**SPACE
DIVISION**

ESP ENERGY SYSTEMS PROGRAMS
MULTI-HUNDRED WATT PROGRAM

Document No. GESP-7107
GEMS-418

MULTI-HUNDRED WATT RADIOISOTOPE THERMOELECTRIC GENERATOR PROGRAM

Part I

Annual Report

1 January 1973 - 31 December 1973

MASTER

**ENERGY SYSTEMS PROGRAMS
SPACE DIVISION
MULTI-HUNDRED WATT RTG PROGRAM
KING OF PRUSSIA PARK
P.O. Box 8661 • Philadelphia, Penna. 19101**

GENERAL ELECTRIC

DISCLAIMER

This report was prepared as an account of work sponsored by an agency of the United States Government. Neither the United States Government nor any agency Thereof, nor any of their employees, makes any warranty, express or implied, or assumes any legal liability or responsibility for the accuracy, completeness, or usefulness of any information, apparatus, product, or process disclosed, or represents that its use would not infringe privately owned rights. Reference herein to any specific commercial product, process, or service by trade name, trademark, manufacturer, or otherwise does not necessarily constitute or imply its endorsement, recommendation, or favoring by the United States Government or any agency thereof. The views and opinions of authors expressed herein do not necessarily state or reflect those of the United States Government or any agency thereof.

DISCLAIMER

Portions of this document may be illegible in electronic image products. Images are produced from the best available original document.

MULTI-HUNDRED WATT RADIOISOTOPE THERMOELECTRIC GENERATOR PROGRAM

Part I

Contract No. AT29-2-2831

Annual Report

1 January 1973 - 31 December 1973

prepared for:

**US Atomic Energy Commission
Space Nuclear Systems Division
Washington, D.C. 20545**

by:

**ENERGY SYSTEMS PROGRAMS
SPACE DIVISION
MULTI-HUNDRED WATT RTG PROGRAM
KING OF PRUSSIA PARK
P.O. Box 8661 • Philadelphia, Penna. 19101**

NOTICE
This report was prepared as an account of work sponsored by the United States Government. Neither the United States nor the United States Atomic Energy Commission, nor any of their employees, nor any of their contractors, subcontractors, or their employees, makes any warranty, express or implied, or assumes any legal liability or responsibility for the accuracy, completeness or usefulness of any information, apparatus, product or process disclosed, or represents that its use would not infringe privately owned rights.

GENERAL ELECTRIC

THIS DOCUMENT IS UNLIMITED

April

NOTICE

"THIS REPORT WAS PREPARED AS AN ACCOUNT OF WORK SPONSORED BY THE UNITED STATES GOVERNMENT. NEITHER THE UNITED STATES NOR THE UNITED STATES ATOMIC ENERGY COMMISSION, NOR ANY OF THEIR EMPLOYEES, NOR ANY OF THEIR CONTRACTORS, SUBCONTRACTORS, OR THEIR EMPLOYEES, MAKE ANY WARRANTY, EXPRESS OR IMPLIED, OR ASSUMES ANY LEGAL LIABILITY OR RESPONSIBILITY FOR THE ACCURACY, COMPLETENESS OR USEFULNESS OF ANY INFORMATION, APPARATUS, PRODUCT OR PROCESS DISCLOSED, OR REPRESENTS THAT ITS USE WOULD NOT INFRINGE PRIVATELY OWNED RIGHTS."

