A SunCam online continuing education course

Spacecraft Subsystems Part 3 – Fundamentals of Thermal Control

by

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1. Introduction

Spacecraft are man-made machines that operate in space. An orbiting spacecraft is normally referred to as a satellite, although it is manmade (aka "artificial") as opposed to a natural satellite like our moon. A spacecraft is typically subdivided into two major parts, the payload and the bus. Where the mission can be defined as the purpose of the spacecraft and is usually identified as the payload part of the spacecraft (e.g. scientific instruments, communications). The thermal control subsystem and other subsystems (e.g. attitude control, electrical power, structures) are part of the bus. With the primary goal of achieving a successful mission, most bus design constraints focus on maximizing the effectiveness of its payload [ref. 1].

Thermal Control in the context of this course refers to the ability of a spacecraft to maintain its temperatures – similar to your home heating system. You may have a temperature range that you feel most comfortable, say 70°F to 72°F, most spacecraft components also have a range which must be maintained in order to protect the components reliability. If a component fails sooner than expected, from excessive overheating for example, the spacecraft's mission could also be shortened.

Note: All figures in this course have no scale.

Apollo 13

Thermal control issues can have serious impacts to the spacecraft's mission. Knowledge of both external and internal heat sources must be included in design considerations in order to maintain thermal balance. One well known example of this is in the Apollo 13 mission, where an internal heat source was lost because much of the equipment in the command module had to be turned off to conserve electrical power during its return to earth. The following is a sequence of events intended to include those attributing to the oxygen tank explosion.

Apollo 13 Sequence of Events:

Pre-Flight

1. O_2 tank 2 was damaged during removal for modification from Apollo 10
2. Design change (voltage) to oxygen tank heaters
3. Did not change thermostatic switches to accommodate for the voltage change
4. Attempt to normalize liquid O_2 level in O_2 tank 2 to 50% did not work using gaseous O_2
5. Normalized liquid O_2 level in tank 2 to 50% using heaters – an atypical method
6. O_2 tank 2 fan motor wiring insulation damaged by excess heater on time

In-Flight

7. Astronaut stirs O_2 tanks in routine housekeeping task
8. Current applied to O_2 tank 2 fan motor
9. Electrical short circuit occurs in fan motor wiring with arcing
10. O_2 tank 2 exploded

See the following figure for oxygen tank locations located in the service module.

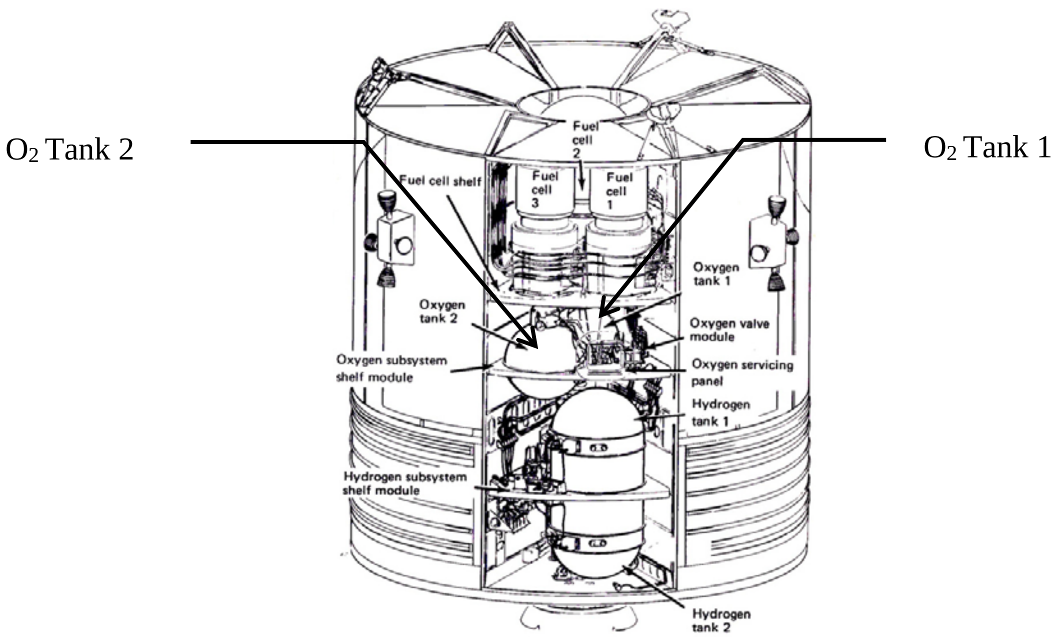

Figure 1.1: Apollo Oxygen Tanks
[Reprint from source: NASA @ nasa.gov]

The root cause of the explosion was due to thermostatic switches not being modified for the voltage change to oxygen tank heaters from 28VDC to 65VDC (sequence 3). This allowed the heaters to be on too long during normalization of liquid oxygen levels (sequence 5), thus damaging wiring insulation near the heaters by exposing the wiring to temperatures of 1000°F (sequence 6). Tank 1 was normalized successfully (sequence 4) and therefore it's fan wiring was not exposed to excessive heater usage.

Orbit Types

The spacecraft operating environment largely depends on its orbit type, which is primarily driven by its intended mission. The three common earth orbiting satellite types are geosynchronous orbit (GEO), highly elliptical orbit (HEO), and low earth orbit (LEO) as seen in the following figure [ref. 1].

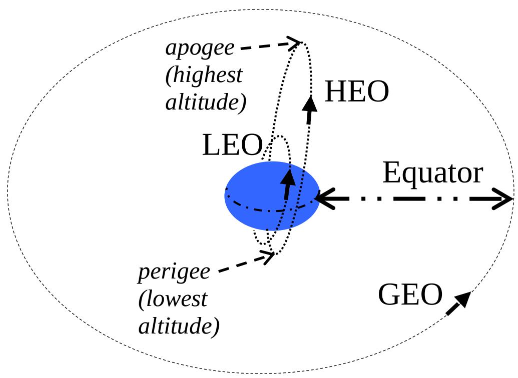

Figure 1.2: Common Earth Orbit Types
[Reprint from source: ref. 1]

LEO satellites have an altitude of less than one thousand miles. Satellites in HEO can have a wide ranges of altitudes for perigee (lowest altitude) to apogee (highest altitude).

