Thermal Design of a Solar Thermal Thruster for Piggyback Satellites

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A method of thermal analysis for a solar thermal thruster was created to aid in the thermal design of the thruster. The method consists of two types of thermal analysis: an analysis program for propellant flow, and an analysis of the temperature distribution of the thruster wall using Pro/Engineer. The numerical results were compared with experimental results to confirm the validity of the method, and there was good agreement between them. A thermal design was created using this thermal analysis method to estimate the performance of a solar thermal thruster for the orbital transfer of piggyback satellites mounted on an H2A rocket. When the thruster is made from heat-resistant steel and the propellant is water, the analytical results showed that the Isp is 203 s, the thrust is 16.6 mN, and the maximum temperature of the thruster is 1088 K. The diameter of the concentrator also was calculated, and it was found to be small enough for the concentrator to be mounted on piggyback satellites.

Key Words: Solar Thermal Propulsion, Thermal Analysis, Piggyback Satellite

Nomenclature

\begin{align*}
A & : \text{area, m}^2 \\
a_1 & : \text{constant for calculation of thermal conductivity} \\
a_2 & : \text{constant for calculation of thermal conductivity} \\
b & : \text{constant for calculation of viscosity, Pa}\cdot s/K^{1/2} \\
C & : \text{Sutherland’s constant, K} \\
C_p & : \text{isobaric specific heat, J/(kg}\cdot K) \\
C_p^0 & : \text{isobaric specific heat of ideal gas, J/(kg}\cdot K) \\
d_e & : \text{equivalent diameter, m} \\
F & : \text{geometric factor, -} \\
h & : \text{heat transfer coefficient, W/(m}^2\cdot K) \\
h_i & : \text{enthalpy, J/kg} \\
H-H^f & : \text{enthalpy departure function, J/kg} \\
L & : \text{latent heat, J/kg} \\
m & : \text{mass flow rate, kg/s} \\
Nu & : \text{Nusselt number, -} \\
P & : \text{pressure, Pa} \\
\dot{q} & : \text{heat flux, W/m}^2 \\
\dot{Q} & : \text{heat flow rate, W} \\
R & : \text{gas constant, J/(kg}\cdot K) \\
S & : \text{surface tension, N/m} \\
T & : \text{temperature, K} \\
u & : \text{velocity, m/s} \\
x & : \text{position from the inlet of a feed pipe, m} \\
X & : \text{quality, -} \\
\varepsilon & : \text{emissivity, -} \\
\lambda & : \text{thermal conductivity, W/(m}\cdot K) \\
\mu & : \text{viscosity, Pa}\cdot s \\
\rho & : \text{density, kg/m}^3 \\
\sigma & : \text{Stefan – Boltzmann constant, W/(m}^2\cdot K^4) \\
\text{Subscripts} \\
b & : \text{boiling point} \\
bot & : \text{bottom of rectangle} \\
f & : \text{surface}
\end{align*}

1. Introduction

A solar thermal propulsion (STP) system is proposed as a new-generation propulsion system for orbital maneuvering. In the STP, a propellant is heated by concentrated sunlight through the wall of the thruster. The STP uses a concentrator to concentrate sunlight, enabling the STP to heat the propellant to a high temperature without using chemical reactions. Figure 1 shows a schematic of the STP.

Recently, there has been an increase in the development of microsatellites by universities and small-medium businesses. A piggyback system is the most practical launch system for microsatellites. In the piggyback system, microsatellites are mounted in an open payload space. The piggyback satellites cannot have any flammable propellants onboard, as this might have an adverse effect on the performance of the main
satellite. As the orbit of the piggyback satellite is constrained by the orbit of the main satellite, the STP gives the piggyback satellites orbital maneuvering capabilities such as deorbit and altitude control. The STP provides for flexible mission designs for piggyback satellites.

