_2 c \right] + \frac{2 \tau (1 + d)}{\pi \tau_{lm}} \sin^{-1} \left( \frac{T_{nsm}}{T_{ns}} \right). \quad (5)$$

Equation (5) can be used to evaluate $M_S$ as a function of $F$ for a fixed exhaust velocity. The constants may be derived from the parameters of a well-developed thruster system, preferable in the middle of the thrust range of interest. For many thrusters, the value of $\tau_{lm}$ to be used would have to change with thrust, but this is not necessary with T5, because the four control loops enable the performance to be maintained at the same high value over the thrust range of interest$^{13,18}$. 

![Figure 11. Total thrust, north-south thrust component, and total thrusting time as functions of angle of thrusting about the nodes](image1)

![Figure 12. Installed thrust as a function of thrusting time](image2)
For the UK-10 IPS, the following values were adopted for subsequent analysis:

\[
\begin{align*}
    a_1 & = 0.9 \text{ kg} \\
    a_2 & = 2.1 \times 10^6 \text{ s} \\
    d & = 0.1 \\
    F_h & = 18 \text{ W} \\
    c & = 2.038 \times 10^8 \text{ Ws/kg} \\
    M_e & = 7.4 \text{ kg} .
\end{align*}
\]

However, the values of \( b_1 \) and \( b_2 \) are strongly dependent upon assumptions made about the design and packaging of the PCCE. Although the mass of the 25 mN PCCE is known, the trend that might be followed with change of \( F \) is not clear. However, the plot of mass against power level shown in Figure 13, using historical and present-day data, indicates that reasonable values are \( b_1 = 4 \text{ kg} \) and \( b_2 = 8.2 \times 10^{-3} \text{ kg/W} \).

Results are presented in Figure 14 for \( \phi = 30^\circ \) and mission durations of 10, 15 and 20 years. Contrary to expectations, optimum values of M_r were not found, the trend observed being a steady decrease in this parameter with reducing \( F \). However, if the analysis had been continued to very low values of \( F \), the large thrusting times corresponding to increasing \( \beta \) would have caused an upward turn in the curves, as indicated by the dotted line in Figure 14. On the basis of these results, it can be stated that the optimum thrust level is determined by the qualified thruster life, as indicated by the hatched region, not by minima in the plotted curves.

### 3.4 Selection of exhaust velocity

If equation (5) is differentiated with respect of \( v \), assuming constant \( F \), an optimum value of \( v \) can be derived by putting \( (dM_r/dv) = 0 \). If the mass of the power source is excluded, this gives

\[
\sqrt{v} = \frac{2}{b_2} \left[ a_2 + b_2 c + \frac{\tau M(1+\phi)}{\pi n_m} \sin^{-1} \left( \frac{T_{ns m}}{T_{ns}} \right) \right].
\]

Because it contains no effective limit on power consumption, apart from the increase of power conditioner mass with \( v \), equation (6) yields values of \( v \) which are unrealistically high. Consequently, an additional constraint must be introduced; this is the total power that can be drawn from the spacecraft's bus or battery.

If the total power consumed by the thruster is \( P_T \),

\[
P_T = F(\frac{v}{2} + \frac{c}{v}) + P_h.
\]
This relationship has been solved for \( v \) as a function of \( F \) at fixed values of \( P_T \) between 300 W and 1000 W. The results, plotted in Figure 15, define a range of thrusts accessible at a given power level. This range is very restricted at low power, especially if technological or operational considerations prohibit the use of beam accelerating potentials greater than a certain value. The lower limit at each power level, 25 to 30 km/s, represents the point at which the thrust can just be supplied; this is no longer possible at lower velocities. The energy consumption per operating period under a particular set of conditions may be derived from Figure 15 if \( t \) is known.

It can be seen from Figure 15 that, if a 25 mN thrust level is selected, \( v \) should be between 36 and 46 km/s if 600 to 700 W are available. Thus, under such circumstances, 40 km/s is a reasonable compromise. This corresponds to a beam-accelerating potential of 1.1 kV, the value selected for qualification of the UK-10 system and for flight on ARTEMIS.