GEO satellites, orbiting about the equator at some small angle (inclination angle), have an altitude of ≈ 23,000 miles above earth. At this altitude, satellites have the same period of rotation as the earth, appearing fixed relative to earth. Because of this, these satellites are most commonly used for communications purposes (e.g. television) since there is no need for the receive dish on earth to track the satellite [ref. 1].

External Heat Sources

The environment where spacecraft can operate is very hostile in general, and for thermal control, external heat sources must be known in order to maintain thermal balance. This section describes those external heat sources, with other environmental conditions described in section 2 (Surfaces) that are thermally degrading to spacecraft.

The primary sources of externally generated heat for earth orbiting spacecraft are direct sun, reflected sun from earth, and infrared (IR) energy emitted from earth. For interplanetary space missions, the primary source of external thermal heat is direct sun, secondary is planetary reflected and IR when passing near a planet. This means that most of the time, for interplanetary missions, the heat source intensity will be a function of the spacecraft distance to the sun. Therefore, missions to distant planets (e.g. Jupiter) and further incur very low solar radiances creating an extremely cold spacecraft environment.

For direct sun, the amount of time the sun is "seen" per orbit by the spacecraft also affects its temperature control abilities, especially when extended for long periods of time. In general, as can be expected, the presence of sun is the hot period and the absence of sun is the cold period as can be seen in the following figure.

Each of the orbital types have a cyclical characteristic where the spacecraft goes from light (i.e. the sun is present) to dark (i.e. cannot "see" the sun). Sometimes, the spacecraft may be in sunlight or darkness for extended periods of time; worst cases are known as 100% sun or minimum sun respectively. Percent sun refers to the amount of sun present during each orbit. For example, in 100% sun, the spacecraft is exposed to the sun the entire duration of each orbit; where the sun vector is normal (orthogonal) to the orbital plane, and Sat #2 (in the following figure) will "feel" this as its temperatures rise. Occurring over several days, this has a warming effect on a spacecraft as expected. This is a period where you would see high use of radiators and/or louvers working to cool the spacecraft.

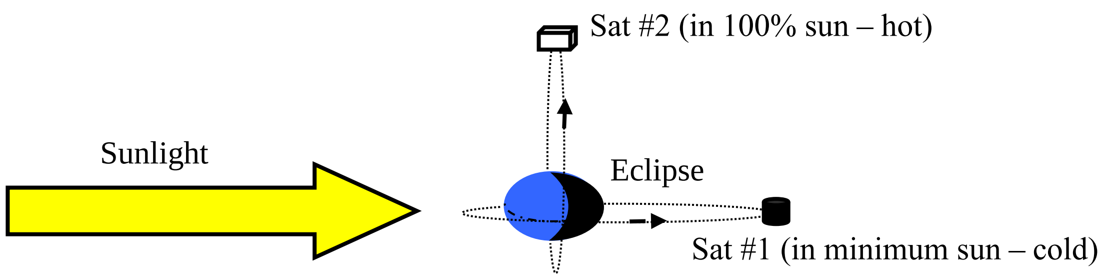

Figure 1.3: Sun vs Eclipse
[Reprint from source: ref. 2]

Minimum sun refers to the least amount of time per orbit the spacecraft will "see" sunlight. For most orbits, the spacecraft will experience minimum sun biannually; when the sun vector is parallel with the orbital plane, and Sat #1 (in the previous figure) will "feel" this as its temperatures drop. Also occurring over several days, this has a cooling effect on a spacecraft as expected. This is a period where you would likely see electric heaters cycling on and off working to warm the spacecraft.

Heat Transfer

Just like on earth, heat transfer for a spacecraft is also governed by the following three fundamental ways for the transference of heat energy: convection, conduction, or radiation. Unlike earth however, air cannot be used as a medium to transfer heat in the zero gravity, vacuum of space. On earth, denser (cooler) air molecules push the lighter (warmer) air molecules up. In other words, since cooler air molecules are denser they fall due to gravity – even though you may have heard the term coined "heat rises". This is an effect from the cooler (heavier) air molecules falling and pushing the heated (lighter) molecules up.

Since a spacecraft cannot just open a window or door to let heat escape, it must employ other methods to transfer heat in order to cool. For example, the payload bay doors of the shuttle (as shown on the cover) open to expose the silver radiators which dissipate heat into space.

For thermal control, a spacecraft will utilize multiple methods as presented in sections 2 and 3 of this course to maintain its thermal balance, passive and active respectively. Where active methods consist of an element that changes, passive methods do not. The following table includes the primary heat energy transfer method used by each.

Table 1.1: Thermal Control Methods - Heat Energy Transfer

Control Method	Heat Energy Transfer Method		
	Convection	Conduction	Radiation
Surfaces			✓
Radiators			✓
Multilayer Insulation Blanket*			
Doublers		✓	
Heat Pipes	✓		
Pumped Fluid Loops	✓		
Electric Heaters			✓
Radioisotope Heater Units			✓
Thermoelectric Coolers		✓	
Louvers			✓

*Ideally, intended to block heat, but there could be small amounts of transfer by conduction and/or radiation.

2. Passive Thermal Control

Passive thermal control is usually characterized by the fact that they don't consist of any moving fluids or parts. But their main attractiveness for spacecraft use is that they do not require power to operate and be effective. Ideally, you want all thermal control devices to be passive in order to eliminate the need to utilize part of the electrical subsystem's power budget. This is because the primary goal of the electrical power subsystem is to maximize the amount of power to the payload. This section describes those methods that passively transfer heat for thermal control of spacecraft.

Surfaces

Thermal control surfaces can be any surface that is used for spacecraft thermal control to include coatings, paints, and finishes. Most internal and external spacecraft components have a thermal control surface to help control its emittance and/or absorptance properties. The performance of these surfaces are characterized by the ratio of absorptivity to emissivity, α / ε, which are characteristics of heat transfer by radiation. For example, white paint has a low ratio and therefore is used as a heat emitter. Ratios greater than 1.0, like blank paint, will get hot when exposed to sunlight.

