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A Thermal Analysis of the ARISE IPT Europa Lander

by

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Logain North

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Student Signature

1/28/23

Date
Abstract

This report is a thermal study of the ARISE IPT Lander designed for a mission to Europa, Jupiter’s icy moon. A brief historical study is done on previous NASA lander missions. A spherical assumption parametric analysis was conducted to determine thermal properties of the designed lander and how those properties can be affected by vehicle size. The study concluded that the design condition temperatures dominate the needs of the spacecraft at the desired spacecraft size. It also concluded that an active pumped heat transfer system is not required for this design.
Introduction

When a spacecraft is sent into space, it is a complex balance of multiple component systems. The most important system is the payload system, which is used to conduct scientific analysis to complete the spacecraft’s scientific mission objectives. However, to complete these objectives, the spacecraft must consist of supporting systems to keep the spacecraft operational. These systems include communications, data handing, power, propulsion, structures, altitude determination and control, and thermals. The researcher oversaw the thermal control system of the vehicle. The vehicle in question was designed to meet the lander objectives of Autonomous Research Investigating the Surface of Europa (ARISE) mission.

This report will begin with a brief overview of what ARISE is and the goal of that program. Then the report will explain the importance of the thermal control system in a spacecraft. Next an explanation on the origins of the thermal control system design, including a historical analysis based on historical NASA missions. Afterwards, there will be a discussion of how the researcher made the preliminary design. Then there will be a discussion on the methodology of the parametric analysis of the effect of changing the surface area of the lander, conducted using MATLAB. Next there will be a discussion on the results of both the preliminary design and of the parametric model. Finally, there will be a conclusion section that will discuss the final findings and how this method could be improved upon, including areas of further research.
ARISE is a mission concept for the Integrated Product Team (IPT) Mission Design class, which is a senior design class for students in the mechanical and aerospace department and for students in the industrial and systems engineering department. This class is split into two semesters, with each semester class further separated by the specific class taken, which is determined by the student’s selected major. This class focuses on the development of a preliminary mission to a deep space body, such as Europa, with a specific focus on how engineers must work as a team on different systems of a vehicle or mission and must consider how their design choices affect other systems and even the mission as a whole.

Thermal control is very important in a spacecraft. This is because all instruments and most pieces of supporting equipment in a spacecraft have an allowable range of temperatures for each specific piece to both remain operational and to survive. If the temperature or the spacecraft exceeds the operational parameters, the piece of equipment stops working. However, if the temperature continues to progress further from the operational range and past the point of survival, the piece of equipment does not survive and becomes inoperable, even if the temperature returns to the operational range. Based upon the operational and survivability requirements of all pieces of equipment on the spacecraft, the researcher determined the allowable service ceiling and floor for temperature. This was determined to be 5-15 degrees Celsius, or 278-288 degrees Kelvin.
The researcher began their design by researching previous spacecraft missions conducted by NASA, with a focus on the lander missions. The main reason for this is to have a starting point to build from. One of the key take-aways is that for most landers, a radioisotope thermal generator (RTG) is required to heat and or power the spacecraft, especially for long term missions, such as a deep space mission to Europa. The researcher primarily considered three main lander missions. Those missions are Viking 1, Viking 2, and Phoenix. It should be noted that all these missions are missions to the surface of Mars. Another point is that the Viking 1 and 2 landers were essentially identical in design, with minor differences in the payload package, if any. This is important because the design of these missions focuses on the environment of Mars which, while still colder than Earth, is much warmer than the surface of Europa.

The first lander style to be discussed is that of Viking 1 and Viking 2. To control temperature in space, the lander was given a light colored coating. In addition, due to the aerodynamic drag that the lander would encounter during the landing sequence, the lander was enclosed in an aeroshell and covered in a cork-like ablative coating. To help maintain temperature both in space and on Mars, the landers had two Snap 19 RTGs, which output 40 We (Watt equivalent), of power, while dispersing excess heat to the lander (Schmidt et al., 2008). This excess heat was used to supplement the heaters and solar heating to maintain the lander temperature at 293 K.
The second lander style to be discussed is that of the Phoenix lander. This lander was used to study the arctic pole of Mars. Due to this goal, it was exposed to much colder temperatures than the Viking landers. However, this lander used a different approach for controlling the temperature. The primary difference is that this lander used electric heaters as the primary heat source. This required the lander to heavily depend on insulation and other exterior passive methods of temperature control. This led to the lander failing to maintain an operating temperature during its mission, which lead to supporting systems, such as the batteries for power failing. This shut down the lander prematurely.

