A THERMALLY-INTEGRATED SPACECRAFT DESIGN APPROACH
USING NUCLEAR DYNAMIC POWER SYSTEMS

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ABSTRACT

The use of dynamic-cycle heat engines for space electric power systems makes it possible to design a spacecraft which is thermally integrated with the power system. The power system fluid loop is used to cool the spacecraft equipment and structure. This results in a spacecraft which is extremely insensitive to changes in sun-angle, eclipse periods, or power dissipation. Resultant temperature variations of equipment is seen to be a few degrees in the worst-case orbits, without the need for active thermal control. Organic Rankine and Brayton cycle generators are considered. Cycle power losses of less than one-percent are experienced in implementing this concept.

Batteries, for example, should typically be controlled within the range of 0-to-20°C. Occasionally, special items of equipment require still tighter temperature control. For example, the precision on-board clocks in the Global Positioning System navigation satellites require the baseplates to which they are mounted to be controlled within $\pm 1^\circ\text{C}$. Similarly, spacecraft which require high pointing accuracies for astronomical observation or other purposes may require very close temperature control of the mounting structure of the sensor equipment or in some cases, of the entire spacecraft structure, in order to avoid the introduction of thermal distortion errors in the pointing control system.

Even the nominal control range of 0 to 50°C generally requires that some active thermal control be applied, usually in the form of louvers and heaters which are thermostatically controlled to operate within the appropriate range. The atypical, but still fairly common mission, which requires the more stringent temperature control for special equipments frequently results in the requirement for unusual spacecraft designs or operational modes. This may result in designs, for example, with unusually high heater power requirements, or in operational modes which continually orient the spacecraft so as to keep a fixed attitude toward the sun. These, in turn, may lead to increased power and/or weight penalties or may cause other spacecraft functions to be compromised.

A spacecraft which uses a nuclear-dynamic electric power system may be designed so that the power system performs the necessary cooling (and heating) function as well as providing the electric power for on-orbit operations. This paper describes how such a design may be implemented specifically for a radioisotope-powered Brayton or Organic-Rankine system.

The chief benefit of such a thermally integrated spacecraft design is its automatic, completely passive thermal control of all internal spacecraft equipment. Moreover temperatures throughout the spacecraft will vary only a few degrees at most, even in the worst-case orbits.

Other unusual features of the design are:

THE ADVENT OF NUCLEAR DYNAMIC (Brayton cycle or Organic-Rankine cycle) power systems as a spacecraft power option introduces a novel capability to spacecraft designers: The power system can be used to cool spacecraft equipment. This is, the working fluid of the power system can be used as a heat transfer medium to carry heat from internally-mounted equipment to external spacecraft radiators. This frees the spacecraft designer from many of the constraints he now faces on locating dissipative equipment for efficient and effective cooling, and on assuring a relatively uniform (and predictable) temperature throughout spacecraft life and through various sun-angles and eclipse periods.

In typical three-axis-controlled spacecraft designs, a number of surfaces are designated as equipment radiators, and essentially all of the power dissipating spacecraft electrical equipment is mounted with good thermal conduction to these radiators. The remainder of the exterior of the spacecraft is thermally insulated, usually by multi-layer foil blankets. Typically the specifications call for equipment temperatures to be maintained within the range of 0 to 50°C. Not uncommonly, certain items of equipment require much closer temperature control.

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- Essentially all external surfaces are uniformly covered by the same high emittance-low absorptance thermal control material.

- No louvers, heaters, or thermostats are needed for thermal control.

- In non-eclipse orbits, temperature of all internal equipment automatically remains within a 1°C band regardless of sun angle.

- During eclipse periods, internal temperatures typically decline by less than 3°C.

- Temperatures as low as 0°C or less can be readily achieved for specific items of equipment (using the Brayton system), limited only by the radiator size provided.

- Equipment may be arranged internally to suit functional as well as cooling requirements.

- Spacecraft orientation or attitude control technique may be selected without regard for equipment cooling requirements.

- The same spacecraft may be flown at any altitude, in any inclination, with any orientation to the sun, without changing its thermal/structural design.