Black and white paints are the most common color. Most paints have a high emittance, with varying absorptance and electrical conductivity properties. Black paints have the following characteristic: $\alpha \approx \varepsilon \approx 1.0$, which is close to thermal equilibrium. This means that most of the heat that is absorbed is then emitted. This makes black paint an effectively neutral color, thermally. Because of this and its performance ratio of >1.0 as previously discussed, most internal spacecraft components are painted black. Most external spacecraft surfaces are painted white to minimize solar energy absorptance.

Thermal control surfaces are also a key part of other thermal control methods to be described in this section, which include multi-layer insulation (MLI) blankets and radiators. Aluminized Kapton is commonly used for the MLI blanket external layer. Surfaces with a performance ratio of less than 0.4, like white paints or optical solar reflectors (OSRs), make them effective radiators. OSRs include quartz mirrors, silvered or aluminized Teflon.

Metallic surfaces are have low absorptance and emittance, such as vapor-deposited aluminum (VDA) and bare aluminum. One use for VDA is as a separator for an MLI blanket (to be discussed later in this section).

The left side of the following table includes ranges based on common thermal control surfaces for the two key thermal control properties: absorptance beginning-of-life (BOL) and emittance; and also includes some common uses. Because the absorptance property of surfaces are prone to degradation due to environmental effects, increasing its absorptivity over time, it is important to distinguish between BOL and end-of-life (EOL) absorptance.

Table 2.1: Thermal Control Surfaces - Properties & Degradation

Surface	Absorptance BOL α	Emittance ε	Common Use	Increases Absorptance from Space Environment (Thermal Degradation)			
				UV Radiation	Atomic Oxygen	Charged Particles	Spacecraft Contamination
White Paints	0.16 – 0.28	0.85 – 0.92	Radiator	✓ (LEO)		✓(GEO)	✓
Black Paints	0.92 – 0.98	0.84 – 0.89	Internal Components				
Aluminized Kapton	0.34 – 0.46	0.55 – 0.86	MLI Blanket		✓(LEO)	✓ (GEO)	✓
Optical Solar Reflectors	0.05 – 0.16	0.66 – 0.80	Radiator		✓(LEO)	✓ (GEO)	✓
Metallic	0.03 – 0.30	0.03 – 0.12	MLI Blanket				✓

The right side of the previous table includes common environmentally degrading effects on thermal control surfaces: UV radiation, atomic oxygen, charged particles (e.g. electrons), and spacecraft contamination (e.g. due to attitude control jets and/or propulsion out gassing). Notice that some of these surfaces are degraded by multiple environmental effects, and can depend on the type of orbit (e.g. LEO, GEO). Observe black paints generally have no thermal degradation, this is primarily due to its higher absorptance and that most of the black painted surfaces are internal to the spacecraft as previously mentioned.

The two Van Allen radiation belt ranges above the earth are 1,000-8,000 miles for the inner belt and 12,000-25,000 miles for the outer belt as can be seen as two donuts in the following figure. Because of this, satellites in GEO (≈ 23,000 miles) are impacted more by the dominant charged particle electrons in the outer belt. LEO (<1,000 miles) satellites would be minimally impacted by the dominant charged particle protons in the inner belt. If mission requirements dictate that an orbit must cover these altitudes, a good choice might be HEO. In HEO, the satellite's dwell time at perigee is much shorter than at apogee. This is because the satellite accelerates as it approaches earth (towards perigee); therefore, the time spent in this damaging region could be limited depending on the altitudes chosen for apogee and perigee.

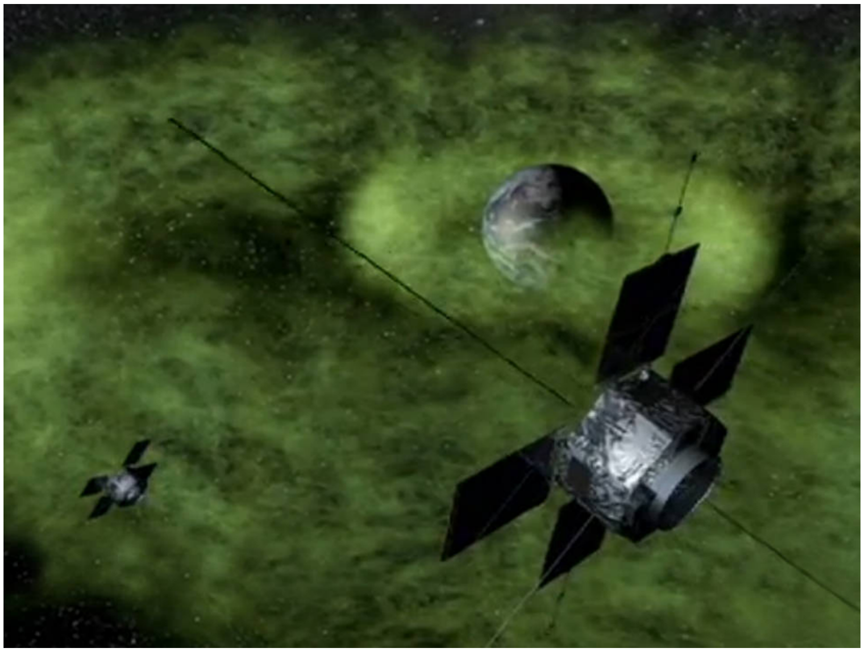

Figure 2.1: Van Allen Radiation Belts
[Reprint from source: NASA @ nasa.gov]

To further explore the Van Allen radiation belts, two probes (Van Allen A and B), formerly called Radiation Belt Storm Probes, are shown in this artist's rendering in the previous figure. Their mission is to collect radiation measurements while orbiting through the radiation belts. These probes are radiation hardened for this mission, in HEO with an apogee and perigee of about 23,000 and 373 miles respectively [ref. 2]. Lastly, in the previous table, notice that satellites in LEO are impacted more by solar UV radiation and atomic oxygen effects.