The conclusion from this historical analysis is that RTGs were to be the heart of the thermal control system, with heaters being used only as an emergency source of heat. This did lead to a problem. The heat output of a SNAP-19 is much too low for the Europa environment. However, RTG technology has improved since the 1960s. One of the modern RTG models is the multi-mission radioisotope thermal generator (MMRTG) (Power Systems, NASA). Unlike the SNAP-19, which could only generate 42 W of electricity per unit, the MMRTG can generate 125 W of electricity and 2000 W of heat per unit. This capability makes it the perfect choice for a deep space mission, such as a mission to Europa. In addition, both landers had coatings that helped it passively control radiative heat transfer. The landers focused on the use of lightly colored coatings to help reflect incoming energy from the sun. However, an active choice by the researcher was to use a darker coating so that more of the incoming heat at Europa can be used. This did lead to a problem during transit.
During the transit phase of the mission, the lander is exposed to not only the solar radiation at Earth’s orbit, but also at Venus’s orbit. This is because it was decided to use a VEGA transfer to get a gravity assist to get to Europa with a smaller delta v requirement. This allows for the vehicle payloads to be larger, however it causes a large strain on the thermal control systems of each vehicle. The solution to this problem is twofold. The first aspect is to use a radiator to disperse more heat into space. The requirement for this is to switch from a passive heat transfer method to a pumped method. The pump system would only be used to help purge heat space during the warm phases of the mission. However, since this introduces an active control method to the thermal system, it also allows for essentially the removal of the radiator in the cold phase, since it can be insulated in a way that only the pipes serve to transfer heat to the radiator. This opens the opportunity for heat recapture during the cold phases at Europa. One of the goals of this study is to determine the overall need of this method of heat control by analyzing the effects of the spacecraft area on the temperature and heat requirements. This analysis is one way to determine if the pumped system is required in this mission, or if it will just add more complexities to the design.
Methodology

The methodology discussed in this study is twofold. First, the researcher will discuss the method of the preliminary analysis used to determine the temperature of the exterior of the lander at different planets, the net heat transfer at different planetary orbital distances, and the amount of heat required to be purged from the lander during the hot phases of the mission. This heat was then used to determine the size of the radiator for the lander. The second methodology to be discussed is how this method was modified to conduct a parametric analysis of the lander in terms of a changing spacecraft area. The analysis of this effect is important, as the final design of the lander will not have the same surface area as was used in the preliminary analysis. However, if the effect of this change is small, and the same design that was determined in the preliminary analysis can still be used in the final design.

Preliminary Analysis

The preliminary analysis of the spacecraft is very important. Not only does it create the basic parameters required for a basic mission architecture, but it also generates some of the initial conditions used for the finite element analysis discussed later. The general approach to this analysis follows the approach discussed in chapter 7 of Elements of Spacecraft Design, Thermal Control (Brown & McMordie, 2002). Some assumptions are used to lay the foundations of this method. The first is that the spacecraft is a spherical shape of the same surface area of the designed spacecraft. Essentially this means that the researcher used the area given to them by the structures leader to determine the radius of the spherical representation of the spacecraft.
The next step is to tabulate the payload and vehicle temperature requirements, in addition to the best and worst case power load. If an RTG is used in the vehicle design, the waste heat wattage should be added to this power load. The next step is to determine the radiative properties of the spacecraft coating. This is typically done by selecting from a table of known values from available coatings.

