Orbit-Raising, Past and Present - the X-Series of Spacecraft and Artemis

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Abstract: Since the advent of the space age, theoretical analyses have shown very clearly that there are significant cost advantages if spacecraft are launched into a low Earth orbit or to some intermediate parking orbit, then are raised to their operational altitude by a low thrust electric propulsion system. Owing to the high specific impulse available from an electric thruster, this procedure allows a smaller, less costly launch vehicle to be utilised, at the expense of a longer transit time. However, there have been very few examples of the use of this technique to date, despite plans to demonstrate its viability in the early 1970s. Those plans involved the UK X-series of spacecraft, which are discussed in this paper. The 5th in the series was intended to exhibit orbit-raising, employing a gridded ion thruster, but was cancelled before this was accomplished. The space community then had to wait for the Artemis communications satellite mission, launched in July 2001, for an actual demonstration of the use of ion thrusters for this purpose.

I. Introduction

During the 1960s, when the UK possessed a vigorous and extensive space programme, with enthusiastic support from the governments then in power, the Royal Aerospace Establishment (RAE) at Farnborough employed some of the most innovative scientists involved in space science and technology within the western world. In the context of this paper, they were the first to realise that low thrust electric propulsion (EP) devices, operating at high specific impulse (SI), would permit the very effective transfer of a spacecraft from a parking orbit to its operational altitude for the expenditure of a tiny fraction of the propellant required using standard chemical engines. The theoretical aspects of this orbit-raising technique using EP were gradually refined and were published in numerous papers in the scientific literature[1-6]. The expansion of elliptical orbits was included[6], as were interplanetary missions[2]. It is interesting to note that, even at this early date, the European (Space Vehicle) Launcher Development Organisation (ELDO) was sponsoring studies of the commercial applications of this general concept[7].

In parallel with this theoretical effort, in the late 1960s the RAE proposed that a series of small spacecraft be designed and flown to demonstrate this concept; this was the 'X' series of satellites[8], eventually scheduled to be launched by the UK's own small rocket, the Black Arrow[9]. This vehicle, with a payload capability to low Earth orbit of up to 150 kg, was developed in parallel with the smaller Black Knight sub-orbital rocket[9], the Blue Streak heavy lift launcher[10] and the Skylark sounding rocket[11], all of which were successful. Indeed, the last launch of Skylark occurred on 30 April 2005[12].

The first two Black Arrow payloads were not intended to be placed into orbit and were designed to monitor the performance of the launcher. The third payload, the X3 technology demonstration satellite, was scheduled to be launched, and was placed into a 556 km × 1570 km orbit with an inclination of 82° by a Black Arrow on 28 October 1971[13]. Designed and built by RAE, with assistance from UK industry, this successful 66 kg satellite was intended to demonstrate the viability of all platform technologies, such as communications, power generation and data

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storage. Sadly, the Black Arrow launcher was cancelled before this one successful launch, but it was able to proceed as planned. The satellite was named Prospero once safely in orbit[14].

Despite this cancellation, the development of the X4 spacecraft continued at RAE. It was intended to demonstrate in orbit the three-axis attitude control system required for an ion thruster-propelled orbit-raising manoeuvre and, most significantly, one of the most advanced solar arrays yet devised[8], which utilised a unique pneumatically driven telescopic tube system for deployment. This satellite was successfully launched by a US rocket on 9 March 1974[15] into a 714 km × 916 km orbit with an inclination of 97.8°, and the array deployment and operation were entirely satisfactory. It was named Miranda in orbit.

This success encouraged continuation of the development of the X5 spacecraft, which was to utilise a higher power version of the same flexible solar array. This programme was also undertaken by Space Department of the RAE, commencing in 1967. The orbit-raising function was to be entrusted to the gridded ion thruster being developed simultaneously by RAE[16-18], with an initial aim of increasing the altitude from 500 km to 1000 km within the first 20 to 30 days of the mission. To accomplish this, the thruster was to operate at 15 mN thrust and an SI of 3000 s, using mercury propellant and consuming 500 W.

However, lack of government interest and thus of funding caused cancellation of this innovative and exciting project, and we have had to wait until the new millennium to see any implementation of these ideas, with the successful orbit-raising exploits of the Artemis[19,20] and SMART-1[21] spacecraft. In the former case, a partial launch vehicle failure left this communications satellite in an incorrect and useless orbit. It was only the availability of the gridded ion thrusters on board, which had been intended to be used for north-south station-keeping (NSSK), that enabled the mission to be saved. With great innovation by the spacecraft controllers, these thrusters were used to raise the orbit to geostationary[20], thereby permitting the primary mission objectives to be met. It should be noted that one of the thrusters utilised on Artemis for this purpose was the T5[22], which was derived directly from the T4 and T4A devices[17,18] originally intended for the X5 satellite. Thus, after a wait of nearly three decades, these gridded ion thrusters were at last able to demonstrate their orbit-raising capabilities, but using xenon rather than mercury as the propellant.

This paper covers this series of missions, together with the development of the associated technologies, in some detail. However, the greatest innovations were clearly the solar array and the ion thruster, so most attention is paid to these topics, and to their relationship to present and future missions.

II. The X3 and X4 Spacecraft

A. The X3 Satellite

The X3 spacecraft was designed and developed by RAE, aided by UK Industry, and was intended to demonstrate several of the technologies required for the later more challenging missions. It followed the X1 and X2 payloads, which were suborbital and were flown to aid launcher development. The technologies flown on X3 naturally included all platform functions, but special attention was paid to the data acquisition and telemetry system, the 3000 silicon solar cells, the thermal control system and the on-board tape recorder. Various experiments were flown, including a micrometeoroid monitor.

As indicated in Fig 1, the spacecraft, named Prospero once in orbit, was configured as a 26-faced polygon, with a length of 0.7 m and a diameter of 1.12 m. Its mass was 66 kg, well within the capabilities of Black Arrow[9]. It was spin stabilised, included a battery for operation during eclipse, and the telemetry system operated at 137 MHz.

