

# Electrostatic Return of Contaminants

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## CONTRACTOR REPORT

### ELECTROSTATIC RETURN OF CONTAMINANTS

#### 1.0 INTRODUCTION

The return of ionized contaminant molecules to charged spacecraft surfaces is very important at all altitudes. It is especially important in geosynchronous and interplanetary environments since it may be the only mechanism by which contaminants can degrade a surface. This study completed an Electrostatic Return Flux (ESRF) model with application to all altitudes and spacecraft geometries. A User's manual is included as Appendix A.

The model has application to large solar sails, thermal control surfaces, optics and solar array surfaces. It has the capability to address deposition on and sputtering of spacecraft surfaces by return of the ionized contaminants. It does not predict the charge state of spacecraft surfaces since other models are already in existence to perform this task. Any charge can be applied to a surface as input to analyzing ionized contaminant return.

#### 1.1 Objectives

The objectives for this proposed study were:

- 1) Develop a model for calculating return flux of ionized contaminants to spacecraft surfaces.
- 2) Develop the capability to address spheres, plates and cylinders.
- 3) Develop the capability to be used in all ambient environments.
- 4) Complete a User's Manual for operation of the model.

All of these objectives were met to varying degrees. This report details the assumptions and algorithms used in developing the model.

## 2.0 MODEL COMPONENTS

### 2.1 Model Inputs

The following inputs are required to determine the electrostatic return flux of ionized molecules that have left a spacecraft. The attempt in this study was to determine which of these can be fixed for specific environments and which ones the user can change at will if they have sufficient knowledge. The danger is that some parameters are influenced by each other so that changing one may not represent the real world situation. The desire was to make sure the results of the ESR model are meaningful and do represent real situations.

The inputs are:

- Surface area of spacecraft
- Outgassing rates of spacecraft surface sources
- Temperature of surfaces that are outgassing
- Molecular species of spacecraft emitted gases
- Velocity and emission rates of gases emitted by vents or engines
- Spacecraft surface charge
- Debye sheath length
- Ionization cross sections of emitted gases from solar photons and ambient electrons
- Sputter yield of various surfaces impinged upon by various ions as a function of energy
- Solar flux of photons responsible for ionization
- Flux of ambient electrons responsible for ionization
- Temperatures of critical surfaces(thermal control, solar arrays, optics, etc)

#### 2.1.1 Surface area

This parameter is required to determine both the overall outgassing rate and the area that is deposited upon or sputtered by incoming ions. The user is able to input an equivalent area for a sphere, cylinder or a flat plate. If the user wants a quick answer, they can input a sphere with a surface area comparable to the spacecraft under analysis. Normally, only shadowed areas are charged negative or when the spacecraft is in eclipse. Because of this, the user may desire to input the area of the surfaces that are charged.

#### 2.1.2 Outgassing Rates

The outgassing rate of a surface in terms of mass per unit area per unit time is one option available for the user. It is also possible to submit a total mass loss over a period of time for a quick look analysis. In this case, the user can identify the percent of the mass returning relative to the total mass emitted during a specific period.

### 2.1.3 Surface Temperature

The surface temperature is required to calculate the emission velocity of the outgoing molecules from the surface. This will determine if they can escape the surface potential of the spacecraft before being ionized and returned by electrostatic attraction.

### 2.1.4 Molecular species

For outgassing, the molecular size is set at 500 amu (atomic mass units) unless the user wants to change it. For other gases such as vents or engine effluents, the user should identify these so calculations can be made as to the velocity and ionization cross section. The mass of the molecule is also required for sputtering estimates of the returning molecule on spacecraft surfaces. Simple gases such as oxygen, nitrogen, carbon dioxide etc, should not be a problem for deposition unless the charged surfaces are at low temperatures to trap these gases. For high negative potentials, these lighter molecular weight gases can cause sputtering.

### 2.1.5 Vent and Engine Emissions

The emission rates and the velocity of effluents from vents or engines are required to determine what fraction is returned to the spacecraft. Typically these gases have velocities on the order of 1 to 3 km per second. The model has values for these for the user to select. These include a small monopropellant hydrazine engine and a bipropellant engine utilizing monomethyl hydrazine fuel and nitrogen tetroxide oxidizer. In summary, the data presented includes major species and approximate velocities during steady state operation for:

- Monopropellant thruster
- Bipropellant thruster

The user will be able to modify these for specific thruster sources not included and there is a large variety of thrusters available. The ESR model is able to accept other engine molecular constituents and velocities as required by the user.

**2.1.5.1 Monopropellant Thruster** The constituents of a monopropellant, hydrazine thruster, plume exhaust are shown in Table 2.1. These were obtained from “Monopropellant Thruster Exhaust Plume Contamination Measurements,” R.K. Baerwald and R.S. Passamaneck, Sept 1977, AFRPL-TR-77-44. The effluents from this reference source are typical for this type of thruster. The relative amounts of each effluent depends on the engine design and operational mode.

This engine is a 25lb thrust engine with a flow rate of 49 grams per second. For a baseline in the ESR model, these fractions are used for a smaller 5 lb thruster with a flow rate of 10 grams per second. The user can input different monopropellant engine characteristics if required.

The  $N_2H_4$  comes from unburned hydrazine.

**Table 2.1. Monopropellant plume species**

Constituent	Mass in sample grams	Mass. %	Condensation temperature at 10 <sup>-4</sup> Pa Pressure
H <sub>2</sub>	2.8 x 10 <sup>-2</sup>	8.1	4K
N <sub>2</sub>	2.3 x 10 <sup>-1</sup>	66.5	26K
NH <sub>3</sub>	8.37 x 10 <sup>-2</sup>	24.2	101K
H <sub>2</sub> O	---	0.71	159K
N <sub>2</sub> H <sub>4</sub>	40 x 10 <sup>-6</sup>	0.012	165K

Velocity of emission=2300m/s.

**2.1.5.2 Bipropellant Thruster** The bipropellant engine used for a baseline in the ESR model comes from the properties of the 25 lb thrust vernier engines used on Shuttle. It uses MMH and N<sub>2</sub>O<sub>4</sub> as the fuel and oxidizer. Table 2.2 summarizes the species in the plume. This data is from, "Shuttle/Payload Contamination Evaluation Program (SPACE Program), MCR-77-106, contract NAS9-14767, April, 1977. Other references for the effluent from bipropellant engines contain the same constituents but vary in the relative amounts because of different engine design and firing pulse widths. Variations to the model baseline can be inputted by the user.

**Table 2.2. Bipropellant Engine Plume Effluents**

Constituent	Mole Fraction	Molecular Weight
H <sub>2</sub> O	0.328	18
N <sub>2</sub>	0.306	28
CO <sub>2</sub>	0.036	44
O <sub>2</sub>	0.0004	32
CO	0.134	28
H <sub>2</sub>	0.17	2
MMH-NO <sub>3</sub>	0.002	46

Velocity of emission = 3505 m/s.

For this engine, the flow rate is 40.8 grams per second. For other size engines, the flow rate and perhaps the percent of each plume constituent will need adjusting by the user. For the ESR model, these fractions and velocities in Table 2.2 are used for a small engine, approximately a 5 lb thrust, with a flow rate of 8 grams per second.

The MMH-HNO<sub>3</sub> is the primary specie that deposits on surfaces at normal temperatures.

### 2.1.6 Spacecraft Surface Charge

Spacecraft at geosynchronous orbit (GEO) experience negative voltages from a few volts to thousands of volts. The user can input any voltage desired. A quiescent level would be between one to ten volts, medium activity would be a few hundred volts, and high activity would be a thousand volts or more.

The ESR model will not calculate the surface potentials. The surface voltages should be calculated by other existing models such as NASCAP GEO, NASCAP POLAR, NASCAP-2K or other surface charge prediction models. Flight data can also be used.

### 2.1.7 Debye Sheath

The Debye sheath is the distance from the spacecraft where the spacecraft potential is shielded from the created ions to 1/e of the near surface value.

The Debye sheath length can be calculated from the atmospheric conditions, if desired by the user, or they can select values. The values used will have to be related to the spacecraft charge.

The Debye sheath is given by

$$\lambda = 69(T_e/N_e)^{1/2}, \text{ meters}$$

where,

$T_e$ =electron temperature in degrees Kelvin

$N_e$ =electron density in number per cubic meter

And by

$$\lambda = 7.34 \times 10^3 (T_e/N_e)^{1/2}, \text{ meters}$$

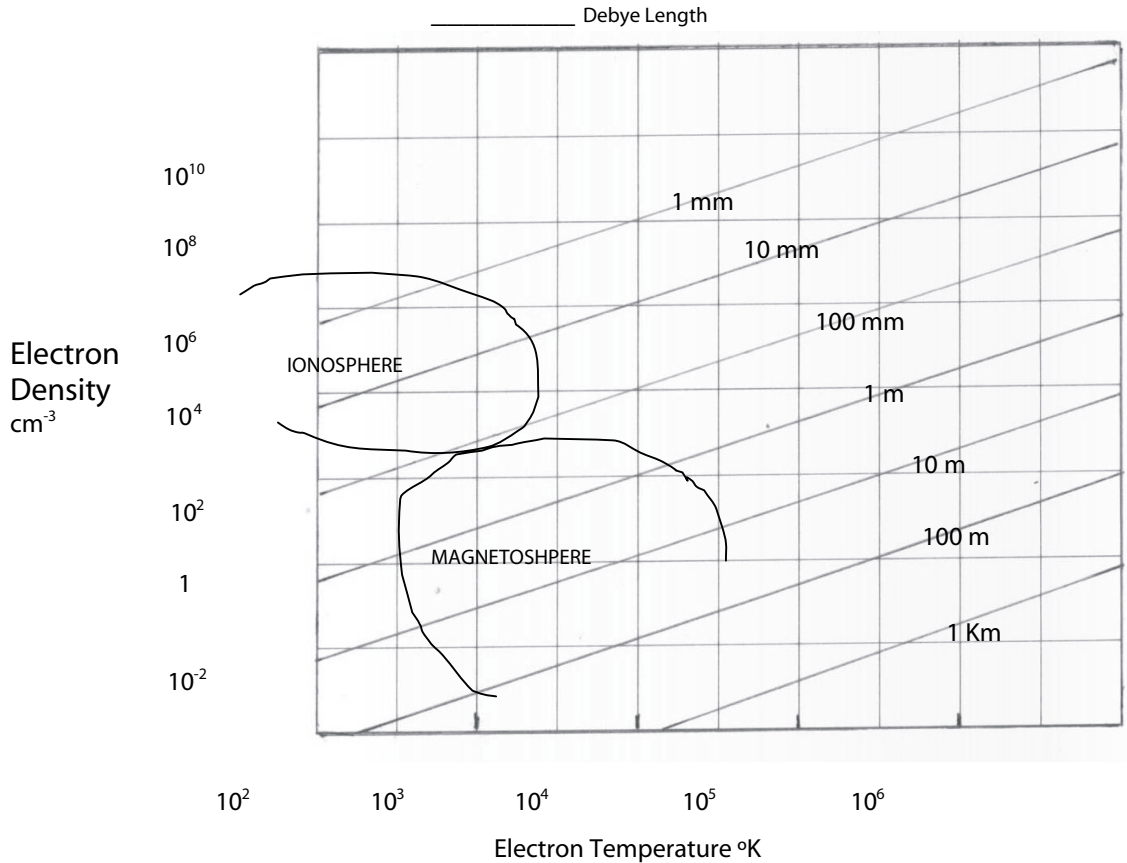
where ,

$T_e$ =electron temperature in eV

$N_e$ =electron density per cubic meter.

The Debye sheath for quiescent periods is on the order of tens of meters at GEO, a few centimeters at LEO, and for active periods on the order of hundreds of meters at geosynchronous orbit.

Figure 2.1 is from “Charging of Large Structures in Space with Application to the Solar Sail Spacecraft”, Hill, Jay R. and Whipple Jr., Eldon C., J spacecraft, Vol. 22, No. 3, May-June-1985. This shows the range of Debye lengths for the range of ambient conditions. Other references show nearly the same results.



**Figure 2.1. Debye Length as Function of Plasma Density and Temperature**

For low Earth orbit (LEO), the Debye sheath is only a few centimeters. The user can apply the model to LEO for surfaces such as solar array interconnects that may be at a negative voltage on the order of 50 to 150 volts. The main concern here would be sputtering of the interconnect material.

It became apparent the Debye sheath is very hard to determine for a general user. Tables 2.3 and 2.4 were developed from existing data to aid the user in selecting an appropriate Debye sheath if needed. If the user has previously run a model to determine the charge on the spacecraft, they will have data available on the number density and energy of the ambient electrons so that the Debye length can be calculated in the model.

