ETS-III Ion Engine Flight Operations in the Extended Mission Period

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An ion engine subsystem with two electron bombardment mercury ion thrusters of 5 cm anode diameter was flight tested on the Engineering Test Satellite-III during an extended mission from September 1983 to March 1985. Three tests were conducted: on/off cyclic tests for the both thrusters, continuous beam test for thruster 1, and continuous discharge test for thruster 2. The flight data were compared with that from the main mission (September 1982 to August 1983). With total operating hours and restarts of 66 h/47 cycles for the neutralizer, 64 h/53 cycles for the discharge, and 53 h/45 cycles for the beam on thruster 1 and 239 h/180 cycles for the neutralizer, 356 h/188 cycles for the discharge, and 220 h/172 cycles for the beam on thruster 2, the ion production and extraction were found normal and unchanged. The hollow cathodes were found to be activated by repeating their operations. The operation of the entire spacecraft was ended with the depletion of the hydrazine used in the reaction control subsystem.

Nomenclature

C_i = ion production cost
CH = high level of main cathode heater current
I_{sp} = specific impulse
J_B = beam current
J_{ck} = main cathode keeper current
J_D = discharge current
J_{nk} = neutralizer keeper current
P_{th} = thruster power
q = propellant flow rate
T_{BP} = power processing unit base plate temperature
T_{cv} = main cathode vaporizer temperature
Th = thrust
T_{nv} = neutralizer vaporizer temperature
T_{TH} = thruster shell temperature
V_B = beam voltage
V_D = discharge voltage
η_p = thruster power efficiency
η_p' = total power efficiency
η_u = propellant utilization efficiency

Introduction

THE ion engine subsystem (IES), which has two electron bombardment mercury thrusters, was flight tested onboard the Engineering Test Satellite-III (ETS-III) during the main mission period, one year from the launch in September 1982. In this period, 100 on/off cyclic tests and over 100 h of continuous beam testing were performed; the functions of the IES were verified as normal. The objectives of the flight test were to demonstrate the ion engine technology in space, to acquire the performance data, to verify the interface compatibilities with other subsystems, and to measure the thrust through the attitude control subsystem data. With these objectives, flight data from the main mission period were analyzed and evaluated in terms of thruster performance, thermal design, thrust generation, and electromagnetic compatibility. The results revealed the same thruster performance in space as during the acceptance testing on the ground, more stable ignition characteristics in space, thermal characteristics as expected, thrust generation equal to the calculated value, and no electromagnetic compatibility problems due to the IES operations.[1-5]

However, the main mission period was not long enough to obtain data on the degradation of the thruster performance even if it should happen. To supplement this, the mission period was extended and additional flight tests of the IES were performed. In the extended mission period, three tests were conducted to detect variations or degradations in thruster operations 1) on/off cyclic tests of both thrusters, 2) a continuous beam test of one thruster, and 3) a continuous discharge test of the other thruster.

This paper describes the thruster flight operations and the results from the extended mission period (September 1983 to March 1985) and discusses the variations in the thruster performance, comparing them with the results from the main mission period.

Since even the total of the main and extended mission operations was much shorter than the time for which ion thrusters should be operational, further tests of the IES were proposed for FY 1985 (April 1985 to March 1986). However, this was not possible because the ETS-III stopped functioning in March 1985 with depletion of the hydrazine used in its reaction control subsystem.

As a space experiment of electron bombardment mercury ion thrusters, flight testing of the ETS-III ion engine followed the Space Electric Rocket Test-II (SERT-II). Although

Presented as Paper 85-2010 at the AIAA/DGLR/JSASS 18th International Electric Propulsion Conference, Alexandria, VA, Sept. 30–Oct. 2, 1985; received Oct. 16, 1985; revision received July 4, 1986.

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these two ion thruster systems are similar in that they are of the electron bombardment type and operate with mercury, each has its own features, both in operation and in system hardware.

The SERT-II thrusters operated for more than several thousand hours before the mercury tanks were emptied.[6,7] On the other hand, the operation of the ETS-III thrusters was of much shorter duration and had to be terminated with much of the mercury left in the tanks because the spacecraft itself stopped functioning. In the SERT-II, long-term continuous operations were possible, because the spacecraft was in near-polar orbit, which allowed a continuous power supply as long as its orbit was in continuously sunlit conditions. The ETS-III thrusters had only about a 100 h of continuous sunlight even in the longest case and, for the most part, were operated in on/off cycles because the Earth's shadow usually eclipsed the spacecraft during part of each orbit. The ETS-III thruster system had an isolator, the operation of which was controlled with programmed sequences; this is a feature that became very common in thruster systems built after SERT-II.

What should be emphasized is that the ETS-III thruster testing has increased the available flight data on ion thrusters, the paucity of which had been considered as one of the barriers to the operational use of such thrusters.

Apparatus and Procedure

ETS-III

In September 1982 an N-I launch vehicle sent the ETS-III into a near-circular orbit of about 1000 km altitude and 45 deg inclination. Weighing 385 kg at launch, the spacecraft has a box-type body of $0.85 \times 0.85 \times 2.1$ m with solar array paddles extending to 0.88×6.0 m. The configuration of the ETS-III in space with the thruster locations and the spacecraft axes indicated is shown in Fig 1. To get maximum solar power to the paddles, the spacecraft changes its attitude around the yaw axis, yaw reference forward or backward, about once a month. In the yaw reference forward attitude, the spacecraft velocity vector has the same direction as the positive roll axis and coincides with the thrust vector of the ion thruster. In the yaw reference backward attitude, the spacecraft velocity vector has the same direction as the negative roll axis and is opposite to the thrust vector of the ion thruster.

