Numerical Thermal Model of a 30-cm NSTAR Ion Thruster

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ABSTRACT

A computer model has been produced for the NSTAR xenon ion thruster using a lumped parameter thermal nodal network model. This model contains 104 nodes on the thruster and was implemented using SINDA and TRASYS on various UNIX workstations. The model includes radiation and conduction heat transfer, the effect of plasma interaction on the thruster, and an account for finely perforated surfaces.

The model was applied to an NSTAR thruster outfitted with approximately 20 thermocouples for thermal tests at the NASA Lewis Research Center. This 30-cm ring-cusp thruster was put through various operating conditions in June and July of 1996. The results of these experiments were used to calibrate and confirm the computer model.

A description of the numeric model of the NSTAR thruster is included in this paper. Comparisons of this model to various experiments are also included.

NOMENCLATURE

\[ A_i = \text{area of } i^{th} \text{ element} \]
\[ A_J = \text{area of } J^{th} \text{ element} \]
\[ C_i = \text{thermal capacitance at node } i, \text{ cal}^\circ \text{C} \]
\[ F_{ij} = \text{form (view) factor} \]
\[ G_{ji} = \text{linear conductor attaching node } j \text{ to node } i, \text{ W}^\circ \text{C} \]
\[ G_{ji} = \text{radiation conductor attaching node } j \text{ to node } i, \text{ W}^\circ \text{C}^4 \]
\[ J_A = \text{ion current hitting grid, A} \]
\[ J_B = \text{ion beam current, A} \]
\[ N = \text{number of nodes} \]
\[ Q_i = \text{heat source or sink for node } i, \text{ W} \]
\[ r_{ij} = \text{distance between the } i^{th} \text{ and } j^{th} \text{ element} \]
\[ T_j^k = \text{temperature of node } j \text{ for the } k^{th} \text{ iteration, } ^\circ \text{C} \]
\[ T_i^{k+1} = \text{temperature of node } i \text{ for the } k+1 \text{ iteration, } ^\circ \text{C} \]
\[ T_j^n = \text{temperature of node } j \text{ at time } t, ^\circ \text{C} \]
\[ T_i^{t+\Delta t} = \text{temperature of node } i \text{ at time } t+\Delta t, ^\circ \text{C} \]
\[ U_+ = \text{ionization energy, eV} \]
\[ V_P = \text{plasma potential, V} \]
\[ \Phi_{sh} = \text{energy deposited in the form of heat, W} \]
\[ \Phi_T = \text{total thruster power, W} \]
\[ \Phi_N = \text{neutralizer power, W} \]
\[ \theta_i = \text{angle between normal of } i^{th} \text{ element and the line connecting the } i^{th} \text{ and } j^{th} \text{ element} \]
\[ \theta_j = \text{angle between normal of } j^{th} \text{ element and the line connecting the } i^{th} \text{ and } j^{th} \text{ element} \]

INTRODUCTION

The 30-cm-diameter ring cusp NSTAR ion thruster represents the state-of-the-art in ion thruster technology. Ion thrusters have long been known to have the highest efficiency at high specific impulse of all electric propulsion devices. The combination of high power utilization at specific impulses in excess of 3,000 seconds has made the ion engine an attractive candidate for high delta-V planetary missions by potentially reducing initial launch costs and decreasing trip times. With its high specific impulse, long life, and high efficiency, the NSTAR thruster is ideal for a number of deep-space mission applications.

Despite these advantages, however, application of ion propulsion to scientific, military, and commercial spacecraft was hampered in the past by high perceived engine development costs and the inability of spacecraft manufacturers to reliably identify potential integration and thruster lifetime issues. The primary concerns that spacecraft manufacturers had in regards to using ion propulsion included the likely impact of thruster operation on spacecraft design and operations, electromagnetic compatibility, spacecraft contamination from thruster efflux, spacecraft damage from the plume, thruster reliability, and thermal loading of the spacecraft from the thruster. Ion propulsion became (and will continue to become) more attractive once tools were
developed (e.g., plume PIC codes) which helped spacecraft manufacturers identify potential spacecraft integrating issues associated with this technology.

