Feasibility of Passive Cryogenic Cooling for Solar Powered Outer Planetary Missions

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Spacescience instruments with cooled detectors require innovative thermal cooling solutions to meet science objectives. Detector sensitivity increases with decreasing temperature and low optics temperatures are often needed to reduce background photon noise. As the detector spectral range coverage increases from the visible to far infrared also requires lower detector temperatures. Increasing demands on detector performance lead to larger format detectors and higher refresh rates resulting in significant increases in power dissipation. Passive coolers rely on emissive power of radiating surfaces to reject heat to space. As the operating temperature requirements of detectors and optics decreases, the ability to reject heat to space becomes increasingly more difficult. Reducing both cooler internal parasitic and external environmental heat loads and maximizing the passive cooler field of view to space will enhance performance. While instrument heat loads and passive cooler parasitic heat loads are controlled by instrument designers, the external environmental heat loads and cooler views to space are governed by spacecraft and mission designers. Solar powered planetary missions require large arrays to generate sufficient power for spacecraft subsystems and payloads. Two or more solar array wings with cell coverage of the order of 40-80 m^2 are often needed to generate sufficient power at 3-6 AU. These large arrays are typically symmetrically configured and can extend tens of meters. The arrays along with spacecraft attitude requirements near the target planetary bodies pose significant challenges for passive cooling at large AU. It is very difficult to provide a clear field of view to space for the cooler with large articulating arrays while keeping solar loads from impinging the cooler and meeting the spacecraft attitude science pointing requirements. This is counter intuitive because of the decreasing solar flux and colder planetary body temperatures at large AU. This paper presents the challenges and opportunities of passive cryogenic cooling versus active cooling for solar powered planetary missions.

Nomenclature

AIAA	=	American Institute of Aeronautics and Astronautics
AU	=	Astronomical Unit
AFT	=	Allowable Flight Temperature
AIRS	=	Atmospheric Infrared Sounder
APL	=	Applied Physics Laboratory
ACE	=	Advanced Cryocooler Electronics
CDR	=	Critical Design Review
CCD	=	Charge Coupled Device
CMOS	=	Complementary Metal-Oxide-Semiconductor
DOD	=	Department of Defense
EOS	=	Earth Observing System
EMI	=	Electromagnetic Inteference
EMC	=	Electromagnetic Compatibility
FPA	=	Focal Plane Array
GSE	=	Ground Support Equipment
GEO	=	Geosynchronous Equatorial Orbit

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Ge	=	Germanium
HEC	=	High Efficiency Cryocooler
HgCdTe	=	Mercury Cadnium Telluride
IR	=	Infrared
JWST	=	James Webb Space Telescope
JIRAM	=	Juno Infra-Red Auroral Mapper
JunoCAM	=	Juno Camera
L2	=	Lagrange Point 2
LEO	=	Low Earth Orbit
LNA	=	Low Noise Amplifier
LWIR	=	Long Wave Infrared
MICAS	=	Miniature Integrated Camera and Spectrometer
MISE	=	Mapping Imaging Spectrometer for Europa
MMRTG	=	Multi-Mission Radioisotope Thermoelectric Generator
MSL	=	Mars Science Laboratory
MEO	=	Medium Earth Orbit
M3	=	Moon Mineralogy Mapper
NGAS	=	Northrup Grumman Aerospace Systems
NIMS	=	Near-Infrared Mapping Spectrometer
OCO	=	Orbiting Carbon Observatory
PDR	=	Preliminary Design Review
PT	=	Pulse Tube
RTG	=	Radioisotope Thermalelectric Generator
RPS	=	Radioisotope Power System
RPM	=	Revolutions Per Minute
SEP	=	Solar Electric Propoulsion
SOA	=	State-of-the-art
Si:As	=	Silicon
TRL	=	Technology Readiness Level
TES	=	Tropospheric Emission Spectrometer
TV	=	Thermal Vacuum
USAF	=	United States Airforce
VIRTIS	=	Visible Infrared Thermal Imaging Spectrometer
VIR	=	Visual and InfraRed Spectrometer
VIMS	=	Visual and Infrared Mapping Spectrometer
TB	=	Thermal Balance
WMAP	=	Wilkinson Microwave Anisotropy Probe

I. Introduction

Power source options for long duration deep-space missions include solar arrays and radioisotope power systems (RPS). Solar arrays with specific power in the range of 40-80 W/kg are currently used in Earth-orbital missions and deep-space missions at distances up to 4 AU.[1] It is expected that both future orbital and deep-space missions will require advanced higher efficiency solar arrays with higher specific power (> 200 W/kg at 1 AU). RPS systems with specific power of ~3 W/kg are currently used in most deep-space missions beyond 4 AU.[1] NASA has long used radioisotope power systems for deep-space systems including Voyager, Galileo, Cassini, MSL Curiosity, New Horizons and will be using an RPS system for Mars 2020. Advanced radioisotope generators are under development by NASA for future space missions.[1]

Outer planetary missions require solar arrays capable of operating in extreme environments (high radiation, low temperatures, high temperatures, and dust). The solar array size needed to provide sufficient power for the spacecraft and all subsystems has increased significantly for spacecraft travel distances in excess of 3 AU. NASA's Juno mission to Jupiter has solar arrays with total area of 65 m^2 to provide 500 W of power at Jupiter.[2] Similarly, ESA's Rosetta mission to the comet 67P/Churyumov–Gerasimenko carries 64 m^2 of solar arrays to produce 400 W at 5.25 AU.[3] Two or more solar array wings with cell coverage of the order of 40-80 m^2 with articulation to track the sun are often needed to generate sufficient power at 3-6 AU. These large solar arrays are typically symmetrically configured and can extend tens of meters from the spacecraft body. The solar arrays along with spacecraft attitude requirements near

the target planets pose significant challenges for passive cooling at large solar distances. It is very difficult to provide a clear field of view to space for the cooler with large articulating solar arrays while keeping solar loads and planetary body infrared heat loads from impinging on the cooler and meeting the spacecraft attitude science pointing requirements. Cleary the large size of these solar arrays represent significant engineering challenges for spacecraft structures, dynamics, navigation and control including for thermal engineers responsible for designing passive cryogenic cooling systems.

Solar powered spacecraft for outer planetary missions with large solar arrays have additional operational constraints once they reach the intended targets. The communication and power subsystems will have orbital spacecraft pointing and attitude constraints to maintain communication links and batteries charged with the solar arrays. Spacecraft pointing for science acquisition will further complicate the spacecraft attitude constraints. Low altitude flybys of planetary bodies for science acquisitions will not only complicate spacecraft maneuvering, but can result in large views to the planetary bodies. This paper presents the challenges and opportunities of passive cryogenic cooling for solar powered outer planetary missions. In addition, the paper presents a brief description of active cooling options being considered for current and future NASA missions.

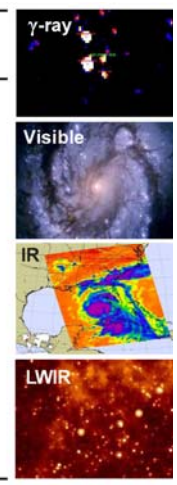
II. Cryogenic Cooling

The production of low temperatures on spacecraft payloads is needed to operate various types of space science instruments. In general, science instruments designed for making observations in the longer wavelengths require cold detectors and cold optics to reduce signal noise. This improves detector dynamic range, extends wavelength coverage, and enables the use of advanced detectors to observe wide range of phenomena from atmospheric dynamics to stellar births. Cryogenic temperatures are needed to take full advantage of space based astronomical observations. Table 1 shows the temperature requirements needed for various detector technologies.

Instruments with cold detectors and optics will often need cold thermal shields to reduce radiative and conductive heat loads on the detector and optics. Generally instrument optics are allowed to operate at warmer temperatures than the detector, but sufficiently cold to be able to neglect photon noise from these warm sources. In some cases, a cold optical shield is used to further reduce this photon noise contribution.

Table 1. Types and wavelengths of electromagnetic radiation, the blackbody temperature with peak emission at such wavelength radiation, and applicable detector types and their required operating temperature (Ref. 4).

Radiation Type	Wavelength (microns)	Blackbody Temp. (K)	Detector Technology	Detector Temp. (K)
γ-rays	10^{-5}	3×10^{8} K	Ge Diodes	80 K
γ-rays	10^{-4}	3×10^{7} K	Ge Diodes	80 K
x-rays	10^{-3}	3×10^{6} K	micro calorimeters	0.05 K
x-rays	10^{-2}	3×10^{5} K		0.05 K
UV	0.1	30,000 K	CCD/CMOS	200-300 K
visible	1	3000 K	CCD/CMOS	200-300 K
IR	2	1500 K	HgCdTe	80-130 K
IR	5	600 K	HgCdTe	80-120 K
LWIR	10	300 K	HgCdTe	35-80 K
LWIR	15	200 K	HgCdTe	35-60 K
LWIR	20	150 K	Si:As	7-10 K
LWIR	50	60 K	Ge:Ga	2 K
LWIR/μwaves	100	30 K	Ge:Ga	1.5 K
microwaves	200	15 K	Bolometers	0.1 K
microwaves	500	6 K	Bolometers	0.1 K



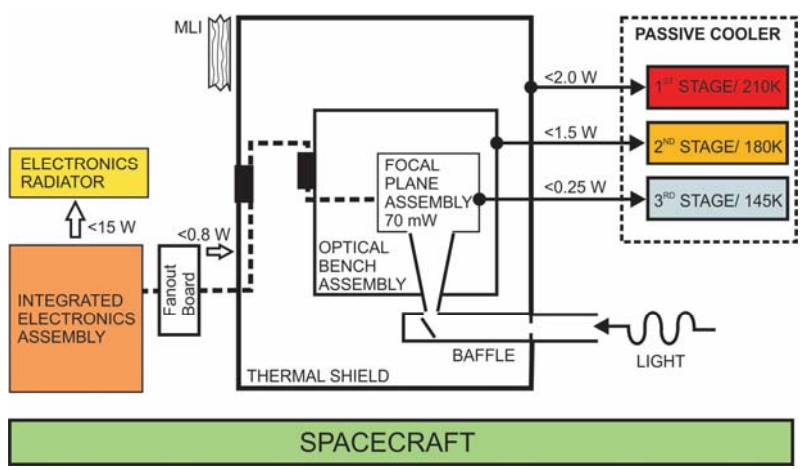


Figure 1. Thermal schematic for typical cryogenically cooled spectrometer: cryogenic staging within instrument minimizes heat load requirements on the 3rd stage passive cooler (Ref. 5).

Along with the various temperature zones needed, the heat lift requirements are equally important for architecting a suitable passive cooler design. The calculations for heat lift needed at various temperature stages involves determining the thermal environment, conductive and radiative parasitics, active dissipations and suitable heat load margins. A thermal schematic can be constructed with this first-order information to help with architecting suitable passive cooler designs (see Fig. 1).

