Lessons Learned During the Integration Phase of the NASA IN-STEP Cryo System Experiment

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ABSTRACT

A key value of this flight experiment has been the opportunity for Hughes and the Jet Propulsion Laboratory (JPL) to identify and resolve cooler and imaging-instrumentation integration issues that will be encountered when these enabling thermal management technologies are integrated in future space cryogenic cooling systems.

BACKGROUND

The CSE is funded by the NASA Office of Advanced Concepts and Technology's IN-STEP and managed by the JPL. Hughes conceived the experiment and completed the equivalent Phase A activity outside of IN-STEP through both government funded and internal research and development programs. Hughes has completed Phase B (mechanical, thermal and electrical design activities), Phase C/D (fabricate, assemble, test, and deliver the CSE to Goddard Space Flight Center), and will support the pre-flight, flight and post-flight data analysis activities for Flight 1 of this two flight experiment. The CSE is currently at the Goddard Space Flight Center (GSFC) awaiting integration onto a Hitchhiker cross-bay carrier.

The overall goal of the CSE is to validate and characterize the on-orbit performance of three thermal management technologies that comprise a hybrid cryogenic system. These thermal management technologies consist of: 1) a second-generation long-life, low-vibration, Stirling-cycle, 65 K cryocooler that will be used to charge a thermal energy storage device (TTTS), 2) a diode oxygen heat pipe thermal switch that enables physical separation between a cryogenic refrigerator and TRP, and 3) a nitrogen, triple-point thermal energy storage device to provide a stable cryogenic temperature source. This experiment is necessary to provide a high confidence zero g database for the future design of cryogenic systems for multi-year space flight applications. The level of confidence that will be provided by this experiment is an important NASA and DoD requirement prior to multi-year mission commitment. The CSE illustrates an important type of NASA space-flight experiment in which an emerging technology is validated to provide the option for subsequent application in near-future space system developments.

The CSE was originally proposed as two separate flights. The first flight, the basis for this paper, includes the ISSC cryocooler and the diode oxygen heat pipe; it utilizes an aluminum sphere to simulate the nitrogen triple-point thermal energy storage device that is anticipated to be included in the second flight.

INTRODUCTION

In this experiment two types of coolers are used. The first is a Stirling-cycle, Hughes long-life, low-vibration, 65 K, 2 W, ISSC suitable for Dual space missions. The technology benefits are significant to a number of science instruments selected for NASA's Earth Observing System (EOS) instruments and a number of space reconnaissance instruments. The unit consists of a compressor connected to an expander by a transfer tube, shown in Fig. 1, and the control electronics.

The second type of cryocooler is a Hughes rotary motion, tactical piston-driven Stirling cooler. This tactical cooler, shown in Fig. 2, has been designed by Hughes to have a relatively large cooling capacity and an anticipated life on the order of 1 000 hours. This cooler has successfully flown on previous shuttle flight experiments to support a rapid cooldown to cryogenic temperatures allowing more time for the actual experiment to be performed at cryogenic temperature. Its predecessors have been used to cool sensors in aircraft and missiles where mechanical robustness and insensitivity to vibration are key requirements.

Figure 3 is a block diagram of the CSE with the ISSC connected to the simulated TTTS mass through a thermal heat strap. One tactical cooler is connected through the diode oxygen heat pipe to the opposite side of the simulated TRP. This physical arrangement was selected to reduce risk should a failure occur within the heat pipe or the ISSC. In an operational system the diode oxygen heat pipe would be located between the ISSC and the thermal energy storage device. A second tactical cooler is used to minimize parasitics by cooling a radiation shield to

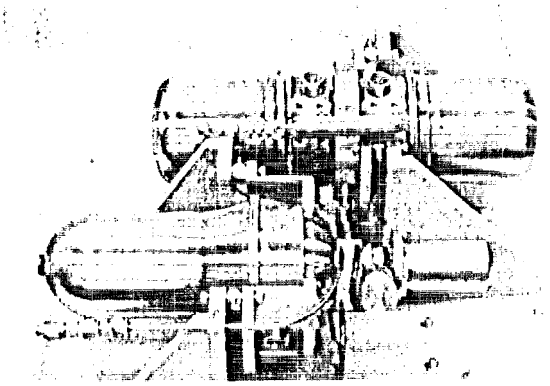


Figure 1. Hughes lo~l~-life, low-vibration, 65 K, 2 w, Improved Standard Spacecraft Cryocooler (ISSC).

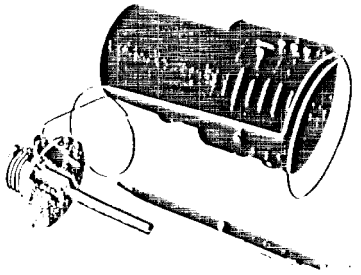


Figure 2. Hughes Model 7044 11 tactical cryocooler.

atemperature of approximately 120 K. In a non-shuttle flight system, shield cooling is generally provided by a cryogenic radiator.

Figure 4 shows the CSE mechanical assembly attached to the upper end plate (UEP). The ISSC and two tactical coolers are heat sunk to the UEP that acts as a radiator. The remainder of the experiment is connected to bipods through support and ballast rings that support the radiation shield and simulated '1'}<}', and the canister electronics box, which contains sensor signal conditioning circuitry. The entire mechanical assembly is attached to, or suspended from, the UEP and secured in a 5 ft^3 canister. The canister and two electronics boxes are mounted on ii cross-bay Hitchhiker carrier assembly with several other experiments, as shown in Fig. 5.