The spacecraft was launched successfully from Woomera in Australia on 28 October 1971[13] and was placed into a 556 km × 1570 km orbit with an inclination of 82°. It was controlled by the RAE's Data Centre at Farnborough and the associated Lasham Ground Station. The ground segment included an innovative early use of automation, with EMR 6130 computers in the Data Centre. These were each equipped with 24K core memories, a 16 Mbit disc drive and tape units.

All spacecraft systems operated successfully for more than a year[14], although difficulties were initially experienced

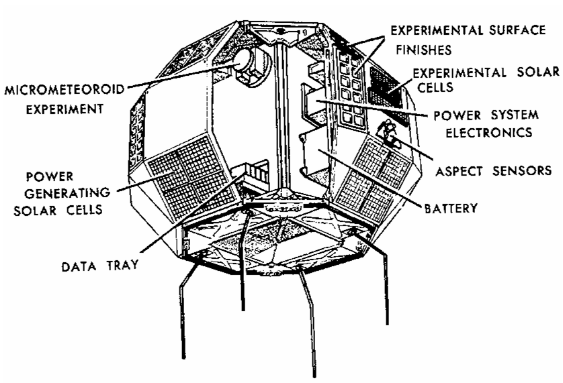


Figure 1. Diagram of the Prospero spacecraft

with the on-board tape recorder. Prospero was then used as a development and training aid in preparation for the launch of X4.

B. The X4 Satellite

The X4 satellite, named Miranda in orbit, was launched on 9 March 1974[15] into a 714 km × 916 km orbit with an inclination of 97.8°, although the cancellation of Black Arrow necessitated the use of a US rocket for this purpose. Its mass was about 93 kg and it was three-axis stabilised, utilising innovative cold propane gas attitude control thrusters and sun and Earth sensors. As indicated in Fig 2, its configuration resembled that of later communications satellites, with long solar array panels on each side of a central body. The latter had dimensions of 0.82 m × 0.66 m × 0.66 m and the span across the array was nearly 9 m. A more detailed diagram, which includes some main dimensions, is shown in Fig 3; in this, the solar arrays are in their stowed position.

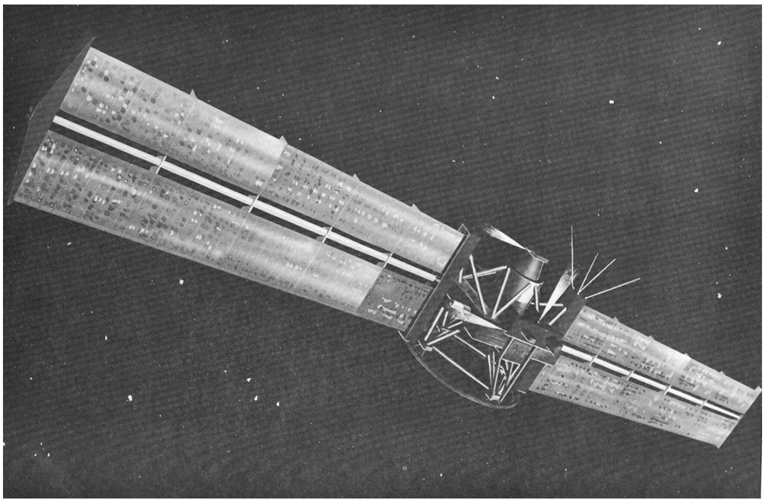


Figure 2. Artist's impression of the X4 spacecraft (RAE picture).

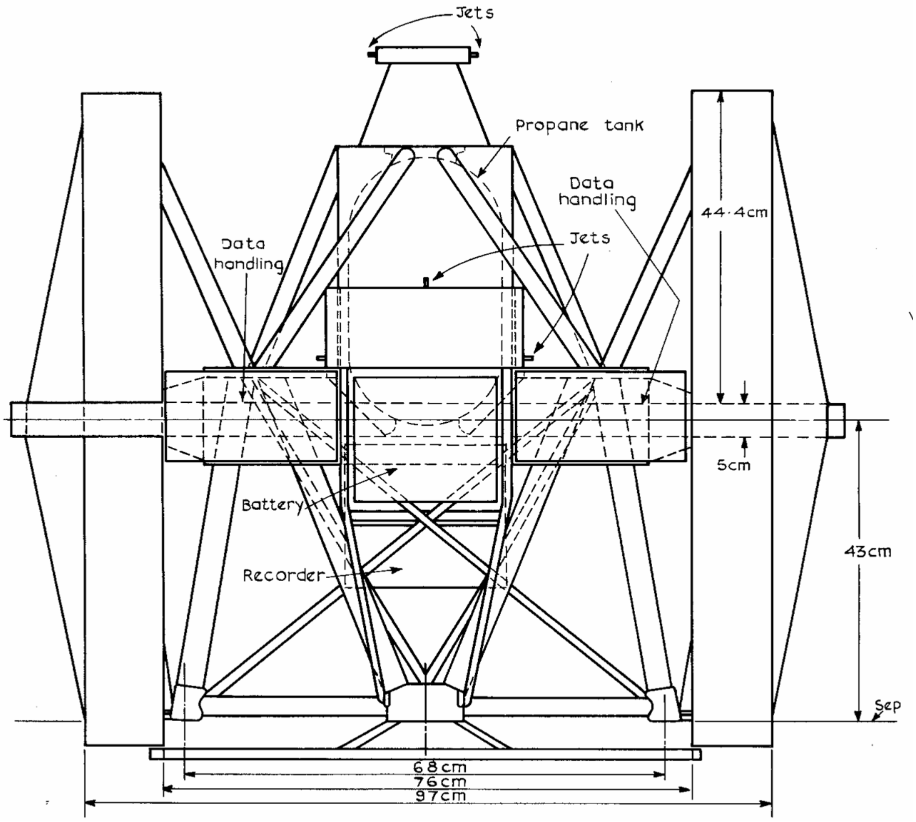


Figure 3. Drawing of the X4 spacecraft with the arrays stowed (RAE diagram).