JPL design principles for cryogenic systems follows the guidelines presented in the Aerospace Corporation paper titled "Thermal Uncertainty Margins for Cryogenic Sensor Systems," Ref. 6. At the conceptual design stage within Phase A/B of a Project, it recommends a minimum of 40-50% heat load margin and no less than 25% at the time of flight hardware testing to assure the cryogenic system meets design requirements. The Aerospace paper reviews the performance of 17 DOD and NASA cryogenically cooled sensor systems developed within 20 years of the paper publication in order to establish design and test guidelines for use on USAF cryogenic system designs. The study includes passive coolers, mechanical coolers and stored cryogen systems. Note that "heat load" includes not only parasitic heat loads but also active sensor dissipation. This is important, because, not only do parasitic heat loads tend to increase, but also, changes in the operational scenarios may result in increased active sensor heat loads. Guidelines are suggested for determining heat load and test margins as a function of Project milestone with the corresponding temperature design and test margins as a function of temperature for passive cooling systems (see Fig. 2). To demonstrate performance robustness during flight hardware testing with heat load, the GSE test hardware will typically include features allowing such testing and in some cases the flight hardware is instrumented with "ground use only" thermal hardware which may need to fly in place. The Aerospace guidelines have been accepted and are used widely in the cryogenic community and have proven very successful in recent JPL applications. We have used these design margin guidelines for both active and passive cryogenic systems on a number of successful flight projects, such as MICAS, AIRS, TES, OCO, OCO-2 and M3 (Fig. 3).

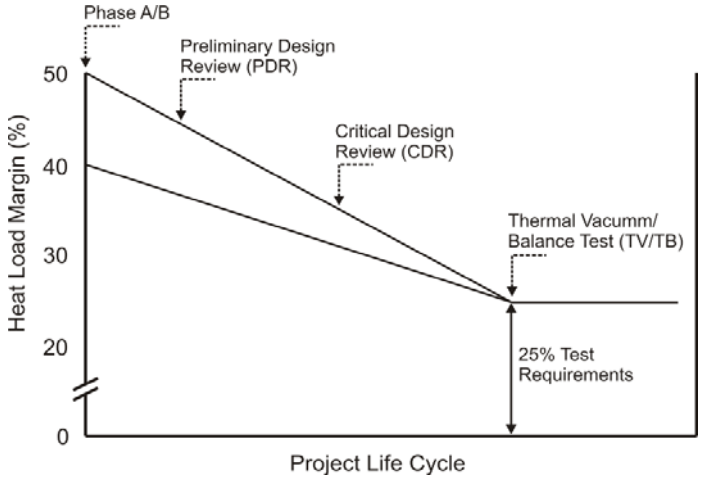


Figure 2. Recommended cryogenic heat load margins (Ref. 6).

III. Passive Cooling

Passive coolers rely on the emissive power of radiating surfaces to reject heat to cold space. As the operating temperature requirements of detectors and optics decreases, the ability to reject heat to space becomes increasingly more difficult because of the fourth power of absolute temperature effect. Lower detector temperatures with increasing power dissipations places higher demands on the passive thermal cooling architectures. Often

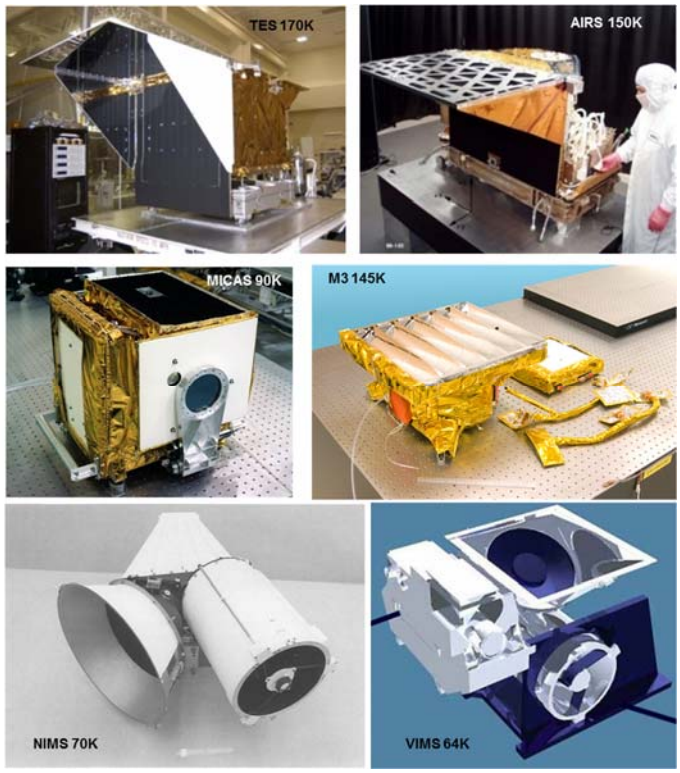


Figure 3. JPL passive cooler flight programs various configurations showing low-emissivity shields. These successful instruments have demonstrated long-term performance in excess of 10 years from LEO, and Moon orbit to outer planetary missions (Ref. 7-10).

Figure 4. NASA's Earth Observing System (EOS) Spacecraft: Left to right, Terra, Aqua and Aura spacecraft with large single-wing solar arrays. The EOS spacecraft are in sun-synchronous orbit at 705 km altitude with ideal accommodations for passive cooling (Ref. 11).

times multi-stage passive coolers are needed when two or more cooling temperature zones are required. This leads to nested configurations where the warmer stages of the cooler provide thermal protection by reducing conductive and radiative heat loads on the lower temperature radiator surfaces. The objective is to reduce parasitic heat loads on all the cooling stages and more importantly on the lowest temperature stages. Reducing both cooler internal parasitic heat loads and external environmental heat loads, and at the same time maximizing the passive cooler field of view to cold space, will enhance performance. While the instrument heat loads and passive cooler parasitic heat loads are internally controlled by instrument designers, the external environmental heat loads and cooler views to cold space are governed by the spacecraft and mission designers. Often times passive cooler designs require deployable shields to guard against solar and planetary infrared heat loads. They typically require specific accommodations on a spacecraft such as placement on the spacecraft anti-sun side with no line-of-sight views to warm spacecraft structures including solar arrays. In some cases, spacecraft orbital flip maneuvers are needed to keep the sun from impinging on the passive cooler radiator surfaces. The passive cooler performance both in terms of radiant capability and/or temperature is often limited by the presence of radiant sources within the cooler radiator surfaces field of view. Spacecraft with large solar arrays pose a significant challenge for passive cooler designers with obscured clear field of view to cold space and additional solar and/or infrared heat loads requirements.

Passive coolers operating in LEO, MEO and GEO orbits are quite common and have a long successful history. The performance of several heritage passive coolers is shown in Fig. 5. The plot shows a comparison of the passive cooler area divided by rejected heat load excluding parasitic heat loads as a function of

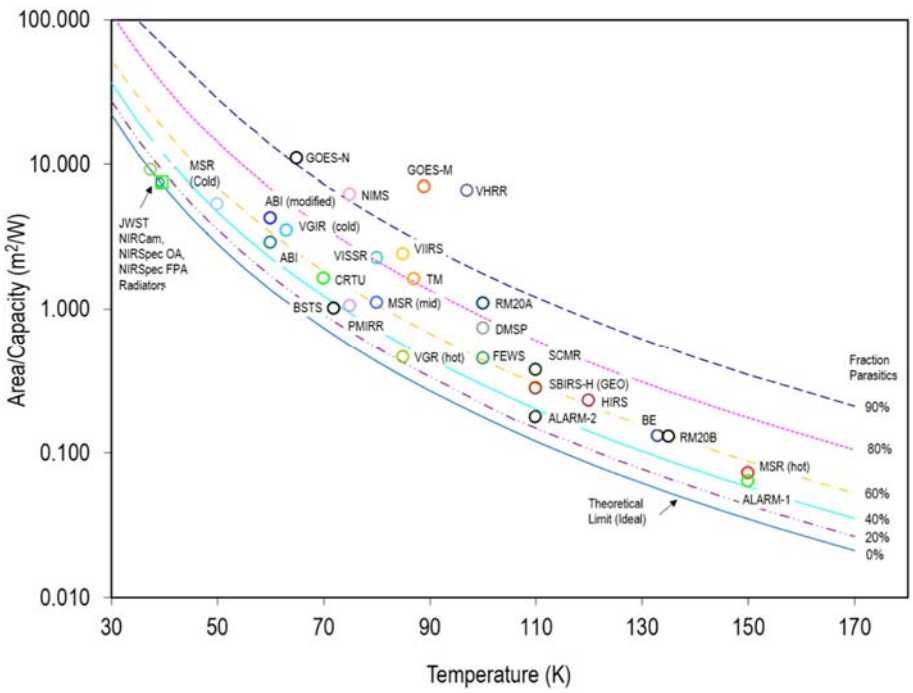


Figure 5. Performance of heritage passive coolers. Comparison of area divided by heat load (excluding parasitic loads) vs. temperature. Moving upwards at a fixed temperature, increased fractional parasitic heat load or reduced emittance causes performance to be reduced compared to an ideal radiator. For a fixed radiator design, increased heat load results in higher temperature and a reduction in the fraction of parasitic heat load. More challenging performance is represented by points at lower temperatures or points that are closer to the theoretical limit. Adapted from Ref. 12-13 (see reference for acronyms used in figure).

temperature. The theoretical ideal performance of a cooler is represented by the solid blue line at the bottom of the plot. As the fractional parasitic heat load increases or the effective emittance is reduced for a radiator at a fixed temperature, the degraded performance compared to an ideal radiator is shown when moving upwards at the radiator temperature. For a fixed radiator design, increased heat load results in higher temperature and a reduction in the fraction of parasitic heat load. More challenging performance is represented by data at lower temperatures or points that are closer to the theoretical limit. For LEO applications, sun-synchronous orbits with nadir pointing spacecraft are ideal because they inherently have an anti-sun side for the passive cooler which keeps the sun out without having to perform orbital flip maneuvers. NASA's EOS spacecraft with single solar arrays took advantage of the sun-synchronous orbit to provide ideal design conditions for instruments with passive cryogenic coolers (see Fig. 4). Earth shields are used to block infrared heat loads from impinging on the cold radiator surfaces. Lower altitude orbits are more difficult because they lead to larger Earth shields, likely needing to be one-time deployable shields in order to fit within the launch vehicle dynamic envelope (see Fig. 3-4). Spacecraft operating in MEO, GEO provide a more benign environment primarily because the Earth view factors and infrared heat loads decrease significantly. It is common for spacecrafts to have a two-wing solar array configuration because it simplifies; 1) spacecraft attitude control, 2) solar array stowing and deployment, and 3) solar array structural and dynamic design. The two-wing solar array spacecraft configuration leads to one of the solar arrays in the field of view of the passive cooler. The solar array not only partially blocks the field of view of passive cooler to cold space, but is a potentially high infrared heat load source. Solar arrays in LEO often operate with cell temperatures in the order of 50-70°C on the front side. The blackbody emissive power of solar arrays at 70°C is 786 W/m^2 and can easily lead to back infrared heat loads on a passive cooler in the order of 10-20 W/m^2 with small view factors of only a few percent. In some instances, the solar arrays can be the source of back scattered solar and infrared heads on the passive cooler. Passive cooler shields, if feasible, may provide significant protection against these external heat loads, but they also reduce the view factors to cold space. Simple directional shields with low-emisssivity specular coatings, to the more complex Winston cones, provide various degrees of protection at the cost of reduced views to cold space and added cost and complexity.

