Mars Sample Thermal Control During Mars Ascent and Orbit

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Although NASA has no official plans at this time for a mission to return samples from Mars, the Program Formulation Office of the Mars Exploration Program sponsors ongoing mission concept studies, systems analyses, and technology investments which explore different strategies for the potential return of samples from Mars, consistent with the charter of the program and stated priorities of the science community. Maintaining the thermal integrity of collected samples would be very important. In general, samples would be collected, sealed inside tubes, and left on the surface for later retrieval. They would then be inserted into an OS (Orbiting Sample), and carried to a Mars or Solar orbit via a MAV (Mars Ascent Vehicle). Subsequently, an Earth return vehicle would rendezvous with the OS and bring it back to Earth. During ascent from Mars, the OS could serve as the nose cone of the MAV and would be subjected to significant aerodynamic heating from the Martian atmosphere. Once the OS is released from the MAV, its external surface would be exposed to potentially several years of sunlight, eclipse, planetary IR, albedo, and space. The challenge is to ensure that these samples are kept at thermally moderate conditions to preserve their integrity in these widely different environments. Various thermal techniques have been investigated to achieve sample thermal control: use of thermal protection shields and surfaces (ablative and non-ablative) to protect them from adverse exposure to ascent heating, as well combinations of thermo-optical coatings during the orbital phase. The work described herein is part of this ongoing effort & will describe the key challenges related to the thermal control of the potential Mars samples during these phases and the corresponding schemes to overcome them.

Nomenclature

A.U.	=	Astronomical Unit
FRCI	=	Fibrous Refractory Composite Insulation
HRSI	=	High Temperature Reusable Surface Insulation
IR	=	Infra-Red
JPL	=	Jet Propulsion Laboratory
MAV	=	Mars Ascent Vehicle
OS	=	Orbiting Sample
PCM	=	Phase Change Material
PICA	=	Phenolic Impregnated Carbon Ablator
SLA	=	Super Lightweight Ablator
SIRCA	=	Silicone Impregnated Reusable Ceramic Ablator
TPS	=	Thermal Protection System
WCC	=	Worst Case Cold
WCH	=	Worst Case Hot

I. Introduction

There are many variations on the Mars Sample Return mission concept, however one possible architecture is summarized in Figure 1. The general idea is that samples would be collected, sealed inside tubes, and left on the surface for later retrieval. They would then be inserted into an OS (Orbiting Sample), and carried to a Mars or Solar

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orbit via a MAV (Mars Ascent Vehicle). Subsequently, an Earth return vehicle would rendezvous with the OS and bring it back to Earth.

