An Innovative Very Low Thermal Power Waste Heat Recovery System for Thermal Control of Deep Space Missions - *A Thermal Flask in Space*

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Future missions to deep space, such as those to the outer planets (Jupiter, Saturn, etc.), which would rely on solar photovoltaic power, would need extremely large solar arrays to produce sufficient power for their operations because solar intensity is so low at those locations. Hence any additional power that would be needed for thermal control is extremely limited. Previous deep space missions like Juno (to Jupiter) required almost 200 W of electrical power for thermal control. This is prohibitively large for many future mission concepts, and leads to them needing very large solar arrays. For Saturn, where the solar flux is 1/4th the flux at Jupiter, this would entail an extremely large increase in the solar array size to accommodate the need for thermal survival power, which would be prohibitively large in size and mass, and very expensive.

Hence there is a need to come up with a thermal architecture and design options that would not need such prohibitively large thermal power levels. One solution relies on harvesting the pre-existing waste heat from all the heat dissipation that would be present from operation of electronics, instruments, etc. for their own functionality. For example, for a generic Saturn mission, the various electronics would already dissipate about 200 Watts of heat that is simply "thrown away" to space from the spacecraft surfaces. The amount of thermal power that would be required for the safe thermal control of components within the spacecraft in deep space would be roughly of this magnitude for this class of spacecraft. So it makes good sense to try to harvest the waste heat and employ it to maintain the temperatures of all the components within their allowable limits. In particular, propulsion systems typically need to be kept above their freezing limits, around room temperature (15°C). Electronics needs to be kept typically above -40°C and batteries above -20°C.

The next question becomes how to harvest this waste heat and direct it to the components that would need it for their survival. The proposed system utilizes a mechanically pumped, single phase fluid loop to pick up the waste heat from components attached to this loop's tubing and then directed to a thermal flask that has tubing attached to it. The thermal flask is cylindrically shaped and contains essentially all systems and components in the spacecraft within it, with the exception of the solar array, antennae, thrusters and various apertures of instruments, etc. to allow them an unobstructed view of space. Waste heat from the heat-dissipating components warms up the fluid and is carried to the flask surface and deposited on it via the fluid loop's flow. The entire flask is covered with Multi-Layered Insulation (MLI) to minimize the heat loss from the flask and allow it to remain warm. Hence the flask essentially creates a thermal environment within which the spacecraft components reside. The temperature of the components within the flask is then essentially the same as the temperature of the flask. This approach could be a very enabling feature for deep space missions.

This paper describes the approach utilized for this thermal architecture, along with its mechanical and implementation aspects. Additionally it will compare and contrast this approach with the more conventional solutions utilized earlier.

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Nomenclature

ATLO	=	Assembly, Test, and Launch Operations
A.U.	=	Astronomical Unit (1.5×10^{11} m)
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
TRL	=	Technology Readiness Level
WCC	=	Worst Case Cold
WCH	=	Worst Case Hot

I. Introduction

Deep Space Missions, like the ones going to outer planets (e.g., Jupiter, Saturn, etc.) which would rely on solar photovoltaic power, need extremely large area & heavy solar arrays (to produce that power for their operations) because the solar intensity is so low at those locations. Hence any power that is needed for thermal control is extremely limited. Previous deep space missions like Juno (to Jupiter at ~5 A.U.) and the currently envisioned Europa Clipper Mission (to Europa, a moon of Jupiter) would require as much as 200-300 W of electrical power for thermal control. This is prohibitively large and leads to needing very large and heavy solar arrays (on the order of ~20-30 m^2 just for thermal reasons). For Saturn where the solar flux is 1/4th of at Jupiter, it would entail a further extremely large increase in the solar array size (4x larger than for Jupiter) – as much as 100 m^2 - to accommodate the need for this level of thermal survival power, which would be prohibitively large in size and mass, and very expensive. Hence there is a need to come up with a thermal architecture and design that would not need such prohibitively large thermal power levels.