**Table 2.3. Average Parameters for Energetic Electrons\***

Parameter	ATS-5	ATS-6	SCATHA
Number density cm <sup>-3</sup>	0.80	1.06	1.09
Average Temperature, Kev	1.85	2.55	2.49
Debye length, meters	353	360	351

**Table 2.4. Recommended Worst Case Conditions at Geosynchronous for Energetic Electrons\***

<b>Number density cm-3</b>	<b>1.12</b>
<b>Temperature, Kev</b>	<b>12.0</b>
<b>Debye length, meters</b>	<b>760</b>

\* Data taken from: NASA Technical Paper 2361, "Design Guidelines for Assessing and Controlling Spacecraft Charging Effects" September 1984.

### **2.1.8 Ionization Cross Sections**

Ionization cross sections of the emitted gases by solar photons and ambient electrons are selected from existing data for use in the model. The cross sections and the flux of the photons and electrons is also required for the ionization rate and is selected from existing data. Reference Cauffman, David, P., "Ionization and Attraction of Neutral Molecules to a Charged Spacecraft", NTIS AD-76641, 9 Aug. 1973, shows that reasonable values of the ionization frequency for outgassing type molecules is  $1.2 \times 10^{-4}$  ionizations/molecule per second for solar photons and  $2.2 \times 10^{-4}$  ionizations/molecule per second for electrons. This was based on a photon cross section of  $1.2 \times 10^{-20} \text{m}^2$  and an electron ionization cross section of  $5.2 \times 10^{-20} \text{m}^2$ . These values are used as the baseline ionization. The user can adjust these if necessary if better data is available on the ionization cross sections and the flux of photons and electrons. The ionization frequency input can be called in the model input section and modified as necessary.

Table 2.5 shows data obtained from "The Photochemistry of Atmospheres, Earth, the other Planets and Comets", Levine, J.S., pages 165-278, Academic Press 1985.

**Table 2.5. Ionization Cross Sections of N<sub>2</sub>, O and O<sub>2</sub>.**

Wavelength Range in Angstroms	Photon Flux x10 <sup>9</sup> cm <sup>2</sup> /sec	Ionization Cross Section x10 <sup>-18</sup> cm <sup>2</sup>		
		N <sub>2</sub>	O	O <sub>2</sub>
100-150	0.34	2.3	3.5	3.6
150-200	4.9	5.4	6.0	7.3
200-250	3.7	8.2	7.6	10.5
250-300	8.0	10.1	8.8	13.6
300-350	19.0	14.6	9.7	17.2
350-400	4.0	17.5	10.7	18.2
400-450	0.9	21.1	11.2	19.4
450-500	2.0	21.9	11.6	21.2
500-550	1.5	24.5	11.9	24.1
550-600	7.0	22.4	11.9	25.0
600-650	5.0	23.2	12.2	25.9
650-700	0.45	25.1	10.0	22.1
700-750	1.0	23.2	8.0	23.8
750-800	4.8	16.8	4.2	9.3
800-850	4.9	0.0	4.4	6.1
850-900	13.0	0.0	4.2	4.7
900-950	12.5	0.0	2.0	9.3
950-1000	17.5	0.0	0.0	12.2

These show the photon flux from the sun and the ionization cross sections for N<sub>2</sub>, O, and O<sub>2</sub>. By integrating over the spectrum, the ionization cross section can be expressed by:

Number of ions produced per second per cubic centimeter of species i =  $N_i \int F(\lambda) \sigma_i(\lambda) d\lambda$   
where.

$N_i$  = neutral gas density for species i,

$F(\lambda)$  = solar photon flux at wavelength  $\lambda$

$\sigma_i(\lambda)$  = photoionization cross section of species i at wavelength  $\lambda$ .

The results are

$$I(N_2) = 9.7 \times 10^{-7} [N_2]$$

$$I(O) = 7.8 \times 10^{-7} [O]$$

$$I(O_2) = 1.5 \times 10^{-6} [O_2]$$

Where  $[N_2]$ ,  $[O]$ ,  $[O_2]$  are densities of each species.

For other simple gases, the values for masses closest to these three is used. As better data is available, other cross sections can be used.



### 2.1.9 Sputter Yield of Returning Ions

The sputtering caused by the returning ions is input from existing data. The user is encouraged to input their own sputtering yield for the material of concern.

The sputtering by returning energetic ions requires detailed information on the ion and the material being impinged upon. Since this detailed data does not exist or is unknown for the combinations that may be used, a set of data was accumulated to aid the user in selecting sputtering yields. The following references, "Theory of Sputtering. I. Sputtering Yield of Amorphous and Polycrystalline Targets", Sigmund, P, Physical Review, Volume 184, Number 2, August 1969 and "A review of Spacecraft Material Sputtering by Hall Thruster Plumes", Boyd, I.D. and Falk, M., AIAA paper 2001-3353, were used for model inputs.

The data used was for various ions impinging on different targets in the energy range from 100 to 1000eV. The ion-target combinations reviewed are:

Xe+♦ Cu	Xe+♦ Ag	Xe+♦ Pd	Ar+♦ Au
Kr+♦ Cu	Kr+♦ Ag	Kr+♦ Pd	Ar+♦ Ge
Ar+♦ Cu	Ar+♦ Ag	Ar+♦ Pd	Ar+♦ Si
Ne+♦ Cu	Ne+♦ Ag	Ne+♦ Pd	

These give a range of data, which is surprisingly similar for the different combinations. Figure 2.2 shows the range of the sputtering yield versus energy and is the average over all the values. The spread on the curve data is about 0.5 units on the sputtering yield axis. Actual values should be used whenever possible.

The sputtering yield is defined as the number of target atoms removed per incident ion. It was determined the most applicable equation would be the calculation of the surface erosion rate. This can be expressed by

$$dx/dt = S(M/d)J\cos\emptyset$$

where,

$dx/dt$  = surface depth erosion rate, cm/s

$S$  = sputtering yield , atoms/ion

$M$  = weight of one target atom, grams

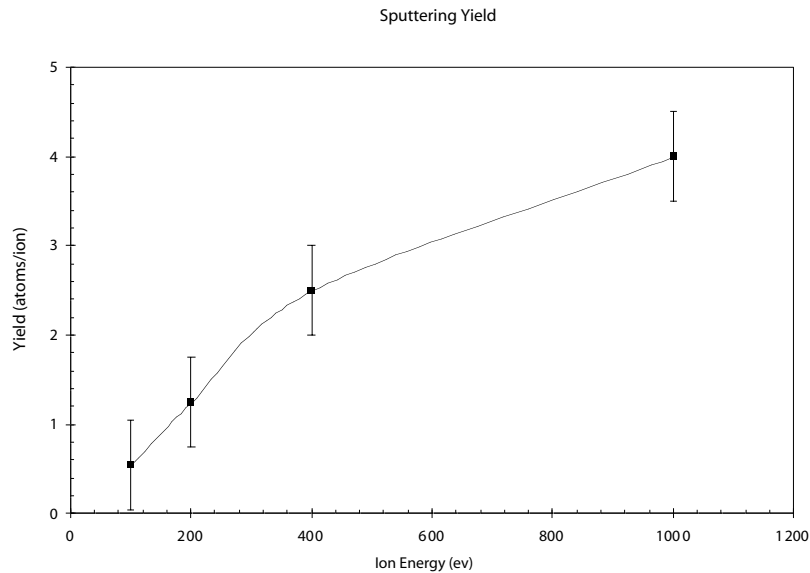
$d$  = density of material impinged upon, grams/cm<sup>3</sup>

$J$  = ion flux, ions/cm<sup>2</sup>.second

$\emptyset$  = angle of incidence

The angle of incidence is assumed to be normal so that  $\cos\emptyset=1$ .

This equation then allows the model to determine the amount of material depleted from ion sputtering effects.



**Figure 2.2-Sputtering yield versus ion energy**

### 2.1.10 Solar Photon Flux

The spectral photon flux of the solar photons is used to calculate ionization by the photons in conjunction with the ionization cross sections associated with each specie to a range of solar photon energies. The values used are shown in the Table 2.5 above.

### 2.1.11 Ambient Electron Flux

The flux of ambient electrons is compiled from existing data on the geosynchronous environment. Some of these values are shown in the Figure 2.1. If the electron density and resulting flux is different from the baseline in the model, the Debye length can be calculated for these conditions and the ionization rate modified as required.

### 2.1.12 Critical Surface Temperature and Type

The critical surfaces that are charged negatively and susceptible to contamination may have a resulting deposited layer from the return of the ions. The temperature of the critical surface is important in determining if the returning ions will stay. Also important is the situation where the critical surface may be sunlit. The sun light can cause increased deposition to occur or allow deposition to occur when it normally would not without solar exposure. This is true for complex outgassing molecules. The model assumes a unit sticking coefficient for the deposited species.

### 3.0 MODEL STRUCTURE

The model is structured so that the predictions are made for an instant in time. In order to apply integrated predictions, the user has to multiply by the time the spacecraft is under the conditions used. As the ambient conditions vary, additional runs will be required to incorporate these periods into a total mission evaluation.

#### 3.1 Conductor Geometry Effect on Electric Fields

At the point of ionization, a molecule immediately comes under the influence of the spacecraft generated electric field. In previous ESR work<sup>1</sup>, we developed the mathematics of electrostatic return of contaminants based on the assumption that the spacecraft could be modeled as a charged conducting sphere. The question that naturally arose was how other common spacecraft geometries would affect the electrostatic return calculations. The question can be answered by examining how the electric fields of different common spacecraft geometries vary as a function of shape. In this section, the electric fields generated by charged conducting spheres, cylinders, and rectangular plates are compared.

##### 3.1.1 Electric Field of a Sphere

The electric field produced by a uniformly charged conducting sphere at a point external to the sphere is equivalent to the electric field produced by a point charge with the same total charge located at an identical radius.

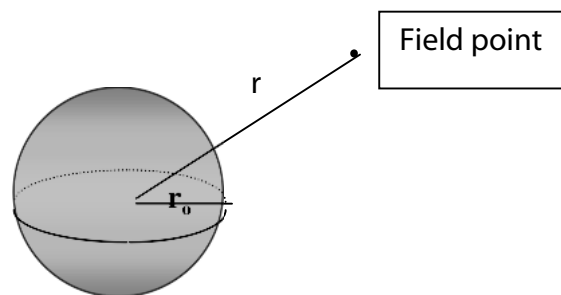


Figure 3.1 – Charged Conducting Sphere

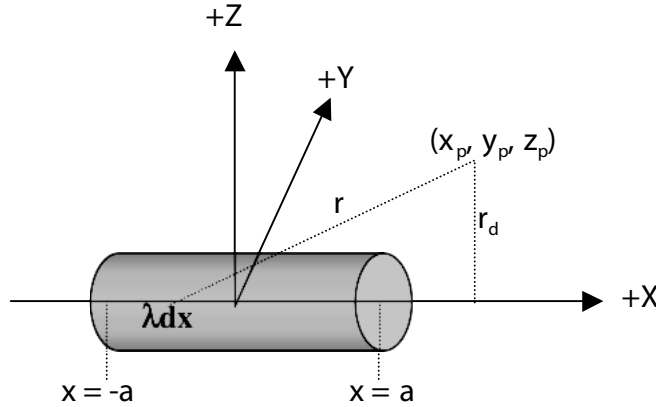
The electric field surrounding a uniformly charged sphere is given by

$$E = Q / (4\pi\epsilon_0 r^2) \quad \text{Eq. 3.1.1}$$

where,  
E = field strength (N/C or V/m)  
Q = total charge on sphere (C)  
 $\epsilon_0$  = permittivity of free space =  $8.85 \times 10^{-12} \text{ C}^2/\text{N}\cdot\text{m}^2$   
r = radial distance (m) where  $r > r_0$

### 3.1.2 Electric Field of a Cylinder

The electric field of a conducting cylinder can be approximated as a finite line charge. Figure 3.2 shows the mathematical setup used for the derivation of the cylinder electric field.