C. The X4 Solar Array

As mentioned earlier, the array flown successfully on X4 was innovative, in that it was of flexible design, with pneumatic deployment[8]. It is shown in diagrammatic form in Fig 4, which also includes a photograph of one wing of the array after deployment in the laboratory. It should be emphasised that this array was actually dimensioned for the X5 spacecraft, on which it was intended to supply the power required by the ion thruster. As this thruster was not included on X4, the power requirement was much lower, so the array was populated with only 7440 2 cm × 2 cm Si cells of 125 µm thickness, rather than the 14,880 specified for X5. The other spaces were occupied by dummy cells. Each panel was 4.18 m long and 0.91 m wide, and the total area was 7.6 m^2.

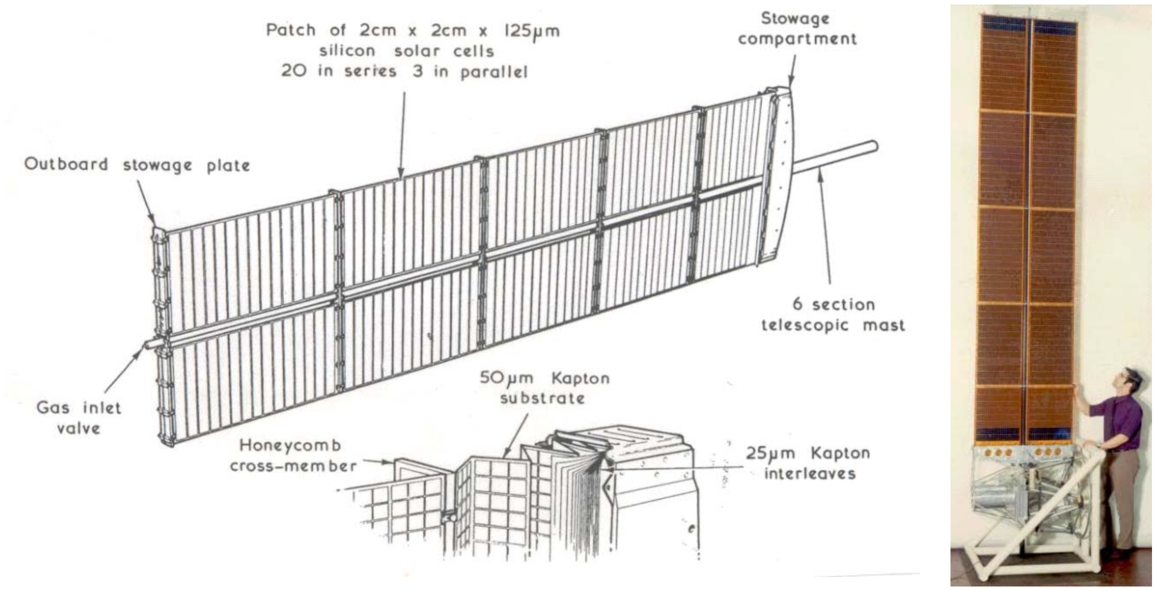


Figure 4. X4 spacecraft solar array (RAE diagram and photo).

The solar cells, of 7.7% efficiency at beginning of life, were mounted on a flexible Kapton substrate of 50 µm thickness and each was protected by a 100 µm thick cover glass. The substrate was broken into separate sections by the central deployment boom and aluminium honeycomb cross members. Each section contained cells connected in a series and parallel arrangement to provide an output of 56 V and 500 W for the ion thruster in the X5 version[23], together with two auxiliary supplies of 7 V and 14 V, with a power capability of 9 W and 45 W, respectively. The total output was calculated to be 561 W in the case of the fully populated X5 array and 310 W for the version flown on X4.

As indicated in Figs 4 and 5, the flexible substrate was arranged to fold concertina-fashion into the stowage compartment. Thin Kapton interleaves of 50 µm thickness were inserted between each pair of layers to prevent damage. This arrangement withstood severe vibration testing[24] in the laboratory, reaching 40 g, and also the actual launch environment.

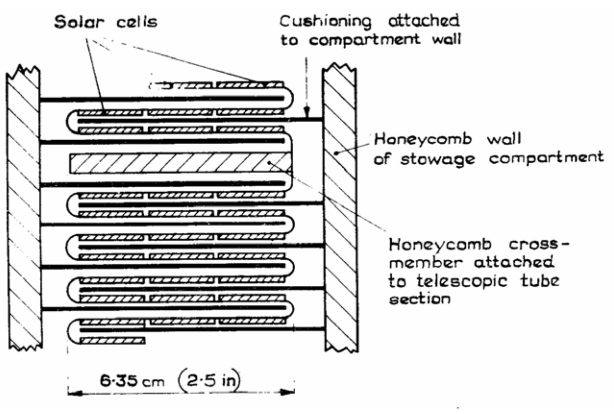


Figure 5. Illustration showing solar array stowage technique (RAE diagram).

The deployment mechanism is illustrated in Fig 6. It consisted of 6 thin-walled aluminium tubes fitted into each other to form a telescopic mast. Each tube was about 1 m long and of 0.4 to 0.5 mm wall thickness, and was provided with a gas seal. The smallest was of 35 mm outside diameter and the diameter of each succeeding tube was increased by 3.2 mm. The complete assembly fitted within a 50 mm diameter outer tube mounted within the body of the spacecraft. Each tube terminated at one of the honeycomb cross members shown in Fig 4. Nitrogen gas contained within the inner tube at a pressure of 2 atmospheres caused deployment at a controlled rate when a pyrotechnic release mechanism was triggered.

