# N-S Stationkeeping by 10-cm Ion Thruster

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**An analysis is presented of the application of the U.K. 10-cm mercury ion thruster to the North-South stationkeeping mission, with particular reference to the proposed European Television Broadcast Satellite of 750-kg mass and 7-year life. It is shown that the nominal thrust of 10 mN is close to optimum for missions of this type. Various potential problem areas, such as solar array contamination, attaining an acceptable reliability, and thermal control of the installation, are discussed. It is demonstrated that the thruster is fully compatible with this and other similar applications.**

## Introduction

IT is widely acknowledged that very large benefits would result from replacing hydrazine North-South stationkeeping (NSSK) thrusters on geostationary communications satellites with ion thruster systems. The much higher specific impulse then available would allow substantial quantities of propellant to be eliminated,[1,2] thus enabling considerably greater payloads to be carried. Such a stationkeeping mission has recently been defined by the European Space Technology Centre (ESTEC). This envisages a 750-kg Television Broadcast Spacecraft (TVBS) having a life of 7 yr, with power for the ion thruster system being provided by the excess capability of a 5-kW (beginning-of-life) solar array.

A detailed analysis has been carried out which shows that the U.K. T4A/T5 ion thrusters[3-5] are well suited to this and similar missions. The nominal thrust of 10 mN represents an excellent compromise between the high power demand associated with larger values of thrust and the long thruster lifetimes necessitated by the choice of lower values. Different optimum thrust levels may, of course, be deduced for other missions, but they should not present difficulties, because the thrusters may be throttled from 7 to 17 mN without appreciable performance degradation.[5]

An assessment has also been made of the thrusting strategies that might be employed for the TVBS mission and of the problems that might be encountered in integrating a thruster system with a typical communications spacecraft. It is concluded that none of the areas considered are likely to give rise to serious difficulties.

## Selection of Thrust Level

The gradually increasing oscillatory North-South drift of a geostationary spacecraft[6] can be corrected by thrusting in a North or South direction around the orbital nodes. However, an ion propulsion system must operate for an appreciable time to provide the necessary impulse, and the efficiency $\eta_T$ of this process decreases with increasing angle $\beta$ from the nodes. Consequently, since reducing the thrust level decreases the mass of the propulsion system while also degrading its efficiency, a compromise must be reached between thrust and operating time.

In arriving at such a compromise, criteria other than system mass and thrusting efficiency 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 during startup and shutdown. With regard to the latter point, it is likely that the shortest times for these parts of the operational cycle will be about 5 and 3 min, respectively, and so it is reasonable to assume that the equivalent of 2 to 3 min of normal propellant flow will be wasted during each thrusting period. This represents a 1.5 to 2.5% reduction of mass utilization efficiency $\eta_m$ over a 2-h cycle,[7] which is probably acceptable and effectively fixes the upper thrust level.

In most envisaged thruster installations, an additional inefficiency occurs, because, to avoid ion beam impingement on the solar arrays, the thrusters are mounted at an angle $\phi$ to the N-S direction. This reduces efficiency by a factor $\cos\phi$ to $\eta_T'$; the installed North-acting or South-acting thrust $T$ is also reduced by the same factor to $T_{NS}$.

The minimum North-South thrust $T_{NS\,min}$ that can be used may be derived by assuming that thrusting takes place continuously around each orbit.. For an annual velocity increment $\Delta V = 50$ m/s, $T_{NS\,min} = 1.87$ mN in the case of the TVBS mission, and, with $\phi = 30$ deg, $T = 2.16$ mN.

In practice, the total operating time of any thruster must be kept within its proven lifetime. Because of the need to restrict development and space qualification costs, this lifetime probably cannot initially be as great as would be dictated solely by physical degradation. Thus, although it has been shown that thrusters can easily surpass 5000 h[8,9] and that lifetimes considerably in excess of 10,000 h are feasible,[10] it is likely that cost will limit the qualified lifetime to around 5000 h, at least for early applications.

It will be shown later that a preferable thruster installation scheme consists of one pair of thrusters on the North-facing side of the spacecraft, with a second pair on the South-facing side. Under normal circumstances, the South-pointing thrusters operate together about one node, whereas the North-pointing ones operate about the other. Consequently, each of the four installed thrusters has to operate for one-half of the total running time $\tau$. For this configuration, a relationship between thrust $\tau$ and angle of thrusting on each side of the nodes, $\beta$, was derived by equating the momentum transferred to the spacecraft to the total impulse provided by the propulsion system. As shown in Fig. 1, it was found that operating times become prohibitively long at only moderate values of $\beta$.