An important thrust of the CSE is the integration of the cryocoolers into this cryogenic system. Not only must the cryocoolers provide the necessary cooling capacity, they must also be thermally, mechanically, physically and electrically compatible with the rest of the instrument. As long- life., low-vibration cryocoolers transition from an emerging to an enabling technology, the focus shifts from cooler performance issues (volume, mass, thermal, input/output power, electrical) to system compatibility and integration issues. The following sections summarize the lessons learned during the system integration phase of this shuttle-flight experiment.

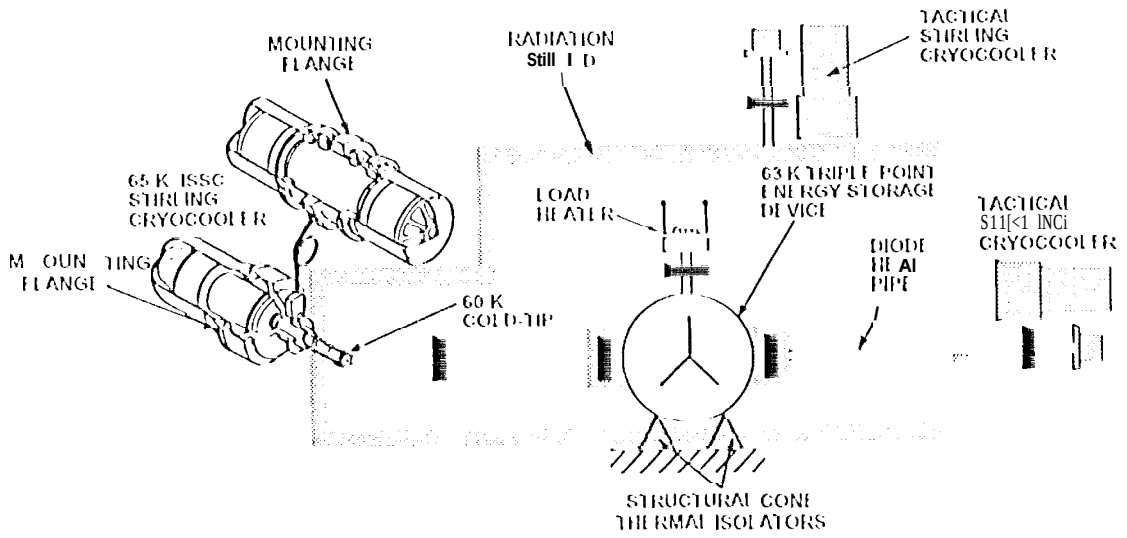


Figure 3. Cryo System Experiment block diagram

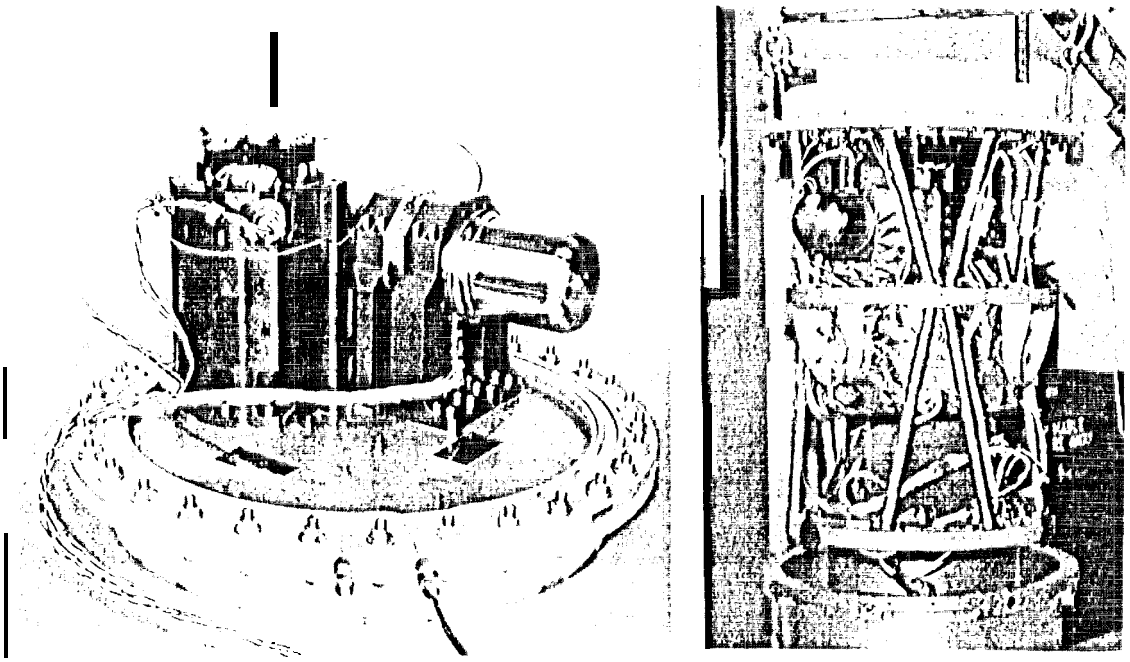


Figure 4. The CSE mechanical assembly attached to the upper end plate.