**Figure 3.2 – Charged Conducting Cylinder Geometry**

The electric field at point  $(x_p, y_p, z_p)$ , due to a uniformly charged conducting cylinder, has two orthogonal components, one radial and the other axial. For this analysis, the coordinate system has been arbitrarily chosen such that the X-axis lies along the center axis of the cylinder with the origin at the center of the cylinder. The cylinder of radius  $r_0$  extends from  $x=-a$  to  $x=a$ . We can equate the electric field generated by a conducting cylinder to a line charge of equivalent length with the same total charge. Any field point external to the cylinder volume can be designated by the Cartesian coordinates  $(x_p, y_p, z_p)$ . The radial distance ( $r_d$ ) from the cylinder centerline to the field point is

$$r_d = (y_p^2 + z_p^2)^{0.5} \quad \text{Eq. 3.2.1}$$

The distance ( $r$ ) from an incremental element of the line charge along the x-axis to the field point is

$$r = [(x - x_p)^2 + y_p^2 + z_p^2]^{0.5} \quad \text{Eq. 3.2.2}$$

The electric field axial component ( $E_a$ ) at the field point  $(x_p, y_p, z_p)$  due to the incremental line charge element  $\lambda dx$ , where  $\lambda$  is the linear charge density is

$$dE_a = (k \lambda dx / r^2) (x - x_p) / r \quad \text{Eq. 3.2.3}$$

where,  $k = 1 / (4\pi\epsilon_0)$

$\epsilon_0 =$  permittivity of free space  $= 8.85 \times 10^{-12} \text{ C}^2/\text{N}\cdot\text{m}^2$

$\lambda =$  linear charge density  $\text{C}/\text{m}$

The electric field axial component at field point  $(x_p, y_p, z_p)$  is

$$E_a = \int_{-a}^{+a} (k\lambda dx / r^2) (x - x_p) / r = \int_{-a}^{+a} k\lambda(x - x_p) dx / [(x - x_p)^2 + y_p^2 + z_p^2]^{3/2} \quad \text{Eq. 3.2.4}$$

Evaluating the integral for the axial component yields

$$E_a = k\lambda \left[ 1 / (r_d^2 + (a+x_p)^2)^{0.5} + 1 / (r_d^2 + (a-x_p)^2)^{0.5} \right] \quad \text{Eq. 3.2.5}$$

The electric field radial component ( $E_r$ ) at the field point ( $x_p, y_p, z_p$ ) due to the incremental line charge element  $\lambda dx$ , where  $\lambda$  is the linear charge density is

$$dE_r = (k \lambda dx / r^2) r_d / r \quad \text{Eq. 3.2.6}$$

where,  $k = 1 / (4\pi\epsilon_0)$

$\epsilon_0 =$  permittivity of free space  $= 8.85 \times 10^{-12} \text{ C}^2/\text{N}\cdot\text{m}^2$

$\lambda =$  linear charge density  $\text{C}/\text{m}$

The electric field radial component at field point ( $x_p, y_p, z_p$ ) is

$$E_r = \int_{-a}^{+a} (k\lambda dx / r^2) (r_d / r) = \int_{-a}^{+a} k\lambda r_d dx / [(x-x_p)^2 + y_p^2 + z_p^2]^{3/2} \quad \text{Eq. 3.2.7}$$

Evaluating the integral for the radial component yields

$$E_r = (k\lambda / r_d) \left[ (a-x_p) / (r_d^2 + (a-x_p)^2)^{0.5} + (a+x_p) / (r_d^2 + (a+x_p)^2)^{0.5} \right] \quad \text{Eq. 3.2.8}$$

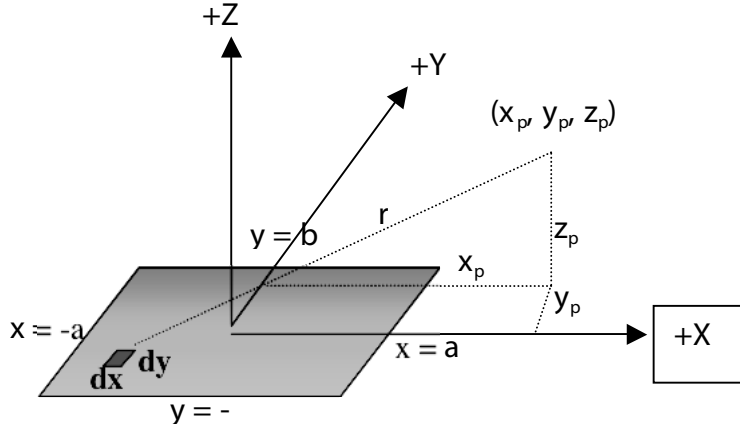
where,  $k = 1 / (4\pi\epsilon_0)$

$\epsilon_0 =$  permittivity of free space  $= 8.85 \times 10^{-12} \text{ C}^2/\text{N}\cdot\text{m}^2$

$\lambda =$  linear charge density  $\text{C}/\text{m}$

### 3.1.3 Electric Field of a Plate

The equations for the electric field components at a point near a uniformly charged rectangular conductor is developed in this section. We arbitrarily define the rectangular conductor as being in the XY plane with its center at the origin. The plate is defined as extending from  $-a$  to  $a$  in the X dimension and from  $-b$  to  $b$  in the Y dimension. Figure 3.3 shows the mathematical setup for the plate analysis.



**Figure 3.3 – Charged Conducting Rectangular Plate Geometry**

The electric field at point  $(x_p, y_p, z_p)$ , due to a uniformly charged conducting ,

$$r = [(x - x_p)^2 + (y - y_p)^2 + (z - z_p)^2]^{0.5} \quad \text{Eq. 3.3.1}$$

The x, y, and z components of the electric field at the field point  $(x_p, y_p, z_p)$  due to incremental charge element  $\sigma dx dy$ , where  $\sigma$  is the area charge density is

$$dE_x = (k \sigma dx dy / r^2) (x - x_p) / r \quad \text{Eq. 3.3.2}$$

$$dE_y = (k \sigma dx dy / r^2) (y - y_p) / r \quad \text{Eq. 3.3.3}$$

$$dE_z = (k \sigma dx dy / r^2) (z - z_p) / r \quad \text{Eq. 3.3.4}$$

where,  $k = 1 / (4\pi\epsilon_0)$

$\epsilon_0 =$  permittivity of free space  $= 8.85 \times 10^{-12} \text{ C}^2/\text{N}\cdot\text{m}^2$

$\sigma =$  area charge density  $\text{C}/\text{m}^2$

The electric field orthogonal components are

$$E_x = k \sigma \int_{-b}^b \int_{-a}^a (x - x_p) dx dy / [(x - x_p)^2 + (y - y_p)^2 + (z - z_p)^2]^{3/2} \quad \text{Eq. 3.3.5}$$

$$E_y = k \sigma \int_{-b}^b \int_{-a}^a (y - y_p) dx dy / [(x - x_p)^2 + (y - y_p)^2 + (z - z_p)^2]^{3/2} \quad \text{Eq. 3.3.6}$$

$$E_z = k \sigma \int_{-b}^b \int_{-a}^a (z - z_p) dx dy / [(x - x_p)^2 + (y - y_p)^2 + (z - z_p)^2]^{3/2} \quad \text{Eq. 3.3.7}$$

where,  $k = 1 / (4\pi\epsilon_0)$   
 $\epsilon_0 =$  permittivity of free space  $= 8.85 \times 10^{-12} \text{ C}^2/\text{N}\cdot\text{m}^2$   
 $\sigma =$  area charge density  $\text{C}/\text{m}^2$

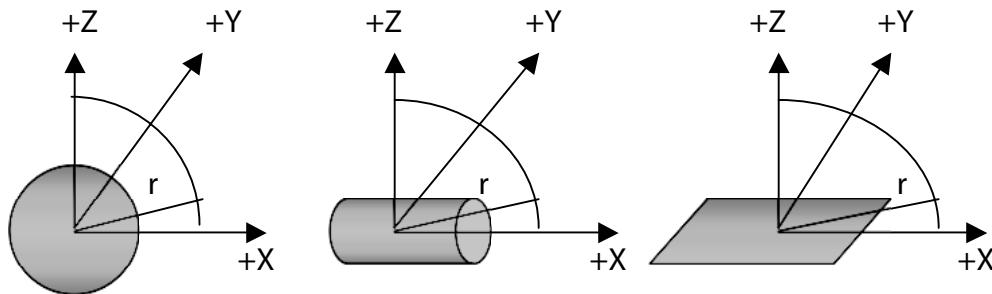
The double integrals in Equations 3.3.5, 3.3.6 and 3.3.7 are messy to evaluate. For this reason, we have chosen to solve these integrals using standard numerical techniques.

### 3.1.4 Electric Field Comparisons

It is intuitive that at large distances, any shaped charged conductor will look like a point charge. It is equally intuitive that at very close distances (relative to the overall dimensions of a surface), a uniformly charged surface will look like an infinite plane. The interesting electric field shape comparisons occur at distances that are on the order of the surface geometry dimensions. This is the range of distances where we expect to see geometry effects on the electric field.

To compare the electric fields for different geometries in this range, we calculate the electric field strength for a 1-meter diameter sphere, a 1-meter long cylinder and a 1-meter square plate. In the calculations, all three geometries are given the same total charge. The electric field strength is computed and compared for points along a 90 degree arc of constant radius.

Figure 3.4.1 shows the arc of computed electric field points relative to the different conductor geometries.



**Figure 3.4.1 – Electric Field Comparison Geometries**

The electric field strengths were computed for different radial values along the arc from 0 to 90 degrees with 0 degrees being coincident with the +Z-axis and 90 degrees being coincident with the +X-axis. For comparison purposes, the computed electric fields were normalized to the field value of the sphere at the given radial distance. Figure 3.4.2 shows the electric field comparison for the different geometric shapes at  $r = 1$  meter.

Normalized E Fields for Different Conductor Geometries (R=1 m)

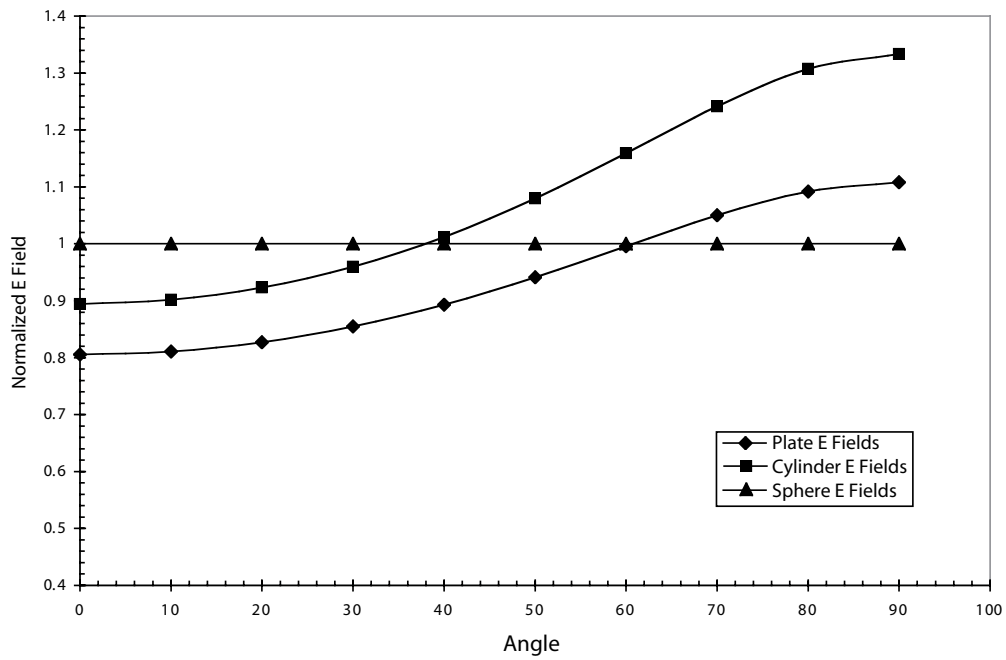


Figure 3.4.2 – Electric Field Comparison at Radial Distance of 1 meter

Figure 3.4.3 shows the electric field comparison for the different geometric shapes at r = 2 meters.