Power needs for thermal control are related to keeping typical spacecraft components within their allowable temperature limits (these contain appropriate margins against their qualification limits):

Propulsion System: +15°C → +50°C

Electronics: -40°C → +50°C

Battery: -20°C → +30°C

II. Typical Environments Encountered

The cruise from Earth to these outer planets leads to inherently a progressively and extremely cold set of environments during the journey. This cold environment near Jupiter, Saturn, and even more distant planets essentially results in typical effective sink temperatures approaching that of cold space (from a T^4 difference based radiation heat loss point of view). The lack of any substantial "free" heat from the Sun and the requirements to maintain components in a relatively warm range leads to large heat losses to space per unit area of the spacecraft. In addition, these ambitious missions would also need very large area spacecraft. So this leads to very large losses that have to be overcome by supplying heat to the spacecraft surfaces.

In order to conserve fuel, these missions could also use gravity assist maneuvers to gain speed by swinging by planets like Earth and Venus. Swinging by Venus also leads to very warm environments that have to be accommodated in the thermal design because the solar flux is almost twice of that on Earth and Venus has a very high albedo.

The thermal design then gets complicated to accommodate these two very extreme requirements. In particular, passive designs (no fluid loop, heat pipes, etc.) are subject to significant design, configurational and implementation challenges to overcome these environments.

The range of environments is summarized below:

Near Earth Cruise
- Warm (benign); 1 A.U.

Venus flyby (Venus Gravity Assist)
- Very hot (Worst Case Hot) Case; ~0.7 A.U.
- 2x Sun flux from Sun (Direct) and 1.5x Sun Flux from Venus (Albedo)

Cruise to Saturn
- Very cold Worst Case Hot Case (WCC)**;** as far as ~10 A.U.
- 90-130 K sink at Saturn

III. Description of Proposed Architecture

Since electrical power is so precious for the functionality of components in the spacecraft, alternative schemes were investigated to minimize the use of electrical heat for thermal survival of the spacecraft components. The solution, which came across as very attractive, relies upon harvesting the pre-existing waste heat from all the heat dissipation that would be available from operation of electronics, instruments, etc. For example, in a Saturn mission, the various electronics would already dissipate about 200 W of heat that is simply "thrown away" to space from the spacecraft surfaces. The amount of thermal power that would be required for the safe thermal control of components within the spacecraft would also be roughly of this magnitude in deep space for this class of spacecraft. It made good sense to try to maximize harvesting of the waste heat and employing it to maintain the temperatures of all the components within their allowable limits.

The harvesting of this waste heat along with supplying it to the thermally controlled components led to the scheme described below. The proposed system utilizes a mechanically pumped single-phase fluid Heat Rejection System (HRS) loop to pick up the waste heat from components attached to this loop's tubing and then direct it to a thermal flask that has tubing attached to it. The thermal flask is cylindrically shaped and contains all systems and components in the spacecraft within it with the exception of the solar array, antennae, thrusters and various apertures of instruments, etc. to allow them an unobstructed view of space (Fig. 1). Waste heat from the heat dissipating components warms up the fluid and is carried to the flask surface and deposited on it via the fluid loop's flow. The entire flask is covered with Multi-Layered Insulation (MLI) to minimize the heat loss from the flask to allow it to remain warm. Hence the flask creates a thermal environment within

