JOURNAL OF SPACECRAFT AND ROCKETS
Vol. 33, No. 5, September–October 1996

Evaluation of the United Kingdom Ion Thruster

Mark W. Crofton*
The Aerospace Corporation, El Segundo, California 90245-4691

A test and evaluation study examining the United Kingdom's 10-cm-diam xenon ion thruster system, known as the UK-10, has been performed at The Aerospace Corporation. The purpose of the study was to determine whether the UK-10 is a suitable replacement on U.S. Air Force satellites for presently employed propulsion devices. The most essential aspects of the UK-10 pertaining to its potential use were evaluated, but a conventional life test was beyond the scope. An examination of performance, lifetime, spacecraft integration issues, mission analysis, and computational modeling and a comparative technology assessment were performed. Emphasis was placed on the creation of a database that will be useful for the integration and operation of satellites using the UK-10 or other gridded electrostatic ion thrusters. Advanced electric-thruster characterization methods produced detailed measurements, in real time, of many fundamental properties of the thruster. The test program concluded that a UK-10 ion propulsion system will be acceptable for use on suitable spacecraft, subject to the considerations discussed, and of considerable potential benefit.

Nomenclature

I_{sp} = specific impulse
V = electrical potential
η_e = electrical efficiency, %
η_m = mass utilization efficiency, %
η_T = thrust efficiency, %
σ = standard deviation

Introduction

MANY geostationary satellites require and/or benefit from longer life, precision stationkeeping, and the capability to reposition on orbit. DSCS III (Defense Satellite Communications System) satellites, for example, have an N–S and an E–W stationkeeping margin of ±0.1 deg. Commercial communications satellites generally have the same requirements. National space policy includes the following goals: reduced spacelift operations and maintenance costs, operational flexibility, routine access to space, and compatibility with the environment. Electric propulsion systems offer great potential to reduce spacelift operations and maintenance cost, increase satellite lifetime, and increase operational flexibility. Also, a reduced launch mass of individual satellites and the possible use of smaller boosters may reduce the environmental impact of space system operations.

The principal objective of the Foreign Comparative Test (FCT) Program of the Department of Defense (DoD) is the identification, testing, and evaluation of non-U.S.-produced products that are potentially cost-effective and timely alternatives to satisfy DoD requirements. One product for evaluation in this program was the United Kingdom 10-cm-diam xenon ion thruster system known as the UK-10. This electric propulsion system is designed for satellite stationkeeping and on-orbit repositioning missions where a substantial velocity change (ΔV) is required to control satellite position over the mission lifetime. The purpose of this program was to determine whether the UK-10 could beneficially perform these missions for U.S. Air Force (USAF) satellites. The Aerospace Corporation was chosen to obtain independent, unbiased ground testing and analysis results.

The UK-10 propulsion system consists essentially of the thruster, propellant supply and monitoring equipment, and power conditioning and control equipment (PCCE). The U.K. Defence Research Agency (DRA), Farnborough, England, provided an engineering model thruster operating on xenon propellant. The term UK-10 Ion Propulsion System (IPS) will refer here to the flight systems being manufactured at Matra Marconi Space UK.

To determine readiness and suitability of the UK-10 IPS for USAF missions, a complete ground test and evaluation (T&E) of the UK-10 thruster, examining 1) performance, 2) operating lifetime, and 3) spacecraft integration/interaction issues, was required. This ground-based evaluation coupled with the European Space Agency's (ESA) operational UK-10 IPS experience on the ARTEMIS geostationary satellite beginning in 1998, will enable a decision to be made concerning UK-10's readiness and suitability for operational use on U.S. missions. Since the instrumentation on ARTEMIS will provide few of the precise data obtainable during ground-based T&E, and the scope of the ground testing did not include an actual life test of a complete flight unit, both ground and space testing had to be essential elements of the evaluation. ARTEMIS is an essential part of ESA's space technology program, and will perform operational laser optical data relay throughout its 10-year mission. The UK-10 IPS and German RIT-10 ion thruster system will share N–S stationkeeping duties. It is expected that UK-10 IPS flight experience on ARTEMIS will resolve the few remaining concerns stemming from inherent differences between ground and space tests, and the ground testing of individual components rather than complete satellite systems.

The ground-based evaluation was a comprehensive, non-life-test study. This program evaluated all essential aspects of the UK-10 pertaining to its potential use on USAF satellites. An examination of performance, lifetime, spacecraft integration issues, mission analysis, computational modeling, and a comparative technology assessment has been performed.

This paper is intended to accomplish three main purposes: 1) summarize and disseminate UK-10 test and evaluation results, 2) link various program technical papers and results, and 3) provide further information on a few technical topics that were not fully treated elsewhere. As such, the discussion will range from superficial summary to detailed treatment.

Test and Evaluation Program Structure

The thruster provided by the DRA, Farnborough, is designated a T5 Mk3 by the United Kingdom. The unit, an engineering model thruster using xenon propellant, was transferred to The Aerospace Corporation prior to the decision by ESA and DRA to use a triple grid set for ion beam extraction on ARTEMIS. Since a triple grid set was not available, a thruster with the baseline twin grid set was provided, and a triple grid set was retrofitted to the thruster during the test program.

Received Oct. 4, 1995; revision received May 13, 1996; accepted for publication May 13, 1996.

*Member, Technical Staff, Technology Operations, Mechanics and Propulsion Department, M5-754, P.O. Box 92957, Los Angeles, CA 90009-2957. Member AIAA.

In view of the nature of the T&E program, engineering or flight unit propellant supply and power conditioning equipment were not required. Laboratory power supply and propellant feed equipment with the required versatility was built at The Aerospace Corporation.

Test results of the program have been documented.[1–15] These results were presented at technical conferences to help facilitate the dissemination of information to the propulsion and satellite design communities, including personnel at satellite manufacturers. A test and evaluation report[15] and a disposition report have been provided to the USAF. The main program results are discussed in this paper.

Transfer and installation of the thruster and the development of a test plan began in September 1992. Thruster operations began in March of 1993. During the subsequent two years, hereafter referred to as the T&E period, a comprehensive evaluation was conducted based on a written test plan. The level of effort at The Aerospace Corporation was approximately 2.3 full-time technical staff and one full-time technician for a period of 2.6 years. Of the total funds provided by DoD, a portion went to NASA for the development and application of modeling software to analyze engine operation. This work was subcontracted to Colorado State University.

Because the world market has several electric thrusters under development with similar performance characteristics, a comparative technology assessment has been made of these thrusters. It was necessary as well to perform simple mission analyses to assess the benefit of utilizing electric propulsion technology. Present and projected future requirements of the USAF regarding propulsion have been considered in the analysis of UK-10 IPS suitability for USAF spacecraft.

Summary of Results and Discussion

Mission Analysis

Many studies have shown that propulsion devices with high thrust per unit mass flow, such as the UK-10, offer dramatic cost reduction, payload and repositioning capability enhancement, and/or increased mission lifetime, depending on how the engineering trade is executed.[1,16–18] These benefits become possible when the UK-10 is used for north–south stationkeeping (NSSK) instead of conventional chemical propulsion systems.