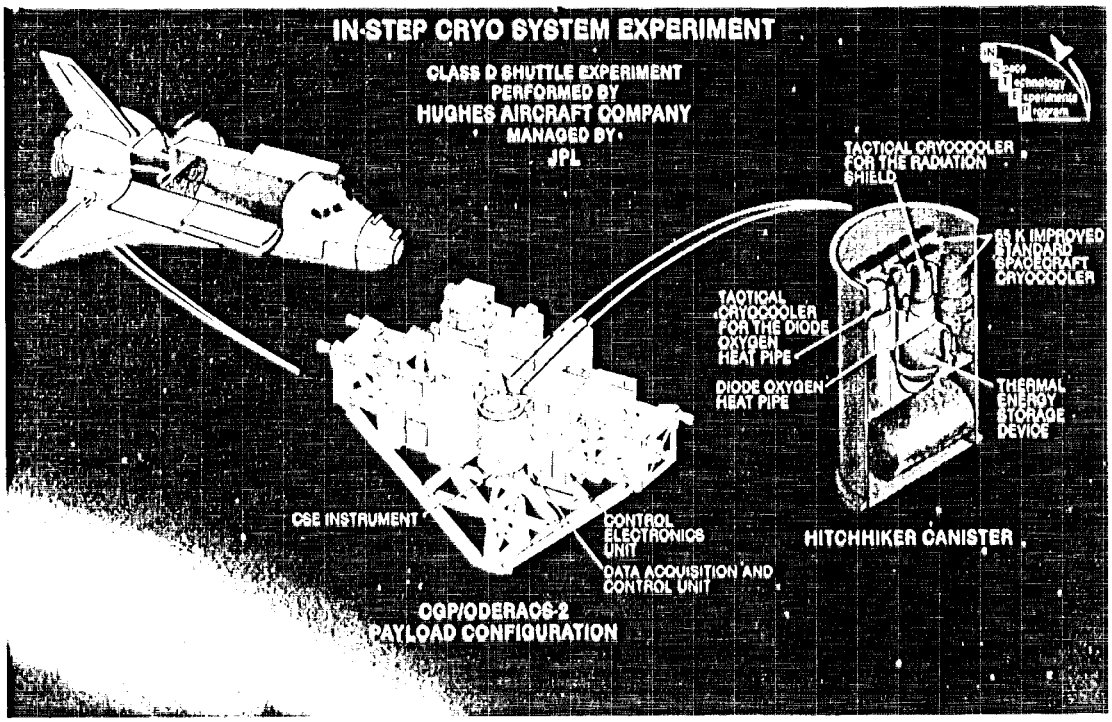


Figure 5. Cryo System Experiment mounted on a Hitchhiker cross-bay carrier within the orbiter.

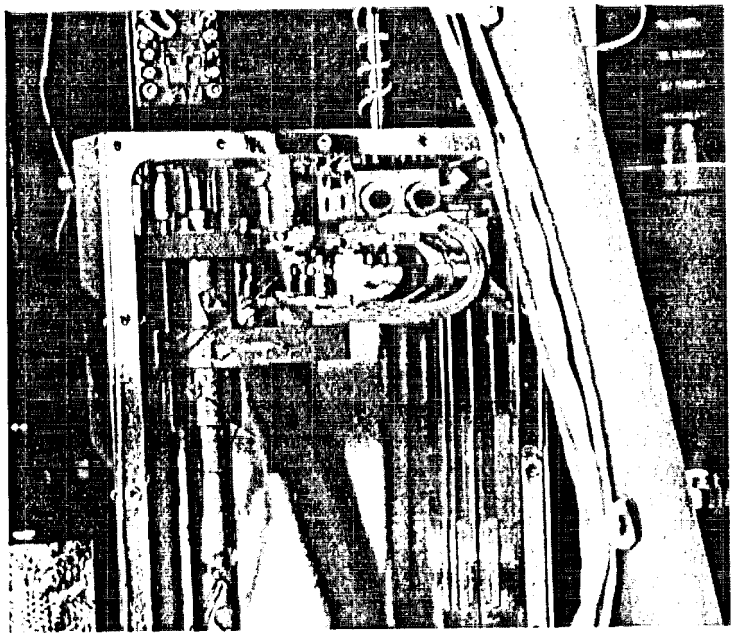


Figure 6. Heat pipe tactical cooler expander, interface mass, and thermal strap.

LAUNCH VIBRATION

A key challenge was to resolve the issue of launch vibration and the impact to the cryocooler expanders. The expander cold finger, a relatively thin-wall, hollow tube, in which the expander piston shuttles, is necessarily thin to minimize parasitic thermal loads due to axial heat transfer. Although by itself the expander cold finger may be considered relatively robust, it is necessary to include a thermal transfer path to provide cooling to the load. This is accomplished by an interface between the expander cold-tip (usually a copper block) and a flexible thermal strap to provide a path to the thermal load. Shown in Figure 6, for example, is the heat pipe tactical cooler expander, interface mass, and thermal strap. Shown in Table 1 are the wall thicknesses, expander material, and supported interface masses at the cold-tip of each type of cooler.

The sizable interface masses attached to the cold finger tip can create a significant dynamic load on the expander cold finger during launch. After completing one axis of the three-axis vibration test, the radiation shield (RS) cooler did not perform correctly. Subsequent investigation of the hardware and removal of the RS cooler revealed a crack at the base of the expander that resulted in loss of helium. The expander cold finger was replaced using parts from a spare tactical cryocooler. The cooler was re-installed onto the UHP of the canister. To prevent future damage due to the launch vibrations, all three of the expanders (ISSC, radiation shield, and heat pipe cryocoolers) were fitted with constraint mechanisms designed to limit cold finger movement. Shown for each of the coolers in Table 2, with and without constraints, are the anticipated bending stresses on the cooler cold fingers during launch. The system was subsequently vibrated to proto-flight qualification levels without incident.