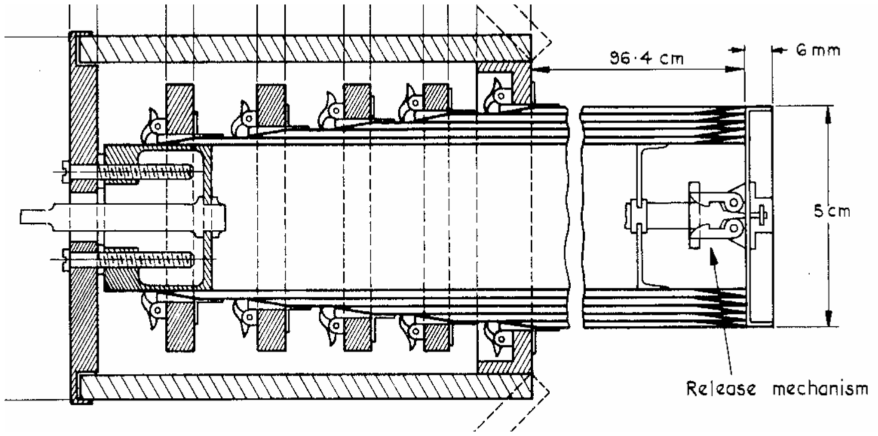


Figure 6. Illustration of the telescopic mast principle (RAE diagram).

It is interesting to compare the proven performance of this array with the equivalent values which would be obtained using modern solar cells of much greater efficiency. This comparison is given below in Table 1, in which it is assumed that the dummy solar cells were of the same mass as the actual devices. The mass quoted for the original arrays[23] includes the complete deployment mechanism and also the stowage compartments. From Table 1, it can be seen that this technology remains close to the state-of-the-art, with values of the power/mass ratio as high as available from almost any other design. Thus the fully populated array, with its mass increased to account for the greater mass of modern cells and the heavier conductors required to carry the additional output power, will produce up to 2 kW, with a power/mass ratio of 130 W/kg and a power per unit area of 268 W/m^2. For comparison, the 2.5 kW array on the Muses-C spacecraft[25] provides 53 W/kg and 236 W/m^2.

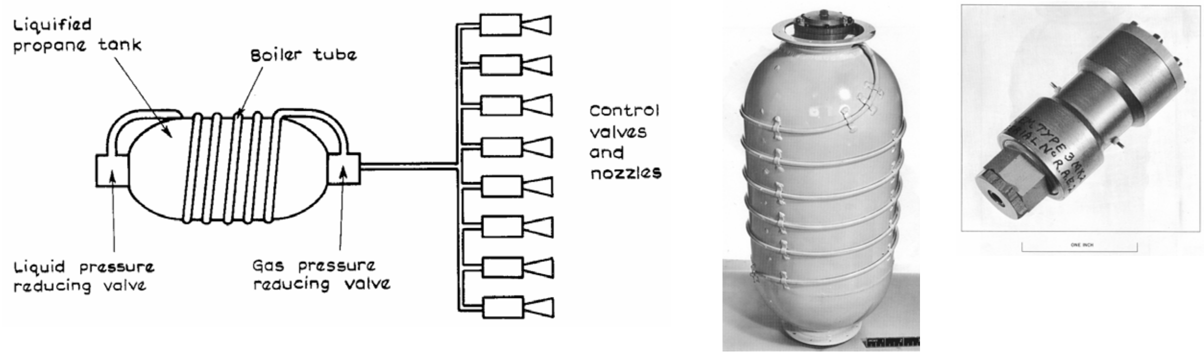
Table 1. Performance characteristics of the solar arrays designed for the X4 and X5 spacecraft with original and modern solar cells.

Cell Type	Number of Cells	Cell Efficiency	Mass (kg)	Output (W)	Power/Mass Ratio (W/kg)
X4 ARRAY (partially populated with solar cells)					
Original Si	7440	7.7	11.83	310	26.2
High Eta	7440	16	12.97	644	49.7
GaAs/Ge	7440	19	15.22	765	50.3
Triple Junction	7440	28	15.63	1127	72.1
X5 ARRAY (fully populated with solar cells)					
Original Si	14,880	7.7	11.83	561	47.4
GaAs/Ge	14,880	19	15.22	1384	91.0
Triple Junction	14,880	28	15.63	2040	130.5

D. Propane Propulsion System

Another innovation which was proved during the flight of the X4 spacecraft was the use of a propane cold gas thruster system for attitude control[8]; a schematic diagram of this is shown in Fig 7a.. This was very simple, with the major advantage that the propellant gas was stored at high density without the need for high pressure; the maximum in flight was 330 psi. Thus an aluminium tank of relatively low mass was satisfactory (see Fig 7b). This had a helical propellant flow tube brazed to its external surface. As the propane flowed through this tube, initially as a liquid, it evaporated, taking its latent heat from the tank. The tank was of 21 cm diameter and 42 cm length, and had a mass of 1.22 kg for a capacity of 4.77 kg. The maximum propellant flow rate was 0.33 g/s.

This system proved to be very effective, although the SI provided was not large at 62 s. The maximum power input was only 4 W and the risetime to full thrust 4 ms. The latter parameter was achieved by the propellant control solenoid valve in each thruster branch. This valve, shown in Fig 7c, incorporated the electroformed nozzle (at the bottom of the photograph). The valve/nozzle assembly was 62 mm long and had a mass of 100 g. At the design input pressure of 10 psi, the thrust was 46 mN. To achieve this input pressure two reducing valves were employed. Each had a diameter of 66 mm, a mass of 190 g, and the control accuracy achieved by the gas output device was ±0.5 psi.



a. b. c.

Figure 7. The propane propulsion system flown on the X4 spacecraft: a. System schematic. b. Propellant tank. c. Propellant control valve (RAE diagram and photos).

III. The X5 Mission

As stated previously, the major aim of the X5 mission was to test the orbit-raising principles which had earlier been explored in depth theoretically. To do this, the design of the spacecraft was changed as little as possible from that successfully demonstrated by X4[8]. In particular, the solar array was identical apart from fully populating it with active solar cells and altering the wiring scheme to cater for the relatively high power demand of the ion thruster. The latter also necessitated an upgrade of the power conditioning and distribution system.