An enhanced repositioning capability and reduction in total satellite wet mass devoted to NSSK are made possible by high-specific-impulse electric propulsion. These important benefits have already been detailed (see especially Figs. 3 and 4 in Ref. 15).[1]

During the Persian Gulf conflict several years ago, a DSCS II (geostationary satellite with 612 kg and 520 W, beginning of life) was repositioned by roughly 100 deg to augment existing communications capability. The transfer required about 1 month at a total propellant expenditure of approximately 6 kg. The fact that shorter transfer times require additional propellant, and this was already an appreciable fraction of propellant remaining, seriously constrained the repositioning parameters. On-orbit maneuver rates for geosynchronous longitude repositioning maneuvers using electric thrusters exceed those of chemical thrusters, with much higher fuel efficiency, when the spacecraft has an adequate power-to-weight ratio. If 500 W had been available for the same DSCS II maneuver, the UK-10 IPS could have repositioned it within the same time period using about 0.5 kg of propellant (see Fig. 4 of Ref. 15). With higher available power, the transfer could be made more quickly and with much less propellant. As a general rule, only the most modern geostationary satellites have a high enough power-to-weight ratio to perform on orbit repositioning with both less time and less propellant.[1] Given the observed power-to-mass trend,[1,19] however, new satellites will have little difficulty meeting this requirement.

Performance Envelope

The UK-10's performance meets the anticipated requirements of many USAF geostationary satellites. The thruster is capable of high propellant mass utilization, high efficiency with respect to usage of electrical power, autonomous operation for long periods, and a wide throttling range. The optimized operating points for nominal 25- and 18-mN operation, established as a direct result of the T&E program, are given in Table 1, with the associated utilization and efficiency figures in Table 2. These operating points were chosen to maximize thruster lifetime and minimize spacecraft contamination. Some variation in the parameters is allowable, since the lifetime and contamination are not sharply dependent on the control parameters about the chosen operating point.[3,4,14] For example, a 0.3-A reduction in main discharge current, with compensating increase in magnet current, produces the same nominal 25-mN thrust level with lower energy cost per extracted ion. This benefit comes at the expense, primarily, of a small increase in anode voltage.

Table 1 Operating-point parameters for 25- and 18-mN nominal thrust levels

	25 mN		18 mN	
Component	Voltage, V	Current, mA	Voltage, V	Current, mA
Beam	1100	457	1100	329
Main discharge	43.5	3000	43.0	2250
Main cathode, keeper	10.5	520	11.2	550
Solenoid	12.1	162	9.2	138
Neutralizer cathode, keeper	19	900	19	900
Accelerator grid	−225	1.20	−225	0.90
Decelerator grid	−50	0.90	−50	0.65
	Flow rate, mg/s			
Cathode	0.097[a]		0.0849[a]	
Main chamber	0.584		0.438	

[a]Includes an estimated 20% reduction due to flow restriction of operating cathode (the flow was measured with cathode off).

Table 2 Calculated and corrected parameters

	25 mN		18 mN	
Calculated parameter	Calculated	Corrected	Calculated	Corrected
Thrust F, mN	25.0	23.7[a]	18.0	16.9[a]
Specific impulse I_{sp}, s	3744	3359[b]	3510	3065[b]
Power to thruster P_T, W	658	658	483	483
Mass utilization efficiency[a] η_m, %	91.3	86.4[c]	85.6	79.5[c]
Electrical efficiency η_e, %	76.4[d]	68.8[e]	74.9[d]	67.4[e]
Thrust efficiency η_T, %	69.8	59.3[f]	64.1	52.6[f]

[a]Corrected for beam divergence, multiply charged ions, etc., using correction factors given in Ref. 13. The direct thrust-stand measurement (also Ref. 13) gave a lower result, but if correct involves an unknown thrust loss mechanism.
[b]Includes neutralizer flow rate of 0.04 mg/s and total thrust correction factor (see Ref. 13).
[c]Includes neutralizer flow rate of 0.04 mg/s.
[d]Does not consider PCCE losses.
[e]Assumes PCCE is 90% efficient.
[f]Uses the corrected thrust level given above.

Thruster performance figures obtained by The Aerospace Corporation improved during the two-year T&E period. The performance improvement resulted from refinement of cathode and main flow ratios, grid conditioning, use of a triple grid set, and the optimization of operating parameters. Reference 3, for example, contains earlier performance data at operating points with poor performance compared to the results of Table 1. An increased ratio of cathode to main flow rate, better grid conditioning, and the selection of a lower cathode current and higher discharge current permitted higher thrust levels and greater propellant mass utilization while retaining good stability and a main discharge voltage within acceptable limits. However, performance figures gradually improved with time as well, as the thruster aged. It is believed that this phenomenon resulted from the grid running-in process that enlarged accelerator grid aperture diameters and chamfered the decel grid apertures on the downstream side. Existing data suggest that the anode voltage is significantly elevated by direct impingement effects. At a given setpoint for the anode voltage, if the thruster grids experience direct impingement, higher gas flow is required to achieve the same beam current and performance figures suffer.

Engine operation is complex, because of the coupling of three propellant flows and three different plasmas. Operating parameters such as the cathode and anode currents and the ratio of cathode to main discharge chamber flow rate required significantly different settings for optimum performance at The Aerospace Corporation than were used in U.K. life tests. The U.S. performance figures are generally better than those determined by U.K. testing. This may be due to facility effects, primarily the better vacuum obtained in The Aerospace Corporation test chamber, and direct impingement effects, but errors in flow-rate measurement cannot be ruled out.[20] The influence of facility background xenon pressure on discharge voltage and grid currents is described in Ref. 3. The Aerospace Corporation equipment used for performance measurement was periodically recalibrated to ensure accuracy, and performance figures were verified by U.K. personnel during visits to The Aerospace Corporation facility.

In addition to an analysis of the performance envelope, thruster performance was characterized with respect to 1) thrust-vector migration,[11–13] 2) beam divergence,[11,14] 3) charge-state composition of the primary ion beam,[11,14] and 4) absolute thrust level as determined by a thrust stand.[13]

The UK-10 IPS performance is appropriate for a variety of USAF and other U.S. satellites with adequate end of life (EOL) solar array power per kilogram of spacecraft mass. Geosynchronous satellites with ≥ 2 W/kg power-to-mass ratio can easily accommodate a UK-10 IPS for the purpose of NSSK. A set of four UK-10 thrusters can readily perform NSSK for satellites of the 2000-kg class, one thruster firing for about 2 h at a time, twice per day except during eclipse periods. For satellites with a low ratio of solar array power to satellite mass (≤ 1 W/kg is certainly in this category), the selection of a UK-10 IPS rather than a thruster system with a higher ratio of thrust to electrical power is highly unlikely. Spacecraft designed to use electric propulsion may incorporate array power in excess of that required for all spacecraft functions excluding stationkeeping and repositioning. This improves flexibility and decreases the importance of the thrust-to-power ratio. In practice, UK-10's suitability for spacecraft should be handled on a per case basis.

Lifetime

The Aerospace Corporation testing of the UK-10 thruster has demonstrated adequate projected thruster lifetime for precision stationkeeping on typical USAF satellites. The primary failure points of the thruster are grids, cathode, neutralizer, and internal components.