Table 1. Cryocooler material, wall thicknesses, and supported cold-tip masses.

Cooler	Expander material	Expander wall thickness	Mass supported at cold-tip
ISSC	Stainless steel, 17-4 PH	0.152 mm (0.006 in)	250 g (0.55 lb)
Tactical	Stainless steel, 17-4 PH	0.152 mm (0.006 in)	164 g (0.36 lb)

Table 2. Stress summary for tactical cooler and ISSC cold cylinders.

Component	Original design			Revised design w/constraints		
	Pressure -1 Dynamic load			Pressure+1 Dynamic load		
	σ_{max}	$(M.S.)_Y$	$(M.S.)_U$	σ_{max}	$(M.S.)_Y$	$(M.S.)_U$
Tactical cooler cold cylinder	137.2	-0.24	-0.37	55.5	1.1	0.99
ISSC cold cylinder	108.4	0.07	0.02	100.6	0.1s	0.10

$$(M.S.)_Y = \frac{\sigma_{Max}}{1.25\sigma_Y} - 1; \qquad (M.S.)_U = \frac{\sigma_{Max}}{1.25\sigma_U} - 1$$

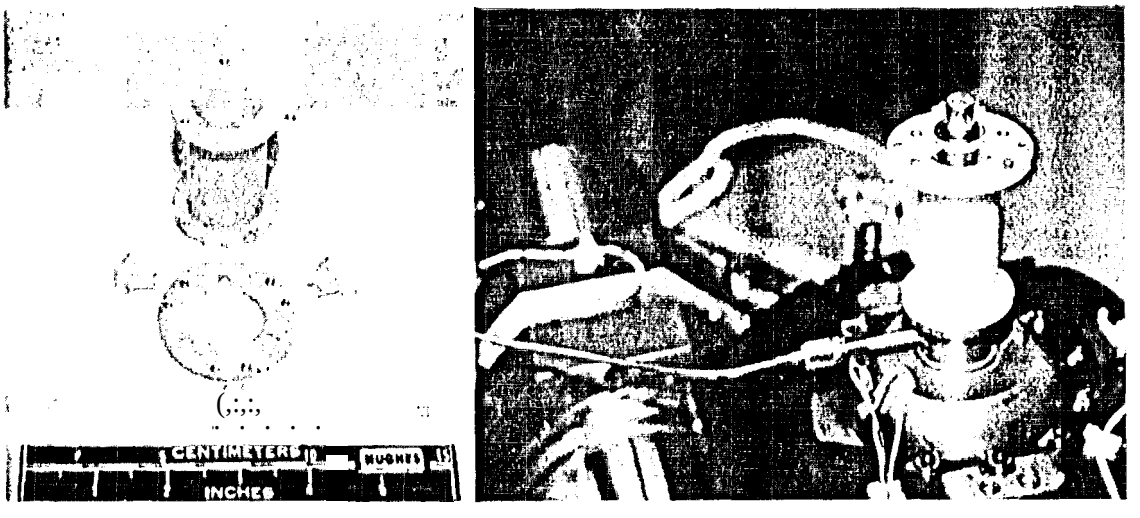
The CSE vibration constraint designs consist of two different G-10 fiberglass configurations: one for the ISSC, and [me. for the two tactical cryocoolers. Shown in Fig. 7 is the design for the ISSC. Note that the orientation of the fiberglass material must be considered to account for differences resulting from thermal contraction/expansion.

ISSC DIAGNOSTIC MEASUREMENT CAPABILITY

Contact resistance

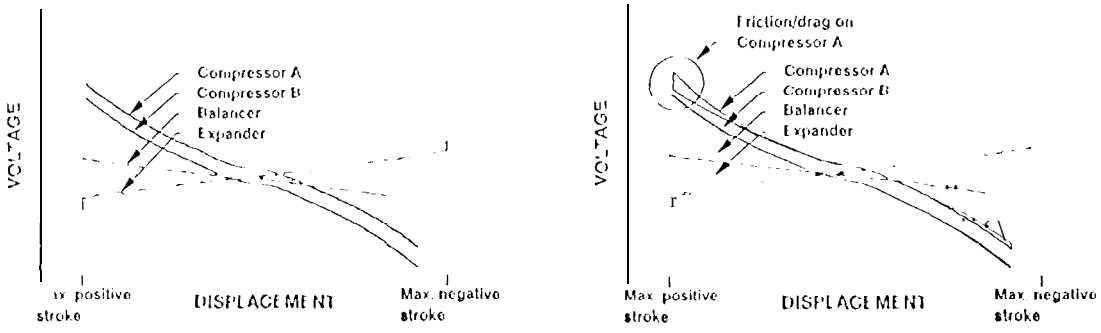
The Hughes ISSC was designed with wry close tolerance, non-contacting, metal-to-metal surfaces between the expander regenerator and its cylindrical coldfinger sidewall, included in the design is an electrical resistance measuring circuit that indicates any momentary or prolonged contact between the expander regenerator and its cylinder sidewall, or its end stops. This circuit has proved essential in a number of diagnostic situations during the system integration process.

