Robust Thermal Control of Propulsion Lines for Space Missions

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In spacecraft that have propulsion lines that are located externally with open bus architecture, the lines are typically insulated by Multi Layer Insulation (MLI) blankets to protect them thermally from the cold space environment. In addition to heat loss through the insulation, mechanical supports used to attach the lines to the spacecraft structure also create heat leaks from the lines. These lines typically have very low thermal conduction in the axial direction, so the heat balance in the lines tends to be very local without much heat spreading. The typical allowable temperature range for hydrazine-based lines is +15/+50ºC. This tight temperature range has to be maintained for every location on these lines. For typical spacecraft, these lines can be several meters long. Temperature control is typically achieved by closed loop monitoring of temperatures along the lines and the corresponding powering of the heaters in a bang-bang approach to maintain the temperatures within the dead band of the control loop. The temperatures of propulsion lines are a function of several parameters with heat loss characteristics of the MLI being the key one. Unfortunately, this same key characteristic (MLI effective emittance) has a large variation along its length due to its dependence on workmanship, which in turn leads to large uncertainties in the propulsion lines' local temperatures. Because of the poor conduction along the axial direction, heat balance along the length varies dramatically from one location to the next, even few inches apart, depending on the combination of the controlling parameters. This paper describes various robust design and implementation approaches that have been investigated to greatly reduce the randomness associated with predicting the temperature of these propulsion lines.

Nomenclature

HRS	=	Heat Rejection System
JPL	=	Jet Propulsion Laboratory
MER	=	Mars Exploration Rovers
MLI	=	Multi-Layer Insulation
MMRTG	=	Multi-Mission Radioisotope Thermal Generator
MPF	=	Mars Pathfinder
MSL	=	Mars Science Laboratory
PRT	=	Platinum Resistance Thermometer
STV	=	Solar Thermal Vacuum
WCC	=	Worst Case Cold
WCH	=	Worst Case Hot

I. Introduction

PROPULSION lines employed in spacecraft require active thermal control to ensure that they remain within their safe allowable operating temperature limits during the entire mission. For Hydrazine, that is commonly employed for propulsion, the typical allowable temperature range is +15ºC to +50ºC to ensure that they have reasonable and adequate margins against freezing (2ºC) at the low end of the temperature range and against hardware operational limits (~+70ºC) for their high temperature limit (thermal hardware constraints and engine

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performance). This tight range of temperature control then necessitates a corresponding thermal design and thermal control system which can make this feasible. This tight temperature range has to be maintained for every location on these lines. These lines crisscross the spacecraft (in their path from tanks to engines) at several locations with varying thermal environments and boundary conditions. Thus the need for tight temperature control conjoined with the large changes in their thermal environments poses a significant challenge in creating and implementing their thermal design.

The typical lines are metallic in construction (Stainless Steel/Titanium alloys) and several meters in length for a reasonable size spacecraft. The most common approach is to glue electrical heaters to these lines to supply heat to them. Temperature control is typically achieved by closed loop monitoring of temperatures along the lines and the corresponding powering of the heaters in a bang-bang approach to maintain the temperatures within the dead band of the control loop. The control of these heaters is done by using mechanical thermostats strategically placed on the lines to automatically duty cycle them without a computer or by employing a spacecraft computer along with a strategically placed temperature sensor (typically a PRT or Platinum Resistance Thermometer) on the line for feedback control. Mechanical thermostats don't need a computer because the thermostats are actuated directly by a bimetallic sensor in a passive way; but they also have hardwired temperature triggers and cannot be changed in flight, which reduces their flexibility. PRTs on the other hand do provide that flexibility of change in the heater control set points to overcome variations in the environments during flight, any surprises or differences between predictions and flight performance, and biasing/fine-tuning of their temperatures to create an optimal balance between the low and high temperature limits during flight.

To minimize the required heater power, the propulsion lines are typically insulated by Multi Layer Insulation (MLI) blankets to protect them thermally from the cold space environment. In addition to heat loss through the insulation, mechanical supports used to attach the lines to the spacecraft structure also create heat leaks from the lines. These lines typically have very low thermal conduction in the axial direction due to the poor thermal conductivity of the metals employed. The temperatures of propulsion lines are a function of several parameters with heat loss characteristics of the MLI being the key one. Unfortunately, this same key characteristic (MLI effective emittance or e*) has a large variation along its length due to its dependence on workmanship, which in turn leads to large uncertainties in the propulsion lines' local temperatures.

The temperature of propulsion lines is a function of several parameters: heat loss characteristics of the MLI, local radiative thermal environment on the external surface of the MLI, local heat loss via the mechanical supports, set points of the thermal control of heaters, heat conduction along the axial direction and the local heater power density (see Figure 1 for typical heat balance of propulsion lines). As discussed above, due to typically poor conduction along the axial direction (small heat spreading), heat balance along the length varies dramatically from one location to the next, even few inches apart, depending on the combination of the parameters delineated above, which control their temperature. Hence to control every location to the tight temperature limits would require an inordinate number of separate sensors, heaters zones and computer channels. So typically only one temperature sensor controls different patches of heaters along a long (a meter or so) section of the line – one location of the line is carefully chosen as the controlled location, which energizes all heater patches with a single computer channel (when PRTs are used) or a single mechanical thermostat performs the same function.

Hence, because of the variation in the combinations of the above parameters that control the local heat balances and temperatures, thermal engineers designing the thermal control of these lines inevitably get caught between trying to avoid freezing at one location of the controlled zone and overheating of another location of the same controlled zone. It always becomes a trade of how much positive (or negative) margin one can achieve or live with for freezing versus the corresponding one for overheating.

