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Contents

8. Active Spacecraft Potential Control System Selection for the Jupiter Orbiter With Probe Mission

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Abstract

The Jupiter orbiter with probe (JOP) spacecraft is briefly described. It is shown that the high flux of energetic plasma electrons and the reduced photoemission rate in the Jovian environment can result in the spacecraft developing a large negative potential. The effects of the electric fields produced by this charging phenomenon are discussed in terms of spacecraft integrity as well as charged particle and fields measurements. The primary area of concern is shown to be the interaction of the electric fields with the measuring devices on the spacecraft. The need for controlling the potential of the spacecraft is identified, and a system capable of active control of the spacecraft potential in the Jupiter environment is proposed. The desirability of using this system to vary the spacecraft potential relative to the ambient plasma potential is also discussed. Various charged particle release devices are identified as potential candidates for use with the spacecraft potential control system. These devices are evaluated and compared on the basis of system mass, power consumption, and system complexity and reliability. The results of this comparison are then used to identify the optimum particle release devices which are capable of actively controlling the spacecraft potential.

I. INTRODUCTION

The scientific objective of the proposed Jupiter orbiter with probe mission is to conduct intensive investigations of Jupiter's atmosphere, satellites, and magnetosphere. If the mission is approved, the spacecraft will be launched by the Space Shuttle/Interim Upper Stage in early 1982 and will arrive at Jupiter some 3 years later. The nominal mission length is 12 months, which provides for multiple encounters with Jupiter and its satellite Ganymede as well as a possible encounter with the satellite Io.

The proposed spacecraft is a dual spin configuration consisting of an orbiter and an atmospheric entry probe. The probe will pass through the Jovian atmosphere on the sunlit side of the planet, and during its 30 min lifetime will transmit atmospheric data to the orbiter which will relay this information back to Earth. The orbiter will continue along its trajectory collecting scientific data in the Jovian magnetosphere and the vicinity of the Jovian satellites.

One of the primary science objectives of the orbiter is to obtain the charged particle distribution functions in the planetary magnetosphere and satellite ionospheres. In order to obtain distribution functions which are representative of the undisturbed plasma, the perturbation in electric potential caused by the presence of the spacecraft must be minimized. Potential variations in the region near the charged particle detectors could result in erroneous information regarding the distribution functions of the charged species. The principal source of error caused by the presence of an electric field in the vicinity of these detectors is a perturbation in the energy and direction of the low-energy particles and a perturbation in the direction of the high-energy particles. Even if the spacecraft potential is close to that of the undisturbed plasma, local potential depressions or barriers[1] may lead to erroneous interpretation of low-energy particles data. For example, the existence of a potential well of magnitude ϕ near these detectors invalidates any electron flux measurements in the energy range from zero to ϕ. The reason for this is, of course, that ambient electrons in this energy range cannot reach the spacecraft, while photoemitted electrons having energies in this range cannot escape the spacecraft. Therefore, in order to obtain accurate information regarding the distribution functions of the charged species, spacecraft design practices should be enforced to (1) maintain the spacecraft at or near the ambient plasma potential and (2) eliminate the presence of differentially charged areas of the spacecraft, thereby eliminating localized electric fields and minimizing potential barriers produced by the release of low-energy secondary and photoemission electrons. The former requirement can be met by providing a return current to space equal to the difference between the incoming electron current and the sum of the incoming ion

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current plus secondary and photoemission currents. The latter requirement can be satisfied by the use of conductive coatings and surfaces on the spacecraft.

Some of the quantitative results to be presented reflect the emission current requirements of the Jupiter orbiter with probe mission. However, the particle release devices and active potential control systems described herein have wide application and their use is not restricted to a specific mission.

2. SPACECRAFT CHARGING IN THE JOVIAN ENVIRONMENT

The equilibrium potential of a passive spacecraft subjected to Jupiter's charged particle flux has been calculated by Goldstein and Divine[2] using a plasma environmental model derived from Pioneer 10 and 11 measurements. The results of their calculations indicate the JOP spacecraft (modeled as a passive conducting sphere) will attain negative floating potentials as high as 10 kV when in eclipse. Spacecraft potentials of this magnitude will surely invalidate proton measurements in the energy range up to 10 kV and will not allow electron measurements in the energy range less than 10 kV. Goldstein and Divine's analysis also indicated that sunlit portions of the spacecraft will discharge, and they may lead to differential charging of adjacent eclipsed and sunlit areas of the spacecraft. These large potentials clearly indicate the need for an active potential control system on the spacecraft as well as the requirement for a conductive spacecraft coating.