The first set of calculations to be made is the maximum and minimum temperatures of the spacecraft. These temperature calculations are dependent on the internal wattage \(W\) (power consumption and RTG output), the value of the solar constant at the orbital body \(G\), the diameter of the spherical spacecraft \(D\), the Stefan-Boltzmann constant \(\sigma\), the solar absorptivity of the spacecraft \(\alpha\), and the spacecraft emissivity \(\varepsilon\). The maximum temperature is given by Equation 1, while the minimum temperature is given by Equation 2. It should be noted that for the minimum temperature case, the solar constant term is missing. This is because it is assumed to be zero at the surface of Europa. In addition, these equations are lacking albedo terms. This is because the albedo effects are near zero for both the hot and cold cases because the spacecraft is far enough away from a planetary body for it to have a negligible effect.

\[
T_{\text{max}} = \left[ \frac{G\alpha}{4\pi D^2} + \frac{W}{\sigma \varepsilon D^2} \right]^{\frac{1}{4}}
\]

Equation 1

\[
T_{\text{min}} = \left[ \frac{W}{\sigma \varepsilon \pi D^2} \right]^{\frac{1}{4}}
\]

Equation 2
The next step is to compare the calculated temperature values with the tabulated maximum and minimum temperatures of the spacecraft. If the calculated maximum temperature exceeds the maximum temperature requirements of the spacecraft, then a pumped heat system is likely to be required to cool the spacecraft. The next calculation to be made is the size of the radiator, in terms of area (A). This equation is dependent on the working temperature of the radiator in the hot case, the emissivity of the radiator, and the waste heat to be emitted from the radiator. The working temperature of the radiator is either the spacecraft maximum temperature allowance or the calculated maximum spacecraft temperature, whichever is larger. Equation 3, shown below, expresses this relationship.

\[
A = \frac{W}{\sigma \varepsilon T^4} \tag{3}
\]

Then, the required heater power for the cold case needs to be calculated. This is essentially another solve of Equation 3, however instead of solving for the radiator area, the goal is to determine how much power is required to maintain the spacecraft at the lower temperature boundary. This boundary is either the spacecraft temperature design lower limit or the calculated minimum temperature, whichever is lower. This relationship is given in Equation 4.

\[
W = A \sigma \varepsilon T^4 \tag{4}
\]

Essentially, this wattage is the minimum power consumption required to maintain the temperature of the spacecraft. If this number is less than the minimum power output, then no additional considerations are required. If this number is greater than the minimum power output in this situation, then the difference of the two values is considered the additional power required for the thermal system to consume by using electric heaters to maintain the temperature of the spacecraft.
Parametric Analysis

In this analysis, the spacecraft area was varied. As observed from the previously expressed equations, the effect of this variation can be expected to be in the temperatures during transit and at Europa. In addition, it is expected to affect the size of the radiator and the minimum required power output to maintain operational temperatures in the lander.

Results

Table 1: Preliminary Analysis Inputs

<table>
<thead>
<tr>
<th>Variable</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Spacecraft Area (m²)</td>
<td>17.3599</td>
</tr>
<tr>
<td>W_RTG (W)</td>
<td>2000</td>
</tr>
<tr>
<td>W_Power_Transit (W)</td>
<td>500</td>
</tr>
<tr>
<td>W_Power_Europa (W)</td>
<td>1024</td>
</tr>
<tr>
<td>Venus Solar Constant (W/m²)</td>
<td>2611</td>
</tr>
<tr>
<td>Earth Solar Constant (W/m²)</td>
<td>1366.1</td>
</tr>
<tr>
<td>Mars Solar Constant (W/m²)</td>
<td>588.6</td>
</tr>
<tr>
<td>Jupiter Solar Constant (W/m²)</td>
<td>50.5</td>
</tr>
<tr>
<td>Lander Emissivity</td>
<td>0.8</td>
</tr>
<tr>
<td>Lander Absorptivity</td>
<td>0.163</td>
</tr>
<tr>
<td>Stefan-Boltzmann’s Constant (W/(m²*K⁴))</td>
<td>5.67*10⁻⁸</td>
</tr>
<tr>
<td>T_Max_Lander (K)</td>
<td>288.15</td>
</tr>
<tr>
<td>T_Min_Lander (K)</td>
<td>278.15</td>
</tr>
<tr>
<td>Radiator Emissivity</td>
<td>0.924</td>
</tr>
</tbody>
</table>