Given the wide range of thermal environments an ion thruster on a deep-mission will likely encounter, it is essential that computer tools are developed to predict the temperatures of thruster components over the expected range of operating and thermal conditions. Some critical areas of concern include the degaussing of permanent magnets from excess heating, freezing of xenon in propellant lines, distortion of the ion optics from thermal gradients, and spacecraft integration issues in general (e.g., thermal soakback). Although work has been done in the past on modeling the thermal behavior of 20-cm-diameter and 30-cm-diameter ion thrusters utilizing mercury propellant, no thermal model has been developed for the NSTAB engine. The most recent approaches used to develop the numerical models started with previous work containing the self-heating of the thruster from the plasma and then used data from experiments to adjust the numerical model to fit the temperatures. This is the approach which was used to determine the self-heating terms on the NSTAR thruster. However, analytical work is in progress to determine self-heating values independent of experimental data. The tests used for calibrating this NSTAB thruster model were based on experiments performed at NASA Lewis Research Center in June and July of 1996.

This calibrated model can then be used to investigate other operating conditions. It has already been used to alert of the possible dangers of overheating the magnets. Other issues investigated but not presented here include enclosing the thruster in an adiabatic surface, changing materials on the thruster, and the influence of space conditions on the thruster.

**SETUP OF MODEL**

**Thermal Model**

There are two major modes of heat transfer which take place in the NSTAR thruster. Although the dominant process is radiation heat transfer, conduction plays a major role in establishing thruster component temperatures. The interaction of the plasma with the thruster will be discussed later. In order to handle a model of significant size and to study the thermal response of the thruster to various steady-state and periodic external radiation loads over its full range of operating conditions, a computer model was utilized using two well-used codes.

SINDA (Systems Improved Numerical Differencing Analyzer) analyzes thermal systems represented in electrical analogy, lumped parameter form. The "conductors" based on the conductive and radiative properties of the system are calculated between nodes and then included in the SINDA input file. The equation used for steady state analysis in SINDA is:

\[ 0 = Q_i + \sum_{j=1}^{i-1} \left[ G_{ij}(T_j^{k+1} - T_i^{k+1}) + G_{ji} \left( T_j^{k+1} \right)^4 - \left( T_i^{k+1} \right)^4 \right] \]

\[ + \sum_{j=1}^{N} \left[ G_{ij}(T_j^{k+1} - T_i^{k+1}) + G_{ji} \left( T_j^{k+1} \right)^4 - \left( T_i^{k+1} \right)^4 \right] \]

which is solved by a "successive point" iterative method. The transient equation used is based on an implicit forward-backward differencing method:

\[ \frac{2C_i}{\Delta t} \left( T_i^{n+1} - T_i^n \right) = \]

\[ 2Q_i + \sum_{j=1}^{N} \left[ G_{ij}(T_j^n - T_i^n) + G_{ji} \left( T_j^n \right)^4 - \left( T_i^n \right)^4 \right] \]

\[ + \sum_{j=1}^{N} \left[ G_{ij}(T_j^{n+1} - T_i^{n+1}) + G_{ji} \left( T_j^{n+1} \right)^4 - \left( T_i^{n+1} \right)^4 \right] \]

For Equations (1) and (2) the radiation terms are linearized before solution routines are initiated.

The second piece of software used is TRASYS (Thermal Radiation Analyzer SYstem). TRASYS uses geometry and surface characteristics to provide radiation conductors for SINDA. TRASYS computes the radiation view (shape) factors using the Nusselt Sphere and double summation techniques. Both of these calculation methods are based on the equation:

\[ F_{ij} = \frac{1}{A_i} \int \int \frac{\cos \theta_i \cos \theta_j}{\pi_2} \cdot \frac{1}{A_j} \cdot dA_i \cdot dA_j \]

which gives the form factor for two finite areas.

The NSTAR model contains 104 thruster nodes with conductors connecting the nodes for conduction and radiation heat transfer. The thruster is essentially broken up into 4 quadrants. However, two of the quadrants are further subdivided in half to
accommodate the gimbal pads. Since the neutralizer has been shown to be insignificant in its thermal impact to the thruster\(^5\), a simplified model of it was used. Figure 1 shows the nodal layout of the thruster.