The cathode and neutralizer are both hollow-cathode devices, relying on the same materials and technology but differing in geometrical details. Each can produce a single point failure of the IPS. A defect in the manufacturing process for some cathodes and neutralizers has caused heater failure after less than 1000 h of cyclic operation.[21,22] The devices are still operable after heater failure using a cold-starting procedure, which is expected to shorten the useful lifetime, perhaps to several thousand hours. Lifetests in progress at DRA have successfully completed a few thousand hours of cyclic operation on several main cathodes in parallel, and more than 5000 h on several neutralizers.[22,23] Neutralizers and cathodes have recently been fabricated by DRA according to a more robust design, and life tests are scheduled to begin imminently.[24] An ignition failure of a similar neutralizer hollow cathode occurred during a MELCO ion thruster endurance test for the ETS-VI satellite, after some 7000 h of operation.[25] After a detailed examination it was concluded that degradation of the insert had occurred, and that this was probably caused by impurities in the propellant, principally water. It is well known that the operation of impregnated porous tungsten dispenser cathodes of the type commonly used in gridded ion engines is extremely sensitive to the effects of O_2 and H_2O contaminants.[26] Ground-test cathode lifetime results are believed to represent a lower limit in view of the residual gas environment of the test facility. Xenon hollow cathodes similar to the IPS devices have been successfully operated far in excess of 10,000 h.[27] The SERT II neutralizer cathodes operated successfully on orbit with mercury propellant for over 17,000 h before the propellant reservoir was emptied.[28]

A high erosion rate of internal components, principally the baffle disk and pole pieces, was observed in the thruster examination following the first 500-h life test, conducted at AEA Technology.[21] Internal erosion was the life-limiting mechanism at the chosen operating point, rather than the anticipated accelerator-grid mechanical failure caused by erosion effects. The life test was conducted under nonideal circumstances, however, with up to 50 V across the discharge for extended periods. A conservative 40-V upper limit for the UK-10 anode (discharge) voltage on ARTEMIS was subsequently imposed, to ensure that internal erosion rates would not limit thruster life. The Aerospace Corporation results regarding the dependence of the relative overall erosion rate of internal components indicate that this restriction is more stringent than necessary,[4,11,14] resulting in an operating point with reduced grid lifetime and lower projected lifetime for the thruster.The internal erosion rate increases rapidly with discharge voltage, and the accelerator grid erosion rate decreases relatively slowly when beam current or propellant flow rates are fixed. Even up to 43 V on the anode, the thruster lifetime at 18- and 25-mN thrust levels is not limited by internal erosion.[4,11,14] At this discharge voltage, the possibility of rapid screen-grid erosion was also of some concern. However, theoretical modeling results[20,22] as well as screen-grid and discharge-chamber erosion-rate data[14,15,21,22] strongly suggest that screen-grid and other internal erosion will not limit the thruster lifetime at the 43-V operating point (32–33 V between anode and cathode keeper; see Table 1).

In cases where very high discharge voltage is required to increase utilization and grid lifetime to perform a given mission, a minor modification of the IPS might enable such operation without unacceptably high internal erosion rates. Reduction of internal erosion rates due to trace amounts of N_2 in the propellant or as residual vacuum chamber gas was observed long ago, but the underlying mechanism is still poorly understood.[29] When several percent N_2 was added to the T5 propellant at the conclusion of the study described in Ref. 4, it was observed that quartz crystal microbalance (QCM) mass deposition rates fell by more than 20% at small angles off the grid plane. At these angles, iron and its oxidation products accounted for 30–40% by weight of the deposition material.[4] A small reduction in QCM signal, comparable to 2σ in the relative error, was observed at angles where molybdenum mass deposition products are some 5 times higher than iron. The magnitude of the signal decrease was roughly that of the iron contribution when pure xenon propellant was used. It seems likely, in view of past work (see citations in Ref. 29) and the potential for large changes in sputtering yield near threshold,[30] that the N_2 dramatically reduced internal erosion rates, while having little effect on the erosion of molybdenum from the accelerator grid because of the high ion impingement energy. It is not known whether the lifetime of other components, such as hollow cathodes, would be affected by the nitrogen.

Erosion of grid and internal thruster surfaces results in the formation of deposit layers, which may exhibit flaking that leads to modification of grid electric fields and grid shorting. The problem may be worse in space, since the weightless environment facilitates flake migration; grid shorting associated with flakes was a serious problem for the SERT II spacecraft.[28] The use of trace N_2 levels in the xenon propellant might mitigate this problem. The UK-10 IPS includes a robust grid short clearance capability to eliminate grid shorts caused by flakes.[31] Anomalous accelerator-grid current levels, observed several thousand hours into the life test of a MELCO thruster and thought to result from flakes formed in the discharge chamber and deposited on the screen grid, were apparently reduced through the use of a mesh surrounding the discharge chamber.[32]

Several evaluations in the past have generated estimates of the T5 grid lifetime. These have been based on physical measurements of grid aperture modification over extended time periods, and a set of assumptions concerning facility effects and other factors.[21,22,33] It was estimated after the AEA triple grid life test that a reduction in operating vacuum chamber pressure from 10^{-5} Torr to space conditions would reduce the accel-grid erosion rate by 50% because of the lower charge exchange ion production and impingement rates.[33] The Aerospace Corporation studies of the pressure dependence of grid currents and the QCM signal shows that the reduction is less than 15%.[3,4]

The definition of grid lifetime is not universally agreed upon. The definition used here is that end of life involves mechanical failure of the grids associated with total erosion of the grid webbing. Table 3 lists the various estimates of grid lifetime obtained in The Aerospace Corporation and U.K. studies. The results of the SAPPHIRE computational model developed by AEA Technology and validated recently by excellent agreement with experimental data concerning accelerator-grid current dependence on background

Table 3 T5 predicted component lifetimes[a]

	Lifetime, h			
Component	The Aerospace Corporation	AEA–SAPPHIRE[b]	AEA 500-h Test[c]	DRA life test
Accelerator grid:				
25 mN	10,700[d] ± 3000	6,075[e]	9355	—
18 mN	15,000 ± 4000	7,560[e]	N/A	—
25 mN, optimum configuration, self-protection	18,000[f] ± 5000	16,200[g]	N/A	—
Main cathode, neutralizer cathode	≥10,000[h]	—	—	>2000[i], >5000

[a]The Aerospace Corporation 18- and 25-mN operating points are given in Table 2.
[b]AEA Technology computational model, validated by The Aerospace Corporation grid current measurements (see Refs. 20, 22).
[c]Approximately 80% propellant mass utilization.
[d]Mechanical failure, at 91% propellant mass utilization. Predicted life at 80% utilization would be much less, as indicated by the AEA model[20] and The Aerospace Corporation data.[4]
[e] Does not include a self-protection effect (this effect is produced by the redeposition of sputtered molybdenum onto the accelerator grid).[26] Also, grid configuration is slightly different from the one tested at The Aerospace Corporation. 80% utilization.
[f]Prediction based on nonoptimized grid set.
[g]85% utilization.
[h]Assumes manufacturing issues are solved.
[i]Life tests in progress, as of June 1995.

Table 4 Main feature comparison for electric thrusters[a]

Feature	UK-10	SPT-100	XIPS-13	ARCJET	BIPROP
Required thrusting hours[b] for $\Delta V = 500$ m/s, $M_{SC} = 2000$ kg	13,469; 3367 each[c]	4910; 1228 each	17,919; 4480 each	1604; 401 each	Short
Beam divergence	Low	High	Medium	Medium	Medium
Ground support level	Low	Low	Low	Low	High
Approximate lifetime, h	10,000[c]	>5,000[d]	10,000[e]	1000	20
Thermal shield required	No	Advisable	No	Advisable	No
Contam. shield required	Advisable	Complex issue	Advisable	No	No
Thrust-level operating point	Variable	Some variability	Fixed	Some variability	Fixed
Thrust level, variation	Steady	Oscillates	Steady, may recycle	Some noise	10% variation typical
Typical thrust level	25 mN	80 mN	18 mN	200 mN	100 lbf
Body of flight data	None	Substantial	None	Substantial	Large
Approx. thrust efficiency[f], %	60	45	50	≤35	—
Approx. specific impulse, s	3000	1600	2600	≥500	300
Complexity	High	Moderate	Moderate	Low to moderate	Low

[a]A set of four is assumed for each electric thruster.
[b]Cant angle 30 deg. $\Delta V = 500$ m/s corresponds to a 10-year (NSSK) mission for a geosynchronous spacecraft.
[c]Projected lifetime at 25-mN operating point (see Tables 1, 2). Lowering the propellant mass utilization reduces the thruster lifetime because of the increased accelerator grid erosion rate.
[d]See Ref. 46.
[e]Claimed lifetime at 18 mN—not demonstrated by life test or other published analysis, but consistent with UK-10 test results at The Aerospace Corporation.
[f]Includes PCCE, neutralization losses (if any).

pressure[20,22] are included. SAPPHIRE is an electrostatic particle-in-cell code representing the plasma as a set of particles interacting through a calculated force field.