The primary objective of the ETS-III mission was to establish the technology common to future high-power spacecraft. In addition to achieving its primary objective, four onboard technological experiments were conducted on the ETS-III. The IES was one of them, which shared the primary objective in the sense that it required high power.

Ion Engine System

The IES consists of two ion thrusters, two power processing units, and a power control unit. The two thrusters are installed on mission panel 2, which is one of the roll faces, with their thrust vectors parallel to the roll axis, as shown in Fig. 1. The power processing units and the power control unit are installed on the inside face of mission panel 2.

Ion Thruster

The two thrusters are of an identical mercury electron bombardment type of 5 cm anode diameter. Figure 2 is a longitudinal section drawing of the thruster including the propellant tank. In nominal operating conditions, the thruster is capable of producing a 30 mA beam of mercury ions at an energy of 1000 eV and giving a thrust of 2 mN at a specific impulse of 2200 s by consuming 68 W of electric power. Both the main cathode and the neutralizer have the same configuration of a hollow cathode with an enclosed-type keeper and an impregnated insert. The propellant tank contains 600 g mercury when full.

Power Processing Unit and Power Control Unit

The power processing unit (PPU) contains 10 power supplies to drive the thruster. The voltages of the beam/accel power supplies and the current of the discharge power supply can be changed in magnitude. The heater power supplies for the main cathode and the isolator have two output levels, nominal and high. Keeper power supplies for both the main cathode and the neutralizer have the output characteristics of decreasing voltage with increasing current. The discharge voltage and the neutralizer keeper voltage are controlled by closed loops with the main vaporizer and neutralizer vaporizer currents, respectively, to make them equal to the reference values. Table 1 shows the major output specifications of the 10 power supplies of the PPU.

The power control unit (PCU) controls the PPU to operate the thruster in programmed sequences. The main sequence goes in the order of start of the neutralizer, the main cathode/discharge, and the beam acceleration. When the neutralizer or the main cathode is extinguished, a sequence for reignition is made automatically.

Operation Procedure and Test Data

Although the extended mission period of the ETS-III lasted about 18 months, the total operating hours assigned to the IES were not as long. This is because the operation of the ETS-III itself was reduced and because continuously sunlit periods, the most appropriate for the IES operation, were decreased in number and shortened in length due to

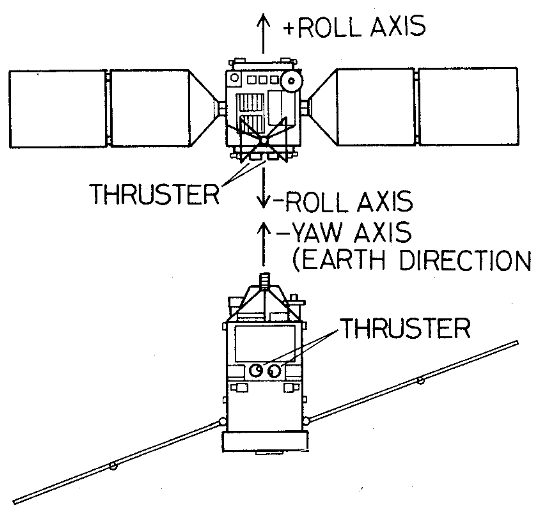


Fig. 1 Diagram of the ETS-III in orbit showing thruster location.

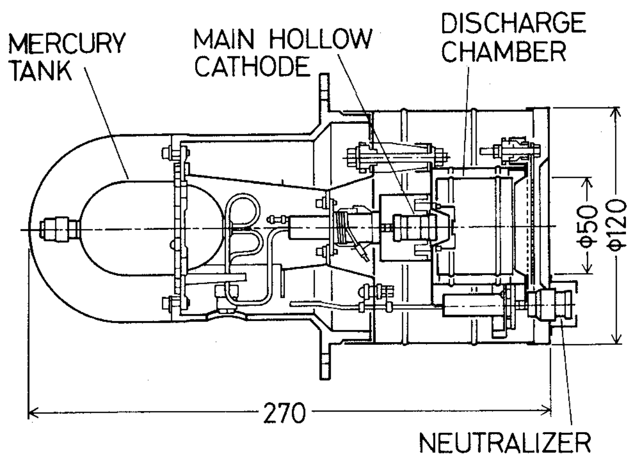


Fig. 2 Longitudinal section drawing of the thruster (dimensions in millimeters).

Table 1 Major specifications of the power supplies of the PPU

Power supply	Nominal operating level Voltage, V	Current, A	Type	Remark
PS1: beam (screen)	1000	0.03	dc	800–1400 V by command
PS2:accelerator	1000	0.001	dc	800–1400 V by command[a]
PS3:discharge	40[b]	0.35	dc	0.2–0.5 A by command
PS4:main cathode heater	5	5	ac	6V and 6A for high level
PS5:main cathode keeper	15	0.3	dc	Negative impedance of −41 V/A
PS6:main cathode vaporizer	0.88–3.5	0.5–2.0	ac	Closed loop control with PS3 voltage[b]
PS7:neutralizer cathode heater	5	5	ac	
PS8:neutralizer keeper	24	0.25	dc	Negative impedance of −75 V/A
PS9:neutralizer vaporizer	0.88–2.1	0.5–1.2	ac	Closed loop control with PS8 voltage
PS10:isolator heater	3	1	ac	5 V and 1.6 A for high level

[a]Voltage linked to PS1. [b]PS3 voltage reference can be varied by command.

correction maneuvers made to attain its nominal orbit, lower than the initial one, during the extended mission period.

