Politecnico di Milano

`Fifth Assignment`

Space Systems Engineering and Operations

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1 Introduction

Perhaps the single most challenging aspect the MESSENGER Mission, was the spacecraft's Thermal Control Subsystem (TCS) design. The probe must in fact be able both to withstand Mercury's harsh thermal environment as well as conciliate this aspect with its survivability along the travel towards the innermost planet.
More in depth:

- Mercury's orbit brings the planet, and the spacecraft, to within 46 million kilometers from the Sun, about 0.3 to 0.47 AU, corresponding to a solar flux variance from 11.1 to 4.5 times stronger than the one of our planet. The flux is unidirectional and is effectively attenuated by the sunshade.
- A secondary problem, that nevertheless cannot be underestimated, is the presence of the hot, broiling surface of the planet itself, 433°C at perihelion and 298°C at aphelion, with a reflected solar intensity of approximately four times that on Earth. In contrast with the Sun's flux this thermal flux is omnidirectional.

2 Architecture

The MESSENGER Thermal Control Subsystem (TCS) is designed to work in completely passive way, requiring no louvers or other mechanisms, and requires little heater power, minimizing both weight and power demands. The materials used are readily available and proven low risk.
MESSENGER TCS has three main design elements:

- A ceramic-fabric sunshade
- Multilayered insulation
- Heat Radiators

2.1 Sunshade

MESSENGER's peculiar and most important tool of thermal defense is a heat-resistant and highly reflective sunshade, fixed on a titanium frame to the front of the spacecraft. Measuring about 2.5 m tall and 2 m across, the thin and light, around 20 kg, shade has front and back layers made of 3M Nextel 312-AF 10 ceramic cloth surrounding several inner layers of aluminized Kapton plastic insulation. While temperatures on the front of the shade could reach 325-370° C when Mercury is closest to the Sun, behind the shade the spacecraft will operate at a temperature around 20° C. The ceramic cloth is rated for temperatures in excess of 1000°C. Adequate testing was done on the sunshade, that was soaked, not cleaned, at 355° C for 18 days, around 432 hours [2].

2.2 Multilayered insulation

Made with Kapton and other insulating hardware, MLI covers most of the surfaces of the spacecraft to protect it against incident thermal radiation and insulate it against internal heat loss. This second level of thermal protection assumes a key role against heat radiation coming from Mercury's surface.

2.3 Radiators

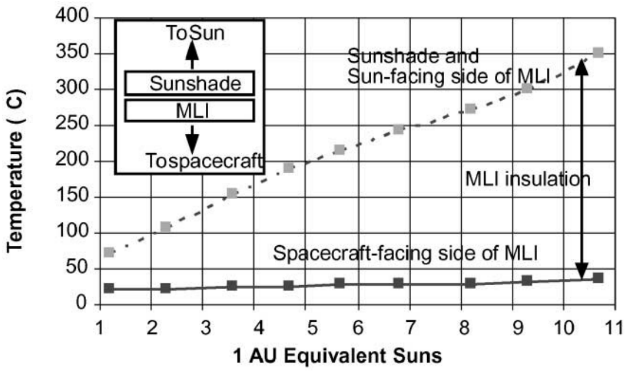
OSRs covered, connected to diode "one-way" heat pipes, radiators are installed on the sides of the spacecraft to carry heat away from the spacecraft body, channeling the heat on the side of the spacecraft that is not in view of the planet. Radiators are isolated from the spacecraft structure and electronics, being coupled only through radiation. A total of 1.2 m^2 of radiator area is needed to dissipate internal heat. They are located on the +X and -X faces of the spacecraft since the -Y face contains the sunshield and +Z and -Z faces contains the LVA and the launch vehicle adapter [3] [6].
The choice of diode heat pipes is dictated by the fact that when a radiator is facing Mercury's hot side, heat flux would be inverted and this would lead to an increase in temperature of the internal electronics connected to the heat pipes. The unsymmetrical heat transfer coefficient of the diode pipe prevents this from happening, effectively detaching the radiator from the rest of the spacecraft's thermal subsystem. Nominal heat flux is restored once the radiator is back in full view of deep space and has cooled down sufficiently.

2.4 Heaters

Other important hardware components in the thermal subsystem design are the 33 redundant heaters, divided in primary and secondary heaters. The set-points of the primary and secondary heaters are offset so that the secondary heaters are never energized unless a primary heater fails. Battery temperature relies on two redundant heater that are controlled by mechanical thermostats.

2.5 Structure

The structure of MESSENGER was design to endure to the extreme heat, featuring a graphite composite construction. The spacecraft is also thermal insulated during burns, the large bipropellant thruster is surrounded by a gold-plated heat shield [1].



