EFFECT OF SOLAR ARRAY PLUME INTERACTIONS ON HALL THRUSTER CATHODE COMMON POTENTIALS

Ira Katz (1), Alejandro Lopez Ortega (1), Dan M. Goebel (1), Michael J. Sekerak (1), Richard R. Hofer (1), Benjamin A. Jorns (1), John R. Brophy (1)

(1) Jet Propulsion Laboratory, California Institute of Technology, Pasadena, CA USA, Email:ira.katz@jpl.nasa.gov

ABSTRACT

In typical implementations of spacecraft electric propulsion (EP) systems the electrical ground of the EP power processing units are resistively isolated from the main spacecraft electrical ground. Telemetry measuring the potential difference between EP power processing unit and the spacecraft electrical ground is usually referred to as the "cathode common" potential. In this paper we review flight and laboratory data on natural and EP generated plasma interactions with solar arrays, present the theory of spacecraft floating potentials, show how this theory is consistent with flight data, and extend this theory to spacecraft with high power Hall effect thrusters.

1. INTRODUCTION

In the near future, spacecraft, such as the proposed Asteroid Redirect Robotic Mission (ARRM), would be propelled by tens of kilowatt solar electric propulsion (SEP) systems using high power Hall effect thrusters. These electric propulsion (EP) systems would generate a denser plasma cloud than previous SEP spacecraft with gridded ion thrusters, such as Dawn [1]. If the potentials between spacecraft surfaces and the EP generated plasma are large, they may accelerate plasma ions enough to cause significant erosion of spacecraft surfaces by ion sputtering. Both gridded ion and Hall effect thrusters have hollow cathode plasma sources in contact with this plasma cloud. The potential difference between the spacecraft chassis electrical ground and the hollow cathode, usually referred to as "cathode common voltage", is a good measure of the potential between the spacecraft and the plasma [2].

It is important during the design process to have a good method of predicting this potential to prevent ion erosion from damaging critical spacecraft components. In this paper we identify the components and interactions that control the "cathode common voltage" along with the ground tests and models necessary to make preflight predictions. We will start from the simplest basics of plasma interactions and build up a complete model by adding the multiple components of a high power, Hall effect thruster, solar electric spacecraft. We begin with the plasma in Low Earth Orbit (LEO) and how it interacts with insulating and conducting spacecraft surfaces. We then add solar arrays with exposed interconnects at a range of voltages. Next comes the EP system generated plasma, followed by the resistor used to isolate the EP system power processing unit from spacecraft chassis ground. We combine all these components to show what controls "cathode common" in several configurations, including a spacecraft with gridded ion thrusters (Dawn [3]), and ones with high power Hall thrusters with differing hollow cathode locations and thruster body surface coatings and grounding schemes.

2. SPACECRAFT FLOATING POTENTIALS

The LEO ionosphere plasma density is within an order of magnitude or two of EP generated plasma plumes. Typical plasma density and temperatures are the order of

$$n_e = n_i \approx 10^{11} m^{-3}$$
$$T_e \approx 0.1 eV$$

The Debye length, λ_D, a good measure of the distances over which the plasma screens out potentials, is about a centimetre, smaller than typical spacecraft dimensions.

The electron thermal velocities are much faster than orbital velocities, and the electron current density to a surface at plasma potential is proportional to the electron thermal velocity, while ion thermal velocities are smaller than orbital velocities, the ion current density on ram facing surfaces is proportional to the spacecraft velocity.

$$j_e = -en\frac{u_e}{4} = -en\sqrt{\frac{eT_e}{2\pi m_e}} \approx -0.85\ \frac{mA}{m^2}$$
$$j_i = enu_{s/c} \approx 0.12\ \frac{mA}{m^2} \tag{1}$$

The "floating potential", steady state potential on an insulating surface exposed to plasma currents, is reached when the ion and electron current densities balance and there is no net current to the surface. Since the ambient electron current density is much higher than the ion current density, surfaces usually charge negative to repel most of the electron current. For a thermal plasma, the current density decreases exponentially with potential.

$$j_e(\phi) = j_e \exp\left(\frac{\phi}{T_e}\right) \qquad (2)$$

The ions have about 5eV kinetic energy from the relative orbital velocity, and their current density varies little with potential.
The floating potential of an insulating surface in LEO is within a volt of the ambient plasma potential.

