Space Station Active Thermal Control Technical Considerations.

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A description of recent and planned thermal control technology developments at the Johnson Space Center and the other NASA Centers in support of Space Stations will be presented. The program is centered around satisfying the needs of the users. Preliminary results of proof-of-concept high capacity heat pipes and two-phase devices will be included which indicate that large amounts of energy (100 kw) can be transported long distances (50m) with very small temperature differences. The presentation will summarize preparations for an "evolutionary test bed" for advanced development of thermal technology which will provide data on components and systems for incorporation into the Space Station designs in the late 1980's. The results of the recently flown Heat Pipe Experiment aboard STS-8 will be presented.

The Space Station must have the capability to reject at least as much energy as it generates and utilizes in order to achieve an energy-balanced system. The major drivers of the design of the thermal control system are established by the multi-year mission duration, large quantities of waste heat to be dissipated, long physical distances involved and variety of payloads and missions which must be accommodated by the Station. The Space Station is especially unique in that it must be capable of accommodating widely varying heat loads, with heat source locations which can be reconfigured by the crew as dictated by mission objectives. Therefore, in addition to system size, long-life reliability, maintainability, versatility, and modularity for growth several years. Therefore, in addition to system size, long-life reliability, maintainability, versatility, and modularity for growth are thermal management requirements unique to the Space Station. Since the thermal system must ultimately accommodate heat rejection up to 100kw or higher and heat transport distances up to 165 feet, it will require comparably large heat rejection systems with radiator areas of hundreds of square feet. Because of its large size and dependence on a good view to the space environment, the radiator will be a principal driver on the overall configuration of the spacecraft.

The three primary technology challenges that must be met to support the Space Station in the area of active thermal control are long-life heat rejection; highly versatile thermal acquisition and transport; and efficient overall thermal utility system integration.

Heat rejection focuses on the requirements for final transfer of waste heat from the spacecraft to the ambient environment by radiation. Large, deployed radiators for heat rejection are required for any large manned Space Station concept. As a result, the radiator is by a significant margin, the largest and most exposed portion of the Space Station thermal system. Thus, radiator size, complexity and efficiency improvements are mandatory in order to produce viable, long-life Space Station thermal subsystem cost, weight and reliability.

Thermal acquisition and transport requirements encompass the collection and movement of thermal energy from the Space Station's heat sources to the radiator heat sink at required temperature levels. Current thermal subsystems require precise ordering of equipment within the thermal transport circuit to maintain temperature control (i.e., equipment requiring cold temperatures must be located first in the circuit, with equipment that can tolerate higher temperatures located later in the circuit). The modularity/growth concept of the Space Station requires that it accept multiple heat loads of varying magnitudes, heat flux density and locations without causing adverse heat source interactions (i.e., the thermal system should be insensitive to multi-disciplinary user loads and their locations). Thus, the key to Space Station thermal acquisition and transport lies in the creation of a highly versatile thermal "utility" or bus system analogous to municipal public utilities, where basic "trunk"
Thermal system integration focuses on the requirements involving the cumulative performance of the elements within the active thermal control system and the system's integration with other Space Station subsystems. Current thermal systems make large use of electrical heaters and require significant crew involvement to change system operating configurations as power profiles and heat loads change and system failures occur. The Space Station thermal system must 1) make judicious use of waste heat by making it readily available to subsystems so as to minimize or eliminate electrical heaters and 2) minimize crew involvement by providing an integrated highly reliable, automated thermal utility system. A further thermal system level challenge is that on-orbit maintainability and serviceability complemented by periodic growth and refurbishment are required to achieve realistic operational life and costs.