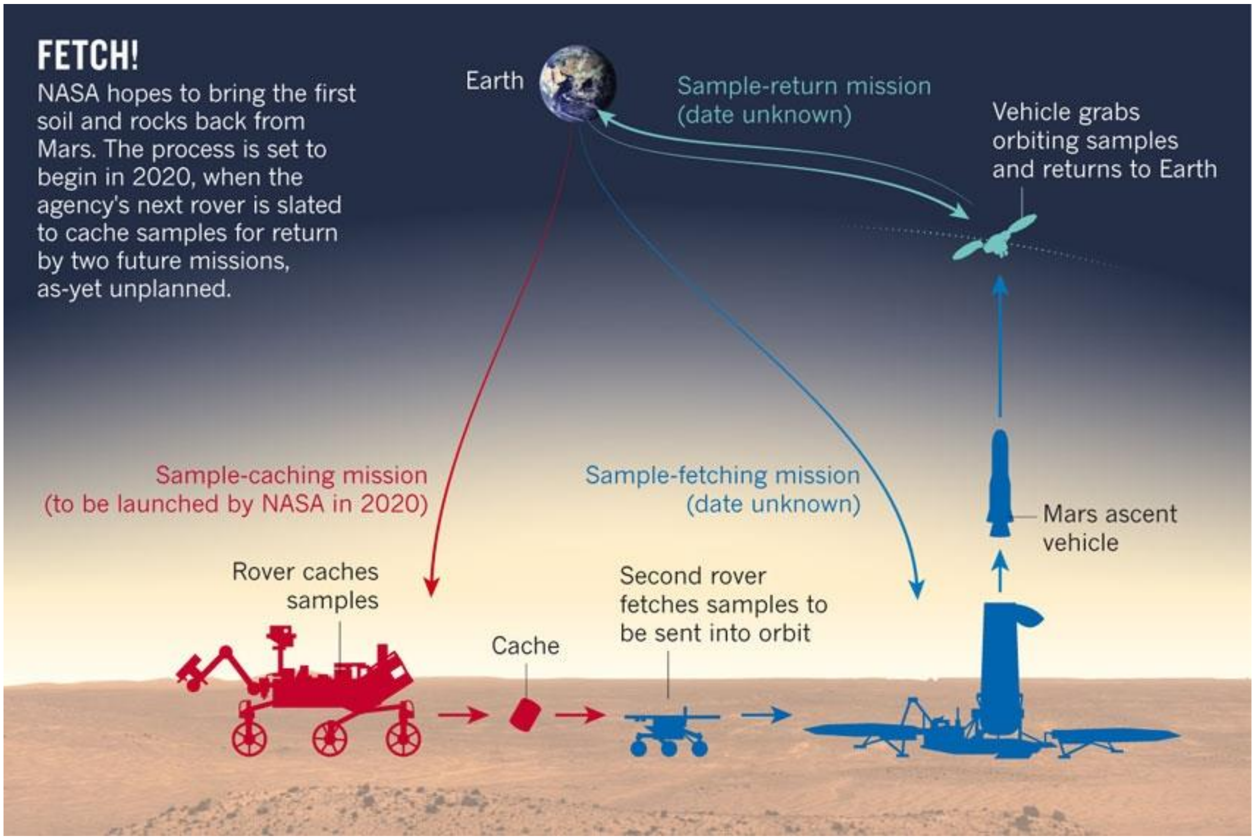

Figure 1: Mars Sample Return Mission Concept.
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Martian samples collected for potential return to earth would have substantial scientific value. Because of this, great care would need to be taken to ensure the samples maintain their scientific integrity. For example, contamination control and co-mingling of the samples is a significant concern, as is their temperature stability and the potential loss of volatiles or water vapor which could occur in the vacuum environment on the return trip to Earth. As a result, thermal control and hermetic sealing of the collected samples is required to maintain the scientific integrity of the samples [2]. Once a sample is collected, it could remain on the surface of Mars for up to 10 years before it might begin its trip back to Earth. Some of the methods which might be used to return a Mars sample to Earth involve placing the sample in Martian orbit, Solar orbit, and/or in transit back to Earth for up to an additional 10 years. During this time, which could total as much as 20 years, the sample would be expected to go through a large number of diurnal and orbital thermal cycles.

The maximum allowable temperature of the sample is still under negotiation within the scientific and engineering community. However, the maximum sample temperature is tentatively set at 60 °C. Maintaining sample temperatures much below this is desirable, but may not be possible. Complying with this limit during the full lifecycle, from extraction on the Martian surface all the way to retrieval on Earth, presents significant engineering challenges. The minimum temperature limit of the sample would likely be similar to what the sample would have already experienced on Mars, or about -125 °C. Therefore, for the collected samples, the primary driver of the thermal design during all the phases is essentially the maximum limit, rather than the minimum.

The orbiting sample (OS), which is approximately a sphere of ~ 20 cm diameter and ~ 5-10 kg mass, and the sample containment system could have different temperature limits than the sample. As an example, if the OS contained a radio beacon for tracking instead of just optical tracking during its orbit, it could have a minimum temperature limit of as high as -40 to -60 °C to ensure the integrity of the radio's electronics & battery.

During ascent from Mars, the primary driving environment is the aero-thermal heating at the nose cone. This is described in more detail in Section II. This phase is short lived and relies on insulating materials and/or the thermal mass of the OS to ride out the severe transient aero-thermal heating. During the orbiting phase, the OS would be primarily exposed to thermal loading from Mars surface (IR & reflected solar), direct solar loading from the sun, and deep space. This phase of the mission is described in more detail in Section III.