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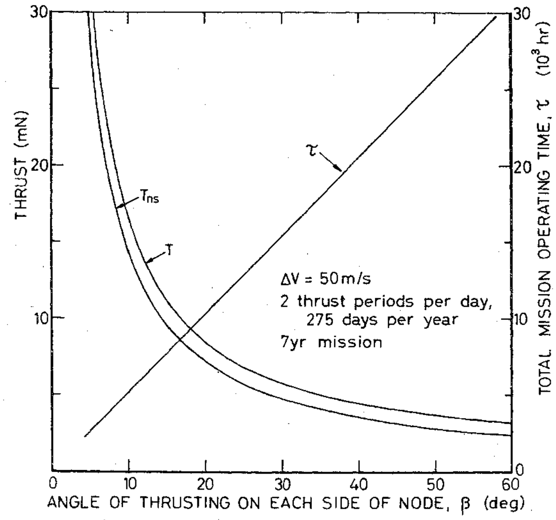


**Fig. 1 Total installed thrust, North-South thrust component, and total thrusting time as functions of angle of thrusting about the nodes.**

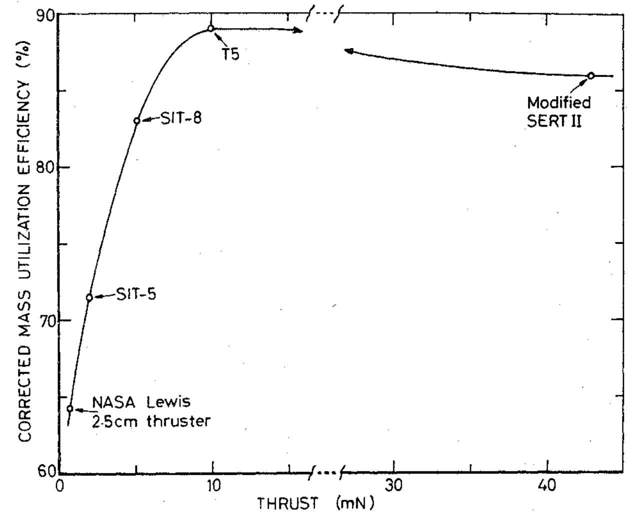


**Fig. 2 Mass utilization efficiency as a function of thrust for Kaufman thrusters.**

## Optimum Thrust Level

It has been shown[11,12] that an optimum thrust exists for any particular mission, partly because total thruster system mass $M_s$ is not a linear function of thrust level. 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 insultators, become of increasing importance. Conversely, at high thrust, the mass of the power conditioner and of the power source tends to dominate.

The analysis reported here is based on the approach adopted by Day and Palmer.[11] These authors derived an expression for $M_s$, including the propellant for the whole mission, as a function of thrust and exhaust velocity, assuming that operation occurs about both nodes. They showed that the mass $M_t$ of a thruster can be related to thrust $F$ and exhaust velocity $v$ by $M_t = a_l + a_2 F/v$, where $a_l$ is the mass of those components not dependent on thrust, such as vaporizers, cathodes, and valves, and $a_2$ is a constant. Similarly, the mass of a power conditioner $M_{pc}$ can be divided into a constant part $b_l$ together with a variable part dependent on power output, the constant of proportionality being $b_2$.

From conservation of momentum, the propellant used by one thruster during the mission is $F\tau/\eta_m v$, where $\eta_m$ is the mass utilization efficiency, corrected for doubly charged ions and propellant flow through the neutralizer. The total mass of the propellant system, excluding any correction to account for the mounting angle $\phi$ of the thrusters, becomes

$$M_{ps} = 4(1+d)F\tau/\eta_m v$$

where the tank mass is represented by a fraction $d$ of the propellant mass. The system mass is then

$$M_s = 4M_t + 4M_{pc} + M_e + M_{ps}/\cos\phi \tag{1}$$

where $M_e$ is the mass of the electrical supply. In the present case, it is assumed that excess solar array power is available, so that $M_e = 0$. Recalling that $T_{\rm NS} = 2F\cos\phi$, Eq. (1) can be expanded to give

$$M_s = 4(a_l + b_l + b_2 P_h) + \frac{4F}{v}\left[a_2 + \frac{b_2 v^2}{2} + b_2 \epsilon + \frac{\tau_M(1+d)}{\pi \eta_m}\sin^{-1}\left(\frac{T_{\rm NS\,min}}{T_{\rm NS}}\right)\right] \tag{2}$$

where $P_h$ is the power consumed by those components, such as vaporizers, not strongly dependent on $F$ or $v$, $\epsilon$ is the