The initial flight objective was modest, being to raise the altitude from the initial 500 km to 1000 km in 20 to 30 days. This was imposed by an unnecessary restriction to operating only with continuous thrust; the reason for this is now obscure, and is hard to understand since the early version of the ion thruster to be flown, the T2[16,26], had by that time demonstrated an excellent restart capability. A thrust of 15 mN was initially assumed, with a SI of 3000 s, using mercury propellant; both values were shown to be entirely feasible, although the thrust objective was later reduced to 10 mN for NSSK after the cancellation of X5. The thruster system, when operating at 15 mN, was intended to be supplied with 500 W, leaving about 60 W from the array for housekeeping functions.

Apart from operation of the thruster, the most challenging aspect of the mission was anticipated to be attitude control, since it was necessary to maintain the array as closely normal to the sun vector as possible while thrusting to achieve the desired power output. Moreover, for efficient orbit-raising, the thrust vector had to be simultaneously aligned with the velocity vector of the satellite. Mission analysis suggested that this could best be accomplished by using an orbital inclination of 81°.

Secondary objectives of the mission were to check that the measured thrust equaled the value calculated from the ion beam parameters, and to assess the impact of the thruster on other spacecraft systems, such as communications and sensors. There was also some residual concern about the possible deposition of Hg on spacecraft surfaces, in particular on the solar array. It was also anticipated that information concerning the stability of the thrust vector might be obtained. It should be mentioned that the earlier SERT-2 mission[27] in the USA had dispelled the more important concerns, since that flight had demonstrated conclusively that ion thrusters using Hg propellant cause no detectable adverse interactions with a spacecraft.

IV. The X5 Mission Ion Thruster

The X5 mission originated in the theoretical orbit transfer analyses performed at RAE[1-6], together with the original suggestion[28,29] in 1962 to work in the EP field by Bryan Day, also of the RAE, with the support of his Division Head, Allan Earl, and the Head of Space Department, George Burt. Although this suggestion initially favoured cesium bombardment[28] and plasma thrusters[29], it led to an experimental programme to study a 10 cm diameter electron bombardment gridded ion thruster[29], designated T1, with tests commencing in early 1968. Mercury was selected as the propellant, to take advantage of its high atomic mass and ease of storage. The T1 thruster was followed by the much improved T2[26], which incorporated hollow cathode electron sources[30] for both the main discharge and the neutraliser[31]. The next step was to design a space-compatible thruster, the T4[17], then the improved T4A[18], with the subsequent T5[32] being accepted for flight in the NSSK role on ESA's L-Sat communications satellite[33]. Unfortunately, funding problems caused development to be cancelled in 1978, so the L-Sat application was abandoned; it was resurrected many years later in the context of the Artemis programme[19].

E. The T1 and Elliott Brothers' Thrusters

Bryan Day's successful suggestion[28,29], in 1962/3, that RAE should commence a programme to develop an EP system was initially based on the perceived need at that time to provide an attitude control and station-keeping capability. However, a later priority became the requirement for a high energy upper stage[23] for the Black Arrow launcher[9], which resulted in the design of the X5 spacecraft. Interestingly, as already mentioned, at that time ELDO was considering the augmentation of the payload capability of its launch vehicle by the same means[7].

This first design, the T1[34], was influenced by US studies by Kaufman[35] and by the work on gridded ion accelerators performed at Fort Halstead by Clayden and Hurdle[36]. A diagram of their ion source is shown in Fig 8 and a section through the T1 in Fig 9. Fig 10 is a photograph of T1. As a result of this earlier experience, a filament electron source was initially employed, together with an axial magnetic field, and the importance of adequate beam neutralisation was recognised[28,36]. A porous tungsten vaporiser produced and controlled the flow of mercury vapour propellant, but the single accelerator (accel) grid was replaced by a twin grid system soon after testing began in early 1968. An ion accelerating potential of about 1.5 kV was chosen, giving an SI of close to 3000 s at 15 mN thrust.

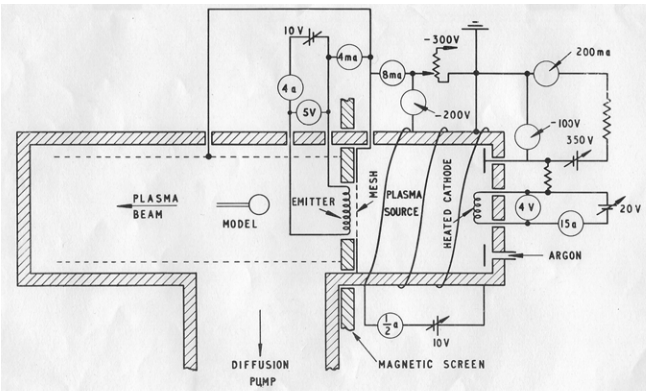


Figure 8. Ion source designed by Fort Halstead.

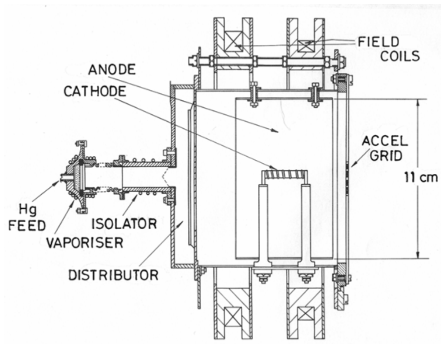


Figure 9. The initial configuration of the T1 thruster (RAE Diagram).

Somewhat earlier and independently, work commenced in the same field at Elliott Brothers[37] (London) Ltd (later to become part of Marconi Space and Defence Systems (MSDS)) at Frimley, near Farnborough; this company later played a major role in the L-Sat programme[33]. Their development was also aimed at orbit transfer applications, and followed the initial operation of a small laboratory thruster as early as 1963, using mercury propellant. A diagram of one of their earlier designs, using a twin grid ion extraction system, is reproduced in Fig 11. However, this independent programme was discontinued in about 1970, owing to lack of funding, but only after successfully demonstrating a 500 W thruster using a hollow cathode electron source and a 1.6 kV beam accelerating potential[38].