The Aerospace Corporation estimate of the accelerator-grid lifetime was obtained by integrating the molybdenum flux determined from QCM and elemental analysis data. The Aerospace Corporation error limit on the grid lifetime is estimated to be ±3000 h.[4] This includes the effects of several uncertainties, among them the following: 1) the point at which grid function is too degraded for practical use is not well understood, and 2) material sputtered from individual grids has a wide angular distribution subject to change over time and loss rate varies over the grid diameter. A fraction of that material, which can only be estimated, redeposits on adjacent grids and the discharge-chamber wall.

In addition, the common approximation that the grid erosion rate is constant over time has been used. The erosion rate of the grid set provided varied dramatically during its 500-h lifetest at AEA.[21,34] However, two measurements of the erosion rate at The Aerospace Corporation, separated by several hundred hours, yielded similar rates. Similarly, the qualification program of the MELCO ion thruster for ETS-VI indicated a reasonably constant rate of mass loss for the accelerator grid.[25] An extrapolation over time can therefore be justified, although some time dependence is likely as erosion modifies the grid geometry, and this dependence is not known.

On the basis of an analysis of all known failure mechanisms, the UK-10 IPS lifetime is predicted to be roughly 10,000 h at the 25-mN operating point in Table 1, which is more than sufficient for NSSK of modern satellites. This prediction assumes that any manufacturing and minor design problems are corrected. The potential device lifetime at 18 mN appears to be well in excess of 10,000 h, but insufficient life testing, particularly of the cathode and neutralizer, and some manufacturing issues result in considerable uncertainty concerning the overall thruster lifetime as presently manufactured. On xenon, hollow-cathode lifetimes exceeding 10,000 h have been demonstrated, even at much higher currents than employed by IPS cathodes.[27] A typical 2000-kg geostationary satellite that requires precision NSSK will need about 13,000 h of thruster operation at 25 mN, over a 10-year mission lifetime. This requires about 3400 h of operation from each UK-10 thruster (see Table 4), for a normal set of four UK-10 thrusters with two each on the north and south panels. A margin much more than 100% exists in the lifetime necessary to perform the mission. For very massive spacecraft, a set of six or eight thruster units may be desirable to maintain a high lifetime margin.

Spacecraft Integration

A number of issues pertain to the integration of an ion propulsion system on a spacecraft. Spacecraft–thruster interactions and plausible configuration of the UK-10 IPS on a DSCS III type of spacecraft are shown in Fig. 1 for illustration. In practice, the DSCS III is a marginal candidate for ion propulsion, because of its low power-to-mass ratio. As oriented, the solar arrays will receive the maximum

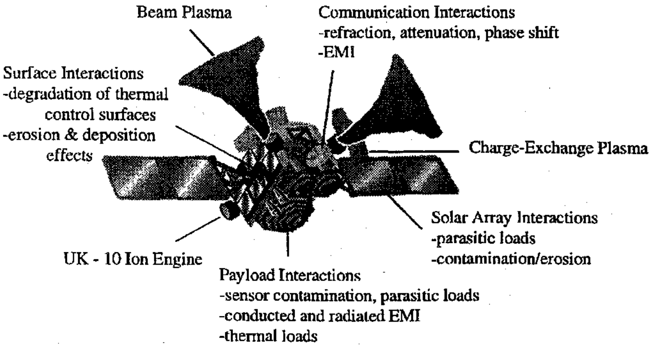

Fig. 1 Sketch of spacecraft–thruster interactions and plausible configuration of UK-10 IPS on a DSCS III type spacecraft.

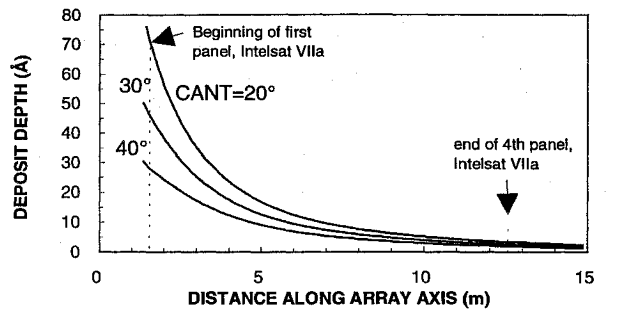

Fig. 2 Predicted molybdenum deposition depth along an Intelsat VIIa solar array axis.

deposition of thruster material (array normal in the plane containing thrust and array axes), the worst-case scenario. The angle between array and thruster axes for a north or south face (the arrays point north and south) is the cant and would ideally be small to produce more thrust along the N–S axis. Thruster positioning on the north and south thermal radiators is the most likely configuration. ARTEMIS has N–S radiator thruster pallets adjoining the anti-Earth face, rather than the east and west faces.

Contamination

The issues of lifetime and contamination are strongly linked, because the same process that accounts for the life-limiting erosion of the accelerator grid produces most of the material that contaminates the spacecraft environment. The deposition of material on surfaces in the thruster environment was monitored using a rotatable QCM and a diverse collection of sample spacecraft materials.[4] The effect of thruster environment on these materials samples during a 65-h exposure was analyzed according to modification of solar absorptance and emittance characteristics. The elemental composition of the deposits was also determined. The results indicated that levels of contamination exist that can quickly alter the characteristics of samples in the thruster-induced environment that view the extraction grids. Samples located behind the thruster were unaffected to within experimental error. In general, spacecraft thermal control materials, sensors, and all other major spacecraft components except solar arrays and possibly antennas will be positioned behind the grid plane. Owing to the inverse square distance dependence of deposition rate, shorter, less powerful solar arrays will suffer higher percentage degradation for a typical configuration.

The level of degradation should be reducible by the incorporation of a suitable sputter shield to eliminate the solar array direct line-of-sight view of the grids. Even at locations more than 90 deg off the thrust axis, where ions constitute the predominant flux of material, a shield can be beneficial. A contamination shield reduced the flux of mercury charge exchange ions from Teal Ruby's 8-cm IAPS thruster by 60% in this location.[35] There are no plans at present to include a shield on ARTEMIS. The generation of an optimized sputter shield design and its qualification are not trivial. The shield will act as a source of contamination itself when struck by highly divergent ions at beam energies. This must be balanced with the shielding effect to reduce degradation rates for critical surfaces while keeping any increase in contaminant flux to other surfaces at acceptable levels. The Japanese ion engine onboard the ETS VI satellite launched in 1994 did have a sputter shield,[32] as did the 8-cm ion engines on the Teal Ruby spacecraft, which was built but never launched.

