NASA Technical Memorandum 100120
AIAA-87-2028

Electromagnetic Emission Experiences Using Electric Propulsion Systems—A Survey

James S. Sovey and Lynnette M. Zana
Lewis Research Center
Cleveland, Ohio

Steven C. Knowles
Rocket Research Company
Redmond, Washington

(NASA-TM-100120) ELECTROMAGNETIC EMISSION EXPERIENCES USING ELECTRIC PROPULSION SYSTEMS: A SURVEY (NASA) 37 p Avail: NTIS HC A03/MF A01 CSCL 21C

N87-22805

Unclas
G3/20 0077457

Prepared for the
23rd Joint Propulsion Conference
cosponsored by the AIAA, SAE, ASME, and ASEE
San Diego, California, June 29—July 2, 1987

NASA

ELECTROMAGNETIC EMISSION EXPERIENCES USING ELECTRIC PROPULSION SYSTEMS - A SURVEY

James. S. Sovey and Lynnette M. Zana
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

and

Steven C. Knowles
Rocket Research Company
Redmond, Washington 98073

SUMMARY

As electric propulsion systems become ready to integrate with spacecraft systems, the impact of propulsion system radiated emissions are of significant interest. Radiated emissions from electromagnetic, electrostatic, and electrothermal systems have been characterized and results synopsized from the literature describing 21 space flight programs. Electromagnetic radiated emission results from ground tests and flight experiences are presented with particular attention paid to the performance of spacecraft subsystems and payloads during thruster operations. The impacts to transmission of radio frequency signals through plasma plumes are also reviewed.

E-3618

INTRODUCTION

As electric propulsion systems become ready to integrate with spacecraft systems, the characterization of propulsion system radiated emissions are of significant interest. This paper briefly summarizes communications and dynamic electromagnetic experiences using electromagnetic, electrostatic, and electrothermal propulsion systems. Electromagnetic radiated emission results from ground tests and flight experiences are presented with particular attention paid to the performance of spacecraft subsystems and payloads during thruster operations. The impacts to transmission of radio frequency signals through plasma plumes are also reviewed.

Over the last 30 years more than 60 spacecraft were flown using electric propulsion systems for drag makeup, stationkeeping, and experiments (refs. 1 to 3). The major flight qualified thruster systems are ablative pulsed plasma thrusters (PPT), ion thrusters, magnetoplasmadynamic (MPD) thrusters, and resistojets. Average power dedicated to the propulsion system ranged from about 3 W for early pulsed plasma devices to about 1 kW for each SERT II ion thruster (table I) (refs. 1 to 21). Table I does not characterize all electric propulsion flights; only the publications which make reference to thrusters EM emissions or RF interactions with plumes were considered in this survey. Except for some resistojet systems, most electric propulsion systems require power processing equipment to tailor the battery or solar array power to that of the thruster. The thruster/power processor produces an electromagnetic environment that could potentially impact spacecraft systems such as communications, guidance, navigation and control, payloads, and experiments. The EM environments may have permanent and varying magnetic fields along with radio frequency and conducted electrical emissions. Prior to flight, electromagnetic interference (EMI) measurements must be made, and hardware must be immunized or the level of EMI reduced to satisfy the compatibility requirements.

Previous papers have surveyed the particle and field interactions using ion thrusters and MPD arcjets (refs. 22 to 24). This paper will review RFI component test specifications, results of spacecraft integration tests, and radiated emissions from flight systems using electromagnetic, ion, and electrothermal thrusters. Spacecraft schematics and ground test configurations will be discussed, general test procedures will be synopsized, and spacecraft integration test results will be reported. Most of the EM emission results were obtained from four flights using ablative pulsed plasma thrusters, six ion propulsion flights, and three flights using electromagnetic devices (table I).

Only published or publicly available information was used in compiling this report. Most of the electric propulsion work, pertaining to EM emissions characterization was performed in the United States, Japan, Russia, Germany, and China.

SYMBOL LIST

ACS	attitude control system
AFAL	Air Force Astronautics Laboratory
AGC	automatic gain control
ASC	American Satellite Company
ATS	Applications Technology Satellite
C-band	4 to 6 GHz range
COMM	communications
CW	continuous wave
dBm/Hz	decibels above one milliwatt per hertz
dBpT	decibels above one picotesla
dBμV/m	decibels above one microvolt per meter
dBμV/m/MHz	decibels above one microvolt per meter per megahertz
DC	direct current
EM	electromagnetic
EMC	electromagnetic compatibility
EMI	electromagnetic interference
ETS	Engineering Test Satellite
E/W	east/west
GTE	General Telephone and Electronics
HV	high voltage
INTELSAT	International Telecommunications Satellite
IR	infrared
Ku-band	10.7 to 18 GHz
LES	Lincoln Experimental Satellite
MIL-STD	military standard
MJS	Mariner Jupiter Saturn
MPD	magnetoplasmadynamic

MSFC-SPEC	Marshall Space Flight Center specification
N/S	north/south
PCU	power conditioning unit
pps	pulses per second
PPT	pulsed plasma thruster
PPU	power processing unit
RE01	radiated emission tests for magnetic field induction
RE02	radiated emission tests for electric fields
RF	radio frequency
RFI	radio frequency interference
RIT	Radio Induction Thruster
S-band	2.2 to 2.6 GHz
S/C	spacecraft
SCATHA	Spacecraft Charging at High Altitudes
SEP	Solar Electric Propulsion
SEPAC	Space Experiment with Particle Accelerators
SERT	Space Electric Rocket Test
S-K	station-keeping
SNAP	Space Nuclear Auxiliary Power
SPEC	specification
STS	Space Transportation System
TM	telemetry
TV	television
UHF	ultra high frequency
VHF	very high frequency
VO	Viking Orbiter
X-band	7 to 8 GHz range

ELECTRIC PROPULSION SYSTEM CHARACTERISTICS

Most of the electric propulsion devices flown to date require less than 1 kW of power per thruster and produce thrust less than 0.5 N. Low voltage power (16 to 50 V) has been provided to the electric thruster's power processing system from the spacecraft bus or battery system. Power converters tailor power to heaters, ignitors, discharge and accelerator electrodes, and neutralizers. Some resistojet systems simply couple to the batteries through switches and load resistors to eliminate start-up current surges.

Some of the basic characteristics of the Teflon PPT, the MPD arcjet, ion thruster, and resistojet are described in table II (refs. 18 to 25). The Hall current thruster and thermal arcjet parameters are also listed for comparison purposes. The PPT accelerates fluoropolymer gas products by establishing a discharge between two electrodes which bound the Teflon propellant. A capacitor, charged to 1 to 2 kV, is discharged across the electrodes after initial breakdown is produced by an ignitor plug. The polymer products are accelerated by gas dynamic forces and Lorentz forces produced by the discharge current and its self-generated magnetic field. The primary mission application of the Teflon PPT has been altitude control and stationkeeping of satellites. The NOVA-1 spacecraft, for example, uses an electro-optical sensor to activate two PPT's which null atmospheric drag and radiation pressure forces. Thrusters are mounted fore and aft on the spacecraft and produce a "drag free" environment (fig. 1) (ref. 8).

A MPD arcjet was flight tested in the pulsed mode in the SEPAC experiment aboard the STS-Spacelab-1 (ref. 11). The SEPAC MPD experiment provided data on basic plasmadynamic phenomena, vehicle charge neutralization, and air glow excitation. The MPD arcjet uses coaxial electrodes to produce a 240 V, 8 kA discharge, which ionized and electromagnetically accelerated argon propellant. The arcjet power processor input was provided by a 28 VDC bus. Basic elements of the power processor were a 480 V capacitor bank and charger, pulse forming network, and switches. The MPD arcjet processes nearly 25 KJ during a 1 ms pulse. The peak power (2×10^6 W) processed by the arcjet represents the highest power level of any thruster/PPU system flown to date. Figure 2 shows the location of the MPD arcjet assembly relative to the other diagnostic subsystems on the SEPAC pallet.

Ion thrusters have been integrated on spacecraft and demonstrated as compatible with the satellite subsystems. For example, geosynchronous satellite operations were accomplished on the ATS-6 flight of 2 cesium ion thrusters (ref. 17). Flight objectives were to demonstrate N/S stationkeeping, unloading momentum wheels, and spacecraft charge control. Each thruster provided 4.5 mN thrust at 2500 sec specific impulse. Sixteen power outputs to the thruster system were provided by 4 DC converters operating from an 18 VDC bus. The ion beam converter had a 500 V, 113 mA capability. Figure 3 shows the location of an ATS ion thruster relative to the spacecraft body, reflector, and antenna (ref. 26). About 2.6 mN of the thrust was applied normal to the orbital plane, and the two thrusters were operated so these thrust components would cancel. It was estimated that about 90 percent of the exhaust plume would not impinge on spacecraft components or structures.

Communication satellites, using up to 24 transponders for C-band or dual C-band/Ku-band communications, have employed resistojets for N/S stationkeeping (ref. 20). The RCA spacecraft use two resistojets simultaneously with two redundant thrusters in a standby condition. Hydrazine resistojets produce a mission average specific impulse of 290 sec and a total impulse capability consistent with an 8 to 10 year satellite mission life. The power to the thruster is provided from a nominal 28 VDC battery system to a solid state controller, resistance banks, and switches used to limit current spikes due to the increasing heater resistance during start-up. Operational heater resistance is achieved in less than 8 sec. In this case, thrusters are symmetrically body-mounted on the RCA spacecraft near the center of mass.

Because of the lack of detailed information on Russian flight systems, only the USSR flight of the Hall current thruster on the Meteor spacecraft is included in this compilation (ref. 9). Reference 9 reported, however, that at least 13 electric propulsion space tests have been undertaken by the USSR dating back to 1964.

RADIATED EMISSIONS FROM ELECTRIC PROPULSION SYSTEMS

Characterization of the electromagnetic environment produced by an electric propulsion system is an important consideration in component level testing as well as spacecraft systems integration. The propulsion system, along with all major spacecraft subsystems must demonstrate that radiated and conducted emissions do not pose a problem to susceptible spacecraft systems. Detailed ground testing is required to verify spacecraft subsystems do not produce conducted or radiated emissions that might upset digital circuitry, introduce false signals in command and telemetry, or impact other communication systems. For example, typical communications carrier signals are at frequencies of 2 GHz or higher. Radiated and conducted emissions from thruster power conditioning equipment can be well controlled or reduced with careful attention to the design of shields, gaskets, and filters. Table III lists the potential radiated emission sources for each of the electric propulsion concepts surveyed in this paper. As an example, the ion thruster, during normal operations, can produce RF emissions from the thruster discharge chamber, the neutralized ion beam, or the neutralizer plasma.

Application of the MIL-SPECS

Prior to flight, the propulsion systems usually undergo some type of ground-based EMI characterization at the component level and also upon spacecraft integration. The standards and test procedures (or some variation thereof) set forth by the Department of Defense were usually followed for EMI documentation. The two primary publications are MIL-STD-461B and MIL-STD-462 (ref. 27). The former establishes both the electromagnetic emission limits and susceptibility requirements for all electrical, electronic, or electromechanical equipment while the latter defines test procedures and measurement techniques. Table IV lists specific radiated and conducted emission limits as well as susceptibility requirements as established by MIL-STD-461B, along with the frequency range of interest for each specification.

Reference 27 provides a fundamental description of electromagnetic interference and compatibility principles and serves as a practical guide to implementation of the Standards.

Since this paper focuses on radiated emissions, only RFI test specifications will be discussed. Figure 4 displays the specification limit set by RE01, MIL-STD-461B for magnetic field emissions from 30 to 50 kHz. The limit is given in terms of magnetic flux density, and decreases at a rate of 40 dB per decade, from 140 dBpT at 30 Hz to 20 dBpT at 50 kHz. The reason for the decrease in allowable density is that magnetic field inductive coupling, such as between two or more wires, increases with frequency.