II. Thermal Challenges of Maintaining OS Within Its Temperature Limits During Ascent

During ascent from Mars on its way to reaching its orbit around Mars or the Sun, the OS would be subject to aero-thermal heating from the Martian atmosphere. An artistic rendition of this concept is shown in Figure 2. Even though the CO_2 atmosphere is very thin (~ 1 kPa at the Martian surface), the high speeds encountered during ascent would cause significant aero-thermal heating. For example, with a 10 cm radius nose cone, peak heat fluxes of 15 to 40 W/cm^2 (depending on the MAV's propulsion system and trajectory) could be encountered (Figures 3 & 4), with an integrated energy flux of 0.25 to 1.5 kJ/cm^2 over a period of 100 to 200 seconds during ascent. The two extremes of this heat flux are based on two different assumed MAV architectures' (solid or hybrid propellant) trajectories during ascent. This translates to a total input energy in excess of 1000 kJ for a 10 cm radius nose cone, which is substantial for a relatively small object (~ 5-10 kg) to safely sustain without compromising the integrity of the sample. At the start of the ascent, a WCH temperature was estimated to be ~ 15 C, assuming a white painted spherical OS at the WCH 27S Martian environment during summer, with ~ 40% dust accumulation. This temperature was used to scope the architecture of the thermal designs to ensure the thermal integrity of the sample. Note that the exact value of this temperature is not very important to arrive at potential architectures.

Figure 2: Artistic rendition of Mars Sample Return Concept [3]

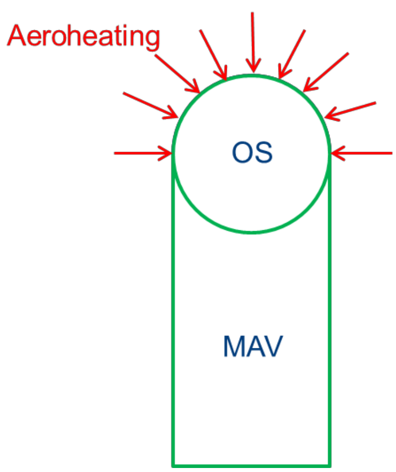

Figure 3: Aero-thermal heating of OS (without fairing) during ascent from Mars

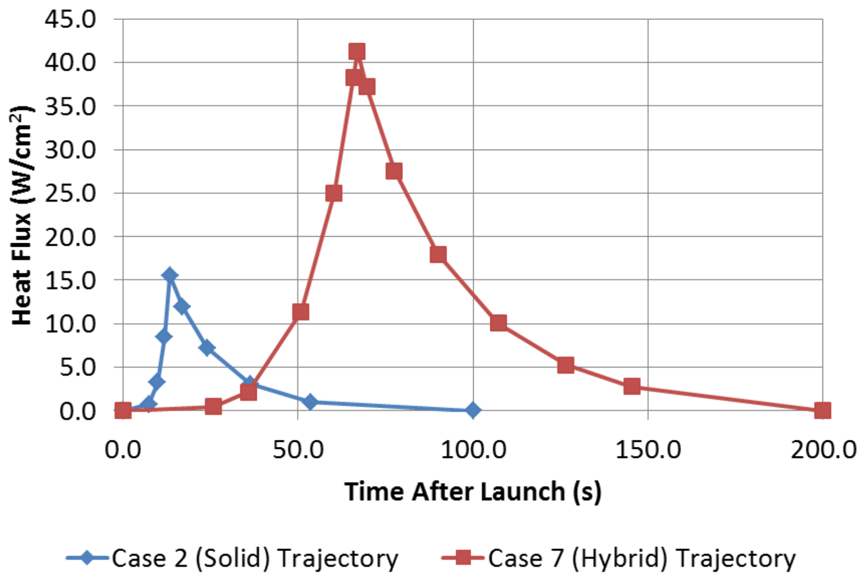

Figure 4: Aero-thermal heating profile during ascent from Mars for a 10 cm radius nose cone

Recognition of this large heat flux then leads to the investigation of possible thermal architectures that would entail for the thermal control of the OS. The primary process of this investigation is to start with the simplest approach and incrementally increase the sophistication until the basic requirements can be meet with a couple of reasonable options. The simplest and most obvious option is to not alter its thermal design and simply rely on the thermal mass of the OS to absorb the incident energy without overheating it. If that is unsatisfactory, the next option is to utilize small improvements by augmenting the thermal mass or increasing the radiation area to reduce the temperature rise. For thermal mass augmentation, simple mass increases or phase change materials (PCM) based mass increases were investigated. The third and more robust option is to use a thermal protection system (TPS) directly on the OS to insulate the OS from the high heat fluxes and protect the OS inside the TPS. The TPS could be ablative or could be non-ablative High Temperature Reusable Surface Insulation (HRSI). Finally, the most robust (and complex) option is to have a fairing surrounding the OS, with a CO_2 gas gap between the OS and fairing. The fairing in the last option may or may not have TPS on its exterior surface. The nature and performance these schemes are discussed below.

