

ELECTROCHROMIC RADIATORS FOR MICROSPACECRAFT THERMAL CONTROL

Anthony Paris, Kevin Anderson
 Jet Propulsion Laboratory, California Institute of Technology
 4800 Oak Grove Dr, Pasadena, CA 91109-8099; 818-393-6732
 Anthony.D.Paris@jpl.nasa.gov, Kevin.R.Anderson@jpl.nasa.gov

Prasanna Chandrasekhar, Brian Zay, Terrance McQueeney
 Ashwin-Ushas Corporation, Inc.
 500 James Street, Suite 7, Lakewood, NJ 08701; 732-462-1270
 chandra.p@ashwin-ushas.com, zay.b.2@ashwin-ushas.com, mcqueeney.t@ashwin-ushas.com

ABSTRACT: Limitations on electrical power for survival heating and reduced thermal mass can often lead to challenges in maintaining allowable flight temperature limits and ensuring temperature stability for microspacecraft hardware. To address these thermal issues, technologies such as variable-emittance thermal radiators based on thin-film electrochromic materials are being investigated at the Jet Propulsion Laboratory (JPL) for microspacecraft applications. Electrochromic materials feature the ability to alter their reflectance in the infra-red wavelengths in response to changes in an applied bias voltage. As a result, thin-film electrochromic materials can be packaged as low-mass, low-power devices that possess the same thermal control functionality as conventional mechanical louvers. The present paper details work done at JPL to develop a microspacecraft thermal control architecture featuring an electrochromic radiator as the primary thermal control device. The microspacecraft application is an ultra-low mass (less than 3 kg) free-flying microsatellite designed for host-vehicle inspection. An electrochromic surface developed by Ashwin-Ushas Corporation is used to control the waste heat lost to space by modulating the effective emittance of the thermal radiator.

INTRODUCTION

While miniaturized or small form-factor spacecraft have the potential to offer increased capabilities and decreased costs for future space missions, system design for these spacecraft can often be quite challenging. This is particularly true for the design of the spacecraft's thermal control system. Whereas the goal of any thermal control system is to maintain all spacecraft hardware within allowable temperature limits during all phases of a mission, the means by which this may be accomplished can differ significantly based on the size of the spacecraft and the allocations for mass, area, and power available to the thermal control system.

Microspacecraft thermal control design differs significantly from that for traditional, larger spacecraft for several reasons. First of all, it is not uncommon for there to be severe limitations on the amount of electrical power available for survival heating of microspacecraft components. For small spacecraft designed to subsist only on battery power or that possess limited exterior area for housing solar cell

arrays, the majority of the electrical power available is likely to be budgeted for avionics and/or instrumentation. The consequence of such power limitations on the thermal control system is that traditional thermal architectures employing a combination of insulation (*e.g.* low emissivity surfaces, multi-layer insulation blankets, *etc.*) and electrical survival heaters may be impractical. This is particularly true for microspacecraft spacecraft that have instruments requiring direct views to deep space (*e.g.* cameras) or that have one or more components that may overheat when well insulated (*e.g.* lasers.) In these cases, such components will necessarily reject heat directly to deep space and create a pathway for heat to leak continuously from the spacecraft. Consequently, the heat lost from these components when not operating must be balanced by direct heating from electrical survival heaters to maintain the components at temperature.

Another factor that complicates microspacecraft thermal control is the low thermal capacity inherent in most low mass spacecraft. A low thermal capacitance may subject a spacecraft to large temperature swings

when changes occur in the internal heat generation rate, the thermal sink temperature, or irradiation loads from solar or infra-red sources. If the primary power dissipation mode on these spacecraft lasts only a small fraction of the mission time, the thermal control system must not only remove waste heat, but also conserve energy and maintain component temperature control. To complicate matters, a thermal control system with low thermal capacitance may need to respond rapidly to changing environmental conditions to prevent pronounced swings in the hardware temperatures.

The highly integrated nature of microspacecraft hardware may also complicate the thermal control system. As a means to miniaturize various systems and decrease mass, some microspacecraft may employ structures or components that serve multiple purposes or combine functionality. For example, an avionics package may combine structural, electronic packaging, and thermal control functions within one monolithic unit. While at times such tight integration of systems or hardware may ease the burden of thermal control (*e.g.* it may be easier to spread waste heat from avionics chips), it is also possible for tightly integrated or miniaturized components to make thermal control more difficult. When tightly integrated systems combine components with wide, but differing, temperature limits, each individual component may be forced to share a much tighter common maximum and minimum temperature limit due to proximity or common packaging. Additionally, in even low power systems, the power densities of miniaturized electronics, instrumentation, and avionics may exceed those of traditional spacecraft by an order of magnitude or more. While the total waste heat from such systems would not be excessively large, advanced thermal control hardware may be needed to manage the high heat fluxes present.

