Colloid Microthruster System Life Test

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Results from over 1750 hr of microthruster system operation are presented. The system consists of a three-needle thruster, bare tungsten wire filament neutralizer, bladderless blowdown propellant feed system, and 5-w power and signal conditioning unit. Neutralizer operation was terminated after 950 hr because of electron leakage to high voltage through some insulator holes. The test was stopped when expulsion pressure was lost from the trapped volume of pressurizing gas in the feed system. Normal operation at 8-μlb thrust and 950-sec specific impulse demonstrated extremely smooth steady-state performance and reproducible turn-on and turn-off characteristics.

Introduction

Colloid propulsion subsystems are well suited for performing satellite stationkeeping and attitude control functions in Earth orbital operations.[1] To demonstrate their flightworthy nature, a colloid microthruster system was developed for space testing in yaw control of a multipurpose satellite in synchronous orbit.[2] When plans for the satellite were cancelled, system development activities were redirected toward ground-test demonstration of colloid microthruster compatibility with realistic satellite interfaces and launch environments.

The yaw control system consisted of two thrusters, a spring-loaded piston feed system, and a constant-current power and signal-conditioning unit.[3] Each thruster was nominally rated at 8 μlb. During the development program, the microthruster system was mounted on a thrust stand that enabled direct thrust determinations.[4] Direct mass-flow measurements were made with this system by weighing it before and after 166 hr operation. The measured mass flow was 15% lower than the average computed time-of-flight mass flow, but only 0.3% higher than the calibrated mass-flow data previously reported.[2,3]

Acceleration and vibration tests to Titan III-C booster vehicle requirements were conducted. After vibration, the feed line to the thruster was found severed at its coupling. The coupling was not used in subsequent design.

Electromagnetic compatibility characteristics of the system were compared with MIL-STD-461A specifications. It was concluded that lightweight, readily available components could be inserted between a spacecraft power source and the input inverter to the power conditioning unit to effectively suppress electromagnetic conduction. The microthruster complied with specification limits for electromagnetic emissions from its exhaust beam.

The yaw control configuration thermal analysis was performed for a three-axis stabilized spacecraft in synchronous equatorial orbit. Thruster heater peak and average power levels required to maintain the needles at given operating conditions were determined from the analysis. This information impacted on subsequent system design.

Based on engineering data obtained from the earlier microthruster system, a single-thruster, flight-prototype system was developed. This paper presents the results of 1792 hr of life testing.

System Description

The prototype system (Fig. 1 and Table 1) consists of a three-needle thruster, zero-gravity propellant storage and feed system, and power and signal conditioning (P/SC) unit. The propellant is a mixture of 19.3% (by weight) sodium-iodide in glycerol.

Thruster

The thruster uses three conventional positive needle sources.[5] A 0.0015-in.-diam bare tungsten wire filament provides exhaust beam neutralization. A second filament serves as a redundant neutralizer.

Thruster assembly details may be seen in Fig. 2. The three capillary needles are equally spaced on a $\frac{1}{4}$-in.-diam circle. The manifold block includes a small brazed plenum chamber which, in turn, is brazed to the feed tube. High-voltage isolation is achieved through use of a machined

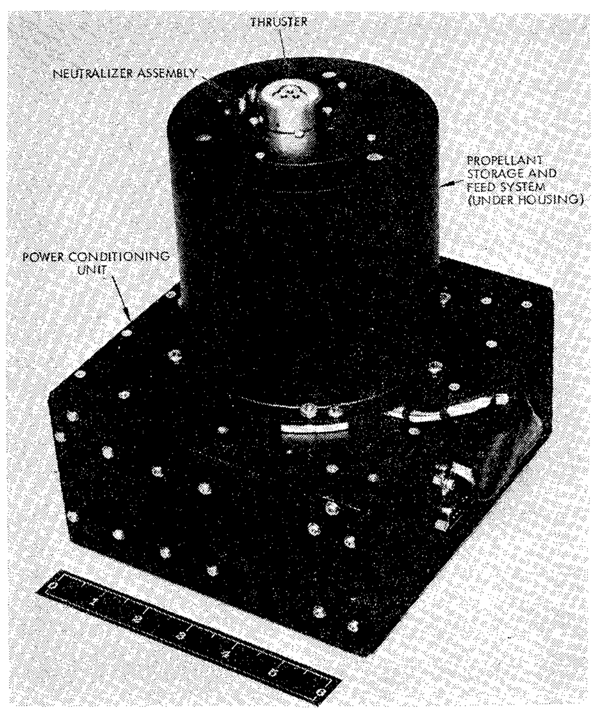

Fig. 1 Microthruster system.