$$\begin{aligned} & j_e(\phi_f) + j_i = 0 \\ & \frac{\phi_f}{T_e} \approx \ln\left(\frac{0.12}{0.85}\right) \approx -2 \\ & \phi_f \approx -0.2\,eV \end{aligned} \qquad (3)$$

The floating potential on conducting surfaces exposed to the surrounding plasma is determined by the surface integrals of the currents.

$$\int (j_e(\phi_f) + j_i)\, dA = 0 \qquad (4)$$

The current densities vary around the spacecraft due to ram/wake effects, but the floating potentials on typical spacecraft still within a few volts from plasma potential. (On large spacecraft, such as the International Space Station (ISS), near the poles the orbital motion through the Earth's magnetic field can generate ten volts or so potentials.)
Conducting interconnects on solar arrays expose a range of potentials, from 0 to ϕ_{SA}, to the space plasma. The floating potential is then determined by the surface integral including the array generated potentials. Since solar array generated voltages are larger than either electron thermal or ion ram energies, we can assume that positive surfaces collect electrons and negative surfaces collect ions.
From Eq. 1, the electron current density is about 7 times the ion current density, so current balance is achieved when 1/8 of the interconnect area is positive and 7/8 of the area is negative. If the maximum solar array voltage were 100V, then the array floating potential is about -75 V. Because the spacecraft body has significant exposed conducting surfaces connected to chassis ground, spacecraft with solar arrays float considerably less negative than solar arrays by themselves as shown schematically in Fig. 1. In Fig. 1 there are two potential scales shown. On the left are the potentials with respect to spacecraft chassis ground; on the right, potentials with respect to the local plasma. The difference between the two scales is just the spacecraft chassis floating potential. Plasma potential is indicated by the dashed horizontal line and the current loop is shown by the grey ellipse.

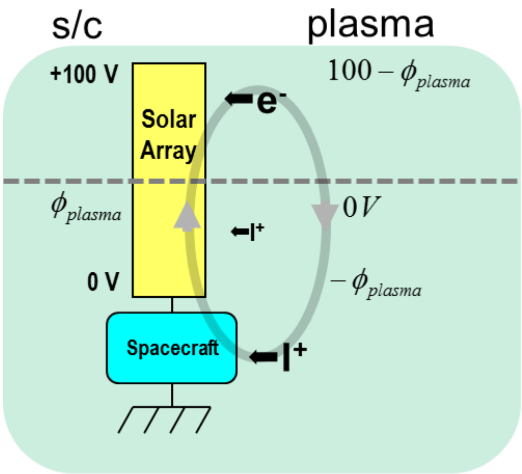

Figure 1. Spacecraft floating potential where ion current to the chassis balances electron collection to the solar arrays

This result is consistent with data from UARS [4] and SERT-II [5], two spacecraft instrumented to measure the spacecraft floating potentials with respect the ionosphere. A notable exception to these results is the International Space Station (ISS), whose high voltage solar arrays collects very little electron current due to their unique construction. Rather than exposed metallic interconnects, the space station arrays use wrap-through interconnects that are not exposed to the surrounding plasma.

3. POTENTIALS ON SPACECRAFT WITH ELECTRIC PROPULSION GENERATED PLASMAS

We have reviewed how spacecraft with high voltage solar arrays interact with the ionosphere. Now we introduce the plasma generated by on-board electric propulsion systems. We use data from SERT-II [5] to illustrate the physics. The potential of the plasma produced by SERT-II gridded ion thruster's hollow cathode as a function of the cathode bias voltage was measured using an emissive probe. While on SERT-II, the cathode bias voltage was controlled by a power supply, the bias potential is exactly the "cathode common" voltage.
As shown in Fig.2 the plasma potential varies linearly with the bias voltage with an offset of about 15V. This offset is due to the sheath and resistive potential drops inside the hollow cathode as shown schematically in Fig. 3. For most hollow cathodes this drop is the order of 10V - 20V. Both gridded ion thrusters and Hall effect thrusters use hollow cathodes, the plasma potentials in the thruster generated low energy plasma plume at high angles relative to the main thrust beam are 10V - 20V of "cathode common".