- *Thermal mass of OS:*

The thermal mass of the OS (MC_p) was estimated to be so small that its temperature would rise by ~150 to 400 °C due to the aero-thermal heating, hence this was completely unviable to thermally protect the OS. Even if the mass was doubled or tripled, the corresponding temperature rise would be prohibitively large to be deemed as a viable option, hence this was deemed to be an unattractive option. Additionally, the associated mass penalty, even if it helped in abating the temperature rise, would be prohibitive and unacceptable.

- *PCM to augment OS thermal mass:*

Phase Change Materials (PCM) are another way to augment thermal mass with smaller mass increases than using sensible heat storage alone. Simple back of the envelope calculations were made to estimate the mass of PCM required to absorb the heat and not exceed the maximum temperature limit of the OS and it showed a mass increase of 1 to 1.5 kg. This is a very large value (as much as 20% of the OS total mass) and was deemed to be an unattractive option. Additionally, typical PCMs have very low thermal conductivity, which will lead to a very large temperature gradient within the PCM thickness (or very high temperatures at its exterior surface).

- *Addition of Radiation Area to increase the heat loss:*

The addition of heat rejection area to the side of the OS (without simultaneously increasing the area exposed to aero-thermal heating) was investigated to understand its effectiveness. Since the radiation heat loss at the desired maximum temperature of the OS (<60 °C) is small (~0.1% of the input heat flux) due to its relatively low temperature, it was also deemed to be an unattractive option.

- *OS covered with TPS:*

Using a Thermal Protection System (TPS) directly on the outside surface of the OS, in the form of very low thermal conductivity insulation, was investigated as an option to reduce the heat input to the OS. This is shown schematically in Figure 5. Two methods currently under consideration to locate the OS in orbit around Mars are optical tracking and/or using a radio beacon. As a result, it is desirable for the TPS to have a high solar reflectance to aid in optical tracking. HRSI in the form of FRCI-12 (Fibrous Refractory Composite Insulation) shuttle tiles are an excellent option because they have very low thermal conductivity and density. One drawback is that typically these tiles are deliberately made to be black in order to improve their emissivity. They could also be left bare (white or gray), but there is no guarantee that they would not turn black by charring at high temperatures. Also, white or gray tiles would have a slightly lower emissivity than black tiles, which would make their maximum heat flux capability smaller.

However, a hybrid option of ½ black and ½ white/gray HRSI tile could be attractive because if the face of the exposed OS that is subjected to aero-thermal heating is black whereas the unexposed is white/gray, then once the OS is jettisoned from the MAV and is in its orbit, it would tumble and would have both white and black faces exposed to the sun making it optically trackable. Even though the entire surface of the OS would not be reflective, the tumbling OS may be adequately reflective for tracking.

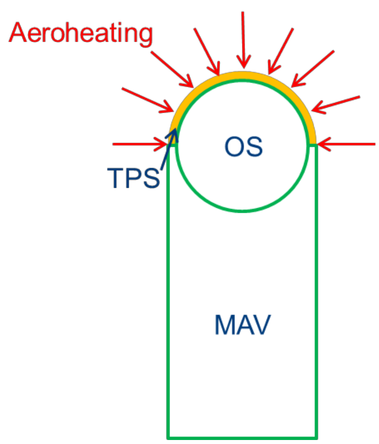

Figure 5: TPS directly on OS (no fairing)