It is necessary to provide a large amount of energy to the propellant in order to obtain a high specific impulse (Isp). In the solar thermal thruster, solar energy is transferred to the propellant through the thruster wall. It is essential to know the temperature distribution of the thruster wall in order to estimate the thrust, and the specific impulse of the thruster. Some ground testing of the STP system has been carried out\(^1\). However, it is difficult to simulate a space environment on the ground, and the tests have been performed only in conditions where there was abundant sunlight. A comprehensive thermal analysis of the solar thermal thruster is necessary to accelerate its development.

In this paper, a thermal analysis method that analyzes the temperature distribution of the solar thermal thruster and the propellant flow is constructed. A thermal design for the solar thermal thruster is then attempted using this method. It is assumed that the STP system is mounted on a piggyback satellite atop an H2A rocket. The piggyback satellite is a 500-mm cube with a mass of 50 kg. The solar thermal thruster uses water as a propellant, and heat resistant-steel as the main construction material.

2. Thermal analysis method

This analysis method is based on the assumption that the propellant is water. The advantages of using water are that it is easy to handle and that it exists in the liquid phase at room temperature. The method consists of two forms of thermal analyses. One is an analysis program that calculates the distribution of the temperature, pressure, density, and heat transfer coefficient of the propellant. The other is an analysis of the temperature distribution of the thruster wall made using Pro/Engineer, a 3-D CAD-analysis software program.

2.1. Configuration of flow channel

Figure 2 shows the configuration of the flow pipe. The propellant is provided via a feed pipe. In the thruster, the propellant flows into the spiral channel where it is heated. Then, the propellant flows into the Laval nozzle, where it expands and accelerates.

2.2. Assumptions in analysis of propellant flow

Assumptions in the analysis of propellant flow are as follows:

**Assumptions in common**
- The flow is a one-dimensional, steady-state flow.
- A boundary layer is fully developed.
- Only convection heat transfer is involved.
- The influence of pipe friction is neglected.
- Ambient pressure is zero.
- There is no chemical reaction or deposition.

**Assumptions in the liquid-phase flow**
- The flow is laminar. In this analysis, the maximum value of the Reynolds number in the liquid phase flow is 42. Thus, this assumption is appropriate.
- The propellant pressure is constant, and any pressure loss is neglected.
- When the propellant temperature reaches saturation temperature, boiling begins.

**Assumptions in the two-phase flow**
- Nuclear boiling and forced convection boiling are considered as boiling phenomena.
- Propellant pressure is constant, and pressure loss is neglected.
- The velocity of the liquid and gas is the same.
- Only when the degree of superheat between the wall and the propellant satisfies the condition of the generation of bubbles, the boiling phenomenon is in nuclear boiling.

**Assumption in the gas-phase flow**
- Any influence of axial heat conduction is neglected.
- The flow is continuous flow. In this study, the maximum value of Knudsen number is $5.0 \times 10^{-2}$.

2.3. Assumptions in analysis of wall temperature

Assumptions in the analysis of the wall temperature of the thruster are as follows:
- The thruster is exposed to a space environment.
- Concentrated sunlight irradiates uniformly on the bottom of the cavity.
- Any heat exchange in the flow pipe due to radiation is neglected.
- As for the feed pipe, the temperature of the inlet is constant and is the same as the inlet propellant temperature.

2.4. Analysis program for propellant flow

Based on these assumptions, we constructed an analysis program for propellant flow. In the flow channel, the propellant changes from the liquid-phase to the gas-phase via the two-phase flow. A thermal model is applied to each phase.
Liquid-phase flow

A basic equation of the propellant is the law of conservation of energy, that is,

\[
\frac{d}{dx} \left( \rho u A_c \right) + \frac{d}{dx} \left( \rho u^2 A_c \right) + \frac{d}{dx} \left( \frac{\rho u}{A_c} \right) + 1 \frac{dQ}{dx} = 0
\]  

(1)

In the analysis program, Eq. (1) is solved using TDMA23.

Two-phase flow

The temperature of the propellant is the same as the saturation temperature. A quality is a parameter that represents the process of boiling. The quality is defined as the following equation.

\[
X_j = \frac{\dot{Q}_{h-x,j}}{\dot{m} L} = 0
\]  

(2)

When the quality becomes 1, boiling is completed.