For example, during initial syslc.in checkout, the ISSC expander piston was being stroked using dc power to verify operation prior to a full system cooldown. When the piston approached to within ?() % of maximum stroke, contact between the regenerator and cold finger wall was



a). Hughes ISSC vibration constraint. b). Hughes ISSC vibration constraint installed

Figure 7. CSE vibration constraint for the Hughes ISSC cryocooler.



a). No contact or drag.

b). Friction at maximum positive stroke of compressor A,

Figure 8. Hysteresis plots of piston position as a function of current.

observed. The cause., described in detail in the. next section, resulted from an excessive side load imposed cm the cold finger by the thermal strap. The circuit was also useful when re-attaching the modified thermal strap to verify acceptable side loads on the cold finger. Additionally, the circuit was invaluable during system integration to verify no-contact in the final as-built state and during environmental extremes.

Hysteresis

The cooler compressor and expander designs incorporate the capability to use the position of the piston as a function of drive current to obtain a plot of the hysteresis or drag of the pistons against the. cylinder sidewall, Figure 8 shows two examples of hysteresis plots in which Fig. 8a represents a normal response with no flit.tion or contact at either compressor A or B or at the expander or its balancer, and Fig. 8b represents friction or drag on compressor A at maximum positive stroke and drag starting to occur on compressor B at maximum negative stroke. This circuit was used throughout systc.in integration to validate cooler clearances before and after component and system vibration and thermal vacuum testing. This validation is the only means to confirm that the compressor piston suspension system is operating satisfactorily following system integration and qualiflc--alim tests.

HIGH-COMPLIANCE THERMAL S'J'J{A]'

Design Of the thermal strap connecting the. cooler cold-tip to the cryogenic load must be carefully considered to achieve. system performance goals. The two critical characteristics Of the strap are thermal conductance and mechanical stiffness. It is desirable. that the conductance be high (> ().S W/K) due. to the strong dependence of cooler efficiency on cold-tip temperature, and that the spring rate be low due to the. sensitivity of the expander cold finger to mechanical side. loads. II is important that these two conflicting alternatives be carefully evaluated when designing the the.]mal strap. The original ISSC thermal strap was designed to provide a ?.() W/K conductance and a 5.8 lb/in spring rate in the. radial direction.

When the ISSC was physically integrated onto the canister UHP and mechanically attached to the the.rmal strap, the expander piston was found to contact the cold finger side wall. With the thermal strap connection adjusted to a free position and the system under vacuum at room temperature, the. contact reappeared when the. piston traveled to 80% of maximum stroke.

Table 3. Predicted cold-tip loads for the original and modified thermal strap designs.

Load	Original (lb)	Modified (lb)
Vacuum induced lid load - 0.01?" cold-tip movement	0.?5	0.12
Thermal contraction load - 0.020' contraction of "1'1<1'	0.43	0.?0
Total anticipated Loads	0.68	0.32
Unanticipated mechanical integration load - Preload	>1.1	0.51
Total Load	>1.4	0.83
Margin @ 1.4 lb, max. acceptable load	0%	40%

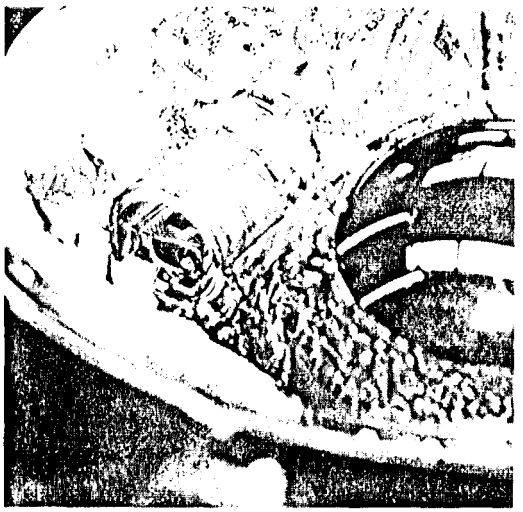
The contact remained through the, TRP mass cooldown to approximately 80 K. The contact persisted with the system warmed up to room temperature, but still under vacuum. When backfilled to mbit.nt pressure, the contacting disappeared, and then reappeared when vacuum credit ions were. reestablished.

The measured deflection at the center of the. UHP due to vacuum was on the order of 0.00'7", which resulted in a calculated ISSC cold-tip movement Of 0.01 ?". The deflection due to thermal contraction of the simulated TRP mass was calculated to be on the order of 0.02.0".

Table 3 summarizes the. predicted cold-tip loads for each of the contributive. forces for both the original and modified thermal strap designs. The "unanticipated loads" arc the result of unavoidable, but small alignment errors introduced when mechanically integrating the. thermal strap into the. system. A lesson learned is that this load must be allowed for when specifying tile. acceptable thermal strap spring rate. The conclusion was that the 1.4 lbs maximum side load imposed on the. expander cold-tip by the original thermal strap was too large.

It was shown that due to the close tolerances within tile. ISSC, the total combination of forces from the thermal strap preload, thermal contraction loads and the additional force created by the canister lid deflection, was sufficient to cause. contact. The force imposed by the thermal strap preload could not be directly measured, but the small amount of force resulting from the UHP deflection indicated that the original thermal strap was producing unacceptable side-load forces that resulted in contact under vacuum conditions. Although the canister vacuum would not rc.suit in lid deflection once in space. (zero differential pressure), the relatively small amount of additional force created by the vacuum indicated that the thermal strap spring constant was too high.