TABLE OF CONTENTS

Section		Page
	PART I	
1	INTRODUCTION AND SUMMARY	1-1
1.1	Introduction	1-1
1.2	Summary	1-1
3	SAFETY	3-1
3.1	Safety Test Program	3-1
3.1.1	Test S-1, Explosion	3-1
3.1.2	Test S-2, Liquid Propellant Fire	3-2
3.1.3	Test S-3, Solid Propellant Fire	3-8
3.1.4	Test Planning	3-10
3.1.5	Other FSA/PISA Tests	3-21
3.1.6	Safety Sequential Tests (SST's)	3-21
3.1.7	Safety Verification Tests (SVT's)	3-22
3.2	Safety Analysis and Reports	3-25
3.2.1	Explosion Environment	3-25
3.2.2	Nuclear Criticality	3-26
3.2.3	Liquid Propellant Fireball/Afterfire	3-26
3.2.4	Abort Environments - General	3-28
3.2.5	Pad Abort Blast Analysis	3-32
3.2.6	Flight P74-1 Mission Requirements/ Characteristics	3-36
3.2.7	Supporting Data for Safety Reports	3-38
3.2.8	V, γ Probability Study	3-39
3.2.9	Re-entry Breakup Study	3-42
3.2.10	Updated Safety Analysis Report	3-46
3.3	Operational Safety	3-46
3.3.1	Radiological Protection Program	3-46
3.3.2	Radiological Safety Training	3-47
3.3.3	Safety Analysis Report of MHW Isotope Heat Source Operations at GE - Valley Forge Space Center	3-47
3.3.4	Safety Analysis Report for Packaging (SARP)	3-48
3.3.5	Test Procedures	3-48
3.3.6	IHS Test Facilities	3-49
3.3.7	Operational Safety Analysis	3-50

TABLE OF CONTENTS (Cont'd)

Section		Page
4	RTG SYSTEMS	4-1
4.1	System Interface, Environmental and Performance Requirements	4-1
4.1.1	Requirements Documentation and Data	4-1
4.1.2	Thermal Analyses and Test	4-2
4.1.3	Dynamic Analyses and Test	4-3
4.2	System Analyses	4-4
4.2.1	Orbital Thermal Analysis	4-4
4.2.2	LES 8 & 9 MHW Thermal Integration Analyses	4-6
4.3	Systems Integration	4-25
4.3.1	Thermal Integration	4-26
4.3.2	Mechanical Integration	4-27
4.3.3	Electrical Integration	4-27
4.3.4	Procedural Integration	4-28
5	HEAT SOURCE	5-1
5.1	Heat Source Design Requirements	5-1
5.1.1	Operational Requirements and Constraints	5-2
5.1.2	Safety Constraints	5-4
5.2	Reference Design	5-7
5.2.1	Design Description	5-8
5.2.2	Assembly Procedure	5-13
5.3	Isotopic Fuel	5-16
5.3.1	Fuel Form	5-16
5.3.2	Basic Requirements	5-16
5.3.3	Fuel Characterization	5-16
5.3.4	Compatibility Tests	5-35
5.3.5	Fuel Fabrication	5-36
5.3.6	Fuel Simulant	5-52
5.4	Post Impact Sphere Assembly (PISA)	5-53
5.4.1	Design Objectives and Changes/Fabrication Experience	5-54
5.4.2	PISA Venting	5-56
5.4.3	Impact Testing	5-67
5.4.4	Impacts at LASL	5-73
5.5	Fuel Sphere Assembly (FSA)	5-79
5.5.1	Description	5-79

TABLE OF CONTENTS (Cont'd)

Section	Page
5.6 FSA Integration Hardware	5-84
5.7 Aeroshell	5-87
5.8 Outer Clad	5-89
5.8.1 Design	5-89
5.8.2 Fabrication	5-92
5.9 Emissivity Sleeve	5-95
5.9.1 Description	5-95
5.9.2 Function	5-95
5.9.3 Assembly	5-95
5.10 Gas Management Subsystem	5-95
5.10.1 Subsystem Requirements	5-97
5.10.2 GMS Design	5-99
5.10.3 Sequence of GMS Operations	5-106
5.11 HSA Analysis and Test	5-109
5.11.1 Operational, Assembly and Launch Temperatures	5-109
5.11.2 Pad Abort Thermal Response.	5-113
5.11.3 Post Impact Temperatures	5-146
5.11.4 Re-entry Aerothermal Analyses.	5-156
5.11.5 Aerodynamic Studies	5-254

PART II

6	CONVERTER	6-1
6.1	Converter Requirements	6-1
6.1.1	Performance Requirements	6-1
6.1.2	Physical Requirements.	6-2
6.1.3	Design Requirements	6-2
6.1.4	Environmental Conditions	6-3
6.2	Converter Reference Design	6-4
6.2.1	Converter Configuration	6-4
6.2.2	Thermopile	6-9
6.2.3	Outer Shell Subassembly	6-15
6.2.4	Heat Source Support System	6-17
6.2.5	Instrumentation and Cabling	6-22
6.2.6	Generator Assembly.	6-23
6.2.7	Converter Materials.	6-26
6.2.8	Converter Weight Breakdown.	6-28
6.3	Converter Design and Development.	6-32
6.3.1	E1 and E2 Converter Design	6-32
6.3.3	Cables and Connectors	6-47