The test set-up for the RE01 magnetic field test has a magnetic-loop transducer placed 7 cm from the test sample or 7 cm from the wire harness. The loop sensor consists of multiple loop of Litz wire with a 13.3 cm diameter. The purpose of the test is to ensure that the magnetic field from the test sample will not inductively couple to nearby, susceptible components or wiring. It should be noted that the specification limits use the decibel system of notation where the picotesla (10^{-12} T) is the basic unit of magnetic flux density. The magnetic flux density must be converted to a meaningful EMI receiver reading by calibration of the loop-transducer.

Figure 5 displays the specification limits for both narrowband and broadband electric field emissions, as set by RE02, MIL-STD-461B. Here, the terms narrowband and broadband have a definite, quantitative meaning and describe the emission bandwidth with respect to a reference bandwidth, that of the EMI measurement receiver. The specification limits are given in terms of electric field intensity (dB μV/m). The RE02 test set-up has the antenna located 1 m from the test sample; both devices are located in a shielded enclosure. The purpose of the test is to confirm that electric fields produced by the test specimen or its wire harness are less than the narrowband or broadband emission limits over the 0.01 MHz to 10 GHz frequency range.

Scientific payload experimenters may have additional EMC requirements. For example, space plasma experiments can be affected by narrow band electric field emissions due to EM signals and their harmonics. Users from the University of Iowa suggest that the upper bound of allowable emissions at the instrument location be 10 μV/m for satisfactory performance of plasma diagnostic instruments (ref. 28). A thrusting/science time-sharing philosophy would probably produce the optimal EM environment.

It should be mentioned that the MIL SPECS are intended as a guide to ensure EM compatibility between all electrical and electronic components in use as an integrated system onboard the spacecraft. Clearly, there are components and circuitry that can withstand electromagnetic environments in excess of the limits imposed by the MIL SPECS. Overall system compatibility will govern the actual implementation of each specification. In many cases, the specifications may be relaxed or waivered. Further, definitive criteria do not exist for impulsive noise, such as that which would be generated by the pulsed plasma thruster, so special specifications are sometimes applied (ref. 29).

Ground-based EMI Documentation of Electric Propulsion Systems

All of the electric propulsion systems surveyed underwent some type of EMI characterization prior to flight. References 6, 18, 22, 30, and 31 present detailed information on the types of tests conducted, the experimental set-ups employed, and the ground test data measured along with discussions of experimental uncertainties. Although it is apparent that a full-scale, integrated documentation is necessary in order to ensure EM compatibility between all spacecraft sub-systems, meaningful information may be obtained at the component level, in order to prevent and/or minimize the communications impact due to EMI related problems. The primary engineering difficulty encountered in ground-based testing is the adequate simulation of the full-scale, spacecraft configuration in a near-space (vacuum) environment. Concerns about propellant contamination of sensitive spacecraft surfaces or about vacuum tank reflections of EM waves often precluded the positioning of the thruster and antennae in their actual flight orientation. The reflection of EM emissions from metal vacuum facility walls may give inaccurate EM field measurements and also can prevent conclusions on time-dependent EM characteristics. For example, about a 10 dB reduction in S-band signal variations was observed after absorber material was installed in a metal vacuum facility. If no reflections were present, the received signal would be nearly constant over the frequency range (ref. 32). Frequency shifts attributed to reflections in metal vacuum facilities were also detected.

Figure 6 displays the test configuration of a novel EMC facility developed specifically for the EMC characterization of the mercury ion thruster propulsion system, which was flown onboard the Engineering Test Satellite III (ETS-III) in 1982 (ref. 18). The ion thruster and the entire satellite were placed inside a large screen room. The ion thruster exhausted into an EMI simulator, a cryo-pumped vacuum environment of approximately 4×10^{-4} Pa (3×10^{-6} torr), during normal thruster operation. The simulator was fabricated with "fiber reinforced plastic", a high strength material of sufficient electromagnetic permeability. Despite some facility related effects that caused fluctuation of the AGC level on an uplink receiver, the test configuration demonstrated the ability to adequately document and assess EMI related impacts of the ion thruster to the satellite.

RESULTS OF GROUND BASED AND FLIGHT TESTS

Radiated emissions from electric propulsion systems can cover a broad range of electromagnetic frequencies. This paper will focus attention on the 10 Hz to 10 GHz range, where propulsion system electromagnetic emissions or plasma plumes might impact the operation of command, control, telemetry, or communication systems. A significant amount of published data exists from component level tests, integration tests, and flight experiences to characterize thruster produced electromagnetic fields, plume effects, and interactions with spacecraft subsystems. This information is synopsized in table V (refs. 3 to 63). These data have been obtained from spacecraft that, on average, provide less than 1 kW to each thruster. Peak power in the case of the MPD arcjet has exceeded 1 MW (ref. 11). The discussion of results will be organ- ized under 5 propulsion system classes, namely: ablative PPT, Hall current and MPD thrusters, ion thrusters, resistojets, and thermal arcjets. Since the arcjet is an attractive candidate for N/S stationkeeping, it is included as a subset even though there is no known flight experience. Each section will briefly discuss the results of propulsion system ground and flight tests as they pertain to radiated emissions and RF communications through thruster plasma plumes.

Ablative Pulsed Plasma Thruster

Published results of three flights using PPT's have indicated no evidence of EMI effects on spacecraft subsystems. The LES-6 spacecraft had four PPT's for E/W Stationkeeping (ref. 4). During component level and integration tests, it was concluded that the induced noise pulses must be less than 60 dB above the antenna receiver threshold noise and not exceed µ50 sec in duration (ref. 34). Test results of the PPT in a vacuum anechoic chamber satisfied the requirement. One of the PPT's operated at least 8900 hr over a 5 year period, and there was no evidence of interference with spacecraft communications or telemetry. The MDT-2A, a 37 min ballistic flight, was China's first electric thruster flight test (ref. 7). The PPT system, which has an average input power of about 5 W, did not disturb the telemetry or communication systems during flight. No obvious effects on spacecraft subsystems were noted during integration testing prior to flight.

The NOVA-1 spacecraft used two PPT's to cancel drag at an altitude of 1170 km (ref. 8). Although there was no published information on ground tests at the component level or on integration tests, during five months of flight experience and more than 1×10^{6} thruster pulses, no EMI effects on spacecraft subsytems were reported.

The LES 8/9 development program for a PPT flight system included extensive testing at the component and system level. These PPT's had improved thrust to power ratios compared to the LES-6 devices (ref. 38). LES-8/9 component EM emission tests were performed in a vacuum tank with RF absorbing walls. The radiation intensity and spectral distribution in the 0.2 to 20 GHz range was to be determined in this case with an accuracy of about an order of magnitude. Results indicated a radiated energy density of 6 to 60 μW/MHz which was estimated to be sufficient to saturate an antenna receiver in the vicinity of the PPT. The RFI was 0.5 to 1.0 sec duration (ref. 37). During spacecraft integration tests, EM radiation and electrostatic pickup other than plasma return currents affected operation of an infrared sensor and some logic circuits (ref. 38). The basic problem was inadequate shielding of electronic boxes and the wire harness. Because of the RFI difficulties and problems encountered in thruster thermal cycling in the spacecraft configuration, the PPT's were replaced on the spacecraft with cold gas ammonia thrusters in order to expedite the LES 8/9 flight. Although the PPT's were not used on LES 8/9, they were successfully integrated to the spacecraft after RFI problems were resolved.

Radiated emission from the ETS IV PPT was reported in 1981. Tests were conducted in a glass vacuum facility according to the MIL-STD-461 specification (ref. 6). After the prototype thruster system was retrofitted with additional shielding and high voltage cable filters, and the PPT was electrically floated, the average noise level was below the RE02 broadband specification except in the 100 to 120 MHz range. RE02 was satisfied at the uplink frequency of 148 MHz and the communication frequency of 1 to 3 GHz. Later, an EM compatibility test was conducted with the PPT and VHF and S-band antenna/receivers similar in characteristic to the ETS IV system. A 20 hr test at 1.4 pps was conducted. Occasional noise was estimated to be the same level as the command signal. The noise pulse was very narrow, and it was predicted that it would not interfere with uplink communications. Published space flight results of ETS-IV indicated that 31 propulsion system tests were conducted over a 3 year flight period. About $4x10^5$ thruster system firings were undertaken in a total operation time of about 100 hr. The PPT system was EM compatible with ETS-IV (ref. 40). There were no EMI effects on telemetry, command signals or spacecraft instruments. The AGC levels of VHF and S-band command receivers were unaffected by PPT operations.

In summary, an excellent description of EM emissions ground testing of the PPT was reported during the ETS-IV program. The LES-6, NOVA-1, and ETS-IV have demonstrated long-term flight operation of the PPT with no apparent indication of EMI effects on spacecraft subsystems.

Hall Current and MPD Thrusters

Three plasma thruster flights made some reference to RF emissions. The USSR Meteor-18 spacecraft accumulated over 600 hr of operating time on two Hall current thrusters with a total of 273 activations. Reference 9 indicates that there was no disruption of spacecraft communications produced by thruster system RF emissions. It was observed that "close location of receiving antennas and thruster in some cases brought a slight influence on the propagation of the radio signals." Generally, the RF signals were significantly higher than the signal noise and the quality of telemetry was not influenced.

The MS-T4 flight experiment of a 240 N MPD thruster operated in the pulse mode was performed by Japan in 1980 (ref. 10). During flight, the MPD thruster operated at a low duty cycle over a period of 5.6 hr. Pulses synchronized with thruster firings (1.5 ms) were observed on the AGC signals of the 400 MHz antenna system; however, the AGC signals of the S-band antenna system were not affected by thruster operations. Apparently, discharge current pulses of about 700 A did not seriously impact seven telemetry and five real time command signals.

At least 6 papers described EMI related component tests, integration tests and flight experiences of the SEPAC experiment aboard the STS Spacelab-1 (refs. 11, 30, 31, 45, 46, and 53). The MPD arcjet aboard this flight was coupled to a 240 V capacitor bank and produced a current pulse of about 8 kA for 1 ms. Arcjet peak power was about 2 MW during the 1 ms pulse. Based on the authors' review of the literature, the SEPAC MPD arcjet was the highest peak power electric propulsion device ever flown on a spacecraft. The basic objectives of the SEPAC test were associated with spacecraft charge control, airglow excitation, and plasmadynamics experimentation. Although specific propulsion experiments were not performed, the MPD arcjet discharge characteristics and on-orbit operational experiences were documented.

During ground based component tests, the MPD device was operated in a metal vacuum chamber. EM radiation levels exceeded the narrow and broadband RE02 SPEC (ref. 31). The EMI generally lasted as long as the MPD arcjet discharge period and was, therefore, thought to originate from the arcjet plasma. It should be noted that antenna locations were not optimized and there were probably significant reflections from vacuum facility metal walls, which may have enhanced the EM noise. Integration tests with the MPD arcjet mounted on the Spacelab pallet were later conducted in a metal vacuum chamber with the antennae located in the chamber. There were no malfunctions of flight hardware due to EMI during the course of this integration testing. RF emissions were above the MSFC-SPEC-521 at frequencies less than 100 MHz. The MSFC SPEC was satisfied over the 100 MHz to 10 GHz range implying no S-band linked RFI. Some click noises were observed on the UHF communication link.

The SEPAC flight experiment was also intended to serve as a testbed for MPD arcjet influences on UHF and S-band communications. Any EMI impacts on the STS Orbiter subsystems due to MPD arcjet operation would also be apparent. During the flight test, the arcjet was operated during thirteen time segments with a total of 260 firings. The MPD arcjet was usually operated in conjunction with an electron beam accelerator. Operations were usually conducted with a plasma diagnostics package, a TV monitor, and many other Shuttle Orbiter and Spacelab subsystems. Space plasma experimental data were readily obtained during the course of MPD arcjet operations. Experiments were related to vehicle charge neutralization, ambient gas ionization, plasma wave effects, and plasma plume dynamics (ref. 64). Reference 44 simply states that the MPD arcjet "induced neither hazard nor EMI with the Spacelab or other payloads." These results are particularly interesting in view of the MPD arcjet firings with peak power levels approaching 2 MW during the 1 ms pulse. As the MPD results to date have been accomplished using pulsed mode operation, radiated emissions from steady state propulsion systems will probably have different characteristics.