Recalculating the. required the.rlllal conductance and verifying margin adequacy enabled consideration of reducing the cross-sectional area of the thermal strap to lower the resultant side-load force on the cold-tip. Two solutions were proposed: reducing the cross-sectional area Of the. existing copper foil the.rmal strap; and using an alternate thermal strap based on a copper braid material as the. flexible element. 'The goal was to increase. the mechanical compliance at the expense of somewhat lower thermal conductance, and thereby to reduce the side-load force imposed on the cold finger.



a). Original design.

b). Modified design.

Figure 9. Thermal straps for the CSE Hughes ISSC.

The original design, shown in Fig. 9a, was for a thermal resistance of 0.5 K/W and consisted of 39 layers of 0.004" x 1.5" of OFHC copper, giving a total cross-sectional area of 0.234 in^2. The layers at each end of the strap were soldered together. Additionally, the TRP interface holes were slotted to enable alignment adjustments. Figure 9b shows the design concept after modifications that resulted in a thermal resistance of 1.1 K/W and had 34 layers of 0.004" x 0.8" of OFHC copper; the resulting total cross-sectional area is 0.109 in^2.

Consideration was also given to an alternate design that involved the use of 32 compliant copper braids 1 in. in length whose design was targeted at a thermal resistance of 1 K/W. The ends of the copper braids were soldered and bolted to copper lugs. Figure 10 shows the modified design and identifies the reference axes used in Table 4. Table 4 is a comparison between the measured thermal strap spring rates for the original, modified and alternate-braid designs. It can be seen that the measured free-free spring rate of the alternate-braid concept in the Y-Axis is extremely stiff. As a result this design was not used.

The modified thermal strap was reinstalled into the CSE. There was no contact between the displacer and cold cylinder sidewall after installation of the strap, or after attaching the UEP and establishing vacuum conditions within the canister. The mechanical system was vibrated to proto-flight qualification levels and subjected to the thermal vacuum test without incident.

Table 4. Measured thermal strap spring rate for the original, modified, and alternate braid designs.

Axis	Original free-free (lb/in)	Modified free-free (lb/in)	Alternate-braid free-free (lb/in)
X	N/A	1.4	7.3
Y	4.0	1.6	28.5
Z	11.9	4.2	3.5

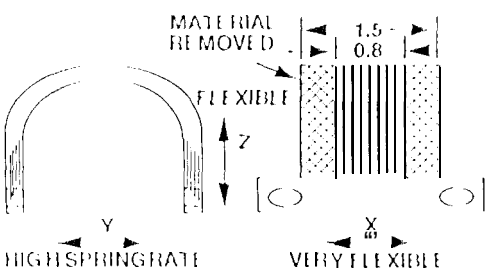


Figure 10. Modified design and references axes.

'J'able 5. Comparison of mass of coolers vs. thei mal mass required for heat transfer.

Item	ISSC	Rad shield cooler	Heat pipe cooler
Cooler	19 Kg (41.8 lb)	2.3 Kg (5,0 lb)	2.3 Kg (5.0 lb)
Thermal Path	23.4 Kg (51.4 lb)	5.7 Kg (12.6 lb)	7.6 Kg (16.6 lb)
TOTAL	42.4 Kg (93.2 lb)	8.0 Kg (17.6 lb)	9.8 Kg (21.6 lb)

MASS CONSIDERATIONS

Another system integration issue. deals with the mass of the. ISSC. "1'0 reduce the, mass by approximately 15%, the. w second-generation cryocoolers are using integral back-to-back compressor- and integral back-to-back expander-designs. The problem is that, although the. mass/volume has been reduced, consderation must be given to the. thermal paths needed to conduct away the waste heat resulting from the input electrical power. 1 'or the CSE, the thermal path to the radiator, the UEP of the canister, resulted in a significant mass penalties for each of the three coolers, Shown in Table 5 is the mass of each cooler and the additional mass of the copper thermal path needed to maintain acceptable cooler operating temperatures.

It should be noted, however, that there arc reasons for initially excluding the. weight of the thei mal path in a cryocooler mass calculation: 1) the required mass is dependent upon the. distant.c from the compressor and expander to the heat rejection surface; 2.) the required operating temperature (lower temperature, larger cross-sectional area) varies with the mission and radiator design; and 3) the ambient temperature to which the. heat is being rejected also varies with design. The key to an optimum solution is to recognize that these. factors will have to be addressed during the system integration and to include a substantial margin for mass.

CONTAMINATION/PARASITICS

One of the issues on a short mission is the effect of contaminants from the surrounding environment in the proximity of a cryogenic experiment. On a short shuttle flight (typically 7 to 10 days), there may not be adequate time for contaminants to dissipate in the vacuum environment. Since. contaminants will migrate. toward the coldest regions, parasitic thermal radiation loads will increase, and the test results may be negatively affected. To minimize contaminants and thermal radiation 1o ssc.s from the simulated 'l'J{l', an Ml .l-covered, anodized aluminum shield is maintained at 12.0 K by a tactical cooler.

Prior to installation in the canister, the mechanical system was vacuum baked at Hughes to remove all contaminants. The canister was them backfilled with clean, dry nitrogen gas at approximately 3 psig. This "clean" condition will be maintained within the canister until after launch when the shuttle bay doors arc opened. At this time the vent valves located on the. canister lower end plate will be opened to establish vacuum conditions and the experiment timeline will be started.