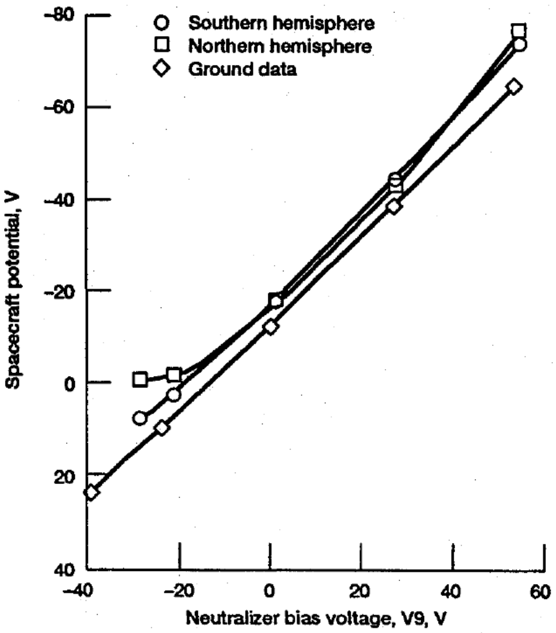

Figure 2. SERT-II spacecraft potential control

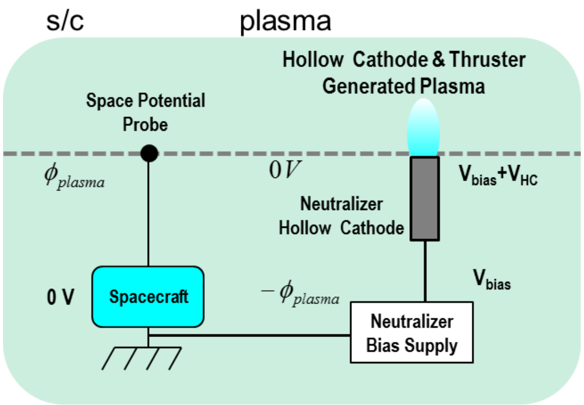

Figure 3. Potential in a hollow cathode produced plasma is about 15 V positive of cathode common

A mission in NASA's Discovery Program, Dawn orbited and explored the giant protoplanet Vesta in 2011-2012, and now it is in orbit and exploring a second new world, dwarf planet Ceres. The Dawn spacecraft was launched on September 27, 2007. Its ion propulsion system allowed Dawn to go into orbit around two different solar system bodies, a first for any spacecraft. The Dawn spacecraft is propelled by three xenon gridded ion thrusters based on an evolution of the NSTAR technology used by the Deep Space 1 spacecraft [6], but uses only one at a time. Over the course of the mission, the ion thrusters changed the Dawn spacecraft velocity by more than 11 km/s, far more than any previous spacecraft achieved with an on-board propulsion system after separation from its launch vehicle. The spacecraft, including the ion propulsion thrusters, is powered by a 10 kW (at 1 AU) solar array consisting of two wings, each 8 m long and 2.3 m wide.

As part of the initial checkout after launch, during an attempt to run one of the ion thrusters, an unexpected shutdown occurred when telemetry indicated that the neutralizer common voltage – the voltage between spacecraft ground and the neutralizer cathode – exceeded a preset limit of +40 V. There was no fault. The preset limit had been set without taking into account that with the high value of the isolation resistance, the small electron current collected by the solar arrays in certain orientations, results in a voltage drop of more than 40 volts.

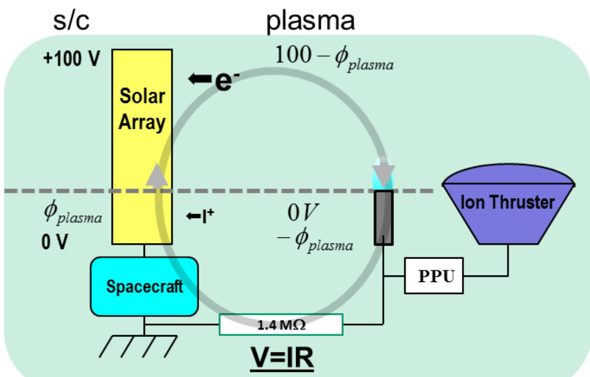

Figure 4. Electrons collected by the Dawn solar arrays cause a voltage drop across the isolation resistor

Few thruster generated ions are collected on conducting surfaces connected to spacecraft chassis ground. As a result, most of the electrons collected by the solar arrays complete the circuit by flowing through the resistor and out the hollow cathode neutralizer as shown in Fig. 4. It only takes about 30 µA to exceed the 40V limit. As part of the thruster start-up sequence, the thruster discharge is run without the accelerating voltage. This operation, known as diode mode, produces a denser low energy plasma plume at high angles then when thrusting. In diode mode, the neutralizer cathode common voltage occasionally exceeds the telemetry limit, +50V.