Normalized E Fields for Different Conductor Geometries (R=2 m)

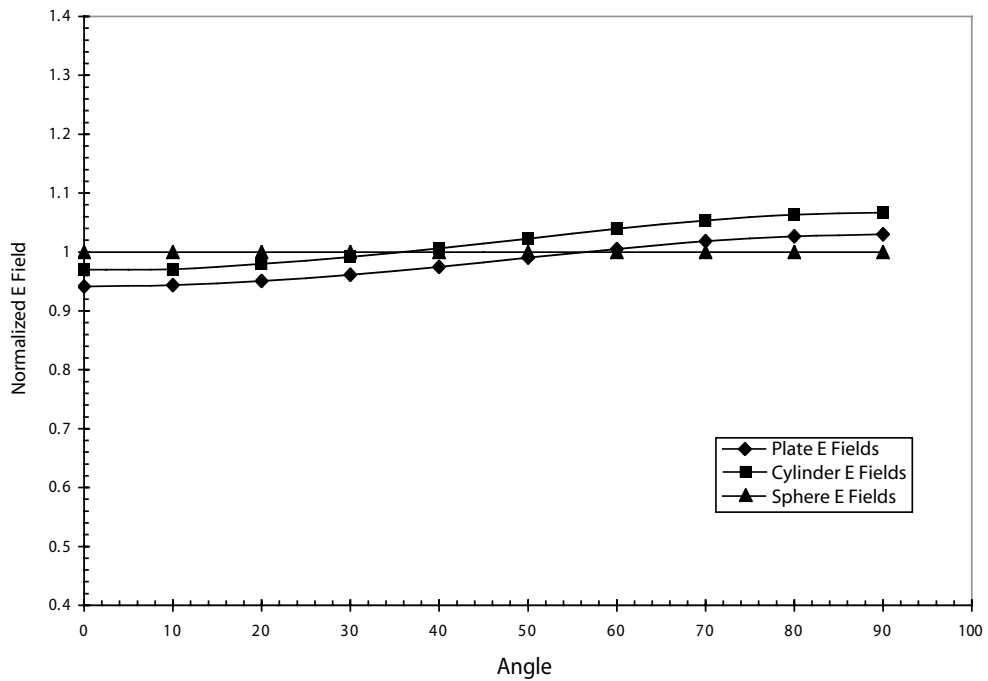


Figure 3.4.3 – Electric Field Comparison at Radial Distance of 2 meters



Figure 3.4.3 clearly shows that once the radial distance is at or beyond about 2 times the characteristic dimension of the conductor, the conductor shape has little effect on the electric field, and probably any shape can safely be modeled as a sphere. Figure 3.4.2 shows that the difference between the electric field of the sphere, a cylinder or plate, can differ as much as 30% at a distance roughly equal to the characteristic dimension of the geometry. Only portions of the spatial field vary by the maximum amount. How much error this causes in the ESR calculations would be a function of what fraction of the ESR molecules are ionized in these regions. The error would manifest itself as errors in the return energy of the ions from these regions. The magnitude of the Debye sheath will also determine whether or not the error is significant. If the Debye sheath is very much larger than the characteristic dimension, then only a tiny fraction of the total number of returning ions will be generated in the regions where the electric field is skewed. It is interesting to note that the electric field of a cylinder or plate is less than that produced at the same radius by a sphere in some places, and more in other places (see Figure 3.4.2). So, for a uniformly outgassing spacecraft, the errors would tend to cancel each other out and the average energy of the returning ions would be nearly the same as for the sphere.

For most anticipated uses of the ESR analysis tool, substituting a charged sphere for a cylinder or plate would cause minimal errors in the ESR calculations. The magnitude of errors in most cases would be much less than the normal uncertainties in other inputs such as the outgassing and ionization rates. However, there may be special cases where this type of geometry substitution is unacceptable. From a calculational standpoint, it would not be much more difficult to allow the user to generate a collection of charged spheres, plates and cylinders, because electric fields generated by multiple charges can be summed as vector additions. The concern is in regards to the increase in complexity of the required user inputs and whether that increase in complexity is justified by a small increase in accuracy. To use a range of cylinders, plates, etc, complex geometry modeling will be required to calculate gas densities around the spacecraft. In light of the findings of the comparison of the fields for the different shapes, it does not appear to be worth the very complex effort. The uncertainties in other input parameters are likely to be a greater source of error than using a sphere approximation.

The little difference between the electric fields, for the different surfaces, was a surprise. A much greater difference was expected.

There are two areas where substituting a sphere will not be sufficient. The first is for a large rectangular surface, such as a solar sail, where the Debye length is less than the dimensions of the sail. The other is at LEO, where the Debye length is only a few centimeters. Both of these situations can be treated the same mathematically. The approach used is to treat the surface as an infinite plate. This will allow calculation of the gas density to be performed without complicated spacecraft geometry modeling.

In summary, the model is made up of a sphere for most applications and an infinite plate for very large surfaces at synchronous or interplanetary and at LEO. This will allow the whole range of spacecraft to be analyzed.

## 4.0 TWO MODEL APPROACH

The computation of electrostatic return of contaminants to a spacecraft requires knowledge of the spacecraft, its contaminant source characteristics and its space environment. Many of the computational inputs can be difficult if not impossible to accurately know during the design phase of a spacecraft. It is envisioned that many users will be interested in using a quick look tool which could determine, with a minimum of difficult inputs, whether or not ESR is likely to be a problem for their spacecraft. For a few users, especially those for whom the quick look analysis showed a potential problem, the capability to do a more detailed analysis would be necessary. In order to accommodate both types of users, we have decided to provide the user with a quick look tool and a more detailed analysis tool in the same code.

The ESR tool provides two different calculational routines to compute the electrostatic return of contaminants based on the user's needs.

The first computational routine assumes a spherical electric field for the charged spacecraft. This routine should be used for cases where the Debye sheath is on the order of, or larger than the spacecraft characteristic dimensions. It can also be used for cases where the Debye sheath is smaller than the spacecraft dimensions as long as the spherical electric field is a reasonable assumption for the actual spacecraft being modeled.

The second computational routine assumes a planar electric field. This routine should be used for cases where the Debye sheath is considerably smaller than the spacecraft dimensions. A classic case where this routine should be used would be the computation of ESR to solar arrays in low earth orbit where the Debye sheath is much smaller than the solar array dimensions.

### 4.1 ESR Due to Spherical Conductor Electric Field

The Debye length (or sheath) is the distance over which the plasma between the ion and spacecraft reduces (by shielding) the strength of the spacecraft electric field by a factor of  $1/e$ . The effects of the Debye length can not be implemented in the ESR calculations as a post calculational correction. The effect of the Debye length is a continuous and incrementally cumulative effect that must be included within the computational integrals.

The electric field surrounding a uniformly charged sphere is given by:

$$E = Q/(4\pi\epsilon_0 r^2) \quad \text{Eq. 4.1-1}$$

where,  $E$  = field strength (N/C or V/m)  
 $Q$  = spacecraft charge (C)  
 $\epsilon_0$  = permittivity of free space =  $8.85 \times 10^{-12}$  C<sup>2</sup>/N-m<sup>2</sup>  
 $r$  = radial distance (m)

However, in the presence of plasma, a shielding effect occurs. The plasma shields the ions from the potential of the spacecraft, effectively reducing the electric field as a function of distance from the spacecraft by a factor of  $e^{-r/\lambda}$ .

$$E = Q/[4\pi\epsilon_0 r^2 e^{(r-r_0)/\lambda}] \quad \text{Eq. 4.1-2}$$

where,  
 E = field strength (N/C or V/m)  
 Q = spacecraft charge (C)  
 $\epsilon_0$  = permittivity of free space =  $8.85 \times 10^{-12} \text{ C}^2/\text{N}\cdot\text{m}^2$   
 r = radial distance (m)  
 $r_0$  = spacecraft radius (m)  
 $\lambda$  = the Debye length (m)

#### 4.1.1 Escape Energy Required for Ion

The energy or work required to move a charge (q) from point B to point A in an electric field is:

$$\text{Work} = q \int_{r_b}^{r_a} E \, dr = q \int_{r_b}^{r_a} Q/[4\pi\epsilon_0 r^2 e^{(r-r_0)/\lambda}] \, dr \quad \text{Eq. 4.1-3}$$

Simplifying:

$$\text{Work} = qQ/(4\pi\epsilon_0) \int_{r_b}^{r_a} 1/[r^2 e^{(r-r_0)/\lambda}] \, dr \quad \text{Eq. 4.1-4}$$

where,  
 Work = energy in Joules  
 Q = spacecraft charge (C)  
 q = charge in spacecraft electric field (C)  
 $\epsilon_0$  = permittivity of free space =  $8.85 \times 10^{-12} \text{ C}^2/\text{N}\cdot\text{m}^2$   
 $r_b$  = radial distance to point B (m)  
 $r_a$  = radial distance to point A (m)  
 r = radial distance (m)  
 $r_0$  = spacecraft radius (m)  
 $\lambda$  = the Debye length (m)

It should be noted that the electric field is measured from the center of the spherical spacecraft, but the Debye length is measured from the surface of the spacecraft, hence the  $r-r_0$ .

Unfortunately, the integral in equation 4.1-4 has a messy solution which contains an infinite series involving factorials. Because of this, in the ESR tool, we have chosen to solve the integral using standard numerical techniques.

Work can also be expressed in terms of potential difference

$$\text{Work} = q(V_B - V_A) = qQ/(4\pi\epsilon_0) \int_{r_b}^{r_a} 1/[r^2 e^{(r-r_0)/\lambda}] \, dr \quad \text{Eq. 4.1-5}$$

### 4.1.2 Absolute Potential

The absolute potential at a point in the electric field can be determined by letting  $r_a$  go to infinity where  $V_A=0$ . The absolute potential at the surface of the sphere ( $r_0$ ) is

$$V_B = Q/(4\pi\epsilon_0) \int_{r_0}^{\infty} 1/[r^2e^{(r-r_0)/\lambda}] dr \quad \text{Eq. 4.1-6}$$

Solving for the spacecraft charge

$$Q = V_B 4\pi\epsilon_0 / \int_{r_0}^{\infty} 1/[r^2e^{(r-r_0)/\lambda}] dr \quad \text{Eq. 4.1-7}$$

### 4.1.3 Ion Return Impact Velocity

The kinetic energy of a neutral molecule leaving the spacecraft is given by:

$$KE = mv^2/2 \quad \text{Eq. 4.1-8}$$

where,  
 E = kinetic energy (j)  
 m = mass of molecule (kg)  
 v = thermal velocity of molecule (m/s)

Assuming that the kinetic energy of the neutral molecule is not changed by a collision prior to ionization, then at the instant of ionization, the new ion has a kinetic energy given by Eq 4.1-8 and immediately comes under the influence of the electric field given by Eq 4.1-2. If we let  $r_i$  be the radial value at which ionization occurs, then the work in joules required to move the newly created ion from  $r_i$  to infinity (using Eq 4.1-4) is given by:

$$\text{Work} = qQ/(4\pi\epsilon_0) \int_{r_i}^{\infty} 1/[r^2e^{(r-r_0)/\lambda}] dr \quad \text{Eq. 4.1-9}$$

where,  
 Work = energy required to escape spacecraft potential in Joules  
 Q = spacecraft charge (C)  
 q = ion charge (C)  
 $\epsilon_0$  = permittivity of free space =  $8.85 \times 10^{-12} \text{ C}^2/\text{N}\cdot\text{m}^2$   
 $r_i$  = radial distance to ionization point (m)  
 r = radial distance (m)  
 $r_0$  = spacecraft radius (m)  
 $\lambda$  = the Debye length (m)

The work computed in Eq 4.1-9 represents the amount of energy the newly created ion will need to escape the electric field of the spacecraft. Consequently, if the energy computed by Eq 4.1-9 is less than the kinetic energy computed by Eq 4.1-8, then the ion will escape the electric field of the spacecraft, but if the computed kinetic energy is less than the energy calculated in Eq 4.1-9, then the ion will be trapped by the spacecraft electric field and will subsequently be drawn back to the charged spacecraft. The radial

distance at which the energy required for escape is equal to the kinetic energy of the molecule is computed by the ESR code and is referred to as the critical ionization radius,  $r_c$ . Any molecules, which become ionized at a radial distance less than  $r_c$ , will return to the spacecraft.

For ions which lack sufficient energy to escape the electric field of the spacecraft, the velocity at spacecraft impact of the returning ion can be computed by equating the potential energy of the ion at the point of ionization to the kinetic energy at impact.

$$mv_s^2/2 = qQ/(4\pi\epsilon_0) \int_{r_0}^{r_i} 1/[r^2 e^{(r-r_0)/\lambda}] dr \quad \text{Eq. 4.1-10}$$

where,  $Q$  = spacecraft charge (C)  
 $q$  = ion charge (C)  
 $\epsilon_0$  = permittivity of free space =  $8.85 \times 10^{-12}$  C<sup>2</sup>/N-m<sup>2</sup>  
 $r_0$  = radius of spacecraft (m)  
 $r_i$  = radial distance to ionization point (m)  
 $m$  = mass of ion (kg)  
 $v_s$  = spacecraft impact velocity of ion (m/s)  
 $\lambda$  = the Debye length (m)

Solving Eq 4.1-10 for the impact velocity gives:

$$v_s = [qQ/(2\pi\epsilon_0 m) \int_{r_0}^{r_i} 1/[r^2 e^{(r-r_0)/\lambda}] dr]^{0.5} \quad \text{Eq. 4.1-11}$$

Clearly the solution to Eq 4.1-11 is a function of the ionization radius. Therefore, molecules that are ionized close to the spacecraft will return with a smaller impact velocities than molecules that are ionized near the critical ionization radius,  $r_c$ . The ESR code computes the maximum impact velocity for the returning ions.