Three tests were made during the extended mission period: cyclic, continuous beam, and continuous discharge. Thruster 2 was exclusively operated for the cyclic test during the main mission period, but both the thrusters were used for the cyclic test in the extended mission period. The continuous beam and discharge tests were performed on thrusters 1 and 2, respectively.

The general operating procedures are given in Refs. 1 and 2. In the continuous discharge test, the thruster was operated continuously in the discharge mode for a series of orbits. Although the beam and neutralizer were not working, a long discharge operating period was possible because the batteries on the spacecraft could supply the electric power for the discharge operation while the spacecraft was in the Earth's eclipse.

In the extended mission period from September 1983 to March 1985, thrusters 1 and 2 were operated for 7 and 21 days, respectively. Table 2 summarizes the dates and the tests performed. The start numbers in Table 2 indicate the discharge operation number counted for the thrusters from the beginning of the main mission period.

Steady-State Characteristics

During the extended mission period, the thrusters were operated normally and showed no eminent tendency to performance degradation. The total beam exhausting hours for both thrusters, from the beginning of the main mission period, was 272.7 h. Table 3 shows the thruster operating hours and the restart numbers for each thruster, mission period, test item, and operational mode.

The first operation of thruster 1 during the extended mission period on May 14, 1984, was found normal. This operation was after one year of dormancy, because thruster 2 was usually used only for the main mission tests, proving the thruster to have been well preserved in space.

Thruster Characteristics

During the extended mission period, the thruster characteristics (changes in operation parameters due to the changes in the discharge voltage or current) were obtained as a result of a whole series of cyclic tests. Different values of the discharge voltage or current were set on each day of operation for the cyclic tests. In the main mission period, such characteristics were obtained during parameter changing tests in which parameter settings were changed by ground commands while the satellite was visible.

Figure 3 shows the effects of the discharge current and voltage on other thruster parameters, during both the extended and main mission periods. Variations in the beam current and the main vaporizer temperature from the values in the main mission period are very small, which means that

Table 2 IES operation dates and test items in the extended mission

Thruster	Test	Date	Start no.
1	Cyclic	5/14–5/19/84	22–52
	Continuous beam	7/4/84	53
2	Cyclic	9/1/83	120–121
		10/31–11/4/83	122–141
		3/12–3/13/84	142–150
		3/19/84	151–160
		3/22–3/24/84	161–169
		11/27/84–	170–171
		11/30/84	173–186
	Continuous discharge	11/27/84	172
		1/31–2/4/85	187

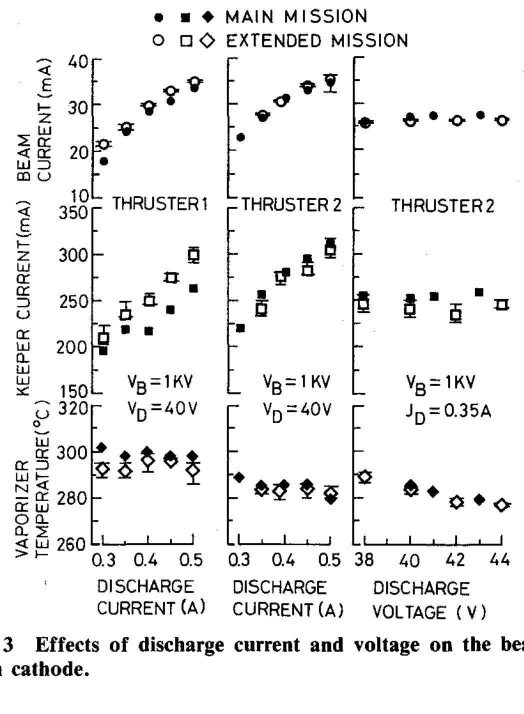


Fig. 3 Effects of discharge current and voltage on the beam and main cathode.

variations in the thruster performance (i.e., thrust, ion production cost, and propellant utilization efficiency) are also very small because these are functions of only the values shown in Fig. 3.

As for the main cathode keeper current of thruster 1, the agreement of the results in both periods is poor. However, in comparison with data from the acceptance test on the ground, the keeper currents in the main mission shown in Fig. 3 are lower than those in the acceptance test, but the keeper currents in the extended mission are in good agreement. These facts indicate that some degree of degradation in the main hollow cathode occurred after the acceptance test, but that a recovery was achieved with repeated operations in space. In fact, the data on the main mission in Fig. 3 were obtained during one of the parameter changing tests, which were performed early in the main mission; after that the hollow cathode operations were repeated 10 times for 28.7 h in total until the end of the main mission.

Thruster operating characteristics were calculated from the same data on the extended mission shown in Fig. 3. The results are shown in Fig. 4 for the propellant flow rate, propellant utilization efficiency, thrust, specific impulse, beam ion production cost, thruster power, thruster power efficiency, and total power efficiency. In the calculations, the propellant flow rate was estimated from the main vaporizer temperature data using the calibration curve obtained in the ground test. The beam ion production cost shown is just the discharge power divided by the beam current. The thruster power efficiency was defined by the ratio of the beam power to the thruster input power and the total power efficiency by that of the beam power to the IES input power.