TABLE OF CONTENTS (Cont'd)

Section	Page
6.4 Design and Development (RCA)	.6-50
6.4.1 Module and Converter Testing	.6-50
6.4.2 Converter Design and Analysis	.6-50
6.4.3 Material Analysis and Process Development	.6-76
6.4.4 Silicon Nitride Chemical Vapor Deposit (CVD) Hot Shoe Coatings	.6-82
6.5 Degradation and Long-Term Performance Analysis	.6-88
6.5.1 Thermoelectric Couple Effects	.6-88
6.5.2 Performance Projections	.6-119
6.6 Structural Analysis and Test	.6-153
6.6.1 Thermal Stress Analysis	.6-153
6.6.2 Stress Analysis of RTG's in the Stacked Configuration	.6-153
6.7 Reliability	.6-161
6.8 Component and Module Testing	.6-163
6.8.1 C-11G Test Program	.6-163
6.8.2 Dynamically Simulated RTG Models	.6-175
6.8.3 Thermal Simulators	.6-176
6.8.4 C-10 Oxygen Back Diffusion Tests	.6-177
6.8.5 D5A3 Dynamic Module Test	.6-182
6.9 Test Bed Converter No. 3 (TBC-3) Testing	.6-185
6.9.1 Acceleration Test (1-3-73 to 1-11-73)	.6-185
6.9.2 Post-Dynamic Performance	.6-186
6.9.3 Special Development Tests	.6-190
6.9.4 Transient Venting Tests (2-24-73 to 3-19-73)	.6-194
6.9.5 Leak Detection (3-6-73 to 3-8-73)	.6-197
6.9.6 Disassembly/Inspection (3-20-73 to 3-30-73)	.6-198
6.9.7 Magnetic Signature Evaluation (4-2-73 to 4-11-73)	.6-199
6.9.8 Life/Stability Test and Associated Tests	.6-204
6.9.9 Refurbishment to TBC-3A Condition (8-3-73 to 9-5-73)	.6-212
6.9.10 Second Life/Stability Test	.6-214
6.10 Test Bed Converter No. 4 (TBC-4) Testing	.6-216
6.10.1 Receiving Inspection Tests	.6-216
6.10.2 Mass Properties Tests	.6-217
6.10.3 Degassing Process	.6-218
6.10.4 Thermal-Vacuum I-V Characteristic Measurement	.6-219
6.10.5 Thermal-Vacuum Stability Test	.6-219

TABLE OF CONTENTS (Cont'd)

Section		Page
8	PRODUCT ASSURANCE	8-1
8.1	Reliability	8-1
8.1.1	Reliability Requirements	8-1
8.1.2	Failure Mode, Effects and Criticality Analysis (FMECA) of the RTG and Supporting GSE	8-2
8.1.3	Failure Mode Analysis	8-2
8.1.4	Subcontractor Reliability Activities	8-12
8.1.5	Failure Analysis and Reporting	8-12
8.1.6	Materials Data Program	8-12
8.1.7	Reliability Assessment.	8-12
8.2	Configuration Management.	8-14
8.3	Quality Assurance	8-15
10	CONVERTER FABRICATION.	10-1
11	ACCEPTANCE TESTING	11-1
11.1	Acceptance Test Plans and Procedures	11-1
11.1.1	Test Plans	11-1
11.1.2	Test Procedures	11-1
11.2	Q-1 ETG Acceptance Testing.	11-4
11.2.1	Receiving Inspection.	11-4
11.2.2	Thermal-Vacuum Performance	11-6
11.2.3	Air Performance/"Hot" Pressure Decay	11-9
11.2.4	Q-1 Additional Testing	11-11
12	QUALIFICATION TESTING	12-1
12.1	Qualification Test Plans and Procedures	12-1
13	GROUND SUPPORT EQUIPMENT	13-1
13.1	Electric Heat Source (EHS)	13-1
13.1.1	EHS Design	13-1
13.1.2	Electric Heater Units	13-4
13.1.3	GDEHS Storage and Handling Containers	13-8
13.1.4	EHS Acceptance Test Fixture.	13-8
13.1.5	EHS High Temperature Vacuum Furnace	13-9