(a) Temperatures in the various phases, according to distance from the Sun

(b) A view of the sunshade prior to launch

The only elements of the spacecraft, not effectively protected by the sunshade are the solar panels. For this reason they are placed on low-conductive struts, 0.9 m from the side radiator panels to reduce heat coupling to the spacecraft. The back faces are covered by Kapton with a vapor deposited aluminum outer layer, reducing the expected temperature by 150°C over that of a fully-packed array.
All this solutions, combined together with the addition of a smart mission design, tailored towards heat management strategy, largely removed the thermal burden from the other subsystems, allowing the spacecraft to operate without the use of special high-temperature electronics, as well as traditional components and design[5].

3 Phases

This particular mission is strongly constrained under the thermal viewpoint, comprehending extreme as well as disparate environments. Due to these aspects, mission design plays also a key role in the thermal management strategy, both during the cruise phase and the orbiting around Mercury, bringing to a precise formulation of the various phases[9] [12].

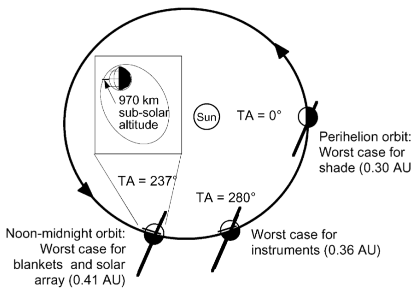
3.1 Cruise phase

During the outer cruise phase of the mission, the focus is switched towards the necessity to prevent an over cooling of the spacecraft. At the beginning of the mission, after launch with the sunshade pointing towards the Sun, the spacecraft will use heater power to make up for the correct thermal condition that cannot be achieved otherwise.
In order to reduce heater power consumption and increase solar array power margin, MESSENGER is designed to be flipped such that the side opposite to the sun shield can be illuminated during outer cruise. This capability has allowed MESSENGER to operate easily between 0.95 and 1.08 AU with little heater power required, allowing unconstrained outer solar distance flexibility when the mission design team was planning for back-up trajectories without complicating the spacecraft thermal or power designs. The spacecraft remained directly exposed to Sun light until 0.95 AU, in the so called outer cruise. To counteract the cooling, a series of "flips" and "flops" were planned. The "flip" maneuvers positioned the sunshade towards the Sun. The "flop" maneuvers turned the spacecraft and pointed the sunshade away from the Sun (see fig:2c).

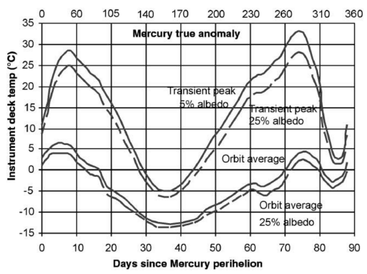
3.2 Orbital phase

The science orbit has been designed to limit the spacecraft exposure to the heat re-radiating from the surface of Mercury. MESSENGER will only spend approximately 25 minutes of each 12-hour orbit crossing at low altitude, around 200 km, from Mercury's surface, in order to prevent overheating. The maximum surface temperature of Mercury reaches $433°C$ at the subsolar point and drops off as a cosine function from the maximum to $-173°C$. The IR heat flux is omnidirectional and cannot be effectively attenuated by the sunshield, thus, in order to minimize this problem, a choice was done in the spacecraft's orbit design: it was designed to be highly elliptical with a periapsis latitude at 60°N. In this way it was possible, in most of the cases, both to place the periapsis of the orbit far from the subsolar point, both to pass the subsolar crossing point with a high velocity. Furthemore, when the orbit periapsis is the nearest to it, Mercury is at 0.41 AU

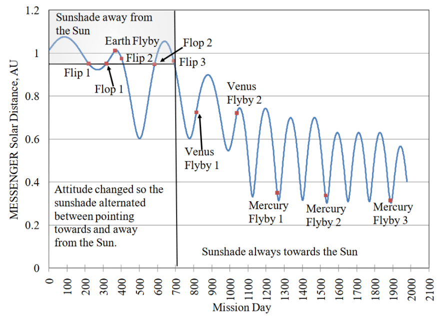
from the Sun or 273° true anomaly: far enough from its perihelion. The spacecraft attitude is controlled to keep the sunshade between the body of the spacecraft and the Sun. This restricts the spacecraft pitch to ±12° and yaw to -17° through +13.5°.
The worst thermal case for the instruments occurs when the spacecraft periapsis is near the subsolar point, day 75, at a distance 0.36 AU from the Sun.
The worst transient, so the broadest variation in temperature experienced by the instruments, is 33°C and lasts for only 25 minutes, with a maximum temperature gradient of 1.5°C per minute. Fortunately this occurs only during a small portion of the orbit. In fact in the remaining part of the 12-hour orbit, the temperature is close to the orbit-average value. [12] Thus the payload is kept within its operating temperature range over 99.5% of the orbit phase, with survival margins in excess of 20°C. Also, spacecraft electronics are always within their -29°C to +60°C operating range, with less than a 5°C per minute gradient. The battery always remains within its -10 to +25°C operating range. For the majority of the mission, the battery temperature is maintained by heaters to be between -5 and 0°C. [12] During the eclipse, the battery temperature does peak to 19°C, but lifetime concerns are mitigated because only 200 discharge cycles are expected.