In Fig. 4, small differences in performance are found between thrusters 1 and 2. For example, a higher flow rate of the propellant is required in thruster 2 than in thruster 1 to maintain the same discharge voltage. Such differences were also found during the ground acceptance test, which suggests that one of the causes of such differences is variations in the thruster hardware fabrication. Small variations in the thruster hardware could occur within the tolerances, although the specifications were the same for the both thrusters.

Performance Variations

Figures 5 and 6 show the major thruster parameter variations in steady state with respect to the start number; each data set was taken just before each operation was stopped. For both the thrusters, the values of the discharge voltage and current are in good agreement with the reference values obtained by ground commands (shown by the dashed lines in Fig. 5) for normal control functions. The beam currents of both thrusters have almost the same values for the same values of the discharge current, discharge voltage, and beam voltage, which means that the thruster functions of ion production and extraction are unchanged.

The main cathode and the neutralizer, however, have rather scattered data in spite of the stable characteristics of the discharge and beam. Because hollow cathodes tend to change their characteristics easily due to degradation, keeper voltages or currents are good indexes for finding signs of such degradation if it occurs. Thus, the data for thruster 2, which has much larger start numbers in total, were examined.

Main Hollow Cathode

For the main cathode, the keeper current for the same value of discharge current was examined for variations with the start number during the both mission periods, because it is sensitive to the discharge current, but not to the discharge

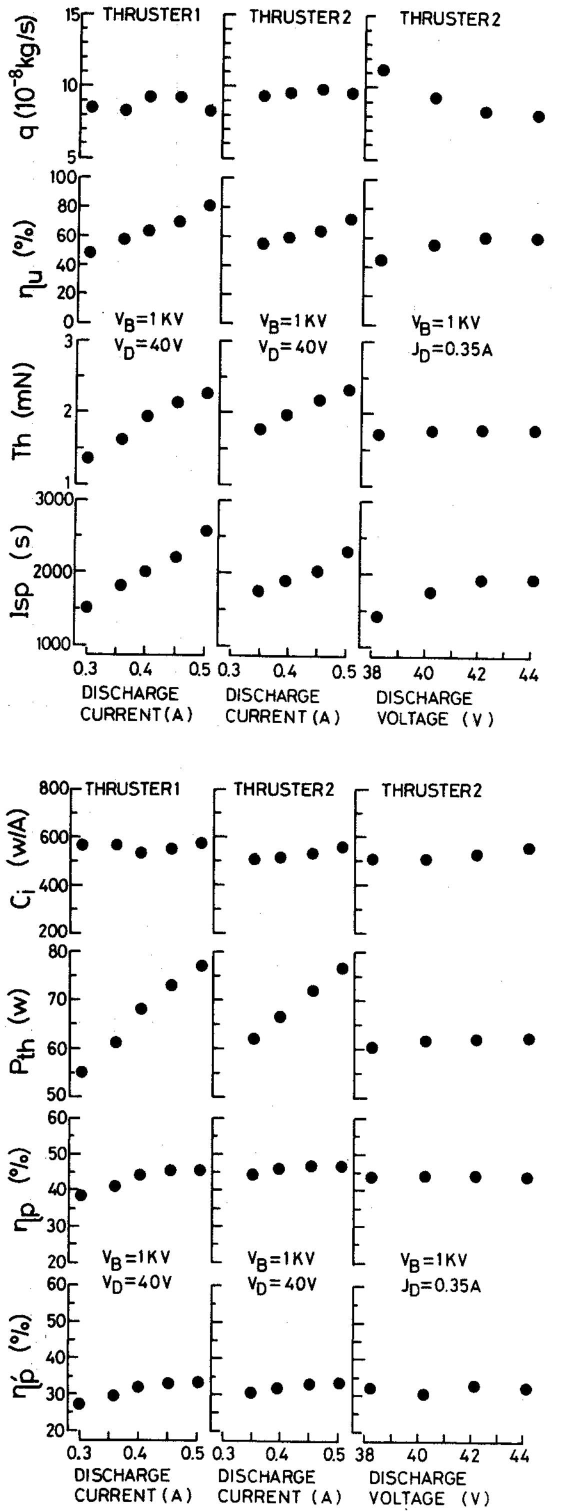


Fig. 4 Effects of discharge current and voltage on the thruster operating characteristics.

Table 3 Summary of the thruster operating hours and the restart numbers

Mission period	Test item	Thruster 1, h/times			Thruster 2, h/times		
		Neutralizer	Discharge	Beam	Neutralizer	Discharge	Beam
Main	Cyclic	—	—	—	82.9/100	80.7/100	74.8/100
9/82–8/83	Cont. beam	24.4/1	24.0/1	23.9/1	104.4/2	104.4/2	104.3/2
	Others	9.1/14	12.1/20	3.2/12	7.2/11	12.4/17	3.3/9
Extended	Cyclic	23.6/31	19.7/31	17.5/31	44.4/67	44.0/67	37.4/61
9/83–3/85	Cont. beam	8.4/1	8.4/1	8.3/1	—	—	—
	Cont. discharge	—	—	—	—	114.8/2	—
Total for both periods		65.5/47	64.2/53	52.9/45	238.9/180	356.3/188	219.8/172

voltage (which corresponds to the mercury flow rate). The results revealed the following. For the 0.35 A discharge current, the keeper current decreased during the extended mission period compared with the early main mission period; but for the 0.39 A discharge current, which was the value in the most cases during the main mission period, the keeper current was almost constant within a scattered range and increased late in the extended mission period. As a whole, these results show no tendency to degradation in the main cathode.