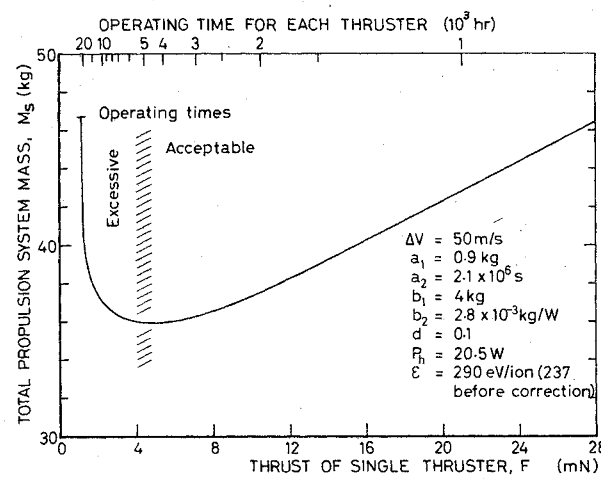


**Fig. 3 Propulsion system mass, including propellant, as a function of thrust.**

product of the discharge chamber ($eV$/ion) and the ionic charge-to-mass ratio, and $\tau_M$ is the total mission duration.

Equation (2) was used to evaluate $M_s$ as a function of $F$ for fixed $v$, with constants derived from the parameters of the T5 thruster system. Contrary to previous practice,[11] it did not appear reasonable to assume $\eta_m$ to be ? constant. Consequently, published data were employed to produce the graph of $\eta_m$ as a function of $F$ shown in Fig. 2. These data are summarized in Table 1, where $I_B$ and $V_B$ are the beam current and voltage, and $V_D$ is the discharge voltage.

A comment should be made concerning the SERT II values. Although this thruster was designed to work with $V_B = 3.3$ kV, $I_B = 250$ mA, and $F = 29$ mN, subsequent studies[17] have shown that $I_B$ and $F$ can be increased substantially, while decreasing $V_B$, by careful grid redesign and other modifications. Using high-perveance grids, $\eta_m \sim 91\%$ at 300 eV/ion was achieved, with $V_B = 1$kV. These values have been assumed in compiling Table 1, with an allowance of 0.03 mg/s for the neutralizer flow rate.

The results, shown in Fig. 3, indicate that the minimum value of $M_s$ occurs at about 5 mN and that the operating time criterion dictates that $F$ should be greater than 4.5 mN. Thus, a thrust of 10 mN appears to be larger than necessary, but the mass penalty is a mere 1.3 kg ($M_s = 37.3$ kg), and the required lifetime is reduced to a very attractive 2100 h. Thus, for a mass increase of 1.3 kg, the cost of life and qualification testing can be drastically reduced.

**Table 1 Published and derived thruster data used in producing Fig. 2**

| Thruster | $V_B$, kV | $I_B$, mA | $V_D$, V | Uncorrected $F$, mN | Uncorrected $\eta_m$ | Corrected[b] $F$, mN | Corrected[b] $\eta_m$ | eV/ion | Ref. |
|---|---|---|---|---|---|---|---|---|---|
| NASA Lewis 2.5-cm | 0.5 | 17 | 42 | 0.78 | 0.68[a] | 0.75 | 0.644 | 400 | 13 |
| SIT-5 | 1.2-1.6 | 25.6 | 45 | 2.1 | 0.76[a] | 2.02 | 0.715 | 465 | 14 |
| SIT-8 | 1.22 | 72 | 36 | 5.16 | 0.86[a] | 5.01 | 0.83 | 276 | 15 |
| T4A(T5) | 0.94 | 167 | 40 | 10.5 | 0.89(0.94) | 10.3 | 0.84(0.89) | 287 | .... |
| SERu II(mod.) | 1.0 | 664 | 36-37 | 42.8 | ~0.89[a] | 41.6 | ~0.86 | ~300 | 16,17 |

[a] Corrected for neutralizer flow.
[b] Corrected for doubly charged ions, using data from Ref. 15 where appropriate, for neutralizer flow, and for discharge chamber keeper power.