A conservative estimate of the net spacecraft collection current during eclipse conditions can be made using the Jupiter environmental model of Goldstein and Divine and neglecting the current contribution due to secondary electron emission. A stationary collector maintained at the same potential as the surrounding plasma produces no disturbing sheath effects and its collection current is the product of the net plasma current density and the collector surface area. The estimated collection area of the JOP spacecraft is about 49 m^2. Using this area and the Jupiter environmental model, the net current collected by the spacecraft when maintained at the same potential as the surrounding plasma can be calculated. The results of these calculations are presented in Figure 1 which indicates the net spacecraft current, as well as the high-energy and Maxwellian electron contributions, as a function of Jupiter's magnetic shell parameter. The net current is the sum of the two electron currents shown in the figure minus the proton current. These results indicate a spacecraft collection current on the order of 0.3 mA in the region near 2-6 R_J (Jupiter radii), and this value drops nearly two orders of magnitude at the nominal mission perijove of 15 R_J. The initial mission perijove for delivery of the atmospheric entry probe is 6 R_J, and according to Figure 1 this corresponds to the region of highest electron collection. These results also indicate the dominant term in the current collection at this location is the Maxwellian electron current.
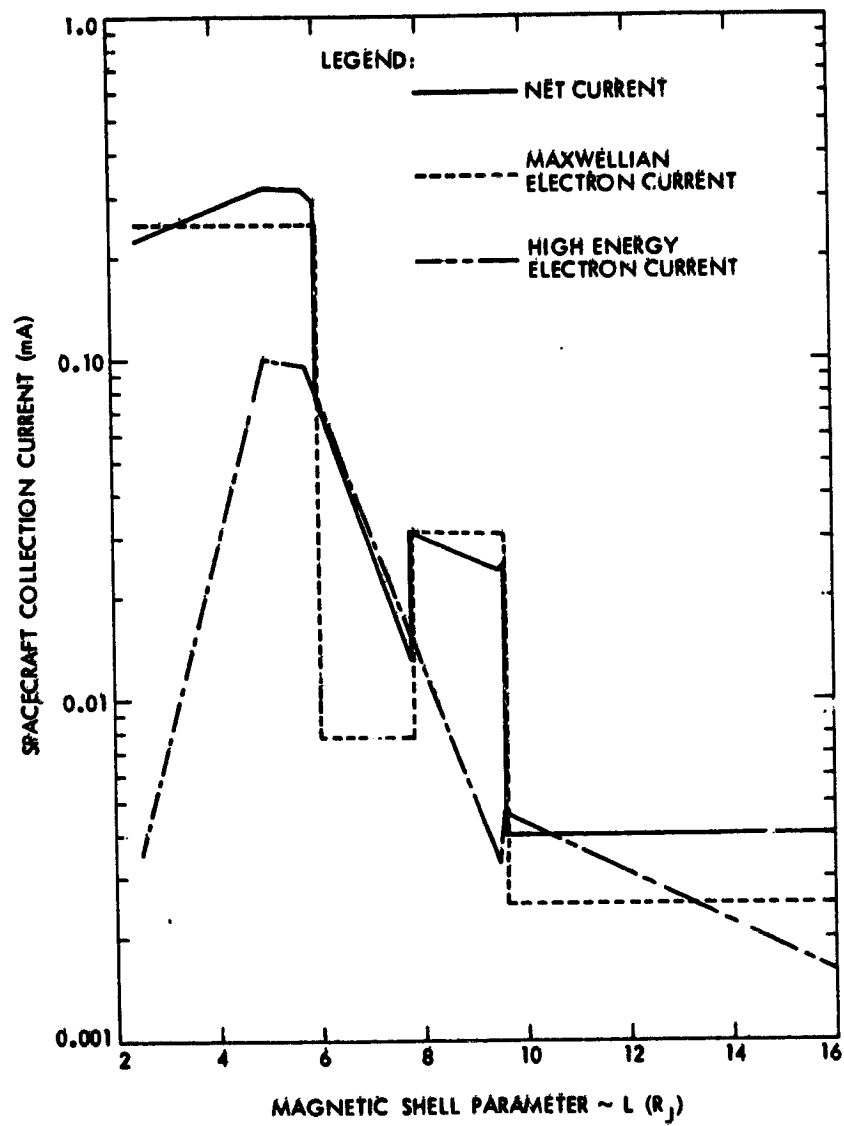
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Figure 1.   Calculated Spacecraft Collection Current as a Function of Jupiter's Magnetic Shell Parameter

Figure 1 can be used to estimate the release current requirements of a charged particle release device since the spacecraft collection current at the plasma potential must be returned to space by the particle release device. This is a conservative estimate since the contribution due to secondary and photoelectron emission has been neglected.

3. ACTIVE SPACECRAFT POTENTIAL CONTROL SYSTEM

The two basic functions of an active spacecraft potential control system are (1) sensing the potential difference between the spacecraft and the surrounding plasma and (2) releasing a current of the proper sign and magnitude to maintain a desired spacecraft potential. In addition, there are two possible control schemes which could be used to couple these functions: (1) closed loop and (2) open loop. The closed loop control system would employ a potential sensing device such as an electric field meter or floating emissive probe to measure the potential difference between the spacecraft and its surroundings. This potential difference can then be maintained at a preselected value by proper biasing of the charged particle release device. The control circuitry which couples the output of the voltage sensor with the biasing power supply serves as the link to form a closed loop control system. An open loop system, on the other hand, would employ a current sensing device to monitor the current through the biasing power supply and, hence, the current-voltage characteristics of the spacecraft. Periodic analysis of this characteristic allows one to determine the bias potential which corresponds to a spacecraft potential equal to the local plasma potential, and one can then adjust the bias potential to give the desired spacecraft potential relative to the ambient plasma. In order to operate in this open loop mode the bias potential must be known relative to a stable reference, and this identifies a general requirement of the charged particle release device: The current-voltage relationship of the ideal charged particle emitter should have infinite slope so that the emission current of the device is essentially independent of its voltage. Two additional requirements of an active potential control system are (1) the particle release device should be mounted so as to minimize any interaction between the released particles and sensitive spacecraft surfaces or science instruments and (2) the thrust produced by the ejected charged particles should not result in any disturbing forces or moments on the spacecraft.

In either, the closed or open loop control scheme one can employ the charged particle release device not only to discharge the spacecraft, but to act as a science instrument as well. As a plasma diagnostic tool[3] the spacecraft potential control system should enable the local Maxwellian electron density and temperature to be determined by analysis of the current-voltage characteristics of the spacecraft.

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4. CHARGED ARTICLE RELEASE DEVICES

The charged particle release devices which are considered suitable for use in an active spacecraft potential control system fall into two categories: (1) electron devices and (2) plasma devices. These categories may be further divided according to the energy of the released or ejected particles. The primary reason for identifying the two major categories is related to the direction in which the devices can drive the spacecraft potential. For instance, a negatively charged spacecraft can be discharged to zero or even a positive potential by the release of negative charge from either an electron emitter or a plasma device. A positively charged spacecraft, on the other hand, can be discharged to zero or negative potentials only by the release of positive charge from a plasma device.

Another reason for identifying the two major categories is that unless the spacecraft is an equipotential surface, the successful control of the spacecraft potential may dictate the release of both negative and positive charge; even though the net release current requirement is almost always negative. This is based on the results of attempts to control actively the potential of the ATS-5 and ATS-6 spacecraft in Earth's geosynchronous orbit.[4] In these tests it was found that electron release alone was sometimes unsuccessful in maintaining the spacecraft at near-zero potential. Operation of the ion thruster, on the other hand, proved successful in clamping the spacecraft potential to approximately 4 V negative with respect to the ambient plasma potential. The reasons for the failure of the electron device and the success of the plasma device in controlling the spacecraft potential in these tests are not fully understood. One explanation is that the release of electrons alone may result in space charge effects which limit the release of negative charge and thus limits the ability to maintain the spacecraft at zero potential. The use of the ion thruster, on the other hand, may eliminate the space charge limitation by providing charge carriers of both signs. Another explanation is closely related to the classification of charged particle release devices into the low-energy and high-energy groups: Potential barriers may form in the vicinity of the spacecraft and limit the release of low-energy electrons. However, by accelerating the electrons to energies in excess of the potential well value, it may be possible to attain a near-zero spacecraft potential by release of electrons alone. Discharging the spacecraft, however, does not necessarily eliminate the potential barrier; this phenomenon is generally thought to be caused by differential charging of adjacent spacecraft surfaces.