At present, only mature thermal technologies such as heaters, thermostats, heat pipes, mechanical louvers, *etc.* are considered sufficiently reliable for use in the majority of thermal control systems. However, these technologies are not always adequate in either performance or resource consumption for use on microspacecraft. Most mature thermal control technologies were designed for large spacecraft systems and tend to be too massive, too power consumptive, and too difficult to scale to very small sizes. In order to meet the unique thermal challenges presented by microspacecraft hardware, new thermal control technologies and architectures are needed. In general, they must allow for active control, possess low mass, require little power, and integrate well with miniaturized systems. The present work seeks to develop such a technology by adapting thin-film variable-emittance radiator technology for microspacecraft applications.

A number of researchers have identified the controlled variance of thermal radiator optical finishes as an attractive technology for development¹. This is largely due to the fact that a commanded variation in the emissivity of a radiator can modulate the heat lost to space while maintaining a constant temperature of the radiating surface. This is demonstrated by the following equations. The total heat emitted by a radiator surface is given by:

$$Q = A\epsilon\sigma T^4 \quad (1)$$

where A is the surface area, ϵ is the emittance, σ is the Stefan-Boltzmann constant, and T is the absolute temperature of radiating surface². If the emissivity of the surface is varied, the above equation can be used to describe the before (1) and after (2) states as,

$$Q_1 = A \epsilon_1 \sigma T_1^4 \quad (2)$$

and

$$Q_2 = A \epsilon_2 \sigma T_2^4 \quad (3)$$

If the radiated heat loads for both cases are equivalent ($Q_1=Q_2$), the relationship between the ratios of emissivity and radiator temperatures is as follows:

$$T_2/T_1 = (\epsilon_1/\epsilon_2)^{0.25} \quad (4)$$

If the temperature of the radiator, and likewise its conductively coupled components, is held constant, then the heat loss from the radiator in the changed state is directly proportional to the ratio of the emissivities:

$$Q_2 = (\epsilon_2/\epsilon_1)Q_1 \quad (5)$$

The result is that the total amount of heat radiated to space may be modulated while the radiator maintains the same temperature. For variable-emittance radiators with low-emissivity states in the 0.1 to 0.2 emissivity range and a high emissivity state near 0.7, a change in the radiative heat loads can be effected up to five times the original value. The magnitude of this change indicates that a variable-emittance radiator with a similar emittance range could be used as a heat switch to modulate the heat loads lost through a radiator.

Mechanical louvers have long been used to provide such variable-emittance functionality for large spacecraft radiators. However, traditional mechanical louvers have a number of disadvantages that become pronounced when incorporated on spacecraft with extreme volume and mass constraints. These include complicated mechanical assemblies, reliance on bimetallic strip actuation, solar trapping, and operational hysteresis². A number of emerging

technologies seek to provide variable-emittance capability for spacecraft without the drawbacks associated with these complicated and bulky mechanical systems. Many of these variable-emittance technologies are being developed internally by private industry and through DOD and NASA sponsored Small Business Innovation Research (SBIR) grants.

One approach to miniaturizing variable-emittance capability is to preserve the operational concept of louvers but greatly decrease the size of the mechanical systems. These micro-scaled mechanical louver assemblies or “microlouvers,” are fabricated with Microelectromechanical Systems (MEMS) machining techniques³. While microlouvers have the potential to offer a range of emittance-variation on par with large louver systems (e.g. from 0.2 to 0.8), the technology to date has demonstrated a smaller effective range ($\Delta\epsilon$ of approximately 0.3.)⁴ Another approach uses nominally high emissivity films that are physically offset from radiator surfaces to form a radiation barrier. When this high emissivity exterior surface is collapsed onto the radiator surface (e.g. by electrostatic forces) the radiator switches states, from a low to high effective emissivity⁵. NASA’s ST-5 mission will demonstrate both these approaches (MEMS micro-louvers and electrostatically-actuated louvers) when it is launched in early next year⁶ (2006.)