Table 2: Preliminary Analysis Findings

<table>
<thead>
<tr>
<th>Variable</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Diameter (m)</td>
<td>2.3507</td>
</tr>
<tr>
<td>Radiator Area (m²)</td>
<td>6.9217</td>
</tr>
<tr>
<td>Maximum Temperature (K)</td>
<td>288.15</td>
</tr>
<tr>
<td>Minimum Temperature (K)</td>
<td>278.15</td>
</tr>
<tr>
<td>Excess Heat (W)</td>
<td>853.39</td>
</tr>
</tbody>
</table>
A Thermal Analysis of ARISE Lander

Figure 1: Spacecraft Temperature

Figure 2: Radiator Area
The results of this study determined the preliminary design values for the thermal system of the lander. Table 1 includes most of the input values that define the lander used in this analysis. These numbers are included so that they can be compared with those determined by another methodology in potential future research. These values included the spacecraft area, as given by the structures team lead of the ARISE mission, among other important values that define the thermal system. Table 2 shows the results of the preliminary analysis. Important values here are the maximum and minimum temperatures. It was determined that the design parameters for the lander dominate the thermal design needs. The diameter value is the diameter of a spherical spacecraft with the same surface area as that of the mission lander. The most important value determined here, however, is the required surface area of the radiator. This is because it is a difficult part to integrate into the vehicle.
The parametric analysis data is shown in Figures 1-3. Figure 1 shows how the maximum and minimum temperature of the vehicle changes based upon the vehicle surface area. It can be seen that the surface temperatures of the vehicle are larger the smaller the vehicle becomes. However, as the vehicle approaches the design size, the design parameters become the temperature values. This continues as the spacecraft becomes larger. Figure 2 shows how the size of the radiator changes with the size of the spacecraft. As the spacecraft gets larger, the radiator size gets larger, however, the size of this increase is comparatively very small. When the design constraints take control, the radiator size levels out to just below 7 square meters. Figure 3 shows the required amount of waste heat from the spacecraft heating system required to keep the temperature at the minimum temperature. It should be noted that the large requirement for the smaller vehicle sizes corresponds to minimum temperatures that are larger than the design minimum, but larger than the design maximum, so this represents heat that needs to be purged from the system. Looking at sizes closer to the design size, the level stabilizes to that of the design. Looking at the temperatures again, it can be concluded that the pumped cooling system is not required and would only add more complexities and ways for the thermal system to fail. However, passive heat pipes should still be used to distribute heat throughout the spacecraft.
What does this data mean? One of the key takeaways here is that if the spacecraft needs to increase in size, then that largely doesn’t affect the thermal system of the lander. Another key point is that the spacecraft will be trying to purge heat for most of the mission, even at Europa. So, vehicle failure by thermal failure, such as what happened in the Phoenix mission, is not to be expected. However, some key assumptions were made that limit the scope of the data, and the conclusions thereafter. The first assumption is that the surface temperature is the temperature for the entire vehicle. This is not the case, as heat transfer will change depending on if that portion of the vehicle is in sunlight or has electronics near it. Also, this does not consider how the fuel tanks for the vehicle will transfer and store heat. Some of the heat that is purged may need to be distributed to the fuel tanks.

In conclusion, the researcher was able to successfully design a thermal control system for the ARISE IPT lander mission using historical NASA missions as reference designs. The methodology used in this analysis used a standard assumption of evenly distributed temperatures on the surface of a spherical representation of the lander. A study was conducted using this methodology of the designed lander, and a parametric analysis was conducted to further understand the effect of changing the spacecraft area on this design. Further research, such as specific finite element modeling of the spacecraft should be conducted, not only to verify the preliminary analysis, but to further define heat transfer throughout the spacecraft. Examples of this could be how heat is transferred to the electronics on board or to the exterior fuel tanks. Another effect to take into consideration in future research is how the lander may affect the icy surface of Europa by the process of purging heat.
References