Ion Propulsion

Reports describing results of six flights of ion thrusters or ion sources contain information concerning RF emissions. The first ion thruster flight was the SERT-1 ballistic flight (ref. 12). A cesium ion thruster failed to start while the mercury thruster operated 31 out of the 46 min. trajectory. A total of 53 electrical breakdowns occurred during thruster operation. Nonetheless, during the course of steady state ion beam operation or during the HV faults, no anomalies were produced in the telemetry, command, or control systems.

The SNAP-10A was a reactor powered satellite which included a 9 mN cesium ion thruster as an experiment (ref. 13). Abnormal thruster arcing of high voltage produced severe effects on telemetry, the control system and horizon sensors. Severe vehicle slewing occurred, abnormal amounts of propellant were used by the attitude control system, and telemetry multiplexer synchronization was upset. Arcing at high voltage terminals produced power supply cycling, which in turn caused deleterious conducted EMI below 1 MHz. Such damaging impacts reinforced the need for detailed system integration testing.

In the SERT-II program, the prototype spacecraft was thermal vacuum tested in excess of 3200 hr with thruster operation of a period greater than 2400 hr (ref. 16). These spacecraft integration tests established long term electrical compatibility and also provided an evaluation of thruster arcing problems. In the actual flight demonstration of SERT-II, the thrusters were operated with ion beam extraction for at least 6900 hr over the 11 year life of the spacecraft (ref. 50). RF emissions produced by the thruster system had no effect on spacecraft command or control. Up and down link data (136 to 149 MHz) was also unaffected. Further, thruster operation was not detected on 1.7 and 2.1 kHz RFI sensors; sensor output was about the same as one would expect from thermal-generated Earth noise (refs. 16 and 47).

Two cesium ion thrusters were aboard the ATS-6 spacecraft (refs. 17 and 26). System integration tests were conducted with a thruster simulator and a flight power processing unit. No interference was observed between the thruster simulator/PPU and other spacecraft subsystems during normal high voltage operation or during HV overload situations. During the ATS-6 flight, a thruster operated 92 hr with no evidence of RFI to telemetry, command systems, television relay, or communication systems. The basic communication frequencies were 153 MHz, 2.25, 4, and 6.15 GHz. High voltage breakdowns normally occurred during the thruster start-up period. During these events, the communication equipment did not detect the presence of the ion thruster, indicating compatibility of the two subsystems during both transient and steady state operation (refs. 15 and 52).

Component level tests of a mercury ion thruster flown on ETS-III were conducted in a glass bell jar with the receiver antennae and PPU at atmospheric pressure (refs. 18 and 30). Broadband noise in the 3 to 100 MHz range exceeded the RE02 SPEC. The RE04 SPEC was also exceeded in the 20 to 70 KHz range. Consequently, noise specifications for the thruster system on ETS-III were relaxed slightly. The electromagnetic compatibility of the ion thruster system and the ETS-III was later confirmed using an EMI simulator or plastic vacuum chamber. The spacecraft was tested in an anechoic room with the thruster mated to the vacuum chamber flange (fig. 6). During the course of the integration testing, the AGC level of the VHF command receiver indicated poor reception. This effect, however, was thought to be associated with the test configuration. No fluctuation of the AGC level was observed with S-band. During actual flight test operations, there was no indicated increase in the AGC level of VHF signals implying compatibility of the ion thruster with VHF communications (ref. 53). One of the thrusters had operated at least 27 hr with 13 restarts, while a second thruster operated at least 182 hr and was restarted 111 times. All spacecraft operations and parameters were reported to be within allowable ranges during ion thruster operation implying EM compatibility with ETS-III subsystems including power, telemetry, tracking, command, attitude control, and thermal control(ref. 54).

A xenon ion source was flown on the P78-2, SCATHA satellite (ref. 19). The spacecraft integration test program was conducted with the ion source mated to a small vacuum chamber (ref. 56). Only conducted emissions and susceptibility tests were conducted on the component level. Spacecraft integration testing was performed for 2 hr, with all system power, experiments, telemetry, and command systems functioning properly. No problems were encountered with RFI or experiment performance. In flight, the ion source was operated in the pulsed mode on 392 occasions during March/April of 1979. The ion source operated at low duty cycle for more than 1 year with no indications of spacecraft anomalies (ref. 57).

The German high frequency ion thruster, RIT-10, will be flown on a direct broadcast satellite TV-SAT (ref. 58). MIL-STD-461 SPECS were applied for EM compatibility tests. The thruster was mounted in a glass vacuum chamber with antennae and power processing hardware located outside at atmospheric pressure. The background noise was below the MIL-STD-461 limits over nearly all of the frequency range, so RFI tests of the thruster could be undertaken. RF emissions from 10 MHz to 10 GHz did not exceed the RE02 SPEC. Some broadband noise, however, did exceed the RE02 SPEC at less than 10 kHz and a waiver concerning EMI requirements may be necessary. No RFI was observed in frequency ranges of command or telemetry during steady state or HV fault clearing operations.

A significant number of EMC test programs were undertaken by the Jet Propulsion Laboratory during the solar electric propulsion development program. Early tests in 1973 were conducted in atmosphere with a PPU, solar array simulator and a resistive load (ref. 59). Significant RF emissions were recorded during steady state operation and HV breakdown simulation (table V). The primary cause of the high RF emissions was poor shielding of a 5 kHz power cable. EMC tests in a metal vacuum chamber were later reported in 1981 (ref. 22). The RF emissions were at least 40 dB above ambient noise from 5 kHz to 1 MHz. In the frequency range of 10 kHz to 80 MHz, the RF broadband emissions were independent of ion beam current over the range of 0.75 to 1.75 A. This beam current range is about 40 to 90 percent of the design beam current level. The warm-up mode, which only has the discharge chamber plasma in operation, was much noisier than the ion beam extraction operation over the frequency range of 500 kHz to 80 MHz. Electric fields in space vacuum will probably differ from the aforementioned results, because of field perturbation by the metal vacuum tank walls. Additional EMC testing was undertaken to evaluate the effectiveness of spacecraft S-band communication through the ion beam exhaust. These data are reported in a separate section of this paper.

Resistojets

In the years 1965 to 1971, over 20 spacecraft utilized more than over 50 resistojets for stationkeeping, orbit maintenance and experiments. Propellants were either nitrogen, ammonia or hydrazine. Resistojets operate directly from spacecraft batteries or use the most rudimentary power processing. Since resistojets do not produce a plasma, any EM emissions would come from the heater coil, wire harness or power processor. The first mission application of resistojets was for orbit adjustment of the Vela III spacecraft using nitrogen resistojets (ref. 60). No system anomalies or failures during resistojet operation were reported. In 1968, the ATS IV used an ammonia resistojet system with 18 mN thrusters (ref. 14). Each low power resistojet on ATS IV used a dc/ac power converter and a transformer closely coupled to thruster with a tubular heater element. The resistojet propulsion system operated over 800 hr without interference to command and telemetry operations. The INTELSAT V series of spacecraft used four hydrazine resistojets to perform N/S stationkeeping (ref. 21). The thrusters were first operational in January 1981. Mission average specific impulse was 295 sec with each thruster consuming about 1.1 W/mN over a thrust range of 220 to 490 mN. EM emissions from the INTELSAT-V resistojet were probably minimized since the heater coil was immersed in the flow in a metal vortex chamber upstream of the nozzle. Assessment of early spacecraft maneuvers indicated that operation and performance of the spacecraft was "entirely nominal for system and orbit conditions." There were no apparent impacts of RFI to spacecraft command or telemetry subsystems.

Spacecraft developed by RCA use hydrazine resistojets with a coiled wire heater which radiates to an annular heat exchanger which surrounds the coil. The heaters are operated directly from a DC power bus. Many successful hours of operation on the SATCOM IR and SATCOM IIR spacecraft have been reported (ref. 20). No anomalous behavior of the thruster or spacecraft subsystems was detected. Thrusters of this genre are qualified and operational on a number of RCA developed spacecraft.

Thermal Arcjets

Arcjet thrusters have application for satellite N/S stationkeeping at 0.5 to 1.5 kW power level and also primary propulsion, where the power levels per thruster may be in the 30 to 100 Kw range. A program to develop a low-power hydrazine arcjet thruster for spacecraft stationkeeping is underway (refs. 25, 65, and 66) The 1.5 kW hydrazine arcjets are expected to operate at a specific impulse of more than 500 sec and at thrust levels in the range of 120 to 180 mN. Thirty kilowatt ammonia arcjets are under development for a space nuclear power system reference mission (ref. 24). Nearly 100 kW electric power would be provided to produce about 8.5 N thrust from the arcjet thrusters.

Arcjet radiated emissions may span from 10 kHz to 30 GHz. There are several potential sources. Broadband voltage oscillations of 15 to 200 kHz have been reported by investigators of 30 kW arcjets (refs. 67 and 68). These fluctuations were attributed to rapid movement of the arc attachment point. Ground shielding will probably contain radiation caused by arc induced broadband emissions.

The arcjet plume is a second possible EM emission source. Recombination of ionized and dissociative species and thermal energy level transitions are accompanied by some level of emission It is expected that, because of the broad frequency range that extends past the visible region, the energy level of any one frequency will be very low. However, no EMI measurements of an arcjet plume has been conducted to date.

The PPU, which is common to most electric propulsion systems, is another source of radiated EM emissions. Anticipated PPU switching frequencies will be in the 20 to 100 kHz range. A narrow, high voltage pulse is used to initiate the arc discharge. It is typically less than 10 μs wide and has very little available current. After the initial breakdown, the arc current increases to its steady state value with over a millisecond. This sudden current rise may induce EM emissions. However, the PPU can be designed to control the rate of increase to an acceptable level. The process of starting an arc is not unique to arcjet thrusters. Arc welders employ similar techniques and are often operated in EMI sensitive environments. Starting an arc does not produce prohibitive EMI in these applications.

Emissions caused by arc attachment at the electrodes will likely be attenuated because of the surrounding metal anode. Energy radiated from the plume is distributed over a broad range of frequencies in the infrared to ultra violet spectrum because of the recombination energies. No individual frequency will transmit at a significant energy level. Further investigation of the EM effects of plasma plumes are required.

TRANSMISSION IMPACTS DUE TO ELECTRIC PROPULSION SYSTEMS

Problem Definition

It is often necessary for carrier signals for both uplink and downlink communications to be transmitted through some portion of the thruster plume. The potential interaction between the radio frequency (RF) signal and the charged species of the thruster plume may present a serious communications impact. Typical RF signal-plume interactions include: reflection of the transmitted signal by the plume; attenuation and phase shift of the signal as it passes through the plume; and generated noise on both signal and amplitude and phase.

Free electrons in the plume, because they are light weight and very mobile, are the major source of interaction with RF transmissions. They readily absorb more incident energy than their ionized counterparts. However, at lower frequencies, particularly below 500 MHz, the ion contribution also becomes important.

Reflection of the transmitted signal by the plume depends upon the dielectric characteristics of the plume boundary. Generally, if the plume can be considered to be adiabatic, that is, if the plume characteristics vary slowly near the boundary, reflection losses are minimal (ref. 32).

RF attenuation and phase shift result from a combination of physical mechanisms including absorption, refraction/scattering, and diffraction (beam spreading). Absorption, or line-of-sight attenuation, removes energy from the electromagnetic fields in the vicinity of the plume and converts it to other forms of energy. If the RF signal must pass through a region of the plume which is "overdense," e.g., plasma electron frequency is greater than the RF frequency, severe attenuation may occur, particularly at lower frequencies (ref. 13).