4.1.1 FIELD EMISSION

Electron field emission from tungsten surfaces is appreciable for electric field strengths on the order of 10^9 V/m. Thus a sharp tungsten rod (radius of curvature of 10^{-6} m or less) will emit appreciable current at potentials on the order of 1 kV. A spherical cluster or brush composed of 100 of these needles can yield emission currents on the order of milliamperes when biased[*] to a potential of a few hundred volts.

Grard[5, 6] has proposed the use of an electron field emitter for actively controlling the spacecraft potential. His device consists of a cluster of small diameter tungsten wire bristles which is attached to the spacecraft by means of a boom. A separation between the spacecraft and probe on the order of twice the characteristic dimension of the spacecraft is sufficient to yield field strengths at the tips of the wires which are within 25 percent of the values obtained for infinite separation. Emission currents as high as 6 mA can be drawn from the device, with the current limitation imposed by the thermal properties of tungsten, rather than a space charge limit. A schematic diagram of Grard's arrangement is presented in Figure 2 which indicates the spacecraft collection current I and illustrates the return of this current to space by means of the electron field emission probe and biasing power supply.
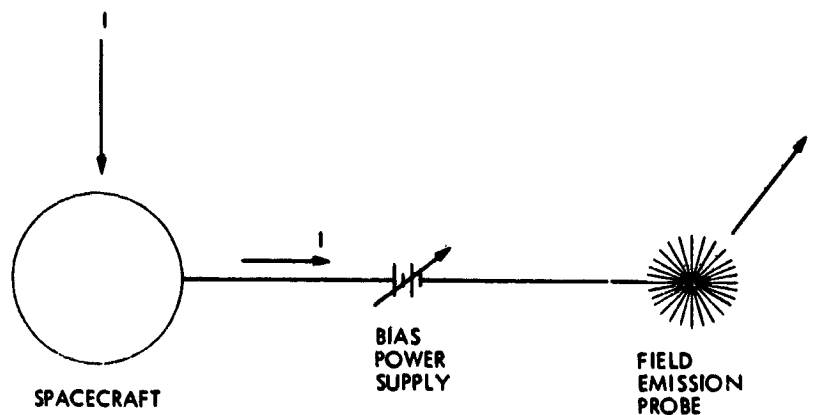

SPACECRAFT

I

BIAS
POWER
SUPPLY

FIELD
EMISSION
PROBE

I

Figure 2. Schematic Diagram of an Active Spacecraft Potential Control System which Utilizes an Electron Field Emission Probe

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[*]Unless stated otherwise, the bias and filament potentials mentioned throughout the paper are negative with respect to the surrounding plasma.

The electron field emission current is governed by the Fowler-Nordheim equation, and the current-voltage characteristics calculated using this relationship are presented in Figure 3 for a probe consisting of one hundred $0.1\text{-}\mu m$ diameter tungsten needles. The general trend illustrated by this figure is a voltage threshold of about 400 V beyond which the current increases rapidly with voltage. This feature of the current-voltage characteristic enables the field emission probe to be used as a stable voltage reference for biasing the entire spacecraft. At the initial mission perijove the release current required to maintain the JOP spacecraft at the ambient plasma potential is about 0.3 mA, and from Figure 3 a biasing potential of 625 V is required to achieve this emission level. Thus the power requirement to maintain the spacecraft at the ambient plasma potential is a relatively modest 200 mW. This is a conservative estimate since the actual current required to maintain this potential would be somewhat less than 0.3 mA due to secondary and photoemission currents.
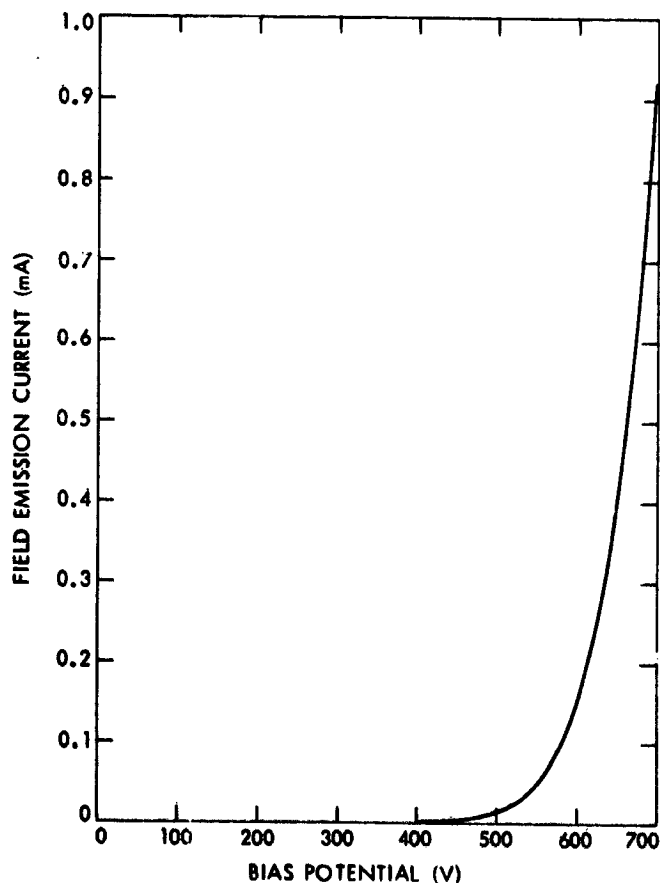

Figure 3. Calculated Electron Field Emission Current From a Probe Consisting of One Hundred 0.1-μm Diameter Tungsten Needles

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The electron field emission probe is an attractive system for use in active spacecraft potential control since it requires no expellant, is lightweight (Grard[6] estimates 150 g for the emitter and boom), relatively simple, and requires only a biasing power supply. However, there is a drawback associated with the device: Tungsten needles having dimensions on the order of 0.1 μm are fragile and not visible to the naked eye. Thus the use of an electron field emitter for active potential control on a spacecraft may not be practical due to problems associated with handling and launching the device.