The solar array deposition rate is also strongly affected by the thruster cant angle with respect to the face of the spacecraft. Figure 2 indicates the predicted molybdenum deposition depth along an Intelsat VIIa solar array axis, due to accumulated contamination from the T5 thrusters during 10,000 h of total operation. Computed EOL degradation of the total solar array output due to thruster-produced deposition is less than 10%. The nominal 25-mN operating point of Table 1 and data from Ref. 4 were used for the calculation. A set of four canted thrusters mounted on north and south radiator panels, 1.5 m from the array mounting point, was assumed (5000 h for each panel). The thruster operated at a nominal 25-mN thrust level, with no preferred average angle between thrust vector and array normal over the mission duration. Not included is the contribution of other metals, especially iron (see Refs. 4 and 14). The degradation of spacecraft materials and solar array output can be reduced through the use of a sputter shield on the thrusters or an increase in the thruster cant (see Fig. 2). The latter exacts an obvious penalty, since the thrust along the desired axis varies with the sine of the cant angle, and the thrust applied along the orthogonal axis must be corrected (for discussion of correction schemes for ARTEMIS, see Ref. 36).

Electromagnetic Interference (EMI)

Levels of T5-generated EMI at the communications frequencies 1–20 GHz used by the vast majority of satellites are very low.[37–39] The ARTEMIS narrow-band electric field limit is 30 dB (μV/m) in each of its L, S, and K_u communications bands and 20 dB (μV/m) in the 400–500-MHz B band. Outside the communications-band notches, the ARTEMIS limit in the 1–20-GHz range is much higher. T5 noise levels easily meet the ARTEMIS specification from 1 to 20 GHz.

The use of ion propulsion will tend to result in the placement of thruster, power conditioner, and other spacecraft electronics on north and south thermal radiators, in close proximity, because of the necessity of efficiently radiating waste heat away from the source points. This close proximity between the UK-10 IPS and spacecraft payload electronics increases the likelihood of EMI effects. If standard engineering practice regarding electronics shielding is inadequate for this configuration, the initial operational experience of ARTEMIS will indicate this unexpected result immediately. If necessary, EMI shielding can be augmented with the principal penalty of increased spacecraft mass. The flight experience of ion propulsion devices to date indicates little difficulty with EMI.[39]

Communications signals transmitted through the thruster plume at these frequencies undergo a small phase shift of several degrees or less, with attenuation and refraction effects also of little significance.[8–10] The maximum electron density in the plume is 1.0×10^{10} cm^{-3} at the 25-mN thrust level, and the greatest density–path-length product is 1×10^{11} cm^{-2}. At the higher densities existing in the plumes of multikilowatt ion thrusters, phase-shift and refraction effects are potentially more troublesome.[39–41] The UK-10 IPS is expected to meet all USAF requirements concerning EMI levels at standard communications frequencies. Satellites such as FLTSATCOM and UHF-FO, which use 240–270 MHz, are more likely to have an EMI problem than those satellites using only $\geq$1-GHz signals. As in some other electric propulsion devices, EMI levels are considerably higher at lower frequencies, and DoD noise specifications are exceeded by the T5 ion thruster over large portions of the spectrum below 100 MHz. This low-frequency noise is largely due to the neutralizer, with most of the noise below 20 MHz. USAF satellites do not utilize frequencies below 100 MHz for communications.

Sensor Interactions

Because the temperatures of thruster external surfaces are on the order of 200°C, any sensitive infrared sensors required on the spacecraft cannot be allowed to view the thruster. Similarly, visible, ultraviolet, and low-energy x-ray sensors would be affected by the emission from the thruster plume,[3] without appropriate precautions.

In addition, the low-density plasma, which extends well outside the primary ion beam, may have an impact. Electric potential gradients inside the beam cause charge-exchange ions to migrate away from the beam, forming an extended charge exchange plasma. Outside the beam, some charge exchange ions assume trajectories that take them behind the exit plane of the thruster, eventually striking spacecraft surfaces with translational energy believed to be predominantly in the range of 0–15 eV.[7,14,42] The flux of charge exchange ions to spacecraft surfaces behind the thruster may be as high as 10^{12} ions cm^{-2} s^{-1}. The flux level of Mo^+ is about five orders of magnitude lower.[43] Given the low impingement energy, sputtering damage to sensors will be minimal. However, some sensor types may require shielding to mitigate ion–sensor interactions, including current drain associated with Xe^+ neutralization. If contaminants are already present on spacecraft surfaces, ion-assisted deposition may occur, resulting in the potentially troublesome formation of surface films. Cryogenic sensors present a special problem if Xe^+, when neutralized, condenses on the surface, forming xenon snow. A thruster sputter shield will help to protect sensors,[35] but since the contaminant flux is almost exclusively ions, a better approach may involve electrostatic shielding.[44]

Thermal Control

The sum of excess thermal energy is about 200 W per 700 W of UK-10 IPS power consumption. The dominant thruster heat load on the spacecraft interior is associated with thermal energy conducted from the PCCE, which is about 90% efficient. This load is minimized by thruster placement on a north or south thermal radiator, with an unobstructed view of deep space so that radiative losses from the 200°C thruster body are maximized. The PCCE can be mounted directly on the interior of the radiator, which can be thermally isolated from the payload radiator to achieve undisturbed payload thermal control. PCCE efficiencies are similar across the range of electric propulsion devices, except that resistojets can eliminate the power conditioner by operating at spacecraft bus voltage. Hydrazine arcjets operating at 1.8-kW input power constitute a more difficult problem than UK-10 ion thrusters, but have been successfully integrated on geosynchronous communications spacecraft. Thermal integration of gridded low-power ion thruster technology on USAF and other spacecraft designed from inception to use ion propulsion is a manageable problem under standard engineering practice, but retrofitting a spacecraft equipped with chemical thrusters for stationkeeping would be a serious challenge.

Thrust Vector Direction and Gimbaling

The direction of the thrust vector migrates on the order of 0.1 deg during thruster warmup and normal operation.[11–13] ARTEMIS uses gimbal mounting with steering capability of plus or minus several degrees to eliminate any concern associated with thrust-vector migration during the life of the thruster and shift in the spacecraft's center of mass over its mission duration. Whether these gimbals are necessary is debatable.[11–13] In any case, thrust-vector stability during operation is likely better than for chemical and other electric thrusters (measurements at the same level of accuracy have not been made on these devices), and the UK-10's thrust-vector stability will clearly meet USAF requirements.

Safety

The design pressure of xenon contained in the ARTEMIS propellant tank is 56 bar, which is low compared to most regulated chemical propellant systems. The use of extreme storage pressures is not required.

A high voltage (about 1100 V) is applied to the thruster body, obtained by dc-to-dc conversion of solar array output power. While the combination of high voltage with high current is unusual on a spacecraft, considerably higher voltages are routinely in use on most communications satellites. For the ion thruster, suitable precautions must be taken to shield the high voltage from generation to application sites. The safety issues do not constitute a serious concern, since all have been successfully managed on existing satellites and on past flight tests of ion thrusters.

The vast majority of geosynchronous communications satellites launched to date with precise N–S stationkeeping requirements carry hundreds of kilograms of hydrazine propellant (or a hydrazine derivative such as monomethyl hydrazine). The launch-site processing of hydrazine is a serious and costly safety hazard. In some cases, the replacement of satellite hydrazine with xenon propellant systems may produce a significant savings in launch processing costs and eliminate a safety problem, but the impact would be minimal in the common case where boosters or upper stages, including apogee kick motors, utilize much larger quantities of hydrazine. Eventually, routine use of new propulsion technologies for orbit transfer as well as stationkeeping may eliminate hydrazine from satellites.