If the RF frequency is greater than the plasma electron frequency, absorption of the transmitted signal is not expected to occur and the primary transmission loss mechanisms are diffraction and refraction. These mechanisms may redirect the incident energy through large angles. Given the required pointing accuracies of the transmitted beam and receiver antennae for deep-space communications, refraction of even a few degrees becomes significant (ref. 13).

RF-Plume Interaction Calculations

A guide for the prediction and evaluation of the interaction effects caused by a rocket exhaust may be found in reference 69. Generally, the plume may be thought of as a homogeneous, isotropic plasma which modifies the existing electromagnetic field state in space. The two primary plume properties which affect the steady state RF-plume interaction are electron number density and electron collision frequency. Other factors include: the RF power density and frequency, overall spacecraft geometry (propagation path), static magnetic field intensity, and ion density and mass.

Sellen calculated the refraction effects on 2.3 GHz waves transmitted through the exhaust of an ion thruster beam (ref. 13). He found that the principle refraction of the wave occurs in the encounter with the dense, uniform core of the ion thruster plume. However, some refraction may also result from the "exponential wing region" of the plume, especially if it is broad. Total refraction of the wave front in excess of 1° occurred if the propagation path encountered the thrust beam within the first five meters of the plasma column.

The effects of the exhaust plasma of a 30 cm mercury ion thruster on S-Band signals was investigated both experimentally and analytically (refs. 32 and 70). An illustration of the experimental arrangement is shown in figure 7. It was found that signal amplitude loss and phase advance increased with ion beam current. Maximum amplitude loss was 0.38 dB or 8 percent. Phase shifts advanced as high as 67 degrees. The effect of this phase shift is only significant during fast ion beam current transients (ref. 70). No additional noise was observed on the carrier frequency during normal thruster operation, and the carrier frequency was not offset due to plasma effects. Poor space simulation may have existed in these experiments due to exposed metal vacuum tank surfaces causing signal reflections and an abnormal EM environment. Improvements in analytical modeling of electron density distributions and RF beam spreading are needed in order to confidently predict plume/RF carrier interactions with space flight geometries.

The SERT-II spacecraft had a wideband RF antenna that viewed Earth through the ion thruster plume. Antenna/receiver bands at 1700 and 2110 MHz indicated noise levels expected from a thermal Earth indicating no apparent communication impact to S-band (refs. 16 and 48).

Corrective Actions

Minimization of the potential impacts to RF transmissions caused by interaction of the RF signal with the thruster exhaust requires spacecraft design and operational constraints. In general, most transmission impacts may be avoided if the RF propagation path encounters only the less dense regions of the thruster plume. This can be accomplished in practice by placement of the thrusters and communications antennae on the spacecraft such that there is a wide angular separation between the RF lines-of-sight and the thrust axes. The spacecraft itself may be used as a shield. Further, using higher frequencies for transmission will negate the absorption loss mechanism. A thrusting/user-time sharing philosophy would produce the optimal environment for spacecraft uplink/downlink communications.

CONCLUDING REMARKS

Characterization of the electromagnetic environment produced by an electric propulsion system is an important consideration in component level testing as well as spacecraft systems integration. The propulsion system, as with all major spacecraft subsystems, must demonstrate that radiated and conducted emissions do not pose a potential problem to spacecraft systems such as communications, guidance, navigation and control, payloads, and experiments. Also, because of the potential impact to communications, the interaction of the thruster plume with RF signals transmitted through the plume must be well understood. This survey paper has provided a brief summary of communications and dynamic electromagnetic experiences using electromagnetic, electrostatic, and electrothermal propulsion systems. Five propulsion system classes were reviewed: namely, ablative pulsed plasma thrusters, Hall current and MPD thrusters, ion thrusters, resistojets, and thermal arcjets. Results of ground-based component and systems level tests, along with the results of 21 flight experiences, have been presented. Some of the significant experiences with regard to EM emission characterization and impacts are highlighted.

In 1986, the LES-6 spacecraft using pulsed plasma thrusters demonstrated at least 8900 hr of flight operation over a 5 year period with no indication of EMI effects on spacecraft subsystems. Other PPTs, such as the two onboard the NOVA-1 spacecraft launched in 1981, have produced at least 1×10^6 operation cycles with no evidence of EMI. Also, an excellent documentation of ground-based EMI measurements during the ETS-IV development program was reported. Component level tests conducted in accordance with military specifications indicated that radiated emissions from the PPT were above overall compatibility requirements. Once countermeasures to reduce the level of radiated emissions were employed, the prototype PPT successfully completed a 20 hr, systems level compatibility demonstration.

A USSR plasma thruster has accumulated over 600 hr operating time and 27 activations without major impact to spacecraft subsystems. An MPD arcjet operated at 8 kA in the pulsed mode aboard Spacelab-1 for 260 firings without hazard to the spacecraft or its payloads. The MPD arcjet peak power was about 2 MW during the 1 ms pulse and, thus, was the highest peak power electric propulsion device flown to date. During the course of MPD arcjet operations, space plasma experimental data were readily obtained, and there were no reports of abnormal Shuttle Orbiter or Spacelab-1 operations.

Ion thrusters have been operated in excess of 6900 hr over an 11 year period on the SERT-II without impacting spacecraft command or control. A broadband S-band antenna viewing Earth through the ion thruster plume did not show evidence of communications impact. Detailed EMC ground tests were conducted during Japan's ETS-III ion thruster development program. The electromagnetic compatibility of the mercury ion thruster system and the ETS-III were confirmed with the use of a novel EMI simulator. During flight operations, ion thrusters accumulated more than 200 hr of operation demonstrating EM compatibility with ETS-III subsystems.

In terms of electromagnetic contamination, the resistojet is the most benign of the electric propulsion systems surveyed in this report. During the last 20 years, over 30 flights using resistojets for N/S stationkeeping, orbit adjustments, or experiments have been successfully accomplished with no indication of EMI associated with the thruster heater coils, wire harness, or power processing units.

The thermal arcjet, although it has no record of flight experience or ground-based EMI testing, was included as a subset to this survey because it is an attractive propulsion alternative for N/S stationkeeping of communications satellites. It is expected that the experiences and practices developed for ion and plasma propulsion systems can be applied to the arcjet thruster.

Available literature documenting quantitative impacts to transmission of RF communications signals through the electric thruster plume was scarce; however, many of the references cited in this survey indicated that the presence of a plasma plume did not significantly affect the transmission of RF signals for command and telemetry. The effects of the exhaust of a 30 cm mercury ion thruster was investigated both experimentally and analytically during the Solar Electric Power development program. It was reported that signal amplitude loss and phase advance increased with ion beam current.

In summary, the electric propulsion systems which were surveyed in this report had distinctly different operating characteristics and power processing requirements. Average power per thruster ranged from 3 W to 1.6 kW, with peak powers as high as 2 MW. The mode of operation was either steady state or repetitively pulsed. Nonetheless, it may be concluded that, although electromagnetic contamination from electric propulsion systems is a serious integration concern, the bulk of flight experiences to date indicated that MPD, PPT, ion thrusters, and resistojets do not adversely impact spacecraft mission completion. It must be stressed, however, that comprehensive EMI documentation at the component and systems level is a critical element to the successful integration of the electric propulsion system to the spacecraft. The primary engineering difficulty encountered in ground-based EMI testing is the adequate simulation of the real-space, vacuum environment. Further advancement in the field of electromagnetic compatibility requires more sophisticated ground test facilities to accurately measure electromagnetic emissions during developmental compatibility tests. It has been demonstrated that better EM backgrounds can be provided through the use of nonmetallic vacuum facilities with RF absorbing liners. Further, detailed descriptions of the pulsed plasma, ion, and arcjet plumes are needed to better understand the potential interaction between the plumes and communications signals.

REFERENCES

1. Molitor, J.H., "Considerations on Technologies and Missions for Non-Chemical Propulsion," IAF Paper 83-400, Oct. 1983.

2. Vondra, R., Nock, K., and Jones, R., "A Review of Electric Propulsion Systems and Mission Applications," JSASS/AIAA/DGLR 17th International Electric Propulsion Conference, 1984, pp. 600-613.

3. Mirtich, M.J., "Resistojet Propulsion for Large Spacecraft Systems," AIAA Paper 82-1948, Nov. 1982. (NASA TM-83489).

4. Guman, W.J. and Nathanson, D.M., "Pulsed Plasma Microthruster Propulsion System for Synchronous Orbit Satellite," Journal of Spacecraft and Rockets, Vol. 7, No. 4, April 1970, pp. 409-415.

5. Guman, W.J. and Kowal, S.J., "Pulsed Plasma Propulsion System for the TIP-II Satellite." 1975 JANNAF Propulsion Conference, Vol. 1, CPIA-PUBL-266-VOL-1, T.M. Gilliland, ed., Chemical Propulsion Information Agency, 1975, pp. 443-458.

6. Hirata M. and Murakami, H., "Electromagnetic Noise Measurement Study of Pulsed Plasma Engine," AIAA Paper 81-0722, Apr. 1981.

7. An, S., Wu, H., Feng, X., and Liu, W., "Space Flight Test of MDT-2A," AIAA Paper 82-1874, Nov. 1982.

8. Brill, Y., Eisner, A., and Osborn, L., "The Flight Application of a Pulsed Plasma Microthruster; The NOVA Satellite," AIAA Paper 82-1956, Nov. 1982.

9. Zhurin, V.V., Porotnikov, A.A., and Borisova, S.B., "Recent Developments in Electric Propulsion in the USSR," AIAA Paper 83-1397, June 1983.

10. Kuriki, K., Nakamaru, K., and Morimoto, S., "MPD Thruster Test on Engineering Test Satellite," AIAA Paper 79-2071, Oct. 1979.

11. Kuriki, K., Kawashima, N., Sasaki, S., Yanagisawa, M., and Obayashi, T., "Space Experiment with Particle Accelerators (SEPAC) Performed in Spacelab First," AIAA Paper 85-1996, Sept., 1985.

12. Cybulski, R.J., Shellhammer, D.M., Lovell, R.R., Domino, E.J., and Kotnik, J.T., "Results From SERT I Ion Rocket Flight Test," NASA TN D-2718, 1965.

13. Sellen, J.M., Jr., "Interaction of Spacecraft Science and Engineering Subsystems with Electric Propulsion Systems," AIAA Paper No. 69-1106, Oct. 1969.

14. Lazar, J., "Review of the NASA Program in Electric Propulsion," AIAA Paper 69-248, Mar. 1969.

15. Olsen, R. C., "Experiments in Charge Control at Geosynchronous Orbit-ATS-5 and ATS-6," Journal of Spacecraft and Rockets, Vol. 22, No. 3, May-June 1985, pp. 254-264.

16. Rulis, R.J., "SERT-II: Design Requirements for Integrating Electric Propulsion Into a Spacecraft," Journal of Spacecraft and Rockets, Vol. 8, No. 3, Mar. 1971, pp. 209-213.

17. Worlock, R.M., James, E.L., Hunter, R.E., and Bartlett, R.O., "The Cesium Bombardment Engine North-South Stationkeeping Experiment on ATS-6," AIAA Paper 75-363, Mar. 1975.

18. Kitamura, S., "Development of the Engineering Test Satellite-3 (ETS-3) Ion Engine System," NASA TM-77538, 1984.

19. Masek, T.D. and Cohen, H.A., "Satellite Positive-Ion-Beam System," Journal of Spacecraft and Rockets, Vol. 15, No. 1, Jan.-Feb. 1978, pp. 27-33.

20. Feconda, R.T. and Weizman, J.I., "Satellite Reaction Control Subsystems with Augmented Catalytic Thrusters," AIAA Paper 84-1235, June 1984.

21. Dressler, G.A., Morningstar, R.E., Sackheim, R.L., Fritz, D.E., and Kelso, R., "Flight Qualification of the Augmented Electrothermal Hydrazine Thruster," AIAA Paper 81-1410, July 1981.

22. Carruth, M.R., Jr., ed., "Experimental and Analytical Evaluation of Ion Thruster/Spacecraft Interactions," JPL Publication 80-92, Jan. 1981. (NASA CR-163975).