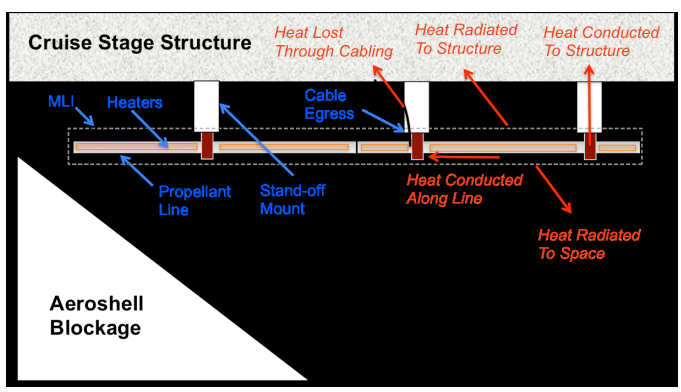

Figure 1. Typical Propulsion Line Heat Balance

If the above parameters were deterministic in nature, at least one would have a reasonable chance to design against their variations, and meet the allowable temperature limits. But the key parameter that controls the heat balance and temperatures of these lines also happens to be the most unpredictable and random. This parameter is the effective emittance of the MLI, which is simply the radiative heat loss characteristic of the MLI. The effective emittance of MLI is a fraction

of the actual heat loss via the MLI when compared to the maximum ideal heat loss from the propulsion line metal surface if it were black.

Propulsion lines are typically small in diameter (6 mm or 1/4") and when they are wrapped with MLI, the external area of the can be significantly larger than the line itself. The heat loss is then larger because of the larger external surface area. Also the workmanship in typical MLI applications on these lines can lead to large variations in the "tightness" of the MLI – loosely wrapped layers will lead to a lower effective emittance, whereas tightly wrapped ones lead to higher effective emittance. The reason is that MLI is not simply layers of radiation shields but also series of thermal "shorts" between these layers due to physical contact between them – so the tighter the fit, the more the thermal shorting between the layers.

Not only does the effective emittance vary for straight sections of lines due to workmanship, it also varies around tight corners or bends when the MLI layers are even more crunched up and less deterministic. Finally, near the mechanical supports, there is a lot of variability of covering these locations with layers of MLI.

For typical well constructed MLI for space flight missions, tests done to determine the effective emittance of MLI for the same kind of shapes or configuration (straight to straight, bend to bend and support to support), factors of 5x variations of effective emittance have been observed. When coupled with variations between straight vs. bend vs. support locations these lead to a wide dispersion of effective emittance.

Since the effective emittance of MLI is the single most important determinant of the heat balance and the resultant temperatures of the propulsion line as a function of location, these large variations lead to very un-robust thermal designs, particularly when this is coupled with wide variations in the external thermal environment (lines in the neighborhood of warm vs. cold objects, being in the sun vs. viewing cold deep space).

Hence one of the primary goals is the numerical understanding of these temperature variations, analytical means to accommodate them, and better yet, to create designs that are not affected by these variations, in order to create a robust and predictable thermal control system.

II. Typical Configurations & Designs for Thermal Control of Propulsion Lines for Space Missions

Two examples of the propulsion line configurations for the Mars Science Laboratory (MSL) mission are shown in Figures 2 (Cruise Stage) and 3 (Descent stage). The cruise stage has an open bus architecture with propulsion lines seeing vastly varying views of space, solar array, propulsion tanks, radiators & miscellaneous hardware, which in turn have vastly different temperatures that also change significantly during cruise to Mars. This represents the most challenging set of environments of the varying and different temperatures of components that are in view of the propulsion lines. The descent stage also has a variety of different temperature components that are viewed by the propulsion lines; however, since the descent stage is an enclosed aero-shell which blocks the view to space, the differences in the thermal environments encountered by the propulsion lines are smaller when compared to the cruise lines. Other spacecraft buses have similar configurations, although they tend to be typically enclosed buses rather than open to space; however, even for open buses, propulsion lines do traverse some distance between the enclosed bus that contains the propellant tanks and the outboard engines/thrusters, so these sections of lines are also subjected to space views too, albeit to a smaller extent than the open bus lines.

In either case, the propulsion lines view a variety of thermal boundary conditions (spatial and temporal). And these environmental boundary conditions significantly affect the heat balance of the lines, plus each location has a different view to them. Hence, as discussed above, since the heat balance of the lines are very localized due to poor heat spreading in the lines, differing boundary conditions at each location create different temperatures for the same heat input.

In addition to the differences in the boundary conditions, as explained above, the thermal characteristics of the lines also vary along their lengths from one location to the next. Some of these are deterministic (e.g., line mechanical supports and temperatures of objects in view of the lines) and can at least be analyzed reasonably accurately with detailed computer models to represent them. On the other hand, some characteristics are not deterministic and have significant variation at every location along the line lengths (e.g., MLI effective emittance) with the variation being primarily due to local workmanship differences. Since these workmanship variations are relatively random in nature, but can be bounded within a range by analogy to previous similar applications, the thermal analyses can only accommodate their differences as error bars, which translate to error bars in temperature predictions of the lines. Since the allowable temperature range is very narrow (+10°C to +50°C), the combination of the (non-deterministic) variations of the thermal characteristics of the lines in conjunction with the more deterministic set of variations in the boundary temperatures in view of the lines thus represents a very significant challenge in the design, prediction and implementation of thermal control of propulsion lines for space missions.