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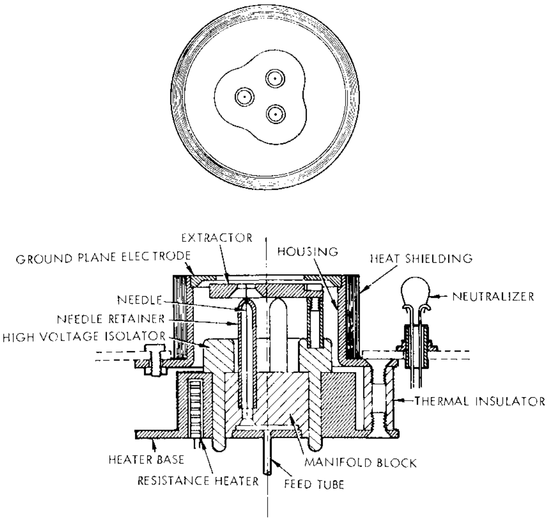


Fig. 2 Thruster schematic.

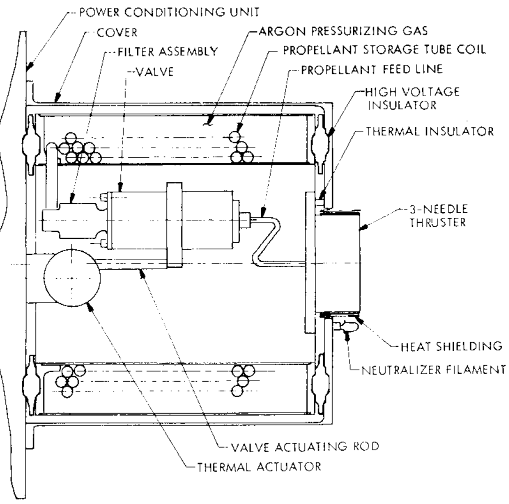


Fig. 3 Microthruster schematic.

epoxy casting. This isolator also provides manifold block and extractor support. The thruster housing mounts the ground plane electrode. Its apertures are enlarged to form a single irregularly shaped opening. The thruster housing is thermally insulated on its outer diameter with layers of metallized Mylar† to reduce heat transfer from areas external to a spacecraft skin. Thermally, the thruster housing is clamped to the prototype structure. The thruster heater base is mounted from the housing by low-thermal-conductivity standoffs to reduce the power requirement. Four carbon resistors encapsulated in the heater base provide active thermal control. These resistors are connected in parallel for added reliability. Two thermistors for monitoring and controlling temperature are also embedded in the base.

The neutralizer filament is shielded from viewing the thruster needles in order to prevent filament carbiding from exhaust beam products during space operation. This is accomplished by placing each filament alongside the thruster housing approximately one inch from the thruster centerline.

Feed System

The feed system includes a propellant storage and expulsion tank, filter, thruster valve, and thermal actuators. Its component parts and their arrangement within the microthruster are illustrated schematically in Fig. 3. The bladderless blowdown feed system concept employed relies on the extremely low diffusion rate of argon pressurizing gas in propellant to achieve high expulsion efficiency. A trapped volume of pressurizing gas provides continuous expulsion pressure. Propellant is stored in a tube coil, and is retained by capillary forces. A gas-to-liquid interface exists between the pressurizing gas and propellant. During launch, the gas-to-liquid interface is oriented to take advantage of lift-off acceleration to retain propellant. After launch, this interface only has to withstand vehicle accelerations that are encountered during spacecraft maneuvers.

The propellant coil consists of an inner 0.09-in.-i.d. Inconel‡ tube coil and two 0.22-in.-i.d. Inconel outer tube coils. Propellant is loaded through the small diameter coil first, enters the large diameter coils, and then is fed to a line passing through the pressurizing gas chamber wall to a valve. The valve mounts on a bracket that is adhesively bonded on its outer diameter to the pressurizing gas chamber inner wall. Upon assembly, the pressurizing gas annulus encloses the propellant storage coils. The propellant-to-gas interface initially resides in the small diameter tube coil. The gas volume is twice the initial propellant volume, so that feed pressure is reduced about 33% (from 36 to 24 torr) near complete expulsion.

The thermally actuated ball seat valve developed for propellant protection from atmospheric conditions, and for on-off flow control during microthruster operation is described in Ref. 6. System response to valve actuation is limited by thermal actuator characteristics (excluding initial turn-on which is limited by the time it takes to fill the thruster with propellant). Elastomers are eliminated from propellant contact in the feed system to preclude atmospheric permeation into propellant during storage.

The feed system is loaded as shown in Fig. 4. All parts are located inside a heated enclosure that is used to outgas the feed system at 50°C and transfer propellant at 35°C. Well-filtered, out-gassed propellant is contained in a flask that can be raised in order to insert a propellant transfer tube. This tube slides freely inside a vacuum feed-through.