(a) Where the described conditions occurs along the orbit



(b) Average and peak instrument deck temperatures



(c) Flip Flop manoeuvres

4 Reverse Sizing

Reverse sizing of the thermal control subsystem will be performed by analyzing and justifying through numerical models the choices adopted by the engineering team of the MESSENGER mission.

4.1 Thermal environment

In the cruise phase, the main source of heat is represented by the radiation coming form the Sun. In the outer cruise ($d_{Sun} > 0.95\ AU$), the solar flux has been modeled as a function of the distance between the spacecraft, or the planet Mercury as in the following case, and the Sun

$$q_{Solar}(r_{SC}) = q_0 \left(\frac{r_{Earth}}{r_{SC}}\right)^2 \quad \left[\frac{W}{m^2}\right] \tag{1}$$

where q_0 is the solar flux ad 1 AU, .
In the inner cruise instead, the solar flux has been decreased by a factor to account the effect of the sunshield blocking a large part of the solar radiation coming form the Sun. The value of the sunshield thermal efficiency η_{SS}, intended as the ratio between the solar radiation that would hit the spacecraft with the sunshield and without it, has been estimated by retrieving the values of absorptivity and emissivity of the outer layer both of the shield and of the spacecraft.
In the Orbital phases, different sources of radiation are present and have been modelled as follows.
The albedo heat flux has been modelled as:

$$q_{Albedo} = q_{Solar} \cdot a_M \cdot \cos\theta \cdot \left(\frac{r_M}{r_{SC}}\right)^2 \quad \left[\frac{W}{m^2}\right] \tag{2}$$

where a_M is the planetary albedo, known to be 0.08 [14] [13] , and θ is the irradiance angle between the spacecraft and the planet.
The IR emission of Mercury's surface is modelled as

$$q_{IR} = \sigma \cdot \epsilon_M \cdot T_{eq}^4 \cdot \left(\frac{r_M}{r_{SC}}\right)^2 \quad \left[\frac{W}{m^2}\right] \tag{3}$$

Mercury shows a huge difference in temperature between the subsolar point $\approx 700\ K$ and its dark side $\approx 100\ K$ with high temperature gradient in the illuminated face. Moreover, the orbit periapsis is at 200 km

altitude at 60° with respect to the subsolar point [6]. As a consequence, in order to obtain meaningful results in a first approximation, it is necessary to consider Mercury infrared radiation as a black body spectrum ($\epsilon_M = 1$) with *equivalent* temperature $T_{eq} = 434\ K$ [7].

4.2 Spacecraft modeling

In first approximation the spacecraft has been modeled as a single spherical node with total area equal to the main body area $A_{SC} = 2(l_1 \cdot l_2 + l_1 \cdot l_3 + l_2 \cdot l_3)$ and cross section equal to $A_{cross} = \frac{A_{SC}}{4}$. The total thermal power entering and exiting the spacecraft has been computed as:

$$\begin{aligned} Q_{IN} &= Q_{Solar} + Q_{IR}^{IN} + Q_{ALB} + Q_{GEN} && \text{Input power} \\ Q_{OUT} &= Q_{IR}^{OUT} + Q_{RAD} && \text{Exit power} \end{aligned} \tag{4}$$

In particular regarding the thermal power entering the spacecraft:

$$Q_{Solar} = \begin{cases} \dfrac{\sigma\left(T_{SS,in}^4 - T_{SC}^4\right)}{\dfrac{1-\epsilon_{SS}}{\epsilon_{SS}A_{SS}} + \dfrac{1-\epsilon_{SC}}{\epsilon_{SC}A_{Cross}} + \dfrac{1}{A_{Cross}}} & \text{If the sunshield points towards the Sun} \\ q_{Solar} \cdot \alpha_{SC} A_{Cross} & \text{If the sunshield points opposite the Sun} \end{cases} \tag{5}$$

$$Q_{IR}^{IN} = q_{IR} \cdot A_{SC} \cdot \epsilon_{SC} \cdot F_{M \hookrightarrow SC} \tag{6}$$

$$Q_{ALB} = q_{Albedo} \cdot A_{SC} \cdot \alpha_{SC} \cdot K_a \cdot F_{M \hookrightarrow SC} \tag{7}$$