The tests which were used to calibrate the NSTAR model took place at the NASA Lewis Research Center (LeRC). The experimental setup at LeRC included the thruster enclosed within a 116 cm-diameter liquid nitrogen cooled shroud contained in a 4.6-m-diameter by 19.5-m-long vacuum chamber. The model used temperature measurements along the shroud and tank walls to establish boundary conditions. These boundary nodes consisted of 37 nodes making up the shroud and experimental setup, and 6 nodes for the tank wall. The thruster was modeled as being isolated from the shroud and its test stand. The model does not include feed lines, electrical lines, or the isolator box as those are predicted to have minimal impact on the thermal characteristics of the thruster. Figure 2 shows the shroud/thruster setup in the model.

The major form of heat transfer within the thruster is radiation. This underscores the importance of using accurate surface property values. Also, changing materials or surface properties could modify the thermal characteristics of the thruster significantly. These properties could also change over the life of the thruster further complicating matters.

For this model the emissive values were assumed to be constant throughout the temperature range examined. For most materials this is a valid assumption for the conditions seen by the thruster. The values used also corresponded to the infrared temperature range. These were obtained from published sources and also from sampling parts of the NSTAR thruster (Table I).
Table I - Assumed Physical Properties of Ion Thruster Materials

<table>
<thead>
<tr>
<th>Material</th>
<th>Density g/cm³</th>
<th>Conductivity W/cm°C</th>
<th>Emisivity</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aluminium 5052</td>
<td>2.77</td>
<td>0.20</td>
<td>1.80</td>
</tr>
<tr>
<td>Pure Titanium</td>
<td>4.43</td>
<td>0.15</td>
<td>0.23</td>
</tr>
<tr>
<td>Carbon Steel</td>
<td>7.81</td>
<td>0.13</td>
<td>0.60</td>
</tr>
<tr>
<td>304 Stainless Steel</td>
<td>7.92</td>
<td>0.125</td>
<td>0.20</td>
</tr>
<tr>
<td>Molybdenum</td>
<td>10.19</td>
<td>0.20</td>
<td>1.20</td>
</tr>
<tr>
<td>Tantalum</td>
<td>16.16</td>
<td>0.035</td>
<td>0.60</td>
</tr>
<tr>
<td>Tungsten</td>
<td>19.38</td>
<td>0.035</td>
<td>1.50</td>
</tr>
<tr>
<td>Alumina (Al₂O₃)</td>
<td>3.79</td>
<td>0.20</td>
<td>0.17</td>
</tr>
<tr>
<td>Kovar</td>
<td>8.36</td>
<td>0.105</td>
<td>0.15</td>
</tr>
<tr>
<td>6Al-4V Titanium</td>
<td>4.43</td>
<td>0.15</td>
<td>0.10</td>
</tr>
</tbody>
</table>

*Grit blasted surface.
Meshed surface

Another surface characteristic which had to be modeled dealt with the perforated surfaces present. TRASYS was not designed to model perforated surfaces. To approximate these surfaces transmissive values were assigned to allow the appropriate percent of the incident energy to pass through. The value used for the transmissivity corresponded to the open area fraction of the perforated surface. However, it was not clear how accurately this assumption was for modeling these surfaces. It is thought that transmissive surfaces in series will artificially block radiation which would normally travel through the aligned open areas of two perforated surfaces.

For comparison the experiments which involved cold soaking the thruster modeled the optics using transmissive surfaces and also a course checkered surface with alternating areas of no material. The layout of the checkered surfaces in TRASYS is shown in Figure 3. Discussion of perforated surfaces will follow later in the paper.

Other properties which affect the model deal with conduction in the thruster. While of lesser influence than the radiative characteristics, they are of importance. Most conductivities of materials present in the thruster remain approximately constant over the temperature range experienced. The values were taken from various published sources and are also given in Table I. The value used for joint conductances was 0.0057 W/cm²°C and is based on experiment.