Table 1 Prototype characteristics

Dimensional envelope	$7.76 \times 8.00 \times 9.25$ in.
Input power	5.0 w
Valve command pulse	750 joules
Thrust (two levels)	8 or 4 μlb
Specific impulse (at 8 μlb)	950 sec
Weights, lb	
Thruster	0.1
Propellant storage and expulsion tank	1.7
Valve assembly	0.3
Propellant	0.5
Power and signal conditioning unit	2.9
Support structure	1.5
Total system	7.0 lb

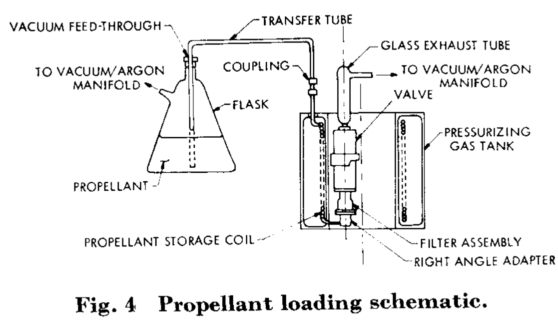


Fig. 4 Propellant loading schematic.

† Trade name, DuPont.

‡ Trade name, International Nickel Co.

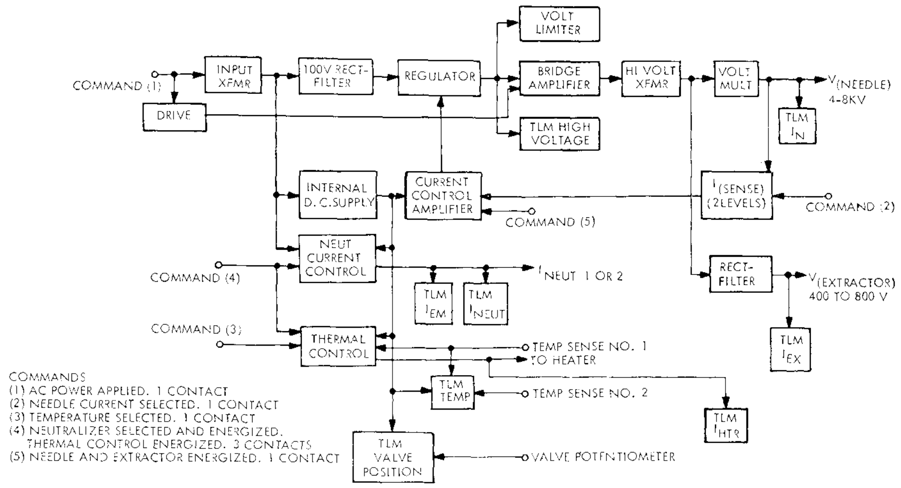


Fig. 5 Power and signal conditioning flock diagram.

Argon, at 1 psig, is used to push propellant up through the transfer tube, into the propellant storage coil, right angle adapter, filter assembly, valve, and out to the glass exhaust tube maintained under vacuum. Exhaust propellant is observed for clarity, color, and gas bubbles. After loading, the propellant valve is closed and the open propellant coil end is pushed into the pressurizing gas chamber. The chamber access hole is plugged and the chamber backfilled with argon at approximately 1 psia to provide expulsion pressure.

The epoxy insulators at the end of the pressurizing gas tank provide electrical isolation of the feed system from the microthruster cover and the P/SC unit. The thermal actuator assembly is mounted from the latter so that its heaters and valve position sensor can operate near ground potential. Each actuator has a Teflon§ tipped piston which bears against an insulator rod in the valve drive screw yoke.

Power and Signal Conditioning

Spacecraft a.c. power is provided to accomplish all functions, with the exception of propellant valve power which utilizes unconditioned spacecraft d.c. power. External commands in the form of relay contact closures sequence the various power and signal conditioning outputs for thrust level selection and turnon and turnoff of the microthruster system. Telemetry signals are provided for needle voltage and current, extractor current, neutralizer heater current, neutralizer emission current, thruster heater current, thruster base temperature and valve position. The prototype P/SC unit (Fig. 5) differs from previous design[2] in that it furnishes conditioned power for one thruster (rather than two). Two current settings for constant-current operation at 18 and 36 μa are available, corresponding to 4- and 8-μlb thrust levels, respectively. Two corresponding heater settings, nominally 25° and 30°C, are also provided.

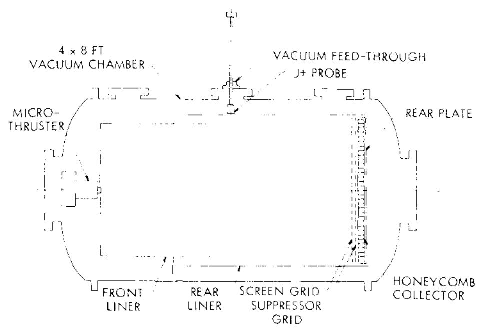


Fig. 6 Life test schematic arrangement.

§ Trade name, DuPont.

Life Test Facility

Microthruster system life testing was performed in a 4- × 8-ft vacuum test station. Figure 6 shows the schematic test arrangement of front and rear chamber liners and the collector assembly.

The collector assembly includes screen and suppressor grids, a honeycomb collector, and rear mounting plate. Screen and suppressor grids were fabricated from 4 × 4 mesh aluminum screen. Approximately $\frac{1}{8}$-in.[2] honeycomb, 1 in. deep, is employed for the collector.