4.1.2 THERMIONIC EMISSION

The refractory metal cathode has been used for the active control of space-craft potential since the first ion thruster flight test was conducted in 1964.[7] In-flight thrust measurements conducted during this test verified the effectiveness of the heated tantalum filament neutralizer in producing a neutral exhaust beam and preventing spacecraft charging. More recent tests using the filament neutralizer on the ATS-5 spacecraft have shown that operation of the neutralizer filament alone can (at least in some instances) restore the potential of the spacecraft to a value near zero, even after having been initially charged to negative potentials on the order of a few thousand volts.[4]

Grard et al[8] have proposed a spacecraft potential control and plasma diagnostic system which employs a thermionic electron emitter. The system consists of a heated filament and biasing power supply as illustrated schematically in Figure 4. The current-voltage characteristics of the device can be calculated by assuming space charge limited flow conditions between two concentric spheres. The inner sphere represents the boom-mounted emitter, and the outer sphere represents the plasma sheath boundary. The space charge limited flow solutions obtained for the spherical geometry are presented in Figure 5 for a concentric sphere diameter ratio of 30. This ratio is representative of the plasma sheath thickness and emitter dimensions, although (as will be shown later) the results are not too sensitive to this parameter. The rapid rise in emission current with voltage, as illustrated in Figure 5, suggests the thermionic emitter can be used as a stable voltage reference for biasing the entire spacecraft. Figure 5 indicates a biasing potential of about 10 V is sufficient to maintain the spacecraft at the ambient plasma potential by supplying an emission current of 0.3 mA. This results in a bias power requirement of about 3 mW, which is significantly less than the 200 mW requirement of the field emission probe. However, this difference is more than offset by the filament power requirement of about 500 mW for a tantalum filament operated at 2100°K.
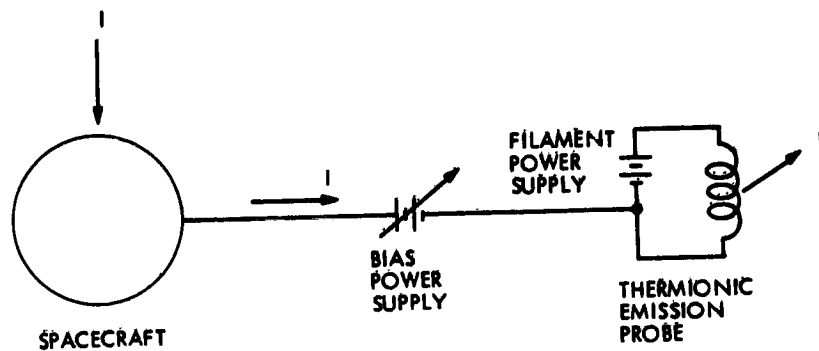
Figure 4. Schematic Diagram of an Active Spacecraft Potential Control System which Utilizes an Electron Thermionic Emission Probe
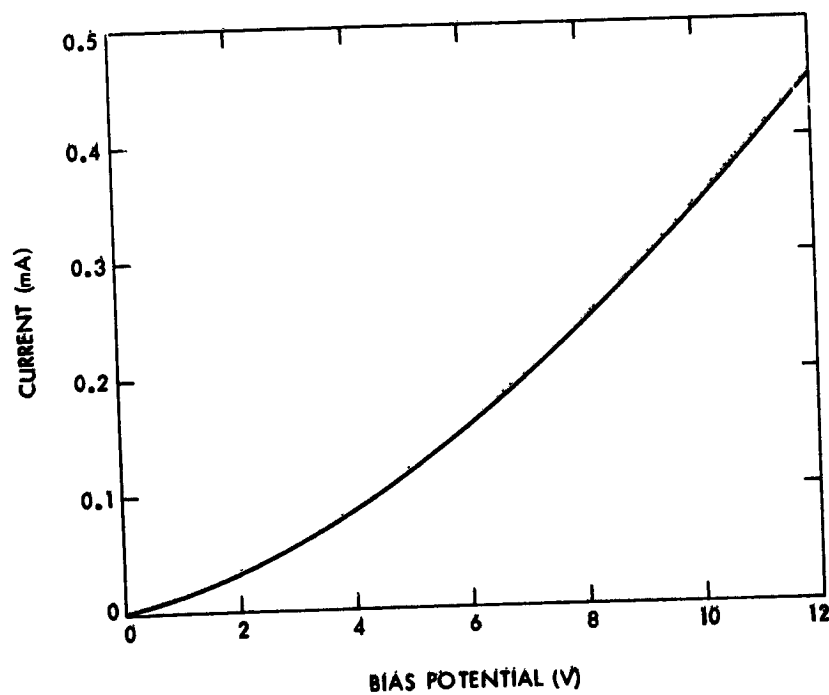

Figure 5. Calculated Space Charge Limited Electron Current Between Concentric Spheres Having a Diameter Ratio of 30

The refractory metal cathode is a viable candidate for a low-energy electron release device and has several outstanding features. The system requires no expellant, has a power consumption of less than 1 W, and is lightweight (Grard et al[8] estimate 150 g for the emitter and boom). In addition, the device is relatively simple and thermionic electron emitters have been used successfully in space for many years. The results of the ATS-5 spacecraft potential control demonstration have shown that the thermionic emitter can reduce the potential of a spacecraft (initially charged negatively) to a value near zero. However, in some instances the ATS-5 neutralizer filament was ineffective in maintaining the spacecraft potential near zero. There are at least two plausible explanations for these failures: (1) A potential well may exist near the surface of the spacecraft which suppresses the emission of low-energy electrons from the heated filament and (2) the ATS-5 neutralizer filament is recessed about 2.5 cm within a 5-cm diameter aperture located on the surface of the spacecraft, and under some conditions local space charge effects may reduce the filament emission current obtained with this geometry. However, the problems discussed above may well be eliminated by use of the system illustrated in Figure 4, since the filament is mounted on a long boom and can be biased relative to the ambient plasma.

4.1.3 EMISSIVE CLAMP

An electron emissive clamp for use in actively controlling the spacecraft potential has been proposed by Roy et al[9] and is discussed by Sellen and Fitzgerald.[10] The device is illustrated schematically in Figure 6 and consists of a heated filament nested within two concentraic spherical grids. The filament is maintained at the spacecraft potential and the grids are biased positive with respect to the filament. Electrons emitted by the filament are accelerated radially outward by the electric fields existing between the filament and grids. These electrons form a cloud or virtual cathode near the outer grid radius. Some of the electrons in the cloud are collected by the grids, and the remainder escape to space. The difference between the escaping electron current and the net plasma collection current represents the emission current of the device.