It can be concluded that a thrust of about 10 mN is a reasonable compromise between minimizing operating times and development costs while keeping close to the minimum of the curve of $M_s$ vs $F$. Moreover, this thrust appears to satisfy the requirements of a wide range of projected spacecraft; the range of applications is further increased by the excellent throttling capability of the T4A and T5 thrusters.[5]

This analysis can be extended to deduce an optimum exhaust velocity for a given mission. This may be done by differentiating Eq. (2) with respect to $v$, assuming constant $F$, and putting $(dM_s/dv) = 0$. If the power source is included, reasonable values of $M_e$ indicate that, at 10 mN, an exhaust velocity of about 30 km/s is close to optimum. However, the results vary strongly with the assumptions made concerning the power source, and so this must be accurately defined if exact answers are required. In particular, any limitations on power or energy consumption must be included.

## Thruster Installation

If a spacecraft is designed with the aim of optimizing the usefulness of an ion propulsion system in the NSSK role, thruster mounting configurations should be selected[11,18] which avoid contamination of the solar arrays by effluent from the thrusters and eliminate the inefficiency represented by the angle $\phi$. However, such configurations imply the use of novel geometries or the positioning of the thrusters on the ends of the arrays, presenting structural, operational, and dynamic problems.

It is more acceptable for the thrusters to be body-mounted. However, only in arrangements similar to that shown in Fig. 4 is it possible to thrust through the spacecraft's center of mass, minimizing the disturbance torques caused by thrust vector misalignments and thrust imbalance. In addition, thrust components not in the N-S direction can be cancelled by operating pairs of thrusters simultaneously, and redundancy is provided by employing two pairs of thrusters; the second pair can point in the opposite direction to the first, as shown, or in the same direction. The latter alternative, however, restricts thrusting to one node of each orbit.

Having selected the installation scheme, it remains to decide whether the thrusters should be mounted in the plane containing the spacecraft-Earth radius vector (radial mounting) or the plane normal to this, which contains the orbital velocity vector (tangential mounting). The main difference between these arrangements emerges when considering orbital perturbations due to thrust vector misalignments, thrust variations, and thruster failures.

In carrying out a conventional analysis of these perturbations for 10-mN thrusters, it was assumed that the maximum thrust vector misalignment is 3 deg and that the thruster control system will maintain a specified thrust level to within ±3%. Although mounting each thruster on a gimbal system could eliminate the first source of error, it can be shown that the mass penalty of doing this would usually cancel the benefits obtained.

The analysis showed that the average velocity increment per orbit due to both errors combined is about $4\times10^{-3}$ m/s. If this is tangential to the orbital path, the resulting perturbation of the semimajor axis will lead to a longitude drift rate of about 0.0014 deg/day, if the error is in the same sense on both sides of the orbit, but no eccentricity change. If the errors are of opposite sense on either side of the orbit, there will be no change in semimajor axis and hence zero drift rate. There will, however, be a small change of eccentricity which leads to a daily libration about the mean satellite longitude of around 0.0003 deg, which can be ignored. In the case where the velocity increment is purely radial, the perturbation of the semimajor axis and the eccentricity change are both negligible. It was concluded that orbital perturbations due to thruster errors and variations may need correction with tangential mounting but that they are negligible in the alternative installation scheme.

The analysis was repeated for the case where one thruster of a pair has failed. When the thrusters are mounted tangentially, there will be a considerable thrust imbalance in the E-W direction, which, for $\phi = 30$ deg, will result in a change of 1 km per day in the semimajor axis and a longitudinal drift rate of 0.014 deg per day, which is unacceptable. Conversely, with radial mounting, the coupling effects are much less severe for synchronous orbits with low eccentricity. These considerations suggest that it is preferable to mount the thrusters in the plane containing the Earth radius vector. This offers greater redundancy and a reduced consumption of E-W stationkeeping propellant.

## Thrusting Strategy

With an installation of four 10-mN thrusters, each must run for under 2200 h to complete the TVBS mission; this is a modest aim, considering the proven capabilities of Kaufman thrusters.[10] A thrusting cycle time of 2 to 4 h is preferable, and this results from operating about each node each three days (2.52 h, 852 starts) or each four days (3.36 h, 639 starts). The thrusting efficiency $\eta_T$ is still acceptable, at 0.982 and 0.968, and the relatively low number of starts should enhance reliability and lifetime.