4. THE BENEFITS OF ION PROPULSION

The high exhaust velocities provided by using an IPS for NSSK result in very substantial propellant mass savings, when compared to chemical propulsion systems. Assuming operation at point P in Figure 7, the potential benefits are indicated in Figure 16 for a spacecraft of 1500 kg dry mass, which equates to Intelsat VII with an allowance of 75 kg for the IPS, including mountings, cables and gimbals for each thruster. The mass saving becomes significant for lifetimes exceeding about 5 years, reaching over 170 kg at 10 years and 370 kg at 20 years. These values are considerably increased when the need to raise the perigee of the initial geostationary transfer orbit is taken into account, because the propellant required for this task is also reduced owing to the fall in mass in the operational configuration.

The possible additional gains to be made by reducing the propellant required for orbit perigee-raising are indicated in Table 3, which lists the masses required for each propulsion function for the nominal Intelsat VII mission. The apogee motor consumes 1560 kg, which is more than the dry mass of the spacecraft. A detailed analysis suggests that the 170 kg mentioned above at 10 years would be increased to 260 kg for an Ariane launch from Kourou, or 290 kg if the Atlas Centaur is used from the Eastern Test Range (ETR).

In order to make further substantial gains, the IPS could, in principle, be used for the perigee-raising task, saving possibly as much as 1200 to 1300 kg. However, this process would occupy a long period of time, probably many months, and would necessitate a change in deployment philosophy.

Figure 16. NSSK propulsion system mass as a function of mission duration for chemical and ion propulsion systems

<table>
<thead>
<tr>
<th>Function</th>
<th>Assumed SI</th>
<th>Propellant mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>NSSK</td>
<td>289</td>
<td>446</td>
</tr>
<tr>
<td>EWSK</td>
<td>285</td>
<td>21</td>
</tr>
<tr>
<td>Attitude control</td>
<td>&lt;285</td>
<td>19</td>
</tr>
<tr>
<td>Station changing</td>
<td>285</td>
<td>24</td>
</tr>
<tr>
<td>De-orbit</td>
<td>285</td>
<td>5</td>
</tr>
<tr>
<td>Raising perigee</td>
<td>311</td>
<td>1560</td>
</tr>
<tr>
<td>Circularisation</td>
<td>311</td>
<td>66</td>
</tr>
<tr>
<td>Residual propellant</td>
<td>–</td>
<td>19</td>
</tr>
</tbody>
</table>

The above analysis is specifically applicable to the baseline Intelsat VII mission. It is clear that the benefits of using an IPS for NSSK become greater as spacecraft mass is increased. This is illustrated in Table 4, which is derived from Ref 4 and concerns spacecraft with beginning of life (BOL) masses in GEO of 1510, 1812 and 2114 kg, and missions of 10, 15 and 20 years duration. As can be seen, launch mass saving range up to 700 kg.

4.1 Economic implications

If the mass savings from the detailed analysis are related to Ariane launch costs of $30000 per kg, a reduction of $7.8M should be possible for a 10 year mission and $13.1M for 15 years. Slightly higher figures would apply for launches from ETR.
Table 4