The basis for the emissive probe design can be understood by considering the expression for space charge limited flow between two electrodes of arbitrary geometry

$$I = kV^{3/2} \tag{1}$$

where I is the space charge limited current, and V is the potential difference between the electrodes. In general the perveance k is a constant for a given electrode pair and is determined by the geometry and size of the electrodes. For the case of spherical geometries, the value of k is determined by the ratio
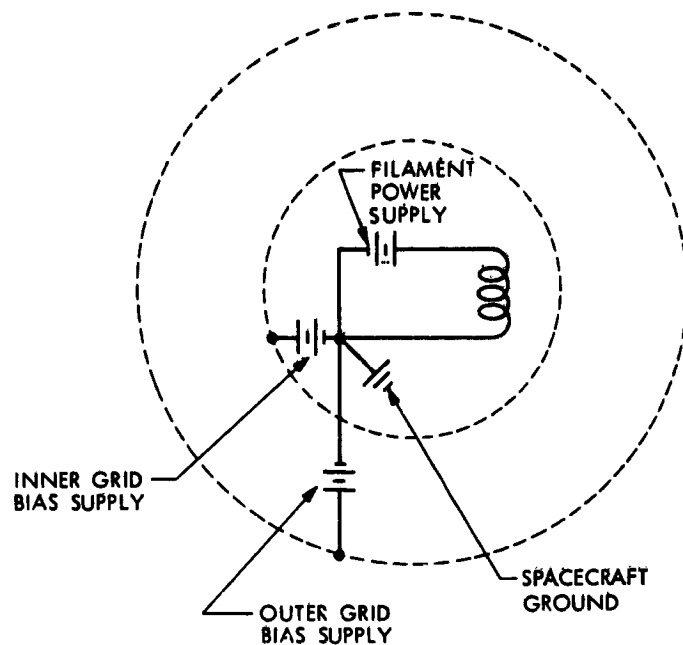
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FILAMENT
POWER
SUPPLY

INNER GRID
BIAS SUPPLY

OUTER GRID
BIAS SUPPLY

SPACECRAFT
GROUND

Figure 6.   Schematic Diagram of the Emissive Clamp
Device of Roy et al[9]

of the diameter of the outer and inner concentric spheres. In this case the
outer spherical diameter is determined by the plasma Debye length λ_D, and the
inner diameter is determined by the radius of the virtual cathode R_o. The varia-
tion of k with the diameter ratio is presented in Figure 7, where it is seen that for
diameter ratios greater than about 20 the value of k remains fairly constant. How-
ever, for diameter ratios less than about 10 the value of k increases rapidly with
decreasing diameter ratio. For an outer spherical collector diameter determined
by the plasma Debye length λ_D, a reduction in the diameter ratio corresponds to
an increase in the diameter of the inner spherical emitter. Thus, in order to
achieve significant emission current levels at low spacecraft potentials, it is
desirable to emit the electrons from a relatively large spherical cathode. The
emissive clamp geometry accomplishes this by use of a set of nested spherical
grids which surround a thermionic emitter. This arrangement produces a virtual
cathode some 30 cm in diameter[10] and allows substantial emission currents to be
drawn at relatively low spacecraft potentials.

Laboratory data obtained with the emissive clamp are presented in Figure 8
which shows the emission current as a function of negative spacecraft (filament)
potential for various values of the voltage applied to the outer spherical grid. The
data presented in Figure 8 indicate release currents on the order of tens of
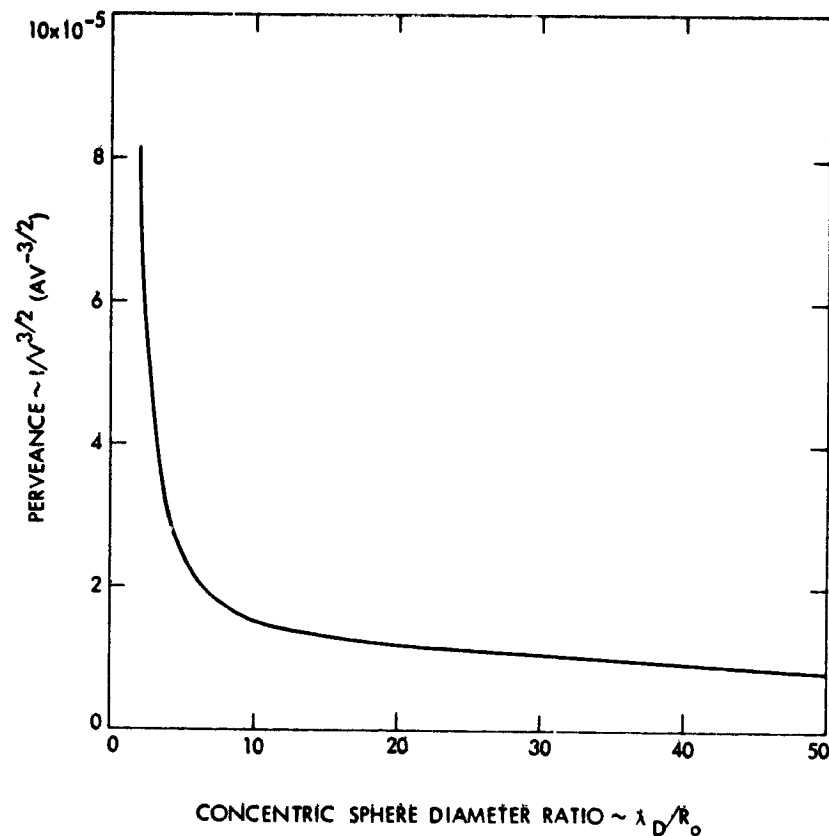
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Figure 7. Variation of the Perveance k with Diameter Ratio for Space Charge Limited Electron Flow between Concentric Spherical Electrodes

microamperes can be achieved at spacecraft potentials on the order of 1 V. The spacecraft potential required for a given release current is dependent on the potential applied to the outer spherical grid, and this permits the spacecraft to be biased by varying the outer grid potential.

In order to obtain the emission current required for the JOP application while maintaining the spacecraft at the ambient plasma potential, the emissive clamp must be biased negative with respect to the spacecraft. An estimate of the biasing potential required to achieve the desired emission current was made by extrapolating the experimental data of Figure 8. These results are presented in Figure 9 which indicates the required biasing potential as a function of the outer grid potential. The extremes of Figure 9 correspond to a biasing power requirement in the 3.0 to 7.5 mW range. The total power requirement of the emissive clamp and
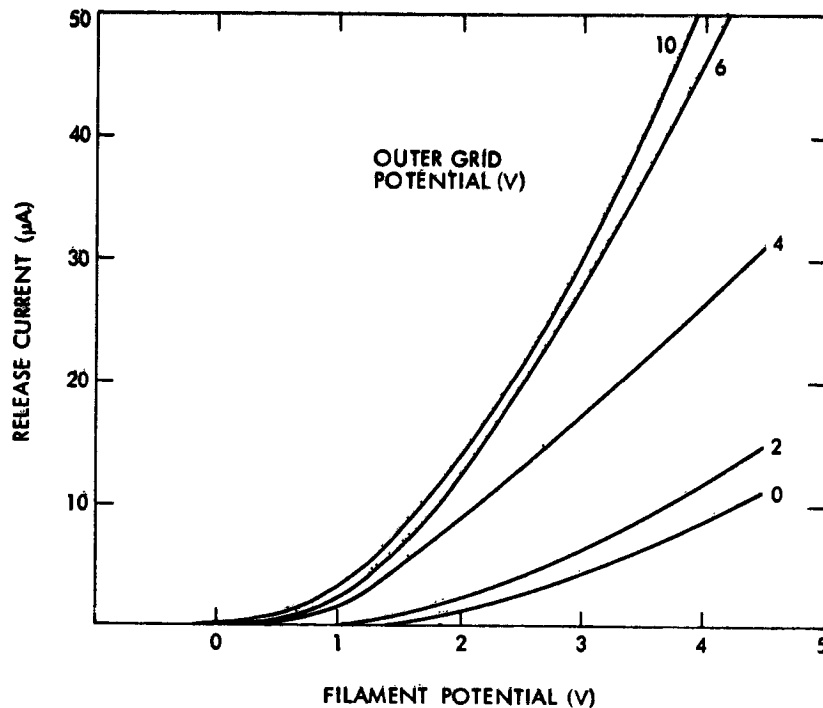
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Figure 8.   Release Current From Electron Emissive Clamp as a
Function of Filament Potential.   (From Reference 10.)