The Thermal Options

System Level - At the integrated thermal system level, the option of primary importance to the overall vehicle configuration is the degree of centralization. In a decentralized Space Station thermal system concept, each module of the Station collects and rejects all of the waste heat it generates with no module interconnects necessary. Each module would have complete flexibility in the selection of its control temperature and in how its heat is rejected. It would be possible for modules to use different working fluids and devices to accommodate their differing heat rejection requirements. However, the decentralized approach would not allow waste heat from one module to be used by another. Also, since each module rejects its own heat, it would require its own radiator system and thus impose its own orientation and location restraints on the overall vehicle. Since several modules may require radiator deployment to achieve heat rejection, overall system thermal complexity and weight would tend to be high.

A centralized Space Station thermal system, on the other hand, would provide for a much more integrated approach to the thermal system design and has many operational and functional advantages over the decentralized system. A centralized system allows full utilization of waste heat generated by one module to be used by another. It minimizes the size, and thus the cost, of the thermal system since waste heat can be utilized and not rejected thru the radiator. Also, because the system can take advantage of the "peaks and valleys" of the module heat loads, it can be sized for a more average heat load level, and thus be significantly smaller in overall size and capacity than a decentralized system. A centralized system also allows for an efficient radiator design that can be located in an "out of the way" minimum environment position on the Space Station. Inherent in the centralized approach is the necessity to transport heat across module boundaries. Several methods of accomplishing this interface have been investigated. These include direct fluid interface through fluid disconnects or on-orbit welding or brazing, typical compact core heat exchangers with connectors, and contact heat exchangers. The latter two methods would not require intermixing of module heat transport fluids.

Heat Rejection - All Space Station concepts envisioned to date require large deployed radiators to reject waste heat. It is necessary to minimize the size of the radiators not only because of obvious weight and cost considerations, but also because of vehicle design and operational considerations. Smaller radiators enhance initial vehicle launch and construction, vehicle attitude control and stability maintenance, payload space sensor viewing, resupply vehicle docking, and micrometeoroid/space debris environment exposure.

In general, to minimize the size of the deployed radiator panels they should be located in an orientation that is edge-to-sun as much as possible. If the radiator is not located edge to sun, the radiators must reject heat to a substantial external environment. The advantage of a fixed radiator configuration is that no rotary fluid joints are required. An alternate location for the radiators is on the same or a similar gimbaled mechanism as the solar array. In this configuration, the radiator location would be constant relative to the solar arrays and thus always have the same minimum view of the solar arrays. The radiators would also be fully edge-to-sun at all times and thus would be exposed to a very minimum external environment. However, since the radiators are not fixed relative to the basic Space Station structure, rotary fluid joints would be required. If a deployed radiator is gimbaled so that it is continuously directed away from the solar flux and the earth flux as much as possible, its area can be reduced as much as 60% over that of a fixed orientation radiator. Another advantage of a gimbaled radiator is that it would be much less sensitive to thermal coating degradation because it would not be exposed to a significant solar environment. Also, since the solar flux is a major cause of the degradation, the rate of coating deterioration would be reduced for a gimbaled radiator.
System complexity considerations preclude simple extension of the Shuttle Orbiter radiator technology for a Space Station that has sustained on-orbit operations. The Orbiter spacecraft rejects its waste heat by mechanically pumping fluid through a space radiator system that contains over 400 separate fluid tubes with a combined length of over 5500 ft. (ref. 1). System reliability becomes unacceptable for missions much greater than about 30 days, since the system is vulnerable to failure from a single meteoroid or space debris penetration of a radiator tube. High reliability for long duration missions can be achieved with a Shuttle type fluid radiator if it is adequately shielded from the meteoroid and debris environment. However, the resulting space radiator system would be heavy and complex because of the required redundant plumbing, pumping and valving hardware. For example, an existing Shuttle type fluid radiator system would require over 19,000 ft. of tubing consisting of over 1500 individual pumped fluid tubes, over 50 fluid manifolds, over 75 isolation valves, fluid disconnects and fluid swivels or flex lines for a 100kw heat rejection system. Therefore, it will be necessary to minimize heat rejection system complexity for the Space Station by incorporating more inherently reliable concepts.