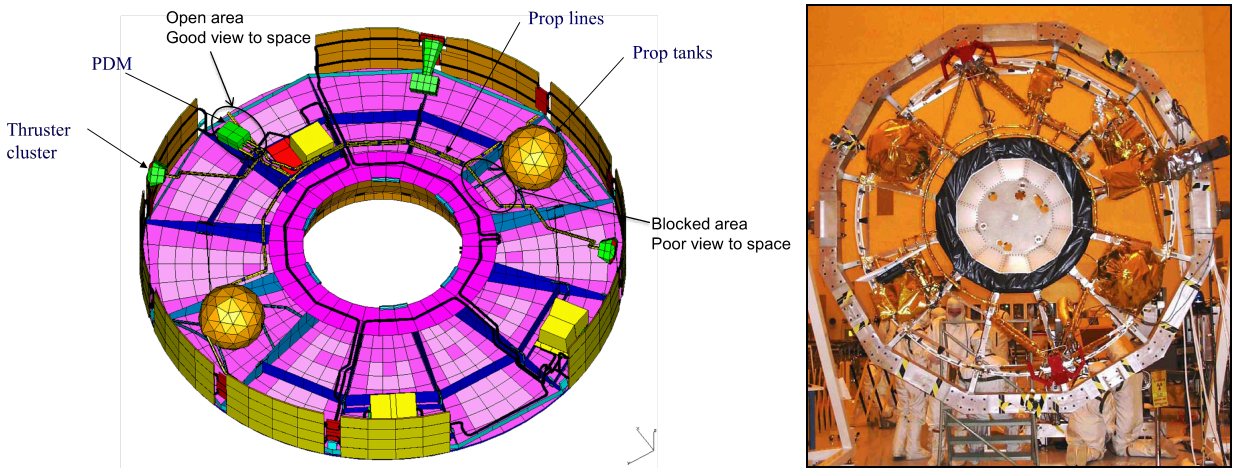

Figure 2. MSL Cruise Stage Propulsion Line Layout & Thermal Model

III. Historical Context of Problem Statement

Thermal control of propulsion lines has been an ongoing challenge for missions at JPL as well as other institutions and companies over several decades. Some recent examples were the Mars missions[1,2,3,4] at JPL: Mars Pathfinder (MPF), Mars Exploration Rovers (MER) – Spirit & Opportunity and the Mars Science Laboratory (MSL). During Solar Thermal Vacuum (STV) testing MPF had several locations that were found to be below freezing temperatures and some that were very hot violating their maximum limits. Many local changes (re-blanketing, shields over mechanical thermostats, aluminum tape heat spreaders, etc.) were made after testing and still some locations violated their allowable limits. The use of mechanical thermostats also aggravated the problems because their set points could not be adjusted in flight. MLI e* variations was judged to be the biggest source of these problems. For MER, in spite of the lessons learned from MPF, and the improved design that employed computer control of heaters (with PRTs providing the local line temperatures for feedback), several problems were uncovered during testing and were partially mitigated by changing set points for control. But even then some problems occurred during trajectory maneuvers with the risk of some lines going below freezing. Fortunately, knowledge of this risk just before the maneuvers allowed for its mitigation by raising the set points of the control PRTs (although that came with some sections of lines overheating, which was deemed to be a smaller and more tolerable risk than frozen propulsion lines). Again the variations in the MLI e* were judged to be a key contributor of these problems.

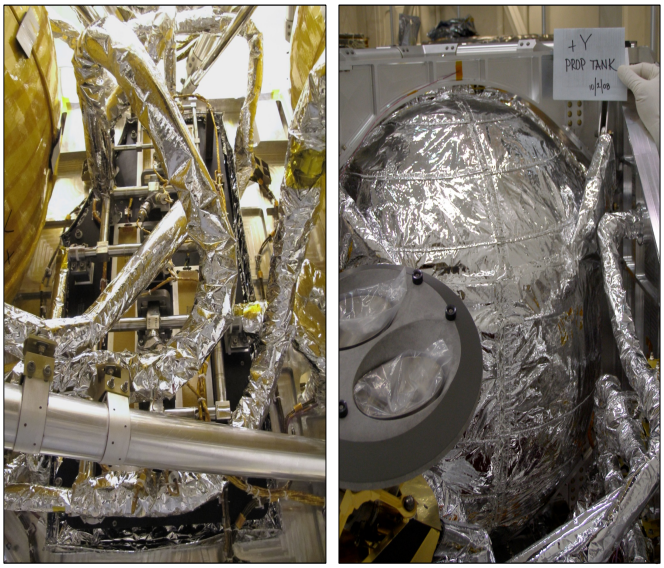

Figure 3. MSL Descent Stage Propulsion Line Layout

For MSL, again in spite of lessons learned from MPF/MER, during testing a propulsion line section (controlled by a single PRT) exhibited simultaneous violations of both the minimum & maximum temperature limits *within the same section*. The support isolation was increased & control set points were adjusted upwards (again with the attendant consequence of overheating a portion of the line above its allowable limit). These changes

necessitated an expensive retest to ascertain that the intended results were as expected and to ensure that no deleterious unintended consequences were created. Variations in the MLI e* were judged to be a key contributor of these problems.

For a non-Mars mission at JPL, the DAWN mission also encountered propulsion line thermal control issues. Simultaneous violations of low and high temperature limits were observed during testing and several PRT locations and change in set points were required before launch to overcome these problems. Lack of detailed accounting for local mount thermal conductances, thermal environments and of course the variations in the MLI e* were judged to be the source of these problems.