Another approach to developing variable-emittance radiators is to utilize electroactive materials that show a reversible optical property change when an electric field is applied. A number of these ‘electrochromic’ materials may be processed into extremely light weight thin-films or coatings. A radiator coated with an electrochromic material has the potential to offer the same variable-emittance capability as a mechanical louver along with decreased mass, cost, and mechanical complexity. Due to these attributes, radiators based on electrochromic films and devices show great promise for microspacecraft applications.

ELECTROCHROMIC DEVICE TECHNOLOGY

Electrochromic devices for spacecraft thermal control are based upon materials that change their reflectance in the infra-red (IR) wavelengths upon oxidation or reduction. When a small bias voltage is applied to such a device, charge migration occurs within the electrochromic material and electrons are either collected or removed. This change results in a modification in the intrinsic infra-red reflectance of the material. An electrochromic device is composed of a number of layers which behave much like the anode, cathode, electrolyte and mutual electrodes in a battery. When a bias voltage is applied to the anode- and cathode-like layers, charge is transferred by means of the electrolyte. In general, devices made from these

materials have the ability to change the visible and IR reflective and/or transmissive characteristics in the visible to near-IR wavelengths (0.4 to 1.1 μm) and/or IR wavelengths (2 to 45 μm).

Ashwin-Ushas Corporation of Lakewood, NJ is presently developing electrochromic devices based on conducting polymers. Details on the electrochemistry of these devices and performance descriptions may be found in the reference literature^{7,8}, but a short summary is provided here for the reader. The IR signature variation observed in these conducting polymer (CP) electrochromic devices is the result of the unique polymer/dopant combination and the design of the device. Figure 1 shows a schematic of an Ashwin-Ushas “dual-electrode” device design (US Patent 6,033,592). Each electrode is composed of three layers: the conducting polymer/dopant layer, a thin gold layer, and a microporous membrane. The front or working electrode is the surface that is visible to an observer or external environment while the back electrode serves as a counter electrode to the front. The electrodes are stacked such that the microporous membrane of the top electrode is in contact with the conducting polymer layer of the back electrode. Once assembled, the microporous membrane is filled with an ionic electrolyte—a room temperature molten salt—that is liquid from -100°C to + 280°C and has a very low vapor pressure. Capillary action causes the electrolyte to travel through the pores of the microporous membrane allowing backside electrochemical contact with the conducting polymer surface on the front electrode. The presence of the electrolyte allows for charge migration between the two electrodes once an electric potential is applied.

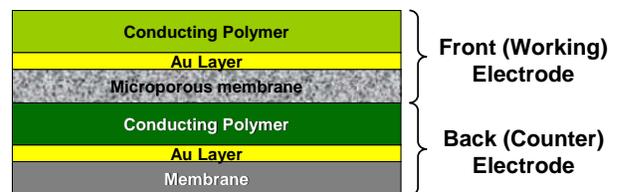
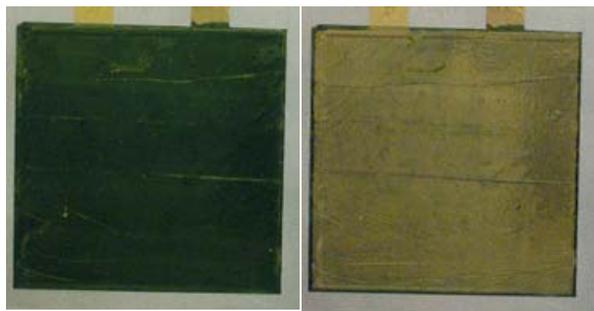


Figure 1. Schematic of a dual-electrode electrochromic device (Ashwin-Ushas Corp., Inc.)

The IR signature modification in the conducting polymer stems from reduction-oxidation reactions occurring in a complementary fashion at the working and counter electrodes. When a negative potential of approximately -1.0 V is applied to the electrodes, the conducting polymer of the front electrode becomes substantially IR-transparent (called the ‘light’ state) the highly IR-reflective gold layer with corresponding low emissivity is exposed. When a small positive potential of approximately +0.2 V is applied, the conducting polymer of the front electrode enters a ‘dark’ state and becomes IR-absorbing with a high

infra-red emissivity. Although the back electrode operates counter to the front electrode (absorbing and donating charge as needed), the state of the conducting polymer in the back electrode does not influence the emissivity of the device as its view to the environment is obscured by the front electrode.