TABLE OF CONTENTS (Cont'd)

Section	Page
13.2 Power Supply Consoles	13-17
13.3 Readout Consoles	13-17
13.4 RTG Shipping Container.	13-18
13.4.1 Design Changes	13-18
13.4.2 Fabrication	13-18
13.4.3 Documentation	13-18
13.4.4 Development Testing.	13-20
13.4.5 Acceptance and Certification Testing	13-20
13.5 RTG Loading and Assembly Station (LAS)	13-21
13.5.1 Description	13-21
13.5.2 Procurement	13-36
13.5.3 Design	13-36
13.5.4 Pre-Acceptance Testing	3-41
13.5.5 Installation and Testing of LAS No. 1 at GE-Valley Forge	13-42
13.5.6 LAS No. 2 Activities.	13-44
13.6 Portable Test Unit (PTU)	13-45
13.7 Shorting Modules	13-45
13.8 RTG Handling Sling	13-46
13.9 Storage Protection Container (SPC)	13-46
13.9.1 Purpose	13-46
13.9.2 Design Description	13-46
13.9.3 Design Requirements.	13-47

LIST OF ILLUSTRATIONS

Figure	PART I	Page
3-1	8-Pack Installed on Support Pedestal	3-3
3-2	Oven Closed Around Specimen in Shock Tube	3-4
3-3	S-1 FSA's Following Test	3-5
3-4	S-2 Test Setup Prior to Live-Simulant Test (Specimens Near Drop Point)	3-6
3-5	MHT 16, 17, 18 and 19 Subsequent to Test S-3	3-7
3-6	Solid Propellant Fire Test-Dual Proximity Test Setup	3-11
3-7	Pre-Heat Oven and Test Specimen Holder.	3-12
3-8	Propellant Block for Contact Test	3-13
3-9	Recovered Contact Test Specimens	3-14
3-10	Recovered Dual Proximity Test Specimens	3-15
3-11	PISA From MHT-5 After Contact Test	3-16
3-12	FSA MHT-9 After Dual Proximity Test	3-17
3-13	SVT Proposed Plan	3-24
3-14	Alternate SVT Plan	3-24
3-15	Shock Wave and Particle Velocity vs Distance From Explosion Center Transtage	3-25
3-16	Static Overpressure vs. Distance From Explosion Center Transtage	3-26
3-17	Payload Configuration	3-27
3-18	Static Overpressure vs. Time After Liftoff Transtage	3-28
3-19	Static Impulse vs. Time After Liftoff Transtage	3-28
3-20	Static Impulse and Positive Phase Duration vs. Distance From Explosion Center Transtage	3-29
3-21	Blast Wave Duration vs. Time After Liftoff Transtage	3-29
3-22	Fireball and Afterfire Model Titan IIIC Liquid Propellant	3-31
3-23	RTG Lateral Impact Effect of Lateral Pressure Ratio on Yield	3-36
3-24	RTG Storage and Assembly Building No. 800.	3-51
3-25	Loading and Assembly Stations in Building 800	3-52
3-26	MHW RTG Vibration Facility - Building 400	3-53
3-27	Dynamic Test Equipment in Building 400 - MHW RTG Facility	3-53
3-28	MHW RTG Shock and Vibration Test Cell	3-55
3-29	Thermal-Vacuum Facility - Building 300	3-56
3-30	Thermal-Vacuum Chamber for MHW RTG Testing.	3-57
4-1	LES 8/9 Orbital Positions for Thermal Analysis	4-5
4-2	Orbital Temperatures - Lower LES 8 RTG Steady State Temperatures at Each Position	4-7
4-3	Orbital Temperatures for Top LES 8 RTG	4-8
4-4	Orbital Temperatures for Lower LES 9 RTG.	4-9
4-5	Orbital Temperatures for Top LES 9 RTG	4-10
4-6	Pad Cooling Concept.	4-13