Instead of using low thermal conductivity TPS, like FRCI-12 tiles directly on the OS, the following ablative materials were also investigated as alternatives: SIRCA, SLA-561 & PICA. All of these have heritage in flight use. Their main advantage is their large heat flux capability when compared to FRCI tiles. SIRCA & SLA-561 have ~ 3x higher maximum heat flux than FRCI tiles, whereas PICA has almost two orders of magnitude larger maximum heat flux. FRCI tile heat flux capability could be marginal in terms of accommodating the expected heat fluxes during ascent, whereas the others would have sufficient headroom against it. FRCI Tiles, SIRCA, and SLA-561 would require comparable masses to protect the OS (~ 0.3 kg with ~ 1 cm thickness), but PICA has a much higher thermal conductivity and would thus need to be much thicker and heavier. SIRCA, SLA-561, and PICA could char due to ablation, but a hybrid approach of a ½ covered with TPS and ½ white painted OS may be acceptable and this may not be a discriminator. The temperature dependent thermal conductivity of these options are shown in Figure 6, and Table 1 shows a summary of the key features, assumptions and relative pros and cons of the various concepts investigated. In conclusion, initial assessment indicates the SIRCA or SLA-561 would be the most attractive options, whereas FRCI tiles may be marginal in the heat flux capability and PICA would be much heavier. Further work will be needed in the detailed design phase to confirm this preliminary result.

	FRCI-12 Shuttle Tile	SIRCA	SLA-561	PICA
Estimated TPS Mass	0.1 – 0.3 kg	0.2 – 0.3 Kg	0.3 Kg	0.3 – 0.6 Kg
Maximum Heat Flux	~ 43 W/cm^2	~ 170 W/cm^2 [4]	~ 200 W/cm^2 [5]	2000 W/cm^2 [6]

Table 1: Comparison of TPS options

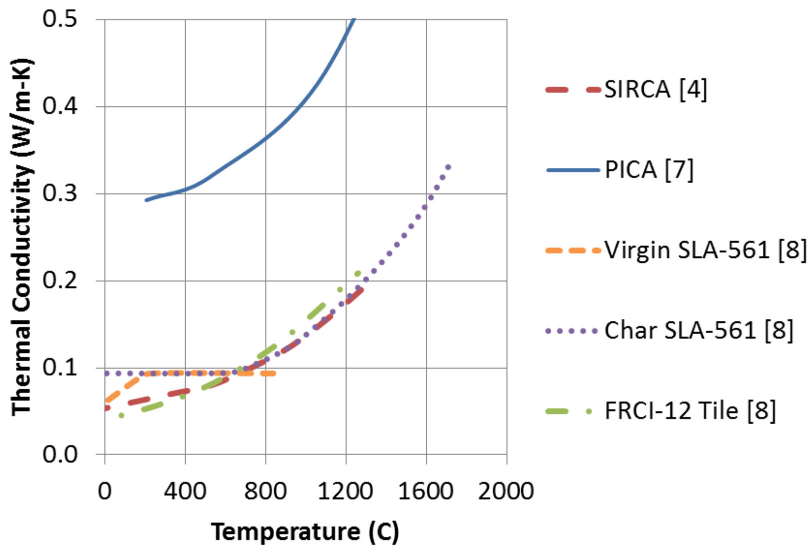

Figure 6: Temperature Dependent Thermal Conductivity of TPS

- *Fairing or Shroud covering OS:*

A more robust option is the use of a fairing or shroud that covers the OS, thus sacrificially protecting it (Figure 7). A fairing would be a clamshell which is released from the MAV upon completion of ascent aero-heating. A shroud would simply be a door-based opening in the MAV nose cone that would allow the OS to be ejected from it. These two concepts are thermally similar even though they are geometrically, operationally and configurationally different because they both have a gas gap separating the exterior surfaces (fairing or shroud) and the OS. Rough dimensions of these concepts would be 20-30 cm diameters, ~3 mm of metallic or composite structural backing, with or without TPS. The gap between the fairing/shroud and the OS would be automatically filled with CO_2 gas at ~1 kPa while on the Martian surface. The interior of the fairing/shroud would also have a low emissivity surface to provide further thermal isolation from the shroud/fairing.