A sample of an electrochromic device fabricated by Ashwin-Ushas of New Jersey is shown in Figure 2 below. As seen in by the color shift in these photos, the devices also show a change in the visible spectrum along with the IR wavelengths. A typical variation in the IR emissivity of these devices is in the range from 0.3 to 0.75, with a maximum measured $\Delta\epsilon$ of 0.55. Development is currently underway to broaden this change to 0.2 to 0.8, a range exceeding that of current mechanical louvers. Aside from their variable-emittance capability, the electrochromic devices developed and manufactured by Ashwin-Ushas Corp possess a number of attractive features for spacecraft thermal control. The maximum and minimum emissivity states are tailorable, from 0.15 on the low side, up to approximately 0.85 on the high side. The devices are comprised of thin film materials and as assembled have thicknesses on the order of hundreds of microns. The devices are extremely low-weight (substantially less than one gram per square cm) and feature no moving parts. Switching times, as defined as the time interval between 10% and 90% of steady-state emissivity values, are as low as 3 seconds at room temperature and under one minute at -35°C . Activation voltages on the order of one volt are well within the range provided by most spacecraft electrical buses and the power draw is low. Peak transient power consumption is on the order of few milliwatts per square centimeter during device state change and steady-state power consumption is on the order of tens of microwatts per square centimeter.



Dark State (+0.2 V)

Light State (-1.0 V)

Figure 2. Dual-Electrode electrochromic device (Ashwin-Ushas Corp., Inc.)

MICROSPACECRAFT APPLICATION

Researchers at the Jet Propulsion Laboratory (JPL) and Ashwin-Ushas Corporation are working to develop electrochromic device technology for use on a microspacecraft application funded by NASA's Exploration Systems Mission Directorate (ESMD). ESMD is seeking to develop novel system concepts to enable safe, affordable and effective human and robotic exploration. In particular, the agency is interested in maturing technologies that enable in-space assembly, servicing, and maintenance. As a part of this effort, researchers at JPL are investigating the use of microspacecraft for future ESMD missions and seek to mature the technology of highly miniaturized spacecraft for the purpose of remote inspection. This inspection microspacecraft, or "micro-Inspector", will extend the reach of human explorers by providing safe inspection of space operations over a range of host spacecraft, applications, and destinations. While specific mission scenarios and performance requirements are being developed for this application, a basic system architecture has been established based upon a pre-cursor microspacecraft project at JPL (the Low Cost Adjunct Microspacecraft or LCAM.)⁹ The general form of the LCAM microspacecraft and a depiction of host inspection and is shown in Figure 3.

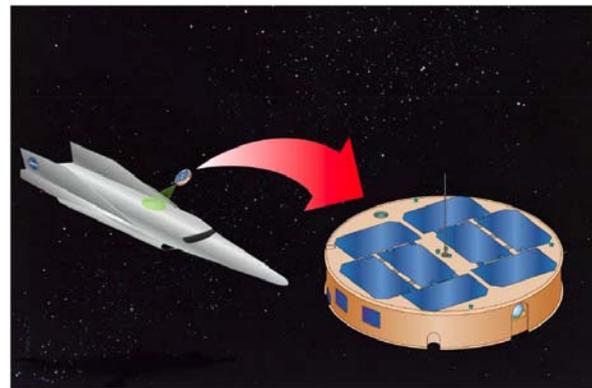


Figure 3. Depiction of the Low Cost Adjunct Microspacecraft (LCAM)⁹

The micro-Inspector spacecraft is conceived to be a deployable, mobile camera platform used primarily to inspect exterior surfaces of a host spacecraft¹⁰. The micro-Inspector will be small and compact (less than 3 kg and 25 cm³) in order to minimize cost and reduce integration complexity with the host vehicle. As a result, the amount of exterior surface area available for solar cells will be small and the amount of generated electrical power will be likewise limited. The salient features of the micro-Inspector from a thermal control perspective are its limited power budget for survival heating, its liquid butane-based propulsion system, and a design requirement to reclaim waste heat from the

avionics and instruments to provide thermal energy to the propulsion system. Figure 4 shows a conceptual view of the micro-Inspector spacecraft. As this figure indicates, the micro-Inspector is essentially a stacked assembly of three major components: a solar array; a circuit board populated with avionics, cameras, and thrusters; and a tank for the butane propellant. In addition to housing the butane, the propellant tank also serves as the main structure and thermal radiator for the spacecraft.