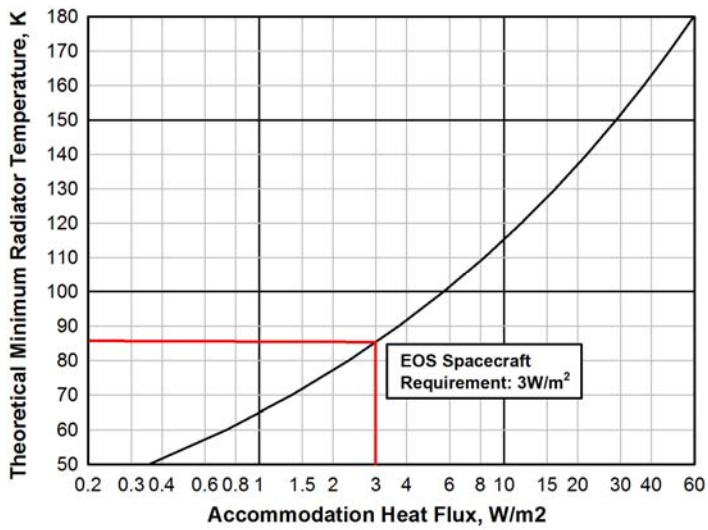


Figure 6. Passive cooler theoretical minium radiator temperature as a function of spacecraft accommodation heat flux. NASA's EOS used a spacecraft accommodation requirement of 3 W/m^2.

For LEO missions, a maximum instrument accommodation heat flux requirement is typically placed on the spacecraft of the order of less than a few watts per meter square. This requirement is levied on the spacecraft and accounts for all external radiative heat loads on the passive cooler radiator surfaces due to the spacecraft accomodations. This requirement typically does not include the conductive and radiative heat loads across the mechanical mounting interface between the instrument and spacecraft. A requirement was placed on all of NASA's EOS spacecraft of less than 3 W/m^2 and is consistent with passive cooler designs requiring cooling temperatures above 90 K (see Fig. 6). Accommodation heat flux requirements between the spacecraft and instrument needs to be established early in the design phase so that passive cooler designers can proceed with preliminary designs. The heat flux requirement will set the theoretical minimum temperature achievable for passive cooler designs as shown in Figure 6.

Two-wing solar array spacecraft pose significant challenges for passive cooler architectures and may lead to designs with minimum temperatures much above 100 K and low heat rejection capability. In general for LEO, if passive cryogenic cooling is needed a single-wing solar array spacecraft configuration is needed such as NASA's EOS spacecraft Terra, Aqua and Aura.

Astrophysics missions with spacecraft operating at the Sun-Earth Lagrange point 2 (L2), Earth-trailing or similar orbits are ideal for passive cooling. These missions by design are far away from planetary bodies and typically require partial or full-sky surveys to meet the scientific mission objectives. From L2, the Earth is aligned with the Sun and

therefore, the Earth contributions are negligible even though it is only at 0.01 AU from L2. This greatly simplifies the passive cooling architectures because the solar arrays can be placed on the sun-side of the spacecraft and the passive cooler directly opposite with a guaranteed 100% view to cold space and no environmental heat loads. Past missions such as Planck, Herschel, Gaia, and WMAP have operated at L2 and NASA's JWST will also take advantage of this orbit and reside at L2.

IV. Survey of Solar Powered Planetary Missions

Spacecraft designed for operation in the inner solar system usually rely on the use of photovoltaic solar arrays to generate electricity from sunlight to power the spacecraft and all payloads. In the outer solar system, where the solar flux is too low to produce sufficient spacecraft power, radioisotope thermoelectric generators (RTGs) are used as a power source. RTGs provide electrical power for spacecraft by converting the heat generated by the decay of plutonium-238 (Pu-238) fuel into electricity using thermocouples. Since they have no moving parts that can fail or wear out, RTGs have historically been viewed as a highly reliable power option. To date, solar power has been practical for spacecraft operating no farther from the Sun than the orbit of Jupiter at 5 AU with a solar flux of ~50 W/m^2. The spacecraft Juno, Magellan, Mars Global Surveyor, and Mars Observer used solar power as does the Earth-orbiting Hubble Space Telescope. The Rosetta space probe, launched March 2004, used its 64 m^2 of solar panels as far as the orbit of Jupiter; previously the furthest use was the Stardust spacecraft at 2 AU.[14] Another spacecraft is Dawn which went into orbit around Vesta in 2011 and Ceres in 2015.[15-20] The potential for solar powered spacecraft beyond Jupiter continues to be studied.[21-22] An overview of Rosetta, Dawn, and Juno are shown in Table 2.

A. Why does NASA use solar technology in space?

To date, solar power has been used on more than 99% of all space missions.[22] The only currently viable power source alternative in space is radioisotope power systems. Radioisotope power systems have been used in many deep-space missions, where solar power is inadequate or impractical.

NASA is studying the use of solar power to propel spacecraft using Solar Electric Propulsion (SEP).[23] Traveling to the outer planets becomes an increasingly difficult problem for generating sufficient power for the spacecraft and SEP needs. Scientists want to investigate Europa, Jupiter's icy moon—which is probably the only other major water supply in the solar system.[24-28] Europa, at about 780 million kilometers from the Sun, receives only about 1/25 of the amount of light as Earth orbiting spacecraft. The other problem with Jupiter is it's a high-radiation environment.

B. Rosetta Mission

The International Rosetta mission was approved in November 1993 by ESA's Science Programme Committee as the Planetary Cornerstone Mission in ESA's long-term space science programme. Rosetta was originally set for a rendezvous with Comet 46 P/Wirtanen; however, after postponement of the original launch a new target was selected for Comet 67P/Churyumov-Gerasimenko.[3,14] The mission was launched on March 2, 2004, on a 10-year journey towards comet 67P/Churyumov-Gerasimenko. During its 10 year journey towards the comet, the spacecraft passed by two asteroids: 2867 Steins (2008) and 21 Lutetia (2010), before entering deep-space hibernation mode in

Table 2. Outer Planetary Missions (Ref. 1-3, 14-28).

S/C	Mission	SA Power, Specific Power, SA Temp	Solar Array size/ Configuration	SEP	Launch and Arrival Dates	Mission Lifetime
Rosetta	Comet Orbiter Lander	7.1 kW at 1 AU 850 W at 3.4 AU 400 W at 5.25 AU and 143 K ~50 W/kg at 1 AU	64 m^2 solar array area Two 2.25m x 14m wings 32 m tip-to-tip +/- 180 deg rotation	No	Mar 2004 to Sept 2016	12 years
Dawn	Protoplanet Orbiter Vesta/ Ceres	10.3 kW at 1 AU 1.3 kW at 3 AU, 185 K ~80 W/kg at 1 AU	36.4 m^2 solar array area Two 2.2 m x 8 m wings 19.7 m tip-to-tip	Yes	Launch:9/27/2007 Vesta:7/16/2011 Ceres: 3/6/2015	Primary: June 2016 Extended: June 2017 (~10 yrs)
Juno	Jupiter Orbiter	14 kW at 1 AU 500 W at Jupiter ~50 W/kg at 1 AU	65.5 m^2 solar array area Two 24.0 m^2 and one 17.5 m^2 wings, Fixed wings at 120deg apart	No	Aug 2011 to Feb 2018	Farthest solar power S/C from Earth (6 yrs)

June 2011.[3] Rosetta arrived at the comet on August 6, 2014 and following deployment of the lander, Philae, on November 12, 2014, the orbiter accompanied the comet through perihelion and back to deep space until the mission ended on September 30, 2016.[3] ESA's historic Rosetta mission concluded as planned with a controlled impact onto the comet it had been investigating for more than two years.

Rosetta's main objective was to rendezvous with, and enter orbit around, comet 67P/Churyumov-Gerasimenko and to perform observations of the comet's nucleus and coma.[3]

The Rosetta spacecraft design was based on a box-like structure, roughly 2 m × 2 m × 2 m, on which all subsystems and payload equipment are mounted.[3] Two large solar panels, with a combined area of 64 m², each stretch out to 14 m in length.[3] The total solar array span from tip to tip is 32 m.[3] The Rosetta orbiter carries an instrument suite of 11 science instrument packages along with the Philae lander which carries 10 science instrument packages (see Fig. 7).[14]

The large two-wing configuration on the Rosetta spacecraft and complex maneuvering produces changing environment and make it impractical to provide cryogenic passive cooling. The Visible Infrared Thermal Imaging Spectrometer (VIRTIS) instrument on Rosetta was a primary science payload and required cryogenic cooling of the focal plane.[29] The visible channel charge coupled device (CCD) covers the spectral range of 0.25 to 1 µm and along with spectrometer were cooled passively to 130 K.[29] The two instrument infrared channels with coverage of 1 to 5 µm and 1.9 to 5 µm were actively cooled to 70 K with one cryocooler each.[29] Both infrared channels use mercury cadmium telluride (HgCdTe) focal planes.[29]

Figure 7. Top: Rosetta spacecraft with Philae lander at comet 67P/Churyumov-Gerasimenko. Bottom: Rosetta spacecraft with two-wing rotating solar arrays and Philae lander at comet 67P/Churyumov-Gerasimenko (Ref. 14).

C. Dawn Mission

NASA's Dawn spacecraft is on a mission to study the two most massive objects in the main asteroid belt between Mars and Jupiter, Vesta and Ceres.[15-20] At each target, Dawn acquires color photographs, maps the elemental and mineralogical composition, measures the gravity field and searches for moons.[15-20] The data gathered by Dawn will enable scientists to understand the conditions under which these objects formed, determine the nature of the building blocks from which the terrestrial planets (like Earth) formed, and contrast the formation and evolution of Vesta and Ceres.[15-20]

The Dawn spacecraft combines innovative state-of-the-art technologies pioneered by other recent missions with off-the-shelf components and, in some cases, spare parts and instrumentation left over from previous missions (see Fig. 8-9).[15-20] The core of the Dawn spacecraft's structure is a graphite composite cylinder with tanks for the ion engines' xenon gas and the conventional thrusters' hydrazine mounted inside the cylinder.[15-20]

Dawn's futuristic, hyper-efficient ion propulsion system allows Dawn to go into orbit around two different extraterrestrial targets, a first for any spacecraft. Each of Dawn's three 30-centimeter diameter ion thrust units is movable in two axes to allow for migration of the spacecraft's center of mass during the mission.[15-20] This also allows the attitude control system to use the ion thrusters to help control spacecraft attitude. Dawn has used ion propulsion for years at a time, with interruptions of only a few hours each week to turn to point the spacecraft's antenna to Earth.[15-20]

The thrusters work by using an electrical charge to accelerate ions from xenon fuel to a speed seven to 10 times that of chemical engines. The electrical power level and xenon fuel feed can be adjusted to throttle each engine up or down in thrust. The engines use only about 3.25 milligrams of xenon per second (~280 g over 24 hours) at maximum thrust.[15] The Dawn spacecraft carried 425 kg of xenon propellant at launch.[19] Xenon was chosen because it is chemically inert, easily stored in a compact form, and the atoms are relatively heavy so they provide a relatively large thrust compared to other candidate propellants. At maximum thrust, each engine produces 91 millinewtons.[20]

The electrical power system provides power for all onboard systems, including the ion propulsion system when thrusting. Each of the two solar arrays is 8.3 m long by 2.3 m wide.[20] On the front side, 18 m^2 of each array is covered with 5,740 individual photovoltaic cells.[17] Dawn's solar cells are about 28% efficient, and on Earth, the two wings combined could generate more than 10 kW.[17] The arrays are mounted on opposite sides of the spacecraft, with a gimbaled connection that allows them to be turned at any angle to face the sun. When the solar arrays are deployed, Dawn's wing span is almost 20 m, and each wing weighs almost 63 kg.[18]