- The spacecraft will be particularly resistant to destruction by radiation (e.g. laser) weapons; no absorptive solar arrays are deployed and all surfaces are largely reflective to such radiation. The small fraction of absorbed heat is rapidly distributed over the entire spacecraft mass by the fluid loop, thus preventing the development of local hot spots.

CANDIDATE POWER SYSTEMS

At this writing two candidate dynamic turbine-alternator power systems are in competition for development funding by the Department of Energy: the Organic Rankine Kilowatt Isotope Power System (KIPS), and the Brayton Isotope Power System (BIPS). Both systems have been funded through the ground demonstration system (GDS) development phase, and GDS testing is now underway. By the time of this conference, a decision is scheduled to have been made concerning which of the two systems will be carried on to flight system development and flight testing in space.

In the present ground demonstration phase, both power systems are designed for end-of-mission output power of 1300 watts. However, both incorporate the capability for operating in the range of 500-to-2000 watts with some modification of equipment and adjustment of fuel loading and cycle conditions.

The major parameters for the proposed 1300 w(e) flight systems are shown in Table 1.

Table 1. Flight System Major Parameters

	KIPS	BIPS
Thermal input (BOL) watt(t)	7200	4800
Output power (BOL) watt (e)	1300	1300
System efficiency	0.18	0.27
Total weight (incl. radiator), lbs.	475	600
Turbine inlet temp., °C	343	760
Radiator area, ft ²	109	160
Working fluid	Dowtherm A	He/Xe
Radiator outlet temp., °C	50 to 75	-17 to -10

DESIGN CONCEPT

THERMAL DESIGN—In the present concept, an oversized radiator is used to overcool the radiator fluid, in the Organic Rankine case, to approximately 30°C, as shown in Figure 1. From the radiator outlet the liquid Dow Therm-A working fluid is routed to an equipment cooling section internally. Spacecraft equipment is cooled by conduction to the fluid loop, so as not to penetrate the hermetic integrity of the loop. This should raise the loop temperature by some 10-to-20°C, from whence it is returned to the power system. The Organic Rankine system is not affected by the temperature rise introduced by the equipment since it typically requires 50-to-75°C as the inlet to the jet condenser, and normally provides a radiator by-pass loop to control this temperature by mixing the radiator flow with uncooled by-pass flow.

Figure 1. Thermally Integrated Design Approach

The Brayton system, on the other hand, requires a lower inlet temperature to its compressor, and so a modification of this approach is required. This is shown in Figure 2. In this case, the He/Xe working fluid is removed from the radiator at what would be its normal outlet temperature and routed through the equipment section. Its temperature rise due to cooling the equipment, however, is approximately 40-to-50°C. In order to restore the desired compressor inlet temperature, the He/Xe working fluid is then returned

to a separate section of the radiator and allowed to recool to the -15°C range before returning to the Brayton rotating unit. If this is implemented as shown in Figure 2, the radiator temperature will show a continuous variation, so that the skin can be continuous, and no insulator section need be interspersed between the two radiator sections.

Figure 2. Thermally-Integrated Design Approach; Modification

Thus, the entire spacecraft, with minor exceptions, is enclosed by the power system radiator and is cooled by the power system fluid loop. It is a good approximation, therefore, to say that the spacecraft is thermally interconnected and will act as a single thermal mass. Thus, temperature variations in orbit and during eclipse periods will be minimum, as will be illustrated below.

In this design approach, all exterior surfaces should be either part of the radiator, i.e., thermally well-connected to the actively heated portions of the exterior surface, or well insulated. Thus, all external surfaces should have low absorptance and high reflectance to incident radiation. This will minimize temperature variations in orbit.

In a cylindrical configuration, the cylindrical shell is most typically formed into the radiator. The ends of the cylinder should be closed and either insulated or thermally connected to the radiator cylinder. The latter may be achieved by heat pipe conduction, wherein radially disposed heat pipes are embedded within or bonded to a solid conductive disc which forms an end face of the cylinder. The heat pipes are conductively bonded or bolted to a radiator fluid plenum at the periphery of the disc, thus completing the thermal connection. This radiator extension is illustrated in figure 3, which shows that a complete disc or any segment of a disc may be used in this manner.