The scattering in the main cathode keeper current is a result of output variations in the keeper power supply due to the temperature variations and the effects of activating the hollow cathode due to repeating the keeper discharge, as well as differences in the discharge voltage. With the same keeper current data and the known temperature dependence of the keeper power supply output, the keeper voltage was examined for the case of the 0.35A/40 V discharge. The maximum output variation due to temperature variations corresponded to about a 10 mA variation in the keeper current; and the keeper voltage was found to drop from 17.3–18.5 V in the main mission period to 17 V or so in the extended mission period.

Neutralizer Hollow Cathode

The neutralizer keeper voltage was intended to maintain the reference value by a closed-loop control with the vaporizer heater current, but, in fact, the operating point of the vaporizer heater current was kept at the lower limit of the loop. This brought about the variations in the operating point of the neutralizer keeper discharge through variations in the mercury flow rate (or the vaporizer temperature) caused by the changes in the thermal input in orbit. Thus, if variations in the neutralizer hollow cathode characteristics do occur, they might be imbedded in such variations in the operating point.

To avoid this, the neutralizer keeper current was plotted against the vaporizer temperature (or mercury flow rate) based on data from both mission periods. The results are shown in Fig. 7. The neutralizer keeper current has scattered values even for the same value of the mercury flow rate. One of the reasons for this is the temperature variation in the keeper power supply output. The keeper current variation range due to temperature variations is about 25 mA for base plate temperature variations of 10 deg.

In spite of these variations, the neutralizer keeper current did not have a tendency to decrease, as a whole, during the extended mission compared with the values during the main mission. This means that no eminent rises in the keeper voltage occurred, because the neutralizer keeper power supply has an output characteristic of decreasing voltage with increasing current.

Since the neutralizer had no isolator between the cathode and the vaporizer, the cathode potential was kept at the ground (spacecraft) potential and no coupling voltages were measured. The beam neutralization was confirmed by the onboard thrust measurements based on data from the changes in the reaction wheel rotating speed. The measured thrusts were found to agree well with the results obtained during the ground test.

Table 4 Comparison of performance for thruster 1 continuous beam tests in the main and extended mission operations

Current	Revolution 1st	2nd	3rd	4th
Beam, mA				
Expended mission	29.5	29.5	29.5	29.5
Main mission	29.9	29.7	29.7	29.7
Discharge, mA				
Expended mission	389	390	390	390
Main mission	402	402	400	400
Main cathode keeper, mA				
Expended mission	257	267	271	274
Main mission	250	256	253	255
Neutralizer keeper, mA				
Expended mission	271	309	315	323
Main mission	257	275	277	281

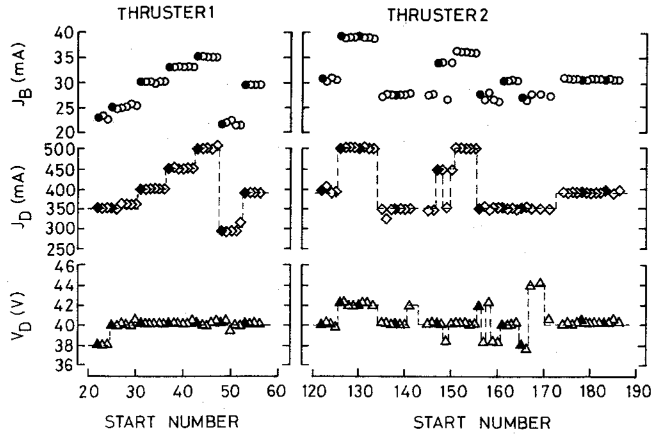


Fig. 5 Thruster parameter variations due to restarts (beam and discharge) (solid symbols denote the first data of a series of operations).

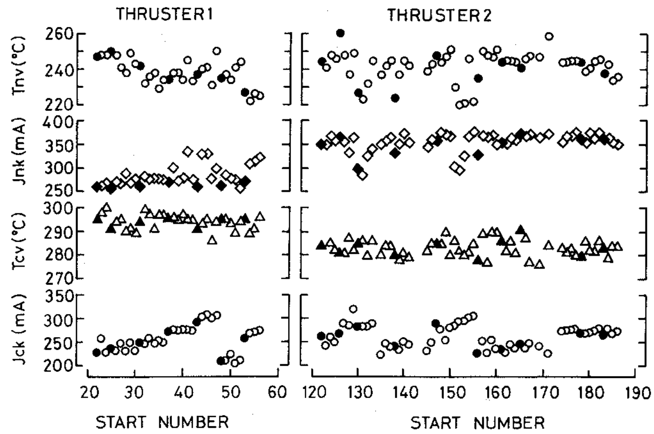


Fig. 6 Thruster parameter variations due to restarts (main cathode and neutralizer) (solid symbols denote the first data of a series of operations).

Startup Characteristics

Hollow Cathode Ignition Time

The ignition time of a hollow cathode is a sensitive indication of the variations in its characteristics and is one of the important items to be evaluated during flight testing. Ignition time variations with restarts of the neutralizer and main hollow cathodes are shown in Figs. 8 and 9 for thrusters 1 and 2, respectively. The discharge started, for all cases, at the same time that the main cathode ignited.