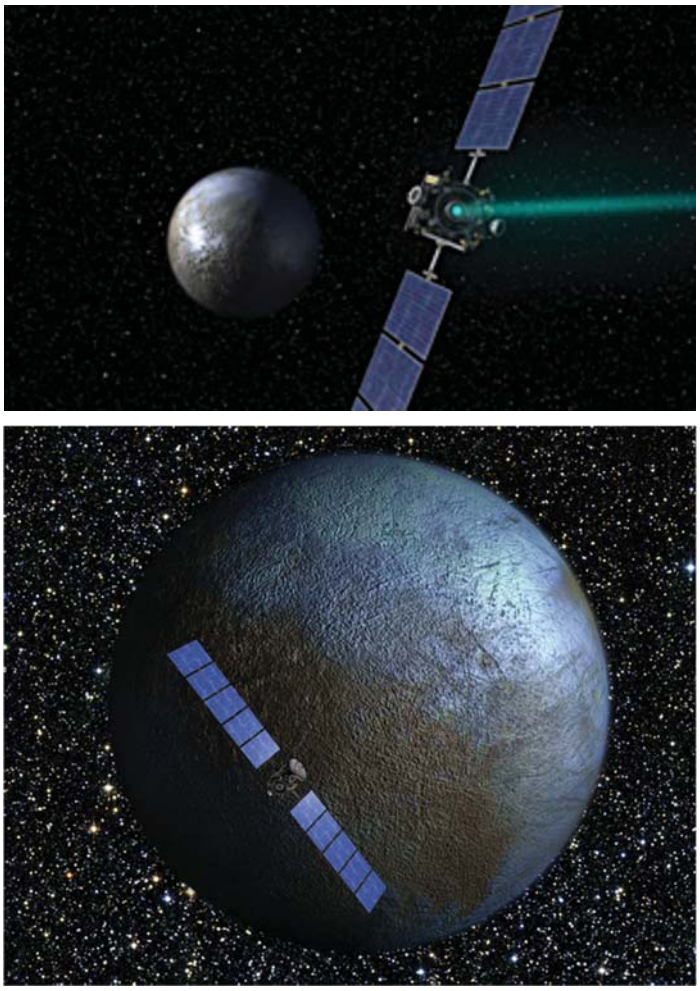


Figure 8. Top: Dawn spacecraft arrives at Ceres (March 2015 top figure). Bottom: Dawn spacecraft in orbit at Ceres (Ref. 15).

The two-wing solar array configuration on this spacecraft along with the environments make it difficult to accommodate a passive cryogenic cooler on this spacecraft. The Visual and InfraRed (VIR) Spectrometer instrument provided by the Italian Space Agency (ASI) uses a crycooler to cool its HgCdTe infrared detector to 70 K and the visible CCD detector and spectrometer are passively cooled to 130 K (Ref. 18). The VIR spectrometer visible channel covers the range of 0.25 to 1.05 µm and the infrared channel 1 to 5.0 µm.[29] The VIR instrument is a re-build of the mapping channel of Rosetta VIRTIS spectrometer and the optical concept is inherited from the visible channel of the Cassini Visible Infrared Mapping Spectrometer (VIMS-V) developed for the Cassini mission.[29]

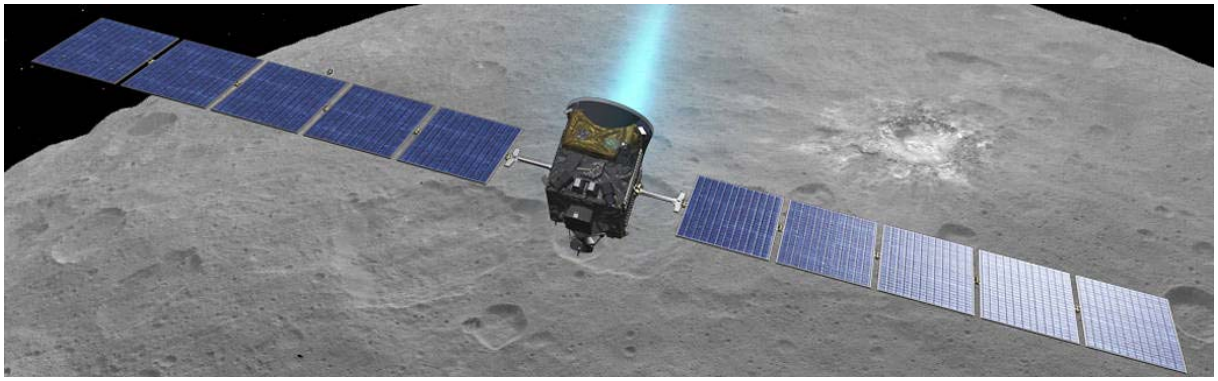


Figure 9. Dawn spacecraft with two wing rotating solar arrays (Ref. 15-20).

D. Juno Mission

Juno is the first mission to Jupiter to use solar panels instead of the radioisotope thermoelectric generators (RTG) used by Pioneer 10, Pioneer 11, Voyager, Ulysses, Cassini–Huygens, Galileo orbiter, and New Horizons. It also recently became the farthest solar-powered spacecraft in the history of space exploration. The Juno mission, launched in 2011, arrived at Jupiter on July 4, 2016. Once in orbit at Jupiter, Juno only receives 4% of the sunlight as it would on Earth, but advances made in solar cell technology over the past several decades, makes it both economically and technically preferable to use solar cells to provide power at a distance of 5 AU. The Juno spacecraft uses three solar panels symmetrically arranged around the spacecraft (see Fig. 10). Each solar panel is 2.7 m by 8.9 m long, the largest on any NASA deep-space probe.[31-36]

The combined mass of the three panels is nearly 340 kg. If the panels were optimized to operate at Earth, they would produce 12-14 kW of power. Only about 486 W is generated when Juno orbits Jupiter, declining to near 420 W as radiation degrades the solar cells.[31-36]

Juno is the second spacecraft to orbit Jupiter, after the Galileo orbiter, which orbited from 1995 to 2003.[2] Unlike all earlier spacecraft to the outer planets,[31-36] Juno is powered only by solar arrays, whereas RTGs are commonly used for missions where they enable or significantly enhance a mission's abilities to meet its scientific goals. For Juno, however, the three largest solar array wings ever deployed on a planetary probe play an integral role in stabilizing the spacecraft as well as generating power. The Juno spacecraft is spin-stabilized with a rotation of 1 RPM for cruise, 2 RPM (angular velocity of 12 deg per second) for science and 5 RPM for main engine maneuvers.[31-36]

Juno was designed for highly elliptical polar orbits of ~14 days, during which it will spend few hours close to the planet so as to maximize science return and minimize radiation exposure in the harsh environment surrounding Jupiter. A total of 37 orbits are planned over a period of 20-months to complete the mission.[31-36] In science mode, the spacecraft spins at 2 RPM and automatically sweep instrument and sensor fields of view through space once per spin cycle. At 2 RPM, the instrument will be able to scan Jupiter's extent about 240 times in the 2 hours that it takes to fly from pole to pole.[31-36]

The Juno Infra-Red Auroral Mapper (JIRAM) instrument acquires infrared images and spectra of Jupiter to investigate auroral structure and troposphere structure, and to perform atmospheric sounding.[37-38] JIRAM consists of both an IR imager and an IR spectrometer. The JIRAM heritage comes from the Italian-made, visual-infrared imaging spectrometers

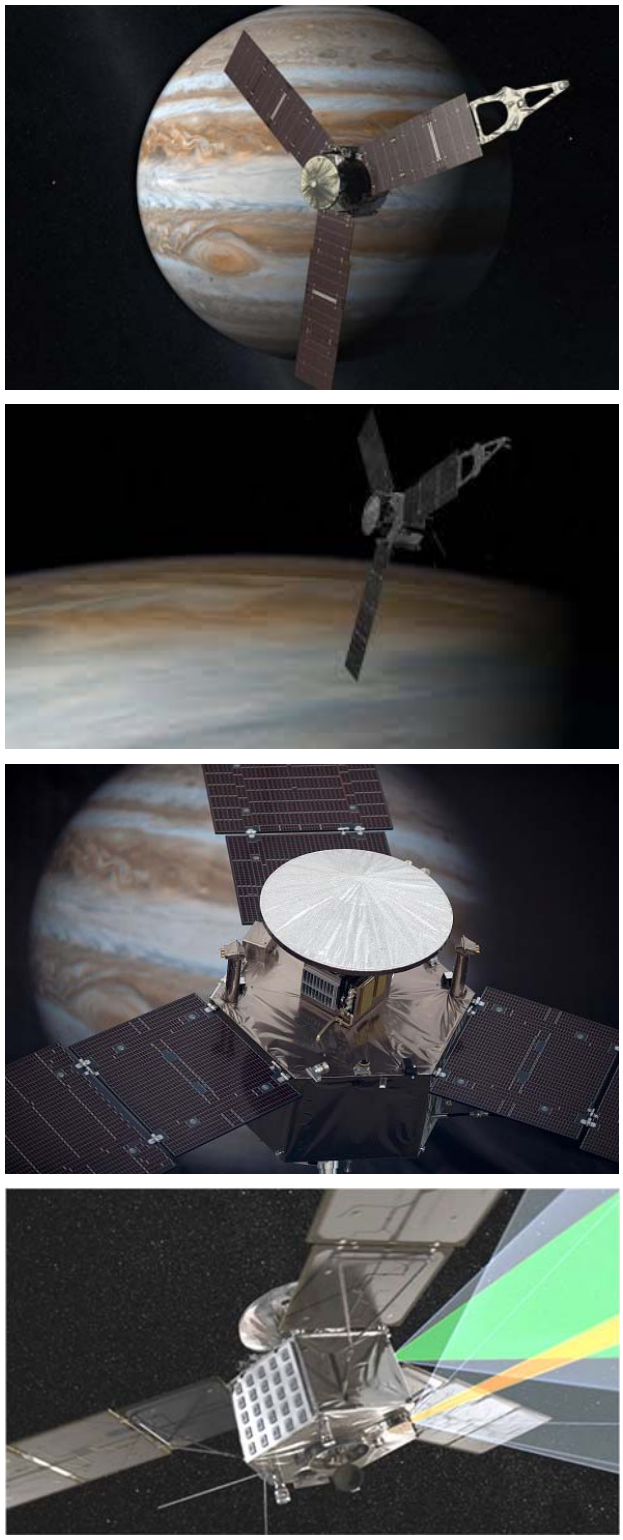


Figure 10. Juno spacecraft with three fixed solar arrays at Europa. Bottom: The JIRAM instrument is mounted to the aft deck of the Juno spacecraft and views in the same direction as the other optical instruments, UVS and JunoCam (Ref. 31-36).

dedicated to planetary exploration, such as VIMS-V on Cassini, VIRTIS on Rosetta and Venus Express, and VIR-MS on the Dawn mission.[37-38] This investigation is provided by the Italian Space Agency (ASI). Two distinct IR focal plane detectors are used for imaging and spectroscopy after the telescope. The infrared detectors covers a spectral range of 2 to 5 µm and use HgCdTe with CMOS mux technology.[37] Both infrared detectors operate at 95 K and the optical head at 130 K.[38] This instrument uses a 2-stage passive cooler design configuration to provide cooling at 95 K and 130 K. The first-stage radiator operating at 130 K protects the second-stage radiator at 95 K with both radiators viewing cold space. The spacecraft thermal and radiation environments posed significant challenges for JIRAM instrument team. The passive cooler was accommodated on the same side of the spacecraft with main engine (see Fig. 11). This appears to be the best possible location for the instrument with relatively good views to cold space. However, it has significant potential challenges such as contamination and infrared heat loads from the main engine, in addition to infrared planetary heating from Jupiter during the low altitude science acquisition pass. Figure 6 shows that an accommodation heat flux of less than 3-4 W/m^2 is needed to passively cool to the 95 K temperature range.