Atomic oxygen (O), remember we breath (O_2), is damaging to LEO satellites because of its abundance at low altitudes and that it is a highly reactive chemical, like hydrogen (H). Single oxygen atoms (O) react to produce CO, CO_2, and H_2O which can wreak havoc on thermal control surfaces, eroding hydrocarbon based thermal control materials.

Another environmental concern that could create a thermal control problem is due to the amount of man-made space debris (aka space junk) as shown in the following figure. This debris, for example, includes old nonfunctional satellites, satellite fragments, or launch related expendables. Primarily impacting satellite in LEO, but also a concern in order for a spacecraft to attain GEO. To attain GEO, a spacecraft must first enter an intermediary transfer orbit, and go through this "mine field" of orbiting debris. For HEO satellites, as previously discussed, this debris field would be more problematic as the spacecraft approaches earth towards its perigee. Generally, overall thermal control damage is proportional to the size and number of the objects damaging a surface. For example, if many 5 mm in diameter or less struck a surface, the overall thermal control damage would likely be negligible. However, the more a thermal control surface area is reduced by either an increase in hits and/or size of the debris, the more its effectiveness is degraded.

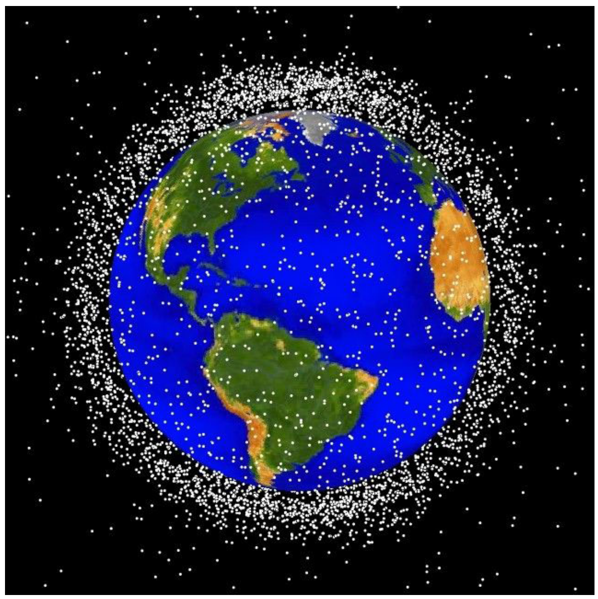

Figure 2.2: Space Debris
[Reprint from source: NASA @ nasa.gov]

Finally, electrical conductivity properties of surfaces used for thermal control must also be considered. As electrical charge increases on spacecraft, electrostatic discharge through a thermal control surface can occur. If this happens, internal spacecraft components are susceptible to damage. Therefore, whether the surface is a conductor or an insulator must be known. Conductive surfaces can provide a safe discharge path to the structure, however, insulating surfaces need to be grounded.

Radiators

A radiator can utilizes an existing part of the spacecraft structure that is exposed to space for heat dissipation or it can be an individual component of the thermal control system that is designed to be a radiator. In both cases, internal excess heat is directed to the radiator for rejection into space as infrared (IR) radiation, thereby cooling the spacecraft.

The space shuttle's radiator was specifically designed as such. The payload bay doors were opened in orbit to expose the radiator panels for heat rejection. This can be seen in the following figure as the space shuttle Discovery approaches the International Space Station for docking. The forward 30ft two-section panels of radiators on both sides were deployable, lifted upwards above the doors, to expose both sides. This allowed for increased heat rejection when necessary.

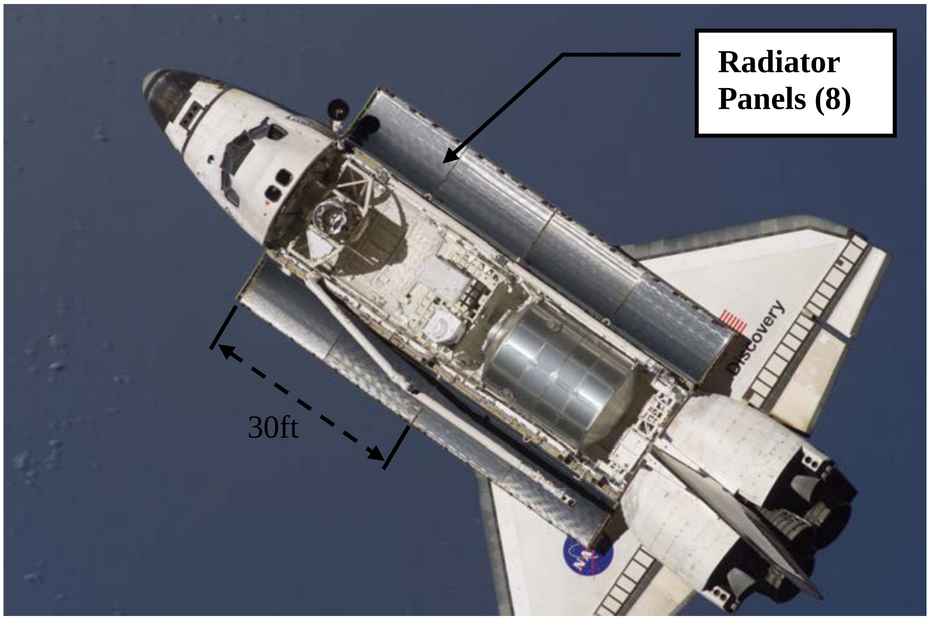

Figure 2.3: Space Shuttle Radiators
[Reprint from source: NASA @ nasa.gov]

Each space shuttle radiator panel consisted of 68 parallel flow tubes, coated with silver Teflon with an emissivity (ε) of 0.76 and an absorptivity (α) of 0.11. Since these radiator panels are comprised of flow tubes, which are part of a pumped fluid loop (to be covered in Section 3), this makes the overall system an active thermal control one.

An important part of any radiator is its surface coating as previously described (*Surfaces*). Reemphasizing here, this coating should have a high emissivity (e.g. white or OSRs) to increase heat dissipation efficiency. In addition, it should have a low absorptivity and be as stable as possible. Stability refers to its resistance to thermal degradation due to the space environment as previously described.