Heat pipes offer an attractive alternative for eliminating many of the single point failures in a space radiator system (ref. 2). A heat pipe radiator concept utilizes multiple independent heat pipes. Therefore, the loss of a single heat pipe is not catastrophic and the need for supplemental meteoroid protection is eliminated. The basic heat pipe radiator concept couples the heat sources to the radiative heat sink through an intermediate array of heat pipes. The heat source rejects its heat to the evaporator portions of the heat pipes. The heat is subsequently removed in the condenser portion of the heat pipes by conduction to fins which make up the surface that radiates the heat to space. Prototype heat pipe radiator panels have been designed, fabricated and successfully tested (ref. 3).

Figure 1 shows a weight comparison of conventional fluid type radiators and heat pipe radiators for a long life Space Station mission (ref. 4). The data shows that heat pipe radiators are relatively insensitive to micrometeoroid penetration probability design requirements and mission life while fluid radiator weight is strongly dependent on these parameters. Heat pipe radiator systems can also significantly decrease overall thermal system complexity when compared to pumped fluid radiator systems. Since single heat pipes can be fabricated to passively reject as much as 2kw of heat, as few as 50 heat pipes could accommodate a 100kw Space Station. This compares to the multi-component, complex pumped liquid radiator system described in the preceding paragraph.

The high capacity monogroove heat pipe concept shown on Figure 2 has been developed to simplify the use of heat pipes on space radiators (ref. 5). The monogroove heat pipe separates the heat transport and heat transfer functions so that each can be optimized separately to provide heat transport capacities on the order of 600,000 to 1,000,000 W-in and high heat transfer film coefficients. It combines the advantages of axial grooves, such as simple construction and large liquid and vapor areas, with the high heat transfer coefficients of circumferential wall grooves. The basic monogroove design contains two large axial channels, one for vapor and one for liquid. The small slot separating the channels creates a high capillary pressure difference which, coupled with the minimized flow resistance of the two separate channels, results in the high axial heat transport capacity. The high evaporation and condensation film coefficients are provided separately by circumferential grooves in the walls of the vapor channel. As indicated on Figure 2, a 55 ft. prototype monogroove heat pipe has been fabricated and successfully tested. In order to provide a practical, compact radiator heat pipe module, the evaporator has been made multi-legged. The evaporator is made up of six 18 inch parallel monogroove heat pipes manifolded together on a monogroove header, which is also attached to the monogroove heat pipe condenser. As shown by the data on Figure 2, this prototype heat pipe can reject greater than 2kw of heat and transport 600,000 W-in. A small 6 ft. version of this monogroove heat pipe was successfully flown on the Shuttle STS-8 mission.

The heat pipe radiator STS-8 flight experiment consisted of a single U-shaped monogroove heat pipe which was bonded to a radiating fin, see Figure 3. Although the monogroove heat pipe is being developed for ammonia fluid, Freon 21 was used in the STS-8 hardware. Heat input for the experiment, limited to 100 W by Shuttle power constraints, was provided by two electrical heaters (30 W, 70 W) attached to the underside evaporator flanges and heat rejection was via a double sided aluminum radiator bonded to the condenser flange. Since time precluded using Shuttle systems for data acquisition, a temperature sensitive liquid crystal film was used to monitor general temperature levels in the evaporator and condenser sections. The film is sensitive in 5°C increments over a 20 to 45°C temperature range and responds by visual color changes which were observed by an astronaut and recorded on photographic film.

The experiment operated on STS-8 in a stable condition with the single 75 W power setting for 2 hrs 35 minutes before being shut off. Evaporator and condenser temperatures, based on reported colors at selected times were in accordance with preflight predictions. Post-
flight inspection of the Tempilabel decals also indicated that the highest registered evaporator temperature was 49°C (120°F). This further confirms proper heat pipe operation at all times. Thus, successful sustained operation of a monogroove heat pipe radiator has been confirmed in the zero-g space environment. No priming or operating problems were experienced at anytime.