23. Byers, D.C., "Electron Bombardment Thruster Field and Particle Interfaces," Journal of Spacecraft and Rockets, Vol. 16, No. 5, Sept.-Oct. 1979, pp. 289-301.

24. Deininger, W.D., "Electric Propulsion Produced Environments and Possible Interactions with the SP-100 Power System," AIAA Paper 85-2046, Sept. 1985.

25. Curran, F.M. and Haag, T.W., "Arcjet Component Conditions Through a Multistart Test," AIAA Paper 87-1060, May 1987.

26. Wales, R.O., ed., "ATS-6 Final Engineering Performance Report, Vol. 2 - Orbit and Attitude Controls," NASA RP-1080, 1981.

27. White, D.R.J. and Hill, J.S., A Handbook Series on Electromagnetic Interference and Compatibility, Vols. 1-5, Don White Consultants, Germantown, MD, 1974.

28. Murphy, G.B., Private Communication, University of Iowa, Department of Physics and Astronomy, May 1986.

29. Sicotte, R.L., "RFI Measurements at UHF on a Pulsed Plasma Thruster," Journal of Spacecraft and Rockets, Vol. 7, No. 3., Mar. 1970, pp. 337-338.

30. Azuma, H., Nakamura, Y., Kudo, I., Kubo, M., and Sasaki, T., "EMI Test Chamber for Ion Engine Mounted on Satellite," AIAA Paper 81-0725, Apr. 1981.

31. Ijichi, K., Yoshida, T., Kudo, I., and Kuriki, K., "Radiated Emission Noise of the Plasma," AIAA Paper 82-1883, Nov. 1982.

32. Ackerknecht, W.E. and Stanton, P.H., "The Effects of an Ion-Thruster Exhaust Plume on S-Band Carrier Transmission," JPL-TM-33-754, 1976. (AIAA Paper 76-1072).

33. Guman, W.J. and Williams, T.E., "Pulsed Plasma Microthruster for Synchronous Meteorological Satellite (SMS)," AIAA Paper 73-1066, Oct. 1973.

34. Sicotte, R.L., "RFI Measurements at UHF on a Pulsed Plasma Thruster," Journal of Spacecraft and Rockets, Vol. 7, No. 3, Mar. 1970, pp. 337-338.

35. An, S.M., Wu, H.J., Feng, X.Z., and Liu, W.X., "Space Flight Test of Electric Thruster System MDT-2A," Journal of Spacecraft and Rockets, Vol. 21, No. 6, Nov.-Dec. 1984, pp. 593-594.

36. Dolbec, R.E., "RFI Measurements on a LES-7 Prototype Pulsed Plasma Thruster," Journal of Spacecraft and Rockets, Vol. 7, No. 7, July 1970, pp. 889-890.

37. Thomassen, K.I., "Radiation from Pulsed Electric Thrusters," Journal of Spacecraft and Rockets, Vol. 10, No. 10, Oct. 1973, pp. 679-680.

38. Vondra, R.J., "The MIT Lincoln Laboratory Pulsed Plasma Thruster," AIAA Paper 76-998, Nov. 1976.

39. Hirata, M. and Murakami, H., "Development of Pulsed Plasma Engine for Engineering Test Satellite IV," IAF Paper 80-F-257, Sept. 1980.

40. Hirata, M. and Murakami, H., "Development of a Pulsed Plasma Engine," JSASS/AIAA/DGLR 17th International Electric Propulsion Conference, 1984, pp. 338-344.

41. Begun, M. and Guman, W.J., "Pulsed Plasma Radio Frequency Interference Studies," AFRPL-TR-77-85, Sept. 1977. (Avail. NTIS, AD-A049581).

42. Artsimovich, L.A., et al., "Development of a Steady State Plasma Engine (SPE) and its Use on a 'Meteor' Artificial Satellite," Cosmic Research, Vol. 12, No. 3, May-June 1974, pp. 414-429.

43. Morozov, A.I. and Shubin, A.P., "Space Electro-Jet Engines," NASA TT F-16542, 1975.

44. Kuriki, K., Morimoto, S., and Nakamaru, K., "Flight Performance Test of MFD Thruster System," AIAA Paper 81-0664, Apr. 1981.

45. Obayoshi, T., et al, "SEPAC Particle Accelerator Test in NASDA Space Chamber," Institute of Space and Aeronautical Science, University of Tokyo, Report No. 562, Vol. 43, No. 8, Aug. 1978, pp. 141-174.

46. Ijichi, K., Harada, H., and Kuriki, K., "MPD Arcjet System for Space Experiment with Particle Accelerator (SEPAC)," AIAA Paper 79-2072. Oct. 1979.

47. Kerslake, W.R., Personal Communication, NASA Lewis Research Center, Cleveland, OH, 1987.

48. Hall, D.F. and Lyon, W.C., "Low Thrust Propulsion System Effects on Communication Satellites," AIAA Paper 72-519, Apr. 1972.

49. Kerslake, W.R. and Domitz, S., "Neutralization Tests on the SERT-II Spacecraft," AIAA Paper 79-2064, Oct. 1979. (NASA TM-79271).

50. Kerslake, W.R., "SERT-II Thrusters - Still Ticking After Eleven Years," AIAA Paper 81-1539, July 1981. (NASA TM-81774).

51. Kerslake, W.R. and Ignaczak, L.R., "SERT II 1979-1981 Tests: Plasma Thrust and Neutralizer Measurements," Journal of Spacecraft and Rockets, Vol. 19, No. 3, May-June 1982, pp. 236-240.

52. Olsen, R.C., "Modification of Spacecraft Potentials by Plasma Emission," Journal of Spacecraft and Rockets, Vol. 18, No. 5, Sept.-Oct. 1981, pp. 462-469.

53. Kudo, I., Machida, K., Toda, Y., and Murakami, H., "Electromagnetic Noise from an Ion Engine System," Journal of Spacecraft and Rockets, Vol. 20, No. 1, Jan.-Feb. 1983, pp. 84-88.

54. Tabata, J., "Electric Propulsion Activities in Japan," JSASS/AIAA/DGLR 17th International Electric Propulsion Conference, 1984, pp. 28-37.

55. Stevens, J.R. and Vampola, A.L., "Description of the Space Test Program P78-2 Spacecraft and Payloads," SAMSO-TR-78-24, Space and Missile Systems Organization, Los Angeles, CA, Oct. 1978. (Avail. NTIS, AD-A061324).

56. Huber, W.B., "Design, Fabrication and Testing of a Satellite Electron Beam System," AFGL-TR-80-0161, May 1980. (Avail. NTIS, AD-A095362).

57. Koons, H.C., "Aspect Dependence and Frequency Spectrum of Electrical Discharges on the P78-2 (SCATHA) Satellite," Spacecraft Charging Technology 1980, NASA CP-2182, 1980, pp. 478-492.

58. Krulle, G., Zayfang, E., Bassner, H., Birner, W., and Muller, H., "RIT-10 System Electric Design, Characteristics, and Compatibility," AIAA Paper 81-0694. Apr. 1981.

59. Whittlesey, A.C. and Macie, T.W., "Electromagnetic Interference of Power Conditioners for Solar Electric Propulsion," JPL-TM-33-623, July 1973. (NASA CR-133229).

60. Jackson, F.A., Stansel, J.C., Jortner, D., and Hagelberg, C.F., "An Operational Electrothermal Propulsion System for Spacecraft Reaction Control," AIAA Paper 66-213, Mar. 1966.

61. Greco, R.V., Bliss, J.R., Murch, C.K., Clark, K.E., and Kelly, A.J., "Resistojet and Plasma Propulsion System Technology," AIAA Paper 72-1124, Nov. 1972.

62. Shaw, R., Pugmire, T.K., and Callens, R.A., "Ammonia Resistojet Station Keeping Subsystem Aboard Applications Technology Satellite (ATS) IV," AIAA Paper 69-296, Mar. 1979.

63. McKevitt, F.X., "Design and Development Approach for the Augmented Catalytic Thruster (ACT)," AIAA Paper 83-1255, June 1983.

64. Burch, J.L., "Space Plasma Physics Results from Spacelab 1," Journal of Spacecraft and Rockets, Vol. 23, No. 3, May-June, 1986, pp. 331-335.

65. Byers, D.C. and Wasel, R.A., "The NASA Electric Propulsion Program," AIAA Paper 87-1098, May 1987. (NASA TM-89856).

66. Knowles, S.C., Smith, W.W., and Chun, S.I., "Low Power Hydrazine Arcjets: A System Description for Near-Term Application," 1986 JANNAF Propulsion Meeting, Vol. 1, CPIA-PUBL-455-VOL-1, K.L. Strange and D.S. Eggleston, eds., Chemical Propulsion Information Agency, Laurel, MD, 1986, pp. 399-408.

67. Todd, J.P., "30 kW Arc-Jet Thruster Research," APL-TDR-64-58, 1964. (Avail. NTIS, AD-601534).

68. "Development of a Thermal Arc Engine," ASD-TDR-62-749, Plasmadyne Corporation, Santa Ana, CA, July 1962.

69. "Plume Electromagnetic Interactions," JANNAF Handbook: Rocket Exhaust Plume Technology, CPIA-PUBL-263-CH-4, Apr. 1977, Chapter 4. (Avail. NTIS, AD-B021093).

70. Stanton, P., "Ion Plume/S-Band Carrier Interaction Study," Experimental and Analytical Evaluation of Ion Thruster/Spacecraft Interactions, M.R. Carruth, Jr., ed., JPL-PUB-80-92, Jan. 1981, pp. 191-215. (NASA CR-163975).

TABLE I. - ELECTRIC PROPULSION FLIGHTS SURVEYED

Propulsion system	Spacecraft	Year launched	Thrust	Average power per thrust, W (Number of thrusters)	Propellant	Application	Reference
Ablative pulsed plasma	LES-6, USA	1968	18 μN	3(4)	Teflon	E/W S-K[a]	4
	TIP II, USA	1976	400 μN	30(2)	Teflon	Drag makeup	5
	TIP III, USA	1981	400 μN	30(2)	Teflon	Drag makeup	5
	ETS IV, Japan	1981	90 μN	20(4)	Teflon	Experiment	6
	MDT 2A, China	[d]1981	------	5(2)	Teflon	Experiment	7
	Nova 1, USA	1981	290 μN	30(2)	Teflon	Drag makeup	8
Hall current thruster	Meteor-18, USSR	1974	20 mN	390(2)	Xenon	Orbit correction	9
MPD thruster	MS-T4, Japan	1980	240 μN	15	Ammonia	Experiment	10
	Spacelab-1, USA/Japan	1983	---------	[c]500(1)	Argon	Experiment	11
Ion propulsion	SERT I, USA	[d]1964	20 mN	1600(1)	Mercury	Experiment	12
	SNAP 10A, USA	1965	9 mN	-------	Cesium	Experiment	13
	ATS 4, USA	1968	20-90 mN	-------	Cesium	S-K	14
	ATS 5, USA	1969	---------	-------	Cesium	S-K	15
	SERT II, USA	1970	28 mN	1000(2)	Mercury	Experiment	16
	ATS 6, USA	1974	4 mN	150(2)	Cesium	Experiment	17
	ETS III, Japan	1982	2 mN	110(2)	Mercury	Experiment	18
Ion source	Scatha P78-2, USA	1979	52 μN	60(1)	Xenon	Experiment	19
Resistojet	20 flight tests, USA	1965-1971	18-360 μN	3-100	Nitrogen Ammonia Hydrazine	Experiment ACS Orbit adjust	3
	6 GTE and ASE, RCA S/C	1980-1985	220 mN	[e]500(2)	Hydrazine	N/S S-K[b]	20
	INTELSAT V	1980-1985	220 mN	[e]500(2)	Hydrazine	N/S S-K	21

[a]East/west stationkeeping.
[b]North/south stationkeeping.
[c]Battery charger.
[d]Ballistic flight.
[e]Plus two redundant thrusters.