#### 4.1.4 Electrostatic Return Mass

If we assume that the spherical spacecraft uniformly outgasses at a rate of  $F_o$  (kg/m<sup>2</sup>/s), then the total mass leaving the spacecraft is described by:

$$M_o = F_o 4\pi r_o^2 \quad \text{Eq. 4.1-12}$$

where,  $M_o$  = total mass leaving spacecraft (kg/s)  
 $F_o$  = surface outgassing rate (kg/m<sup>2</sup>/s)  
 $r_o$  = spacecraft radius (m)

In the steady state, the total mass flux through any sphere of radius  $r$ , for  $r > r_o$  must be equal to  $M_o$ . The flux per unit area through a sphere of radius  $r$  then is:

$$F_r = M_o / (4\pi r^2) = (F_o 4\pi r_o^2) / (4\pi r^2) = F_o r_o^2 / r^2 \quad \text{Eq. 4.1-13}$$

where,  $F_r$  = flux at radius  $r$  (kg/m<sup>2</sup>/s)  
 $F_o$  = surface outgassing rate (kg/m<sup>2</sup>/s)  
 $r_o$  = spacecraft radius (m)  
 $r$  = radial distance (m)

The density at radial distance  $r$ , for  $r > r_o$ , is:

$$D_r = F_r / V = F_o r_o^2 / (V r^2) \quad \text{Eq. 4.1-14}$$

where,  $D_r$  = density at radius  $r$  (kg/m<sup>3</sup>)  
 $F_r$  = flux at radius  $r$  (kg/m<sup>2</sup>/s)  
 $F_o$  = surface outgassing rate (kg/m<sup>2</sup>/s)  
 $r_o$  = spacecraft radius (m)  
 $r$  = radial distance (m)  
 $V$  = thermal velocity (m/s)

As was noted previously, there exists a critical ionization radius which is a function of the spacecraft potential, the Debye sheath and the radial velocity of the contaminant molecule. If a molecule is ionized beyond the critical ionization radius then it escapes the spacecraft potential, but all molecules ionized within the critical radius lack sufficient kinetic energy to escape and return to the spacecraft. In the steady state, the total mass of neutral contaminant molecules within the critical ionization radius is described by the integral:

$$M_t = \int_{r_o}^{r_c} [F_o r_o^2 / (V r^2)] (4\pi r^2) dr \quad \text{Eq. 4.1-15}$$

Hence, 
$$M_t = 4\pi F_o r_o^2 (r_c - r_o) / V \quad \text{Eq. 4.1-16}$$

where,  $M_t$  = total mass within critical ionization radius  $r_c$  (kg)  
 $F_o$  = surface outgassing rate (kg/m<sup>2</sup>/s)  
 $r_o$  = spacecraft radius (m)  
 $r_c$  = critical ionization radius (m)  
 $V$  = thermal velocity (m/s)

Neutral contaminant molecules may be ionized by any one of three different mechanisms; solar UV photons, ambient electrons, and energetic ions. Of these three mechanisms, only the solar UV photons and ambient electrons are significant. Therefore, the number of ions produced per second within the critical ionization radius is:

$$M_i = M_t (v_e + v_{ph}) \quad \text{Eq. 4.1-17}$$

where,  $M_i$  = total mass of ions returned to spacecraft (kg)  
 $M_t$  = total mass within critical ionization radius  $r_c$  (kg)  
 $v_e$  = ionization frequency due to electrons  
 (ionizations/molecule/s)

$\nu_{ph}$  = ionization frequency due to solar UV photons  
(ionizations/molecule/s)

The contaminant return mass fraction (Mf) is then given by:

$$Mf = M_i / M_o \quad \text{Eq. 4.1-18}$$

The ESR code computes the contaminant mass that returns to the spacecraft in terms of total g/s and also in terms of g/cm<sup>2</sup>/s based on the charged area. Deposition is then computed assuming a worst case sticking coefficient of unity and a contaminant density of 1 g/cm<sup>3</sup>. The computed deposition thickness is provided in angstroms.

Although all of the calculations within the ESR tool are made using mks (meter, kilogram, seconds) units, some inputs and outputs are cgs (centimeter, gram, seconds) units, for parameters where cgs units are more commonly used by the contamination community.

## 4.2 ESR Due to Planar Conductor Electric Field

For cases where the Debye length is much smaller than the characteristic dimension of the spacecraft and where the spacecraft charged surface is not well represented by a sphere, such as a solar panel in low earth orbit, a different approach is warranted. The Debye length in low Earth orbit can be just a few centimeters. For such cases, the electrostatic return of contaminants is a local effect relative to the size of the spacecraft and can best be modeled using the electric field equations for an infinite planar charged conductor.

For an extended planar conductor, the E field is given by

$$E = \sigma / \epsilon_o \quad \text{Eq. 4.2-1}$$

Where,  $\sigma$  = charge per unit area (C/m<sup>2</sup>)  
 $\epsilon_o$  = permittivity of free space = 8.85 x 10<sup>-12</sup> C<sup>2</sup>/N-m<sup>2</sup>

In the presence of plasma shielding

$$E = \sigma / \epsilon_o e^{-r/\lambda} \quad \text{Eq. 4.2-2}$$

Where,  $\sigma$  = charge per unit area (C/m<sup>2</sup>)  
 $\epsilon_o$  = permittivity of free space = 8.85 x 10<sup>-12</sup> C<sup>2</sup>/N-m<sup>2</sup>  
 $r$  = distance above conductor surface (m)  
 $\lambda$  = debye length (m)

The energy or work required to move a charge (q) from point A to point B in the electric field is

$$\text{Work} = q \int_B^A E dr = q \int_B^A \sigma/\epsilon_o e^{-r/\lambda} dr \quad \text{Eq. 4.2-3}$$

### 4.2.1 Escape Energy Required for Ion

If the molecule is ionized at a distance  $r_i$  above the conductor surface, then the work required for the ion to escape is the energy required to move the ion from  $r_i$  to infinity.

$$\text{Work} = q \int_{r_i}^{\infty} \sigma / \epsilon_0 e^{-r/\lambda} dr = q\sigma\lambda / \epsilon_0 e^{-r/\lambda} \quad \text{Eq. 4.2-4}$$

Since,

$$\text{Work} = q(V_B - V_A) = q \int_{r_b}^{r_a} \sigma / \epsilon_0 (e^{-r/\lambda}) dr = q\sigma\lambda / \epsilon_0 (1/e^{rb/\lambda} - 1/e^{ra/\lambda}) \quad \text{Eq. 4.2-5}$$

### 4.2.2 Absolute Potential

The absolute potential at a distance  $r$  can be determined by letting  $r_a = \infty$ , and  $V_A = 0$

$$V_B = \sigma\lambda / \epsilon_0 (1/e^{rb/\lambda}) \quad \text{Eq. 4.2-6}$$

The potential  $V_B$  at the spacecraft surface can be determined by setting  $r_b = 0$

$$\sigma = V_B \epsilon_0 / \lambda \quad \text{Eq. 4.2-7}$$

### 4.2.3 Critical Ionization Distance

The critical ionization distance  $r_c$  can be determined by setting the kinetic energy of the molecule at the instant of ionization equal to the work required to move the newly created ion from the ionization point to infinity.

$$mv^2/2 = q \int_{r_c}^{\infty} \sigma / \epsilon_0 (e^{-r/\lambda}) dr = q\sigma\lambda / \epsilon_0 (e^{-r_c/\lambda}) \quad \text{and} \\ r_c = \lambda \ln [2q\sigma\lambda / (\epsilon_0 mv^2)] \quad \text{Eq. 4.2-8}$$

Where,  
 $r_c$  = critical ionization distance above conductor surface (m)  
 $\sigma$  = conductor charge per unit area ( $C/m^2$ )  
 $\epsilon_0$  = permittivity of free space =  $8.85 \times 10^{-12} C^2/N\cdot m^2$   
 $q$  = ion charge (C)  
 $\lambda$  = debye length (m)  
 $m$  = mass of molecule (kg)  
 $v$  = thermal velocity of molecule (m/s)

### 4.2.4 Ion Return Impact Velocity

The impact velocity of the returning ion can be computed by converting the potential energy of the ion in the electric field at the point of ionization to kinetic energy.



$$mv_s^2/2 = q \int_0^{r_i} \sigma/\epsilon_0 (e^{-ri/\lambda}) dr \quad \text{integrating and solving for } v,$$

$$v_s = [(2q\sigma\lambda / m\epsilon_0) (1 - e^{-ri/\lambda})]^{1/2} \quad \text{Eq. 4.2-9}$$

Where,  
 $v_s$  = surface impact velocity of ion (m/s)  
 $r_i$  = ionization distance above conductor surface (m)  
 $\sigma$  = conductor charge per unit area (C/m<sup>2</sup>)  
 $\epsilon_0$  = permittivity of free space =  $8.85 \times 10^{-12}$  C<sup>2</sup>/N-m<sup>2</sup>  
 $q$  = ion charge (C)  
 $\lambda$  = debye length (m)  
 $m$  = mass of molecule (kg)

#### 4.2.5 Electrostatic Return Mass

Assuming a constant surface outgassing rate, the fraction of mass outgassed in one second that is within the critical ionization distance is  $r_c/v$  where  $v$  is the average thermal velocity of the outgassed molecules. The mass flux ( $F_r$ ) which returns to the spacecraft is

$$F_r = F_o r_c (v_e + v_{ph}) / v \quad \text{Eq. 4.2-10}$$

Where,  
 $F_r$  = returned mass flux (kg/m<sup>2</sup>/s)  
 $F_o$  = surface outgassing rate (kg/m<sup>2</sup>/s)  
 $r_c$  = critical ionization distance (m)  
 $v_e$  = ionization frequency due to electrons  
 (ionizations/molecule/s)  
 $v_{ph}$  = ionization frequency due to solar UV photons  
 (ionizations/molecule/s)  
 $v$  = thermal velocity of outgassed molecules (m/s)

The contaminant return mass fraction  $M_f$  is given by

$$M_f = F_r / F_o \quad \text{Eq. 4.2-11}$$

Where,  
 $M_f$  = returning mass fraction  
 $F_r$  = returned mass flux (kg/m<sup>2</sup>/s)  
 $F_o$  = surface outgassing rate (kg/m<sup>2</sup>/s)

## 5.0 MODEL OPERATIONS

The model can be used in various ways to determine if return flux of ionized contaminants or sputtering may result for a space system. The following examples are intended to assist the user in using the model, attached as Appendix A. The user's manual in Appendix A helps the user select required calculations, input choices and assumptions.

### 5.1 Quick Look

For this case, it is assumed the user has little knowledge of his space system and its outgassing characteristics. What is important to the user is to determine what fraction of the outgassing mass comes back. The user can then present this to the materials engineers to allow an assessment of the return flux. The first screen shown in the model is shown in Figure 5.1. The user selects the whole spacecraft ESR calculations box. The next screen that comes up is shown in Figure 5.2. This is the input screen used for the calculations. It is assumed the user knows from previous charging studies that some of the spacecraft surfaces will charge to ten volts and that the Debye sheath is on the order of 100 meters. It is fortunate that these are defaults in the ESR model. The surface temperature default of 300K is close enough also. Since the user doesn't know the outgassing characteristics of his system at this point, the user can select the "Calculate ESR" box. To determine just the fraction, returning it is also not totally necessary to know anything about the dimensions of the space system. Not knowing the size will introduce an error of only 10 to 20 % over reasonable ranges of one meter to 10 meters in radius for a sphere.

Figure 5.3 shows the output screen for the calculation. The user can check the input assumptions here to make sure they are correct.

The results show that the contaminant mass fraction that returned for a 10 volt negative charge and a Debye length of 100 meters is  $2.71 \text{ E-}04=0.000271$ . Now the user can investigate the outgassing of the system and estimate what fraction will return. If the user determines the outgassing total quantity and the area of the charged surfaces on his spacecraft, then the user can calculate the amount depositing per unit area. This gives an end of life deposition value after most outgassing is complete.

If the rate of deposition is required, the user must return to the model and let different fractions be released over several time periods based on best estimates from the materials engineers.

This example shows that small amount of information may be all that is required to determine if return flux of the contaminant is a problem or not.

### 5.2 Worst Case Assumptions

Many times, the worst case situation is evaluated. The reason being, if the worst case is not a problem, then there is no problem for other conditions.

On the first screen, Figure 5.1, the user selects “Whole Spacecraft” and goes to the screen in Figure 5.2. The user selects “Help me Estimate Debye Sheath”. Figure 5.4 shows the screen for selecting the Debye Sheath. The user selects “Use GEO Flight Data”. Figure 5.5 then comes up and allows selection of the worst case data. After selecting continue, Figure 5.6 shows the values for the worst case Debye sheath length. Selecting continue shows Figure 5.7, which is the input data. After selecting “Calculate ESR”, Figure 5.8 shows the output results for this analysis.