Another technology option that must be considered in the heat rejection area is whether or not to deploy or construct the space radiator. A pumped fluid radiator would probably require deployment since the complexity of making and insuring leak tight fluid connections on-orbit would discourage construction in space. A deployable radiator would simplify initial Space Station on-orbit buildup, but at a significant system complexity, weight, and cost penalties. The deployment mechanism will be inherently complex, requiring drive motors, fluid swivels or flex hoses, and increasing radiator weight by 25 to 50%. Cost penalties result from this increased weight as well as from the cost associated with deployment mechanism development and fabrication. Heat pipe radiators readily lend themselves to on-orbit construction since they are made up of several independent, closed elements. If a leak of a single heat pipe does occur during construction, it does not significantly impact overall system performance. Radiator construction in space would require on-orbit crew time using the Shuttle RMS (remote manipulator system), but would significantly decrease overall radiator system complexity, weight and cost. Also, a space constructed radiator would be inherently maintainable on-orbit, perhaps of overriding importance in its selection for the indefinite life requirements of the Space Station.

A space constructable radiator system that fully meets the challenges presented by the Space Station mission has been conceived. It uses the large (~2 kw capacity) independent heat pipe radiator elements previously described. These elements are coupled and uncoupled to a centralized heat transport circuit as shown in Figure 4. In this concept each of the radiator heat pipe elements is an identical sub-module of the system and comes attached to its own radiator fin and heat transport circuit interface section. Thus, any required radiator area would be formed by simply putting together in a building block fashion the required number of heat pipe modules. The heat pipe radiator modules would be attached to the heat transport circuit without breaking into the Space Station thermal system. The space constructable heat pipe radiator approach has several significant advantages. First, system complexity would be minimized by reducing the number of radiator elements by an order of magnitude (e.g., 50 heat pipes vs 1500 fluid tubes for a typical 200 kw system). Radiator costs would be minimized since the system consists of multiples of identical modules that could utilize longer production runs. The system would have extended, indefinite life capability due to the insensitivity of the design to the micro-meteoroid environment. A penetration of one heat pipe element would not affect the operation of the other heat pipes or significantly affect the overall heat rejection capability (e.g., 2% for 100 kw system). Also, long life would be enhanced by ease of refurbishment since individual heat pipe radiator elements can be easily replaced or upgraded. Launch weight and volume would both be minimized since the radiator can be constructed in space from compactly stowed elements. Therefore, a heavy complex deployment mechanism would not be required.

The thermal interface between the space constructable radiator and the station heat transport circuit also constitutes another critical area. The requirement to transfer up to 100 kw out of the heat transport circuit and into the space constructable radiator system through a contact heat exchanger is one of the major challenges that must be addressed for any type of high-energy space constructable thermal management system. Because these interfaces must be attachable and detachable in a zero-g space environment using the remote manipulators further complicates the design of these elements. Heat transfer with a low temperature drop is required and this necessitates intimate thermal contact across the joint. Mechanical bolted joints are common both for ground and space applications. Interface materials such as indium foil and thermal grease have been used to enhance heat transfer. For remote on-orbit assembly, this kind of joint would be very difficult to implement. In order to provide a more practical technique for in-space construction, two types of thermal contact joints have been investigated. Both approaches rely upon the application of pressure between dry surfaces in intimate contact to provide the required heat transfer. The first is the plug-in heat exchanger shown in Figure 5. The evaporator of the radiator heat pipe panel is inserted into the heat exchanger by the remote manipulator device. Contact pressure between the surface of the heat pipe and the fluid heat exchanger is provided by gas or hydraulic pressure against a diaphragm. Very acceptable heat transfer coefficients of as high as 500 Btu/hr-ft² have been measured for joint pressures of 150 psi. The disadvantage of this concept involves the added complexity of components associated with the gas or hydraulic pressurization system.