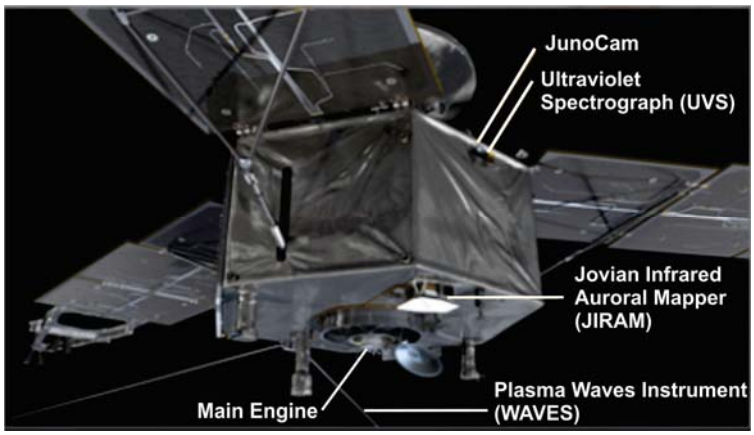


Figure 11. JIRAM instrument installed on Juno on the aft deck (Ref. 31).

There are no known technical papers in the open literature that address the thermal design of this instrument. The highly elliptical orbits with ~2 hr science data acquisition periods with a relatively high altitude of 4200 km and 14 day orbit period provides a benign external thermal environment.[38] The short science acquisition duration along with a fairly benign external environment does permit the use of passive cooling to accommodate a passive cryogenic cooler on this spacecraft. It is noted that previous versions of this instrument concept on the Rosetta and Dawn spacecraft required 70 K for its infrared focal plane whereas the JIRAM instrument infrared focal plane is only cooled to 95 K. It is speculated that the thermal design for the JIRAM team was very challenging as noted in Ref. 20.

V. Solar Array Technology

In order to operate five-and-a-half times farther away from its power source than Earth-observing satellites, Juno is equipped with more than 18,000 solar cells (see Fig. 12).[31-36] The Juno spacecraft needs about 405 W at Jupiter to operate. The thousands of reddish-blue solar cells are located on 11 panels, four on two of the spacecraft's 115 kg wings and three on the third wing (see Fig. 11).[31-36] One of the main processing challenges came in the fact that the wings are so large, they can't support their own weight in gravity. The entire solar arrays combined are almost 340 kg.[35] They're a little more massive than typical solar arrays because of all these various requirements of stiffness

Figure 12. Juno rigid solar arrays wing deployment: ~50-80 W/kg at 1 AU, TRL 9 on Juno (left) and Dawn (right) (Ref. 22).

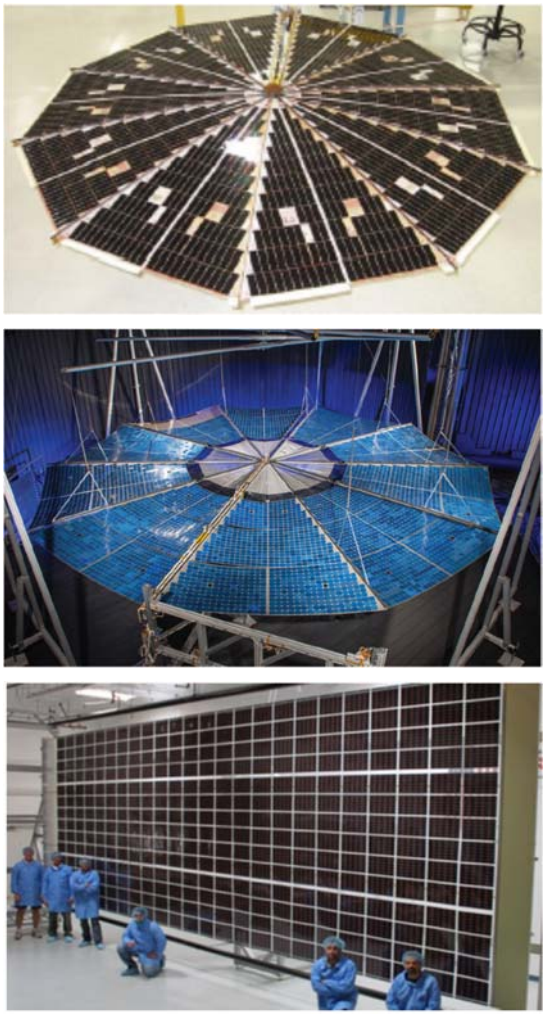


Figure 13. Top: ATK Ultraflex solar array: ~150 W/kg at 1 AU, Ultraflex TRL 9 on Phoenix, Middle: ATK Megaflex, TRL 6. Bottom: Deployable Space Systems (DSS), Rosa/Megarosa, ~150 W/kg at 1 AU, TRL 6 (Ref. 21-22).

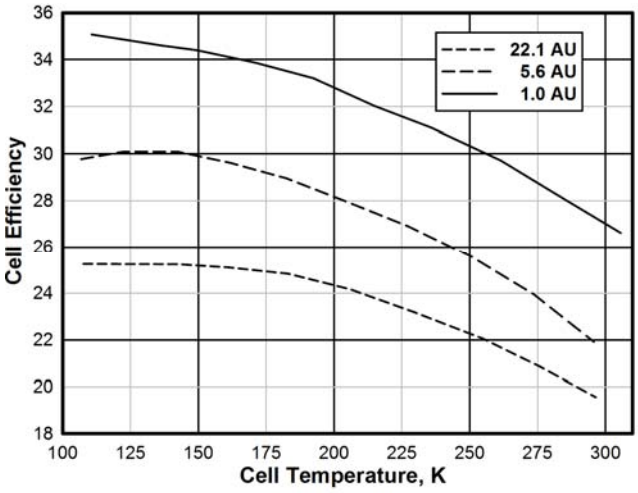


Figure 14. Low light triple junction performance as a function of cell temperature (Ref. 21).

and pointing and carrying the magnetometer at the tip of one of the wings. The wings have to be stiff and strong because Juno is a spinning spacecraft and the wings dominate how true it spins on its axis.

Two types of solar cells are common outside our hospitable atmosphere. Silicon cells covered by thin glass to avoid degradation from radiation make up the 16 arrays flanking the International Space Station. Multi-junction cells made of gallium arsenide and similar materials resist degradation better than silicon and are the most efficient cells currently made, with energy conversion efficiencies up to 34%.

State-of-the-art solar cell technology for space can be expensive compared to commercial terrestrial cell technology. Figure 13 shows the TRL 9 Ultraflex solar arrays and TRL 6 Megaflex. The ATK Megaflex technology is the same as Ultraflex with improved stowed packaging. Figure 13 also shows the Deployable Space Systems (DSS) higher power density solar array technology at TRL 6.

Outer planet missions operate with solar arrays at low irradiance and low temperature (LILT). Figure 14 show the performance of LILT triple junction cell as a function of temperature and AU distance.[21] Manufacturers continue to develop solar cell technology for Earth orbit and test at 1 AU and 28°C as standard conditions. Testing solar cells for LILT is costly, time-consuming and impractical. Flight-qualified solar cells use triple junction cells from 2 US vendors with 29% conversion efficiency at 1 AU and 28°C.[22] However, there is significant variation in cell performance under LILT conditions. Private industry is developing 4 to 6 junction cells and it is expected that with sufficient funding, 33 to 36% cell efficiency will be achieved in the near future.[22]

VI. NASA's Planned Europa Mission Thermal Architecture

NASA's planned Europa mission is designed to conduct detailed reconnaissance of Jupiter's moon Europa and investigate whether the icy moon could harbor conditions suitable for life.[24-28] The mission would put a spacecraft in orbit around Jupiter in order to perform a detailed investigation of the giant planet's moon Europa -- a world that shows strong evidence for an ocean of liquid water beneath its icy crust and which could host conditions favorable for life.[24-28] The mission involves sending a highly capable, radiation-tolerant spacecraft into a long, elliptical orbit around Jupiter to perform repeated close flybys of Europa. Beyond Earth, Jupiter's moon Europa is considered one of the most promising places in the solar system to search for signs of present-day life.[24-28]

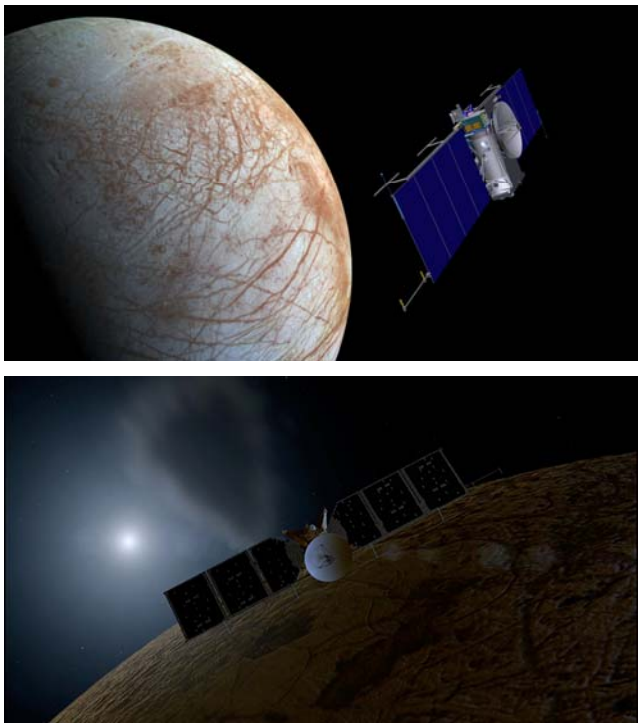


Figure 15. Baseline Europa mission spacecraft with two-wing rotating solar arrays (Ref. 26).

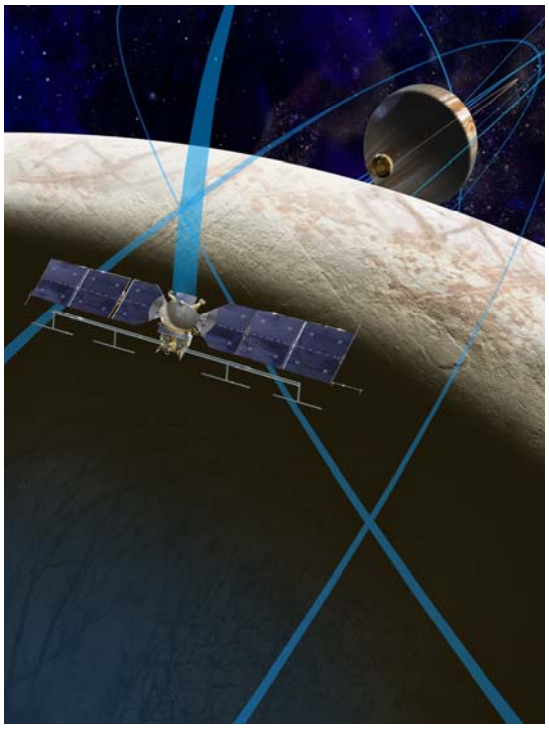


Figure 16. Artist's rendering shows a concept for a future NASA mission to Europa in which a spacecraft would make multiple close flybys of the icy Jovian moon, thought to contain a global subsurface ocean (Ref. 24-28).