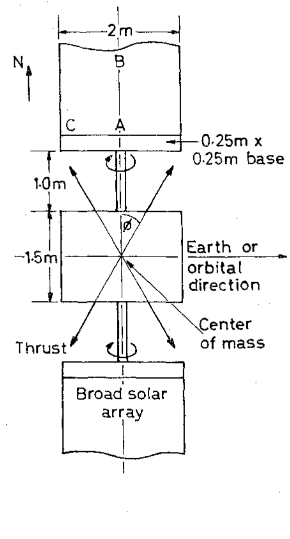


Fig. 4 Proposed TVBS thruster installation scheme.

#### Interactions with the Solar Arrays

Examination of Fig. 4 will show that, in the case of a broad solar array, there will always be ion beam impingement on the array when it lies in or near the mounting plane of the thrusters. This may not occur with a narrow array, and so the operating philosophy discussed below may not be applicable to that case. The degree of impingement will depend on the geometrical configuration and on the ion beam divergence. For the U.K. 10-cm thrusters,[4] the semiangle containing 95% of $I_B$ is less than 11 deg. Unfortunately, the remaining 5% of the ions cannot be disregarded, but, thus far, their distribution has not been accurately determined. To allow for their presence, it is assumed here that they are contained within a conical surface of 20 deg half-angle originating from the edge of the thruster's grid system.

It has been assumed so far that stationkeeping involves operation of the thrusters each three or four days throughout the year. However, at certain times, the positions of the Sun, Earth, and satellite are such that the solar arrays must rotate through the plane containing the thrusters. If direct ion impingement is to be avoided, thruster operation must not occur as the array passes through an angle $\pm\theta$ about this plane. With $\phi=30$ deg, the dimensions in Fig. 4, and an exhaust plume angle of $\pm 20$ deg, the interference zone occupies an angle of $\pm 32$ deg (Fig. 5).

This does not prevent stationkeeping, however, because it is possible to overcorrect for the North-South drift prior to the array entering the interference zone. Thruster operation then is suspended while the array rotates through $2\theta$, and the spacecraft drifts increasingly in the N-S direction. After this "deadband" has been crossed, overcorrection restores the situation. It is common to specify an NSSK accuracy of $\pm 0.1$ deg; the maximum allowed deadband is then equivalent to about 77 days.

The situation is illustrated in Fig. 6. To a first approximation, NSSK occurs as the spacecraft crosses arcs JK and LM, which are centered about 90 and 270-deg right ascension relative to Aries. However, because of lunar perturbations, the maneuver execution points will vary by about $\pm 9$ deg, and the larger arcs EF and GH must be allowed for thrusting. With the radial mounting configuration, the solar arrays become aligned with the plane of installation during thrusting when the sun vector lies within the angle $2\theta$ relative to Aries. Thus thrusting is not permissible when the Sun vector is within arcs AB and CD. For the tangential configuration, the forbidden regions are rotated through 90 deg.

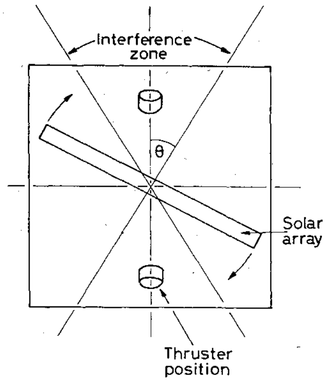


**Fig. 5 View of North or South face of spacecraft, showing inteference zones.**

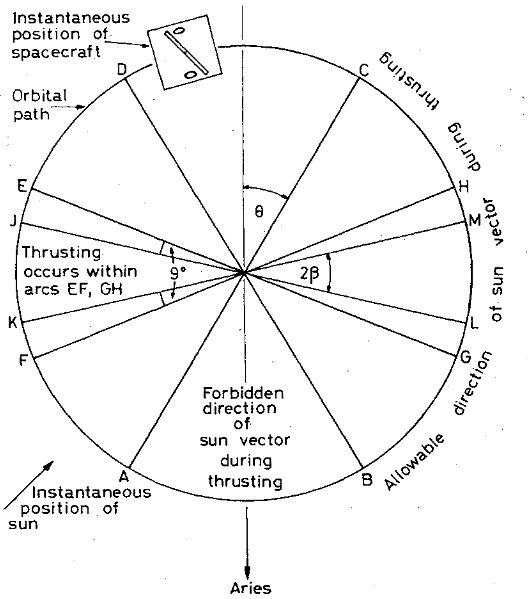


**Fig. 6 Schematic of thrusting strategy avoiding ion beam impingement on solar arrays.**

#### Interactions with the Attitude Control System

As already mentioned, appreciable disturbance forces can be generated during stationkeeping, caused by thrust vector misalignment, thrust imbalance, and movements of the center of mass of the spacecraft. In general, as well as causing orbital drift, these forces will produce torques on the spacecraft which must be counteracted by the attitude control system, thereby using hydrazine propellant.