SYSTEM OPERATIONS/SOFTWARE

The CSE is controlled by firmware., programmable read only memory (PROMs), in the flight electronics and is designed to automatically follow a sequence of programmed steps. The CSE software gathers and storm sensor data, controls and sequences the ISSC and tactical coolers to turn on and off, monitors and controls the operating parameters of tile ISSC, and provides an operator interface for the experiment timeline.

Using Customer Ground Support Equipment (CGSE), an operator is able to monitor the progress of the experiment during the. flight. Additionally, the operator can also use the CGSE to alter or modify the baseline system operat ions at any time during the experiment timeline.

The firmware control approach permitted the design of the electronics, typically a long-lead item, to proceed before all the requirements and the. experiment operating parameters were completel y defined. During system integration, collections to the time.iinc and revisions to the experi ment operat ions were able to be incorporated into the firmware without having to modify the. hardwar e.

Controlling the system operations by software allowed for maximum flexibility in defining the system experiment, the system operational sequence, and the operating parameters of the ISSC coder. For an experiment whose goal is to evaluate an enabling technology, this flexibilit y was invaluable.

ELECTRICAL INTERFACE REQUIREMENTS

Electrical interface requirements for shuttle flight experiments are relatively straightforward, and the. CSE used the NASA-GSFC Hitchhiker Customer Accommodations & Requirements Specifications. The focus, for tertiary Class 11 experiments, is on shuttle and personnel safet y; fusing is provided to protect the. shuttle, not the experiment. 1 'rem the experiment's point of view, the source current is li mited onl y by the fuse.s. The primary issue. is whether or not to provide experiment safeguards and circuitry to preclude excessive current that could result in a Mown fuse, and possible data loss or inability to complete the experi ment.

Hughes chose to follow an alternat ive path: characterize the electrical hardware, understand the operational timeline to preclude operational sequences that could result in excessive currents, and utilize electronics har dware, software and firmware. that would allow relatively easy reprogramming of the experiment timeline. Finally, Hughes validated the experiment during the thermal vacuum test by following the experi ment operational t i meline.

Nevertheless, a key concern was the potential for the "simulated shuttle power source" to provide excessive current or voltage to sensitive circuits during electrical checkout and system integrat ion. To preclude subjecting the flight fuses to electrical stresses induced by shorts or low-resistance paths, Hughes utilized laborator y power supplies that included both current- and voltage-limiting capability. These power supplies provided protection for the flight fuses on several occasions when the. CSE c.x])cric.need system shutdown and reset modes during the. checkout, integration and initial thermal vacuum testing.

1 'or the pre-integration elect rical checkout at GSFC, H ughes has requested power supplies that include both current- and voltage-limiting capability. After integration on the cross-bay earl-icr with other experiments, the CSE will receive power, which will *not* be limited, through the 1 litchhiker avionics. For subsequent pre-flight electrical checkout at KSC, power will continue to be supplied through the Hitchhiker avionics.

SAFETY

Pressurized Components vs. Pressure Vessels

For the CSE, an important safety issue was the classification of the cryocoolers and heat pipe. These components were examined by JPL using criteria from NSTS1700.7B that addresses pressure vessels, pressurized components, and piping, tubing, and fitti ngs. With concurrence from the Johnson Space Center, the cryocoolers and heatpipe were classified as pressurized components. As such, both cryocoolers and the heat pipe. required only proof testing.

Table 6. Orbital comparison between CSE and LPE.

	Elevation	Inclination	Comment
CSE	200 nm	51 °	The CSE orbit crosses the SAA and spends - 5% of its time in the SAA during a 7-day period.
LPE	135 nm	2.8.5 "	The LPE orbit crosses the SAA and spends ~ 2 % of its time in the SAA during a 7-day period.

This classification resulted in a significant cost savings to the Project. It meant that the Project was exempt from providing additional items specifically for burst testing and life-cycle. testing. in addition to the. cost savings, there were savings associated with preparation time for the Flight Safe.ty Reviews and the extensive traceability (paperwork) required to validate pressure vessel qualification

Radiation Exposure Due to the Natural Space Environment - Risk Minimization

A challenge for the CSE was in procuring a cent rol and data acquisition system that would function in the natural space environment. Candidate. vendors were restricted to those having previous shuttle experience; two candidates were located, one on the West coast and one on the East coast. The West coast candidate was selected cm the basis of lower cost and proximity to Hughes. Additionally, since J}'], had used Titan electronics for their Lambda Point Experiment (LPE), it was believed that use of this flight-proven, off-the-shelf design would result in mini mum development costs for CSE.

Since the shuttle flight and orbit were not known at the time of procurement, and because the electronics were a long-lead item, the specification was released using the LPE natural space environment. When the shuttle flight and orbit were announced, it was learned that the orbit was higher and the inclination was greater than original] y anticipated (see Table 6), resulting in a significantly more severe radiation environment. The CSE will be subjected to a proton flux 2 to 3 times greater than LPE, although the cosmic ray environment for both missions is similar.

The key issue was not total integrated radiation dosage, which for a 7- 10 day shuttle mission was negligible, but rather, susceptibility of the electronics to Single Events Effects (SEE), a combination of Single Event Upset (SEU) and Single. Event Latchup (SEL).