The neutralizers of the both thrusters have very stable ignition times and do not tend to delay in igniting. The neutralizer ignition time of thruster 2 becomes rather shorter with restarts, although only slightly. As a whole, the neutralizers had good startup characteristics during both the main and extended mission periods.

The ignition of the main cathodes (or the discharges) are characterized as follows. The main cathode of thruster 1 had a very unstable ignition time during the main mission period

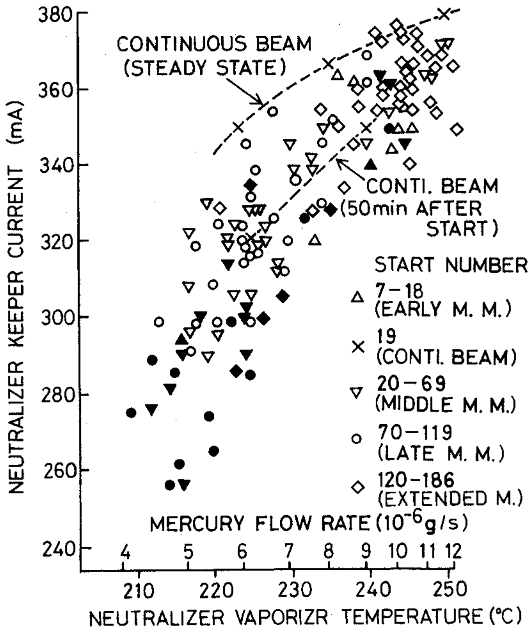


Fig. 7 Neutralizer flow rate vs keeper current (solid symbols denote the first data of a series of operations, MM the main mission).

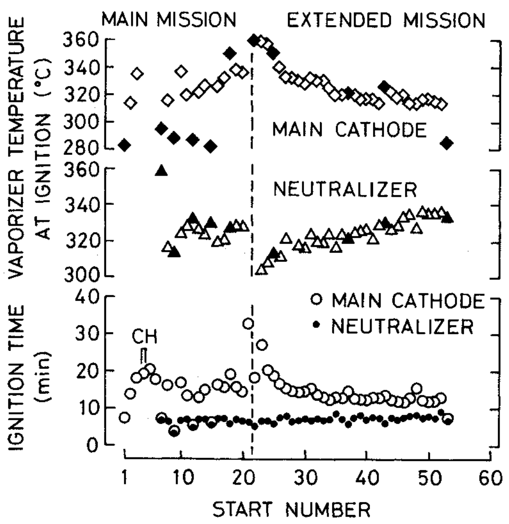


Fig. 8 Hollow cathode ignition characteristics for thruster 1 (solid symbols denote the first data of a series of the operations).

(start numbers 1–21) and took longer than 20 min at the beginning of the extended mission period. With the restarts repeated, however, the ignition time was shortened to 12–14 min and remained stable.

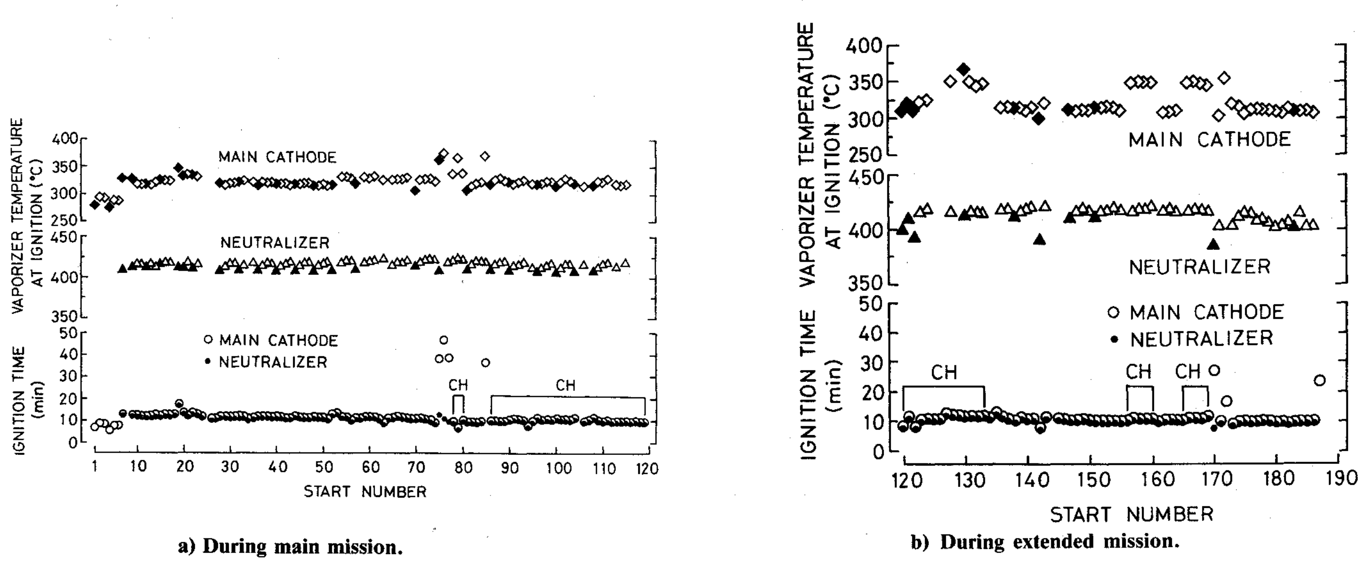
The inherent ignition time of the main cathode could not be obtained for thruster 2, because the main cathode ignited simultaneously with the neutralizer ignition in most cases. This means that the ignition of the main cathode was not determined from its own characteristics, but by the control sequence that allows the main cathode to ignite only after the neutralizer ignites.