During the nominal mission, the spacecraft would perform about 45 flybys of Europa at closest-approach altitudes varying from 25 to 2700 km above the surface.[24-28] The mission is a follow-up to studies made by the Galileo spacecraft during its eight years in Jupiter orbit, which indicated the existence of a subsurface ocean on Europa.[24-28] Plans to send a spacecraft to Europa were initially conceived with projects such as Europa Orbiter and Jupiter Icy Moons Orbiter, in which a spacecraft would be placed into orbit around Europa.[24-28] However, due to the strong impact of radiation from Jupiter's magnetosphere in Europan orbit, it was decided that it would be safer to inject a spacecraft into an orbit around Jupiter and make several close flybys of the moon instead.[24-28] The mission is now formally called Europa Clipper and is a collaboration between NASA's Jet Propulsion Laboratory and the John Hopkins University Applied Physics Laboratory (JHU-APL).

The goals of the Europa Clipper orbiter are to explore Europa, investigate its habitability and aid in the selection of a landing site for a lander.[24-28] Each flyby would cover a different sector of Europa in order to achieve a medium-quality global topographic survey, including ice thickness.[24-28]

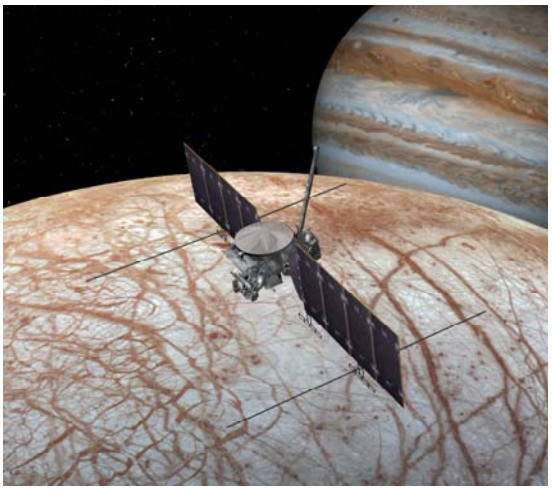


Figure 17. Baseline Europa spacecraft with two-wing rotating sun-tracking solar arrays (Ref. 24-28).

A wide orbit of Jupiter with several flybys of Europa will minimize radiation exposure and increase data transfer speed. Because Europa lies well within the harsh radiation fields surrounding Jupiter, even a radiation-hardened spacecraft in near orbit would be functional for just a few months.[24-28] Another key limiting factor on science for a Europa orbiter is not the time the instruments can make observations, but rather, it is the time available to return science data to Earth.[24-28] Most instruments can gather data far faster than the communications system can transmit it to Earth because there are a limited number of antennas available to receive the data.[24-28] Between each of the flybys, the spacecraft would have seven to ten days to transmit data stored during each brief encounter. That would let the

spacecraft have up to a year of time to transmit its data compared to just 30 days for an orbiter.[24-28] The result would be almost three times as much data returned to Earth, while reducing exposure to radiation.[24-28]

Both RTGs and photovoltaic power sources were assessed to power the orbiter.[25] Late in 2013, solar panels for the spacecraft were baselined for the mission (see Fig. 15-17). The spacecraft equipped with 90 m^2 of solar array recharges its batteries in between flybys while relaying data back to Earth. The Europa Clipper mission orbits will pass through Jupiter's intense magnetosphere, which is expected to gradually degrade the solar cells as the mission progresses.[27]

The Mapping Imaging Spectrometer for Europa (MISE) instrument on the Europa Clipper mission requires cooling to 85 K and 109 K for the focal plane and spectrometer optics, respectively. In Phase A of the Project, a trade study was performed to determine if passive cooling was feasible for this application. At Europa, the environmental heat sources on the passive cooler come from Europa itself and the Sun. The IR and albedo contributions from Jupiter are small and can be neglected. Europa with a mean suface temperature of 102 K and albedo of 0.67 can be a significant infrared heat source and reflected solar heat source. The solar flux at Europa varies from 45 to 56 W/m^2 (see Fig. 18) .[24-28] The spacecraft configuration with a two-wing solar array limits the possibility of accommodating a clear view to cold space for the passive cooler.

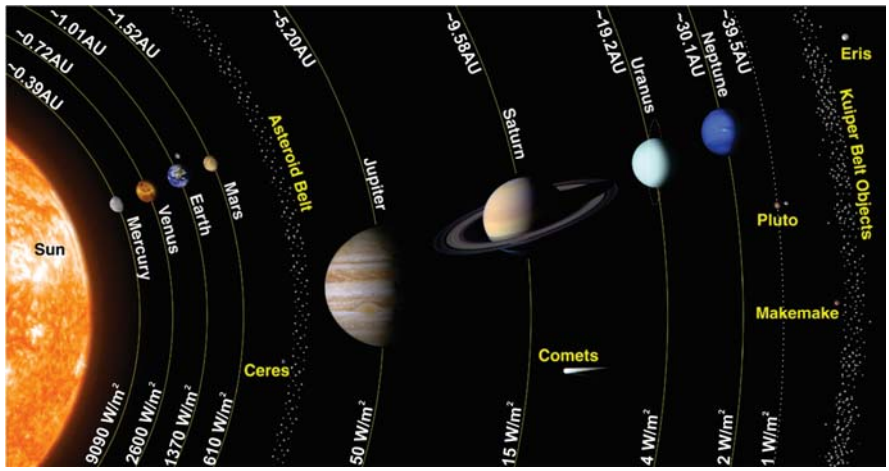


Figure 18. Solar irradiance for planetary bodies as a function distance from the Sun.

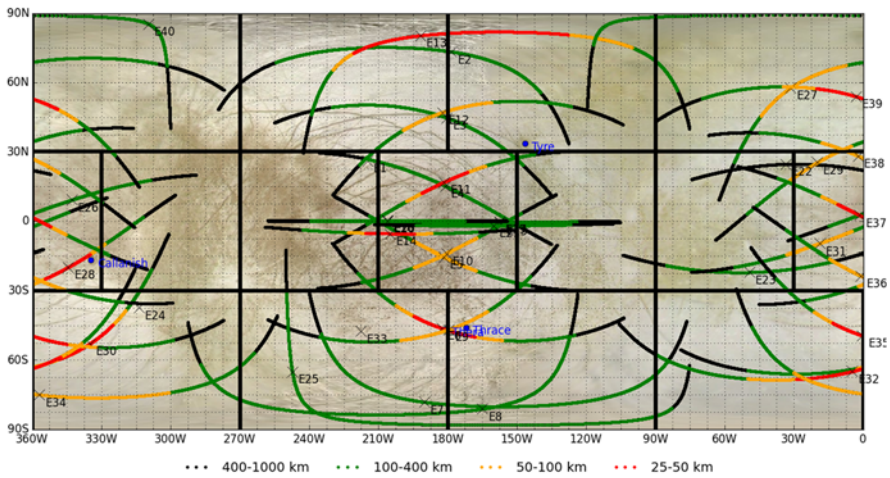


Figure 19. Europa Clipper flyby altitude for a representative tour.

The MISE instrument acquires science data during each flyby for about 8hrs beginning at an altitude of approximately 66,000 km through closest approach and outbound to again 66,000 km altitude. Minimum altitude at closest approach varies from 25 km to 2700 km. The science platform with most of the payloads including MISE is nadir pointing for the 8 hour duration with various longitude and latitude passes to acquire science. Figure 19 shows the flyby trajectories for a typical reference tour. The two-wing spacecraft solar array configuration along with Europa, the Sun and nadir pointing attitude constraints during the flyby revealed that it was impractical to design a suitable passive cooler to provide the cooling needed at 85 K and 109 K. A passive cooler with a large Winston cone to protect against Europa infrared heat loads and solar flux had the best performance for most of the flybys, but failed to meet the temperature and cooling requirements for a significant number of the flybys. Operational temperatures for science acquisition are required during each flyby and during calibration sequences throughout the mission. This concept was therefore discarded because it failed to meet the minimum science requirements. The large Winston cone passive cooler design did not fit within the volume envelope and mass constraints (see Fig. 20-21). At the conclusion

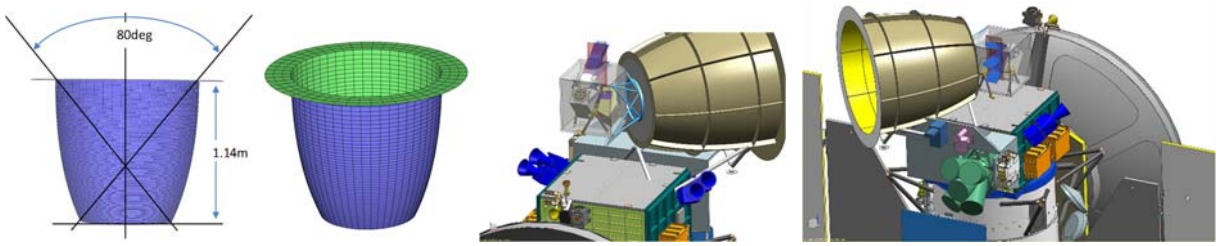


Figure 20. Winston cone passive cooler on MISE instrument.

of the passive cooling trade study, the MISE project was directed to consider active cooling and determine the mass, power, volume and other accommodation resources required to implement an active solution.

VII. Future Use of Solar Power for Space Application

While NASA continues to develop the radioisotope power system technology, it also continues to support and fund solar cell technology for future space use. It is anticipated that, in the forseeable future, solar cell technology will continue to be proposed to power spacecraft bound for the outer solar system. An example includes the NExt Mars Oribiter (NeMO) mission concept planned for launched in the early 2020's.[41-42] An artist conception of the NeMO spacecraft is shown in Figure 22. The moderate size spacecraft with a mission lifetime of 6.5 years requires 20 kW solar arrays to power the spacecraft, payload and the NEXT-C ion engines.[41-42] The launch date is currently planned for July 2022 with a low orbit science phase starting in July 2024. The Mars primary mission would be low orbit at 320 km altitude and 75-93 deg inclination with nadir pointing +/- 30 deg roll for science observation.[41-42] Clearly the size of the solar arrays required, as depicted in Figure 22, show that it will be a challenge to accommodate instruments requiring passive cryogenic cooling. Figure 22 also shows a concept for a Saturn orbiter with the required solar array size as shown in the figure. An Ultraflex 40 m^2 solar array produces 377 W and 335 W at Saturn beginning-of-life and end-of-life, respectively. Orbital attitude constraints along with the low altitude will make it difficult to accommodate a passive cryogenic cooler with a clear view to cold space for these type of missions.