![Checkered Pattern used to Model Ion Optics in TRASYS](image)

**Self Heating due to Plasma Interaction**

One of the more difficult parts of the thruster model is determining the amount of heat which is applied due to the plasma interaction the surfaces. In order to determine analytically the amount of heat that is produced by the plasma, several characteristics need to be well understood. One of which is the precise location of the deposition of charged particles on the various surfaces. The current produced by these particles and their corresponding temperatures are also relevant. Work is underway to determine this analytically.

For this paper a method which was used in past work was applied. This method entailed using previous heat flux data from past work and then adjusting the values until the temperatures in a model corresponded with the experimental data. Once the self-heating values were adjusted to correlate the temperatures to the experiment an overall comparison of self-heating was done analytically. The total heat applied was determined by taking the total energy added to the system and subtracting out the energy which exited the thruster in the beam. Equation 4 shows this energy balance.

\[
\Phi_{sh} = \Phi_T \cdot \Phi_N \left( \frac{J_B - J_A}{V_p + U_+} \right)
\]

The adjusted values of self heating are shown in Figure 4 for the high power 2.3 kW case.
Other power levels were investigated, but the high power case was of most concern since it led to the most extreme temperatures. However, to examine the model without the influence of the plasma, a case representing the thruster being cold-soaked is presented.

**RESULTS AND DISCUSSION**

The computer model will first be compared to the cold soak test which was done at LeRC on June 18, 1996. The shroud or enclosure which is shown in Figure 2 was cooled via liquid nitrogen flowing through tubing which encompassed its surface. The open end of the shroud was closed off by a door which was cooled by the rest of the shroud through radiation since it was not connected to the sections cooled by the liquid nitrogen. A donut or ring piece can be seen in the shroud which was located close to the front of the thruster face (optics end) and was cooled through conduction with the cylindrical part of the shroud. It was used to minimize the amount of the thruster which viewed the vacuum tank wall which was at room temperature.

The temperatures for the boundary conditions in the model consist of the experimentally determined shroud temperatures. A total of 37 nodes were used in modeling the shroud and other experimental setup such as the test stand. The shroud was painted with a commercial, high-temperature, fireplace flat black paint which has a measured emissivity of 0.9.

Figure 5 shows a cross section of the NSTAR thruster with the temperatures determined experimentally and via the SINDA computer model. There are two temperatures derived from the computer model which correspond to different approaches to modeling the optics (checkered and transmissive).

The SINDA model accurately predicted all thermocouple values within 5 °C except at three nodes. One of those three, the neutralizer tip, is within 6 °C. The other two are on the edge of the mask and front edge of the thruster. Both of these predictions are within 15 °C of the data. Two locations are shown in the figure as being double boxed.

This model is important since it shows the accuracy in predicting the thruster temperatures without the influence of the plasma when the thruster is operating. It is difficult to determine the discrepancy of the temperatures in the mask area. This may reflect the difficulty in determining the contact resistance between the mask and the rest of the thruster.

The effect of changing the method of modeling the optics appears minimal in this case. Most of the temperatures changed by only a degree or two Celsius. The most drastic change in temperature was in the optics and this was only by 2-3 °C. This would indicate that modeling the surface as transmissive is sufficient.

The NSTAR thruster has also been modeled in SINDA to predict transient data. Figure 6 and 7 show a comparison between experimentally determined data on February 28, 1996 and the SINDA model with the optics modeled as transmissive surfaces.

The predicted results from SINDA agree to within 10 °C for all of the nodes except 112 (mask), 400 (neutralizer rear), 102 and 104 (plasma screen). The areas of greatest discrepancy tend to be along the plasma screen and mask. This represents the difficulty in determining some of the contact resistances in the system and the modeling of
Figure 6 - Transient Cold Soak Experiment (2/28/96) Compared to SINDA model of NSTAR Thruster

Figure 7 - Transient Cold Soak Experiment (2/28/96) Compared to SINDA model of NSTAR Thruster Discharge Chamber
perforated surfaces. It should be noted that this data is from an older version of the SINDA model, but it still is representative of the current model.