Figure 6 shows a second approach, which utilizes a flat contact heat exchanger. In this concept, 100 psi pressure is applied to the contact interface by thermal expansion bolts and a truss assembly to distribute the load over the contact surface. A prototype heat exchanger of
Heat Acquisition and Transport - In previous manned spacecraft, thermal transport between heat sources and the radiator heat sink has been achieved through the use of pumped fluids. The Shuttle Orbiter and Spacelab are examples of this technology. These conventional pumped liquid loop systems use mechanical pumps to circulate the Freon 21 or water working fluids through the system. Pumped loop technology is available and has been found to be fairly reliable and functional in use. However, when sized to accommodate the very large heat acquisition and transport distances required by the Space Station, severe pump power penalties and operational constraints result. Pump power penalties for a 100 to 150 kw Space Station would be on the order of 3 to 5 kw. This would not only be a very significant energy consumer on the Station but require extremely large and costly space qualified pumps to be developed and fabricated. The large pumps would also be major noise and vibration generators. Furthermore, their periodic maintenance, refurbishment, and/or replacement during the life of the Station would be a continuing drain on operating costs and crew time.

Operational constraints of the pumped fluid system result because of the large system temperature differentials that result around the fluid circuit with reasonable fluid circulation rates. For example, the Shuttle Freon 21 loop operates with a temperature differential of about 50°F, i.e., the fluid is allowed to heat up from 40°F to 90°F during its path around the heat transport circuit. This varying and constantly increasing heat transport fluid temperature requires that equipment be placed in the circuit at the precise position where fluid temperature is adequate to provide the necessary equipment cooling temperature. Thus, equipment must either be physically located in proper order of fluid lines must often double back on themselves to place equipment in proper order relative to the fluid loop. This is an acceptable burden on small spacecraft and is tolerable even on large spacecraft like the Orbiter where heat load locations are fixed. However, even on the Shuttle, the liquid circuit approach results in undesirable limits on payload thermal support. For a large evolving Space Station where heat load location and magnitude variations are an integral part of the inherent mission objectives, a pumped liquid system will have even more severe operational constraints, perhaps significant enough to preclude achieving necessary flight requirements.

A Space Station heat transport system has been conceived that offers the potential of low power consumption with easy adaptability to accommodate multiple heat loads of varying magnitudes and locations without creating adverse heat source interactions. The concept is a two-phase heat transport circuit, as shown in Figure 7, in which the heat transfer into or out of the loop is achieved by evaporation or condensation of a working fluid (e.g., ammonia). The prime mover for the fluid is a small pump located in the liquid portion of the loop. In this concept the heat load is removed from individual subsystems/payloads through evaporative heat exchangers. The vapor from the heat exchanger is fed through a vapor return line to the radiator interface heat exchanger where it is condensed and slightly subcooled. The liquid that comes from the radiator heat exchanger is then circulated back to the heat loads with a small liquid pump. Because heat transport is determined by the heat of vaporization (e.g., ammonia, 589 BTU/LB) rather than the heat capacity of Freon 21 (0.24 BTU/LB/°F), the pump flow requirements are at least 50 times less for this system than they would be for a Freon 21 fluid circulation system. Therefore, many of the components, particularly the pump, of the thermal control system can be significantly smaller in size due to the high heat transfer rates inherent in a condensing/evaporating system. Figure 8 illustrates the overwhelming pump power advantage of a two-phase heat transport circuit over a Shuttle type pumped liquid circuit. The two-phase pump power for a 100 to 150 kw Space Station is less than 100 watts compared to 3 to 5 kw for a pumped liquid system. Long life space qualified pumps of 100 watts or less are readily available and their minimum size and cost significantly enhances system maintainability and reliability. Another major driver for considering a two-phase heat transport system for the Space Station is that the evaporation and condensation processes are essentially isothermal. Thus, the two-phase loop provides a uniform thermal control bus temperature for Space Station subsystems, experimental equipment and payloads. Therefore, sequencing of heat generating equipment is not required and overall Space Station modularity, operational flexibility and evolutionary growth is made a practical reality by taking advantage of the inherent versatility of a two-phase heat transport bus.