Q_{GEN} represents the internal power generated by the spacecraft itself. $T_{SS,in}$ and ϵ_{SS} are taken as the temperature and the emissivity of the sunshield side facing the spacecraft during orbital operations. The power exiting the spacecraft can instead being divided in two contributions: the power irradiated by the spacecraft towards the free space:

$$Q_{IR}^{OUT} = \sigma \cdot \epsilon_{SC} \cdot (A_{SC} - A_{RAD}) \cdot \left(T_{SC}^4 - T_\infty^4\right) \tag{8}$$

and the power irradiated specifically by the radiators:

$$Q_{RAD} = \sigma \cdot \epsilon_{RAD} \cdot A_{RAD} \cdot \left(T_{SC}^4 - T_\infty^4\right) \tag{9}$$

According with the literature [11][10][7], the previous parameters have been estimated as follows:

Parameter	Value	Parameter	Value	Parameter	Value	Parameter	Value
ϵ_{SC}	0.05	$A_{SC}\ [m^2]$	13.56	α_{SC}	0.12	$l_1\ [m]$	1.27
ϵ_{SS}	0.87	$A_{SS}\ [m^2]$	5.00	Ka	0.8	$l_2\ [m]$	1.85
ϵ_{RAD}	0.88	$A_{RAD}\ [m^2]$	1.20	$T_\infty\ [K]$	3	$l_3\ [m]$	1.42

Table 1: Constants values used

4.3 Allowable temperature range

According to literature regarding the mission, all the instruments onboard the spacecraft were designed to operate near room temperature. Taking the highest minimum and the lowest maximum temperature allowed by components, it is possible to determine a safe range of operating temperature for the spacecraft of $0°C < T_{SC} < 30°C$. In this temperature range all of the components are within their temperature limits, even though most of them can also operate well outside this envelope [8]. This will over-constrain the analysis, but will result in a preliminary sizing for the system. A more refined multi-nodal analysis is necessary for this kind of thermal sizing. Normally, a conservative margin of $\mp 15°C$ should be taken over the typical temperature ranges admissible by the various components, but in this case it was possible to retrieve the $0°C$ - $30°C$ was retrieved from literature [2] as the range used for validation of the thermal model.

4.4 Critical cases

Three critical cases have been taken into account to verify properly the TCS subsystem sizing.

4.4.1 Orbital hot case

The first critical case considers the spacecraft in the hottest point of its both interplanetary and planetary trajectory that is the periapsis of the orbit around Mercury.
In this condition the orbital altitude is 200 km and the corresponding view factor between the planet and the spacecraft $F_{M \hookrightarrow SC}$ is equal to 0.309, while θ is taken 60° in the worst case scenario based on trajectory choice, in the described case the the distance of the spacecraft from the Sun is 0.36 AU, as already described in the Orbital Phase (paragraph 3.2) [12] . In this case, the total thermal power exchanged by the spacecraft, divided in its components is:

Inward				Outward
Q_{Solar} $[W]$	Q_{IR}^{IN} $[W]$	Q_{ALB} $[W]$	Q_{GEN}^{max} $[W]$	Q_{IR}^{OUT}
69.08	360.11	199.40	450.00	324.68

Table 2: Heat fluxes without considering the radiators

The required radiator area to keep the spacecraft at $30°C$ is 2.06 m^2 while the spacecraft is known to use only 1.20 m^2 of radiative surfaces that, according with this model, will drive the object to a temperature of $58°C$. This can be accepted since the model implemented is very simple and the maximum temperature aforementioned is referred only to actual components that are insulated from the structure resulting in overestimating the radiative power required to cool down correctly the spacecraft [5].
The use of diode heat pipes and thermal bridging between heat-producing components and the radiators makes it possible to reduce the need for radiator area to a more manageable size, while allowing for the use of relatively common thermal blanketing materials, such as a multi layer insulation covering the body of the spacecraft, composed of several thin layers of aluminized kapton, which is often used in situations where a cold insulator is required.
It is also possible to estimate the thermal efficiency of the sunshade η_{SS}. According to the previous model, the solar radiation would be 5.9 kW, resulting in efficiency $\eta_{SS} = 0.97$.