Appendix

Appendix A: Preliminary Analysis Code

%Honors Capstone-Preliminary Analysis
%Logain North
%housekeeping
clc; clear;close all;
%Inputs
space_area = 17.3599; %spacecraft area m^2
%power values
Q_rtg = 2000;
Q_power_transit = 500;
Q_power_Europa = 1024;
Q_transit = Q_rtg+Q_power_transit; %total power, transit
Q_Europa = Q_rtg+Q_power_Europa; %total power, europa
%solar constants at venus, earth, mars,jupiter
G = [2611 1366.1 588.6 50.5];
lander_emiss = 0.8;
lander_abs = 0.163;
bolt = 5.67*10^-8;
T_max_land = 288.15;
T_min_land = 278.15;
% calculations for min and max temp
D = sqrt(space_area/pi);
G_term = G*lander_abs/4;
power_term = Q_transit/(pi*D^2);
bolt_term = bolt * lander_emiss;
T_surf_tran = ((G_term+power_term)/bolt_term).^.25;
T_max_tran = max(T_surf_tran);
T_min_tran = min(T_surf_tran);
if T_max_tran > T_max_land
    T_max = T_max_tran;
else
    T_max = T_max_land;
end
T_surface = (Q_Europa/(bolt_term*pi*D^2)).^.25;
T_min = T_min_land;
if T_min == T_min_tran
    Q_compare = Q_transit;
else
    Q_compare = Q_Europa;
end
%radiator sizing
rad_emiss = 0.924;
A_rad = Q_transit/(bolt*rad_emiss*T_max^4);
%required/excess power
W = A_rad*bolt*rad_emiss*T_min^4;
Q = Q_compare-W;
if Q_compare <0
    Q_addiional = Q;
else
    Q_excess = Q;
end
Appendix B: Parametric Analysis Code

%Honors Capstone-Spacecraft Area Analysis
%Logain North
%housekeeping
clc; clear;close all;

%Inputs
space_area = 1:.1:25; %spacecraft area m^2
%power values
Q_rtg = 2000;
Q_power_transit = 500;
Q_power_Europa = 1024;
Q_transit = Q_rtg+Q_power_transit; %total power, transit
Q_Europa = Q_rtg+Q_power_Europa; %total power, europa
%solar constants at venus, earth, mars,jupiter
G = 2611;
lander_emiss = 0.8;
lander_abs = 0.163;
bolt = 5.67*10^-8;

T_max_land = 288.15;
T_min_land = 278.15;

% calculations for min and max temp
D = sqrt(space_area/pi);
G_term = G*lander_abs/4;
power_term = Q_transit./(pi*D.^2);
bolt_term = bolt * lander_emiss;
T_surf_tran = ((G_term+power_term)./bolt_term).^0.25;

for n = 1:1:length(T_surf_tran)
    T_max_tran = T_surf_tran(n);
    if T_max_tran > T_max_land
        T_max(n) = T_max_tran;
    else
        T_max(n) = T_max_land;
    end
end

T_surface = (Q_Europa./(bolt_term*pi*D.^2)).^0.25;

for n = 1:1:length(T_surface)
    T_min_tran = T_surface(n);
    if T_min_tran > T_min_land
        T_min(n) = T_min_tran;
    else
        T_min(n) = T_min_land;
    end
end
% radiator sizing

rad_ems = 0.924;
A_rad = Q_transit./(bolt*rad_ems*T_max.^4);

% required/excess power

W = A_rad.*bolt.*rad_ems.*T_min.^4;

% plotting

figure(1)
plot(space_area, T_max, space_area, T_min)
xlabel('Spacecraft Area (m^2)')
ylabel('Temperature (K)')
legend('T(max)', 'T(min)')
title('Spacecraft Area Vs. Temperature')

figure(2)
plot(space_area, A_rad)
xlabel('Spacecraft Area (m^2)')
ylabel('Radiator Area (m^2)')
title('Spacecraft Area Vs. Radiator Area')

figure(3)
plot(space_area, W)
xlabel('Spacecraft Area (m^2)')
ylabel('Heat Required (m^2)')
title('Spacecraft Area Vs. Required Waste Heat')