However, the inherent ignition time of the main cathode is estimated to be below 10 min or so from the hollow cathode test results from the main mission period (start numbers 1–6) and from the fact that simultaneous ignition of the neutralizer and the main cathode could take place even when the neutralizer ignited sooner than the average, e.g., below 10 min. In the sense that the main cathode ignited at the same time as the neutralizer ignition, it had a stable ignition time.

In some cases, however, the main cathode ignition was delayed suddenly from the preceding stable conditions, as in the cases of start numbers 75–77, 85, 170, and 185. This resulted from the coupled effects of the hollow cathode design and possible variations in the ignition conditions. The main cathode and its electrical interface were designed with a very small margin in the cathode tip heating power when the nominal level of the heater current was selected. Thus, small variations in the ignition conditions can cause very long delays in the ignition time.

These situations were expected to take place in space. The ground test had revealed the main cathode heater power to be critical, but the nominal level had not been raised in the IES design because the high level of the heater current was installed as an option. In fact, no delays took place in the main cathode ignition with the high level of the heater current selected (as denoted by CH in Figs. 8 and 9).

The vaporizer temperatures at ignition are also shown in Figs. 8 and 9. Since the telemetry for the vaporizer temperatures was calibrated only up to 400°C, the values over 400°C are not exact and are thus only an index to measure the vaporizer temperatures qualitatively. In spite of the very high telemetry values of the neutralizer temperatures of thruster 2, boiling of the mercury in the vaporizer was not suggested in that the normal ignition of the neutralizer were achieved for all of the restarts. The mercury tank was pressurized up to 0.25 MPa when the mercury and pressurizing gas were loaded on the ground and was effectively kept at that pressure due to the very low consumption of the mercury.



a) During main mission.

b) During extended mission.

Fig. 9 Hollow cathode ignition characteristics for thruster 2.

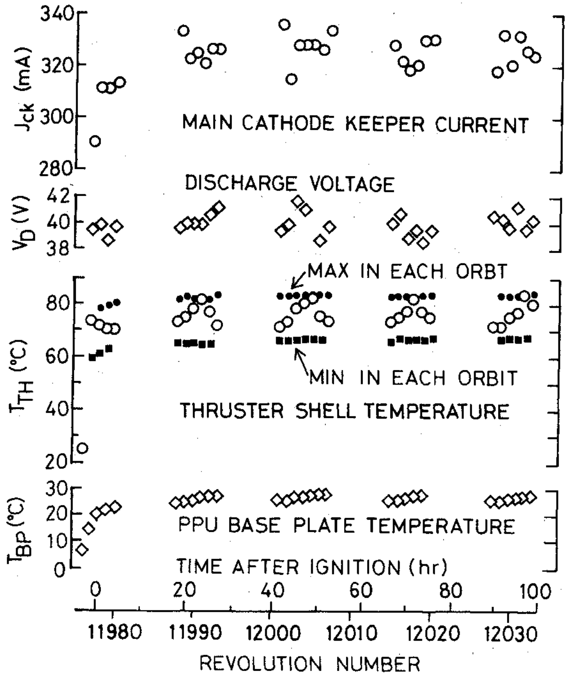


Fig. 10 Overall trends of the main parameters during the continuous discharge test (part 2).

Discharge Extinction

Extinction of the discharge/main cathode occurred quite often for the both thrusters, typically between the starts of discharge and beam acceleration. For thruster 1, it occurred 13 times, all during the extended mission period, and for thruster 2, 12 times during the main mission period and 9 times during the extended mission period.

Neutralizer extinction occurred twice, once for each thruster, both during the extended mission. The neutralizer extinction accompanied the main discharge extinction corresponding to case 3 (described below) and automatic restarts were achieved by the neutralizer reignition sequence of the PCU.

The main cathode discharge extinction can be classified into three cases:

1) Discharge extinction followed by the automatic restart was repeated in the discharge stabilizing phase before the beam start. According to the programmed sequence, the PCU shut off thruster operation as soon as the extinction was repeated up to 10 times. Such cases occurred twice for thruster 2 during the extended mission period, but these were due to too high discharge voltage references of 42 or 44 V. These values were set intentionally to examine the thruster capability.

2) Discharge extinction followed by restart occurred when the beam/accel voltage was turned on and it was repeated until the end of the scheduled operation time, resulting in no beam exhaustion. Such cases happened three times for thruster 2 during the extended mission period.

3) After discharge extinction followed by restart was repeated a few times, the stable discharge was maintained resulting in normal beam exhaustion. All of the discharge extinction, except for cases 1 and 2 corresponded to this case.

In all cases, high-amplitude noises in the discharge voltage were not observed and gradual or sudden variations in the discharge voltage were typical, from the flight data whose sampling rate was 1/1 s.