Manufacturing Assessment

Manufacture of the UK-10 IPS, as of mid-1995, was on schedule for a 1997–1998 ARTEMIS launch. There are clearly some manufacturing issues that need to be worked out by Matra Marconi/DRA concerning the fabrication of the cathode and neutralizer, however. A defect in the manufacturing process has resulted in the premature failure of cathode and neutralizer heater elements used at AEA Technology and The Aerospace Corporation. This defect appears to be understood, and enough defect-free devices are available for the ARTEMIS mission. The defect in question causes heater failure only, so that the devices can still be operated using a cold starting procedure. This alternate starting procedure is undesirable in that it consumes a considerable amount of propellant and may shorten the device lifetime. Production of new cathodes and neutralizers stopped when Philips Components Limited permanently closed the production line. Matra Marconi and DRA are establishing new production lines.

SAPPHIRE modeling results and the experimental test data suggested that the triple-grid design could be improved to reduce the erosion rates of the grids and increase the thruster lifetime.[4,20,22,34] An optimized design (see Table 3) with a SAPPHIRE predicted lifetime 70% greater than the triple-grid set tested at The Aerospace Corporation has been formulated by DRA/AEA and constructed for testing.[20,24]

Evaluation of Comparable Technology

Many studies have established that high-specific-impulse electric propulsion devices constitute an enabling technology for NSSK, repositioning, and other applications. These devices expand payload mass and capability, extend mission lifetime, and/or reduce costs, provided that all issues related to their operational use can be adequately addressed.

In 1992, the only operational U.S. spacecraft system that had accepted a high-specific-impulse thruster ($\geq$500 s) for NSSK was the Telstar 4 commercial satellite. The first of these was launched in 1993, and the position of many spacecraft designers has grown much more appreciative of the potential benefits of electric thrusters. No U.S. manufacturer was making an ion thruster system for a specific satellite. Since then, Hughes Space and Communications, the largest commercial satellite builder, began a transition to operational use of the xenon ion propulsion system (XIPS-13 and XIPS-25) on its own satellites in the near term (see, for example, Ref. 45).

Table 4 compares the main features of several ion thrusters, including the SPT-100, arcjets, and bipropellant chemical thrusters. With respect to these general features, the UK-10 is competitive within the engineering trade space for north–south stationkeeping of geosynchronous spacecraft, and much better than the bipropellant chemical thruster in most respects. Less information is publicly available concerning the other thrusters, except for the SPT-100, on which much effort was expended for characterization and improvement (see, for example, Ref. 46). In view of the similarity of the UK-10 to other gridded electrostatic thrusters such as the XIPS-13, however, the test results of this FCT program aid the understanding of performance, lifetime, and spacecraft interaction potential for those devices as well.

The XIPS-13 (as well as the Japanese MELCO) should be capable of a longer accelerator-grid lifetime than the UK-10 for a similar propellant utilization, although this has not been demonstrated. Longer life is expected due to the lower current and neutral xenon density

Table 5 Summary of The Aerospace Corporation measurements performed on the T5 ion thruster

Measurement	Comments
Performance envelope	Characterization was emphasized at 25 and 18 mN, but operation at thrust levels as low as 0.5 mN and as high as 35 mN was studied briefly. The effect of facility background pressure was also studied (see Refs. 3, 15).
Thrust-vector migration	<0.15-deg migration during turn-on or change of thrust level 12–27 mN (see Refs. 12, 13 for details).
Ion flux, plasma density profiles	Measured at several thrust levels over a wide angular range (see Refs. 7, 14).
Electron, ion energy distribution	Energy spread of beam ions varies from 5 to 10 eV for 13–24 mN and 0 to ± 20 deg off the plume axis. Electron energy distribution is non-Boltzmann (see Refs. 7, 14).
Plasma potential profiles	7–13 V over the operating points and plume coordinates chosen. Peaks at 15 cm downstream (see Refs. 7, 14).
Beam divergence	$\theta_{95} = 16.2$ deg at 122 cm (see Refs. 7, 14 for details). AEA: $\theta_{95} = 12$ deg at 60–120 cm.
Charge-state distribution	Xe^{+2} and Xe^{+} fractional currents measured over wide operating range as function of angle. Ratio of Xe^{+2} : Xe^{+} highest at plume center (see Refs. 11, 14).
Plume-induced microwave phase shift	About 6 deg for 24-GHz transmission through plume center at 25 mN, 20 cm downstream (see Refs. 8–10).
Electron density distribution	Center electron density (25 mN, 20 cm downstream): 2×10^{11} cm^{-3} (see Refs. 8–10).
Grid dynamics	Telemicroscopic grid viewing, studying grid spacing and alignment dependence on operating point. Grid motion correlated with thrust-vector migration (see Ref. 13).
Fe, Fe^{+} efflux rate	Fe^{+} measured by ion mass spectrometry, and Fe by QCM combined with elemental analysis (see Refs. 4, 14).
Mo efflux, erosion rate	Integrated efflux rate is 12 g/10,000 h at Table 1 25-mN operating point (see Refs. 4, 14).
Xe density profile	Xe density has a clear maximum on centerline, even close to grids. Measured by multiphoton laser-induced-fluorescence (Ref. 5).
RF, microwave noise (EMI)	Highest noise from neutralizer at about 10 MHz (see Refs. 37, 38).
Thermal emission distribution and component temperature	Thermal images obtained indicate thermal emission profile. Approximate temperatures available after image calibration (see Ref. 6).
Total mass deposition rate profile	Measured by QCM, 21 in. from thruster, 10–50 deg off grid plane (see Ref. 4).
Spacecraft materials degradation	Δ absorptance and Δ emittance resulting from controlled exposure of 50 materials samples to ion engine environment. XPS analysis of deposit layers (see Ref. 4).
Emission spectrum (uv, visible)	Unnormalized spectrum, 200–850 nm. Xe^{+} lines dominate (see Ref. 3).
Absolute thrust level	Results slightly lower than expected, after allowing for measured thrust loss factors (see Refs. 6, 13, 14).

of the XIPS, which should produce a lower erosion rate per unit area across the grid (the grid area is 70% greater). The cost of having this longer grid life is a larger thruster, which occupies more spacecraft area. The XIPS is less variable in operating point but is also less complex, which may increase its reliability.

The arcjet, now in use for NSSK on several spacecraft, has a specific impulse and propulsion system wet mass bridging the gap between chemical thrusters and ion thrusters. Like the Teflon® pulsed plasma thruster, the thrust efficiency of the arcjets is substantially below typical ion propulsion figures, but a considerable body of flight data exists.

Computational Modeling

To more effectively utilize the relevant data produced by The Aerospace Corporation test program, a computational modeling effort was initiated to focus on the ion acceleration and ejection phase of engine operation. Charge-exchange ion production and accelerator-grid impingement, as well as certain aspects of the neutralization process and the ion production phase in the discharge chamber, were included in the scope. Although each of the relevant models is being developed for application to the UK-10 thruster, the work is of some utility for ion thrusters of all types.

Substantial progress has been achieved to-date in the following areas: 1) three-dimensional computation of ion-beam current density profiles and divergence, 2) numerical analysis of the ion-beam neutralization process, 3) computation of the downstream neutral density field and the spatial distribution of the charge exchange ion production rate, and 4) the impingement current of far-field and near-field charge exchange ions.[47] A computer code to track the motion of charge exchange ions derived from both xenon and contaminant neutrals has been developed.

The model of the discharge-chamber system will compute discharge-chamber performance curves as well as maps of plasma potential and primary and Maxwellian electron density and energy. These results will enable computation of doubly-charged-ion production rates and current densities to the wall. Sputter erosion rates and lifetimes for the various internal components (wall, screen grid, baffle disk) could then be estimated.

It would be useful to also develop a hollow-cathode model describing the current–voltage characteristic in terms of geometry and propellant flow rate. This could be combined with the discharge chamber and beam plasma models to form a model of overall ion thruster operation.