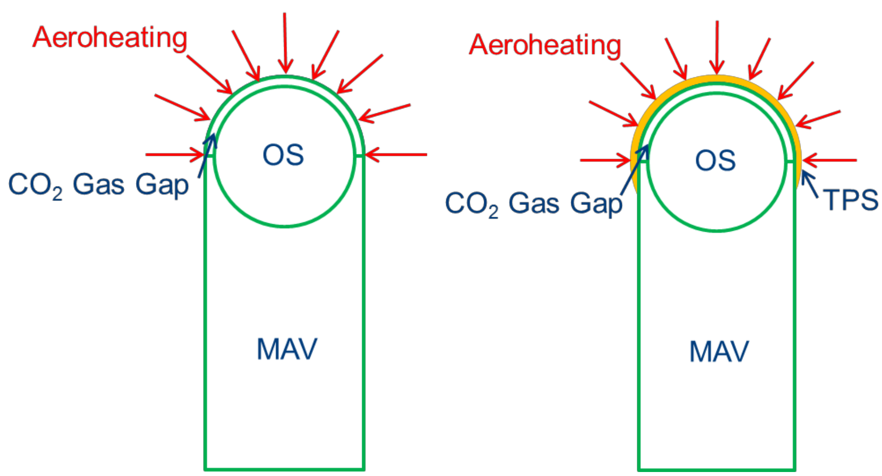

Figure 7: Fairing and OS with and without TPS

A metallic shroud without a TPS would lead to very high temperatures (on the order of 500 °C), which may be acceptable from the standpoint of protecting the OS because of the thermal isolation provided by the CO_2 gas gap between them. However, it has some significant concerns and disadvantages. First, a relatively large temperature rise (30-40 °C) of the OS may not leave much margin in the maximum temperature limit of the OS. This could be exacerbated by the fact that the high temperatures of the fairing/shroud may significantly increase its IR emissivity, which would increase the radiative heat load on the OS (resulting in even higher OS temperatures). Second, the fairing/shroud would need to be as much as 1 kg heavier since the metal would be not only required for structure but also would need sufficient thermal mass to prevent overheating. Third, this type of design would require a highly coupled structural-thermal-vibrational analysis since the material properties are all temperature dependent. This makes for a challenging design. For these reasons a metallic only fairing/shroud without a TPS was not deemed to be an attractive option.

Use of a TPS on the fairing/shroud was examined next. The conclusions were very similar to those for the TPS directly on the OS, which are that SIRCA and SLA-561 would likely be the most attractive TPS option. Also, the fairing/shroud would provide additional protection by further isolating the OS from the direct aero-heating via the CO_2 gas-gap. One other benefit of this architecture is that the extra mass required for the TPS would come from the MAV mass budget rather than the OS budget. Although this does not affect the total mass of the MAV, a lower mass and volume OS would ease the requirements for the Earth Return Vehicle that could carry the OS back to Earth.

On one hand, a fairing/shroud would lead to additional complexity in the design of the MAV and require more mechanisms (push off springs, clamshells, pyro-firing mechanisms, cabling, instrumentation, logistics, etc.) On the other hand, since the OS is completely shielded by the fairing/shroud, it would not be degraded by aero-thermal heating and would remain highly reflective (white or similarly solar reflective) and would be easier to track than a ½ white, ½ - dark OS.

III. OS During Orbit

Once the OS is released from the MAV nose cone, it would orbit Mars or the Sun awaiting its rendezvous with the Earth Return Vehicle on its eventual journey to Earth and could remain in orbit for several years. During this period, the thermal integrity of the sample and the OS has to be maintained. As mentioned before, the notional OS is a sphere about ~20 cm diameter and a mass of ~5-10 kg. It has to be optically trackable and/or needs a radio beacon to provide RF tracking.

To be optically trackable, the OS needs to be reflective in the solar spectrum. White paint or shiny metallic outer surfaces would readily provide that feature. Since the OS is exposed to both sunlight and deep space, it is subjected to both solar heating radiative cooling, and a careful thermal control design is needed to ensure that it stays within its allowable temperature limits (maximum and minimum). The thermal environments are summarized in Table 2. Since the orbit parameters are still undetermined, worst case hot (WCH) and worst case cold (WCC) combinations were utilized to bound the analysis, and are shown in Table 2. These bounding environments use the extreme range of beta angles (0 to 90°), a nominal altitude of 500 km, and a Mars-Solar distance of between 1.381 to 1.665 A.U.