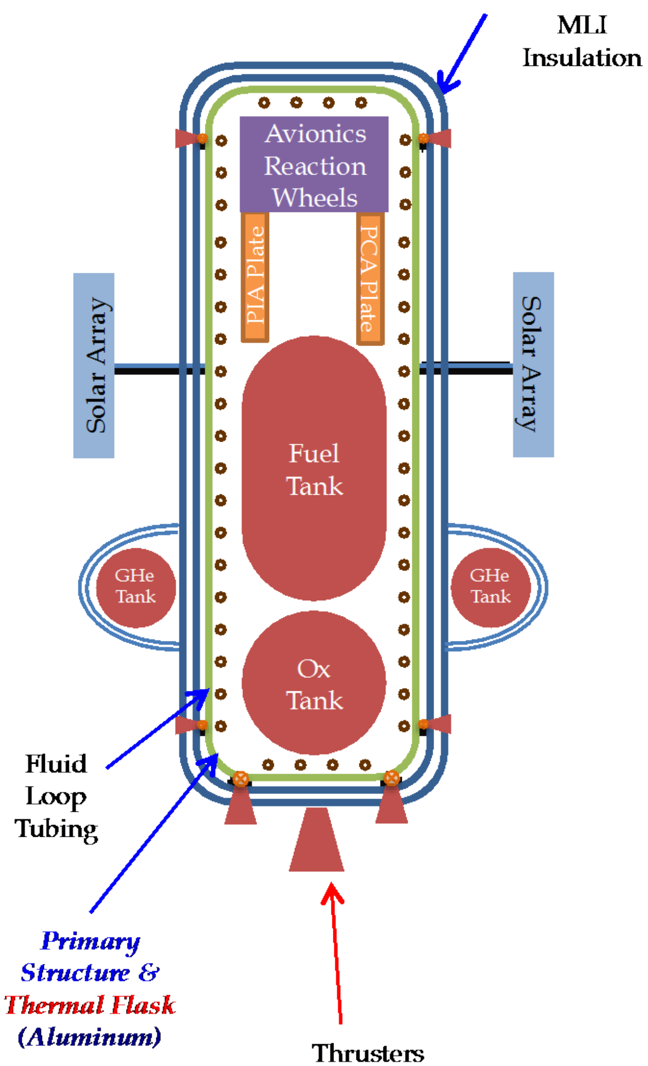

Figure 1. Schematic of Thermal Flask Architecture

which the spacecraft components reside (Fig. 2-4). The temperature of the components within the flask is then essentially the same as the temperature of the flask.