So the general trend in all these examples is the inherent lack of robustness in the traditional thermal design of propulsion systems, lack of predictability due to variations in the lines' thermal characteristics, and the inherently very small heat balance (~10 mW per cm) that varies all along the lengths of the lines due to poor heat spreading. Expensive retesting has also been necessitated as a result of these problems in order to fix them before launch, and in spite of these fixes, problems have been encountered during flight. Hence the need for more robust and deterministically predictable thermal designs of these lines is warranted to ensure continued mission success (luckily no known mission failures have been attributed to propulsion line thermal control).

IV. Numerical Examples to Illustrate Magnitude of Potential Problems

A simple numerical example is presented below to illustrate the nature and magnitude of possible potential for temperature variations due to a major culprit, the MLI e* variations, due to workmanship. From past Mars projects a variation of about 5:1 (highest to lowest value of e*) has been generally observed for the MLI e*, and this is only for similar sections of lines (e.g., within straight sections). As one compares between different kinds of sections of lines (bends vs. at supports vs. straight), these variations are even more pronounced (10:1). This is illustrated in Table 1 that was created by culling data from MPF/MER/MSL data.

Category	Low e*	Nominal e*	High e*
Straight lines	0.022	0.049	0.11
Bent lines	0.049	0.11	0.245
Stand-offs	0.033	0.074	0.165

Table 1. MLI e*variation from past Mars projects at JPL

Basic equation for heat loss via MLI, assuming no local additional losses via supports and miscellaneous paths, and assuming negligible heat spreading along the tubing is as follows (this is simplified and bounding to illustrate the point). It is assumed that the controlled location has a PRT whose measured (and controlled) temperature controls the heater power (duty cycle). This is also the same power inserted at a different location with a different e* than the controlled one.

$$Q_{heater} = \varepsilon^*\sigma A(T_{prop}^4 - T_{sink}^4)$$

So for a given Q_{heater} and space sink, T_{sink} (4 K, close to zero), T_{prop} is proportional to $(\varepsilon^*)^{1/4}$. If the ε* range ratio is 5:1, it implies that the resultant T_{prop} ratio (max to min) is 1.5:1, by solving the above equation. In other words, if at the controlled PRT location, $T_{prop,1}$ = 0ºC, then along the same section of the line at a different location, with an e* different from the one at the controlled location (and the same heater power), $T_{prop,2}$ = 137ºC! To carry this further, if the two locations have different boundary temperatures in their view, the situation is further exacerbated, and the hotter location could be warmer yet depending on the differences between the two local sink temperatures. Figure 4 illustrates these points graphically to show the trends as a function of e* ratios. The figure on the left shows that if location 1 is controlled to 30ºC and if the e* ratio of location 2 (uncontrolled) vs. that of location 1 was allowed to vary from factor of 1 to 5, the corresponding temperature of location 2 could be well below the freezing point of hydrazine even if the ratio were as small as 2:1. Similary, in the inverse situation, to ensure that the uncontrolled location (2) does not drop below 30ºC, location 1 would have to be held at well above the max AFT limit of 74ºC. In fact, one would violate the max AFT limit of 74ºC at location 1 for e* ratios as small as ~2 between locations 2 and 1 to ensure that location 2 does not drop below 30ºC. Hence from this simplified analysis it is quite evident that non-deterministic variations in e* of MLI can have a profound effect on the variations in the temperatures of the propulsion lines. This same figure illustrates the above concerns with freezing and overheating of the propulsion lines due to MLI e* variation by performing simple back of the envelope calculations.

Figure 5 illustrates the results of a detailed computer modeling of the MSL cruise stage propulsion lines in which the actual flight configuration and the corresponding geometry of the spacecraft was simulated to get a realistic assessment of the temperature distribution of the lines. This clearly shows the very significant effect of

temperature distribution on the lines due to the MLI e* variation, where the same thermally controlled zone can be below freezing at one location within it while another location could be extremely overheated.

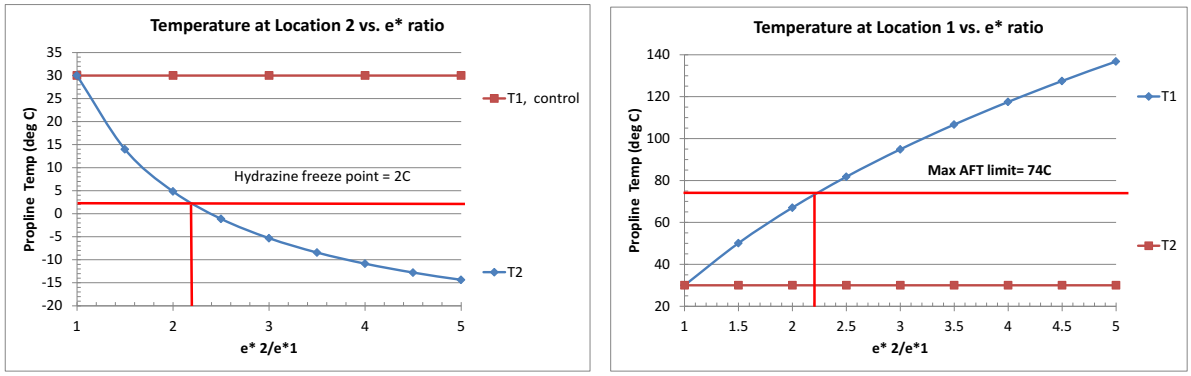

Figure 4: Freezing (left) and overheating (right) concerns due to variations in MLI e*