TABLE II. - TYPICAL THRUSTER CHARACTERISTICS

Propulsion system	S/C, year	Average thrust, propellant	Repitition rate, Hz	Specific impulses, s	Average/ (peak power), W	Total[a] impulse capability, Ns
Pulsed plasma (ref. 8)	Nova 1, 1981	290 μN, Teflon	1	----	30 (-)	2×10^3
Hall current thruster (ref. 9)	Meteor, 1974	20 mN, Xe	cw	1000	390 (390)	> 4×10^4
MPD arcjet (ref. 11)	Spacelab 1, 1983	-------, Ar	0.067	----	200 (2×10^6)	-------
Ion thruster (ref. 17)	ATS 6, 1974	4.5 mN, Cs	cw	2500	150 (150)	7×10^4
Ion source (ref. 19)	Scatha, 1979	52 μN, Xe	cw	----	60 (60)	> 60 Ns
Resistojet (ref. 20)	Spacenet, 1984	220 mN, N_2H_4	cw	290	500 (500)	2×10^5
rcjet (ref. 25)	----------	130 mN, N_2H_4	cw	450	1000 (1000)	2×10^5 (projected)

[a]Total impulse of all thrusters on S/C.

TABLE III. - RADIATED EMISSION SOURCES FROM ELECTRIC PROPULSION THRUSTERS

Thruster	Potential radiated RFI/EMI source(s)
Pulsed plasma thruster	Ignitor discharge Main discharge Exhaust plasma Wire harness
Ion thruster	Discharge chamber plasma Neutralizer plasma bridge Beam plasma Electric breakdown at ion optics Wire harness
MPD thruster	Main discharge Exhaust plasma Wire harness
Arcjet thruster	Arc discharge Exhaust plasma Wire harness
Resistojet	Heater or wire harness
Power processor	Plasma ignitor circuits Power switches Cables and wire harness

TABLE IV. - EMISSION AND SUSCEPTIBILITY REQUIREMENTS FOR ELECTRICAL, ELECTRONIC, OR ELECTRO-MECHANICAL EQUIPMENT AS SET-FORTH BY MIL-STD-461B; PART 1; APRIL 1, 1980

Requirement	Description
CE01	Conducted emissions, power, and interconnecting leads, low frequency (up to 15 kHz)
CE03	Conducted emissions, power, and interconnecting leads, 0.015 to 50 MHz
CE06	Conducted emissions, antenna terminals 10 kHz to 26 GHz
CE07	Conducted emissions, power leads, spikes, time domain
CS01	Conducted susceptibility, power leads, 30 Hz to 50 kHz
CS02	Conducted susceptibility, power leads, 0.05 to 400 MHz
CS03	Intermodulation, 15 kHz to 10 GHz
CS04	Rejection of undersired signals, 30 Hz to 20 GHz
CS05	Cross-modulation, 30 Hz to 20 GHz
CS06	Conducted susceptibility, spikes, power leads
CS07	Conducted susceptibility, squelch circuits
CS09	Conducted susceptibility, structure (common mode) current, 60 Hz to 100 kHz
RE01	Radiated emissions, magnetic field, 0.03 to 50 kHz
RE02	Radiated emissions, electric field, 14 kHz to 10 GHz
RE03	Radiated emissions, spurious and harmonics, radiated technique
RE04	Radiated emissions, magnetic field, 0.02 to 50 kHz
RS01	Radiated susceptibility, magnetic field, 0.03 to 50 kHz
RS02	Radiated susceptibility, magnetic induction field, spikes and power frequencies
RS03	Radiated susceptibility, electric field, 14 kHz to 40 GHz
UM03	Radiated emissions, tactical and special purpose vehicles and engine-driven equipment
UM04	Conducted emissions and radiated emissions and susceptibility engine generators and associated components UPS and MEP equipments
UM05	Conducted and radiated emissions, commercial electrical and electromechanical equipments

TABLE V. - RADIATED EMI RESULTS FROM STUDIES, GROUND TEST, AND MISSION

Type of propulsion, spacecraft, year launched, S/C operations	Ground test specifications and equipment	Ground test results	Flight results	Reference
Pulsed plasma, 4 thrusters, Ft ≈ 320 Ns, LES-6 S/C, 1968, E/W S-K	• Limit RFI to −110 dBm/Hz UHF • Signals > 50 μs above −169 dBm/Hz would cause interference • Used metal vacuum tank with absorbing material	• RFI time duration above −169 dBm/Hz was 15 μs • RFI was not detrimental to S/C or experiments	• One PPT operated 8900 hr during 5 years • No interference with TM or COMM	4,33,34
Pulsed plasma, 2 thrusters, MDT-2A, 1981, Ballistic test, 37 min	• S/C integration tests conducted	• PPT had no obvious effects on S/C subsystems	• PPT did not disturb TM or COMM	7,35
Pulsed plasma, 2 thrusters, NOVA-1, 1981, drag cancellation			• More than 1×10^6 thruster pulses during 5 months operation • No EMI effects on S/C subsystems	8
Pulsed plasma, 2 thrusters, Ft ≈ 2200 Ns, TIP II, 1976, drag cancellation	• RFI measurments made at specific frequencies and positions • Atmospheric tests	• RFI generated at discharge initiation 100 to 300 μs pulses Exceeded MIL-STD-461 cw specifications		5
Pulsed plasma, ground tests of LES-7 system, 1970	• Broadband (100 MHz) measurements at 8 GHz • Used metal vacuum facility with broadband absorber	• No absolute measurements • Peak noise ± 25° from PPT boresight	ORIGINAL PAGE IS OF POOR QUALITY	36
Pulsed plasma, ground tests of LES 8/9 system, 1973, 1976	Order of magnitude measurements 0.2 to 20 GHz • Used vacuum tank with absorbing walls • Broadband antenna 2 to 3 m from PPT outside vacuum tank	• Radiated energy density: 6 to 60 μw/MHz • Sufficient to saturate antenna receiver near PPT • RFI of 0.5 to 1 μs duration		37
	Spacecraft integration tests	• IR sensor affected by RFI • Logic circuits affected • PPT replaced by cold gas ammonia thrusters		38
Pulsed plasma, 4 thrusters, ETS IV, 1981	• MIL-STD-461 • PPT in glass vacuum belljar • Power supplies external to belljar	• Average noise level exceeded RE02 only at 120 MHz • RE02 satisfied at uplink frequency of 148 MHz and COMM frequency of 1 to 3 GHz	• 31 tests over 3 year flight period • 4×10^5 firings during 100 hr operational period • No EMI effects on TM, command signals, or S/C instruments • No changes on AGC levels of VHF and S-band	6,39,40
Pulsed plasma, AFAL study of 4 mN PPT, 1977	• MIL-STD-461 • MIL-STD-1541 • Thruster simulator in screen room • RFI measured from 5 KHz to 10 GHz	• No absolute measurements • Shielding of PPT system was effective if covers are well fitted or gaskets are used		4

TABLE V. - Continued

Type of propulsion, spacecraft, year launched, S/C operations	Ground test specifications and equipment	Ground test results	Flight results	Reference
Hall current plasma thrusters, Meteor-18, 1974, System interactions experiment			• Operating time of thrusters > 600 hr with 273 activations • Thruster RF emissions did not disrupt COMM • Close location of thruster and antenna caused some RFI	9,42,43
MPD thruster, Ft ≈ 170 Ns, MS-T4 S/C, 1980, experiment			• Thruster operated 5 hr, 39 min at low duty cycle • Pulse observed on AGC signals of 400 MHz antenna system • AGC signals from S-band (2.28 GHz) not affected by MPD thruster	10,44
MPD arcjet STS-9, Spacelab-1, SEPAC Experiment, 1983	Used metal vacuum chamber; No special precautions for EMI suppression • MIL-STD-461 • Spacelab standard for antennae locations	• EMI levels exceeded Spacelab and RE02 specifications • EMI lasted as long as the MPD discharge period • EMI was thought to originate mainly from the arcjet plasma • No estimate of vacuum chamber reflections	• MPD arcjet operated during 13 time segments with a total of 260 firings • Operations were done with a plasma diagnostics package and TV monitor • MPD arcjet induced neither hazard nor EMI with Spacelab or other payloads	11,30,31, 46,49,53, 45
	Integration test • MIL-STD-461 • MSFC-SPEC-521 • Metal vacuum facility used with Spacelab pallet and arcjet inside	• No malfunction of flight hardware due to EMI • RF emissions above MSFC-SPEC-521 at < 100 MHz • Specs satisfied for 100 MHz to 10 GHz • No S-band link RFI • Slight VHF RFI click noise on voice COMM		
Ion thruster, SERI-I, 1964, Ballistic flight experiment			• Thruster operation and HV faults produced no anomalies in TM, command or control systems • Thruster operated 31 min • Ion beam apparently did not interfere with commands to or transmission from S/C	12
Ion thruster, SNAP-10A, 1965, experiment			• Thruster arcing produced severe effects on TM, control system and horizon sensors • Probable cause: conducted EMI at < 1 MHz • Reinforced need for system integration testing	13
Ion thruster, 2 thrusters, SERT II, 1970, experiment		• Prototype S/C thermal vacuum tested > 3200 hr; Operational thrusters on > 2400 hr	• Up/down link data (136 to 149 MHz) unaffected by thruster operation	16 47 to 51

TABLE V. - Continued

Type of propulsion, spacecraft, year launched, S/C operations	Ground test specifications and equipment	Ground test results	Flight results	Reference
		• Established electrical compatibility with total S/C • Evaluated thruster arcing problems	• No effects on S/C command or control • Thruster operation not detected on 1.7 and 2.1 GHz RFI sensors • No apparent effect on COMM with wideband RF antenna viewing Earth through the ion beam • Operated over 11 years • Over 6900 hr of thruster operation with ion beam extraction	
Ion thruster, 2 thrusters, ATS-6, 1974, experiment	• S/C integration test in anechoic chamber • S/C fully assembled and deployed	• System integration tests conducted with thruster simulator and flight PPU; No EMI detected on S/C subsystems during normal operation or HV breakdowns	• No RFI on command link (150 MHz) or TV relay (4 GHz) • Thruster operated 92 hr. No evidence of RFI on TM, command systems, or COMM (153 MHz, 2.25 GHz, 6.15 GHz).	15,17, 26,52
Ion thruster, ETS-III, 1982, experiment	• MIL-STD-461 • Thruster in glass vacuum belljar • PPV at atmosphere on ground plane for component test	Component test • Broadband noise 3 to 100 MHz exceeded RE02 specifications • RE04 exceeded in 20 to 70 KHz range • Thruster, and S/C compatibility confirmed using EMI simulator Integration test • AGC level of VHF command receiver increased to worst case reception level • No fluctuations of AGC level for S-band	• No indicated increase in AGC level of VHF signals due to thruster operations • One thruster was 182 hr and 111 cycles • S/C operations and parameters within allowable ranges during ion thruster operation implying thruster compatibility with S/C subsystems.	18,30, 53,54
Ion Source, SCATHA, P78-2, 1979, experiment	• MIL-STD-461A conducted emissions and susceptability only • S/C integration test conducted with ion source in vacuum chamber	• S/C integration test for 2 hr with S/C power experiments, TM, and command systems functioning; no problems with RFI or experiment performance	• In March/April 1979 there were 392 ion source pulse operations • Ion source operated at low duty cycle for more than 1 year; no indication of S/C anomalies while operating.	19 55 to 57
Ion thruster, RIT-10 mN Hg thruster, ground test of TV-SAT system	• MIL-STD-461 • Used glass vacuum chamber	• RF emissions below SPEC from 10 MHz to 10 GHz • Less than 10 MHz some broadband noise exceeds RE02. • No RFI observed. Infrequency ranges of COMM or TM during steady state or fault clearing operations.		58