Multilayer Insulation Blankets

Multilayer insulation (MLI) blankets can typically be seen covering all spacecraft. These blankets are used to form a thermal barrier to prevent heat transfer. They also provide protection from external environmental damage and are an effective insulator for propellant tanks and lines. These blankets consist of an external layer, multiple insulating layers, and an internal layer.

In addition to the surface properties as previously discussed in the *Surfaces* section, the external layer material of an MLI blanket should be chosen based on its solar temperature response, to ensure that it does not get too hot when in sun, and have a medium α/ε ratio. The materials commonly used as the external layer are Kapton, Aluminized Teflon, or Beta cloth. Another consideration should be the effects of excessive cycling between cold and hot each orbit. This can be damaging to an MLI blanket exposed on a satellite in LEO.

One example is the spacecraft Hubble's multilayer insulation blankets damage due to the extreme temperature ranges over very short periods of time. Due to its low orbit – it "feels" extreme temperatures many times a day. This damage can be seen on bay 8 in the following figure. From left to right, are bays 6, 7, 8, and 9, with new outer blanket layers (NOBLs) on Bays 6 and 9. These two NOBLs included stainless steel sheets to provide additional thermal protection. The damaged insulation on bay 8 was replaced during service mission 4 in 2009. Stainless steel sheets were installed on some of the bays for additional thermal control.

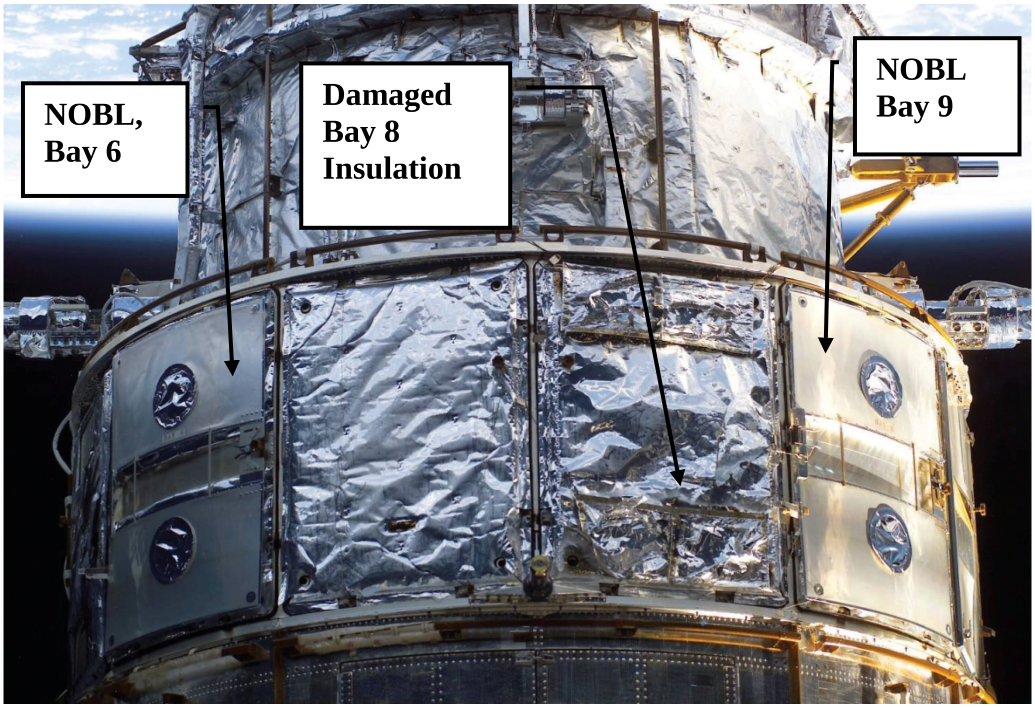

Figure 2.4: Spacecraft Hubble - Thermal Insulation
[Reprint from source: NASA @ nasa.gov]

In between the external and internal layer is the insulating layer, which contains multiple parallel layers of material with low-emissivity surfaces in order to minimize transfer of thermal radiation. To achieve 100% thermal blockage (or near it), multiple layers of this insulating material is needed to make up the middle, the insulating layer of the blanket. This number of layers is normally a tradeoff between insulating performance and mass.

Mass is another important consideration besides power consumption. Not necessarily a concern in zero gravity space, mass can add considerable cost to the launch. Therefore, mass is a key factor in calculating the cost of any spacecraft launch. The greater the mass of the spacecraft on the rocket, the more fuel is needed to launch the spacecraft into space. Hence, mass (weight) minimization is a goal that applies to all subsystems (e.g. thermal control) and subsystem components (e.g. MLI blanket). Since the MLI is protected between the external and internal layers, each insulating sheet can made very thin to save weight.

The insulator material is typically Mylar or Kapton and commonly consists of 15-20 layers, each between 0.0051–0.127 mm thick. Where performance can be measured by emissivity, an exponential function, the following figure shows the cumulative effects of having multiple layers, where each layer allows less heat to transfer (in both directions) than the one below it. Since the number of layers vs emissivity is an exponential function, effectiveness peaks at about 20 layers. Therefore, it would not be cost effective or practical to add more layers. Insulating performance is also dependant on the effectiveness of the separation method used between each insulator.