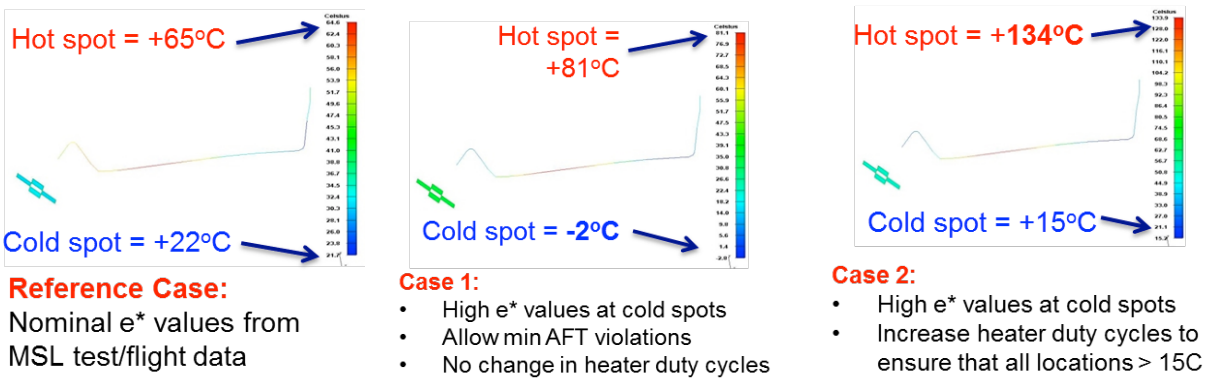

Figure 5: Effect of variations in MLI e* even with realistic analyses

V. Typical Analytical Means to Accommodate Effect of MLI e* Variations to Predict Propulsion Line Temperatures

Lacking the employment of more robust solutions, the approach used by JPL is to simply use the MLI e* range as an error bar in the analysis to predict the resultant line temperatures. Detailed thermal models are first constructed that account for all the configurational details (line layout, line supports, heater power capacity, PRT location, line properties, neighboring components, cables, heat spreader tape, external environments and other boundary conditions). In these models the MLI e* is then allowed to be a random variable within the previously observed ranges (based on straight vs. bends vs. supported portions, as shown in Table 1). The models are then run to understand the robustness of the line temperatures to variations in the appropriate MLI e* values. Heater duty cycles are then computed to ensure that under any combination of e* values within their range, no location (node in the model) violates the min temp limit against freezing, since freezing any location is catastrophic (overheating is less catastrophic, within limits, though).

Since e* at any location is pretty much a random number based on workmanship differences, worst case bounding combinations of e* values are chosen in the robustness check to find the possible range of temperatures in all the line nodes with the constraint of no location falling below their minimum allowable temperature limit. As mentioned above, a single PRT is used to control the entire section of every separate line, and the entire propulsion line system is divided into several sections, each about a meter long. All heater patches on any section then share the same duty cycle (based on what the control PRT location demands to maintain its temperature), because a single software switch in the flight computer controls the on/off duty cycle of all heater patches in a section. So within any controlled section of every line, a high e* value is used for cold portions of the line section and low e* value is used for the warmer section. This then forces the cold spots (which also are near the control point) to run at a high duty cycle to overcome the leakier MLI at those locations (high e*), which then also leads to the less leaky MLI locations

(low e*) to run warmer (using the same duty cycle for all nodes within a controlled section). This then crystallizes all the resultant hot spots in the lines.

Then the hot spots temperatures are compared to the high allowable temperature limit to judge the robustness of the thermal design. All this is done prior to thermal testing which then unravels the actual e* values at every instrumented location. If the combination of propulsion line temperatures during the test and exercising of the model for flight conditions yields acceptable temperatures for all location, then no further changes are necessary. Otherwise adjustment of control points is tried to investigate if any changes to the propulsion line MLI or physical location of control point is necessitated (usually followed by a retest) or adjustments of control point values is sufficient.

VI. Development Tests

The workmanship variations that have been discussed above are relatively random, even though they tend to follow a bias with straight sections tending to be lower e* values, whereas bends have the highest ones and those near support somewhere in between (as shown in Table 1). They also roughly have a 5:1 ratio between the maximum and minimum values for each kind of section (straight to straight, bend to bend, etc.). Unfortunately, development tests tend to be not very useful for validation of the propulsion line thermal design because of the variation of the MLI e* values being so workmanship related. Also development tests typically use simplified and idealized configurations as compared to the actual flight system which is significantly more complex to be characterized reasonably well in a development test.

VII. Solar Thermal Vacuum Testing & Impact on Mission Operations

Since the temperature of the lines are very hard to predict analytically due to the variations discussed above, solar thermal vacuum testing of the flight spacecraft including the flight rendition of these lines is extremely important to ensure that adequate thermal mapping of these lines is conducted in environments that simulate the extremes that would be encountered during flight. Adequate number of temperature sensors (many for test plus some for flight) placed at reasonably closely spaced locations on all the lines is essential to map the thermal characteristics accurately at several locations. The testing also anchors the thermal models to then make predictions and adjustments for flight conditions.

But in spite of extensive testing due to the workmanship related variations, as discussed above for earlier flight projects, problems are still encountered in all these missions, where the temperatures remain in violation of their allowable limits, in spite of adjustments of their control points and even after post-test changes of the flight units. Sometimes that leads to an expensive retest of the entire spacecraft because any changes that are made to the design, if they are related to workmanship reasons, deem it necessary because the change itself (however well engineered and thought through) leads to a change in thermal balance of the zones modified.