TABLE V. - Continued

Type of propulsion, spacecraft, year launched, S/C operations	Ground test specifications and equipment	Ground test results	Flight results	Reference
Ion thruster, 30 cm diameter Mercury thruster, SEP development, ground tests	• MIL-STD-461, VO-75, MJS-77			
	Atmospheric tests of PPU and solar array simular using resistive loads	• MJS-77 RFI SPEC exceeded by 50 db over 0.1 to 10 MHz • RF emissions 10 to 20 dB avove normal during HV breakdown • Primary cause: poor shielding of 5 KHz power harness		59
	Metal vacuum chamber used	• RF emission > 40 dB μV/m above ambient from 5 KHz to 1 MHz • RF emissions independent of beam current (0.75 to 1.75 A) from 10 KHz to 80 MHz • Warm-up mode with only discharge chamber plasma was noisier than beam extraction operation from 500 KHz to 80 MHz		22
Ion thruster 30 cm diameter Mercury thruster, ground tests	• Used steel vacuum tank with some microwave absorbing material • Data obtained on S-band (2.1 to 2.3 GHz) at ion beam currents from 1.0 to 1.7A and discharge plasma only • Measure S-band phase shift attenuation due to communication through ion exhaust beam	• Transmission loss of S-band signal through beam was measured as high as -1.3 dB and measured phase shift was between 20 and 45° • Signal reflections in metal tank may be significant		32
	• Used steel vacuum tank with microwave absorber • Same goals as preceding test	• Signal amplitude loss and phase advance increased with ion beam current. • Maximum amplitude loss was 0.38 dB or 8 percent • Phase shifts advanced as high as 67° • No adidtional noise observed on carrier frequency • No carrier frequency offset was observed due to plasma • Metal tank effects may exist		22

TABLE V. - Concluded

Type of propulsion, spacecraft, year launched, S/C operations	Ground test specifications and equipment	Ground test results	Flight results	Reference
Resistojet, early flight experiments, ACS and orbit adjust (two examples)			First operational resistojet used for orbit adjustment on Vela S/C, 1965, N_2 propellant; no failures indicated	3,60,61
			ATS-IV NH_3 resistojet system operated for over 800 hr before reentry; TM data indicated successful operation	14,62
Resistojet RCA developed S/C GSTAR, Spacenet, ASC and Satcom (4 thrusters per S/C)			First operational S/C maneuver of N_2H_4 resistojets on RCA developed S/C in April 1983; many hours of operation reported on Satcom IR and IIR; no anomalous behavior of thrusters reported	20,63
Resistojet INTELSAT V S/C, TRW thrusters, (4 thrusters per S/C)			First INTELSAT V operational firings of resistojets in January 1981; operation and performance of S/C maneuvers was "entirely nominal for the system and orbital conditions"; no apparent RFI effects on TM; used N_2H_4 propellant	21

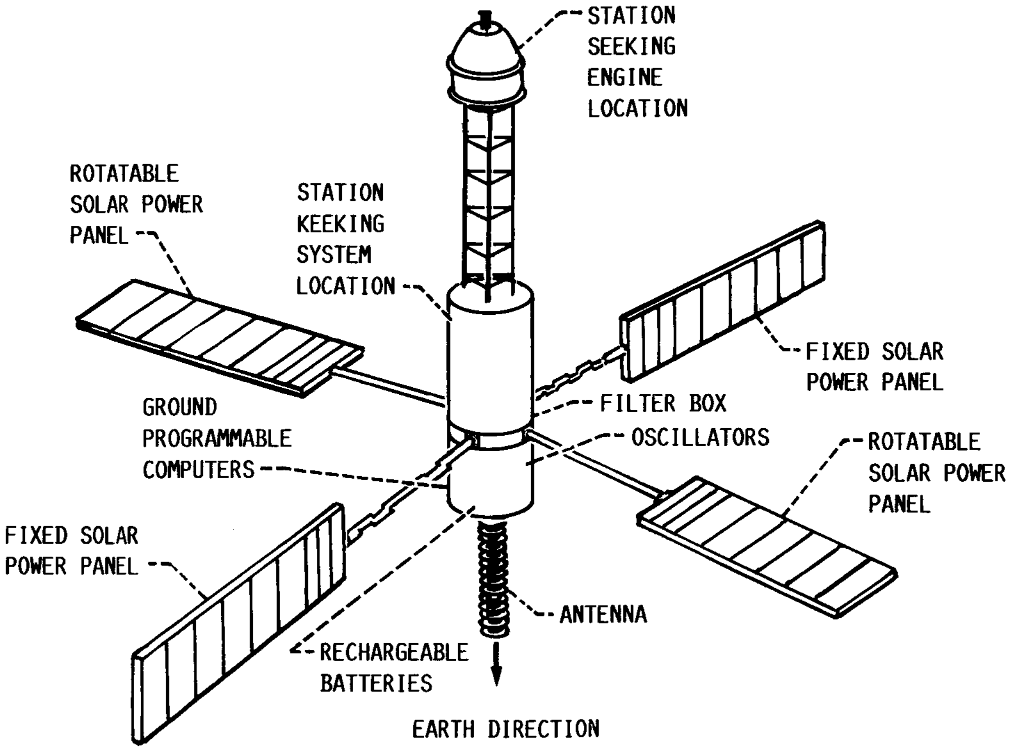


FIGURE 1. - NOVA SPACECRAFT FEATURES.

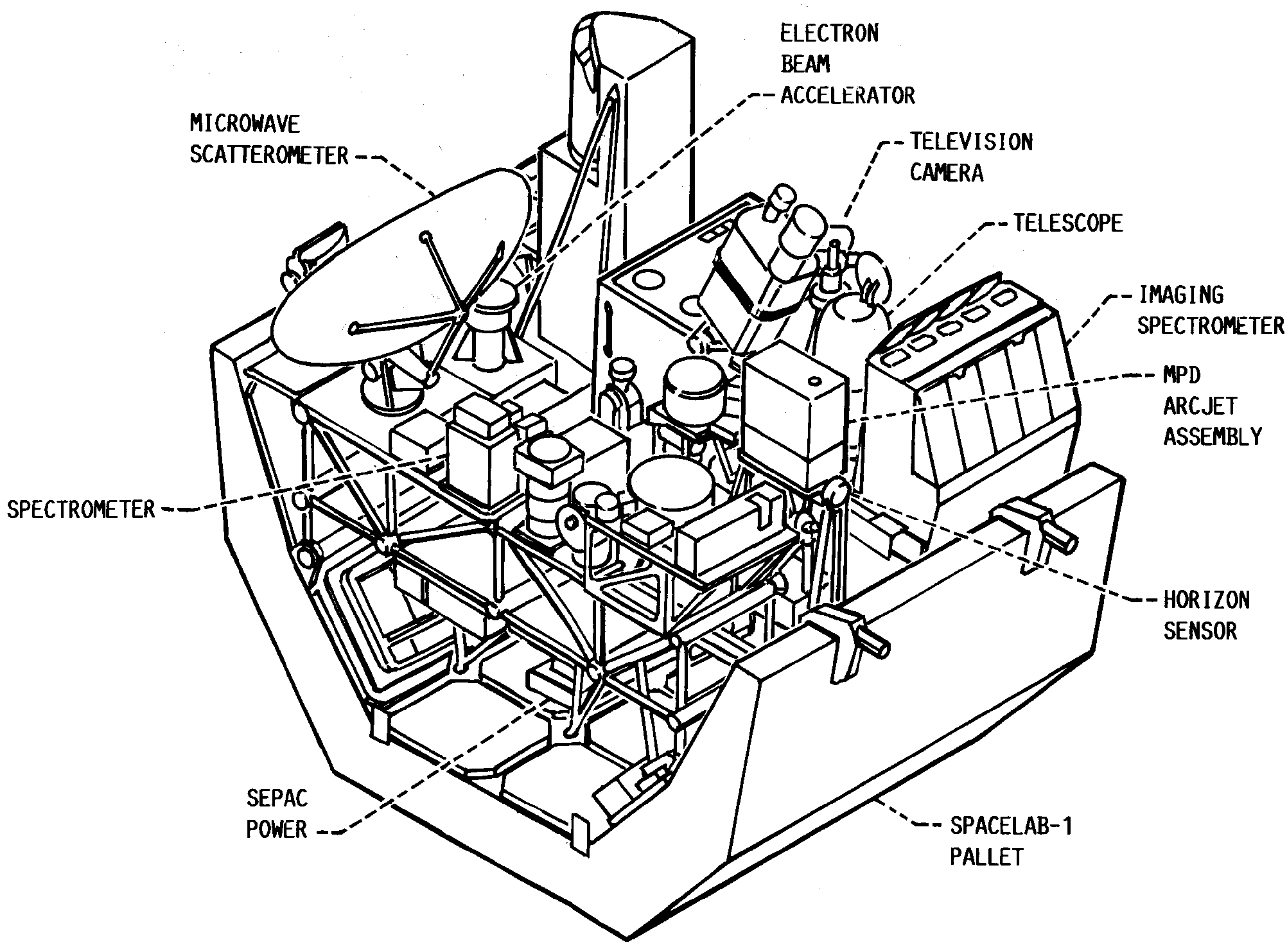


FIGURE 2. - MPD ARCJET ON SPACELAB-1 PALLET.

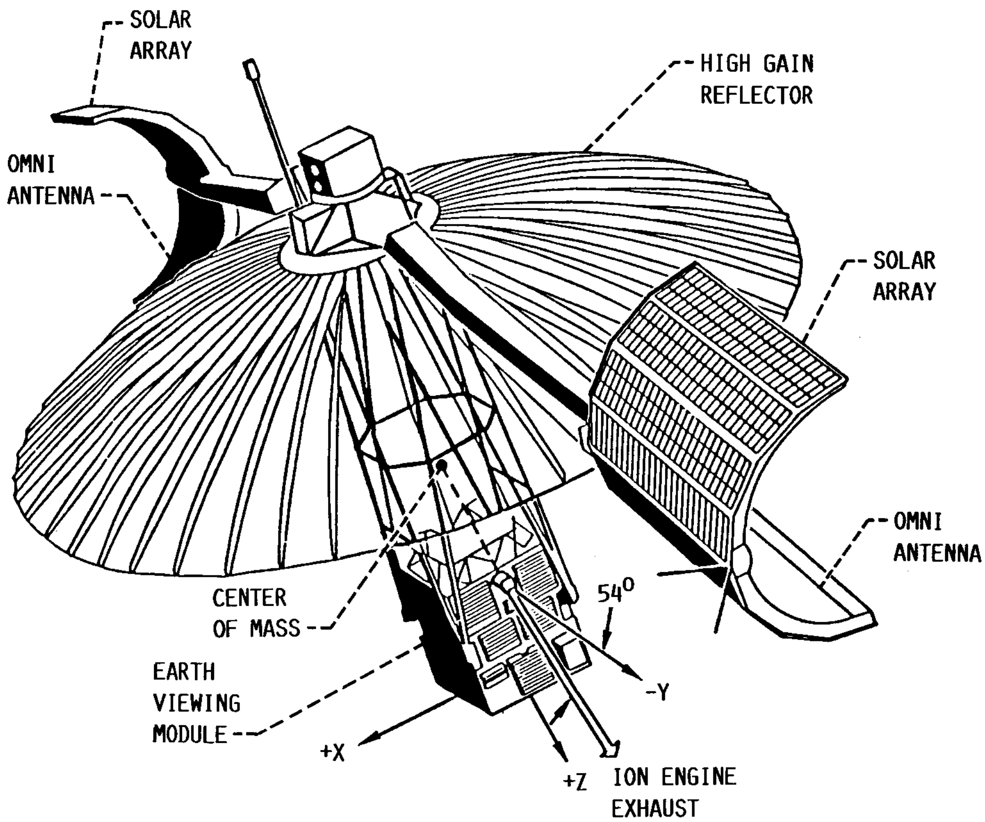


FIGURE 3. - ATS-6 SPACECRAFT AND ION ENGINE CONFIGURATION.