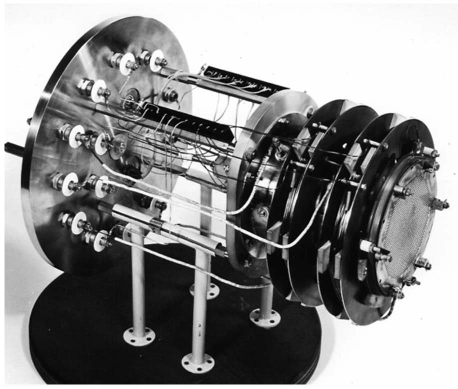


Figure 10. Photograph of the T1 thruster (RAE photo).

F. The T2 and Culham Laboratory Thrusters

At the RAE, the success of the initial tests with the T1 thruster resulted in the design of a new device[26], the T2 (Fig 12), for which a 10 cm beam diameter was selected to provide a reduced thrust of 10 mN, with a beam accelerating potential of 2 kV. The design incorporated a hollow cathode electron source[30] (Fig 13) and employed the primary electron accelerating principle established by Harold Kaufman[39] at the NASA Lewis Research Center in the early 1960s. In this, as indicated in Fig 12, the axial cathode emits electrons into a coupling plasma contained within a cylindrical inner magnetic polepiece. The opening from this polepiece is partly blanked off by a circular disc, the baffle. The electrons, in emerging through the annular gap between these components, have to cross a magnetic field with a strong radial component. This impedes their motion and causes them to gain energy. This configuration, coupled with the use of the correct value of the magnetic field, allows the energy to be adjusted to maximise the rate of ionisation within the discharge chamber[40].

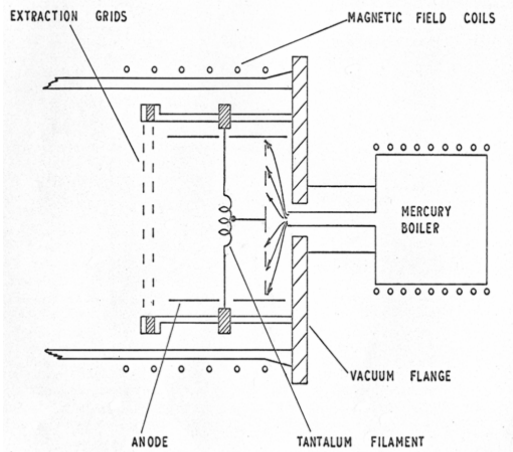


Figure 11. An early Elliott Brothers thruster.

As it is necessary to change the magnetic field to suit performance requirements, such as thrust and efficiency, the decision was made in designing the T2 thruster to employ solenoids rather than permanent magnets, and this policy has been retained to the present day. It has, amongst other advantages, permitted extremely wide throttling ranges to be achieved with all the thrusters developed in the UK[41].

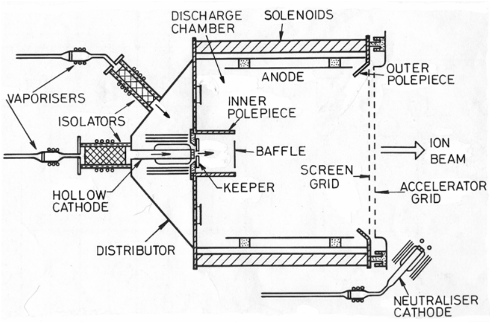


Figure 12. Sectional diagram of the T2 thruster.

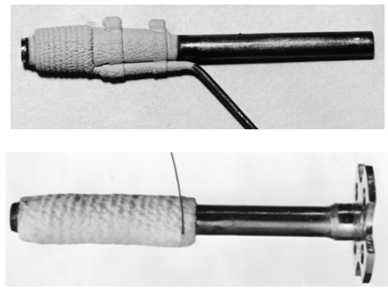


Figure 13. Hollow cathodes for the T2 thruster. Top: made by GEC. Bottom: made by RAE (RAE photos).

The hollow cathodes fitted to this thruster were based initially on those designed in the USA for ion beam neutralisation[42] purposes and for the discharge chamber of the SERT II thruster[27]. The first versions were made by GEC (Fig 13, top), but the RAE soon set up its own manufacturing and test facilities in 1968. An example of an RAE cathode is shown at the bottom of Fig 13. This has a single-ended heater winding, with the return current passing through the cathode body, but all later versions, to the present day, have employed bifilar windings insulated from the body. In parallel with the cathode work at RAE[30], in 1968 Mullard Ltd at Mitcham commenced a major programme of research and development in this field[43], which culminated in cathodes and neutralisers designed for the thrusters to be flown on the L-Sat mission.

The T2 thruster exhibited a much increased performance[26], with a notable early achievement being the realisation that the open area ratio of the grid system has a marked effect on efficiency. It was shown that, at a propellant utilisation efficiency of 80%, the discharge losses were reduced from 540 W/A to 260 W/A by increasing the open area ratio from 50% to 75%. An electrical efficiency of 80.7% was achieved with an input power of 433 W and thrust of 10 mN. One of three T2 thrusters was subjected to life-tests[44] in the early 1970s, achieving about 2000 h, and hollow cathodes and vaporisers were tested for longer times.

While this work was underway, fundamental discharge chamber and ion beam investigations were being undertaken at the UKAEA's Culham Laboratory, using the diagnostic thruster design[45] shown in Fig 14 These studies benefited from the wide plasma physics diagnostics expertise at Culham, together with a vacuum facility modified specifically for this purpose. Initially, the particle fluxes to all parts of the thruster were measured under widely varying conditions[45], which aided considerably the future design process. Later work, on a special diagnostic version of the T4A thruster[18], extended the investigations to cover virtually all physical processes[46], including the acceleration of primary electrons[40], the production of doubly charged ions, the ion extraction and acceleration processes, ion beamlet vectoring and ion beam neutralisation[47]. The accumulated expertise allowed Culham to bid successfully for a major Intelsat contract for thruster testing in 1976, but this was withdrawn at the last moment.