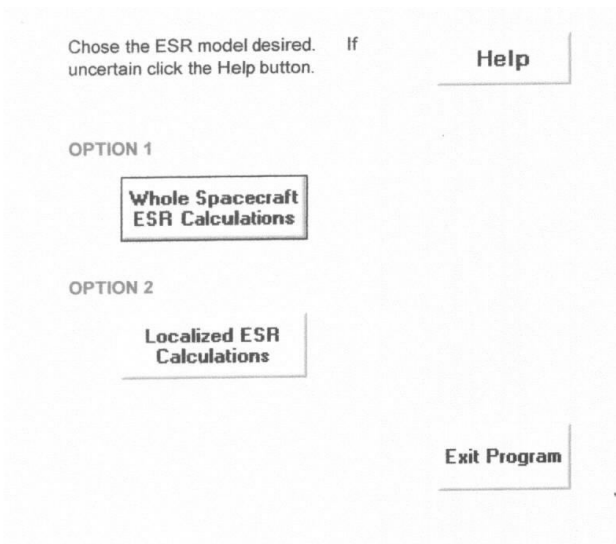
### **5.3 Sputtering**

For this case, the sputtering option will be presented. Here, the user selects from the screen in Figure 5.2 the “Surface Sputter Parameters”. Figure 5.9 shows the screen where molecular weights can be input. The default value is for aluminum. Selecting return brings up the screen in Figure 5.10 where the selection of spacecraft voltage is set to 1000 volts and the box for computing surface sputter is selected. After executing the model calculations, the results are shown in Figure 5.11. The results show that the sputter rate is  $1.57\text{E-}07$  Angstroms per second. For the total erosion, the rate must be multiplied by the time this condition exists.

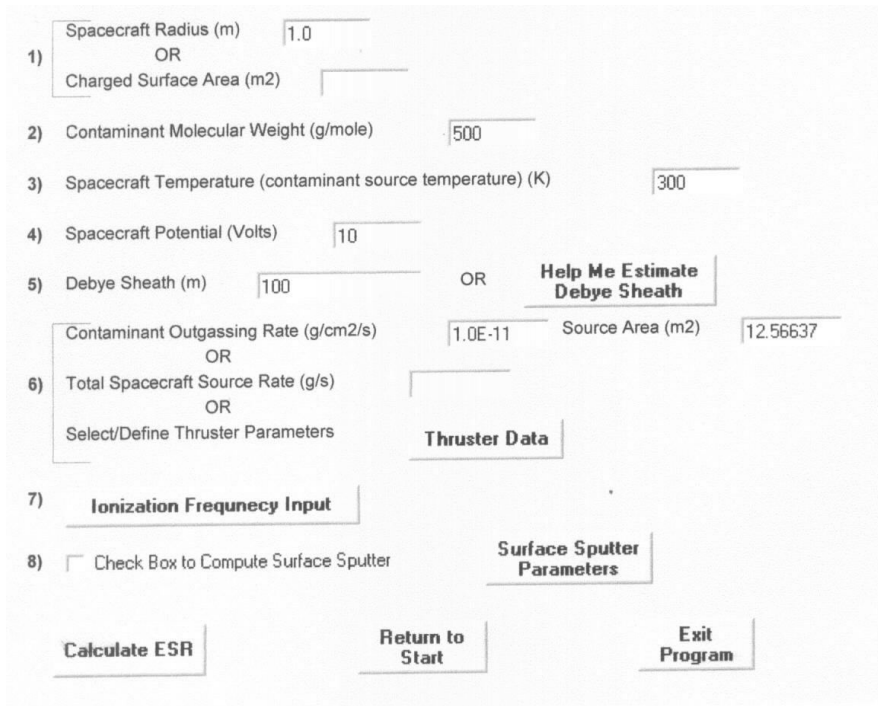
### **5.4 Solar Sail and LEO**

This case shows how to use the model for the second option where localized geometry exists. This corresponds to a case where the Debye sheath is small compared to the geometry of the surface, or the surface is a flat plate. The calculations are for a planar electric field. The primary applications for this option are for solar sails and surfaces at LEO that have a charge such as solar arrays.

Selecting “Localized ESR Calculations”, in the screen in Figure 5.1, Figure 5.12 results. There are less choices since the dimension of the spacecraft is not required. The outgassing rate is for the local environment such as, solar arrays, thermal control surfaces and solar sail surfaces. The voltage of the surface is a necessary input here for sputtering effects. Selecting the “Calculate ESR” button produces the same type of output as seen in the above examples.



**Figure 5.1 First Model Screen**



**Figure 5.2 Input Screen**

## Electrostatic Return (ESR) Of Contaminant Molecules ESR Spherical Model Results

### Computational Input Parameters

Spacecraft Radius (m) = 1.0  
Molecular Weight of Contaminant (g/mole) = 500  
Contaminant Source Temperature (K) = 300  
Spacecraft Potential (Volts) = 10  
Debye Length (m) = 100  
Contaminant Source Rate (g/cm<sup>2</sup>/s) = 1.00E-11  
Contaminant Molecule Ionization Frequency due to Electrons (ionizations/molecule/s) = 2.20E-04  
Contaminant Molecule Ionization Frequency due to Photons (ionizations/molecule/s) = 1.20E-04

### Computational Results

Returned Contaminant Mass Rate = 3.4109E-13 (kg/s) = 3.4109E-10 (g/s)  
Returned Contaminant Mass Rate per Area = 2.7143E-14 (kg/m<sup>2</sup>/s) = 2.7143E-15 (g/cm<sup>2</sup>/s)  
Returned Contaminant Mass Fraction = 2.71E-04  
Deposition (unity sticking coef) = 2.71E-07 (angstroms/s)  
Maximum Return Velocity = 1961.99 (m/s)  
Return Energy per Molecule = 25.57 (ev)  
Critical Ionization Radius = 80.77 (m)  
Critical Ionization Distance above Sphere Surface = 79.77 (m)  
Spacecraft Charge = 1.16E-09 C  
Spacecraft Charge per Square Meter = 9.23E-11 C/m<sup>2</sup>

**Figure 5.3. ESR Output Screen**

The Debye sheath is the distance from the spacecraft where the spacecraft potential drops by  $1/e$  due to shielding by the local plasma. The Debye sheath is a function of the local plasma density and temperature. Chose one of the following 3 options.

---

**OPTION 1 - Input Plasma Density and Temperature**

Plasma Density (electrons/m<sup>3</sup>)  Plasma Temperature (K)   
or  
Plasma Temperature (ev)

**Compute Debye Sheath** Debye Sheath =  meters

---

**OPTION 2 - Determine Plasma Parameters from Geosynchronous Flight Data (this option should only be chosen for cases with high orbital altitudes where geosynchronous data is applicable)**

**Use Geo Flight Data**

---

**OPTION 3 - Chose Approximate Debye Length for Orbital Conditions from Table**

**Use Table Approximation**

---

**Continue**

**Figure 5.4. Help Screen for Selecting Debye Sheath**

The following parameters for energetic electrons at geosynchronous orbits are compiled from flight measurements from ATS-5, ATS-6, and SCATHA.

The first input option represents average conditions as measured by the forementioned spacecraft. This data can be used to approximate pre-storm conditions.

The second input option approximates worst case conditions

Select one of the following

Use Average Geosynchronous Conditions (Pre-Storm)

Use Worst Case Geosynchronous Conditions

Note: If electron density and temperature are known, leave both boxes unchecked and click Return button, then input electron density and temperature in option 1 in the Debye calculation form.

**Continue**

**Figure 5.5. Help Screen for Debye Sheath Flight Data**

The Debye sheath is the distance from the spacecraft where the spacecraft potential drops by  $1/e$  due to shielding by the local plasma. The Debye sheath is a function of the local plasma density and temperature. Chose one of the following 3 options.

**OPTION 1 - Input Plasma Density and Temperature**

Plasma Density (electrons/m <sup>3</sup> )	<input type="text" value="1120000"/>	Plasma Temperature (K) or Plasma Temperature (ev)	<input type="text"/>	<input type="text" value="12000"/>
<b>Compute Debye Sheath</b>	Debye Sheath =	<input type="text" value="760"/>	meters	

**OPTION 2 - Determine Plasma Parameters from Geosynchronous Flight Data (this option should only be chosen for cases with high orbital altitudes where geosynchronous data is applicable)**

**Use Geo Flight Data**

**OPTION 3 - Chose Approximate Debye Length for Orbital Conditions from Table**

**Use Table Approximation**

**Continue**

**Figure 5.6. Data Selected for Debye Sheath Worst Case**



1) Spacecraft Radius (m)   
OR  
Charged Surface Area (m2)

2) Contaminant Molecular Weight (g/mole)

3) Spacecraft Temperature (contaminant source temperature) (K)

4) Spacecraft Potential (Volts)

5) Debye Sheath (m)  OR **Help Me Estimate Debye Sheath**

6) Contaminant Outgassing Rate (g/cm2/s)  Source Area (m2)   
OR  
Total Spacecraft Source Rate (g/s)   
OR  
Select/Define Thruster Parameters **Thruster Data**

7) **Ionization Frequency Input**

8)  Check Box to Compute Surface Sputter **Surface Sputter Parameters**

**Calculate ESR** **Return to Start** **Exit Program**

**Figure 5.7. Input Data for Worst Case**

## Electrostatic Return (ESR) Of Contaminant Molecules ESR Spherical Model Results

### Computational Input Parameters

Spacecraft Radius (m) = 1.0  
Molecular Weight of Contaminant (g/mole) = 500  
Contaminant Source Temperature (K) = 300  
Spacecraft Potential (Volts) = 1000  
Debye Length (m) = 760.0000  
Contaminant Source Rate (g/cm<sup>2</sup>/s) = 1.00E-11  
Contaminant Molecule Ionization Frequency due to Electrons (ionizations/molecule/s) = 2.20E-04  
Contaminant Molecule Ionization Frequency due to Photons (ionizations/molecule/s) = 1.20E-04

### Computational Results

Returned Contaminant Mass Rate = 5.9275E-12 (kg/s) = 5.9275E-09 (g/s)  
Returned Contaminant Mass Rate per Area = 4.7170E-13 (kg/m<sup>2</sup>/s) = 4.7170E-14 (g/cm<sup>2</sup>/s)  
Returned Contaminant Mass Fraction = 4.72E-03  
Deposition (unity sticking coef) = 4.72E-06 (angstroms/s)  
Maximum Return Velocity = 19644.31 (m/s)  
Return Energy per Molecule = 2563.26 (ev)  
Critical Ionization Radius = 1387.27 (m)  
Critical Ionization Distance above Sphere Surface = 1386.27 (m)  
Spacecraft Charge = 1.12E-07 C  
Spacecraft Charge per Square Meter = 8.92E-09 C/m<sup>2</sup>

**Figure 5.8. Output for Worst Case**

The image shows a software interface with two input fields and a button. The first field is labeled '1) Molecular Weight of surface material to be sputtered (g/mole)' and contains the value '27.0'. The second field is labeled '2) Density of Surface Material (g/cm3)' and contains the value '2.7'. Below these fields is a button labeled 'Return'.