LIST OF ILLUSTRATIONS (Cont)

Figure		Page
4-7	Two Wall Jet Concepts	4-14
4-8	MHW 66-Node Thermal Model.	4-19
4-9	MHW 26-Node Thermal Model.	4-20
4-10	TBC-1 Transient Test	4-22
4-11	TBC-1 Transient Test	4-24
4-12	TBC-3 Axial Temperature Profiles in Air	4-24
4-13	TBC-3 Axial Temperature Profiles in Vacuum	4-25
5.2-1	MHW Heat Source	5-9
5.2-2	FSA Sequence	5-14
5.2-3	Retainer Ring Assembly.	5-14
5.2-4	Final Assembly Sequence	5-15
5.3-1	Saturation Concentration of Helium in Solid PuO ₂ (From LASA Data).	5-23
5.3-2	Helium Generation vs. Time for Various MHW Components	5-24
5.3-3	MHW Fuel Sphere Assembly - FSA: Post Impact Shell Assembly - PISA, In Graphite Impact Shell - GIS	5-29
5.3-4	LASL Compatibility Sphere Test Configuration	5-30
5.3-5	PuO ₂ /Ir Compatibility Test Assembly	5-37
5.3-6	LASL-Vented Test Capsule	5-38
5.3-7	Thermal Gradient Test Configuration	5-42
5.3-8	Uniform Temperature Test Configuration	5-43
5.3-9	Tensile Test Specimens.	5-44
5.4-1	Post-Impact Sphere Assembly.	5-53
5.4.1-1	Sample WC28-4 (with KCN). Positive ion IMA-images prepared for cross-section specimen. Sample had been ground with alumina wheel, then stress relieved at 850°C for 15 minutes. It was then cleaned in KCN for 15 minutes at 5 volts. Magnifi- cation = 240X.	5-57
5.4.1-2	Sample WC 14 (cross-section) Before Long High Temperature Treatment. Sample was first heated to 1050°C for one hour before ORNL-super-cleaning. Cleaning procedure used: 10 minutes in hot (90°C) 70% HNO ₃ + 30% HF; then 2 minutes in hot (90°C) HNO ₃ ; then two distilled water rinses; then hot (90°C) distilled water rinse; then air dry. Magnification = 240X.	5-58
5.4.1-3	Sample WC 14 (cross-section), same as sample in Figure 5.4.1-2 except long heat treatment after cleaning. Heat treat- ment: 1500°C for 18 hours. Magnification = 240X.. . . .	5-59
5.4.2-1	MHW Vent Filter - Current Design	5-60
5.4.2-2	MHW Vent Filter - Frit Type - Backup Design	5-66

LIST OF ILLUSTRATIONS (Cont)