MECHANICAL DESIGN- A mechanical design which would allow the non-power-related spacecraft equipment to be assembled and tested separately from the power system, despite their eventual close integration, is required. In concept, this may be achieved as shown in figure 3. The two major subassemblies would be manufactured and assembled separately, but with mechanical and thermal interfaces provided for their eventual mating.

Figure 3. Spacecraft Mechanical Assembly

The mating of the two major spacecraft components allows repeated assemblies and disassemblies to be performed. Mechanical connections are by bolts and other reversible fasteners.

Thermal connections are similarly fastened. As illustrated in Figure 3, the spacecraft electronics boxes are all mounted to equipment shelves within the equipment module. These shelves contain embedded and bonded heat pipes, designed similarly to the radiator extensions previously described and illustrated in detail in Figure 4. In this case, the interior portion of each heat pipe is its boiler section, while the peripheral portion is its condenser section. This peripheral portion is then bolted to a mating section of a plenum which contains the circulating working fluid of the power system.

Figure 5 illustrates how the various working fluid plenums may be arranged to provide heat-transfer surface paths for the equipment shelves, using the Brayton-

Figure 4. Radiator Extension Details

Figure 5. Working Fluid Loop - Brayton Cycle

cyclic system as an example. The heat pipes attach to these plenums as illustrated in Figure 6.

Figure 6. Heat Pipe Attachment to Fluid Plenum

THERMAL PERFORMANCE-Control and implementation of the design with good thermal performance requires that the temperature gradients between the working fluid and the equipment shelves be minimized. The

heat pipes themselves will introduce relatively small gradients. The ΔT due to conductive contact between heat pipes and plenum walls can be similarly small if carefully mated surfaces and good fastening procedures are implemented. It follows that the major temperature drops will be introduced within the plenum-to-working fluid heat transfer segment.

In general, this "film-drop" can be reduced by increasing the pressure drop of the working fluid in the plenum, thus decreasing the efficiency of the power conversion system. The problem, therefore, is that of finding a design point which represents a suitable compromise between temperature drop and system power loss.

ORGANIC RANKINE SYSTEM-Here, the cycle efficiency is only weakly dependent on liquid pressure drop. Thus, a suitable design point can be found at a tube diameter of 1.5 inches, assuming a tube with an annular flow channel as shown in figure 7. For the heat input per unit length shown, and the indicated pressure drop per unit length, the resultant specific pressure drop is 3.76×10^{-3} psi/watt. Therefore, for a total thermal dissipation on the equipment shelves of 1200 watts, (assuming ~ 100 w is radiated as rf power), the total pressure drop would be 4.5 psi. From reference 1, the efficiency loss will be approximately 0.3% or some 4 watts (e) in a 1300 watt (e) system. For a uniform heat input along the tube, the film temperature drop will be some 2°C , as shown in Figure 7, so that the equipment temperatures will follow the working fluid temperatures quite closely.

This is illustrated by assuming a disposition of various electronic equipment boxes on a typical shelf, with the power dissipations shown in Figure 8.

Figure 7. Organic Rankine System Plenum Trade-Off Parameters

Figure 8. Assumed Equipment Shelf Component Dissipation (Watts) Total on Shelf = 530 W (t)

The resultant temperatures in the plenum and on each of the boxes is shown in Figure 9, as determined in a 2-D multi-mode finite-difference calculations.

Figure 9. Organic-Rankine Spacecraft Equipment Shelf Temperature Map

BRAYTON CYCLE-The Brayton Cycle parameters led to a plenum design which utilizes a tube with internal fins as shown in Figure 10. The design parameters were selected to correspond to a 1.75"

Figure 10. Equipment Shelf Header Design Parameters (Brayton Spacecraft)

O.D. tube but, as can be seen in Figure 10, only a slight increase in the diameter is needed to achieve allowable pressure drops without significantly affecting the temperature gradient from the tube wall to the gas.