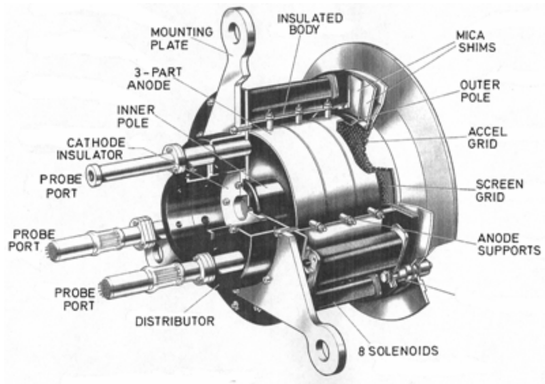


Figure 14. Sectional diagram of the Culham 15 cm thruster (Culham diagram).

Work was also progressing in parallel at the City University in London, where the physical processes in a simple laboratory ion engine were being studied with a variety of diagnostics[48]. Argon was used as the propellant, and much valuable information was gleaned from this programme, which eventually influenced subsequent work at Culham. Another significant contribution was made at this time by the University of Liverpool, where the electrical breakdown of mercury vapour was investigated in detail[49], to aid in the design of electrical isolators (Fig 12).

G. The T3, T4 and T4A Thrusters

Using the experience gained from the T1 and T2 programmes, coupled with the plasma and ion beam diagnostics carried out at Culham, RAE then designed the T3 thruster, which was constructed in Industry in 1971, but was replaced a year later by the T4[17], prior to the commencement of any testing. A sectional view of the T4 is shown in Fig 15 and a photograph in Fig 16. As can be seen, the design featured a conical discharge chamber and an integrated propellant tank; the neutraliser is excluded from the diagram. The small size of the tank is interesting, since the equivalent xenon tank would have 10 to 15 times the volume, depending upon the storage pressure. Cathodes and neutralisers were procured from Mullard[43]. Unfortunately, in testing the conical discharge chamber was found to cause the primary electrons to reach the anode too easily, and the propellant utilisation efficiency was low as a result. As a consequence of this problem, the T4A[18] variant reverted to a cylindrical discharge chamber; a photograph of this, including the neutraliser, is shown in Fig 17. As in Fig 16, the outer earth screen is removed in this picture to show the details of the discharge chamber, solenoids and grid system.

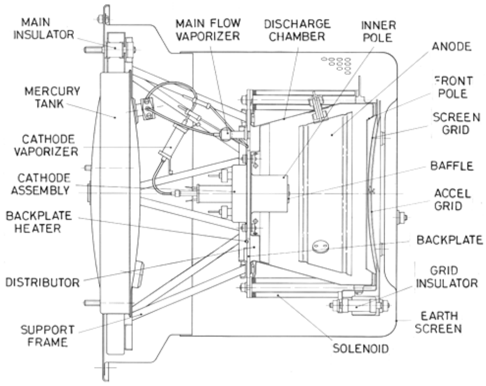


Figure 15. Sectional diagram of the T4 thruster (RAE diagram).

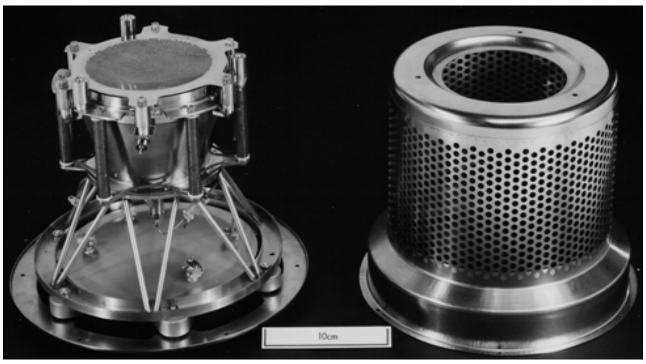


Figure 16. Photograph of T4 thruster (RAE photo).

Figure 17. Photograph of T4A thruster (RAE photo).

The T4A design then much more closely resembled flightworthy hardware, and initial performance assessments were very encouraging. These were conducted in three laboratories, the RAE, Culham and the Fulmer Research Institute (FRI), with essentially identical results. Life testing[50] at the FRI compared two grid sets, with 0.5% and 1% compensation, at 10 mN thrust. With each operated for 1000 h, it was found that the latter, although giving a beam divergence of below 8°, was subjected to some direct ion beam impingement on the accel grid. Thus the former value of compensation was selected for all future work, with a divergence normally in the range 10 to 12°, depending upon operating conditions. Grid lifetimes were predicted to be at least 30,000 h under these 10 mN conditions, with mercury propellant.

By this time the X5 opportunity had been lost owing to funding problems and all available efforts were being directed towards the L-Sat mission[33]. The success of the development work led to the plan to fly both UK and German gridded thrusters on the spacecraft in the NSSK role. The contractors selected to produce the flight hardware were MSDS and MBB, respectively, and the thrusters were designated T5[32] and RIT-10[51]. This communications satellite eventually flew as Olympus[52], but without ion thrusters, the EP project having been cancelled for financial and administrative reasons in 1978. A major feature of this 1970s programme was the successful design of a modular power conditioning and control equipment[53] (PCCE) with a mass of 10 kg, which operated the thruster entirely automatically under microprocessor control.