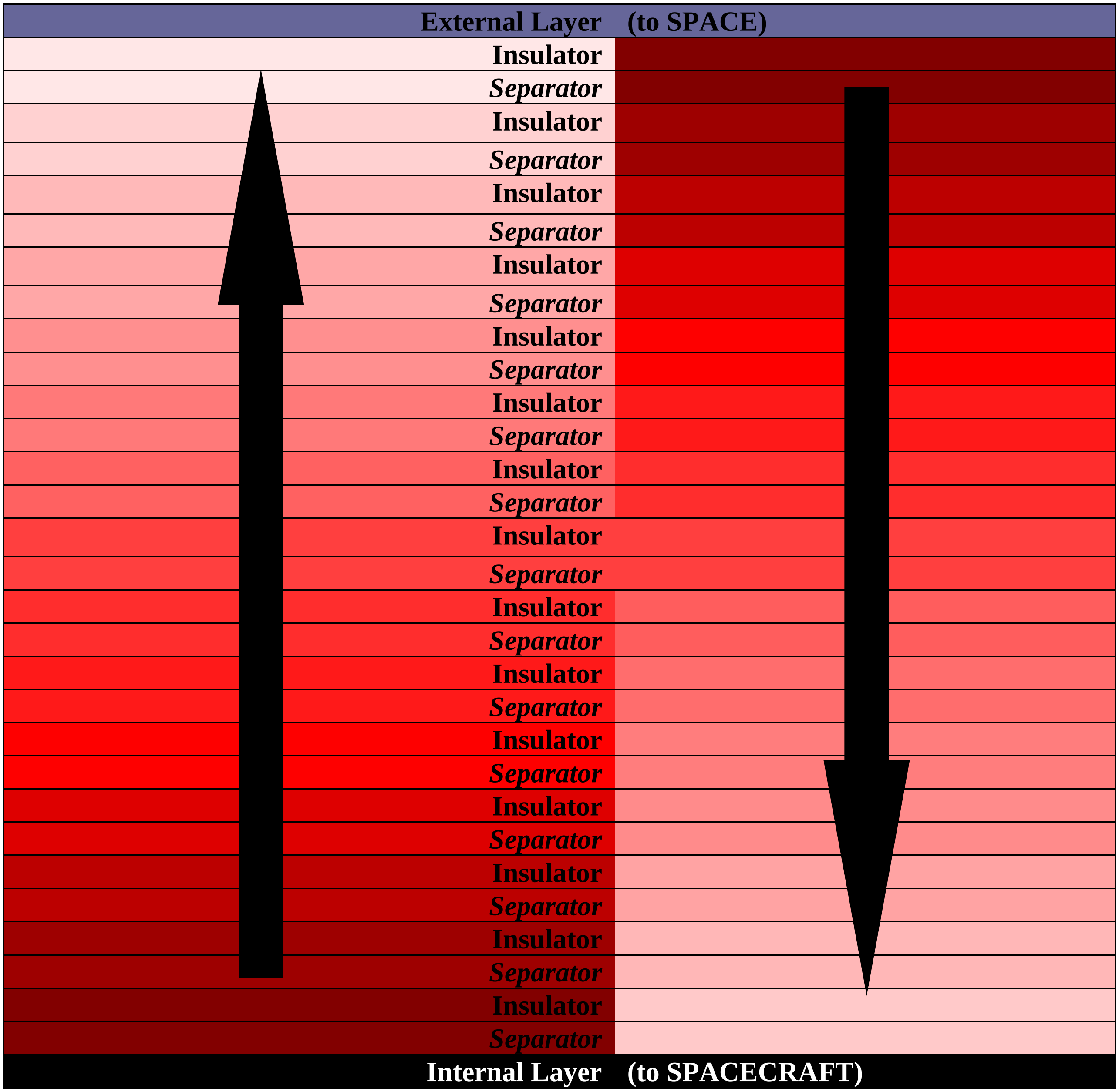

Figure 2.5: MLI Blanket Heat Transfer Effects

The goal of the separator is to minimize conduction between each insulator. To achieve this, the number of contact points between each insulating material layer must be minimized in order to reduce the area that can conduct heat. One method, using Mylar as the insulator, is to emboss each Mylar insulating sheet and coat one side with a vacuum-deposited aluminum (VDA) finish with low emittance and absorptance – this combination acts as a low conductivity separator. This allows the Mylar to act as both a insulator and a separator. Otherwise, you would need another material between each insulating sheet to provide the separation, such as fine netting (Dacron, Nomex, or silk) to minimize contact points.

Lastly, the internal layer is primarily designed to protect the thin insulating layers from damage during spacecraft assembly and is made of either Kapton or Dacron sailcloth. The side facing the spacecraft components is usually not aluminized in order to reduce the risk of electrically shorting the hardware in close proximity to the MLI blanket.

One final important function of the MLI blanket is that it must vent. This is, trapped air (from prelaunch) within the blanket must have a way to escape upon entering the vacuum of space.
To achieve this venting, perforated layers can be used to vent through the external or internal layer. Another approach is to use unperforated layers and vents around the edges. Venting is also important to expel any contamination that might be present.

Doublers

For those electrical components that dissipate high power per unit of surface area, thermal doublers can be used to help minimize the amount of heat transferred directly from a component's baseplate to the spacecraft mounting plate. Thermal doublers (aka spreaders) work by conducting heat laterally from the hot regions before transfer to the spacecraft mounting plate as depicted in the following figure.

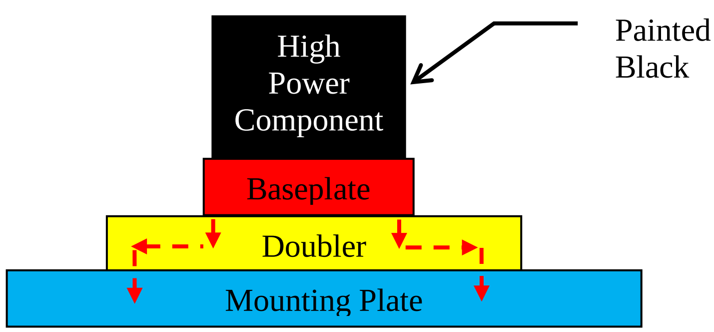

Figure 2.6: Thermal Doubler

The spacecraft mounting plate is typically a face sheet attached to an aluminum honeycomb panel. A thermal doubler in between the mounting plate and a high power component's baseplate adds another layer to the structure (mounting plate) in order to increase conductivity, resulting in decreasing the temperature rise "felt" by the mounting plate. Doublers are usually custom fabricated to cool a hot spot, so there is no standard size or shape. Also, they are made of the same or thermally compatible materials as the mounting and base plates.