The mass penalty for the TVBS mission has been assessed by use of the analysis devised by Pye.[19] It was assumed that the hydrazine thrusters employed have a specific impulse of about 180 s and that thrust moment arms appropriate to the dimensions in Fig. 4 are used. It was shown that between 3.7 and 6.1 kg of hydrazine will be required to correct for thrust vector misalignments, about 0.1 kg for thrust imbalance, and 0.6 to 1.0 kg for movements of the center of mass.

As the dominant contribution is caused by thrust vector misalignment, the hydrazine consumption could be reduced by thrust vectoring, preferably by mounting the thrusters on gimbals. Unfortunately, the mass of the four gimbal systems would be comparable with the fuel saved, and so no overall benefit would result.[19] However, the flexibility of multikilowatt solar arrays causes additional disturbing torques, and these could also be compensated for by thruster gimbaling. Moreover, thrust vectoring could be used to desaturate momentum storage wheels,[19] and so this technique may eventually find wide application.

## Thruster/Spacecraft Interactions

Although the interaction between an ion thruster and a spacecraft is complex, orbital tests[20] and careful analyses[18] have indicated that there is no fundamental obstacle to the successful integration of a thruster system into an operational satellite. The present preliminary study has confirmed that this is the case with the U.K. 10-cm thruster.

#### Thermal Effects

It is probable that it will be necessary to recess the ion thrusters into the North and South faces of the spacecraft body to give adequate clearance for the stowed solar array (Fig. 4). With this assumption, the temperature distribution within the thruster and its immediate environment was determined by the conventional analysis of a node model, including the effects of solar radiation input.

It was found that the maximum discharge chamber temperature is below 200°C, which is quite acceptable, and that the thermal input to the spacecraft never exceeds 12 W. The rear enclosure of the thruster reaches a maximum temperature of under 60°C, and so there is no possibility of vaporizer thermal control being lost. In addition, the minimum rear enclosure temperature is well above the freezing point of mercury. An analysis was also carried out of the thermal characteristics of the packaged power conditioner (pcu). It was shown that the unit will operate satisfactorily, provided that the hot surface of the package is positioned to face the spacecraft control radiator.

**Contamination by Thruster Effluent**

Contamination of spacecraft surfaces by effluent from the thrusters must be minimized, so that the output of the solar arrays is not diminished and the characteristics of thermal control surfaces are not altered. It is considered that the operating constraints discussed earlier are adequate to prevent the impingement of beam ions onto the arrays, but possible contamination from sputtered materials, charge-exchange ions, and mercury vapor remains to be considered.

Neutral mercury vapor is emitted from the thruster because the discharge chamber is not perfectly efficient. For T4A and T5, this neutral flow is about 0.065 and 0.04 mg/s, respectively; to this must be added a smaller contribution of about 0.01 mg/s from the neutralizer. A cosine distribution is to be expected which will interact mainly with the solar arrays. However, any mercury condensing on them will immediately re-evaporate, because they are exposed to the sun whenever the thrusters are operating. Conversely, the small flux reaching the body of the spacecraft may condense on any cold surfaces, with the possibility that their thermal characteristics will be changed. However, it was concluded that this will not be a significant problem, because, for typical radiator temperatures, the evaporating flux is nearly three orders of magnitude more than the likely maximum deposition rate. Only if amalgamation occurs will difficulties arise.

A more serious problem is the production of slow mercury ions by charge-exchange reactions between the primary ion beam and the neutral atom efflux. These ions sputter material from the accel grid and, under some circumstances, cause sputtering of the spacecraft structure. They also form a cloud of tenuous plasma around the spacecraft which can result in significant leakage currents to exposed solar array electrical interconnections. Such currents are likely to be more serious if high-voltage arrays are introduced, although, with the T5 thruster,[5] the very low neutral efflux should assist in alleviating the problem. However, this potential difficulty can be defined fully only by means of a space experiment; the results of ground tests are likely to be masked to some extent by the influence of the vacuum chambers employed.