biasing supply is estimated at about 1.4 W, with most of the power consumed by
the heated filament.   The estimated system mass is 0.9 kg.

The electron emissive clamp is capable of providing relatively large emission
current levels at low spacecraft potential.   However, in order to maintain the
spacecraft at the ambient plasma potential a biasing power supply must be used.
In this sense the emissive clamp is less efficient than a biased thermionic emitter,
since it requires power supplies for both the inner and outer grids as well as the
filament and biasing supplies.   However, in perhaps a more important application
the emissive clamp may be used as a sensitive indicator of the sign of the space-
craft potential relative to the surrounding plasma.   The current-voltage character-
istics of the emissive clamp indicate that a relatively small potential difference
between the spacecraft and its surroundings produces a measurable release cur-
rent, and this feature allows the clamp to act as a sensitive swich for use in an
active control system.   In this application, the absence of any release current
from the emissive clamp would indicate a positively charged spacecraft and could
be used to signal positive ion release from a plasma device.   Likewise, the
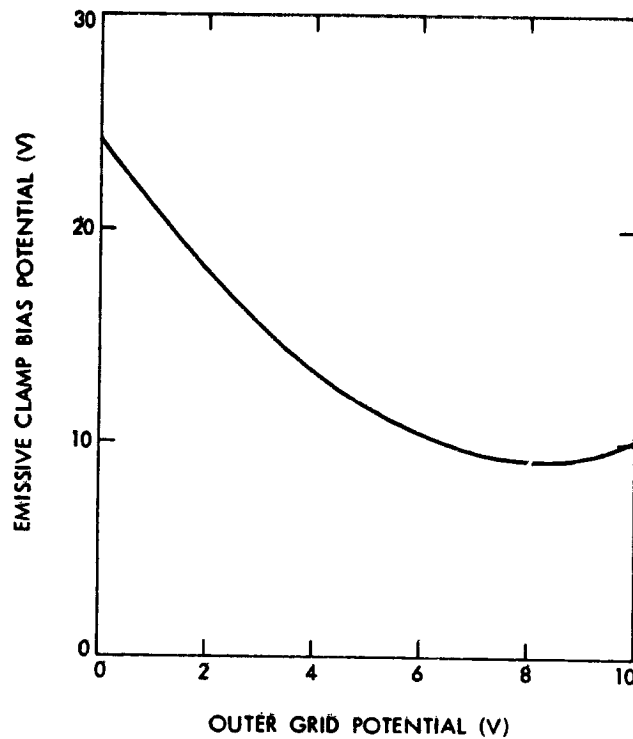
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Figure 9. Emissive Clamp Bias Potential as a Function of Outer Grid Potential for a Release Current of 0.3 mA

presence of a release current from the clamp would indicate a negatively charged spacecraft and could be used to signal electron release from either an electron emitter or plasma device.

4.2 Electron Devices-High Energy

4.2.1 ELECTRON GUN

Electron guns have been used for many years on spacecraft and rockets designed to obtain scientific data such as magnetic field line length and shape, particle drift rates, and various other magnetospheric phenomena. In planning these experiments the problem of spacecraft charging caused by the electron gun operation was recognized, and steps were taken to minimize the potential excursions experienced by the spacecraft. For example, Hess et al[11] describe the use of an inflatable conducting collector which was deployed around the rocket to increase its collection area. The large collector area minimized the positive potential the rocket must attain in order to compensate for the current released by the electron gun.

More recently, Polychronopulos and Goodall[12] have used an electron gun to control actively the potential of a rocket flown in Earth's ionosphere. In this particular application, the electron gun was used to maintain the rocket body at constant potential while making Langmuir probe measurements. The electron gun arrangement employed by these investigators is illustrated in Figure 10 and consists of the electron emitter, accelerating grid, and spacecraft potential sensing device. The electron accelerator grid is electrically attached to the spacecraft structure, and the heated filament can be biased with respect to the spacecraft. A floating probe or collector is used as a voltage reference, and the potential difference between the probe and spacecraft is sensed by a voltage follower. When a potential difference is sensed, the voltage follower develops a voltage at its output and this signal is inverted and amplified. When the potential difference is negative, indicating a negatively charged spacecraft, the amplified and inverted output of the voltage follower biases the filament negative and increases the release current. A positively charged spacecraft, on the other hand, results in a positive
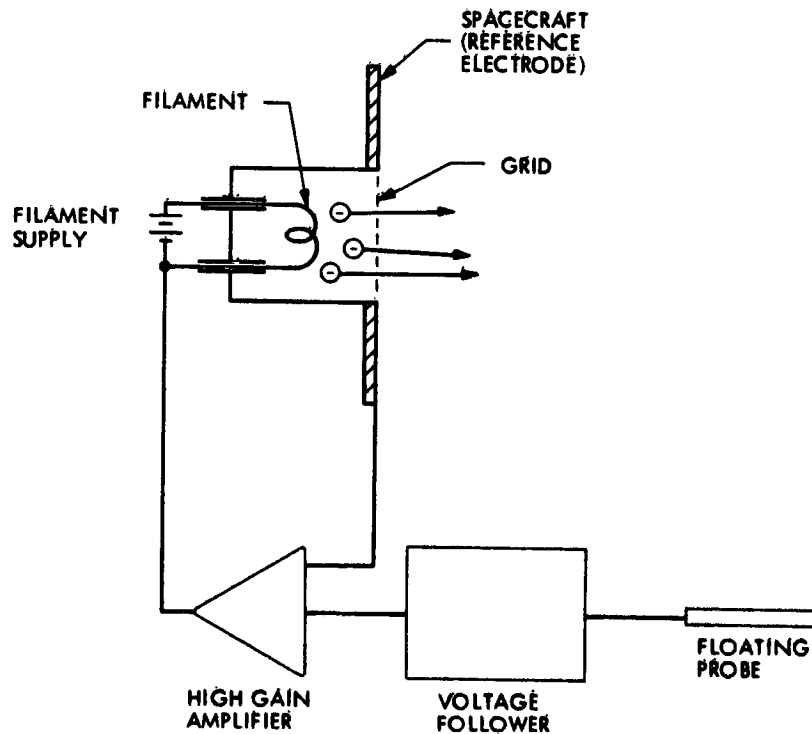