Case	Altitude	Orbit Beta Angle	Solar Irradiation	A.U.
Worst Case Hot	500 km	90°	720 W/m^2	1.381
Worst Case Cold	500 km	0°	495 W/m^2	1.665

Table 2: OS bounding thermal environments during orbit phase

For reliability over several years of life, the thermal design needs to be passive (no heaters, thermal sensors, etc.). For a passive design, the main driver is the external surface finish which determines its thermo-optical properties: solar absorptivity (α) and IR emissivity (ε). In particular, the ratio of these two properties (α/ε) determines the OS surface temperature. White paint has a low α/ε ratio of about 0.25 and would maintain its temperature quite low, well below the maximum limit (60 °C) but above its minimum limit (-125 C), in a desirable range for the sample. It would also make it very trackable optically. However, a white painted OS could be much colder than the minimum allowable limit of -40 °C for a radio beacon based tracking system.Two kinds of thermal models were constructed to help in the thermal design, since the detailed mechanical and thermal design is only a concept at this point. The first model was a hollow shell of the OS with only the shell's thermal mass modeled. This would provide the WCH and WCC temperature extremes of the OS and the material inside it because of the very small thermal mass of the shell (~0.3 kg). The OS shell was placed in the WCC and WCH orbits to bound the temperature ranges that could be experienced. Figure 8 shows the temperature variation during an orbit for the WCC and WCH conditions as a function of α/ε. It is evident that both the average temperatures and the temperature fluctuations are strongly related to the α/ε value, with higher values leading to higher temperatures and vice versa.

An α/ε value of < 3 would keep the OS below its maximum limit, and α/ε values as low as ~1 would keep it above the minimum limit, which is a wide range for choosing appropriate external coatings from past heritage. An α/ε value of 2 to 2.5 would lead to the most desirable range of temperatures: -40 to +30 C, which would not only be desirable for the sample's integrity but also the mechanical design of the OS.

These α/ε guidelines can be used to make conclusions about realistic surface coatings for the OS. White paint would bias it cold approaching the minimum limit, with α/ε = ~ 0.25. Uncoated metals tend to have very high α/ε values, as much as 6, so would not be suitable because they would make the OS very hot, in excess of +100 C. Having a "zebra-stripe" of two different coatings would allow one to tune the average α/ε to a desirable range. For example, a 60% white combined with 40% gold “zebra stripe” finish would yield an average α/ε of about 2.5, which is desirable thermally as well as very reflective in the solar spectrum. More other combinations are feasible with the choice of other surface coatings. Regardless of the surface coating, degradation of thermo-optical properties of the external surfaces of the OS over several years of exposure to the space environment would need to be considered to ensure that the corresponding impact on the OS’s temperatures are understood.