4. HALL THRUSTER SOLAR ARRAY PLASMA INTERACTIONS

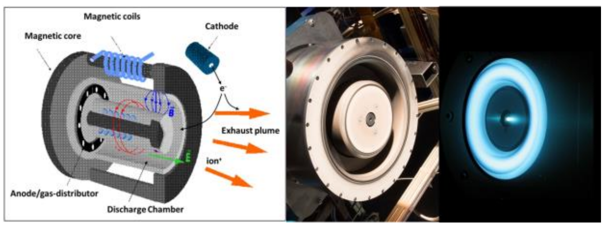

Figure 5. Hall effect thrusters

Hall effect thrusters ionize propellant gas in a cylindrical channel [2]. An axial electric field in a strong radial magnetic field accelerates the ions downstream. Hall thrusters use hollow cathodes to provide electrons to ionize the gas in the channel and charge and current neutralize the main thrust beam. The hollow cathode can be mounted either outside of the thruster channel (Fig.5 left) in the center of the thruster (Fig. 5 center & right). Hall thrusters operate at higher currents and lower voltages than gridded ion thrusters. For example, at 4.5 kW the Aerojet Rocketdyne XR-5 Hall thruster [7] has nearly four times the current of NASA/GRC's NEXT gridded ion thruster [8]. Higher currents lead to higher plume plasma densities. A simplified diagram of a Hall thruster system is shown in Fig 6. Note that the thruster body is connected to spacecraft chassis ground.

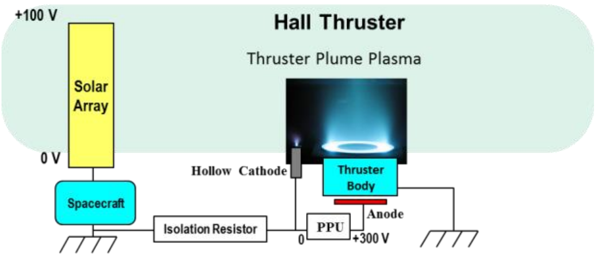

Figure 6. Hall effect thruster system

We examine three electrical configurations of Hall thrusters: 1.) an insulating surface on the pole pieces, as shown in Fig. 7 left and the thruster body tied s/c chassis ground, 2.) exposed conducting pole pieces and the thruster body tied s/c chassis ground shown in Fig. 7 right, and 3.) exposed conducting pole pieces and the thruster body tied to cathode common. In each of the cases the analysis is for a centrally mounted hollow cathode and generalization of the results to externally mounted cathodes is discussed

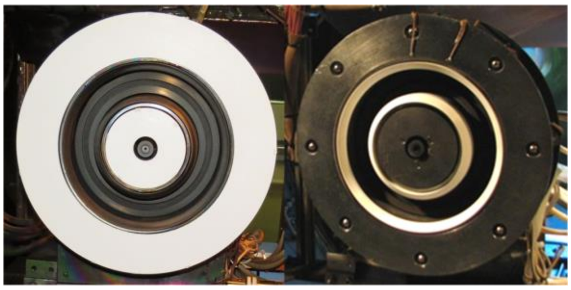

Figure 7 Hall thruster with insulating pole piece surfaces (left) and conducting surfaces (right).

As well as the solar array electron current collection data, the current voltage characteristics of the thruster body in the Hall thruster plasma is needed to predict cathode common potentials. Laboratory data was obtained for magnetically shielded 6kW H6MS Hall thruster [9, 10] with centrally mounted cathode operating at 300V 20A.

With the insulating pole piece surfaces the ion current to the thruster body was relatively small, about 3 mA. This value may be much higher for externally mounted cathodes, particularly if the hollow cathode is mounted outside the thruster body and cathode ions have a direct line of sight to the body.