As regards sputtering of the spacecraft structure, if the neutralizer system is operated efficiently, with small coupling voltages, the energy acquired by the charge-exchange ions in reaching the spacecraft will be lower than most sputtering thresholds. Consequently, there will be very little damage; this

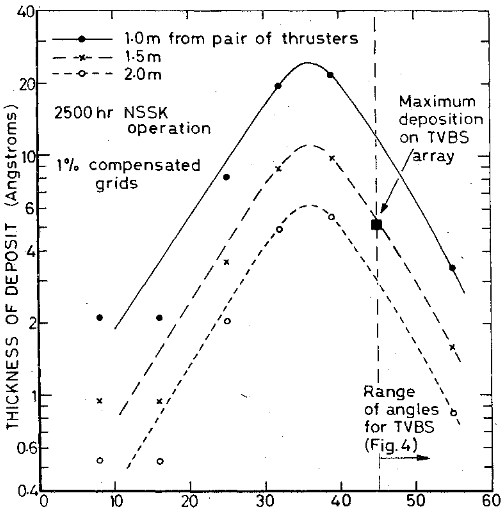


**Fig. 7 Molybdenum deposition in 2500 h as a function of angle to the plane of the grids of the T4A thruster.**

conclusion needs to be tested, however, with representative spacecraft surfaces mounted in an electrically floating experimental system. If damage to sensitive components is detected, it can be avoided thereafter by applying a positive bias of a few volts to them to repel the ions. The permanent contamination of cold spacecraft surfaces by charge-exchange ions is extremely unlikely, because the calculated flux is much less than the flux of neutral atoms. However, amalgamation with sensitive surfaces must again be avoided.

The molybdenum sputtered from the accel grid would normally be regarded as the most serious spacecraft contaminant, because, once deposited, it will not re-evaporate. However, this sputtering has been considerably reduced in the T4A and T5 designs by achieving high values of $\eta_m$, by reducing the accel grid voltage, by minimizing the neutralizer mass flow rate, and by attaining a more uniform discharge chamber plasma distribution. This overall improvement has been confirmed by measurements of the deposition rate of sputtered materials during thruster life-tests.[21] After a running-in period, the maximum deposition rate of molybdenum 20 cm from the center of 1% compensated grids was about $5\times10^{-12}$ g-cm$^{-2}$-s$^{-1}$, at an angle of about 35 deg to the plane of the grids. There is an additional contribution due to sputtering of the stainless-steel ground screen,[21] but this can be reduced to a low level, because the charge-exchange ions are attracted preferentially to the negative accel grid.

In Fig. 7, these measured deposition rates have been converted to thicknesses at various distances from a pair of thrusters, assuming the applicability of an inverse square law, a sticking coefficient of unity, and NSSK operation for 2500 h. To relate these values to the thicknesses that would be deposited on a solar array in actual operation is not straightforward, partly because of the varying geometry as the array rotates. However, for the preliminary TVBS configuration depicted in Fig. 4, the closest part of the center of the array visible from the thruster (point A) is at a distance of 1.6 m and at an angle of 45 deg to the plane of the grids. The deposition thickness at this point will be greater than elsewhere because areas further away from the spacecraft body, such as B in Fig. 4, are at larger distances from the thrusters and also subtend larger angles. Points further from the centerline, such as C, do approach closer than 1.6 m to the thrusters as the array rotates, but the angle subtended to the plane of the grids then becomes greater and, as indicated in Fig. 7, the deposition rate falls rapidly.

It may therefore be concluded that the maximum thickness accumulated on any part of the array during 2500 h of operation is about 6 Å. From Ref. 22, it will be seen that the transmittance of solar radiation through such a layer is 100%, apart from possible thin-film effects; there should, consequently, be no appreciable degradation of the TVBS array due to this cause.

It also is possible for slow molybdenum ions to be created through charge-exchange processes in the exhaust plume. These ions can travel in all directions, and so a small flux will reach most spacecraft surfaces and, unless repelled electrostatically, will condense there. Although this flux has not been evaluated thus far, for the UK thrusters, the low production rate of molybdenum atoms from the grids suggests that this effect should not be a serious problem.

## System Reliability

It has been proposed that the overall reliability aim for the TVBS electric propulsion system should be 0.95. A preliminary reliability analysis established that this can be achieved by a system of four T4A or T5 thrusters, with four power conditioners and two interconnected propellant tanks, assuming that a limited degree of passive redundancy is provided in the pcu.