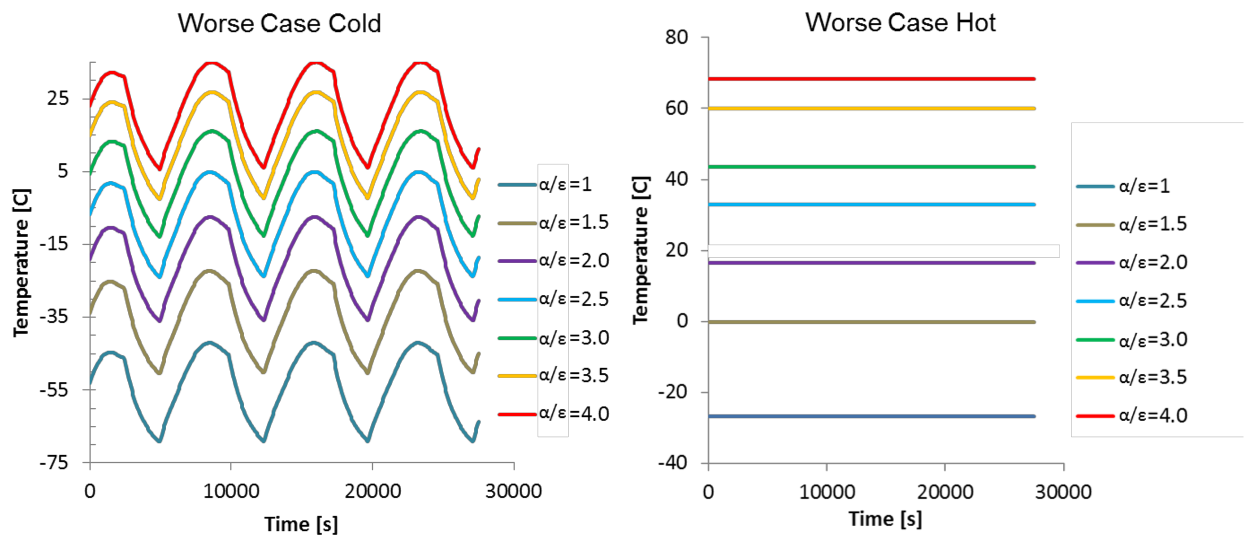

Figure 8: Worst Case Cold and Hot Temperatures of an empty OS shell

Since the above analysis is bounding and very conservative, a second model was also used. This model is a best case since the entire mass of the OS (5 kg in this example) is assumed to be very well coupled to the OS outer shell. Results of this analysis, shown in Figure 9, clearly show that the temperature fluctuations are very small (only a few degrees in an orbit). The damping due to the thermal mass is clearly a big help. This damping effect would limit the minimum temperatures due to the smaller swings. The WCH temperatures wouldn't be affected because of the beta angle of 90^0 leading to a constant and fixed illumination from the sun. The smaller difference between the maximum and minimum temperatures would lead to a wider choice of thermo-optical coatings to maintain the temperatures within their allowable limits.

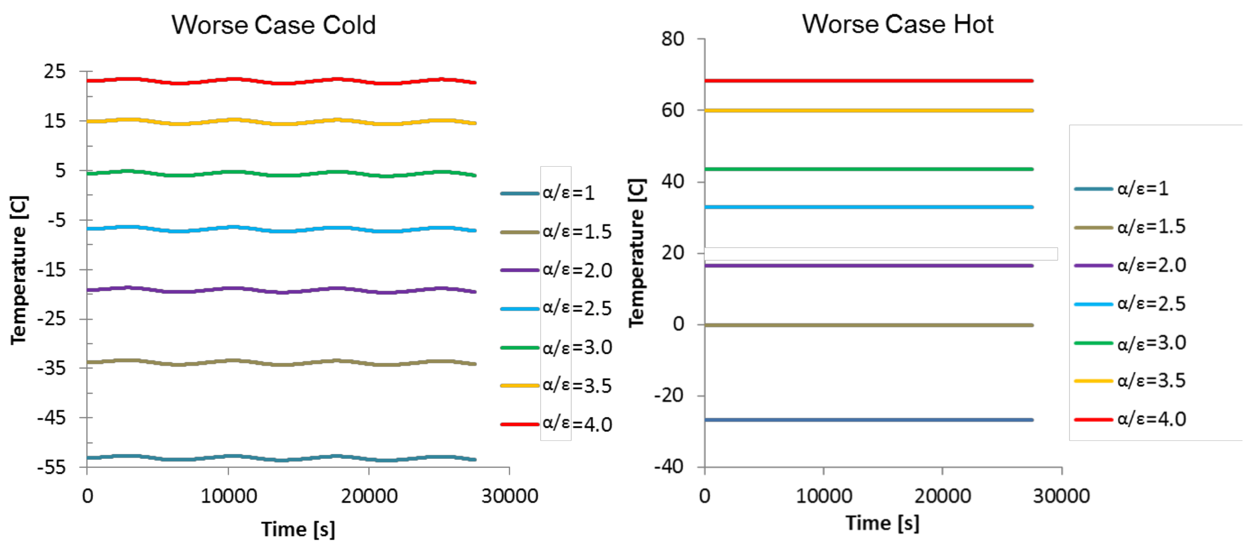

Figure 9: Worst Case Cold and Hot Temperatures of OS shell well coupled to internal mass

IV. Conclusions

Thermal control of the OS during its ascent and orbit phases present a challenging set of requirements and constraints in order to maintain the thermal integrity of the Martian sample. This led of an interesting set of options to meet its requirements. During ascent, TPS directly on the OS or on a fairing/shroud protecting the OS would be attractive options to consider. TPS composed of either ablative (SIRCA, SLA-561, or others) or non-ablative (HRSI tiles) materials would be attractive options. An OS ½ coated with solar reflective material and ½ covered with TPS would allow for optical tracking in addition to thermal protection during aero-thermal heating. During orbit, a "zebra stripe" combination of solar reflective coatings with an α/ε value of 2 to 2.5 would provide for maintenance of temperatures within a relatively temperature narrow range. It should be recognized that these are only conceptual studies of possible options and the final choice of thermal architecture will require significant additional considerations of manufacturing constraints, structural integrity, optical tracking, etc.

Acknowledgments

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