The current voltage characteristic of the thruster with conducting pole pieces, obtained by applying a DC bias to the thruster body, is show in Fig. 8. The ion saturation current is about 300 mA, two orders of magnitude larger than with insulating surfaces on the pole pieces. Note that when the thruster body is connected to chamber ground, 2.5 A of electrons flow from the body to the chamber and help current neutralize the beam. This is much different than in flight, where the main thrust beam ions must be current neutralized by electrons flowing through the beam plasma. The floating potential of the thruster is about -40 V with respect to chamber ground. With conducting pole pieces, the ion currents to the thruster body are orders of magnitude larger than the electron currents collected by the solar arrays. As an example, a calculation for a notional spacecraft with four 9kW Hall thrusters shows the solar arrays are expected to collect less than 10 mA of electrons, far less than the 300mA of ion saturation current for a single 6kW thruster.

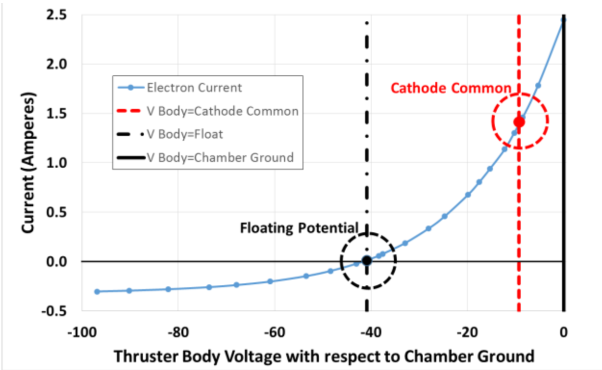

Figure 8. Hall thruster body electron currents with conducting pole pieces.

With a centrally mounted hollow cathode or an external hollow cathode whose orifice is inside thruster magnetic field, the ion current to a Hall thruster body with insulating pole piece covers will be very small. In our laboratory test the ion current to the body was only 3 mA. Ion current to exposed conducting spacecraft surfaces will be also very small, since most such surfaces are behind the thruster exit plane. In this case the spacecraft chassis ground will float negative with respect to the Hall plume plasma so that only a portion of the solar array is positive and collects ions. An upper bound on the cathode common voltage is the resistive drop from the solar array electron collection across the isolation resistor, as shown in Fig. 9. Proper choice of the isolation resistance will limit the cathode common voltage. For example, for our notion example, a 1 kΩ resistor will limit cathode common to less than 10V as shown in Eq. 5. Higher the resistance will drive the spacecraft chassis ground more negative with respect to the plasma, and increase the cathode common voltage. Because ions are accelerated by sheath electric fields, negative chassis potential leads to increased ion energies and therefore increased sputter erosion.

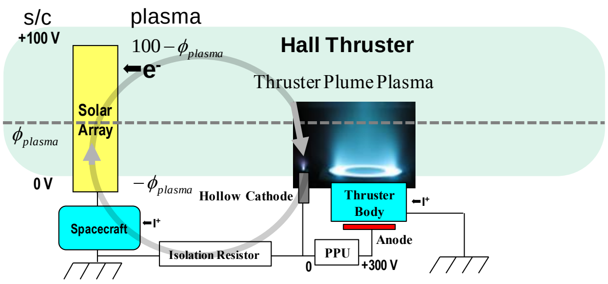

Figure 9. Cathode common circuit for a Hall thruster with insulating pole piece surfaces

$$
\begin{aligned}
&V = I\,R \\
&I < I^{\max} \approx 10\,mA \\
&R = 1k\Omega \\
&V < 10V
\end{aligned}
\qquad (5)
$$

Ion currents to Hall thruster bodies with conducting pole pieces will be quite large. In our laboratory test with a centrally mounted cathode the saturation ion current to the body was about 300 mA. In this configuration, currents to the thruster body dominate, and the thruster body acts like a floating probe as shown in Fig. 10. It is driven negative to repel electrons and balance the thruster body ion current. Since the electron temperatures are high in the dense plasma above the inner pole piece, the floating potential is about -40 V with respect to local plasma. The corresponding cathode common voltage, in this case the potential difference between the hollow cathode and the thruster body, is +30 V.