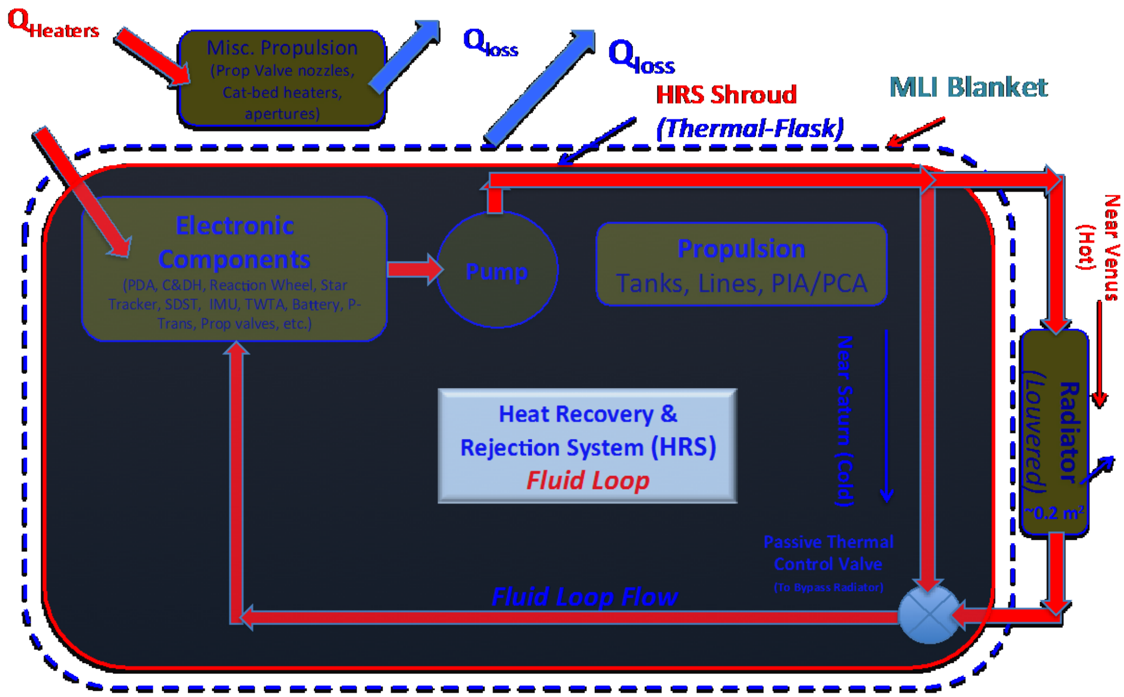

Figure 2. Fluid Loop (HRS) Schematic

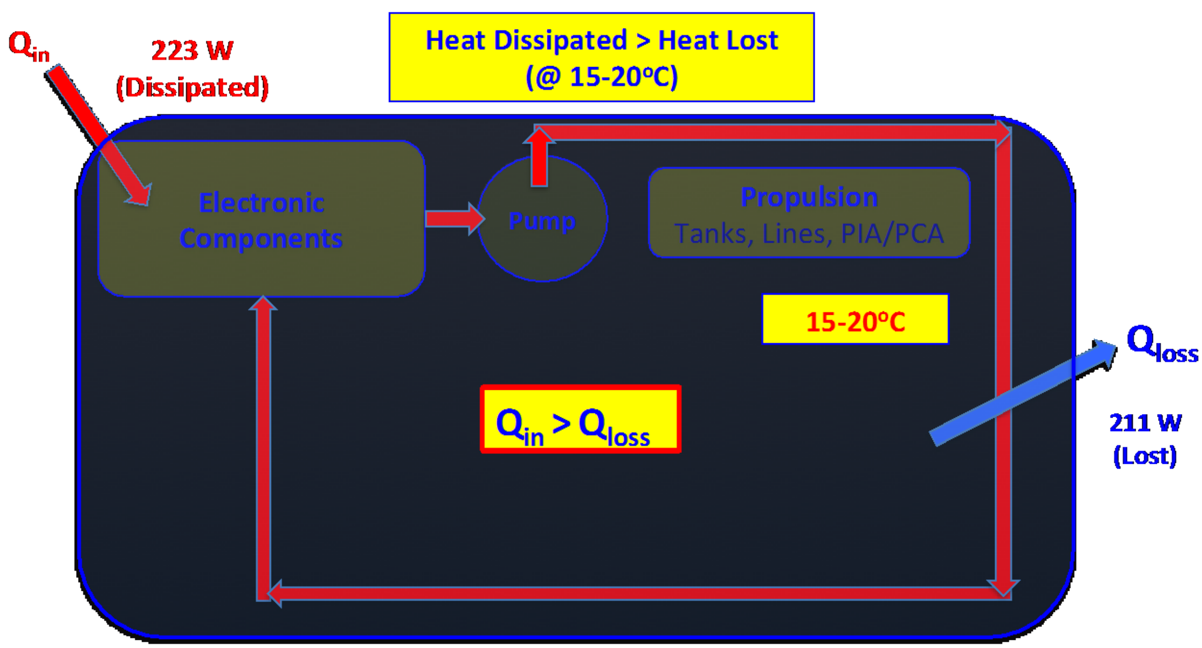

Figure 3. Example Overall Heat Balance of Flask

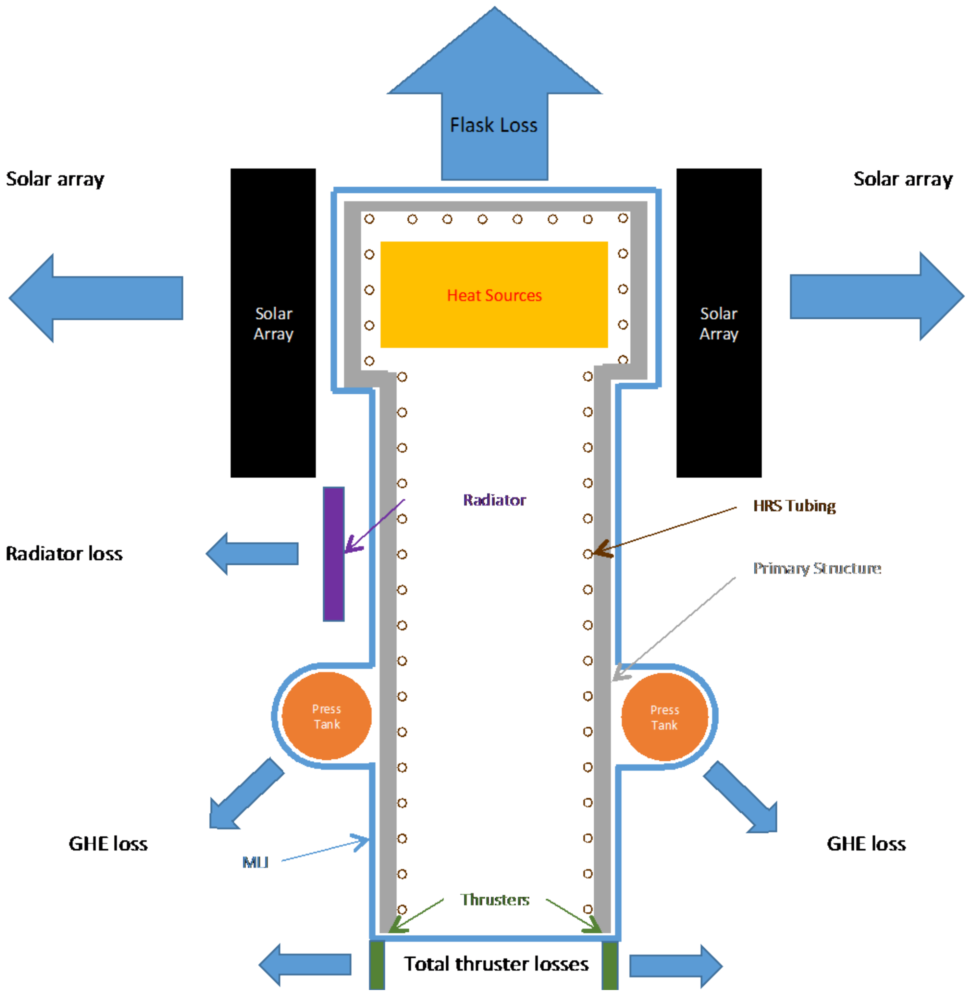

Figure 4. Various Components of Thermal Flask Heat Loss

As an example, for a deep space mission spacecraft that would have all its components fit inside a cylindrical thermal flask 1 m in diameter and 4 m long, the heat loss from the MLI insulation would be approximately 150 W to maintain the flask at ~20°C in a deep space environment. The heat input in the dissipating components (electronics, etc.) for this class of mission would be approximately 200 W. Hence the waste heat would be more than sufficient to overcome the heat loss to maintain it at ~20°C. The choice of a ~20°C temperature for the flask is primary dictated by propulsion components minimum allowable temperature limit of about 15°C.

The flask (constructed of about 1/8th inch or 3 mm aluminum shell with local structural stiffeners) would essentially serve dual functions: provide the structure to support various components within the flask as well as spread the heat coming out of the fluid loop tubing. The flask would then be almost uniform in temperature and create a very uniform environment for the components within the flask. The fluid loop would serve the function of harvesting the waste heat from the components and supplying it to the flask. This would be its function during cold conditions.

In hot conditions (near Earth of during Venus flybys) when heat from the sun or the warm planets would enter the flask via its MLI insulation, the fluid loop would automatically serve the inverse function of picking up this excess heat from the dissipating components and directing it to a small (~0.25 m^2) outboard radiator. Passive thermal control valves that rely on thermally induced volume change of a liquid or wax stored in a bellows would automatically open or close ports (in an analog fashion) to the radiator (hot conditions) or to the flask (cold conditions) to tightly control the temperature of the flask within a narrow range (20°C) to ensure that the components within the flask remain in their desirable range. Hence this architecture is very flexible and automatically accommodates both cold and hot conditions passively.