The direct cause for the discharge extinction is that the cathode mercury flow cannot follow to compensate for a rise in the discharge voltage. There are three related factors for this:

a) Generally, the discharge voltage rises as the beam acceleration begins with a constant mercury flow or a larger mercury flow is required in the beam mode to keep the same discharge voltage.

b) In the discharge stabilizing phase (normally for 3 min from the discharge start to activation of beam/accel voltage), the cathode heater power is not turned off, but rather kept at a low level (about 4 W). This improves the discharge efficiency; without this cathode heating, a larger mercury flow would be needed to keep the same discharge voltage. In fact, this heating is turned off when the beam/accel voltage is turned on, making factor a more severe.

c) Some mercury from the vaporizer is condensed in the isolator before it is sufficiently warmed. As the isolator temperature increases, the condensed mecury is vaporized again and flows into the discharge chamber. As this process continues, the quantity of the condensed mercury decreases and the revaporized mercury flow also decreases. In this process, the discharge voltage increases due to the decrease in the mercury flow rate.

Although discharge extinction is thought to come from the coupled effect of the factors noted above, the occurrence of a particular discharge extinction is not directly related to certain conditions, but seems to be a phenomenon with probability. To confirm this, the ignition time and the vaporizer temperature are chosen as possible controlling parameters and the probabilities of discharge extinction in certain ranges of these parameters were compared with each other, where the probability was defined as the ratio of the discharge extinction number to the total ignition number.

Within ranges having relatively large numbers of ignitions, particular ones were found to have high discharge extinction probabilities for the thrusters 1 and 2. These ranges are 11–13 min ignition time and 315–325°C vaporizer temperature at ignition for thruster 1 and 10–12 min ignition time and 325–330°C vaporizer temperature at ignition for thruster 2. The reasons for these are estimated to be as outlined below.

Mercury condensation and revaporization processes proceed in the three states: 1) condensation, 2) revaporization, and 3) no condensation (normal state). On the way from the second to the third state, the mercury flow to the discharge chamber decreases and the discharge voltage increases. If this overlaps the rise in the discharge voltage due to the beam acceleration [factors a and b describe above], it may result in discharge extinction. The fact that occurrences of discharge extinction are concentrated in certain ignition time ranges supports this.

For thruster 2, occurrences of discharge extinction were also concentrated in ranges of high vaporizer temperature at ignition. This can be related to the fact that the main cathode ignition is delayed by the PCU sequence until the neutralizer ignites. The vaporizer continues to be heated even after the main cathode would ignite if it were operated separately. Thus, on ignition, the vaporizer temperature rises much higher than the temperature for the nominal mercury flow rate, causing the discharge voltage to decrease to the very low level and inversely bringing very low vaporizer temperatures (about 200°C) by keeping the vaporizer current at the lowest level. This process, if emphasized, may also result in discharge extinction by overlapping the discharge voltage rise described above.

The isolator heater power has two levels, nominal and high, to be selected as a command. For thruster 1, the nominal level of the isolator heater power was ordered for all restarts. For thruster 2, the high level was ordered for 18 restarts and the nominal level 169 times during the entire mission. With the high level of the isolator heater power, four discharge extinctions occurred, but three of these were cases where intentionally high values of the discharge voltage reference were selected. The high level of the isolator power appears to be effective in preventing discharge extinction, but it is not definitive because of the few sample cases. The high level is effective only before the beam start and the isolator heater power is lowered to the nominal level after that because of power limitations.

Continuous Operation Tests

Thruster 1 was tested continuously in the beam exhausting and discharge operating modes. The discharge test was performed for a total of 114.8 h in two parts: the first for 16.1 h and the second for 98.7 h. The thruster operation was found normal for this test. In the discharge test, the discharge voltage and the current were stable, but the keeper current was increased toward its steady state for about the first 3 h. These voltage and currents were very noisy, as they usually are in the discharge operation mode. The overall trends are shown in Fig. 10 for the same test. The data could not be obtained while the satellite was in orbits entirely invisible from Japan; the onboard tape recorder can store the data for only one orbiting revolution of the satellite. The thermal conditions reached steady state by the second visible group of orbits.

The continuous beam test of thruster 1 continued for 8.3 h during the extended mission, while it continued for 23.9 h during the main mission period. Table 4 compares the data in the both tests for the first four revolutions. The results are almost identical. The beam exhaustions and discharges were stable, and the main cathode keeper currents increased for the first 2 h or so and were kept constant with small cyclic variations due to the thermal input variations.

However, the keeper currents of both the main cathode and the neutralizer increased from the main mission period. This indicates that the hollow cathodes were activated by repeating their operations. The increasing keeper current means decreasing keeper voltage because both keeper power supplies have negative impedance outputs, as shown in Table 1.

Conclusions

The testing of the ETS-III ion engine system proceeded for 30 months during the main and extended mission periods until March 1985. With much of the mercury left in the propellant tanks and with the thruster functions being normal, the testing was terminated when ETS-III operation ended due to the exhaustion of the hydrazine fueling its reaction control system. During these periods, thrusters 1 and 2 accumulated 53 and 220 h of beam operation and 45 and 175 cycle restarts, respectively. In addition, thruster 2 was tested in the discharge operation mode for a total of over 100 h. These operations proved that the functions of ion production and extraction were normal and unchanged. Both the main and neutralizer hollow cathodes had scattered values of their keeper currents in each operation, but were found to have no eminent tendency to degradation. Effects of the hollow cathode activation were also found. Extinction of the main cathode keeper and the discharge took place more than 30 times in total, but automatic restarts were achieved by the PCU sequence in most cases. This revealed, however, that some improvements are needed in the thruster thermal design and the control of the hollow cathode ignitions.

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