The reliability of the breadboard pcu[23] was assessed for a 7-yr mission consisting of a 20-h launch phase and a 2200-h operational life, using ESTEC standard failure rates for

Table 2 System reliabilities for TVBS mission

| System | Mounting plane | |
|---|---|---|
| | Radial | Tangential |
| 4 thrusters, each with dedicated pcu, no redundancy | 0.932 | 0.691 |
| 4 thrusters, each with fully redundant pcu | 0.999 | 0.984 |
| 4 thrusters, 4 pcu's, fully switchable | 0.934 | 0.931 |

components. The value obtained was 0.74; this can be increased significantly by redundancy techniques and by circuit simplification. It was much more difficult to estimate the reliability of the thruster, because of the lack of statistical data, although thruster and component life-tests strongly suggest that wearout mechanisms are sufficiently slow for there to be no difficulty in achieving the objectives of typical NSSK missions.

Possibly the best approach to adopt in the absence of experimental data concerning random failure of the thruster is that followed by Molitor.[24] He assessed failure rates by compounding values for component parts deduced by comparison with typical airborne equipment. For the SIT-8 thruster,[15] a failure rateof 10,600 in $10^9$ h was obtained. As the technology of the T4A and T5 thrusters is similar, a failure rate of 12,000 per $10^9$ h was assumed for assessing system reliability.

Using these values, the system reliabilities given in Table 2 were derived for the two mounting configurations; the radial scheme is preferable because the failure of one thruster does not preclude the other one of that pair from operating and so the mission can be accomplished with any two of the four thrusters. It is evident that full redundancy is not necessary to meet the stated aim of 0.95 and that switchable pcu's are of benefit only with tangential mounting.

## Conclusions

It has been concluded from the work reported in this paper that the U.K. T4A and T5 thrusters are fully compatible with the TVBS North-South stationkeeping mission. It was shown that a thrust of 10 mN is suitable, allowing the proven thruster lifetime to be as low as 2200 h, with a correspondingly reduced cost of development and of space qualification.

It is suggested that four thrusters should be used, with two on the North face of the spacecraft and two on the South face. They should thrust through the center of mass, at an angle to the North-South direction to avoid impingement of the ion beams onto the solar arrays. If mounted in the plane containing the radius vector to the Earth, the effect of disturbance forces is minimized, and a single thruster of a pair may be operated without unacceptably perturbing the orbit. It was shown that the concept of overcompensation, followed by drift through an acceptable "deadband," avoids the possibility of direct ion beam impingement on broad solar arrays.

The propellant used by the attitude control system in correcting for disturbance torques was calculated for thrust vector misalignments of up to 3 deg, a thrust imbalance of $\pm 3\%$, and an arbitrary shift of the c.g. of the spacecraft of 1 cm; the major cause of attitude perturbations was shown to be thrust vector misalignment. However, assuming moment arms consistent with the proposed TVBS design and a specific impulse of 180 s, the calculated hydrazine consumption was not large enough to justify the mass and complexity of thrust vectoring. However, this conclusion might be reversed if momentum wheel unloading could be accomplished using a vectoring system.

The study of the integration of a thruster system with the TVBS spacecraft showed that no particularly difficult problem should arise. In particular, thruster temperatures are within acceptable limits, both with and without solar radiation input, and the heat flux to the spacecraft is under 12 W from each thruster. Similarly, it should be possible to operate the pcu within the spacecraft, provided that the hot surface of the device can face the control radiator of the spacecraft or have a direct view to space.

The experimental work so far accomplished suggests that solar array contamination by sputtered material should not be a problem for the TVBS mission. As regards the contamination of sensitive surfaces with condensed mercury, it has been shown that all are sufficiently warm for immediate re-evaporation to occur. Similarly, charge-exchange mercury and molybdenum ions are not expected to cause difficulties, although more work concerning the latter is required. A study is also needed of the interaction between the tenuous plasma surrounding the spacecraft and the solar array.

A preliminary assessment of system reliability has indicated that the stated aim for the TVBS mission of 0.95 can be achieved with only a small increase in pcu reliability if the thrusters are mounted in the plane containing the Earth radius vector. In the alternative mounting scheme, a much greater increase is needed, or a switching matrix must be employed to allow any pcu to be connected to any thruster.

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