Figure 10. Schematic Diagram of the Electron Gun Configuration of Polychronopulos and Goodall[12]

filament bias which suppresses the release current. The emitting filament used in their electron gun experiment was a commercially available light bulb with the glass cover removed. The filament power requirement is about 3 W, and laboratory tests indicated the electron gun could provide a release current of 0.2 mA with a filament bias of 10 V. The estimated mass of the electron gun configuration is about 0.5 kg.

Both laboratory tests and rocket flight tests demonstrated the ability of this system to maintain the spacecraft potential to within ±10 mV of the floating probe potential. However, two potential problem areas were identified as a result of the rocket flight tests: (1) the floating probe must be deployed far enough away from the spacecraft to insure that it senses the undisturbed plasma and (2) the contact potential difference between the spacecraft and floating probe must be minimized.

The electron gun is a viable candidate for increasing* the potential of a negatively charged spacecraft. The power and mass requirements of the device are fairly modest, and the system has been successfully employed to increase the potential of a negatively charged rocket in tests conducted in Earth's ionosphere. There are, however, two problems which are recognized and would require some modification of the control system used by Polychronopulos and Goodall: (1) the system uses the undisturbed plasma floating potential as a reference voltage. In the Jovian environment the floating potential is variable and may be several kilovolts negative with respect to the plasma potential. Thus, this reference is unacceptable; and (2) the system suppresses electron release when the spacecraft potential exceeds the reference value, and this prevents biasing the spacecraft positive relative to the reference potential. The former problem can be overcome by use of an emissive floating probe or some other device for measuring the plasma potential. A solution to the latter problem would involve some additional control circuit logic. For example, if a positive spacecraft potential were desirable, the inverting function of the amplifier could be eliminated and the amplifier gain varied until the desired spacecraft potential was reached.

4.3 Plasma Devices-Low Energy

4.3.1 HOLLOW CATHODE

Hollow cathodes have replaced the filament neutralizers in many ion thruster designs and have been operated successfully during the SERT II and ATS-6 flight tests. The low-density plasma produced by the hollow cathode discharge is a conducting medium which allows efficient coupling between the cathode and positive ion beam. This coupling permits an electron current equal in magnitude to the ion

*An increase in potential is, in this context, an algebraic increase.

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beam current to be extracted from the neutralizer plasma with a relatively low bias potential. Current neutralization of an ion beam in this manner prevents the spacecraft from charging to large negative potentials. In a different application, the hollow cathode plasma has been used to couple effectively a spacecraft to the ambient space plasma. In these recent tests, the cesium hollow cathode neutralizer on the ATS-6 satellite was used to prevent this spacecraft from charging to large negative potentials during eclipse periods. The fact that the cathode discharge plasma is quasineutral suggests the device might also be used for lowering the potential of a positively charged spacecraft. However, in this application one would expect larger coupling voltages (and hence higher power requirements) due to the low mobility of the heavier ions.

A sketch of the hollow cathode is presented in Figure 11. The device consists of a cylindrical cavity with an orifice located at the downstream end. The upstream end is attached to a valve or vaporizer which controls the flow of a gas such as mercury, cesium, argon, or xenon.* An anode or keeper electrode is located downstream of the cathode orifice, and an electrical discharge between these electrodes produces the low-density plasma. The plasma acts as a good conductor and electron emission currents on the order of amperes can be achieved at a coupling or biasing potential on the order of a few volts. The steady-state power requirements of ion thruster hollow cathodes are typically 2-10 W for the keeper power supply and 3 W for the bias supply. In addition, a heater power requirement of about 30 W is generally required for startup. The mass of the ATS-6 neutralizer assembly is about 45 g, and the estimated xenon gas required for 1000 hr of hollow cathode operation is 25 g. Hollow cathodes using mercury gas have been operated in laboratory tests for as long as 20,000 hr,[13] and cesium and mercury cathodes have accumulated many operational hours in space. The mercury hollow cathodes on the SERT II spacecraft have been restarted several hundred times and remain fully operational after some 6 years in space.[14]

There remains another aspect of operating a plasma device, such as a hollow cathode or an ion thruster, which may ultimately dictate their use for obtaining meaningful spacecraft potential control. Spacecraft charging tests conducted with the filament neutralizer on the ATS-5 spacecraft indicated that although this device could reduce the potential to a near-zero value, a potential barrier surrounded the spacecraft.[15] The effect of this barrier was to prevent the low-energy plasma electrons from reaching the particle detectors on the spacecraft. The use of the ATS-6 ion thruster, on the other hand, successfully discharged the spacecraft and reduced substantially the potential barrier effect. These results suggest the

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*For spacecraft charging applications an inert, noncondensible gas such as argon or xenon is desirable in order to prevent contamination of the cold spacecraft surfaces.

beam plasma produces a space charge neutralization effect which affects a reduction in the potential barrier height. Operation of the hollow cathode may well produce the same result.
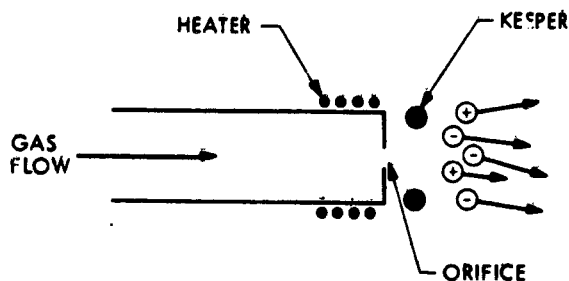

Figure 11. Schematic Diagram of a Hollow Cathode

4.4 Plasma Devices-High Energy

4.4.1 PLASMA GUN

Among the various types of plasma guns, the electron bombardment ion thruster is probably best suited for use as a high-energy plasma source for active spacecraft potential control. This type of source has been operated successfully in space and has been fabricated and tested in a variety of sizes ranging from less than a centimeter to 1.5 m in diameter. The SCATHA satellite[16] will employ a 2-cm diameter xenon ion source for use in spacecraft charging control experiments.