3. Active Thermal Control

As previously mentioned, ideally, you want all thermal control devices to be passive in order to maximize electrical power to the payload. However, this is not practical due to the extreme cold/hot environments "felt" by the spacecraft. Also, for some high power spacecraft components, passive elements alone cannot continuously maintain its operating temperature range. This is where active thermal control is typically employed, to cover those situations where passive thermal control is inadequate.

With the exception of heat pipes and radioisotope heater units (RHUs), all active thermal control methods in this section require electrical power to operate. To improve reliability, most active thermal control systems are redundant in case of failure (e.g. Heater 1A, Heater 1B).

Many active control methods also require continuous monitoring of spacecraft temperatures. To help achieve this, most spacecraft are covered from top to bottom with temperature measuring devices called thermistors. A thermistor is an electrical component that changes its resistance as its temperature changes. Since thermistor resistance is a function of temperature, the local temperature can be measured where the thermistor is located. These temperature measurements can then be used as inputs to an active control system and/or used for thermal trend analysis by an engineer.

Heat Pipes

Heat pipes use a liquid-gas phase change to efficiently transfer heat from one location to another over relatively long distances. Most consist of the following components: evaporator, tube, working fluid, capillary wick structure, and a condenser. Heat pipes function by adding heat to one end of the tube at the evaporator, changing the working fluid from a liquid to a gas, and then removing the heat transported to the other end by the condenser, changing the working fluid back into a liquid. The phase change cycle is summarized by the following sequence:

<u>Phase Change Cycle - Heat Pipes</u>:
1. Heat added at evaporator end
2. Liquid (working fluid) in the wick is vaporized into gas by the evaporator
3. Gas transfers the heat through the tube (attracted by lower pressure) to the condenser end

4. Gas becomes a liquid (condenses) after contacting the cooler condenser
5. Heat is removed at the condenser end
6. Liquid (working fluid) is transferred back to the evaporator end by capillary action

This cycle is repeated continuously.

Using this closed two phase system for thermal control, large amounts of heat can be transferred by convection through the sealed heat pipe tube between two end locations. The operating temperature range to be "felt" by the pipe determines the working fluid to be used which must also be compatible with the material used for the tube. An aluminum tube with an axial groove wick structure, using ammonia as the working fluid has been commonly used for heat pipes.

Pumped Fluid Loops

Like heat pipes, pumped fluid loops (PFLs) can also be used to transfer large amounts of heat, to provide heat transfer cooling by forced convection. The cooling concept is similar to how the coolant is used in your car to cool the engine, where antifreeze is used as the coolant, cooling the engine as it passes through it then releasing heat as it flows through the radiator.

PFLs consist of a the following basic components:

- mechanical pump
- tubing
- working fluid (coolant)

The working fluid (e.g. ammonia, water) in the tubing absorbs thermal energy at a heat source and transfers it by mechanical pump to a heat sink. PFLs often use a spacecraft radiator, as previously discussed for the space shuttle, as the heat sink to dissipate the heat into space. The cooled working fluid returns back to the heat source where the process is repeated.

Electric Heaters

Electric heaters are commonly used to warm-up components to their minimum operating temperature prior to power on and also to provide an instant heat source during spacecraft cold periods. Spacecraft heaters function using the same concept as snow or ice is melted from your auto's rear window defroster/defogger.

Electric heaters aboard spacecraft consist of an electrically resistive element between two insulating layers (e.g. Kapton). When heat is needed as measured by a thermostat or an onboard computer using thermistor inputs, a voltage is applied to induce a current flow through the resistive element that produces radiant heat. If necessary, commands can also be transmitted by the ground system to turn a heater on or off. These heaters are usually available in standard rectangular sizes, but can also be customized to fit around essentially any spacecraft component if necessary.

Radioisotope Heater Units

Radioisotope heater units (RHUs) offer another thermal control choice when others are not practical. *Radioisotope* refers to the nuclear fuel used to produce heat from radioactive decay. RHUs provide the flexibility and efficiency to locate each unit where heat is needed most.

Closely related to their cousin, radioisotope thermoelectric generators (RTGs) used for power, RHUs provide an alternative to the traditional method, converting electrical power to thermal energy. To further explain, if using RTG as a power source, its heat would need to first be converted to electricity and then back into a thermal source using another method such as an electric heater. This would draw additional power from the RTG for thermal control. As previously mentioned, electricity is a scarce spacecraft resource and is closely reserved for the payload.

RHUs are reliable, lightweight, and can provide a continuous or variable heat source. Each RHU consists of a platinum-rhodium metal clad plutonium-238 fuel pellet, within nested graphite protective layers, all fitting within a cylindrical heat shield enclosure depicted in the following figure along with a penny to provide scale.

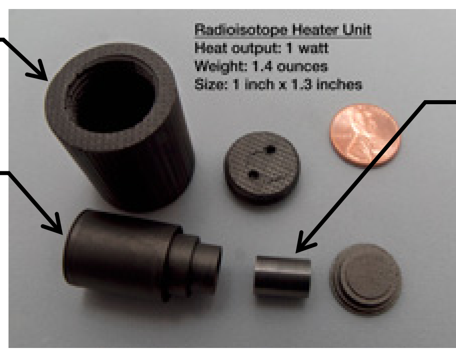

Figure 3.1: Radioisotope Heater Unit
[Reprint from source: NASA @ nasa.gov]

Variable RHUs or VRHUs were developed to provide thermostatic control which consists of up to 5 RHUs that fit within a cylindrical holder. This holder rotates on bearings by two temperature-sensitive bimetal springs to either expose one side that is painted white or the other side that is covered with a 22-layer Kapton MLI blanket. When needed to retain heat (warm the spacecraft), the cylinder is rotated so that the MLI blanket is facing space. When needed to release heat (cool the spacecraft), the cylinder is rotated to expose the white paint to space acting as a radiator.

The following table includes those NASA missions heated by RHUs, with each unit creating about 1W of heat initially which decreases over time.