Figure		Page
5.5-1	Fuel Sphere Assembly Exploded View	5-79
5.5-2	Impact Shell Dimensions	5-80
5.5-3	Impact Shell Random Winding Device	5-82
5.6-1	Retaining Tray Assembly	5-85
5.6-2	Top End Spring Crushup	5-85
5.6-3	Bottom Tray and Laminated Spring Assembly	5-86
5.7-1	Aeroshell Assembly	5-87
5.8-1	Iridium Clad Parts in Fabrication	5-90
5.8-2	Schematic of Gas Management Subsystem	5-92
5.9-1	Emissivity Sleeve Fabrication	5-96
5.9-2	Emissivity Sleeve/Clad Fit-Up	5-97
5.10-1	Schematic of Gas Management Subsystem	5-99
5.10-2	PRD Diaphragm Rupture Characteristics	5-103
5.10-3	PMD System Schematics (Initial Configuration)	5-104
5.10-4	Gas Management Valve Cross Section	5-106
5.10-5	Gas Management Subsystem Sequence of Events	5-107
5.11.1.1-1	MHW Helipak Steady State Temperatures	5-115
5.11.1.1-2	MHW Helipak Steady State Temperatures	5-116
5.11.1.1-3	MHW Helipak Steady State Temperatures	5-117
5.11.1.1-4	Fuel/PICS Temperature Variation with PICS ID Emissivity	5-118
5.11.1.1-5	Fuel/PICS Temperature Variation with PICS OD Emissivity	5-119
5.11.1.1-6	Fuel/PICS Temperature Variation with Gap Condition	5-120
5.11.1.2-1	Response of MHW HSA After Removal from MRC Chill Blocks	5-121
5.11.1.2-2	Thermal Model of MHW HSA Steady State	5-122
5.11.1.1-3	MHW HSA Temperature Profile MRC Test Conditions	5-123
5.11.1.2-4	Temperature Profile MRC ΔT Test Steady State	5-125
5.11.1.2-5	Temperature Profile MRC ΔT Test Steady State	5-126
5.11.1.3-1	MHW HSA Temperature Profile - On Pad Conditions	5-130
5.11.1.3-2	MHW HSA Transient Post Launch Response Using 1/4 Sphere 3-D Model	5-132
5.11.1.3-3	MHW HSA Emissivity Sleeve/Outer Clad Gas Temperature Penalty	5-134
5.11.2-1	Titan IIC Liquid Fire Environment	5-136
5.11.2.1-1	Molybdenum Oxidation Rate	5-138
5.11.2.1-2	Molybdenum Surface Energy Balance Terms	5-140
5.11.2.1-3	Graphite Surface Energy Balance Terms	5-141
5.11.2.2-1	RTG Model Proximity	5-143
5.11.2.2-2	Bare HSA in Solid Fire - Contact	5-144
5.11.2.2-3	Bare HSA in Solid Fire	5-145
5.11.2.3-1	Temperature Response During Liquid Fireball	5-147
5.11.2.3-2	Temperature Response of RTG W/Case During Liquid Fire	5-148
5.11.2.3-3	Temperature Response of RTG W/Case During Liquid Fire	5-149

LIST OF ILLUSTRATIONS (Cont)

Figure		Page
5.11.2.3-4	Bare HSA Temperature Response During Liquid Fireball . . .	5-150
5.11.2.3-5	Bare HSA Temperature Response During Liquid Fireball . . .	5-151
5.11.3-1	Thermal Conductivities Burial Analyses Materials . . .	5-152
5.11.3-2	FSA and PISA Half Burial Model	5-153
5.11.4.1-1	Heating Rate and Recession - Heating Rate Increased 20% . . .	5-158
5.11.4.1-2	Minimum Gamma Re-entry Temperatures - Nominal Heating . . .	5-159
5.11.4.1-3	Minimum Gamma Re-entry Temperature Heat Rate Increased 20%	5-160
5.11.4.2-1	MHW HSA Aeroshell Thermal Gradient Model	5-161
5.11.4.2-2	MHW HSA Aeroshell Steep Angle Reentry Trajectory	5-163
5.11.4.2-3	MHW HSA Aeroshell Steep Angle Reentry Pressure and Heating History	5-164
5.11.4.2-4	MHW Aeroshell Temperature History	5-165
5.11.4.2-5	Aeroshell Margin of Safety.	5-167
5.11.4.2-6	Thermal Stress at Stagnation Point 8.4 Sec	5-168
5.11.4.2-7	Side-on Thermal Model of HSA End Cap Cross Section	5-171
5.11.4.2-8	Side-on HSA End Cap Cross Section.	5-172
5.11.4.2-9	MHW Aeroshell Temperature History 2.25" From Threaded Joint	5-173
5.11.4.2-10	MHW Aeroshell Temperature History at Threaded Joint	5-174
5.11.4.2-11	End-on End Section MHW HSA Nodal Diagram	5-176
5.11.4.2-12	Temperatures at 8.1 Sec (Peak Surface Temperatures)	5-177
5.11.4.2-13	End-Section In-Depth Temperatures	5-178
5.11.4.2-14	End-on End-Section In-Depth Temperatures	5-179
5.11.4.2-15	Thermal Stress Computer Model - Side-on	5-181
5.11.4.2-16	Thermal Stress Computer Model - End-on	5-185
5.11.4.2-17	Tension Modulus of Elasticity vs. Temperature AXF-5Q	5-187
5.11.4.2-18	Tensile Strength vs. Temperature AXF-5Q	5-188
5.11.4.2-19	Compressive Modulus of Elasticity vs. Temperature AXF-5Q . . .	5-189
5.11.4.2-20	Compressive Strength vs. Temperature AXF-5Q	5-190
5.11.4.2-21	Poisson's Ratio vs. Temperature for Tension and Compression AXF-5Q	5-191
5.11.4.2-22	2-D Thermal Stress Circumferential Nodal Locations.	5-193
5.11.4.2-23	Recession History for the Emissivity Sleeve	5-194
5.11.4.2-24	Steep Angle Reference Design Analysis In-Depth Temperature History at Stag Line Actual Heating	5-196
5.11.4.2-25	Steep Angle Reference Design Analysis In-Depth Temperature History - Stag Line 10% Increased Heating	5-197
5.11.4.2-26	In-Depth Aeroshell Temperatures t = 8.0 Seconds Maximum ΔT and Maximum Surface Temperature	5-198
5.11.4.2-27	Thermal Stress History.	5-199
5.11.4.2-28	Margin of Safety	5-200