A schematic diagram of an electron bombardment ion source is presented in Figure 12. The device consists of a hollow cathode, anode, and accelerating electrodes. Electrons produced in the hollow cathode discharge are used to ionize the gas atoms as a result of collisions, and the collision probability is increased by use of a magnetic field arranged parallel to the thruster axis. Some of the ions produced in the discharge chamber drift toward the accelerating electrodes and are drawn out to produce the high-energy ion beam. To prevent excessive charging of the spacecraft on which the ion source is mounted, an electron source is located downstream of the accelerator system. The source of the neutralizing electrons can be either a heated filament or hollow cathode neutralizer. The quasineutral beam acts as a good conductor and assumes a potential near that of the environment. This allows the spacecraft to be biased by controlling the emission current of the neutralizer. A reduction in neutralizer emission current causes the spacecraft to charge negatively, while an increase in emission current
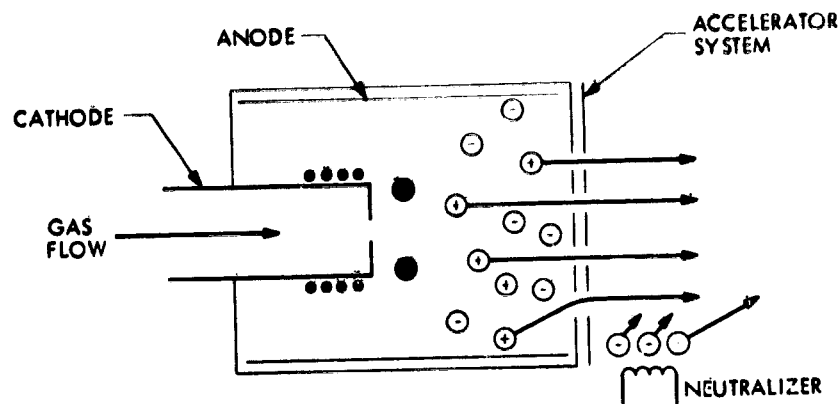
Figure 12. Schematic Diagram of an Electron-Bombardment Ion Source

causes the spacecraft to charge positively. The ability of the ion thruster to increase the potential of a negatively charged spacecraft has been demonstrated in the ATS-6 flight tests.[15] However, attempts to bias a spacecraft positive with respect to the surrounding plasma have not been successful in either laboratory or flight tests.[17] The reason for these failures is thought to be due to the interaction between the ion beam and spacecraft caused by the presence of the charge exchange plasma produced downstream of the accelerator system. Mounting the plasma source on a boom should substantially reduce this interaction, however, since the plasma density is inversely proportional to the square of the distance from a point source.

The steady-state power and mass requirements of the plasma sources are substantially higher than those for an electron emitter. The large power requirements reflect the additional energy required to ionize the neutral gas atoms as well as energy loss mechanisms, such as recombination and radiation, which occur within the ionization chamber. The higher mass requirement is, of course, due to the increased power supply requirements, in addition to the mass of the expellant and its storage and control system.

5. POWER AND MASS ESTIMATES

The extimated power and mass requirements of the charged particle release devices are presented in Table 1. The devices are arranged in the table according to increasing complexity which generally corresponds to increasing power and mass penalties as well as attractiveness as a release device. The majority of the

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Table 1. Estimated Power and Mass Requirements

Device	Power	Mass	Comments
Electron Field Emitter	0.2 W	0.2 kg	Possible Handling and Launch Problems
Electron Therm-ionic Emitter	0.5 W	0.3 kg	Flight Experience. Demonstrated Limited Potential Control Capability on ATS-5.
Electron Emissive Clamp	1.4 W	0.9 kg	Laboratory Tested.
Electron Gun	3.5 W	0.5 kg	Flight Experience Both as a Control Device and Diagnostic Tool.
Plasma Hollow Cathode	13 W	1.6 kg	Flight Experience. Demonstrated Potential Control Capability on ATS-6.
Plasma Gun	25 W	7.3 kg	Flight Experience. Used as a Potential Control Device on SERT II and ATS Spacecraft.

data presented in Table 1 were taken from hardware and design information available in the literature. In those instances where no data were available, the power and mass requirements were calculated based on either the experimental or theoretical emission current characteristics of the device. The power estimates for the hollow cathode were based on experience with ion thruster hollow cathode designs, and these estimates may therefore be overly conservative since ion thruster cathode emission currents are on the order of amperes, while the present application requires an emission current on the order of a milliampere. The requirements shown for the plasma gun were taken from the SCATHA ion source design goals, and it is interesting to note that a significant fraction of the power requirement of this device is consumed by the hollow cathode. Hence the development of an efficient, low-power hollow cathode would greatly enhance the competitive positions of both the hollow cathode and plasma gun devices.

The mass estimates presented in Table 1 include the release device and associated power supplies, and in the case of the electron devices the mass of a support boom is included. The mass estimates for the plasma devices also include a quantity of xenon gas sufficient to provide 1000 hr of continuous operation, as well as the gas storage and flow control equipment.

6. SUMMARY AND CONCLUSIONS

Accurate determination of the charged particle distribution functions in Jupiter's magnetosphere will require an active potential control system on the Jupiter Orbiter spacecraft. A spacecraft in Jupiter's magnetosphere will generally tend to charge negatively with respect to the surrounding plasma, and this dictates a net electron release current in order to increase the spacecraft potential. This current requirement can be met by use of a thermionic electron source or a plasma device.

As a result of this preliminary investigation, the following general conclusions can be drawn concerning the selection of a charged particle release device for use in an active spacecraft potential control system:

(1) Electron release devices have the lowest power and mass requirements and are simpler, but flight experience suggests they may not be as effective as plasma devices in controlling spacecraft potential. Their capability of discharging the spacecraft is apparently degraded by the presence of differentially charged areas of the spacecraft, and this dictates the following design considerations:
(a) Eliminate differencial charging by designing an equipotential spacecraft, and
(b) mount the emitter sufficiently far from the spacecraft to minimize the interaction between the two.

(2) Plasma devices have higher power and mass penalties associated with them, but they are more flexible than electron emitters since they provide the capability of releasing charge carriers of either sign.

(3) The selection of a charged particle release device for use with an active potential control system will ultimately reflect a compromise between the mission science objectives, spacecraft conductivity, and the power and mass requirements of the devices.

(4) Either type device may be considered as a potential science instrument for performing plasma diagnostic studies, since they are capable of varying the potential of the entire spacecraft.

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