1) Molecular Weight of surface material to be sputtered (g/mole)	27.0
2) Density of Surface Material (g/cm3)	2.7

Return

**Figure 5.9. Molecular Weight and Density Selection**

1) Spacecraft Radius (m)  OR Charged Surface Area (m2)

2) Contaminant Molecular Weight (g/mole)

3) Spacecraft Temperature (contaminant source temperature) (K)

4) Spacecraft Potential (Volts)

5) Debye Sheath (m)  OR **Help Me Estimate Debye Sheath**

6) Contaminant Outgassing Rate (g/cm2/s)  OR Source Area (m2)   
 OR  
 Total Spacecraft Source Rate (g/s)   
 OR  
 Select/Define Thruster Parameters **Thruster Data**

7) **Ionization Frequency Input**

8)  Check Box to Compute Surface Sputter **Surface Sputter Parameters**

**Calculate ESR** **Return to Start** **Exit Program**

**Figure 5.10 Screen for Selecting Voltage and Surface Sputter Option**

## Electrostatic Return (ESR) Of Contaminant Molecules ESR Spherical Model Results

### Computational Input Parameters

Spacecraft Radius (m) = 1.0  
Molecular Weight of Contaminant (g/mole) = 500  
Contaminant Source Temperature (K) = 300  
Spacecraft Potential (Volts) = 1000  
Debye Length (m) = 100  
Contaminant Source Rate (g/cm<sup>2</sup>/s) = 1.00E-11  
Contaminant Molecule Ionization Frequency due to Electrons (ionizations/molecule/s) = 2.20E-04  
Contaminant Molecule Ionization Frequency due to Photons (ionizations/molecule/s) = 1.20E-04  
Molecular Weight of Surface Material (g/mole) = 27.0  
Density of Surface Material (g/cm<sup>3</sup>) = 2.7

### Computational Results

Returned Contaminant Mass Rate = 1.3804E-12 (kg/s) = 1.3804E-09 (g/s)  
Returned Contaminant Mass Rate per Area = 1.0985E-13 (kg/m<sup>2</sup>/s) = 1.0985E-14 (g/cm<sup>2</sup>/s)  
Returned Contaminant Mass Fraction = 1.10E-03  
Deposition (unity sticking coef) = 1.10E-06 (angstroms/s)  
Maximum Return Velocity = 19645.11 (m/s)  
Return Energy per Molecule = 2563.47 (ev)  
Surface Sputter = 1.57E-17 (m/s) = 1.57E-07 (angstroms/s)  
Critical Ionization Radius = 323.83 (m)  
Critical Ionization Distance above Sphere Surface = 322.83 (m)  
Spacecraft Charge = 1.16E-07 C  
Spacecraft Charge per Square Meter = 9.23E-09 C/m<sup>2</sup>

**Figure 5.11. Output With Sputter Results**

1) Contaminant Molecular Weight (g/mole)

2) Spacecraft Temperature (contaminant source temperature) (K)

3) Spacecraft Potential (Volts)

4) Debye Sheath (m)  OR

5) Contaminant Outgassing Rate (g/cm<sup>2</sup>/s)

6)

7)  Check Box to Compute Surface Sputter

**Figure 5.12. Input Screen for Localized ESR Calculations.**

**APPENDIX A – ELECTROSTATIC RETURN (ESR) PROGRAM USER’S MANUAL**

## 1.0 INTRODUCTION

Contaminants generated by a spacecraft can return to a spacecraft via a mechanism known as electrostatic return (ESR). ESR can occur when spacecraft generated contaminants become ionized by solar UV or solar electrons and are returned to the spacecraft due to the spacecraft potential.

The ESR tool is a first generation computational code created for the purpose of computing the return characteristics of spacecraft generated contaminants which return to the spacecraft via the electrostatic return mechanism. The ESR code does not compute the spacecraft charging, this task is left to preexisting codes. The ESR code does compute the debye sheath (if necessary), the returned contaminant mass rate, returned mass flux, returned mass ratio, return velocity, and surface sputter (where applicable). To help the user, default values have been supplied for all program inputs. However, the user is encouraged to supply the program with appropriate inputs which accurately reflect the specific spacecraft and environmental conditions being modeled.

This document contains information to help install and operate the ESR program. The user interface for the ESR code was designed to be intuitive, but this document should be useful if confusion arises.

## 2.0 ESR TOOL INSTALLATION

Three files are required for installation of the ESR Tool. These files are:

- 1) Setup.exe
- 2) Setup.lst
- 3) Esr.cab

These three files should be located in the same directory prior to beginning installation. The location of the directory is not important.

Double click the setup executable file **Setup.exe**. Follow the instructions for the installation of the ESR Tool software. Unless the user changes the installation directory when asked during setup, the ESR Tool will be installed as an application in the Programs directory (C:\Program Files\ESR).

After installation, the ESR Tool can be executed via Start menu. To begin the program, click on **Start**, then click on **Programs**, then click on **ESR Tool**.

The ESR Tool software can be uninstalled using the Program Removal Utility found in the Control Panel folder.



### 3.0 PROGRAM OPERATION

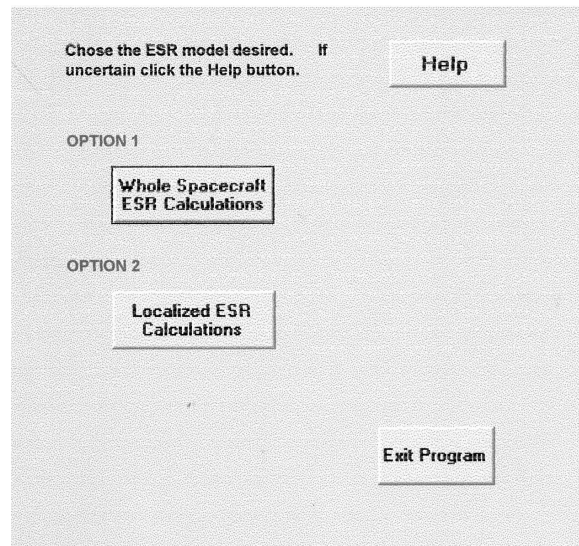
If the ESR Tool has been installed in the default location, to run the program, click on **Start**, then click on **Programs**, then click on **ESR Tool**. If the ESR Tool was installed elsewhere, double click on the ESR.exe file icon which will be located in the directory designated by the user during installation.

#### 3.1 Splash Screen

When program execution is initiated, a splash screen will briefly be displayed as the program is loaded and initialized. No response is required of the user during this screen display. The splash screen should automatically unload after several seconds.

#### 3.2 Model Selection Screen

After the program execution begins and the splash screen has unloaded, the model selection screen will appear. This screen contains four buttons. One of two models should be selected by single clicking the appropriate button. If the user is uncertain which model to choose, click the help button to display a description of the two models and comments provided to help the user choose the correct model for their current needs.



**Figure 3.2 – ESR Model Select Screen**

The ESR tool provides two different calculational routines to compute the electrostatic return of contaminants based on the user's needs.

**Option 1** – The first computational option assumes a spherical electric field for the charged spacecraft. This option should be used for cases where the debye sheath is on the order of or larger than the spacecraft

characteristic dimension. It can also be used for cases where the debye sheath is smaller than the spacecraft dimensions as long as the spherical electric field is a reasonable approximation of the actual spacecraft charge. It can be shown mathematically that the electric field of a spherical conductor provides a reasonable approximation of the electric field for other more complex conductor shapes as the distance from the conductor increases beyond the characteristic dimension of the conductor.<sup>1</sup>

**Option 2** – The second computational option assumes a planar electric field. In general, this option should be used for cases where the debye sheath is considerably smaller than the spacecraft dimensions. A classic case where this option should be used would be the computation of ESR to solar arrays in low earth orbit where the debye sheath is much smaller than the solar array dimensions. This computational option should be used when modeling localized ESR in an environment where the debye sheath is small relative to the spacecraft dimensions.

The mathematical derivations for both models are described in the contract final report and will not be included here.

### 3.3 Spherical Conductor Primary Input Screen

If option 1 is selected on the model selection screen, the next screen displayed will be the spherical conductor primary input screen. This screen contains all of the model input parameters or links to other input parameter screens as needed. There are eight parameters which must be specified prior to calculating the contaminant ESR.

The screenshot displays the Primary Spherical Model Input Screen with the following elements:

- 1) Spacecraft Radius (m)**: Input field with value 1.0. Below it is "OR" and **Charged Surface Area (m2)** with an empty input field.
- 2) Contaminant Molecular Weight (g/mole)**: Input field with value 500.
- 3) Spacecraft Temperature (contaminant source temperature) (K)**: Input field with value 300.
- 4) Spacecraft Potential (Volts)**: Input field with value 10.
- 5) Debye Sheath (m)**: Input field with value 100. To its right is "OR" and a button labeled **Help Me Estimate Debye Sheath**.
- 6) Contaminant Outgassing Rate (g/cm2/s)**: Input field with value 1.0E-11. To its right is **Source Area (m2)** with value 12.56637. Below this is "OR" and **Total Spacecraft Source Rate (g/s)** with an empty input field.
- Below item 6 is "OR" and **Select/Define Thruster Parameters** with a button labeled **Thruster Data**.
- 7) Ionization Frequency Input**: A button.
- 8)  Check Box to Compute Surface Sputter**: A checkbox. To its right is a button labeled **Surface Sputter Parameters**.
- At the bottom are three buttons: **Calculate ESR**, **Return to Start**, and **Exit Program**.

Figure 3.3 – Primary Spherical Model Input Screen

A brief description of each input is provided below.

**1) Spacecraft Radius or Charged Surface Area** - Enter the radius (meters) of the spacecraft. Or, if only a portion of the spacecraft is charged, enter the area (meters<sup>2</sup>) of the charged surface.

**2) Contaminant Molecular Weight** - Enter the contaminant molecular weight (g/mole). This value will be set automatically when a thruster profile is picked for the contaminant source. In all other cases, the user must provide a valid value or the default value may be used. The automatically supplied value for thruster sources is based on the average of the entire thruster effluent and may be changed by the user after thruster selection.

**3) Spacecraft Contaminant Source Temperature** - Enter the spacecraft contaminant source temperature (deg K). If a thruster is selected as the contaminant source then the contaminant temperature input is irrelevant and may be omitted. For all other contaminant sources, a valid value must be supplied by the user.

**4) Spacecraft Potential** - Enter the spacecraft potential (volts). This value must be supplied by the user and will typically be the result of a spacecraft charging analysis.

**5) Debye Sheath** - Enter the Debye length (meters). Note that the terms Debye length and Debye sheath are use interchangeably in the ESR tool and documentation. If the debye sheath is not known, single click the “Help Me Estimate Debye Sheath” button. This will take the user to the Debye sheath computation screen. If the Debye sheath is computed using the Debye sheath computation screen, then the computed value will automatically be supplied to the spherical conductor primary input screen.

**6) Contaminant Source Rate** - The user has three options to choose from to define the contaminant source rate.

The first option is to supply a contaminant source rate in terms of g/cm<sup>2</sup>/s. The units on the input value are cgs units because surface outgassing is generally specified in cgs units. If this option is chosen, then a contaminant source area (meters) must also be supplied. If the spacecraft radius was supplied in input 1, the contaminant source area defaults to the area of the designated sphere size. However, the area should be set to reflect the actual contaminant source area if the entire spherical surface is not an outgassing source.

The second option for the user is to supply the source rate in terms of g/s. This is a total source rate and is useful for describing contaminant sources such as vents, etc.

The third option is for the user to select or specify a thruster. This is done by single clicking on the “Thruster Data” button. This button will take the user to the thruster specification screen. The source input will automatically be supplied to the spherical conductor primary input screen.

**7) Ionization Frequency Input** - The ionization frequency inputs may be changed by the user by single clicking on the “Ionization Frequency Input” button.

**8) Surface Sputter Computation** - It is possible for contaminant molecules to return to the spacecraft surface with enough kinetic energy to remove molecules from spacecraft surfaces in an erosion process known as sputtering. To compute sputtering, the user should check the “Compute Surface Sputter” box and then set the surface material specifications by single clicking on the “Surface Sputter Parameters” button.

Once all eight inputs have been specified, single click the button titled “Compute ESR” to begin the computation process. At this time, any missing user inputs will be brought to the attention of the user. The ESR tool does not attempt to verify the validity of the user inputs beyond checking for proper format. At any time, the button titled “Return to Start” may be single clicked to return to the model selection screen. Likewise, the “Exit Program” button can be used at any time to exit the program. It should be noted that the computation process may require up to a minute to complete, based on the particular user inputs. Please be patient.

### 3.4 Planar Conductor Primary Input Screen

If option 2 is selected on the model selection screen, then the next screen displayed will be the planar conductor primary input screen. This screen contains all of the model input parameters or links to other input parameter screens as needed.

The screenshot displays a software interface for entering model parameters. It consists of seven numbered items:

- 1) Contaminant Molecular Weight (g/mole): Input field with value 500.
- 2) Spacecraft Temperature (contaminant source temperature) (K): Input field with value 300.
- 3) Spacecraft Potential (Volts): Input field with value 10.
- 4) Debye Sheath (m): Input field with value 1, followed by the text "OR" and a button labeled "Help Me Estimate Debye Sheath".
- 5) Contaminant Outgassing Rate (g/cm2/s): Input field with value 1.0E-11.
- 6) Ionization Frequency Input: A button.
- 7)  Check Box to Compute Surface Sputter, followed by a button labeled "Surface Sputter Parameters".

At the bottom of the screen are three buttons: "Calculate ESR", "Return To Start", and "Exit Program".

**Figure 3.4 – Primary Planar Model Input Screen**

There are seven parameters which must be specified prior to calculating the contaminant ESR. A brief description of each input is provided below.

**1) Contaminant Molecular Weight** - Enter the contaminant molecular weight (g/mole). The user must provide a valid value or the default value may be used.

**2) Spacecraft Contaminant Source Temperature** - Enter the spacecraft contaminant source temperature (deg K), a valid value must be supplied by the user.

**3) Spacecraft Potential** - Enter the spacecraft potential (volts). This value must be supplied by the user and will typically be the result of a spacecraft charging analysis.

**4) Debye Sheath** - Enter the Debye length (meters). Note that the terms Debye length and Debye sheath are use interchangeably in the ESR tool and documentation. If the debye sheath is not known, single click the “Help Me Estimate Debye Sheath” button. This will take the user to the Debye sheath computation screen. If the Debye sheath is computed using the Debye sheath computation screen, the computed value will automatically be supplied to the planar conductor primary input screen.