4.4.2 Orbital cold case

The second critical case considers the spacecraft in the coldest position of its orbit around Mercury that is the apoapsis in eclipse condition.
The apoapsis is at 15200 km altitude and it would be over the dark side of the planet. In this condition the entering heat flux will involve only the internally generated heat and Mercury's irradiated power while the exit flux will involve the spontaneous radiation towards the free space but also the component due to the presence of the radiators. Accounting also for the presence of the Sun shade, the expression for the entering heat flux results in

$$Q_{SunCold1} = (1 - \eta_{SS})\, q_{Solar} \alpha_{SC} A_{cross} \tag{10}$$

In this situation the heat fluxes involving the spacecraft, which is considered in the lowest accepted temperature, are the following:

Inward			Outward	
Q_{IR}^{IN} $[W]$	Q_{GEN}^{min} $[W]$	$Q_{Heaters}$ $[W]$	Q_{IR}^{OUT} $[W]$	Q_{RAD} $[W]$
0.31	220.00	308.09	214.01	314.39

Table 3: Heat fluxes in the orbital cold case

$Q_{Heaters}$ was computed as the difference between the outward and inward heat fluxes, as computed through the previously shown formulas, where the single-node model of the spacecraft is fixed at a temperature equal to $0°C$, the lowest admissible one. This results in the minimum power needed to keep the node at the prescribed temperature in steady-state conditions.
The resulting $Q_{Heaters}$ needed in much higher than the one actually required by the mission profile. This is mainly due to the fact that we are considering a steady state condition at the coldest point possible for the mission, while in reality the eclipse phase lasts in the worst case scenario, 35 min, during which steady state condition is not actually achieved[5].
The thermal design of the MESSENGER spacecraft is such that temperature transients take hours to complete, and as such heating power can be limited and directed only to the specific components that need it the most. These include the batteries, fuel tanks, valves, pipings, critical electronic components and processors. The low coefficients of heat transfer of both the MLI and composite structure made it possible to decrease significantly the power requirement during these relatively brief phases.

4.4.3 Interplanetary cold case

The third critical case considers instead the coldest condition encountered during the outer cruise, which happens at a solar distance of 0.95 AU, when the spacecraft is flipped into the configuration where the sunshade is pointed sunwards. In this condition solar heat flux is only slightly higher than the minimum for the mission, which happens ad a distance of 1.08 AU, but the added effect of the sunshield makes the actual heat flux hitting the spacecraft much lower.

Inward			Outward	
Q_{Solar}^{IN} $[W]$	Q_{GEN}^{min} $[W]$	$Q_{Heaters}$ $[W]$	Q_{IR}^{OUT} $[W]$	Q_{RAD} $[W]$
18.48	220.00	289.92	214.01	314.39

Table 4: Heat fluxes in the interplanetary cold case

As can be seen, the required $Q_{Heaters}$ to keep the spacecraft at the lowest acceptable temperature is lower than in the previous case resulting in a non-sizing condition. This is close to the required increase in power output that happened right after a *Flip* maneuver, which was in the order of 250 W [5].

4.5 Mass estimation

The bulk of the mass of the thermal control subsystem comes from the radiator panel, which have an honeycomb aluminum structure with an average density around $\rho_{Rad} = 12\ kg/m^2$ for space applications.
Heaters are typically made up of a thin resistive element protected between two layers of a protective plastic material and do not carry significant weight. MLI mass strongly depends on the number of layers and on the material used, as well as on the design of the supporting structure. A preliminary estimate based on values retrieved from other application returns an average density of $\rho_{MLI} = 1.2\ kg/m^2$. An extra margin in considered to account for the mass of the structure holding the sunshield and the body MLI and connecting it to the main structure .

Component	Radiators	Body MLI	Shield MLI	Margin 30%	Total
Mass [Kg]	14.40	14.83	6.0	10.57	45.80

Table 5: Mass estimations

The total computed mass is lower than the one retrieved from literature of 52.2 kg [8], which is likely due to the fact that this model does not take into account the weight of heat pipes, heaters, and thermostats due to lack of sufficient documentation regarding size and weight, as well as underestimating slightly the mass of the structure holding the sunshield, which also has to support some antennas, thrusters and sensors during the liftoff phase.
This represents however a good initial approximation of the mass of the subsystem.

4.6 Material choice

Reversing the previous equations in order to retrieve the values of α and ϵ of the materials used in the thermal insulation of the spacecraft in not trivial, as with the current simplified model no proven material can satisfy the requests of the mission.
The material with the closest characteristics, however, is aluminized kapton [10], which has a long history in the space industry as a cold coating. This is due to the fact that the hot case of the mission is the most critical one, while heater power can easily be provided during cold cases thanks to the availability of solar power and battery power during the course of the mission.

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