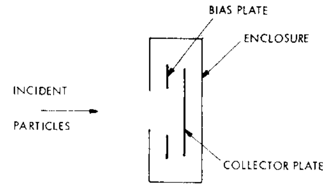
The collector and liners form an aluminum Faraday cage. The cage elements are electrically isolated from each other, and are connected together through external wiring. Polystyrene legs are used to isolate the liners from the tank. Shielded steatites are used to construct the collector assembly and fasten it to the rear liner.

The front liner has a mean diameter of 36 in. and a length of 30 in. It has a 4-in. access hole for the thruster plus two viewing ports which are covered with aluminum mesh screening. The rear liner is 42 in. in diameter and 52 in. long. It

Table 2 Interlock response to trouble signals

Trouble
1) Chiller water fails (action: a,b)
2) Floating collector potential exceeds 50 v (action: a)
3) Excessive temperature of diffusion pump cooling water (action: a,c,d,e,h)
4) Burnout of one element of diffusion pump heater (action: a,c,d,e)
5) Backing pump pressure exceeds 50 μ (action: a,c,d,e)
6) Vacuum tank pressure exceeds 2×10^{-5} torr (action: a,c,d,e)
7) Diffusion pump trap refrigerator failure (action: a,c,d,e)
8) Extractor current of 2.5 μa or more for 15 sec (action: a,c,d,e)
9) Needle voltage less than 5 kv for 5 min (action: a,c,d,e)
10) City power fails (action: a,c,d,e)
11) Vacuum tank pressure exceeds 5×10^{-3} torr (action: a,c,f,g)
12) Extractor current of 2.5 μa for 5 min or more (action: a,c,f,g)
Note: a) alarm rings
b) switch to city water
c) valve closes
d) maximum needle current demanded
e) thruster heater off
f) power to microthruster off
g) diffusion pump gate valve closes
h) diffusion pump power off

Fig. 7 J+ probe schematic.



dovetails with the front liner to prevent electron or particle leakage to the tank in the charged condition. It has one screened side hole for pumping.

A life test control console provides the following functions during test: input power, commands, data monitoring and recording, and interlocks. Microthruster input power is derived from batteries through a d.c. voltage regulator and then an inverter to furnish square wave a.c. input at approximately 1 kHz. The microthruster batteries are trickle-charged by means of a commercial charger. Thus, in the event of facility power failure, there is a slight drop in d.c. voltage, but sufficient stored energy is available to operate the microthruster for many hours. The primary commands are main power on/off, high voltage on/off, and open/close valve. Additional commands are thruster heater on/off, needle current select, manual/automatic temperature select control, and neutralizer select.

During life testing, eight channels of data were recorded continuously on a strip chart recorder. For most of the test, these were 6 telemetry signals (needle voltage, extractor current, needle current, beam current or neutralizer emission current, valve position, and thruster temperature) plus floating collector potential and vacuum chamber pressure. Floating collector potential is developed across a 220 kΩ resistor in series with 100 MΩ to ground. Overvoltage in case of neutralizer failure is prevented by a 70-v neon bulb connected to the common Faraday cage elements in the floating collector condition. When the Faraday cage is grounded, total beam current is recorded instead of the neutralizer emission telemetry signal. The following metered signals were recorded at regular intervals during life test: screen and suppressor voltages, screen and suppressor currents, front and rear Faraday cage currents, collector current, rear plate current (collector backing plate), diffusion pump heater current in each leg, foreline pressure, running hours of operation, input rms voltage and current, and rms voltage across the neutralizer.

The life test was interlocked as outlined in Table 2. This was accomplished by a multiplicity of logic relays in the vacuum control console and the microthruster control panel. Power to the vacuum controller is provided by a trickle charged 12-v battery in conjunction with a 12 v d.c. to 110 v a.c. inverter. This assures operation without fail even with loss of facility power. City water is available to back up the chill water supply to the diffusion pump cooling coil. A standby nitrogen bottle is used to back up the compressed air supply for the diffusion pump gate valve.

Special provisions were made in the event that any one of the three elements in the diffusion pump heater burned out or shorted to ground. Each leg of the heater circuit has multiple breakers and a sensing circuit to monitor heater current. This provides a continual check on the condition of each individual heater element.

Life test performance data were obtained using the time-of-flight method described in Ref. 7. Access to the needle voltage is provided by a high-voltage feed-through on the microthruster mounting end flange on the vacuum chamber. An external electrical circuit is used to short needle voltage to ground through a thyratron tube when time-of-flight photographs are taken. The time-of-flight distance to the collector was 72 in. (1.83 m) during test.

Several efficiency factors need to be taken into account when evaluating time-of-flight data.[8] The data presented herein were not corrected for energy loss[9] nor beam spread.