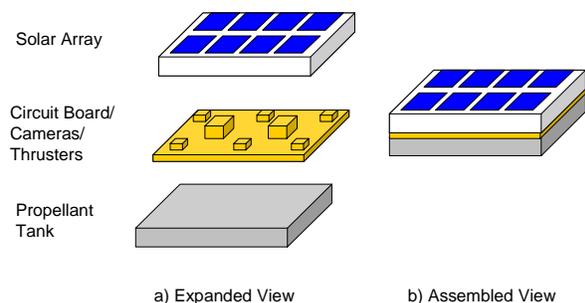


Figure 4. Conceptual drawing of the micro-Inspector assembly

The thermal control system of the micro-Inspector must perform a number of functions including maintaining all avionics/instruments/batteries within allowable temperatures, preventing the densely packed solar array from overheating when exposed to the sun, ensuring a conductive pathway from the avionics and instruments to the propellant tank, and balancing the waste heat load so that the butane propellant does not become over-pressurized. Furthermore, the thermal control system must perform adequately when exposed to a range of disparate thermal environments, since any of the exposed surfaces of the spacecraft may have transient views to the sun, deep space, planetary bodies, or the host vehicle during the course of its inspection.

If either ample electrical heater power or broad temperature limits existed for this application, the design of the thermal control system would be fairly straightforward. The bottom exterior surface of the propellant tank would be designed to have a high emissivity and sized so that it could reject a maximum heat load (*e.g.* all of the heat generated by the payload and parasitic heat loads transferred from the warm solar array) without overheating the spacecraft. For conditions when the payload was not powered or when the solar array was not pointing directly at the sun, the heat load on the propellant tank radiator would be supplemented by electrical survival heaters to maintain its minimum allowable temperature. The amount of this supplemental heating would depend upon the tank's minimum allowable temperature. Unfortunately, this thermal control approach is not applicable for the micro-Inspector because relatively

large amounts of excess electrical heater power are not available and the minimum allowable temperature for the propellant tank is rather high in practice.

The propellant system has a high minimum allowable temperature due to the need for a saturated liquid-vapor butane mixture in the propellant tank. The condensation point for iso-butane is -11.7°C , and this temperature sets an absolute lower limit for the propellant tank. Temperatures lower than this would result in all of the butane in the tank condensing to liquid form with no butane vapor available to the thrusters. For the propulsion system to provide a sufficient level of thrust, the butane vapor must also be kept at a minimum pressure. For this reason the tank must actually be kept above a minimum of 0°C which corresponds to a vapor pressure of approximately 15 psia (103.4 kPa). Applying margin to this lower temperature limit results in a minimum allowable flight temperature for the tank of $+15^{\circ}\text{C}$.

If all the micro-Inspector components had high maximum allowable temperatures (*e.g.* greater than 60°C), then the spacecraft could be insulated with multi-layer blankets and allowed to run hot when operating. However, the lithium-ion batteries chosen for this spacecraft have a maximum allowable temperature of only $+30^{\circ}\text{C}$. Since the batteries, circuit board, and propellant tank are thermally coupled in this compact spacecraft to foster waste heat utilization, the allowable temperature range for the entire tank and circuit board system is only $+15$ to $+30^{\circ}\text{C}$. Thus, the thermal control system must be able to reject the full heat load of the micro-Inspector when the butane system is primed, conserve this heat when it is needed by the propulsion system to vaporize liquid butane, and maintain the spacecraft at $22.5 \pm 7.5^{\circ}\text{C}$ —while in the process consuming very little electrical power itself.

As previously noted, the electrochromic devices from Ashwin-Ushas Corporation are low-power, active devices that can vary infrared-emissivity values by a factor of four or more (*e.g.* from 0.15 to 0.7.) Since emissivity is directly proportional to heat loss at a given radiator temperature, thermal control of the micro-Inspector may be achieved by applying an electrochromic device to the underside surface of the propellant tank to serve as the primary thermal radiator. The electrochromic radiator would reject the waste heat load from the spacecraft in its nominal high emissivity state and switch to a lower emissivity state to conserve a portion of this waste heat as needed for vaporizing the butane propellant. With its ability to be controlled by the flight computer and the relatively rapid switching of emissivity states, the electrochromic device can also be used to compensate for the low thermal capacity of the micro-Inspector by continually adjusting the heat load thermal balance.