And finally, even after an initial test or a retest, differences exist between the flight conditions and the tested ones, plus some surprises are encountered particularly related to transient conditions (trajectory maneuvers, fault conditions, etc.), which are usually not tested. So this leads to constant and expensive vigilance of these lines throughout flight combined with periodic updates to control schemes utilized for ensuring the thermal safety of these lines.

VIII. Potential Solution Approaches

Realizing that the basic thermal design of propulsion lines for spacecraft is not robust, several approaches have been investigated by the author of this paper. They are enumerated below:

- *Black tape on propulsion lines:*

One way to achieve robustness is to simply not use MLI insulation at all (bare lines) and use some very deterministic emittance tape that would be glued to the outside of the heater surfaces. This approach would work in terms of being deterministic because the emittance of this tape would be well quantified (not dependent on being subject to workmanship like MLI would be).

For example, if one used black tape, which is very deterministic in its surface emittance, it would lead to a predictable set of temperatures for the propulsion lines. But it would lead to heater powers that would be 10 to 20X larger than for typical MLI applications. This exorbitant increase in power would be unpalatable because it could lead to several hundred watts of power being required for thermal control for these lines. In terms of robustness, the black tape would be quite robust because typical fingerprints and other contaminants tend to be high in emissivity.

- *Low emittance tape on propulsion lines:*

An alternative approach to avoid these high powers could be based on employing low emissivity tape (like aluminized Kapton), which is deterministic in its emissivity (only for pristine tape) and could lead to low power needs. Unfortunately, there is a significant risk associated with this approach and it has to do with contamination when handling the tape during installation as well as during integration of the spacecraft. Because of the low emissivity of this tape, it is also very sensitive to contamination, e.g., fingerprints, any out-gassing from neighborhood components, etc. So this approach would make the design very vulnerable to being not robust thermally.

- *Clamshell around propulsion lines:*

The novel approach overcomes the concerns associated with all of the above alternative approaches as well as those with the MLI approach. It basically consists of a "clamshell" design in which the inner surface of the shell is coated with low emissivity aluminized Kapton tape and the outer surface is covered with black tape. It is shown schematically in Figure 6. This clamshell completely encloses the propulsion line. The line itself is covered with its heater, which in turn is covered completely with black tape.

This approach would be low in heater power needs because even though the outer surface of the propulsion line (and its heater) are covered with black tape as well as the outer surface of the clamshell, the inner surface of the clamshell is covered with low emissivity aluminized Kapton tape. Hence the heat loss from the line will be small and comparable to the MLI based one, because it is governed by three radiative resistances in series - small resistance for black tape on propulsion lines, large for low emissivity tape on the clamshell's inside surface and small for the clamshell outer surface. The large resistance for heat flow, due to the clamshell's inner surface, then controls the overall heat loss from the line and then keeps the heater power needs to be small and comparable to that for MLI based designs.

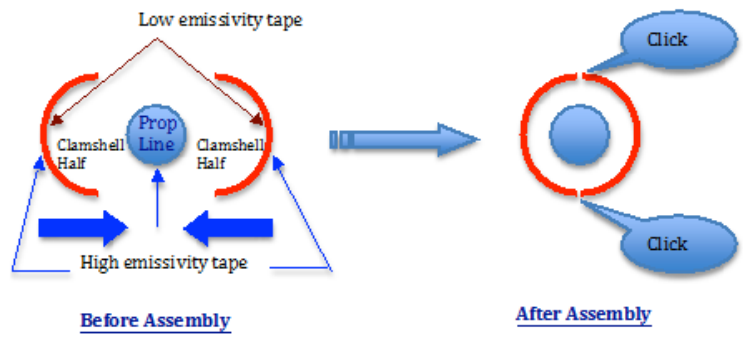

Figure 6: Clamshell concept for Propulsion Line Thermal Control Balance

In terms of contamination changing the radiative properties of surfaces, since the clamshell's inner surface is always protected during handling and is only installed after all the work on the propulsion lines has been completed, the controlling surface, which is the clamshell's inner surface, is always in pristine conditions. Also, in terms of susceptibility to out-gassing, the clamshell by its design protects the low emissivity inner surface from the outgassing products.

For the straight sections of the propulsion lines, it is easy to conceive of two halves of the clamshell that would be "clicked" in place from the two halves of the line as shown in Figure 6 by using straight sections of clamshell halves. For locations near bends or supports, similar clamshells that are approximately similar in shape to the bends could be used to cover the line sections there. Implementation of this scheme on tight bends would, however, be problematic due to concerns with geometrically matching the clamshell shapes to the bends.

The clamshell halves could be made of a low thermal conductivity, lightweight, material like G-10, or Delrin. They could either be supported from the structure on which the lines are hung from or directly from the lines themselves.

One important detail in the design of this concept that would need to be addressed is the local thermal shorting between the lines and the clamshell halves – if that shorting is significant, then it could lead to thermal gradients in the lines. So a careful accounting of these shorts would need to be undertaken to minimize their impact.