The allowable pressure drop was selected to correspond to that which would produce a similarly moderate power loss in the Brayton system; in this case approx-

imately 10 watts(e) in a 1300 w(e) system (Ref.2). Temperature drops, however, are seen to be much larger than those which can be achieved with the liquid Rankine system. Fortunately, the absolute temperature of the gaseous Brayton working fluid is quite a bit lower to begin with, so that the higher temperature drops can be tolerated. This is illustrated in Figure 11. In this case, the working fluid has already experienced a 25°C temperature rise in cooling other equipment prior to entering the reference plenum.

Figure 11. Rankine Cycle Spacecraft - Equipment Shelf Temp. Map (With Auxiliary Circular Heat Pipe)

With the Brayton system in particular, equipment locations can be selected to satisfy thermal preferences, if desired. For example, solid state rf power amplifiers typically experience improved efficiency at lower temperatures. These critical components can, therefore, be located near the inlet side of the equipment shelf, thus achieving efficiency values which would not normally be practical in a direct-radiation-cooled amplifier. With a working fluid temperature of -17°C at the radiator outlet, certain equipments can be held at sub-zero temperatures, if located near the inlet of the cooler equipment shelf.

TEMPERATURE VARIATIONS IN ORBIT

Overall spacecraft temperature variations during the worst case eclipse orbits were determined for a variety of spacecraft surface finishes. Because the selected design approach results in essentially the entire spacecraft being cooled by the power system fluid loop, the vehicle was treated as a single-node isothermal body for the purpose of the calculation. Since eclipse transients have time constants of the order of an hour, and the fluid loop transit time through the spacecraft is of the order of seconds, this is believed

to be a reasonable assumption for the spacecraft as a whole.

The resulting temperature profile for various surface finishes is given in Figure 12, based on a 12-hour noon orbit. An absorptance to emittance ratio (α/ϵ) of 0.4/0.76 refers to a silvered teflon coating which is available in the form of an adhesive tape. $\alpha/\epsilon = 0.3/0.9$ are standard degraded properties values for a limited class of white paints which include SG-13 and Z-93. The aluminum side of vapor-deposited aluminum on Kapton has an α/ϵ ratio of about 0.11/0.35. The Kapton side of a 2-mils sheet has an α/ϵ of 0.36/0.76.

Figure 12. Overall Spacecraft Temperature Transient in Maximum Eclipse Orbit

Figure 12 indicates that a silvered teflon coating offers the most satisfactory selection. It combines a low overall temperature (temperature values have only relative significance) and a low worst-case orbit variation, as seen in Figure 13 (4°C from maximum to minimum). In addition, it is superior to white paints in the stability of optical properties and shows only slight dependence of absorptance on the angle of incidence.

Figure 13. Orbital Temperature Variation

Finally, Figure 14 shows the response of a typical electronic box to the sink temperature variation of Figure 13. This box, which dissipates 24 watts, shows a total temperature decline of $\sim 2^{\circ}\text{C}$ during eclipse, and varies only minimally in non-eclipse conditions.

Figure 14. Typical Electronics Box Temperature Eclipse Period Variation (Box Dissipation = 24 Watts)

Typically such a box rides $\sim 10^{\circ}\text{C}$ above the neighboring plenum segment. Thus, if the box is switched off and its dissipation goes to zero, its temperature will decline only slightly below the associated plenum segment before equilibrating at the stand-by temperature, only some 10°C below its normal operating temperature.

CONCLUSIONS

A thermally integrated spacecraft design approach can be implemented with the nuclear dynamic power systems which makes use of the working fluid of the power systems to maintain spacecraft equipment at very stable, nominal temperatures. Thermal control of all included equipment, is completely passive and automatic, and temperature variations are extremely small, even in eclipse conditions. The output power penalty is less than one percent.

The power system module may be independently assembled, hermetically sealed, and tested. An equipment module may be similarly assembled and tested separately. The two modules need be combined only late in the program assembly process. Thermal-vacuum testing of the total spacecraft can be significantly simplified or dispensed with in view of the non-differentiated nature of the thermal design.

The reference design approach should be particularly valuable for spacecraft with unique thermal stability requirements, or for standard-bus spacecraft. These letters are intended to accommodate a variety of payloads without redesign. A well-controlled and non-differentiated thermal design, insensitive to equipment dissipation or orbital conditions, is an obvious advantage for a standard spacecraft.

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