To maximize the efficiency of the waste heat modulation by the electrochromic device, the total heat

load rejected by the radiator will be minimized. Although the waste heat from the avionics and instruments is largely determined by the maximum electrical power generated by the solar array at any given time, the parasitic heat load from the warm solar array can be mitigated through thermal design. The exposed surfaces of the solar array structure will be coated with a low solar absorptivity white paint to minimize incident solar heating. Additionally, the heat transfer between the solar cell cover and the rest of the spacecraft will be minimized by the combination of thermally insulating structural stand-offs and a low-emissivity coating on the interior of the solar array structure (*i.e.* the surface facing the circuit board.) By minimizing the parasitic heat loads to and from the solar array in the cold and hot conditions, the electrochromic device radiator will be used to primarily manage the avionics and instrument waste heat loads.

To guide the development and design of the electrochromic device, a simple thermal model of the micro-Inspector was developed. Thermal mass nodes were created for the solar array panel; the circuit board, cameras, and thrusters; and the propellant tank. Thermally conductive paths between the solar array and the circuit board and the circuit board and the tank were modeled based on the properties of insulating fiberglass structural stand-offs and thermal gasket material, respectively. Radiation exchanges between the interior solar array and circuit board, camera/thruster and deep space, and propellant tank and deep space were also included. A solar array area of approximately 220 cm² was included in the model along with associated electrical power generation from 0 to 7.5 W depending on the orientation of the spacecraft with respect to the sun. The electrochromic device was modeled with a surface area of approximately 170 cm² and with a high-state emissivity of 0.7. The underside area of the propellant tank not covered by the electrochromic device (approximately 150 cm²) was given a constant emissivity of 0.05, similar to that of highly polished metals. Results of this model are shown in Figures 5 and 6.

Both figures show the magnitude of the relevant heat loads and the temperature of the propellant tank for a number of spacecraft sun angles. Figure 5 shows the baseline design case of an electrochromic device equipped propellant tank radiator. For off-sun angles up to about 20 degrees, there is no need to modulate the emissivity of the electrochromic device in order to maintain the propellant tank temperature at 27°C. Since the absorbed energy on the solar array follows a cosine law with respect to the off-sun angle, there is very little change in the absorbed or rejected heat loads for these small angles. The avionics power load remains constant through these angles (and less than

the maximum 7.5 W generated) because a portion of the power is reserved for recharging batteries.

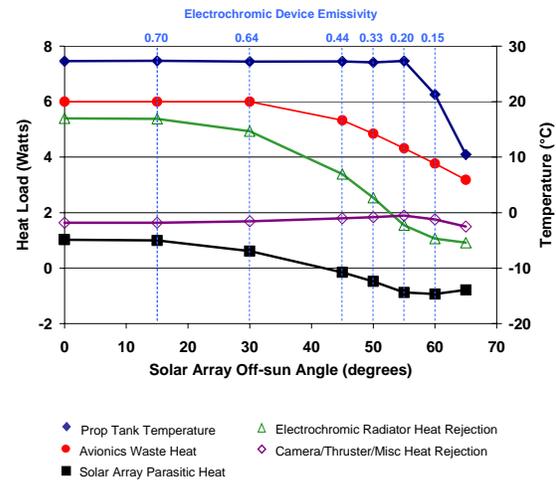


Figure 5. Thermal modeling results for the micro-Inspector as equipped with an electrochromic radiator

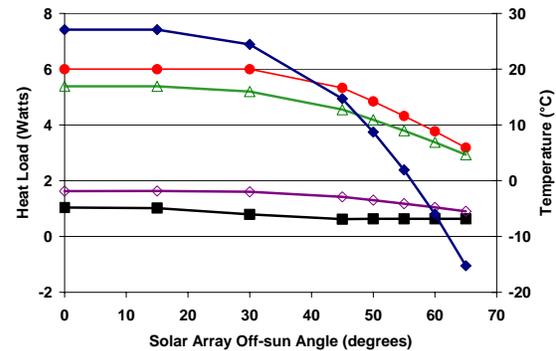


Figure 6. Thermal modeling results for the micro-Inspector as equipped with a constant emissivity radiator (see Fig 5 for chart legend)

At angles beyond 30 degrees, the amount of power generated by the solar array begins to lessen and, thus, the power available to the avionics also decreases. It is beyond this orientation where the electrochromic device is used to modulate the emissivity and decrease the amount of heat lost through the radiator. By decreasing the emissivity in a continuous manner from 0.7 to 0.2, the Electrochromic device can maintain the propellant tank at 27°C while the avionics power consumption drops by approximately 70%. Beyond a 55 degree off-sun angle, the Electrochromic device has reached its lower emissivity limit, and the propellant tank temperature

begins to drop. However, the tank remains above its lower allowable temperature limit of 15°C even after the spacecraft moves beyond 60 degrees off-sun.