- *Thermally Couple a Fluid Loop to the Propulsion lines:*

If a tightly thermally controlled interface pre-exists (or can be created for this purpose), which can be (thermally) coupled to the propulsion lines, and if the propulsion lines can in turn be reasonably thermally isolated from other thermal environments or boundaries, then a very robust thermal control system for propulsion lines can be devised. The propulsion lines would then be essentially "slaved" to the temperature of the fluid loop if the thermal coupling between the propulsion lines and the fluid loop significantly overpowers the corresponding thermal coupling to the

external environment. A fluid loop (mechanically pumped single phase using Freon-11), liked that used for thermal control of the MSL spacecraft and rover, could be additionally employed for the thermal control of propulsion lines by adding extra fluid loop (a.k.a, Heat Rejection System or HRS) tubing to closely follow the propulsion lines and be thermally coupled to them. A schematic of this concept is shown in Figures 7 & 8.

The fundamental concept to implement this thermal coupling is to position the HRS lines close to the propulsion lines and insert an interface filler material like Chotherm that is commonly employed to thermally couple electronic components to their mounting interfaces to create a strong thermal coupling between them. As an example, a 3 mm thick strip of Chotherm which is also about 3 mm wide sandwiched between the two lines would provide sufficient thermal coupling between them, while at the same time provide enough springiness to the Chotherm strip to conformally fill in the surface roughness in the tubing to create good thermal contact. The two lines would be mechanically held together by zip ties or straps to maintain contact and provide Chotherm compression to create a good thermal contact between the two lines. The propulsion line-Chotherm-HRS line assembly would be also covered with standard MLI to minimize the thermal coupling and interactions with the external environment and retain the relative bias between the lines as the predominant thermal coupling. It should be noted that even though the MLI would still have the e* variation even in this system, the much stronger coupling between the two lines via the Chotherm is predominant and as a result, the temperature of the propulsion line is fundamentally governed by the HRS line temperatures rather than the MLI e* values or the environment.

If the HRS pre-exists then no heater power would be required because the HRS recovers waste heat from the various heat dissipating components in the spacecraft and automatically creates appropriate temperature levels in the HRS line that are commensurate with the propulsion line allowable temperature range requirements. As an example for MSL, with the HRS lines at 35°C, detailed computer modeling of this concept show that the propulsion lines can be controlled to a very tight range within 5°C of the HRS line temperatures. This analysis assumed the same bounding extreme range and combination of MLI e* values as well the extreme values of environmental and boundary temperatures.

A variant of the Chotherm thermal contact concept is the use of spaced apart local thermal straps to periodically thermally couple the two lines. The main incentive for the strap concept is to minimize or eliminate any mechanical coupling between the two lines which could lead to undue mechanical stress on either line because of the stiffening effect of the mechanical coupling. Heat spreader layers of aluminum tape on the two lines (discussed in the next section) would allow for larger spacing of the thermal straps, thus reducing the need for too many straps. A thermal analysis of two lines with straps and aluminum tape would yield an optimum spacing of the straps to create a uniform temperature distribution in the propulsion lines.

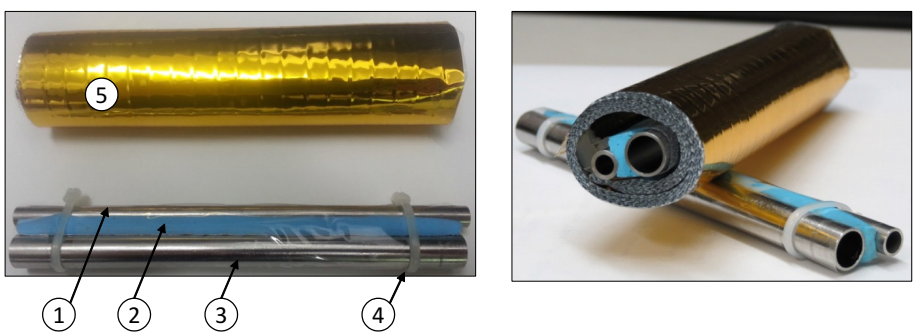

Figure 7: Fluid Loop to Propulsion Line Thermal Coupling Concept. 1) ¼" Propulsion Line, 2) Thermal Interface (CHO-THERM®), 3) 3/8" HRS Line, 4) "Straps," and 5) MLI.