LIST OF ILLUSTRATIONS (Cont)

Figure		Page
5.11.4.2-29	Aeroshell Stress at Stagnation Point	5-201
5.11.4.2-30	Aeroshell Stress at Stagnation Point	5-202
5.11.4.2-31	Aeroshell Stress at Stagnation Point	5-203
5.11.4.2-32	Aeroshell Stress at Stagnation Point	5-204
5.11.4.2-33	Thermal Stress at Stagnation Point During Cool Down	5-206
5.11.4.3-1	Ablation Heating - Laminar Trajectory	5-208
5.11.4.3-2	Ablation Heating - Turbulant Trajectory	5-208
5.11.4.4-1	RTG Breakup Study	5-212
5.11.4.4-2	Altitude and Time Breakup Bands	5-213
5.11.4.4.3	Ratio of the Instantaneous RTG Stagnation Line Heat Flux at Breakup	5-214
5.11.4.4-4	Side-on Stable RTG Breakup Altitude Band and Flux Ratio - Time Bands	5-215
5.11.4.4-5	Tumbling RTG Breakup Altitude Band and Flux Ratio - Time Bands	5-216
5.11.4.4-6	RTG Breakup Study	5-217
5.11.4.4-7	30° Thermal Response Model End-on RTG Breakup Analysis	5-220
5.11.4.4-8	End-on MHW RTG Breakup Altitudes and Times Design Trajectories	5-222
5.11.4.4-9	Qcws Ht Breakup for Various Re-entry Orientations	5-223
5.11.4.4-10	Case I - HSA Release from Side-on RTG	5-224
5.11.4.4-11	Case II - HSA Release from End-on RTG	5-225
5.11.4.4-12	Case III - HSA Release from Side-on RTG	5-226
5.11.4.4-13	Case IV - HSA Release from End-on RTG	5-227
5.11.4.4-14	In-Depth Aeroshell Temperatures at Maximum ΔT and Maximum Surface Temperature	5-229
5.11.4.4-15	In-Depth Aeroshell Temperatures at Maximum ΔT	5-230
5.11.4.5-1	Analytical Methodology	5-231
5.11.4.5-2	Coordinate System and Sign Convention	5-232
5.11.4.5-3	Pitching Moment Coefficient vs. Angle of Attack of Payload System Plus Transtage	5-232
5.11.4.5-4	Pitching Moment Coefficient vs. Angle of Attack of Payload System Plus Transtage	5-233
5.11.4.5-5	Pitching Moment Coefficient vs. Angle of Attack of LES 9	5-233
5.11.4.5-6	Analysis Flow Chart	5-235
5.11.4.5-7	RTG and HSA Heating, HSA Thermal Response, and Release Points of the RTG and HSA (Case I)	5-238
5.11.4.5-8	RTG Domes and Mounting Bracket Response RTG/SC Phase of Re-entry (Case II)	5-239
5.11.4.5-9	Case II Conclusions: Intact Impact of Cladded HSA	5-241
5.11.4.5-10	Case IV Heating, Thermal Response Histories, Release Points and Safety Margins	5-242

LIST OF ILLUSTRATIONS (Cont)