The utility of the modeling codes can be illustrated by results of the beam plasma modeling. For example, the computation of ion trajectories gave strong support to a hypothesis, inspired by experimental data, that direct impingement effects were a strong function of the accelerator-grid voltage.[4] The numerical calculation of the accelerator-grid current level is in good agreement with experimental results. The calculation indicates that comparable impingement rates apply to charge exchange ions produced in the intergrid region and downstream of the grid set.[47] The sputter yield on average will be higher for those ions formed in the intergrid region, lending support to a semiempirical determination that the accelerator-grid erosion rate is considerably higher than the product of grid current and sputter yield at the grid voltage (typically 225 V negative).[4]

Database

One of the primary goals of the UK-10 FCT program was the establishment of a comprehensive database of information. Most of the contents of this database can be found in the technical reports documenting the results of individual program elements. The database includes information produced for all aspects of the program, from mission analysis and the survey of competitive thruster characteristics to the computational modeling work and many laboratory data sets produced by the program's core effort. A complete listing of the measurements performed on the T5 thruster, and a brief comment on each, is given in Table 5.

Concluding Remarks

The UK-10 ion thruster system has been examined for stationkeeping and on-orbit repositioning of USAF satellites. The scope of the evaluation included examination of performance, projected lifetime, spacecraft integration issues, mission analysis, computational modeling, and comparative technology assessment. An extensive database was generated, which will be useful for the integration and operation of satellites using gridded electrostatic thrusters, particularly the UK-10. Propulsion system characteristics should also be considered during the satellite design phase. Included in the database is detailed information concerning performance envelope; thrust-vector migration; ion flux and floating-potential profiles; propellant charge-state distribution; electron, beam ion, neutral xenon, and charge exchange ion densities; grid dynamics; iron and molybdenum erosion rates; absolute thrust level; and spacecraft materials modification in the thruster environment. Advanced diagnostic methods were used in many instances to perform detailed measurements in real time, demonstrating that this approach has considerable utility for electric thruster characterization.

No insurmountable integration issues were found. The thruster lifetime at a 25-mN operating point with standard triple-grid configuration and 91% propellant mass utilization was projected to be about 10,000 h, adequate for many potential applications.

It was concluded that the UK-10 IPS is appropriate for NSSK, repositioning, and drag makeup missions, including precision formation flying. Milstar III, ultrahigh-frequency follow-on (UHF-FO), the Space Based Infrared System (SBIRS), and GPS IIF (Global Positioning System) are all potential users of electric propulsion technology, and the DSCS follow-on satellite system has already baselined the use of ion propulsion for NSSK. Given projected requirements, ion propulsion is a good candidate for all of these new satellite systems, although not as a retrofit.

At the inception of the UK-10 Foreign Comparative Test program, the Air Force had no alternative to chemical thrusters for on-orbit propulsion, with the exception of the unproven arcjet system. It was focused on experimental programs for high-power arcjets to perform orbit transfer missions. Similarly, NASA was focused on the development of multikilowatt ion engines for primary propulsion applications, particularly interplanetary missions. Several attractive options for N–S stationkeeping and maneuvering have become available in the last several years, and the use of ion propulsion is recommended for both military and commercial spacecraft in view of the results of this FCT program. The USAF will directly benefit from utilizing this technology, but in the past has been reluctant because of the risk. One of the current drivers to accept the risk is the dramatic increase in need to reduce costs while maintaining or improving capabilities. The UK-10 FCT program will contribute significantly to the reduction of risk in the use of gridded ion thruster technology on USAF spacecraft, because of the information and expertise obtained. It will aid system definition, up-front design, and anomaly resolution efforts for future satellites.

This test program concluded that the UK-10 IPS is an acceptable propulsion system for USAF spacecraft. Present and projected future requirements of the Air Force regarding propulsion have been considered in the analysis of UK-10 suitability.

Development programs for all future U.S. spacecraft should examine the available ion propulsion options before choosing, on a case-by-case basis, the best propulsion system. The potential benefits are too great to ignore, and this technology, subject to the verification by operational use expected within the next 2–3 years, is ripe for application.

Acknowledgments

This work was supported by the U.S. Air Force Space and Missile Systems Center, Division of Advanced Plans, under Contract F04701-93-C-0094. Funds were provided by the Foreign Comparative Test Program of the U.S. Department of Defense. D. G. Fearn provided the T5 Mk3 thruster and facilitated with N. C. Wallace, also of the British DRA, its installation and operation. The valuable, continuing assistance of Fearn and P. M. Latham of AEA Technology has been greatly appreciated. The test and evaluation team included The Aerospace Corporation investigators J. E. Pollard, S. W. Janson, R. P. Welle, L. K. Johnson, and the author, as well as P. C. T. deBoer, on sabbatical from Cornell University, and L. N. Ahmed, in a one-year Education with The Aerospace Corporation program. These investigators were supported by R. H. Ueunten, E. M. Yohnsee, E. W. Fournier, and L. S. Garman. The work was performed in the Propulsion and Environmental Chemistry Section, managed by R. B. Cohen. The modeling effort has been conducted by P. Wilbur at Colorado State University, and monitored by V. Rawlin at NASA. This work was presented as Paper 95-91 at the 24th International Electric Propulsion Conference, Moscow, Russia.

References

[1]Janson, S. W., "The On-Orbit Role of Electric Propulsion," AIAA Paper 93-2221, June 1993.

[2]Crofton, M. W., "A Comprehensive Test and Evaluation Program for the UK-10 (T5) Ion Engine," International Electric Propulsion Conf., IEPC Paper 93-104, Sept. 1993.

[3]Crofton, M. W., "Test and Evaluation of the T5 (UK-10) Ion Engine: Program Status," AIAA Paper 94-2847, June 1994.

[4]Ahmed, L. N., and Crofton, M. W., "Surface Modification Measurements in the T5 (UK-10) Ion Thruster Environment," AIAA Paper 95-2827, July 1995.

[5]Crofton, M. W., "Laser Spectroscopic Study of the T5 (UK-10) Ion Thruster," AIAA Paper 95-2921, July 1995.

[6]Crofton, M. W., Ueunten, R. H., Harris, L., Yohnsee, E. M., and Cohen, R. B., "Calibrated Infrared Imaging for Electric Thrusters," AIAA Paper 96-2977, July 1996.

[7]deBoer, P. C. T., "Electric Probe Measurements in the Plume of the UK-10 Ion Thruster," International Electric Propulsion Conf., IEPC Paper 93-236, Sept. 1993.

[8]Janson, S. W., "Microwave Diagnostics for Ion Engine Plumes," International Electric Propulsion Conf., IEPC Paper 93-237, Sept. 1993.

[9]Janson, S. W., "Microwave Interferometry for Low Density Plasmas," AIAA Paper 94-2424, June 1994.

[10]Janson, S. W., "Microwave Interferometry for Ion Engines," AIAA Paper 94-2741, June 1994.

[11]Pollard, J. E., "Plume Measurements with the T5 Xenon Ion Thruster," AIAA Paper 94-3139, June 1994.

[12]Pollard, J. E., "Beam-Centroid Tracking Instrument for Ion Thrusters," *Review of Scientific Instruments*, Vol. 65, No. 12, 1994, pp. 3733–3738.

[13]Pollard, J. E., and Welle, R. P., "Thrust Vector Measurements with the T5 Ion Engine," AIAA Paper 95-2829, July 1995.

[14]Pollard, J. E., "Plume Angular, Energy, and Mass Spectral Measurements with the T5 Ion Engine," AIAA Paper 95-2920, July 1995.