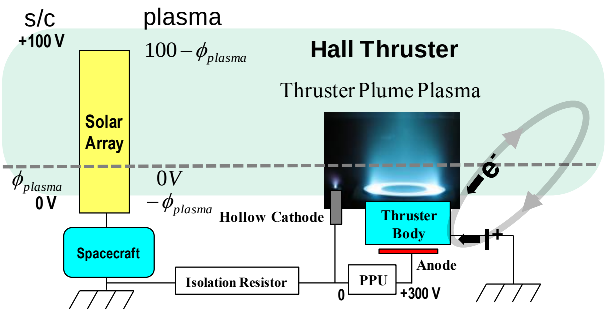

Figure 10. With conducting pole pieces current loop through thruster body closes in the plume plasma

In flight, solar array collection will not play a significant role. The thruster body ion current, about 300 mA, is much greater than the maximum solar array electron collection, less than 10mA. For Hall thrusters with externally mounted cathodes where the cathode plume has unobstructed line of sight to thruster body metal, the thruster body will still tend to anchor the floating potential. However, in this case the voltages will be lower, the order of 10V, since the cathode plasma electron temperature is much lower than in the plasma next to the inner pole piece. If the isolation resistor is a kΩ or larger, the resistor just acts as a voltage probe and won't affect the floating potential or the value of cathode common.

In the first two configurations, the thruster body surfaces can float tens of volts negative with respect to the inner pole plasma. In the first case, insulating pole piece surfaces, while the thruster body potential can be controlled by the value of the isolation resistor, the insulating pole piece surfaces will reach floating potentials near that of the second case thruster body. The negative floating potentials will lead to enhanced sputtering of pole pieces surfaces. The third case, conducting the thruster body tied to cathode common along with conducting pole piece surfaces as shown in Fig. 11, is a new configuration under investigation as a way to minimize the voltage difference between the plasma and the pole pieces.

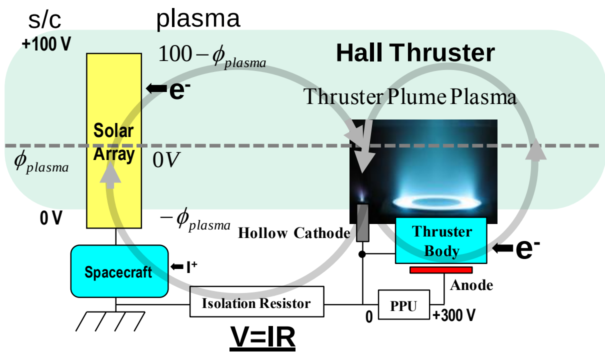

Figure 11. Current loops for Hall thruster with conducting pole pieces and thruster body at cathode common

In this case there are two separate circuits. In the first current loop electrons are emitted from the hollow cathode, flow through the plasma, land on the pole pieces, and are conducted back to the hollow cathode. Laboratory measurements show that about 1.5 A of electrons are collected by the H6MS thruster body and flow through this circuit, less than 10% of the discharge current.

The second circuit is very similar to the first case where the pole pieces had insulating covers where the cathode common voltage is generated by the circuit current flowing through the isolation resistor. The only difference is that now none of the ions hitting the thruster contribute to the circuit and opposed to the 3mA of ions in the first case. The 3mA of ions cancelled out the same amount of electron current prior to the isolation resistor. In the new case, since the electron current flowing through the isolation resistor is increased, we expect the spacecraft chassis ground to float more negative with respect to the Hall plume plasma. Only a portion of the solar array floats positive. An approximate equation for the cathode common voltage is shown in Eq. 6 Again, a 1 kΩ isolation resistor will limit cathode common to 10V.

$$V_{CC} \approx I_{\max} R \left(\frac{V_{array} - V_{CC} - V_{HC}}{V_{array}} \right)^2 \qquad (6)$$

5. CONCLUSIONS

The floating potential on spacecraft in the ionosphere without EP is the balance of solar array electron collection and ram ion collection on spacecraft grounded conducting surfaces. Most EP power systems are isolated from spacecraft ground through an isolation resistor. Electric thruster hollow cathodes generate a local dense ionosphere like plasma at a potential about 15V positive of cathode common. Few EP thruster plume ions hit spacecraft body, many fewer than ram ions in the ionosphere.