The relationship between a critical heat flux in which a bubble occurs and the degree of superheat is shown below.

\[
\dot{q}_{inc} = \frac{\dot{Q}_{h-x}}{ST_h} \left( T_w - T_s \right)_{inc}^2
\]  

(3)

In a case where the degree of superheat in the channel exceeds \((T_w - T_s)_{inc}\) in Eq. (3), it is assumed that nuclear boiling begins. Then, if the degree of superheat becomes less than \((T_w - T_s)_{inc}\), nuclear boiling is constrained and a transition to forced convection boiling occurs.

Gas-phase flow

The basic equations are one-dimensional Euler equations.

\[
\frac{d}{dx} \left( \rho u A_c \right) = 0
\]  

(4)

\[
\frac{dP}{dx} + \rho A_c \frac{du}{dx} = 0
\]  

(5)

\[
u \frac{du}{dx} + \frac{d}{dx} \left( \frac{b_h + \dot{Q}/\dot{m}}{A_c} \right) = 0
\]  

(6)

\[
P = \rho RT_p
\]  

(7)

Eqs. (4) to (7) are mass conservation, momentum conservation, energy conservation, and state equation, respectively. Colonna’s method is adopted as the solution method30.

2.5. Calculation of heat transfer coefficient

In single-phase flow, the heat transfer coefficient can be calculated from the Nusselt number.

\[
Nu = \frac{h_d}{\lambda}
\]  

(8)

In the laminar flow, based on the assumption that the flow is a fully developed laminar flow, \(Nu\) is constant in each section of the channel. In the gas-phase flow, the Reynolds number becomes larger than the transition Reynolds number. If the Reynolds number exceeds 3000, \(Nu\) is calculated from Gnielinski’s equation40, which is used to calculate the heat transfer coefficient in the fully developed turbulent flow.

In the two-phase flow, Kutateladze’s equation is used in the nuclear boiling and Dengler-Addoms correlation is used in forced convection boiling to obtain the heat transfer coefficient41. This equation includes Dittus-Boelter’s equation, which is used in turbulent flow. We assumed that the forced convective boiling is in the annular flow. In the present work, the disturbance of the liquid film may occur in the annular flow.

2.6. Calculation of physical properties

In the liquid-phase and the two-phase flow, we used the program attached to JSME steam tables to calculate physical properties31. In the gas phase, the propellant expands and the pressure exceeds the range of the program from the JSME steam tables. Thus, we introduced the following approximate formulas.

Isobaric specific heat

Isobaric specific heat is described in Eq. (9).

\[
C_p = C_{p0} + \frac{\partial}{\partial T} \left( H - H_0 \right)_T,\text{composition}
\]  

(9)

The equation of \(C_{p0}\) given in Ref. 6 was used and \(H-H_0\) was calculated from Redlich-Kwong equation80.

Viscosity

Sutherland’s equation is used32.

\[
muc = \frac{b(T)}{1 + C/T_p}
\]  

(10)

The known value of the viscosity at 273.16 K and 0.61166 Pa is used to obtain the value of \(b\) in Eq. (10).

Thermal conductivity

We used the equation shown below30.

\[
\lambda = \frac{a_1 + a_2 T_p}{1 + C/T_p}
\]  

(11)

The known values of viscosity at 323.16 K and 0.61166 Pa, and 273.16 K and 0.61166 Pa are used to obtain the values of \(a_1\) and \(a_2\) in Eq. (11).

2.7. Representative value of wall temperature

The flow channel has a different wall temperature not only in the flow direction, but also in the circumferential direction. The representative value of the wall temperature with a rectangle and concentric annulus of the cross section is:

\[
T_w,\text{annulus} = \frac{T_{w,\text{out}} A_{S_{out}} + T_{w,\text{in}} A_{S_{in}}}{A_{S_{out}} + A_{S_{in}}}
\]  

(12)

\[
T_w,\text{rect} = \frac{W \left( T_{w,\text{top}} + T_{w,\text{bottom}} \right) + H \left( T_{w,\text{left}} + T_{w,\text{right}} \right)}{2W + 2H}
\]  

(13)

2.8. Analysis of wall temperature of thruster

We used Pro/Engineer to analyze the temperature distribution on the wall of the thruster.