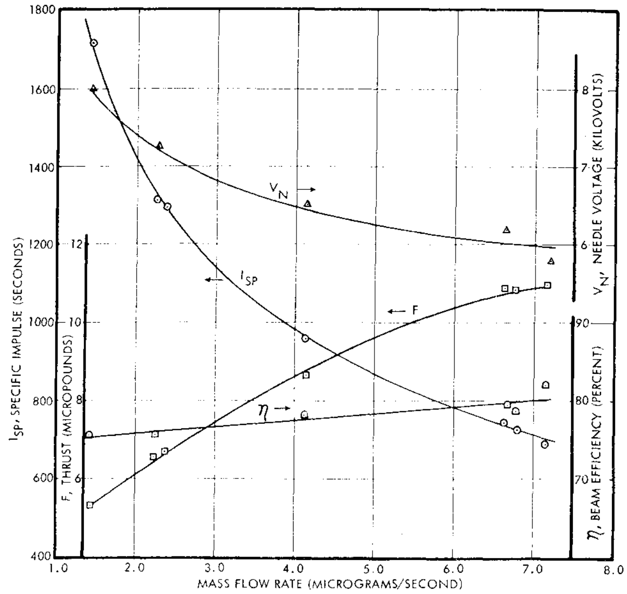


Fig. 8 Thrust, specific impulse, beam efficiency, and needle voltage vs mass flow rate.

Reference 4 presents a detailed discussion of energy loss considerations, and gives a correlation of time-of-flight data with direct thrust measurement. For example, using 400-v loss out of 6500-v accelerating potential at 8.7-μlb time-of-flight thrust, a corrected thrust value of 8.2 μlb is obtained.

The charged particle detector ($J+$ probe, Fig. 7) used for scanning the exhaust beam has a $\frac{1}{4}$-in.-diam aperture to accept incoming particles. These particles are collected on a target plate and are measured by means of a sensitive electrometer placed between the collector and ground. The bias electrode is maintained sufficiently negative to reject incoming electrons and suppress secondaries from the target collector. Thus, the $J+$ probe is used to measure the arrival rates of positively charged particles.

Microthruster Performance

The life test may be divided into five major time periods and post-test analyses, as follows.

Period 1 (Log hours 0–860)

During this period, microthruster operation was smooth and arc free, over a broad range of operating characteristics. Thrust ranged from 5 to 11 μlb and specific impulse from 1750 to 700 sec (Figs. 8 and 9). Extractor current was below detectability limits except on 22 occasions when arcs of about 10 to 30 sec duration occurred with about 2.5 μa peak. Sixteen of these arcs occurred during valve opening.

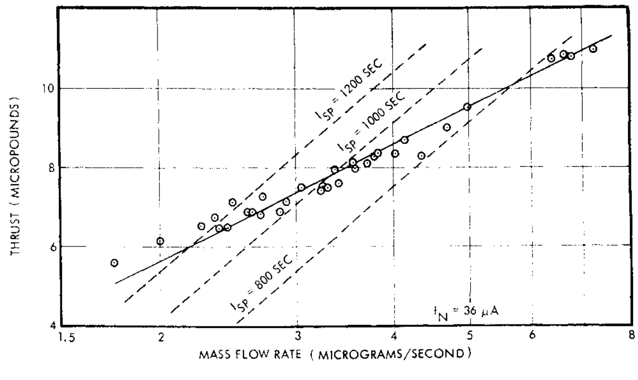


Fig. 9 Thrust vs mass flow at 36 μa needle current.

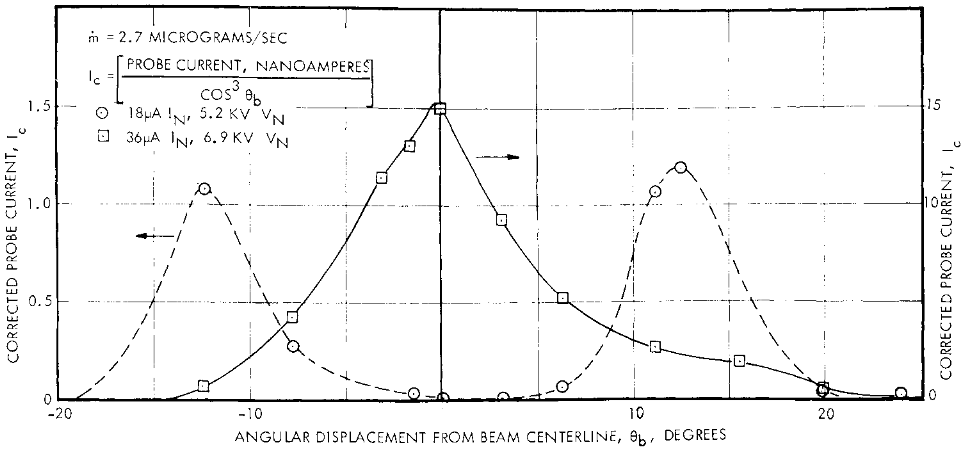


Fig. 10 Beam probe current vs angular displacement at 18 and 36 μa beam current.