In contrast, Figure 6 shows results from the model with a constant 0.7 emissivity propellant tank radiator replacing the electrochromic device. The results are identical for the low off-sun angles where emittance-variation is not employed. However, once off-sun angles eclipse 15 degrees, the propellant tank temperature begins to drop rapidly. At approximately 40 degrees, the tank has reached its lower allowable temperature. Beyond this point and without using battery power reserves, the micro-Inspector would not be able to generate enough pressure in the butane propulsion system to operate the thrusters and correct its attitude with respect to the sun. While this simple steady-state model is of limited fidelity, it indicates that the electrochromic device has the potential to extend the steady-state lifetime of the spacecraft over a desirable range of off-sun angles.

ELECTROCHROMIC DEVICE DEVELOPMENT

In most all cases, emerging thermal control technologies require development to demonstrate that they may function in a reliable way in a spacecraft system. Apart from thermal performance, these technologies must demonstrate durability in space-like environment that may include vacuum exposure, large magnitude thermal cycling, launch load dynamics, and high radiation. The technologies must show compatibility with common materials comprising other spacecraft systems, including chemical propulsion. The hardware must also survive mechanical integration and handling on par with typical aerospace construction. Additionally, as the technology is being developed, attention must be paid to minimizing the mass, power, and cost of the associated hardware.

Work has begun at JPL and Ashwin-Ushas Corporation to develop electrochromic devices in this manner. At this time, the development largely entails testing a number of prototype devices for performance, material compatibility, manufacturability, and lifetime. Performance testing of the electrochromic devices is achieved by one of two means. The first entails using a reflectometer or emissometer instrument to acquire hemispheric reflectance measurements for the electrochromic device. These measurements are obtained at ambient temperature and atmospheric pressure conditions and are generally accurate to within 10% of the actual emissivity values. The second performance measurement technique uses calorimetric methods to infer the total emissivity of the device. Under this technique, an electrochromic device is placed in a vacuum environment (usually a bell jar) with a full view of a temperature controlled

heat sink coated with a high-emissivity black paint. An electrical heater is used to apply a known heat load to the backside of the electrochromic device and the temperature of both the device and the heat sink are monitored. The following equation is then used to determine the emissivity of the device:

$$\varepsilon \cong Q/(A\sigma)(T_1^4 - T_s^4) \quad (6)$$

where Q is the applied heat load, A is the area of the electrochromic device, T_1 is the measured temperature of the device, and T_s is the measured temperature of the heat sink.

Figure 7 shows a picture of a calorimetric test setup at JPL. With this technique, care must be taken to avoid parasitic conductive heat leaks to and from the structure holding the electrochromic device and to radiatively insulate the device from a view to the lab environment. However, as long as these parasitic loads comprise a fraction of the heat input into the device, the calorimetric method is the more accurate method for measuring the emissivity of the devices.



Figure 7. Electrochromic device calorimetric test apparatus (Jet Propulsion Lab)

The electrochromic devices are also being tested for chemical compatibility with the iso-butane propellant baselined for use on the micro-Inspector. Although the thrusters on the micro-Inspector will not directly impinge butane vapor upon the electrochromic device radiator, the possibility exists that the spacecraft may encounter a cloud of expunged butane propellant as it performs its mission. To replicate these conditions, an amount of butane equivalent to the full allotment on the spacecraft was placed in a vacuum chamber along with four electrochromic device samples. The apparatus for this test is shown in Figure 8. The devices were tested with an emissometer prior to butane exposure and then tested again after two weeks of soaking in the butane vapor. The electrochromic devices showed no observable physical changes and retained their emissivity variance

capability to within 5% of the original values—well within the uncertainty of the emissometer.