**5) Contaminant Source Rate** - Enter a contaminant source rate in terms of  $\text{g}/\text{cm}^2/\text{s}$ . The units on the input value are cgs units because surface outgassing is generally specified in cgs units.

**6) Ionization Frequency Input** - The ionization frequency inputs may be changed by the user by single clicking on the “Ionization Frequency Input” button.

**7) Surface Sputter Computation** - It is possible for contaminant molecules to return to the spacecraft surface with enough kinetic energy to remove molecules from spacecraft surfaces in an erosion process known as sputtering. To compute sputtering, the user should check the “Compute Surface Sputter” box and then set the surface material specifications by single clicking on the “Surface Sputter Parameters” button.

Once all seven inputs have been specified, single click the button titled “Compute ESR” to begin the computation process. At this time, any missing user inputs will be brought to the attention of the user. The ESR tool does not attempt to verify the validity of the user inputs beyond checking for proper format. At any time the button titled “Return to Start” may be single clicked to return to the model selection screen. Likewise, the “Exit Program” button can be used at any time to exit the program.

### 3.5 Debye Sheath Computation Screen

The Debye sheath is the distance from the spacecraft where the spacecraft potential drops by  $1/e$  due to shielding by the local plasma. The Debye sheath is a function of the local plasma density and temperature. The Debye sheath computation screen allows the user to estimate the appropriate Debye sheath in one of three different ways.

The Debye sheath is the distance from the spacecraft where the spacecraft potential drops by  $1/e$  due to shielding by the local plasma. The Debye sheath is a function of the local plasma density and temperature. Chose one of the following 3 options.

---

**OPTION 1 - Input Plasma Density and Temperature**

Plasma Density (electrons/m<sup>3</sup>)  Plasma Temperature (K)   
 or  
 Plasma Temperature (ev)

Debye Sheath =  meters

---

**OPTION 2 - Determine Plasma Parameters from Geosynchronous Flight Data (this option should only be chosen for cases with high orbital altitudes where geosynchronous data is applicable)**

---

**OPTION 3 - Chose Approximate Debye Length for Orbital Conditions from Table**

---

**Figure 3.5 – Debye Sheath Computation Screen**

**Option 1** – The first option allows the user to compute the Debye sheath by supplying the local plasma density (electron/m<sup>3</sup>) and temperature. The temperature may be supplied either in degrees Kelvin or in electron volts. The user should single click the “Compute Debye Sheath” button. The computed Debye sheath will appear in the adjacent window and will be automatically supplied to the primary input screen.

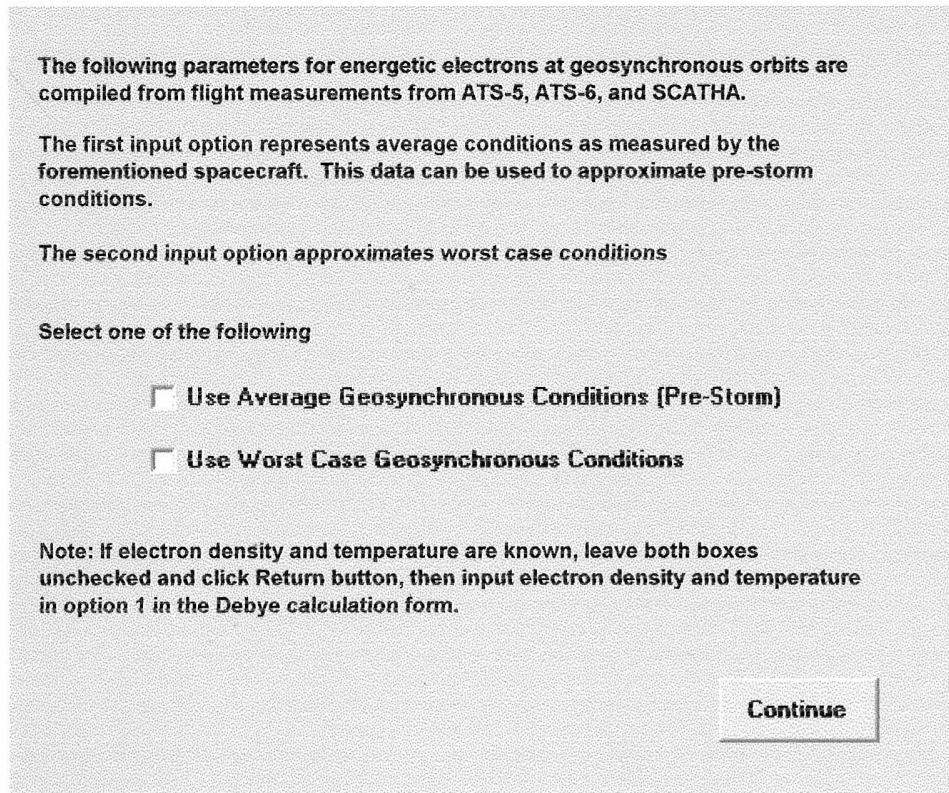
**Option 2** – If the spacecraft being modeled is in a high orbit such as a geosynchronous orbit, and the user does not know the local plasma density and temperature, then the user should single click the “Use Geo Flight Data” button. This button will open the Geosynchronous Plasma Flight Data screen and will allow the user to obtain averaged plasma density and temperature values based on actual flight measurements. For more detailed information on the Geosynchronous Plasma Flight Data screen, see section 3.5.1.

**Option 3** – The third option should be the last resort for obtaining the Debye sheath. This option can be selected by single clicking the “Use Table Approximation” button. This button will open the Debye Sheath Approximation screen which allows the user to select an approximate Debye sheath based on crude orbital ranges and solar environments. For more detailed information on the Debye Sheath Approximation screen, see section 3.5.2.

If options 1 or 2 were selected and completed, single click the “Continue” button to return to the primary input screen. Completion of option 3 will automatically return the user to the primary input screen. In all three cases, the computed Debye sheath value will be automatically supplied to the primary input screen.

### 3.5.1 Debye Sheath Geosynchronous Orbit Flight Data Screen

The GEO flight data screen allows the user to obtain average plasma density and temperature values based on average or worst case conditions for geosynchronous orbit. The average values are based on data which was collected by three different spacecraft, ATS-5, ATS-6, and SCATHA. These average values should only be used for spacecraft in high orbital altitudes where the geosynchronous data is applicable.



The following parameters for energetic electrons at geosynchronous orbits are compiled from flight measurements from ATS-5, ATS-6, and SCATHA.

The first input option represents average conditions as measured by the forementioned spacecraft. This data can be used to approximate pre-storm conditions.

The second input option approximates worst case conditions

Select one of the following

Use Average Geosynchronous Conditions (Pre-Storm)

Use Worst Case Geosynchronous Conditions

Note: If electron density and temperature are known, leave both boxes unchecked and click Return button, then input electron density and temperature in option 1 in the Debye calculation form.

Continue

**Figure 3.5.1 – Debye Sheath Geosynchronous Orbit Data Screen**

The user may select the average pre-storm environmental conditions or the average of the worst storm conditions. The user should select the data to use by checking the appropriate box. After the selection is made, click the “Continue” button to return to the previous screen. The plasma density, temperature and associated debye sheath value will appear in the appropriate text boxes.

### 3.5.2 Debye Sheath General Approximation Screen

The table approximation screen allows the user to generate a rough debye sheath value by selecting an orbital altitude range and a solar activity range.

For a very rough (order of magnitude) approximation of the Debye length, select the case below which most closely describes expected orbital environment.

Chose Orbital Altitude Range

Low (90 km to 600 km)

Middle (600 km to 2000 km)

High (2000 km or above)

Chose Solar Activity Range

Low Solar Activity

High Solar Activity

Approximate Debye Length (m) =

**Figure 3.5.2 – Debye Sheath Table Approximation Screen**

After the altitude range and the solar activity range have been selected, single click the “Get Debye Sheath” button. A Debye sheath value will appear in the adjacent text box. Single clicking the “Continue” button will close the table approximation screen and return the user to the current primary input screen. The Debye sheath value will automatically be supplied to the primary input screen.

It should be noted that the values provided in this routine are rough approximations of the Debye sheath value and should be used only when better information is not available.

### **3.6 Thruster Input Parameters Screen**

The thruster input parameters screen allows the user to input thruster specifications for cases where a thruster is the contaminant source of interest.



Select desired thruster from the drop down list and click "Get Parameters" or enter thruster parameters directly

Select Thruster Type

---

**Thruster Parameters**

Thruster Mass Flow Rate  g/s

Plume velocity  m/s

Average Molecular Mass  g/mole

**Figure 3.6 – Thruster Input Parameters Screen**

Three thruster input parameters are required. These are:

- 1) Thruster Mass Flow Rate (g/s)
- 2) Plume Effluent Velocity (m/s)
- 3) Average Molecular Mass (g/mole)

If these parameters are known for the thruster or vent of interest, they should be inputted directly in the appropriate text boxes. However, if user does not have the required thruster parameters then they may select one of the thruster profiles provided by single clicking on the pull down menu of thrusters. Select a thruster by moving the cursor over the thruster of choice. When the thruster of choice is highlighted, single click to select the thruster. After a thruster is selected, single click the "Get Parameters" button. The thruster parameters will appear in the three text boxes.

Once the thruster parameters have been input, single click the "Continue" button. The thruster input parameters screen will unload and the user will be returned to the primary input screen. The thruster parameters will automatically be supplied to the appropriate primary input screen text boxes. If one of these inputs is later changed, the program will provide a warning to the user.

### **3.7 Ionization Frequency Input Screen**

The ionization frequency input screen allows the user to set the ionization frequency (ionizations/molecule/s) for both solar electrons and solar photons. Default values are provided.

Note: Inputs presented on this form are generally not changed by the casual user. However, these inputs may be changed by knowledgeable users as the need arises.

Contaminant Molecule Ionization Frequency Due to Electrons (ionizations/molecule/s)

Contaminant Molecule Ionization Frequency Due to Solar Photons (ionizations/molecule/s)

**Figure 3.7 – Ionization Frequency Input Screen**

The ionization frequency values are automatically changed as a function of other source inputs. In general, the average user will probably not generally change these values. If the default values have been changed and the user wishes to reinitialize to the default values, single click on the “Return Default Values” button. Single clicking on the “Continue” button will unload the ionization frequency input screen and return the user to the primary input screen.

### 3.8 Surface Sputter Input Parameters Screen

The surface sputter input parameters screen allows the user to specify the surface material parameters for the purposes of computing the surface sputter due to electrostatic return of contaminant molecules.

1) Molecular Weight of surface material to be sputtered (g/mole)

2) Density of Surface Material (g/cm3)

**Figure 3.8 – Surface Sputter Input Parameters Screen**

The required inputs are:

- 1) Molecular weight of surface material (g/mole)
- 2) Density of surface material (g/cm<sup>3</sup>)

The default parameters provided are for aluminum.

Single clicking on the “Continue” button will unload the surface sputter input parameters screen and return the user to the primary input screen.

### 3.9 ESR Computational Results Screen

The ESR calculational results screen provides the user with the computational results and a record of the input parameters. The specific output parameters listed on the ESR calculational results screen will be a function of the user input choices. However at a minimum, all will include the following results:

- 1) Returned contaminant mass rate per area in kg/m<sup>2</sup>/s and g/cm<sup>2</sup>/s
- 2) Returned contaminant mass fraction – this is the fraction of contaminant mass which returns to the spacecraft.
- 3) Deposition thickness of returned contaminant in angstroms/s – this value assumes a sticking coefficient of unity (obvious worst case).
- 4) Maximum return velocity in m/s
- 5) Return energy per molecule in ev
- 6) Critical ionization radius in meters – this is the distance from the spacecraft beyond which a contaminant molecule from the modeled source will escape the spacecraft’s electric field.
- 7) Spacecraft charge per unit area in C/m<sup>2</sup>

If the user has specified that surface sputter be computed, then the output results will include the surface sputter rate in m/s and in angstroms/s.

By single clicking the “Return to Input Form” button, the user can return to the input screen. This feature is useful when performing parametric studies for a range of input values.

Single clicking the “Print Results” button will cause the results screen to be printed on the user’s default printer.

#### **4.0 ESR TOOL APPLICATION NOTES**

If the spacecraft has more than one contaminant source, the user may model the electrostatic return from of each source separately and add the results to obtain a cumulative result.

## REFERENCES

- 1 Rantanen, Ray, Tim Gordon, "Electrostatic Return of Contaminant Molecules", NASA Phase I SBIR Final Report, NAS8-99009, June 1999.
- 2 Rantanen, Ray, Tim Gordon "Electrostatic Return of Contaminants", Final Report, NASA PO H-32496D and H-32497D, February 2003

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