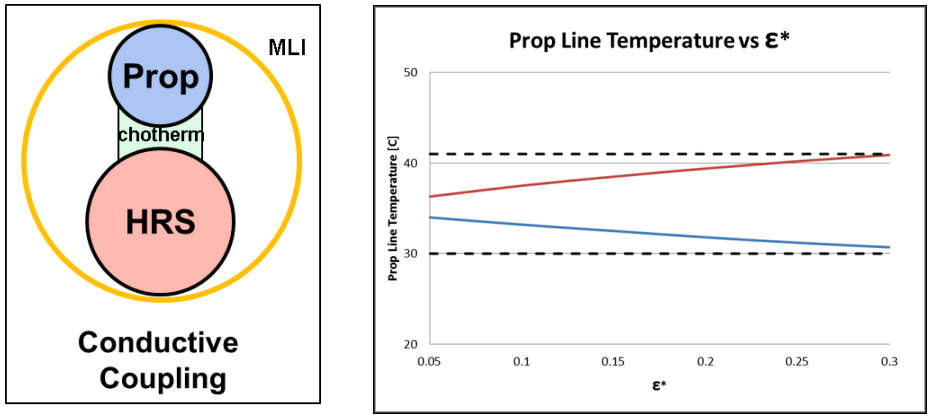

Figure 8: Prop line Thermal control by thermally coupling them to Fluid Loop

- *Heat spreader tape (Aluminum) applied to propulsion lines*

One of the key deficiencies of current thermal designs is the lack of heat spreading along their lengths. So local hot or cold spots remain localized due to the localized thermal balance. If the propulsion lines were constructed of aluminum then they would be thermally very conductive, but unfortunately they cannot be used due to the extremely corrosive nature of the propellant. An obvious scheme is to significantly enhance this lateral conduction to smear out the hot and cold spots and as a result reduce the large range of temperatures experienced by the propulsion lines. Since the thermal conductivity of the metal alloys utilized for the propulsion lines is quite small (16 W/m-K for stainless and 8 W/m-K for Titanium) and other much higher thermal conductivity materials like Aluminum (k = 170 W/m-K) and Copper (k = 380 W/m-K) are not compatible with the propellant fluids, the only option is to cover these lines with several layers of high conductivity Aluminum of Copper Tape (~0.1 mm thick for each layer). So 10 layers of Aluminum tape would be worth about 1 mm of total thickness and that would be ~20 X more conductive than the parent stainless steel tubing of 0.5 mm thickness. Detailed computer modeling of this concept was undertaken and showed that it would greatly reduce the temperature gradient of the propulsion lines as shown in Figure 9 – it also shows sample of tubing covered with increasing number of layers of tape.

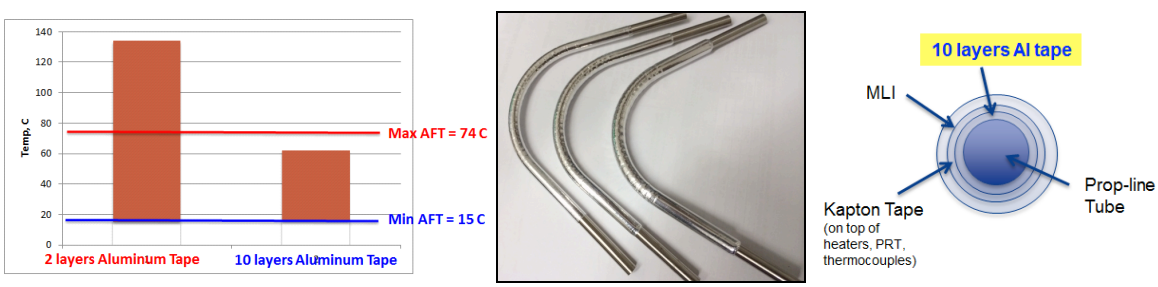

Figure 9: Reduction in propulsion line temperature gradients with ten layers Aluminum Tape (left), sample tubing covered with increasing numbers of tape layers (center), and complete thermal hardware stackup (right)

IX. Applicability & Limitations of Various Approaches

As can be easily gathered from the inherent nature of these significantly different schemes, each of them has advantages and disadvantages associated with them. These differences can be categorized into the following key facets, with Table 2 comparing the proposed schemes in terms of their relative pros and cons:

- Relative robustness
- Thermal control power required
- Sensitivity to contamination
- Mass
- Impact on rest of the spacecraft system

- Thermal modeling complexity
- Ease of implementation
- Mission operations
- Cost
- Schedule

Category	Thermal Mitigation Scheme				
	Black Tape on lines	Low emissivity Tape on lines	Clamshell around lines	Fluid Loop connected to lines	High conductivity Tape on Lines
Relative robustness	Very robust	Not robust due to contamination sensitivity	Robust	Extremely robust	Robust
Thermal control power required	Extremely large, probably not affordable for most missions, 10x to 20x larger than typical	Smaller than typical MLI designs	Small, comparable to typical MLI designs	Small, comparable to typical MLI designs; if fluid loop pre-exists then could be zero if waste heat is harvested from other thermally controlled	Same as for typical MLI designs
Sensitivity to contamination	Not sensitive	Very sensitive	Not sensitive	Very insensitive	Not sensitive
Mass	Small	Small	Small	Small: for mass of extra fluid lines (if fluid loop pre-exists) Medium: for fluid loop if it does not pre-exist	Small
Impact on rest of the spacecraft system	Neglible, except for thermal power	Negligible	Small, comparable to typical MLI designs	Medium: due to analysis required to confirm integrity of mechanically coupling propulsion lines with fluid loop	Negligible
Thermal modeling complexity	Small	Significant due to lack of robustness (contamination	Small	Much simpler due to extreme robustness	Small
Ease of implementation	Relatively easy	Relatively easy	Harder than typical MLI designs, but still realtively easy	Harder than other concepts	Very labor intensive due to several (10) layers that need to be individually installeed on every
Mission operations	Simpler due to thermal robustness	Very complex due to lack of thermal robustness	Simpler due to thermal robustness	Much simpler due to thermal robustness	Similar but simpler than typical MLI designs
Cost	Relatively small impact	Relatively small impact	Relatively small impact	Medium: due to implementation reasons	Relatively small impact
Schedule	Small impact	Small impact	Small impact	Medium: due to implementation reasons	intensive nature of tape installation

Table 2. Pros and cons of Thermal Mitigation Schemes

X. Conclusions

Thermal control of spacecraft propulsion lines has been and continues to be an ongoing challenging problem for current and future missions. The main concern is related to the lack of robustness of the temperature control systems employed for these lines. One of the key reasons for this is the poor conduction along the lines, coupled with the large variations in the heat loss characteristics of the MLI blankets employed for covering these lines. Several alternative schemes to mitigate these concerns have been suggested in this paper. One of them (use of several layers of aluminum tape) is currently being employed for the Mars 2020 mission. Use of the suggested schemes or their variants can significantly improve the robustness, predictability and mission operations for future space missions, thus greatly reducing the risk to these missions due to concerns with thermal control of propulsion lines.

Acknowledgments

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