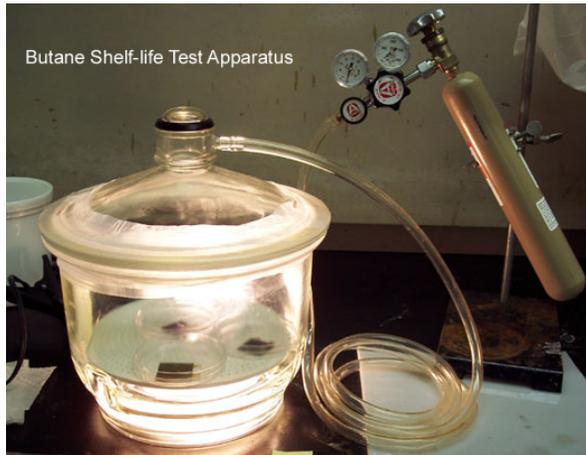


Figure 8. Butane compatibility test apparatus (Ashwin-Ushas Corporation)

In addition to the butane tests, the electrochromic devices are also being tested for long term storage in ambient environments. These tests are to investigate if exposure to air over a long period of time will have a damaging effect on the electrolyte used in the devices. The tests are very similar in concept to the butane compatibility tests. Four device samples are placed in ambient environment storage while four others are kept in a sealed chamber filled with dry nitrogen gas. The devices were tested after an initial 30 days of exposure with the ambient devices showing an approximate degradation of 10% in their emissivity variance while the dry nitrogen soaked showed none. The devices were returned to their respective environments and will be tested at 30 day intervals for approximately six months.

To explore issues with manufacturability, a large electrochromic device was designed and fabricated to fit the dimensions of the pre-cursor LCAM microspacecraft project at JPL. Whereas most of the electrochromic devices used in performance testing have been on the order of 25 cm² or smaller, the electrochromic device for micro-Inspector will have to be an order of magnitude larger to serve as an effective radiator. Additionally, the electrochromic device may require an arbitrary shape to conform to the placement of other structures on the propellant tank underside, such as the docking mechanism. Figure 9 shows an annulus shaped electrochromic device that was procured from Ashwin-Ushas Corp. for the LCAM demonstration hardware. The photo on the right shows the device as installed on the hardware. Emissometer measurements of this surface revealed that the emissivity change was consistent over the entire area of the device, with localized variations at less than 10% of the overall emissivity change of 0.4.

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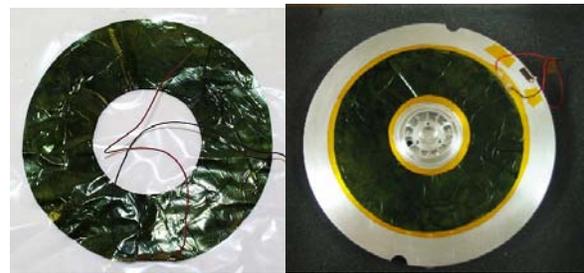


Figure 9. Annulus-shaped electrochromic device for LCAM demonstration hardware.

In addition to the previously mentioned tests, a number of reliability tests have already been conducted on these devices as part of the NASA and DOD Small Business Innovation Research programs. The devices have shown resistance to vacuum environments (little to no outgassing), ultraviolet exposure (over 500 hours), gamma radiation (up to 8 Mrad.) Further development testing as part of the micro-Inspector project will include vibrational tests and thermal cycling tests to gauge environmental stress. While the temperature cycles will not be extensive (-10 to +30°C), the devices will need to be tested to four times expected life, which could entail thousands of switching cycles depending on mission duration. Additionally, the dwell or soak duration for the temperature cycling and the rate of temperature change will be based on expected mission requirements.

SUMMARY

Variable-emittance thermal radiators based on thin-film electrochromic materials are being investigated at the Jet Propulsion Laboratory (JPL) and Ashwin-Ushas Corporation for use on a future microspacecraft. The micro-Inspector spacecraft is an ultra-low mass (less than 3 kg) free-flying microsatellite designed for host-vehicle inspection. With little generated electrical power available, the micro-Inspector relies on waste heat from avionics and instruments to provide thermal energy for its propulsion system. Electrochromic devices developed by Ashwin-Ushas Corporation are used to control the amount waste heat lost to space by modulating the effective emittance of the micro-Inspector's thermal radiator. Thermal models indicate that electrochromic devices can maintain optimal hardware temperatures and decrease heat loss for spacecraft off-sun angles in excess of 45 degrees. A technology development program is underway to assess electrochromic devices for performance, manufacturability, material compatibility, and lifetime issues.

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