Table 3.1: Radioisotope Heater Units (RHUs)

Spacecraft	Launch Year	RHUs	Mission
Pioneer 10/11	1972/73	12 RHUs each	Planetary flybys of Jupiter (10 & 11) & Saturn (11)
Voyager 1/2	1977	9 RHUs each	• planetary flybys of Jupiter, Saturn, plus interstellar space (1) • planetary flybys of Jupiter, Saturn, Uranus, and Neptune, plus interstellar space (2)
Galileo	1989	Orbiter - 103 Probe - 17	Venus and Earth flybys, Jupiter orbit, probe to Jupiter's atmosphere
Mars Pathfinder	1996	3	Mars Exploration
Cassini/Huygens	1997	Cassini orbiter - 82 Huygens probe - 35	Cassini: Venus, Earth and Jupiter flybys, Saturn orbit Huygens: Exploration of Saturn's moon, Titan.
Mars Exploration Rovers: Spirit & Opportunity	2003	8 RHUs each	Mars Exploration

Thermoelectric Coolers

Thermoelectric coolers (TECs) provide a solid-state driven method for thermal control, primarily used to cool areas of the spacecraft where needed. Typical uses have been for cooling low noise amplifiers, star trackers, and IR (infrared) sensors. TECs are also available commercially for our personal use, replacing the traditional "need for ice", for example in the form of portable coolers. Cooling is achieved through the Peltier effect, which is the inverse of the Seebeck effect; where Seebeck uses a heat source to produce a voltage (as in RTGs), Peltier uses a voltage source to induce heat transfer. The Peltier effect occurs by passing an electrical current through two junctions joined by two dissimilar semi-conductors, absorbing heat as it passes through the cold junction and emitting heat as it passes through the hot junction as depicted in the following figure.

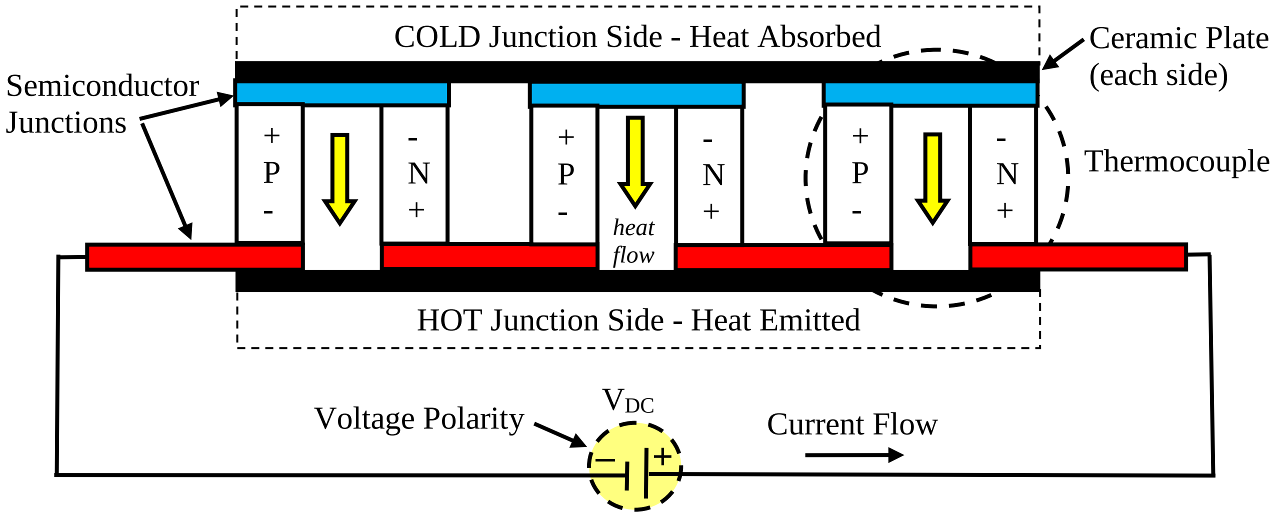

Figure 3.2: Single Stage Thermoelectric Cooler

The previous figure is an example of a single stage TEC, containing semiconductor elements electrically connected in series and physically configured in parallel consisting of six semiconductor elements – three thermocouples. These elements are physically configured in parallel so the heat flows evenly through each element. The series elements, known as N and P legs, contain "n" and "p" doped semiconductor elements respectively. The combination of this material difference (n vs. p) and electrical potential (voltage) across the p-n junction produces a proportional temperature difference. This forms a thermocouple device, where the heat is absorbed from the COLD junctions and emitted by the HOT junctions. Also, the V_{DC} polarity is very important here, since reversing the polarity also reverses the COLD and HOT junctions.

To provide thermal optimization, various configurations can be used, single-stage or multi-stage. Where multi-stage would consist of multiple single stages electrically connected in parallel. In any configuration, all the COLD junctions are connected to one ceramic plate and all the HOT junctions are connected to another ceramic plate.

Louvers

Louvers provide active thermal control by varying the heat transfer from a radiator (most common), between internal surfaces, or the exterior opening of the spacecraft wall (like a window). Heat can be transfer can be adjusted by varying the surface emissivity of the louver. The vane louver is the most widely used type of louver, consisting of the following:

- bimetallic spring actuator
- actuator housing
- rectangular blades (or vanes), VDA coated Kapton
- structural frame (like a window frame)

These louver blades move (like a home's venetian blind) to cover or expose a high-emissivity surface to control the amount of heat transfer. For example, when covering a radiator, the louvers would control the amount of heat rejected into space or retained in the spacecraft. You may also think of this as precision control (i.e. variable), where the radiator alone would provide coarse control (i.e. either on or off) as driven by its own properties and the amount of heat directed to it.

The bimetallic spring actuators thermally coupled to the structural frame, opens or closes the blades based on a set point that is reached in order to warm or cool the spacecraft. Actuation time is usually several minutes, unlike seconds that we may have experienced by opening or closing our venetian blinds. The range of emissivity can be from about 0.1 (fully closed) to 0.7 (fully open). In the fully closed position, the VDA surface is exposed to minimize heat transfer. In the fully open position, the underlying surface or radiator is uncovered to maximize heat transfer through the louver assembly. Also, the louver can be partially opened/closed to provide a range of thermal control positions.

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