[15]Crofton, M. W., "UK-10 FCT Program Test and Evaluation Report," The Aerospace Corp., TOR-94(4507)-2, El Segundo, CA, Aug. 1995.

[16]Chan, A. K., Zondervan, K. P., and Smith, W. B., "Solar Electric Propulsion Assessment," The Aerospace Corp., TOR-93(3501)-2, El Segundo, CA, Dec. 1993.

[17]Fearn, D. G., "Ion Propulsion—a Technology for Improving the Cost-Effectiveness of Large Communications Satellites," *Electronics and Communication Engineering Journal*, Vol. 4, No. 3, 1992, pp. 153–162.

[18]Vaughan, C. E., and Cassady, R. J., "An Updated Assessment of Electric Propulsion Technology for Near-Earth Space Missions," AIAA Paper 92-3202, July 1992.

[19]Pollard, J. E., Jackson, D. E., Marvin, D. C., Jenkin, A. B., and Janson, S. W., "Electric Propulsion Flight Experience and Technology Readiness," AIAA Paper 93-2221, June 1993.

[20]Bond, R., and Latham, P., "The Sapphire Compute Code and Its Use for Ion Thruster Grid Design and Erosion Calculations," AIAA Paper 95-2923, July 1995.

[21]Fearn, D. G., "Ion Thruster Lifetime Limitations Imposed by Sputtering Processes," International Electric Propulsion Conf., IEPC Paper 93-177, Sept. 1993.

[22]Fearn, D. G., "Life-Testing of the UK-10 Ion Propulsion System," UK Defence Research Agency, DRA/CIS(CSC3)/CR/94/43, Farnborough, England, UK, Dec. 1994.

[23]Fearn, D. G., private communication, UK Defence Research Agency, England, UK, June 1995.

[24]Wallace, N., private communication, UK Defence Research Agency, England, UK, April 1996.

[25]Shimada, S., Satoh, K., Gotoh, Y., Nishida, E., Noro, T., Takegahara, H., Nakamaru, K., and Nagano, H., "Ion Thruster Endurance Test Using Development Model Thruster for ETS-VI," International Electric Propulsion Conf., IEPC Paper 93-169, Sept. 1993.

[26]Marrian, C. R. K., and Shih, A., "The Operation of Coated Tungsten-Based Dispenser Cathodes in Nonideal Vacuum," *IEEE Transactions on Electron Devices*, Vol. 36, No. 1, 1989, pp. 173–179.

[27]Sarver-Verhey, T. R., "Continuing Life Test of a Xenon Hollow Cathode for a Space Plasma Contactor," AIAA Paper 94-3312, June 1994; also private communication, April 1996.

[28]Kerslake, W. R., and Ignaczak, L. R., "Development and Flight History of SERT II Spacecraft," *Journal of Spacecraft and Rockets*, Vol. 30, No. 3, 1993, pp. 258–290.

[29]Garner, C. E., Brophy, J. R., and Pless, L. C., "Metal and Metal Nitride Erosion Rates in a Divergent Field Ion Engine," International Electric Propulsion Conf., IEPC Paper 91-097, Oct. 1991.

[30]Berisch, R. (ed.), *Sputtering by Particle Bombardment I: Physical Sputtering of Single-Element Solids*, Vol. 47, Topics in Applied Physics, Springer–Verlag, New York, 1981.

[31]Dunster, R., private communication, Matra Marconi U.K., July 1995.

[32]Shimada, S., Satoh, K., Gotoh, Y., Nishida, E., Noro, T., Takegahara, H., Nakamaru, K., and Nagano, H., "Development of Ion Engine System for ETS-VI," International Electric Propulsion Conf., IEPC Paper 93-009, Sept. 1993.

[33]Fearn, D. G., "Evaluation and Test of Spacecraft/Xenon Ion Thruster Interfaces," INTEL-920 CR, UK Defence Research Agency, DRA/CIS/CSC3/WP94004/1, Farnborough, England, UK, Jan. 1994.

[34]Watson, S. D., Hurford, P., Martin, A. R., Banks, C. R., Eaton, R. W., Harvey, M. S., Moulford, W. E. F. L., and Pierce, A. J., "500 Hour Tests of the T5 Ion Thruster with Dual and Triple Grid Extraction Systems," International Electric Propulsion Conf., IEPC Paper 93-170, Sept. 1993; also Martin, A. R., Banks, C. R., Eaton, R. W., Hurford, P., Moulford, W. E. F. L., and Pierce, A. J., "Erosion Measurements for Two- and Three-Grid Ion Thruster Extraction Systems," International Electric Propulsion Conf., IEPC Paper 93-171, Sept. 1993.

[35]Power, J. L., "Ground Correlation Investigation of Thruster/Spacecraft Interactions to be Measured on the IAPS Flight Test," International Electric Propulsion Conf., IEPC Paper 84-39, May 1984.

[36]Porte, F., Saint-Aubert, P., Mawby, D., and Hsing, J., "Application of Ion Propulsion System to Communications Satellites," International Electric Propulsion Conf., IEPC Paper 93-015, Sept. 1993.

[37]Chandra, S., Mawdsley, F., Brown, R., Watson, S., and Fearn, D. G., "Measurements of the Electromagnetic Emissions from the T5 Ion Thruster," International Electric Propulsion Conf., IEPC Paper 93-234, Sept. 1993.

[38]Johnson, L. K., and Janson, S. W., Aerospace Corp., El Segundo, CA (unpublished data).

[39]Sovey, S. J., Zana, L. M., and Knowles, S. C., "Electromagnetic Emission Experiences Using Electric Propulsion Systems," *Journal of Propulsion and Power*, Vol. 5, No. 5, 1989, pp. 534–547.

[40]Ackerknecht, W. E., and Stanton, P. H., "Effects of an Ion-Thruster Exhaust Plume on S-Band Carrier Transmission," *Journal of Spacecraft and Rockets*, Vol. 14, No. 7, 1977, pp. 445, 446.

[41]Brukhty, V. I., and Kirdyashev, K. P., "Electromagnetic Interference of Stationary Plasma Thruster," International Electric Propulsion Conf., IEPC Paper 93-147, Sept. 1993.

[42]Carruth, M. R., and Brady, M. E., "Measurement of the Charge-Exchange Plasma Flow from an Ion Thruster," *Journal of Spacecraft and Rockets*, Vol. 18, No. 5, 1981, pp. 457–461.

[43]Samanta Roy, R. I., Hastings, D. E., and Gatsonis, N. A., "Numerical Study of Spacecraft Contamination and Interactions by Ion-Thruster Effluents," *Journal of Spacecraft and Rockets*, Vol. 33, No. 4, 1996, pp. 535–542.

[44]Conger, J. C., and Hastings, D. E., "Active Control of Contaminant Particulates in a Low-Earth-Orbit Near-Spacecraft Environment," *Journal of Spacecraft and Rockets*, Vol. 32, No. 1, 1995, pp. 184, 185.

[45]Ashley, S., "Electric Rockets Get a Boost," *Mechanical Engineering*, Vol. 117, No. 12, 1995, pp. 61–65.

[46]Garner, C. E., Brophy, J. R., Polk, J. E., and Pless, L.C., "A 5,730-hr Cyclic Endurance Test of the SPT-100," AIAA Paper 95-2667, July 1995; also Day, M., Maslennikov, N., Randolph, T., and Rogers, W., "SPT-100 Subsystem Qualification Status," AIAA Paper 95-2666, July 1995.

[47]Monheiser, J., Kameyama, I., and Wilbur, P., manuscript in preparation.

J. A. Martin
Associate Editor