Several two-phase heat transport circuit hardware programs have been conducted to prove the feasibility of the thermal bus concept (ref. 6, 7, 8). An effort to design, develop, build, and test a prototype high-capacity, isothermal heat transport subsystem utilizing this thermal bus concept is underway. The current baseline approach in this effort is a series flow, pump assisted wicked evaporator concept. The approach combines elements of the pumped two-phase and capillary pumped
loops previously investigated. The system utilizes two-phase evaporative cold plates and liquid to two-phase evaporative heat exchangers for the heat source interfaces and condensing contact heat exchangers for the heat rejection system interface. The system control temperature will be controllable from 40°C to 40°C with a 5°C control band. Individual heat loads can vary from 1 to 25 kw and, through modularity and growth considerations, will be able to support a total system heat rejection requirement of 150 kw. Thermal bus prototype subsystem testing is expected to occur by mid FY 85 with the initial development effort to be completed by early FY 86.

A Candidate Thermal System

In response to the above discussed Space Station thermal considerations, a candidate thermal management system has been concepted, as shown in Figure 9, which optimizes system weight, power consumption, growth capability, operational feasibility, maintainability, and cost. The two major elements of the system are the heat transport subsystem and the heat rejection subsystem. The pumped two-phase thermal bus provides a heat transport subsystem that operates at a near constant temperature, independent of location in the circuit, for payload/subsystem heating or cooling while requiring less than a tenth of the electrical power of a comparable single phase pumped liquid system. The space constructable heat pipe radiator will allow any size radiator system to be constructed or maintained at any time during the Space Station's evolution by simply combining the required number of identical independent heat pipe radiator modules. The same basic high capacity heat pipe would also be utilized in a combined Space Station module meteoroid shield/radiator design to minimize the amount of deployed radiator area for viewing or docking constraints.

Future Plans

Extensive planning of thermal management system technology development efforts aimed at an eventual application to a Space Station has occurred during the past 4 years. Investigation of some of the most promising major components has been initiated on a fairly modest level during this period. These initial efforts have been successful in providing a significant amount of early demonstration hardware and data and, at the same time, notifying industry of NASA's interest in this area. Variations of existing technology hardware as well as new technologies will now be fed into a Space Station Thermal Management System Test Bed, where various options for particular components and/or subsystems can be evaluated and compared. An example of the integration of the technology development program with the test bed is shown in Figure 10.

References

FIGURE 1 - COMPARISON OF PUMPED FLUID AND HEAT PIPE RADIATORS
FIGURE 2 - PROTOTYPE HIGH CAPACITY MONOGROOVE HEAT PIPE
FIGURE 3 - SPACE CONSTRUCTABLE HEAT PIPE RADIATOR
FIGURE 4 - SHUTTLE STS-8 MONOGROOVE HEAT PIPE FLIGHT EXPERIMENT
FIGURE 5 - PRESSURE CLAMMING HEAT PIPE TO FLUID CONTACT HEAT EXCHANGER
FIGURE 6 - HEAT PIPE RADIATOR MECHANICAL CONTACT HEAT EXCHANGER
FIGURE 7 - TWO-PHASE HEAT TRANSPORT CONCEPT
FIGURE 3 - COMPARISON OF SINGLE VS TWO-PHASE HEAT TRANSPORT CIRCUIT POWER REQUIREMENTS
HEAT TRANSPORT (TWO-PHASE THERMAL BUS)

HEAT ACQUISITION (CONTACT HEAT EXCHANGER EVAPORATORS)

CENTRAL HEAT REJECTION (GIMBALED SPACE CONSTRUCTABLE RADIATOR)

MODULE HEAT REJECTION (INTEGRAL HEAT PIPE RADIATOR/ METEOROID SHIELD)

FIGURE 9 - SPACE STATION ACTIVE THERMAL CONTROL CANDIDATE SYSTEM
FIGURE 10 - INTEGRATION OF TECHNOLOGY INTO THERMAL TEST BED