Figure 7 shows that the agreement of temperatures in the discharge chamber area is very good. The temperatures follow within 5 °C throughout the test. This is crucial to have accurate since most of the components in the thruster of concern are on or near this surface. It also shows that the discharge chamber is interacting with its surroundings as in the experiment. So even though the temperatures of some outer components such as the plasma screen may be less accurate, their impact on the discharge chamber is minimal.

The next step is to examine an operating thruster. Figure 8 gives the temperatures on the NSTAR thruster when it was operating at 2.3kW on July 3, 1996 at LeRC. It also gives the temperatures for the SINDA model for both types of optic surface representations.

As mentioned prior initial values of self heating were used and then adjusted to correspond to the experimental data. Those adjusted values were given in Figure 4. That method resulted in 331.5 Watts being applied to the thruster. After subtracting 28 W used by the neutralizer the thruster has 303.5W of applied heat.

Using equation 4 where $\Phi_T=2274$ W, $\Phi_{N}=23$ W, $I_B=1.75$ A, $J_A=0.01$ A, $V_p=1100$ V, and $U_b=12.13$ A results in the applied heat flux being 316 W. The 303.5 W derived from the model is within 4% of the calculated value and shows good agreement.

The temperatures in the discharge chamber are within 5 °C of the experimental data. However, the temperatures along the plasma screen and mask are off by a considerable amount. The discrepancy is most likely caused by: (1) the difficulty in modeling a finely perforated surface; and (2) modeling contact resistance. The coupling between the discharge chamber and the plasma screen is through isolators which have a high number of contact points.

But as shown earlier, the interaction between the discharge chamber and the environment is accurate. Therefore, this model will give an accurate prediction of the discharge chamber and its components under varying conditions. This is supported by the amount of energy supplied for the model and analytically.

Once the model is calibrated it can be used to predict various situations. One of the major concerns deals with directional heat flux. Primarily how much heat will be directed towards a satellite. To give an idea of the direction the heat flows from the thruster a model was run which encapsulated the thruster in a box. This setup and results are shown in Figure 9. The box was given space conditions; i.e. temperature of -273 °C and an emissivity of 1.0. The thruster was then given the heat distribution corresponding to the 2.3 kW power case.

It can be seen that a majority of the heat is expelled through the front of the thruster out of the optics. It should be noted that the effect of the plasma is only included as the heat applied to the thruster and the power out the front of the thruster does not include the energy in the beam. The sides of the thruster are fairly uniform in power distribution with the back having the lowest amount of heat flux.

However, these values would change if an object of different temperature were on a side. If a satellite were behind the thruster with a much higher temperature than space it would drastically reduce the amount of heat flux in that direction.
CONCLUSION

The SINDA model has been developed and accurately describes the NSTAR thruster discharge chamber and components within 10 °C. There is a larger discrepancy with the temperatures on the plasma screen and mask. However, it has been shown that this has minimal affect on the temperatures of the discharge chamber and its components. There is still an accurate representation of the interaction between the inner surfaces and the environment.

Changing the discharge chamber whether it be by a material change or a change in its layout will have the greatest affect on the temperatures in the thruster. The plasma screen and neutralizer are of lesser impact on the thruster thermal environment.

Limitations of the model include the approximating of perforated surfaces. Currently there are no tools available to help model finely perforated surfaces. Not only is the determining of radiation view factors more difficult through and to the surface, calculating the conduction along the material is also more difficult. Some work has been done to further approximate the perforated surface. The two methods used here were modeling the surface as having a transmissivity equal to the open area and the other was to create a rough checkered pattern of open area. Other work was done but not presented here to show that a rough checkered pattern would give an accurate representation of a pattern with higher resolution.

Other limitations include the determining of contact resistance between parts. While the dominant form of heat transfer is radiation, conduction does play a role. It was shown that contact resistance plays a significant role in the connection of the discharge chamber to the plasma screen.

The self heating terms were developed from experimental data. Further work is being done to determine analytically these terms for various cases.

The model is now capable of being integrated into various environments. It can be used to look at spacecraft integration issues and design changes. Changing the surface characteristics on the thruster and specifically the discharge chamber can have a large effect on changing the temperatures of the thruster.

REFERENCES