VIII. Active Cooling Systems

Active cooling with mechanical cryocoolers below 100 K is feasible given sufficient electrical power and a room temperature radiator to reject the cryocooler and drive electronics waste heat. Pulse tube (PT) cryocoolers with flexure bearing

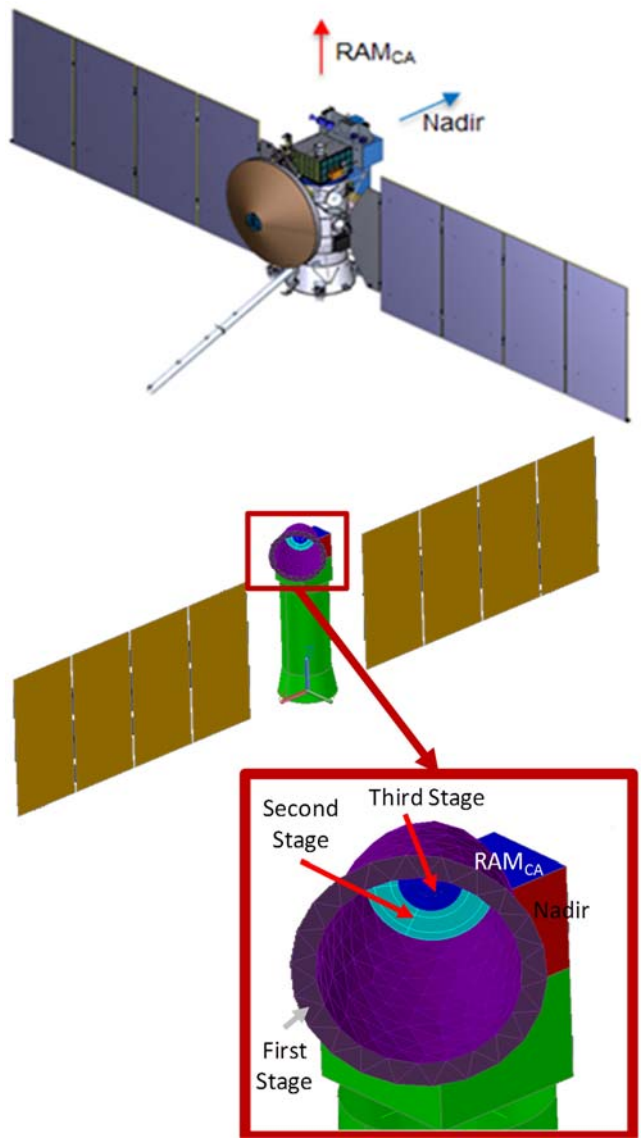


Figure 21. MISE instrument with Winston cone passive cooler.

technology have a long successful space flight history with some coolers operating continuously in space for nearly 15 years without performance degradation.[43-45]

Common design objectives of space cryocoolers include long life, high reliability, and high efficiency. The so-called "Oxford class" cryocooler emerged in the early 1990s as a widely applicable solution to these lifetime, reliability, and efficiency requirements. Though the original cryocooler designed by researchers at Oxford University was a Stirling cycle machine, the term "Oxford class" is now commonly taken to include both Stirling and PT cryocoolers that make use of a Stirling thermodynamic cycle and utilize flexure suspended, non-contacting pistons for the compression mechanisms. Both Stirling and PT coolers use a linear compressor with two opposing voice-coil actuated motors to cancel vibration. Flexure bearings on the linear motors maintain precise radial and axial alignment preventing piston contact with the cylinder walls. The flexures are designed for operation well below their fatigue limit leading to theoretically infinite life for these coolers, and no performance degradation is expected. A Stirling cooler requires a moving displacer at the cold head, whereas the PT achieves the same effect passively with an inertance tube and reservoir volume. Having no moving mechanisms, the PT cold head is more reliable and produces much lower levels of exported vibration than its Stirling counterpart.

PT cryocoolers have demonstrated successful operation in space in excess of 16 years without performance degradation.[43-45] From the early 1990s, JPL has advocated the use of PT coolers with the first set of coolers flown on AIRS in 2002, and two on TES in 2004, and one additional on OCO in 2008. Another PT cooler will also be used on OCO-3 to be launched in the near future and will fly onboard the International Space Station.

Recently, Oxford cooler technology has gained wider acceptance

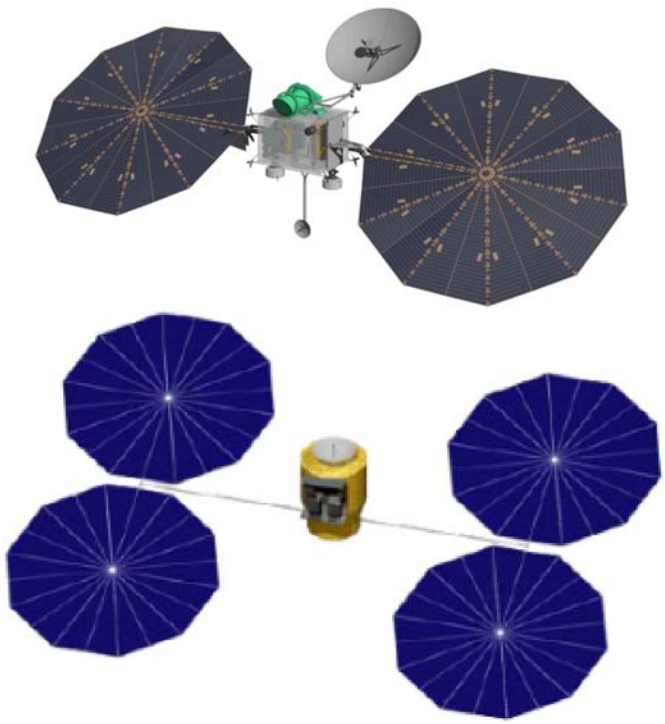


Figure 22. Top: NeMO Spacecraft concept with two-wing Ultra-Flex solar arrays (Ref. 42). Bottom: Saturn orbiter mission solar array sizing example for a Titan/ Enceladus orbit, 40 m^2 solar array, Ultraflex solar arrays MJ cell performance, 48 kW BOL at 1 AU, 377 W at Saturn arrival, 335 W EOL power (Ref. 41-42).

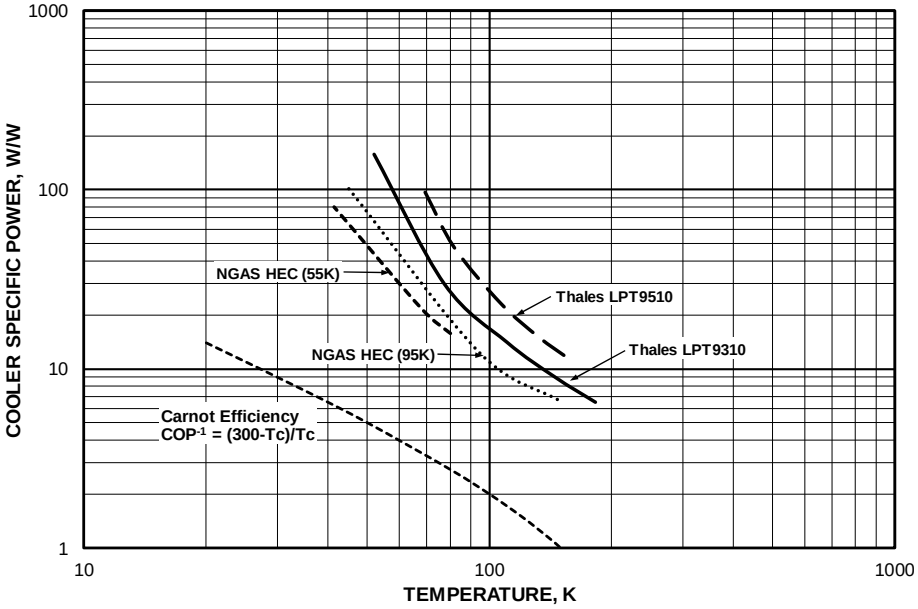
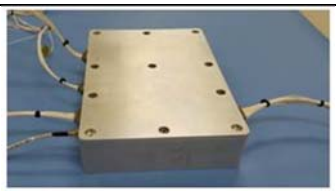


Figure 23. Cooler specific power as function of temperature: data for Thales Cryogenics and Northrup Grumman Aerospace Systems (NGAS) High Efficiency Coolers (HEC).

with low cost tactical cooler manufacturers as a viable option to increase the cooler reliability and life time. With the incorporation of the flexure bearing and the PT technologies, tactical cooler lifetimes have increased tenfold to in excess of 10 years while not appreciably increasing cost.[46] Thales Cryogenics (Eindhoven, The Netherlands) is one tactical cooler manufacturer that has chosen to invest in this technology and offer a low cost PT cooler.[46-51] Figure 23 shows the performance of the state-of-the-art (SOA) flight cryocoolers and the low cost Thales Cryogenics cryocoolers. The Carnot efficiency calculated in terms of specific power is shown at the bottom of the plot and represents the theoretical maximum performance. The cooler input power per watt of cooling (=Specific Power) range is highly dependent on cooler coldtip temperature as shown in the plot. The plot shows that these coolers have an order of magnitude lower performance compared to Carnot's efficiency. JPL has been working with this vendor for over six years to demonstrate flight worthiness of their PT coolers, as a means to provide low cost, reliable coolers to NASA instruments. Table 3 shows a comparison with cooling at 65K between the SOA and low cost options available from Thales Cryogenics. There are other cooler options available that are TRL 6 or above including the Lockheed Martin microcoolers.[52-55]

Table 3. Thales Cryogenics cryocooler compared to the SOA HEC from NGAS for an example application at 70 K with 1 W cooling (Ref. 43, 49-50).

NGAS HEC SOA Flight Cryocooler		Low Cost TRL6 Thales Cryocooler Option	
 Advanced Cryocooler Electronics (ACE)	3.8 kg 200 WAC (out) 3 in x 8 in x 7.8 in 82% efficient plus 12.4 W tare	 Iris Technology, High Power-Low Cost Cooler Electronics (HP-LCCE)	1 kg 200 WAC (out) 1.9 in x 5.6 in x 5 in > 86% efficient plus 1.7 W tare
 High Efficiency Cooler (HEC)	4.1 kg 200 WAC (input) 10 in x 4.5in x 12.1 in Specific power @ 65 K = 24 WAC/W *(65 K coldtip is required to cool detector to 70 K with thermal strap)*	 Thales LPT9310	7.1 kg 180 WAC (input) Compressor: 3 in x 8 in Cold head: 7.5 in x 3 in Specific power @ 65 K = 45 WAC/W *(65 K coldtip is required to cool detector to 70K with thermal strap)*
Power Input for 1 W of cooling at 65 K = 42 W Lifetime > 10 years, Total mass= 7.9 kg		Power Input for 1.0 W of cooling at 65 K = 54 W Lifetime > 10 years, Total mass= 8.1 kg	

IX. Summary

Passive cooling to cryogenic temperature below ~100 K on outer planetary mission with multiple large solar arrays is extremely challenging given the solar array sun-pointing constraints, payload science pointing requirements and spacecraft communication constraints. With all these constraints and warm solar arrays on two sides of the spacecraft, at a minimum, it is improbable that a clear field of view to cold space can be found to accommodate a cryogenic radiator with low spacecraft accommodation heat fluxes. Future outer planetary missions traveling to distances beyond Saturn at 10 AU would require even much larger solar arrays to generate sufficient power for the spacecraft and would make it much more challenging for cryogenic instrument designers wanting a passive solution. In these cases, it's very likely that active cooling with a mechanical cryocooler will be the only viable option available to instrument designers. With the recent advancements of cooler technology over the past 15 years, cryocoolers are now quite common in Earth missions and a few outer planetary missions. A passive solution, if it exsits, is of course the best solution if it does not place undue pointing constraints on the spacecraft attitude for the entire mission. A cryocooler solution is not without technical issues and will require additional resources including electrical power which may lead to having to increase the size of solar arrays to accommodate the additional power needed. If the mission is baselined with SEP, it's very likely that the additional power required by the cryocooler will be a small delta on the the total power subsystem. As noted previously, SEP systems typically require several kilowatts of power.

A heat rejection system with a room temperature radiator will be needed to dump the cryocooler and drive electronics waste heat to space. The active cooling solution removes the requirement of needing a clear field of view to cold space; however, it introduces other design challenges which will need to be addressed including exported vibration, EMI/EMC, and DC/AC magnetics fields. If active cooling is chosen, all cooler options need to be assessed against requirements and compatibility with the spacecraft to select the best cooler candidate option.

Acknowledgments

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References

[1]NASA/JPL Advanced Propulsion and Power: https://scienceandtechnology.jpl.nasa.gov/research/research-topics-list/spacecraft-technologies/advanced-propulsion-and-power.