The only power required to thermally operate this thermal control system is for running the fluid loop pump. For missions of this type, a typical input power for a pump would be ~10 W. Additionally, all of the pump operational power is harvested as waste heat and contributes to the heat inserted into the thermal flask.

IV. Novelty of Architecture

This is a novel concept that enables missions because it almost eliminates the need for thermal control power for a large size spacecraft that is designed for deep space. Without this concept, the missions may be prohibitively heavy and expensive to provide large amount of power for thermal control in the extremely cold environments that such missions would encounter. This concept is completely automatic (requiring no on-board or ground directed control of its function). It also is quite simple in its implementation because all the spacecraft components are placed inside an aluminum shell with standard MLI insulation covering this shell. This work is radically different from previous missions that utilized solar arrays for power (like JUNO) where electrical heaters on the order of 200 W were utilized to thermally control the spacecraft components. Missions that had Multi-Mission Radioisotope Generator (MMRTG) nuclear power sources (e.g., Cassini) did not encounter this problem because of ample waste heat available from MMRTGs. Solar missions do not have that luxury; hence the need to come up with novel schemes like the one proposed here to enable them.

V. Heritage & Technology Assumptions

The fluid loops utilized to make this architecture work have a long and extremely successful heritage from JPL's Mars missions starting from Mars Pathfinder (MPF) followed by the two Mars Exploration Rovers (MERs, Opportunity and Spirit) and the recent Mars Science Laboratory (MSL) Curiosity Rover mission. In particular, the Curiosity Rover utilized waste heat from the MMRTG to thermally control the rover on Mars. HRS (fluid loop) technology is well established (TRL-9) for last ~20 years for application in extremely successful spaceflight projects MPF, MER (2x), MSL (2x). Number of pumps inside an assembly can be further increased for long life missions to increase reliability (MSL Cruise HRS pump assembly has 3 pumps, 2 redundant, only one on at any time) – this would lead to an insignificant mass increase with no change in power. MLI blanket properties have been well characterized for several decades on several flight projects - for large and small size blankets.

VI. Expected Performance for Example Applications

WCC Conditions

- Completely cocoon the entire spacecraft and its components by a thin metal shroud (Thermal Flask) which has HRS fluid loop tubing bonded to it
    - Except solar array, thrusters, antenna, star-tracker/Instrument apertures
- Thermally couple all internal heat dissipating components to HRS fluid loop
    - Except cat-bed heaters and thrusters
- Supply all this picked up heat to external shroud via HRS fluid flow tubing
- External shroud insulated by MLI blankets to minimize heat losses
- Approximate heat balance shows that to maintain shroud at 15-20ºC the heat dissipation exceeds heat losses via blankets – hence this temperature can be maintained in cold conditions
- No survival heater power needed above and beyond that dissipated by spacecraft components for their own functionality
- This essentially zeroes out thermal power burden on s/c during cold conditions
- Completely cocoon the entire spacecraft and its components by a thin metal shroud (Thermal Flask)

WCH Conditions

- During Venus flyby, almost 2x solar flux from Sun and 1.5x from Venus albedo
- Spacecraft MLI blankets (insulation) surrounding shroud minimize net heat insertion in spacecraft
- Parasitic heat inserted into spacecraft via blankets picked up by HRS fluid (along with spacecraft dissipation) and automatically directed to radiator mounted external to blankets
- Attitude Control System (ACS) ensures that radiator does not face Sun or Venus during Venus flyby to allow for safe heat rejection from radiator
- Relatively small radiator (~0.25 m^2) safely rejects heat dissipated within spacecraft and that coming in via blanket from Sun/Venus to maintain spacecraft temperatures within allowable flight limits (< 30ºC)

- Louvers on radiator automatically close or open to minimize heat loss from radiator in cold conditions and maximize heat rejection in hot conditions
- Passive HRS thermal control valve (MSL, MER, MPF heritage) automatically directs HRS flow to radiator or bypasses it to maintain safe operating temperatures in spacecraft in all conditions