It was found that this arcing could be prevented by first opening the valve to a very small opening stroke until a beam current was measured. A typical beam probe scan, shown in Fig. 10, exhibits a narrow exhaust beam. On the other hand, attempting to operate at 18 μa at the same mass flow lowers system operating voltage and gives a broad, hollow exhaust.

Because of high initial feed pressure, much of the operating data were obtained with the valve in the throttled condition. During this period, each valve closure (a total of 20) was accompanied by beam current decay to zero in a few seconds. The accelerating voltage could then be turned off for long periods of time (the longest period was 130 hr).

With the addition of a controlled oxygen leak to the vacuum chamber, neutralizer carbiding in the test environment was eliminated (caused by back-scattered material).

Period 2 (Log hours 860–887)

The microthruster was removed from vacuum, photographed, and placed back in test without cleaning while a faulty feedthrough insulator on the test facility was repaired.

Period 3 (Log hours 887–1297)

Thruster performance continued to be generally excellent. Mass flow showed a decrease with time. The valve could now be stroked to the full open condition. Beam current did not decay to zero for several hours after valve closure, indicating trapped gas in the thruster plenum from atmospheric exposure. Extractor arcing was much more frequent.

This period was concluded by electron leakage from the neutralizer to high voltage. This caused a drop in needle voltage for a 2-hr period, resulting in a propellant short to the extractor.

Period 4 (Log hours 1297–1484)

A constant extractor current from 0.3 to 0.5 μa persisted for a period of 70 hr and was independent of whether the valve was closed or not. The current leakage was finally corrected by operating the thruster with a closed valve and a high operating temperature.

Period 5 (Log hours 1484–2348)

A period of 3 hr after valve opening was required before the thruster exhibited constant beam current. Mass flow was now significantly lower at a given temperature, and it was continually decreasing. The nominal thrust requirements were met by constantly increasing thruster temperature, but $\dot{m}$ decayed faster, the higher the temperature. At the highest operating temperature of 40°C, $\dot{m}$ decreased 37% in a 50-hr period. Feed pressure was very low during this period as verified by turning off high voltage with the valve open for extended periods of time; $\dot{m}$ was primarily caused by capillary feed. Needle clogging may have been partially responsible for the decreasing $\dot{m}$. Thruster performance deteriorated significantly. High acceleration voltages were required to avoid beam spread, and specific impulse was significantly lower than during Period 1.

Post-Test Analysis

Loss of pressure during the test was explained by the location of detectable leaks in the pressurizing gas tank. Electron leakage was traced to holes through support insulators between the feed system and microthruster housing. The original thruster operational characteristics were recovered by cleaning the thruster. The original beam decay characteristics with valve closure evident in Period 1 were repeated after cleaning to the valve seat. Electron leakage was not evidenced after covering the support insulator holes.

Neutralizer Operation

One neutralizer operated a total of 950 hr. During this time, the rms voltage across the neutralizer was monitored. A plot of this voltage with time is shown in Fig. 11. The rapid rise in voltage during the first 25 hr occurred prior to introducing an oxygen leak to the vacuum environment. This voltage and corresponding filament resistance increase was caused by filament carbiding.

Since the oxygen addition was maintained at a constant partial pressure of 6×10^{-7} torr while total chamber pressure was decreasing during test, the ratio of oxygen to hydrocarbon

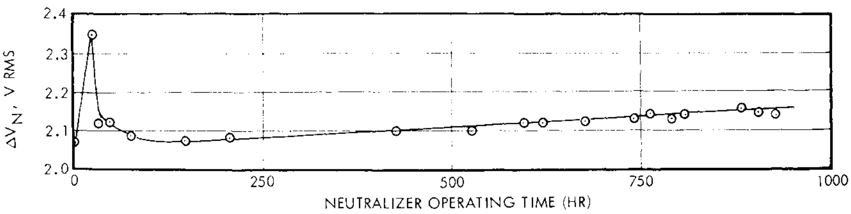


Fig. 11 Neutralizer filament voltage drop vs operating time.

molecules was probably increasing with time (total pressure was typically in the low 10^{-6} torr range). There is, therefore, a possibility that oxidation evaporation took place rather than true evaporation. The latter condition is deemed more likely. Neutralizer resistance will rise 12% during the expected filament life. The 4.5% rise in voltage drop across the filament indicated a resistance rise of between 3 and 4.5% based on previous experiments. Thus, the projected neutralizer life is between 2700 and 4000 hr.

The neutralizer had a measured emission-limited current of 177 μa. If the current had been 50 μa instead, neutralizer lifetime would have been greater than 10,000 hr. In order to increase filament lifetime, small trimming resistors could be added to each neutralizer to yield more optimum emission characteristics. By testing the neutralizer in an oxygenated vacuum chamber, embrittlement caused by filament carbiding can be avoided. Thus, the filament should readily survive a launch environment.