Cathode common potentials on EP spacecraft with gridded ion engines is determined by the voltage drop across the isolation resistor from electron current collected by the solar arrays from the plasma plume. On some EP spacecraft with Hall thrusters having insulating pole piece covers, cathode common potentials are determined by the by the voltage drop across the isolation resistor from plasma plume electrons collected by the solar arrays reduced by ion collection on the thruster body and spacecraft grounded conducting surfaces. On EP spacecraft with Hall thrusters with grounded bodies and conducting poles or external hollow cathodes outside the thruster magnetic field the cathode common potentials are close to the floating potential of the thruster body alone. Finally, if the Hall thrusters body were tied to cathode common and pole pieces were conducting, the cathode common voltage determined by the by the voltage drop across the isolation resistor from plasma plume electrons collected by the solar arrays reduced by ion collection on spacecraft grounded conducting surfaces.

The most important parameters that go into predicting cathode common voltages are solar array electron collection and the thruster body I-V curve. Both of these can be measured in laboratory tests. This data combined with a Hall thruster plume model such as HALLPLUME2D [11] and interactions codes, e.g. NASPAS can be used to choose the value of the resistor that both isolate the spacecraft chassis from thruster power system noise and reduce cathode common voltages to prevent undue sputter damage.

6. ACKNOWLEDGEMENTS

Copyright 2016 California Institute of Technology. U.S. Government sponsorship acknowledged. The research described in this paper was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration. The support of NASA's Space Technology Mission Directorate through the Solar Electric Propulsion Technology Demonstration Mission (SEP TDM) project is gratefully acknowledged. The authors would also like to acknowledge Dr. John T. Yim of NASA/GRC for the notional spacecraft surface element model.

7. REFERNCES

1. Rayman, M. D., Fraschetti, T. C., Raymond, C. A., Russell, C. T., Dawn: A Mission in Development for Exploration of Main Belt Asteroids Vesta and Ceres," Acta Astronautica 58 (2006 605-616).
2. Goebel, D. and Katz, I., Fundamentals of Electric Propulsion: Ion and Hall Thrusters, JPL Space Science and Technology Series, March 2008.
3. Brophy, J.R., Garner, C.E. & Mikes, S., Dawn Ion Propulsion System – Initial Checkout after Launch, AIAA-2008-4917.
4. Carruth, R. & Vaughn, J., Increasing Importance Of Material Electrical Interaction With The Space Environment, International Symposium on Materials in a Space Environment; 8th; 8 Jun. 2000; Arcachon, France.
5. Kerslake, W. R. & Ignaczak, L. R. Development and Flight History of the SERTII Spacecraft, JSR 30, pp 258-290, 1993.
6. Brophy, J. R., et al., Ion Propulsion System (NSTAR) DS1 Technology Validation Report, JPL Publication 00-10, October 2000
7. de Grys, K., et al., 4.5 kW Hall Thruster System Qualification Status, AIAA-2005-3682, 40th AIAA/ASME/SAE/ASEE Joint Propulsion Conference Proceedings, Tucson, AZ, 2005.
8. Patterson, M. and Benson, S., NEXT Ion Propulsion System Development Status and Performance, AIAA-2007-5199, Joint Propulsion Conference, July 2007.
9. Mikellides, I. G., Katz, I., Hofer, R. R., and Goebel, D. M., Magnetic Shielding of a Laboratory Hall Thruster, Part I: Theory and Validation, Journal of Applied Physics, 115, 043303 (2014).
10. Hofer, R. R., Goebel, D. M., Mikellides, I. G., and Katz, I., Magnetic Shielding of a Laboratory Hall Thruster Part II: Experiments, Journal of Applied Physics, 115, 043304 (2014).
11. Lopez Ortega, A., Huang, W., Mikellides, I.G., & Katz, I, A self –consistent numerical code for high-power Hall thruster plumes: improvements,

validation, and integration with spacecraft interactions code, 14th Spacecraft Charging Technology Conference, ESA/ESTEC, Noordwijk, NL, 04-08 April 2016.

12. Sekerak, M.J., Lopez-Ortega, A., Katz, I., Yim, J.T., Modeling of Plasma Plume Interaction with a Spacecraft Operating High-Power Hall Effect Thrusters, 52nd AIAA Joint Propulsion Conference, Salt Lake City, UT, July 25-27 2016.