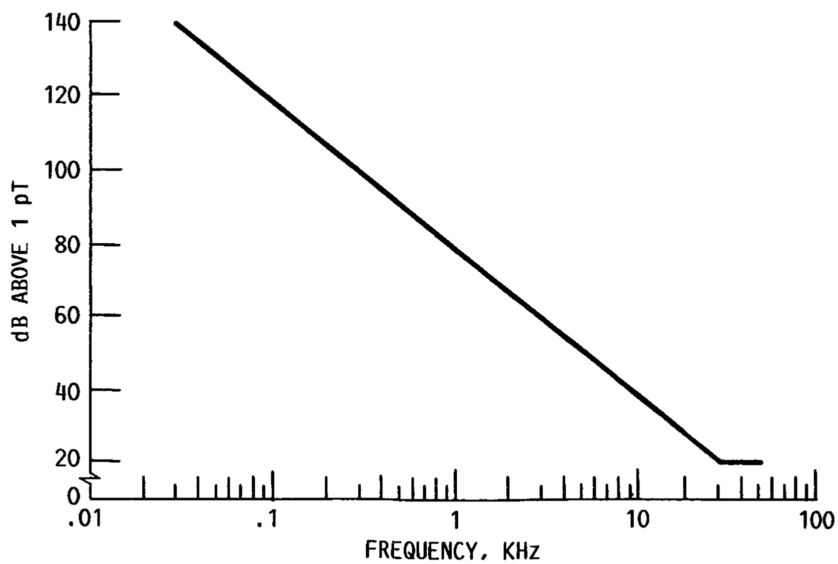
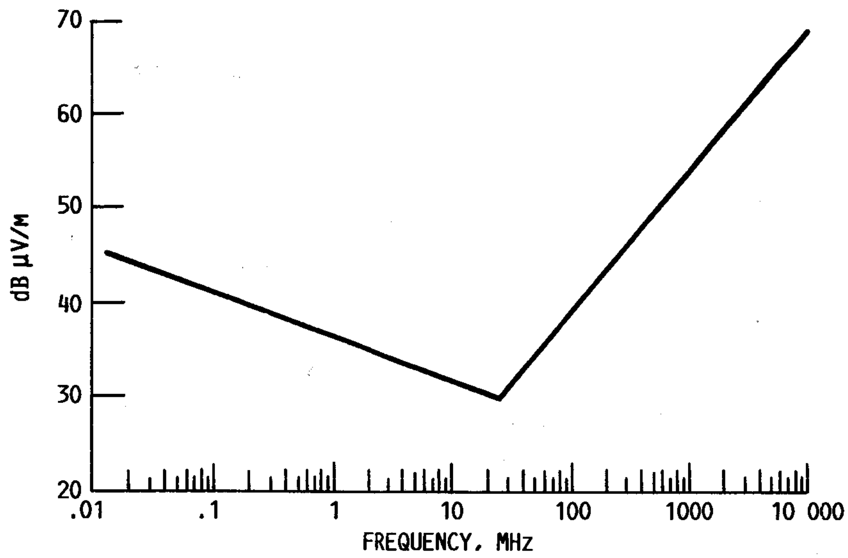
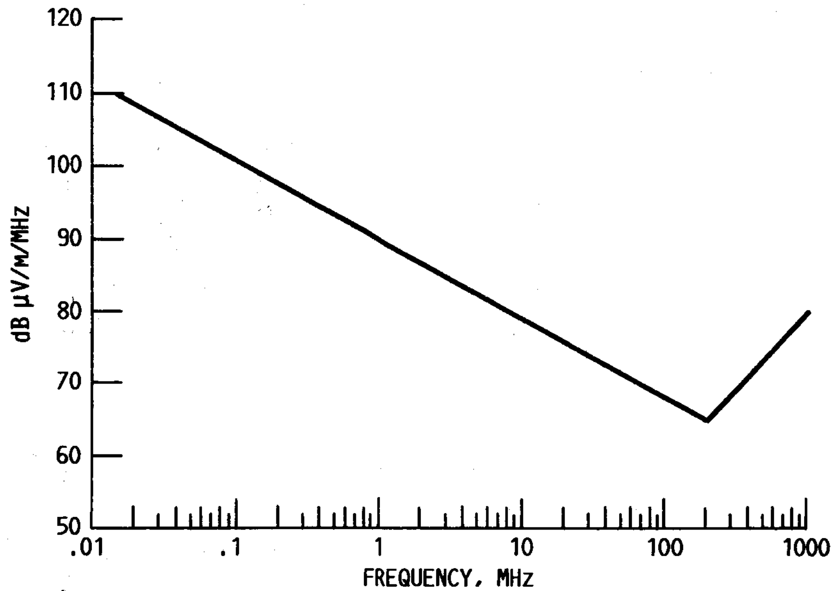


FIGURE 4. - LIMITS FOR MAGNETIC FIELD EMISSIONS (RE01, MIL-STD-461B).



(A) RE02 NARROWBAND EMISSIONS.



(B) RE02 BROADBAND EMISSIONS.

FIGURE 5. - LIMITS FOR ELECTRIC FIELD EMISSIONS.

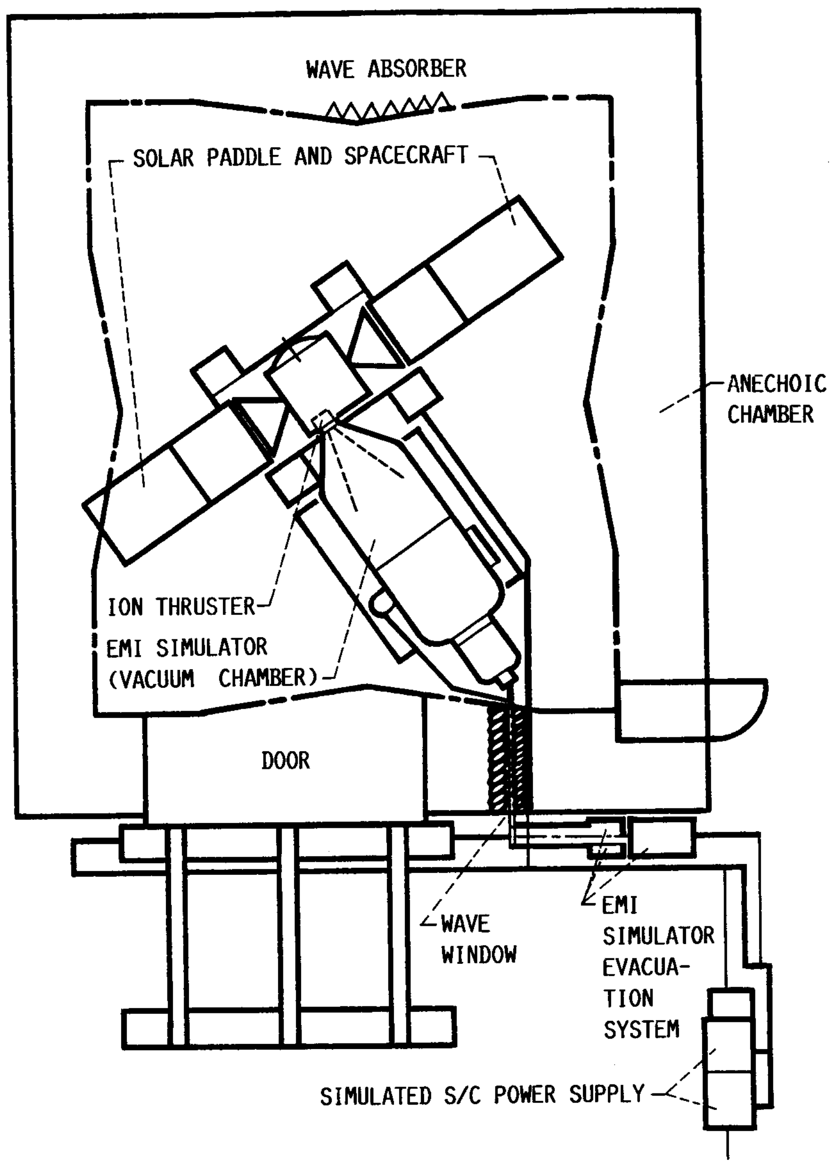


FIGURE 6. - ION THRUSTER INTEGRATION TEST FOR ETS III.

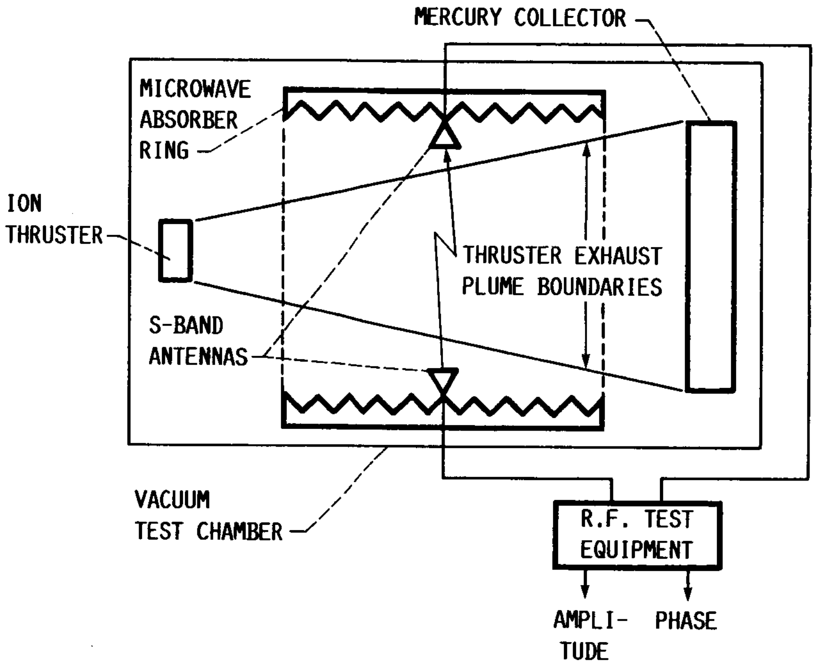


FIGURE 7. - ILLUSTRATION OF EXPERIMENTAL ARRANGEMENT FOR EXHAUST PLASMA EFFECTS ON S-BAND SIGNALS.

NASA
National Aeronautics and Space Administration

Report Documentation Page

1. Report No. NASA TM-100120 AIAA-87-2028	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle Electromagnetic Emission Experiences Using Electric Propulsion Systems - A Survey		5. Report Date
		6. Performing Organization Code 506-42-31
7. Author(s) James S. Sovey, Lynnette M. Zana, and Steven C. Knowles		8. Performing Organization Report No. E-3618
		10. Work Unit No.
9. Performing Organization Name and Address National Aeronautics and Space Administration Lewis Research Center Cleveland, Ohio 44135		11. Contract or Grant No.
		13. Type of Report and Period Covered Technical Memorandum
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D.C. 20546		14. Sponsoring Agency Code

15. Supplementary Notes

Prepared for the 23rd Joint Propulsion Conference cosponsored by the AIAA, SAE, ASME, and ASEE, San Diego, California, June 29 - July 2, 1987. James S. Sovey and Lynnette M. Zana, NASA Lewis Research Center; Steven C. Knowles, Rocket Research Company, Redmond, Washington 98073.

16. Abstract

As electric propulsion systems become ready to integrate with spacecraft systems, the impact of propulsion system radiated emissions are of significant interest. Radiated emissions from electromagnetic, electrostatic, and electrothermal systems have been characterized and results synopsized from the literature describing 21 space flight programs. Electromagnetic radiated emission results from ground tests and flight experiences are presented with particular attention paid to the performance of spacecraft subsystems and payloads during thruster operations. The impacts to transmission of radio frequency signals through plasma plumes are also reviewed.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Electric propulsion Electromagnetic compatibility	Unclassified - unlimited STAR Category 20

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No of pages	22. Price*
Unclassified	Unclassified	36	A03

NASA FORM 1626 OCT 86

*For sale by the National Technical Information Service, Springfield, Virginia 22161