The values input into Pro/Engineer are as follows:

- Input energy from the bottom of the cavity.
- Bulk temperature of the propellant.
- Heat transfer coefficient in the flow channel.
- Heat loss from the surface of the thruster, cavity, nozzle, and feed pipe.

The heat losses are calculated as follows:

Radiation losses

Radiation losses from the thruster surface, cavity, and nozzle are considered. The radiation loss is expressed as Eq. (14).

\[
\dot{Q}_{\text{loss}} = \varepsilon \sigma A_c F T_r^4
\]  

(14)

The view factor when calculating radiation loss from the thruster surface is 1. As for the cavity and nozzle, we used the view factor of the cylinder and the circular truncated cone.

Heat loss from propellant feed pipe

The axial heat flux obtained by using Pro/Engineer is used. The heat loss from the feed pipe is given by the produce of the
heat flux and the cross-sectional area of the feed pipe. This heat loss is added to the heat flow rate provided in the cavity.

2.9. Sequence of thermal analysis
1. The mass flow rate and temperature at the inlet of the feed pipe is decided.
2. Wall temperature distribution is assumed.
3. Under the assumed temperature distribution of the wall, an analysis of the propellant in the flow channel is performed.
4. Iterative calculations are performed until the inlet pressure required to choke the flow in the nozzle is obtained.
5. The heat flow rate, heat transfer coefficient, propellant temperature obtained from the analysis of the propellant flow are input into Pro/Engineer, and the distribution of the wall temperature is obtained.
6. The wall temperature used in the analysis of the propellant flow is modified.
7. The heat losses described in Sec. 2.8. are calculated and an analysis using Pro/Engineer is performed again.
8. Iterative calculations from 2. to 7. are performed until the thruster wall temperature converges.

3. Validation of thermal analysis method
To validate the thermal analysis method, a comparison of the experimental data and analysis results was made. The experiment was performed in a vacuum chamber. Instead of using sunlight, a heater was utilized as the heat source. The thruster used in the experiment was constructed from a corrosion-resistant aluminum alloy (AS056).

Figure 3 shows the measurement points of the temperature in the experiments. Tables 1 and 2 show the typical dimensions of the thruster and the conditions of the experiments. Tables 3 and 4 show comparisons of temperatures obtained in the experiments and in the analysis. Both results correspond with each other within 6%. Thus, this analysis method has adequate validity.

4. Target conditions of thermal design
From the thermal analysis method, the thermal design of the solar thermal thruster is created.

4.1. Thruster material and propellant
Heat-resistant steel (SUH310, 25 Cr–20 Ni steel) was selected for the construction of the thruster. Table 5 shows the physical properties of the steel.

Water was selected as the propellant, which is the same propellant used in the experiment.

4.2. Target Isp
The target Isp is defined in this section. Figure 4 shows the relation of the Isp and mass requirement under the conditions of a satellite with a mass of 50 kg and a velocity change of 1 km/s. In this paper, 200 s of Isp is defined as the target specific impulse.

<table>
<thead>
<tr>
<th>Table 3. Comparison of experiments and analysis A</th>
</tr>
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<tbody>
<tr>
<td>Plenum Temperature, K</td>
</tr>
<tr>
<td>Front</td>
</tr>
<tr>
<td>Side 1</td>
</tr>
<tr>
<td>Side 2</td>
</tr>
<tr>
<td>Nozzle</td>
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</table>

<table>
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<tr>
<th>Table 4. Comparison of experiments and analysis B</th>
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<tbody>
<tr>
<td>Plenum Temperature, K</td>
</tr>
<tr>
<td>Front</td>
</tr>
<tr>
<td>Side 1</td>
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<tr>
<td>Side 2</td>
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<tr>
<td>Nozzle</td>
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<table>
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<tr>
<th>Table 5. Physical properties of 25 Cr-20 Ni steel</th>
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</thead>
<tbody>
<tr>
<td>Density, kg/m³</td>
</tr>
<tr>
<td>Specific heat</td>
</tr>
<tr>
<td>Thermal conductivity, W/(m·K)</td>
</tr>
<tr>
<td>Emission ratio</td>
</tr>
<tr>
<td>Allowable temperature limit, K</td>
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</tbody>
</table>