The neutralizer maintained a floating Faraday cage within the test chamber at about 30 v during test. The cage was normally left floating during operation, and was grounded for time-of-flight and beam current measurements. Neutralizer emission was monitored while the cage floated. When neutralizer emission measurements indicated electron leakage from the cage, neutralizer operation was terminated.

Mass Flow Observations

Normalized $\dot{m}$ and temperature vs time are shown in Fig. 12; the circled numbers at the bottom correspond to the 5 major time periods. Here $\dot{m}$ is normalized to 30.5°C thruster temperature by multiplying by the ratio of propellant viscosity at observed thruster temperature to the viscosity at 30.5°C. This ratio was determined from data in Ref. 7. For laminar flow conditions, with the primary flow impedance in the thruster needles, $\dot{m}$ is inversely proportional to viscosity at constant feed pressure. The normalized data are valid if the primary flow impedance remains within the thruster needles and the flow reaches steady state conditions. The data from 400 to 810 hr log time are somewhat suspect because of operation in a throttled valve condition. These data are therefore *minimum* mass flow rates.

Normalized $\dot{m}$ decreased by a factor of 3 in 1800 hr. Less than 10% of the available propellant was used during this time. This accounts only for less than 5% pressure drop. The decreased $\dot{m}$ was, therefore, caused by some other factor. It was later determined to be loss of expulsion pressure. As previously noted, in period 5 thruster temperature had to be continually increased to maintain $\dot{m}$. At 1500 hr log time, an external potentiometer was added to the set temperature command switch on the control console so that any desired thruster temperature could be maintained. At the final 40°C thruster temperature, $\dot{m}$ decreased faster than at lower temperatures; for a given $\dot{m}$, however, thruster performance remained substantially the same. The accelerated decline in $\dot{m}$ probably was caused by elevated thruster temperature operation in a slow flow regime where propellant was not being drawn away from the needles fast enough to prevent NaI precipitation as glycerol evaporates. At 40°C, the evaporation rate of glycerol from a 12-mil-diam hemisphere is 0.08 μg/sec. For a three-needle thruster, it would be 0.24 μg/sec. At 2200 hr log time, $\dot{m}$ had decreased to 1.7 μg/sec at 40°C; thus, the evaporation rate was 14% of the spray rate. As precipitate builds up, this type of clogging becomes progressively worse.

The slow propellant drainage from the volume between the valve seat and needle tips when gas is present and the valve is closed can also lead to needle clogging by the same mechanism. This condition persisted for the latter portions of the life test after the microthruster was exposed to atmosphere in Period 2.

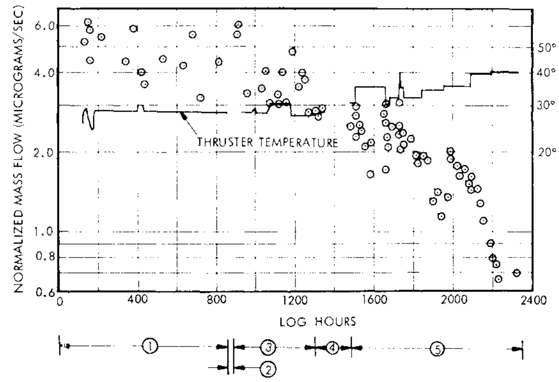


Fig. 12 Normalized mass flow rate and thruster temperature vs time.

System Response to Valve Actuation

The valve was opened in the vacuum test chamber 40 days after completion of propellant loading. During most of this interval the feed system was in atmosphere. The initial startup transient showed that the valve protected propellant from atmospheric permeation during propellant storage prior to test. If gas had leaked past the valve seat, thruster arcing during startup would have been evident. In addition, needle current would slowly decay to zero upon valve closure. Actual valve closure was accompanied by a thrust decay to zero in less than 10 sec. Typically the decay rate was from 90% to 10% thrust in 2.5 sec.[6]

Although beam current dropped to zero within seconds after valve closure, a small incremental beam current often commenced several minutes later, decaying to zero in about 1 hr. This was caused by heat transfer from the thermal actuators to the trapped volume of propellant between valve and thruster. The problem was eliminated by removing heat to the thruster immediately following valve closure to yield a net temperature decrease with time.

System turnon and turnoff response was limited by thermal actuator time constants.[6] With gas-free propellant in the thruster, starting and stopping transients displayed reproducible characteristics.

Power Conditioning Operation

The power conditioning unit required a maximum 5.0 w input for 0.3 w beam power, an 0.8 w neutralizer, and 0.3 w heater.

During prototype life test, a power conditioning unit was operated at 20 v rms input voltage and about 4.5 w input power. It operated without failure in excess of 2000 hr. A variation in the thruster heater cycling rate was interesting but of no significance. Thruster temperature was maintained constant at any given setting within the limits of detectability. A false needle current telemetry signal occurred at voltages above 8 kv. This is a higher voltage than would normally be used.