A 32K x 8 SRAM hat! been categorized as " high risk, " being, very soft to both SEU and SEL; and approximatel y 40 parts had been identified for which no SEE susceptibility data were available at J]']. . Replacing these parts within the CSE control and data acquisition system would require. a major redesign that was unacceptable in terms of both cost and schedule. Substitution of equivalent radiation-hardened parts was not practical based upon lack of compatibility, lack of availability, and relative.ly high cost.

Testing to validate. acceptable performance in the nat ural space environment was too costly, and not consistent with the delivery schedule. Additionally, measured performance of the parts would not resolve the issue, only provide more. detailed knowledge about the parts.

An assessment team, consisting of circuit designers/system engineers from the Hughes CSE Project, and electronics parts reliability and natural space environment (radiation) specialists from J]'],, was tasked with resolving the problem. '1'heir approach was to: 1) determine the shuttle. exposure profile to the South Atlantic. Anomaly (SAA), 2.) re-estimate the. radiation environment and CSE shielding impacts, 3) develop a better understanding of the. questionable. parts, associated manufacturers and circuit technology, 4) reduce the parts list by discarding non-

susceptible parts and assess risk of SEU and SEL for those remaining, 5) identify suspect part flight histories and radiation data, 6) identify and assess impact of hardware and software modifications for enhanced immunity, problem determination and recovery, and 7) establish and compare the. radiation environment between LPE and CSE.

The CSE shuttle exposure. profile. consists of SAA exposure during approximately 5 % of the mission, usually in blocks of 9 consecutive orbits, with cosmic rays accounting for another 5%. The environment is severe enough (5 - 100 MeV) to cause SEUs in non-qualified parts (parts having a threshold of 0.4 McV arc expected to experience 0.1 upsets per 24 -hours of exposure.). Shielding was found to provide little improvement.

1 electronic parts pare down resulted in eliminating a number of parts and identifying 21 remaining parts with unknown susceptibility. Of these 21 remaining parts, 10 were eliminated by Hughes due to circuit function (resets or loss of which is insignificant). Further review by Hughes of SEE circuit operational impact (hardware/flight software modifications) indicated that the. use of bus architecture precluded meaningful assessment, anti that no cost-effective modifications were identified.

The LPE exposure. to radiation was less severe. LPE hardware flew at 135 nautical miles (rim) versus 2.00 nm expected for CSE. At this altitude, the radiation environment intensity (proton flux) doubles with each 100 nm increase in altitude. LPE cxpcrie.need SEUs.

The assessment team indicated that 11 identified parts may experience SEUs (<0.1 SEU per 24 hours of exposure, total mission exposure determined to be 7 hours, 42 minutes). It is costly and late in the program to ret refit a change (Class D experiment assumes manageable risk). To minimize risks associated with the SAA, the assessment team recommended modifications to ground support procedures to: 1) allow operator intervention and custom commanding, 2) utilize power standby or power off modes during SSA exposure (30-minutes down time maxi Inure), anti 3) target benign experiment operations during SAA exposure (critical data gathering to occur during non-exposed orbits).

"1 'he lesson learned is that when a project cannot afford radiat ion-hard electronics, it is necessary to understand the natural space environment based upon the shuttle flight inclination and orbit as early as possible. to allow for a complete assessment of the single e.vent effects upon the electronic parts.

Reliability/Quality Assurance - in-1'rocm Checks/Inspection Points

A challenge with any new design is the establishment of meaningful and timely in-process inspections points, as well as performance. and verification checks. Sonic of the checks that were performed on the CSE included mechanical inspections and fit checks of the hard ware during fabrication, These checks minimized the mechanical integration problems associated with a first-time build. other checks that were. performed included the verification of sensors, continuity checks on the system cabling, and electronic checkouts. Additionally, the software was developed and checked out using an emulator prior to the final design being burned into the flight μprocessor.

in some cases additional checks and inspection points were established after problems had been discovered. One example occurred during the mechanical integration of the thermal strap between the. ISSC expander cold-tip and the simulated TRP, previously discussed. The piston/cylinder contact signal was monitored to validate that the thermal strap was reintegrated in a position with no side loads.

Another example occurred when the vibration constraints were added to the cryocooler expanders after the RS tactical cooler expander failed during the vibration test. In this case, the

tactical cooler input power was monitored for any changes during the reassembly proms. An increase. in input power was indicative that a side. load configuration existed, and the constraint needed adjust ment.

To minimize problems associated with the CSE mechanical integration, Hughes fabricated a full-scale mockup of the, experiment canister and t he mechanical components. A non-flight UHP was used as a template for fit checks of the flight structural members, cryocooler heat sinks and mechanical component clearances during hardware fabrication, An additional benefit of the full-scale mockup was in being, able to chink the. electrical cable routes and lengths prior to installation of the connectors.

The lesson learned was that a well-thou~ht out inspection and in-process verification plan, coupled with a full-scale mockup, saved valuable amounts of time in a fast-track program by anticipating problem areas, identifying critical assemblies and processes, and verifying that the assembly/process was correct before proceeding with the final assembly. Additionally, when a problem did occur, revisiting the inspection timeline and revising inspection points were included as a part of the corrective action process to preclude the problem from reappearing.

SUMMARY

The NASA 1 N-S"I'Ill' CSE has provided an opportunist y to identify and resolve a number of cryogenic system interface integration issues that normally would not be addressed until a cryogenic cooling system was selected for integration into a multi-year space mission. It is hoped that the challenges and problems encountered in the system integration of these enabling technologies will provide insight to system designers of imaging-in strument systems incorporating long-life cryogenic cooling systems.

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