Fig. 4. Relation of specific impulse to propellant mass with a satellite mass of 50 kg and a velocity change of 1 km/s
5. Results and discussions

5.1. Thruster configuration

Figure 5 shows the cross section of the thruster. Table 6 shows the dimensions of the flow channel. Some modifications were made to the thruster used in the experiment, including expanding the cavity to catch more sunlight, shortening the spiral channel, omission of one side of the feed pipe, and expanding the expansion ratio. The thruster has three section shapes. Table 7 shows the Nusselt number for each section shape.

5.2. Calculation results of thruster performance

Tables 8 and 9 show the values at the inlet of the feed pipe and the performance of the thruster. A total of 203 s of Isp was obtained. From Fig. 4, it can be seen that 20 kg of propellant is required to obtain 1 km/s of velocity change.

5.3. Results of heat transfer in thruster wall

Figures 6 and 7 show the distributions of the temperature and heat flux in the thruster, respectively. The maximum temperature of the thruster appears at the bottom of the cavity, where the value is 1088 K. This satisfies the allowable temperature limit shown in Table 5. The heat from the bottom of the cavity flows to the feed pipe where the propellant is boiling via the inner section of the thruster, and the little heat is transferred to the outer section of the thruster, especially in the convergent nozzle. Thus, the temperature of the outside of the thruster is above 200 K, low compared to the bottom of the cavity. It is necessary to transfer the heat to the lower wall of the thruster in order to heat the propellant effectively in the expansion region.

5.4. Result of analysis of flow channel

Figure 8 shows the distribution of the propellant temperature and heat transfer coefficient. The propellant forms a two-phase flow in the vicinity of the peak at the end of the feed pipe. Another peak in the heat transfer coefficient appears at the nozzle throat. In the spiral channel, the propellant temperature increases and decreases. Cooling at one side in the thruster causes this temperature change in the propellant. The maximum temperature of the propellant is 949 K. Compared to the maximum thruster wall temperature, the propellant temperature is lower by more than 100 K.

Figure 9 shows the distribution of the heat flux to the propellant. The change in the heat flux in the feed pipe is caused by taking an approximate wall temperature from Pro/Engineer. The propellant is cooled in the region where the line of heat flux is disconnected.

5.5. Heat balance in thruster

Table 10 shows the heat balance of the thruster. The heat loss is 42.6% of the heat input from the concentrator. Most of the heat loss occurs on the surface of the thruster.
5.6. Concentrator

Thermal analysis shows that the solar concentrator needs to focus 60.3 W of sunlight. Figure 10 shows the parameters of the solar concentrator. It is supposed that the concentrator is a two-dimensional parabolic mirror with an aluminum surface. The dimensions of the concentrator are shown in Table 11. In this thermal design, the size of the piggyback satellite is assumed to be a 500-mm cube. Thus, the diameter of the concentrator is small enough to mount on the satellite. However, the large focal length requires a deployment method.

6. Conclusion

The experimental and the numerical analysis results are consistent with each other. Thus, the validity of this method of thermal analysis is verified.

A thermal design for a solar thermal thruster for piggyback satellites has been achieved. In the case where the material of the thruster is a heat-resistant steel and the propellant used in the thruster is water, an Isp of 203 s is obtained analytically. The maximum temperature of the thruster is 1088 K. The expansion ratio of the nozzle is 225, and the throat diameter is 0.4 mm.

In the case where the concentrator is a two-dimensional parabolic mirror, the diameter of the concentrator is small enough to mount on piggyback satellites. However, the focal length is large. Thus, a deployment method is required.

References

5) The Japan Society of Mechanical Engineers.: JSME Steam Tables, JSME, Tokyo, 1999 (in Japanese)