Conclusion

The test indicated thruster lifetime well in excess of the 1792 hr operation demonstrated. The projected neutralizer life of 2700 to 4000 hr can be improved by adding a small trimming resistor in series with each filament to give optimum emission characteristics. The bladderless blowdown feed system offers a passive and potentially reliable means for storing propellant and expelling it over a reasonable feed pressure range. The existing gas pressurizing chamber assembly methods need revision to prevent repetition of the gas leaks observed. The power and signal conditioning unit operated without failure for over 2000 hr. This life test was conducted as part of an evaluation of the 8μlb thruster system with respect to geosynchronous satellite requirements. As a result, colloid propulsion has been shown to be compatible with typical flight specifications.

References

[1] Benson, R. A., "Earth Orbital Mission Requirements for Secondary Propulsion Systems and Their Impact on Colloid Systems," *Transactions of the ASME*, Vol. 92B, No. 3, Aug. 1970, pp. 621–627.

[2] Huberman, M. N. et al. "Present Status of Colloid Microthruster Technology," *Journal of Spacecraft and Rockets*, Vol. 5, No. 11, Nov. 1968, pp. 1319–1324.

[3] Zafran, S., Beynon, J. C., and Cohen, E., "Colloid Microthruster System Development," AIAA Paper 68-84, New York, 1968.

[4] Huberman, M. N. and Kidd, P. W., "Charged Particle Electrostatic Thrusters," TR AFAPL-TR-69-14, March 1969, Air Force Aero Propulsion Lab., Dayton, Ohio.

[5] Cohen, E., Burson, W. C., Jr., and Herren, P. C., Jr., "Colloid Thruster Technology '69," Eighth International Symposium on Space Technology and Science, Japanese Rocket Society, Tokyo, 1969.

[6] Beynon, J. C., "Valve Development for Colloid Thrusters," AIAA Paper 70-615, San Diego, Calif., 1970.

[7] Kidd, P. W., "Parametric Studies with a Single-Needle Colloid Thruster," *Journal of Spacecraft and Rockets*, Vol. 5, No. 9, Sept. 1968, pp. 1034–1039, and Vol. 5, No. 12, Dec. 1968, p. 1504.

[8] Zafran, S., Meissinger, H. F., and Cohen, E., "Colloid Thrusters," *Aviation and Space, Progress and Prospects*, ASME, New York, 1968.

[9] Huberman, M. N., "Measurement of the Energy Dissipated in the Electrostatic Spraying Process," *Journal of Applied Physics*, Vol. 41, No. 2, Feb. 1970, pp. 578–584.

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Microthruster Development in France under Direction of the Centre National d'Etudes Spatiales (CNES)

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In the CNES program on micropropulsion for orbit control of satellites, priority has been given to geostationary satellites. Low-I_{sp} (60-sec) thrusters (cold gas and subliming solid) will be available in France for satellites to be launched after 1970. Medium I_{sp} (200-sec) thrusters (hydrazine and ammonia) could be used after 1972. Ion engines (cesium contact + thrust vectoring through beam deflection) will be used after 1975. Other French studies, not supported by the CNES, have dealt with chemical or electric thrusters. For the 1970–1975 period, the CNES plans to concentrate its activities on hydrazine monopropellant thrusters, which are particularly simple and reliable, and ion engines, which permit important weight saving.

Nomenclature

DFVLR	= Deutsche Forschungs und Versuchsanstalt für Luft und Raumfahrt (German government aerospace research laboratories)
DRME	= Direction des Recherches et Moyens d'Essais (French Army research organization)
ELDO	= European Launcher Development Organization
ESRO	= European Space Research Organization
LEP	= Laboratoires d'Electronique et de Physique Appliquée (French electronics research company)
MATRA,SNIAS	= French aerospace companies
ONERA	= Office National d'Etudes et de Recherches Aérospatiales (French Air Force research laboratories)
RAE	= Royal Aircraft Establishment (U.K.)
SEP	= Société Européenne de Propulsion (French space propulsion company), formerly SEPR: Société d'Etude de la Propulsion par Réaction
SNECMA	= Société Nationale d'Etude et de Construction de Moteurs d'Avions (French aircraft propulsion company)
SAGEM, CdC THOMSON-CSF	= French avionics companies

Introduction

THE Centre National d'Etudes Spatiales (CNES) decides on the programs for, and implementation of, French space activities. Its annual budget of approximately $100,000,000 covers both national projects and French participation in international organizations such as ESRO and ELDO. At the outset, CNES was founded to support scientific research. In 1966, however, the development of applications satellites appeared to be a more important task, and it was then decided that CNES should be in a position to launch geostationary satellites during the 1970's. Although some attitude control studies had already been done, no microthruster with good performance and long lifetime had been developed in France. CNES created a new department to begin research and testing of hardware constructed by industry under CNES sponsorship and technical supervision. The main target has been to create a know-how rather than the development of one microthruster in particular.

This work was accomplished in collaboration with other organizations, both French (DRME and ONERA) and in-

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