VII. Other Options Investigates

Due to severe power constraints, only options that harvested waste heat from components to service other components (like propulsion) were considered

A. Options that relied on significant use of electrical heat were deemed unattractive due to reasons cited before
B. Fluid loop (HRS) based thermal bus was the most logical option for thermal control for cold and hot conditions
C. Heat pipes or loop heat pipes were not deemed suitable from testing, integration, ATLO, and modularity reasons as well as a multi component serving autonomous thermal bus

Within the HRS approach, an earlier option considered was to employ local HRS plate clamshells to surround propulsion system to maintain their temps.

A. Smaller surface area/mass of plates
B. Would need thermal isolation of these components from the cold structure, which would be difficult to satisfy simultaneous thermal and structural needs
C. Heat losses via conduction and MLI around clamshells would be similar to those from an all enclosing thermal flask
D. Thermal flask approach that encloses the entire spacecraft except solar arrays, Helium tanks, thrusters and some small components was chosen as the baseline
    - Integrated structure and thermal flask system (~3 mm or 1/8” thick aluminum shell with local stiffening features) - reasonable total mass
E. All components inside flask are thermally and mechanically coupled to integrated structure/flask - no need for thermal isolation based complexity

VIII. Rules of Thumb for Sizing & Performance

The heat loss from the flask should be proportional to the external surface area of the flask for a given controlled temperature as shown below:

$$Q = \sigma \varepsilon A (T_{flask}^4 - T_{\sin k}^4)$$

The total heat loss would then be simply the harvested heat plus any heat dissipation in the locations where the heat is supplied (e.g. propulsion module).

IX. Applicability & Limitation

Obviously this architecture is most well suited for missions that are very power-starved and which could encounter very cold environments, like the ones that go to outer planets. The key assumption is that significant amount of dissipation is present from which the waste heat is harvested. This architecture relies on enclosing all (or most) thermally controlled components within the flask. If the configurational constraints present a challenge to this assumption then a single flask concept may not work as well and it would have to be broken into mini-flask-regions, which would make the configuration more complicated to implement. There will inevitably be some components that will have to protrude out of the flask (e.g., solar array and supporting structure, thrusters, apertures of star scanner, sun sensor, instruments, radiators, etc.). There would be some thermal-mechanical design challenges to minimize these parasitics to minimize the waste heat recovery required to overcome all the heat losses.

X. Possible Improvements

Since the primary heat loss from the flask and correspondingly the required harvested heat is directly proportional to the MLI blanket effective emissivity, reduction of this emissivity would directly benefit the thermal design’s efficiency. The large blanket surfaces that would cocoon the flask would be amenable to substantial improvements because the edge seam losses can be minimized (the edge seam losses are the largest contributors of MLI emissivity increase due to local compaction of the blanket layers). Additionally, two mechanically separated blankets, which are not in contact with each other, could greatly reduce the heat loss. Maximization of packing of components within the flask envelope would also minimize the heat loss because the external surface area of the flask is what determines the total loss, so anything that can fit inside would already have its heat loss accounted for by loss from the flask external area. Minimizations of protrusions out of the flask would also be beneficial since those external components would tend to run cold and would increase the parasitics losses from the controlled warm surfaces of the flask.

XI. Conclusion

The thermal flask architecture presented in this paper describes an attractive, practical, simple and power efficient approach for the thermal design of future deep space solar power based missions that are extremely power starved due to the very solar flux available at large distances from the Sun. It relies on harvesting waste heat from heat dissipating components via mechanically pumped fluid loop tubing and then, via the same tubing, directing it to a "thermal flask" that contains all the essential thermally controlled spacecraft components within it. The thermal flask serves as the spacecraft structure as well as a thermal cocoon to create a warm environment that the spacecraft components are slaved to. The simplicity, robustness and low thermal power needs of this approach, along with the ease in its implementation and testing makes it a very enabling architecture for future deep space missions.

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