[2]Wikipedia: https://en.wikipedia.org/wiki/Juno_%28spacecraft%29.

[3]Wikipedia: https://en.wikipedia.org/wiki/Rosetta_%28spacecraft%29.

[4]Ross, R. G. "Aerospace coolers: A 50-year quest for long-life cryogenic cooling in space," In Cryogenic Engineering, pp. 225-284. Springer New York, 2007.

[5]Rodriguez, J.I., Tseng, H., Zhang, B., "Thermal Control System of the Moon Mineralogy Mapper Instrument," 38th SAE Tech. Paper Series, Paper No. 2008-01-2119, San Francisco, CA, July 2008.

[6]Donabedian, J., "Thermal Uncertainty Margins for Cryogenic Sensor Systems," Aerospace Corporation, AIAA-1991-1426-764.

[7]Rodriguez, J.I., Tseng, H., Varanasi, P., Zhang, B., "On-Orbit Performance of the Moon Mineralogy Mapper Instrument," 39th SAE Technical Paper Series, Paper No. 2009-01-2390, Savannah, Georgia, July 2009.

[8]Rodriguez, J. I. and Na-Nakornpanom, A., "On-Orbit Performance of the TES Cryocooler System and Instrument – Six Years in Space," Adv. In Cryogenic Engin., Vol. 56b, Plenum Press, New York, NY, 2011.

[9]Rodriguez, J. I. and Ross, R. G. Jr., "AIRS On-Orbit Cryocooler System Performance," SPIE Proceedings, Vol. 5151, Earth Observing Systems VIII, Barnes, W.L., ed., pp. 220-231, November 2003.

[10]Swanson, K. S., Leader, F. E., "Preparation for Cassini Grand Finale – Validation of Saturn Ring Heating in TSS," 46th International Conference on Environmental Systems, ICES-2016-34, July 2016, Vienna, Austria.

[11]NASA EOS Missions: https://eospso.gsfc.nasa.gov/.

[12]Donabedian, M., ed., "Spacecraft Thermal Control Handbook: Vol II: Cryogenics," Figure 6.13.

[13]Franck, R. A., Gurule, A. P., Brinckerhoff, P. A., McCallan, T. R., Renbarger, M., Brown, A. A., "High Performance Cryogenic Radiators for James Webb Space Telescope," 46th International Conference on Environmental Systems, ICES-2016-141, Figure 11, July 2016, Vienna, Austria.

[14]ESA Fact Sheet, Nov 2014, Rosetta Mission:
http://download.esa.int/esoc/shares/1/Rosetta_PressKit_CometLanding_November2014_v4.pdf

[15]NASA Press Kit, Dawn at Ceres: http://dawn.jpl.nasa.gov/news/pdf/Dawn-Ceres-presskit_20150227.pdf

[16]NASA Fact Sheet, Dawn: http://dawn.jpl.nasa.gov/mission/dawn_fact_sheet.pdf.

[17]NASA Media Site: http://dawn.jpl.nasa.gov/DawnMedia/index.asp.

[18]NASA/JPL Dawn Mission: http://dawn.jpl.nasa.gov/.

[19]Wikipedia: https://en.wikipedia.org/wiki/Dawn_%28spacecraft%29.

[20]Thomas, V. C., Makowski, J. M., Brown, G. M., McCarthy, J. F., Bruno, D., Cardoso, J. C., Chiville, W. M., Meyer, T. F., Nelson, K. E., Pavri, B. E. and Termohlen, D. A., "The dawn spacecraft. Space science reviews, 163, No. 1-4, 2011, pp.175-249.

[21]Benson, S. W., "Solar Power for Outer Planets Study, "Presentation to Outer Planets Assessment Group," NASA Glenn Research Center, Nov. 2007.

[22]Beauchamp, P., Ewell, R., Brandon, E., Surampudi, R., "Solar Power and Energy Storage for Planetary Missions," JPL Presentation, Aug. 2015.

[23]NASA Fact Sheet: Solar Electric Propulsion:
http://bigidea.nianet.org/wp-content/uploads/2016/10/ps-01987_sep_fact_sheet-21.pdf.

[24]NASA Europa Mission: http://europa.jpl.nasa.gov/home.cfm.

[25]NASA Solar System Exploration, Europa: Overview: Ingredients for Life? http://solarsystem.nasa.gov/planets/europa.

[26]NASA/JPL Europa Mission: https://www.jpl.nasa.gov/missions/europa-mission/.

[27]NASA Mission to Europa: https://www.nasa.gov/europa.

[28]Wikipedia: https://en.wikipedia.org/wiki/Europa_Multiple-Flyby_Mission.

[29]Coradini, A., Capaccioni, F., Capria, M.T., Cerroni, P., De Sanctis, M.C., Magni, G., Bonsignori, R., Reininger, F., Encrenaz, T., Drossart, P. and Semery, A., "VIRTIS, Visible Infrared Thermal Imaging Spectrometer for the ROSETTA Mission," In Geoscience and Remote Sensing Symposium, 1995, IGARSS'95.'Quantitative Remote Sensing for Science and Applications', International, Vol. 2, pp. 1604-1606, IEEE.

[30]De Sanctis, M. C., Coradini, A., Ammannito, E., Capria, M.T. and Fonte, S., "VIR, The Imaging Spectrometer for the Asteroid Belt Exploration," Memorie della Societa Astronomica Italiana, 2009; Vol. 80, p.295-300.

[31]Jet Propulsion Laboratory, California Institute of Technology, Juno Orbit Insertion – Press Kit, https://www.jpl.nasa.gov/news/press_kits/juno/.

[32]Bolton, S. J., and Juno Team, "The Juno Mission," International Workshop on Instrumenation for Planetary Missions (2012).

[33]Grammier, R. S., "A Look Inside the Juno Mission to Jupiter," Aerospace Conference, 2009 IEEE, pp. 1-10.

[34]Dodge, R., Boyles, M. and Rasbach, C., "Key and Driving Requirements for the Juno Payload Suite of Instruments," AIAA SPACE 2007 Conference & Exposition, p. 6111, 2007.

[35]NASA Fact Sheet, Juno Mission to Jupiter: https://www.jpl.nasa.gov/news/fact_sheets/JUNO_Fact_Sheet_2016.pdf.

[36]NASA/JPL Mission to Jupiter; Juno: https://www.jpl.nasa.gov/missions/juno/.

[37]Adriani, A., Coradini, A., Filacchione, G., Lunine, J. I., Bini, A., Pasqui, C., Calamai, L., Colosimo, F., Dinelli, B. M., Grassi, D., Magni, G., Moriconi, M. L., Orosei, R., "JIRAM, the Image Spectrometer in the Near Infrared on Board the Juno Mission to Jupiter," Astrobiology 8, 2008, No. 3, p613-622.

[38]Adriani, A., Filacchione, G., Di Iorio, T., Turrini, D., Noschese, R., Cicchetti, A., Grassi, D., Mura, A., Sindoni, G., Zambelli, M. and Piccioni, G., "JIRAM, the Jovian Infrared Auroral Mapper," Space Science Reviews, pp.1-54, 2014.

[39]Soucy, M. A., Chateauneuf, F., "Status of the ACE-FTS Instrument On-board SciSat-1," Mirror, Vol. 11, p. 12.

[40]Soucy, M. A., Walker, K. A., Fortin, S., and Deutsch, C., "ACE-FTS Instrument: activities in preparation for launch," In Optical Science and Technology, SPIE's 48th Annual Meeting, International Society for Optics and Photonics, (pp. 143-152).

[41]Wikipedia: https://en.wikipedia.org/wiki/Mars_2022_orbiter.

[42]Jedrey, T., Lock, R., Matsumoto, M., Conceptual Studies for the Next Mars Orbiter (NeMO), Industry Day, May 2, 2016: http://images.spaceref.com/news/2016/NeMOIndustryDay.pdf.

[43]Raab, J. and Tward, E., "Northrop Grumman Aerospace Systems Cryocooler Overview," Cryogenics 50 (2010) 572-581.

[44]Ross, R.G. and Johnson, D.L., "Effect of Gravity Orientation on the Thermal Performance of Stirling-type Pulse Tube Cryocoolers," Cryogenics 2004; 44: 403-408.

[45]Ross, R.G., Jr., Johnson, D.L., Metzger, A., Kotsubo, V., Evtimov, B., Olson, J., Nast, T., and Rawlings, R.M., "Gamma-Ray Pulse Tube Cooler Development and Testing," Cryocoolers 11, Kluwer Acedemic/Plenum Publishers, New York, 2001, 155-162.

[46]van de Groep, W., van der Weijden, H., van Leeuwen, R., Benschop, T., Cauquil, J. M., Griot, R., "Update on MTTF figures for linear and rotary coolers of Thales Cryogenics", in Infrared Technology and Applications XXXVIII, Bjørn F. Andresen; Gabor F. Fulop; Paul R. Norton, Editors, Proc. of SPIE Vol. 8353, pp. 83531S-1 to S-15, 2012.

[47]Freeman, J.J., Murphy, J.B., Kirkconnell, C.S., "Experimental Demonstration of Cryocooler Electronics with Multiple Mechanical Cryocooler Types," SPIE Proceedings Volume 8353: Infrared Technology and Applications XXXVIII, p. 83531W-1 (2012).

[48]Freeman, J., et. al., "Integrated Testing of the Iris LCCE and NGAS Micro Pulse Tube Cooler," Cryocoolers 17, proceedings of the 17th ICC held in Los Angeles, California, on July 9-12, 2012, Ed. By S.D. Miller and R.G. Ross Jr., ICC Press, Boulder CO, 2012.

[49]Cryogenic Cooler LPT 9310/17 Specification, Thales Cryogenics, June 17, 2013.

[50]Johnson, D.L., McKinley, I.M., Rodriguez, J.I., Tseng, H., Carroll, B.A., "Characterization Testing of the Thales LPT9310 Pulse Tube Cooler," *Cryocoolers 18*, Kluwer Academic/Plenum Publishers, New York (2014), pp. 125-133.

[51]Johnson, D.L., Rodriguez, J.I., Carroll, B.A. Bustamante, J.G., Kirkconnell, C.S., Luong, T.T., Murphy, J.B. and Haley, M.F., "Integrated Testing of the Thales LPT9510 Pulse Tube Cooler and the Iris LCCE Electronics," Adv. in Cryogenic Engineering, Vol 59B, American Institute of Physics, NY, 2014, pp. 1806-1813.

[52]McKinley, I. M., Johnson, D. L., Rodriguez, J. I., "Characterization Testing of Lockheed Martin Micro Pulse Tube Cryocooler," Cryocoolers 19, ICC Press, Boulder, CO (2016).

[53]Nast, T., Roth, E., Olson, J., Champagne, P., Frank, D., "Qualification of Lockheed Martin Micro Pulse Tube Cryocooler to TRL6," Cryocoolers 18, ICC Press, Boulder, CO (2015).

[54]Olson, J., Champagne, P., Roth, E., Nast, T., Saito, E., Loung, V., Kenton, A., Dobbins, C., "Microcryocooler for tactical and space applications," Adv. In Cryogenic Engineering, Vol. 59, Amer. Institute of Physics, Melville, NY (2014).

[55]Nast, T., Olson, J., Champagne, P., Roth, E., Saito, E., Loung, V., McCay, B., Kenton, A., Dobbins, C., "Development of Microcryocoolers for Space and Avionics Applications," Cryocoolers 19, ICC Press, Boulder, CO (2016).