REDUCTION IN PROPULSION SUBSYSTEM MASS DUE TO APPLICATION OF ION PROPULSION FOR NSSK

<table>
<thead>
<tr>
<th></th>
<th>Mass (kg)</th>
<th>10</th>
<th>15</th>
<th>20</th>
</tr>
</thead>
<tbody>
<tr>
<td>Spacecraft GTO mass</td>
<td>2500</td>
<td>3000</td>
<td>3500</td>
<td>2500</td>
</tr>
<tr>
<td>(Ariane launch)</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Spacecraft BOL mass</td>
<td>1510</td>
<td>1812</td>
<td>2114</td>
<td>1510</td>
</tr>
<tr>
<td>mass (kg)</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Bipropellant mass for</td>
<td>213</td>
<td>256</td>
<td>298</td>
<td>324</td>
</tr>
<tr>
<td>NSSK (kg)</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Xenon mass (kg)</td>
<td>25</td>
<td>30</td>
<td>35</td>
<td>39</td>
</tr>
<tr>
<td>IPS dry mass (kg)</td>
<td>75</td>
<td>75</td>
<td>75</td>
<td>75</td>
</tr>
<tr>
<td>Thruster firing time/day (h)</td>
<td>1.43</td>
<td>1.72</td>
<td>2.00</td>
<td>1.43</td>
</tr>
<tr>
<td>Energy required (Wh)</td>
<td>1645</td>
<td>1978</td>
<td>2300</td>
<td>1645</td>
</tr>
<tr>
<td>Reduction in mass saving</td>
<td>113</td>
<td>151</td>
<td>188</td>
<td>210</td>
</tr>
<tr>
<td>propulsion subsystem (kg)</td>
<td>159</td>
<td>217</td>
<td>278</td>
<td>315</td>
</tr>
</tbody>
</table>

Of course the mass savings can be used, wholly or in part, to increase the lifetime of the spacecraft. This would result in an increase in revenue and further savings from the delay in the procurement and deployment of a replacement. Assuming a revenue-earning capability of $2.5M per transponder per year, the lifetime extension of a 36 transponder spacecraft by 5 years immediately offers a large earning potential, $450M. Further, the investment made in the replacement of the spacecraft is effectively reduced, in simplistic terms, by two-thirds, and an entire generation of spacecraft could be eliminated during a period of 30 years.

The spacecraft performance enhancement made possible by the use of an IPS for station-keeping is difficult to estimate. However, if the ratio of numbers of transponders to baseline dry mass is related to the 260 kg saving for an Ariane launch from Kourou, an extra six transponders can probably be carried, together with appropriate mass increases for all other relevant subsystems. At $2.5M per year for each one, the economic benefits of these additional transponders are considerable, reaching $150M at 10 years.

It should be pointed out that the advantages of using an IPS for NSSK are greatest when the spacecraft is designed from the outset with this form of propulsion. In the case of a retrofit, in which an IPS is installed in an existing spacecraft, the benefits become much harder to estimate and are likely to be smaller, although still significant. In particular, the payload is probably then not optimised to the overall spacecraft design. Moreover, the impact on factors such as radiator area, power generation, overall layout and dynamic behaviour may necessitate a considerable amount of requalification activity. Nevertheless, such a retrofit is entirely feasible, as has been shown by studies of the ARTEMIS spacecraft.

5. CONCLUSIONS

It can be concluded that considerable economic benefits result from efforts to reduce the mass of a large communications satellite. These benefits can appear as a lower launch cost, or as increased revenue, the latter from a higher capacity payload or from a longer operational lifetime. If the option of a longer lifetime is selected, further financial savings can be made, because replacement satellites will be required less frequently.

In cases where accurate station-keeping is needed, such as Intelsat VII, a very promising method of reducing mass substantially is to replace the chemical propulsion system used for NSSK by ion thrusters. These have an exhaust velocity, or specific impulse, at least an order of magnitude greater than chemical thrusters, allowing the propellant mass consumed to be reduced by the same ratio. Since the mass in GEO is then reduced by more than 200 kg, the propellant required for circularisation of the initial transfer orbit is also less, giving further savings. For Intelsat VII, the net mass reductions over 10 years are conservatively estimated to be 260 kg for an Ariane 4 launch from Kourou and 291 kg for an Atlas Centaur from the Eastern Test Range. Depending upon how these savings are utilised, the increased revenue could amount to several hundred million dollars over the life of the spacecraft.

Several ion thruster systems are under development specifically for the NSSK mission. Of these, the UK-10 system offers high performance over a very wide thrust range, together with a mass utilisation efficiency which is maintained at well over 80% for the complete mission. This, together with active control of other life-limiting factors, ensures that thruster/spacecraft interactions are well-defined throughout the mission, and can be planned for in advance with confidence.

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