V. Application to the Artemis Spacecraft

Following a decision by the UK Department of Industry, the ion thruster programme was restarted in the mid-1980s. The initial work was based on the original T4A design, since no T5 thrusters were available. In fact, an old T4A had to be retrieved from the University of Bristol, where it had been operated using argon, and a new series of tests commenced using it at the Culham Laboratory, this time with xenon propellant. These were very successful[54] and a second UK team was formed, led as before by the RAE, to further develop this technology. No changes were necessary to the thruster, apart from a modification to the inner polepiece-baffle disc assembly to optimise performance. A 1 m diameter facility was modified at RAE to accommodate part of the testing programme

While very few design changes were necessary, the performance envelope was extended to more than 70 mN thrust and 2 kW input power[55]. The other work concentrated on the achievement of qualified status, and the development of the propellant supply and monitoring equipment[56] (PSME) and of the PCCE[57]. This was mainly undertaken by Marconi Space Systems (MSS) Ltd, which eventually became Matra Marconi Space (MMS) Ltd. Simultaneously, the RAE became the Defence Research Agency (DRA), then the Defence Evaluation and Research Agency (DERA) and, most recently, QinetiQ

A joint bid was made by RAE, MSS and Culham in late 1989 for an Intelsat programme of testing. This was successful and one of the most significant achievements of this contract was the confirmation that a decel grid adds considerably to the durability of a grid system[58]. As a consequence, all subsequent T5 thrusters have been fitted with a triple-grid system, as shown in Fig 18. Also as part of this contract, thruster-spacecraft interactions were studied in great detail, covering sputter deposition, electromagnetic interference (EMI), the emission of infra-red radiation from the ion beam, and spacecraft charging phenomena[58].

Figure 18. T5 Mk 4 thruster without earth screen (DERA photo).

In support of the T5 programme and the Artemis mission, cathode life-testing exceeded of 15,000 h at QinetiQ and limited thruster testing at MMS, designed to validate theoretical models of life-limiting factors, reached 2000 h. The cathodes were initially provided by Philips Components, once Mullard Ltd, but they discontinued work in this field in the early 1990s and QinetiQ now manufacture these devices

An even more extensive set of diagnostic tests became possible in the early 1990s, owing to the implementation of a Foreign Comparative Test Program[59] at Aerospace Corporation, funded by the USAF. This work commenced in 1993 following a year of planning and setting up equipment, and involved a wide variety of ion beam probes, sputter deposition detectors, mass spectrometers, a thrust balance, EMI measurements, laser-induced fluorescence and microwave measurements of plasma parameters, and thermal and optical measurements of the grids during thruster operation. At the end of the programme in 1995, no thruster was probably better characterised. Later work accomplished the full characterisation of the thruster over the throttling range 0.3 to 30 mN[41].

The flight on Artemis[19] was again jointly with the German team responsible for developing the RIT-10 RF ionisation thruster, which was very considerably improved in the intervening years and was also converted to utilise Xe as the propellant[60]. As in the case of L-Sat, the concept adopted by ESA was that this new method of NSSK should have redundancy of both thrusters (4 were flown) and of technology. This turned out to be wise decision, in view of several failures which occurred during the mission[20,61]. Fig 19 depicts both thrusters mounted on the spacecraft.

Figure 19. T5 (bottom) and RIT-10 (top) ion thrusters mounted on the Artemis spacecraft (ESA photo).

The spacecraft was launched on an Ariane 5 from Kourou on 12 July 2001. Unfortunately, instead of being placed in the required geostationary transfer orbit (GTO), a partial launcher failure resulted in an apogee which was much lower than desired; the orbit measured 592 km × 17,529 km. If Artemis had been a conventional communications satellite, it would have been declared a failure at that point. However, a very careful assessment of the situation by the various contractors involved and by ESOC resulted in a concept for a rescue mission. This involved a great deal of detailed planning and the production of a large amount of new software to be uploaded to the satellite, but success was ultimately achieved on 31 January 2003, when Artemis reached geostationary orbit and was declared fit for service[20,61]. It still has a predicted lifetime of 10 years.

This rescue was only possible because the ion thrusters were available on the spacecraft, although they were not designed or positioned for the task facing them. The initial orbit-raising task was undertaken by the chemical apogee motor, which used most of the propellant within the satellite to circularize its orbit at an altitude of 31,000 km; this required 1450 kg of bi-propellants. The ion thrusters were then operated over very long periods of time to follow the spiral orbit-raising strategy first discussed in the early 1960s, with thrust levels, depending on spacecraft orientation and which thrusters were being used, of never more than about 20 mN. Despite failures of ancillary equipments, not of the thrusters themselves, the operation was a success, and represents the first demonstration of this type of orbit-raising, many decades after it was first proposed.

It should finally be mentioned that, subsequent to this success, the SMART-1 spacecraft[21], employing a similar strategy, has successfully reached lunar orbit after being launched into a GTO by an Ariane 5. In accomplishing this very significant achievement, it should be recalled that the spacecraft initially transited through the van Allen radiation belts 4 times each day, yet suffered only software problems as a result of the very severe conditions it encountered. The propulsion system used so successfully on this occasion was a PPS-1350G Hall-effect thruster[62] developed and manufactured by Snecma.

VI. Conclusions

This paper has summarised the accomplishments of the UK X-Series of spacecraft, concentrating on the solar array and ion thruster technologies. The X4 satellite demonstrated the viability of this unique array concept, which is still fully competitive, and of all other platform systems. The T4A ion thruster could have been ready in time to undertake the required orbit-raising manoeuvre, had not the following X5 mission been cancelled. It was, at this stage, designed to produce 10 mN of thrust at an SI of about 3000 s, using mercury propellant, and it had already demonstrated adequate durability.

Following the cancellation of the X5 mission, the thruster, then designated T5, was suggested for NSSK on ESA's L-Sat experimental communications satellite, which later became Olympus. This proposal was accepted, with the condition that the German RIT-10 RF thruster be used in parallel with the T5. Development then made excellent progress, until funding problems led to another cancellation in 1977.

After more than two decades, the same mission was approved for the Artemis satellite, again involving the RIT-10 in a more advanced form. The propellant selected for this application was xenon. However, in 2001 a partial launch failure stranded this spacecraft in a useless elliptical orbit, from which it was rescued by the combined use of the on-board chemical and ion propulsion systems. The final phase of this rescue involved the use of the ion thrusters to conduct a slow orbit-raising manoeuvre from 31,000 km to geostationary altitude, thus demonstrating a capability first predicted in the early 1960s.

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