Figure		Page
5.11.4.5-11	Case V - Release Points	5-245
5.11.4.5-12	Case V - Heating, HSA, Thermal Responses and Safety Margins .	5-246
5.11.4.5-13	Case V - Heating, HSA, Thermal	5-247
5.11.4.5-14	Case VI - Heating, HSA, Thermal Responses, Release Points and Safety Margins	5-248
5.11.4.5-15	Case VII - Heating, HSA, Thermal Responses, Release Points and Safety Margins	5-249
5.11.4.5-16	Case VIII Analysis Results	5-251
5.11.4.5-17	Case X - Heating, Thermal Response, Release Points and Safety Margins	5-252
5.11.5.1-1	MHW-Static Stability Analysis RTG Configuration	5-255
5.11.5.2-1	Results of Initial Trajectory Simulations	5-257
5.11.5.2-2	Results of Second Trajectory Simulation	5-258
5.11.5.2-3	Trajectory Simulation with Pitch Rate = 2 Deg/Sec	5-259
5.11.5.2-4	Angle of Attack vs. Altitude Pitch Rate = 3 Deg/Sec	5-261
5.11.5.3-1	MHW-HSA Hypersonic Static Stability Characteristics	5-263
5.11.5.3-2	MHW-HSA Hypersonic Statis Stability Characteristics	5-264
5.11.5.3-3	MHW-HSA Hypersonic Static Stability Characteristics	5-265
5.11.5.3-4	MHW-HSA Hypersonic Static Stability Characteristics	5-266
5.11.5.3-5	MHW-HSA Hypersonic Static Stability Characteristics	5-267
5.11.5.3-6	MHW-HSA Subsonic Static Stability Characteristics	5-274
5.11.5.3-7	MHW-HSA Subsonic Static Stability Characteristics	5-275
5.11.5.3-8	MHW-HSA Subsonic Static Stability Characteristics	5-276
5.11.5.3-9	MHW-HSA Subsonic Static Stability Characteristics	5-277
5.11.5.3-10	MHW-HSA Transonic Static Stability Characteristics	5-278
5.11.5.3-11	MHW-HSA Transonic Static Stability Characteristics	5-279
5.11.5.3-12	MHW-HSA Transonic Static Stability Characteristics	5-280
5.11.5.3-13	MHW-HSA Transonic Static Stability Characteristics	5-281
5.11.5.3-14	MHW-HSA Supersonic Static Stability Characteristics.	5-282
5.11.5.3-15	MHW-HSA Supersonic Static Stability Characteristics.	5-283
5.11.5.3-16	MHW-HSA Supersonic Static Stability Characteristics.	5-284
5.11.5.3-17	MHW-HSA Subsonic-Transonic-Drag Characteristics	5-285
5.11.5.3-18	MHW-HSA Subsonic Drag Variation with Reynolds Number . . .	5-286
5.11.5.3-19	MHW-HSA Transonic/Subsonic Dynamic Stability	5-287
5.11.5.3-20	MHW-HSA Transonic/Subsonic Dynamic Stability	5-288

PART II

6.2-1	MHW Radioisotope Thermoelectric Generator	6-5
6.2-2	External Envelope MHW RTG	6-6
6.2-3	Internal Envelope MHW RTG	6-7
6.2-4	LES 8 and 9 Interface	6-8
6.2-5	Silicon Germanium Unicouple	6-10

LIST OF ILLUSTRATIONS (Cont)

Figure		Page
6.2-6	SiGe Unicouple Dimensions and Materials	6-10
6.2-7	End Cap Insulation	6-11
6.2-8	Inner Moly Frame	6-13
6.2-9	Completed Insulation Assembly	6-14
6.2-10	Completed Thermopile Assembly	6-14
6.2-11	Outer Shell Subassembly	6-15
6.2-12	Gas Management Valve	6-18
6.2-13	MHW C-Seals	6-18
6.2-14	Heat Source Support System	6-19
6.2-15	Heat Source Support Cap Pre-Assembly	6-20
6.2-16	End Enclosure and Pre-Load Screw	6-21
6.2-17	Thermopile Mounted to Assembly Fixture	6-24
6.2-18	Outer Shell and Thermopile on Assembly Fixture	6-24
6.2-19	Assembled Outer Shell and Thermopile with T/E Screws Installed	6-24
6.2-20	Electric Heat Source Installation	6-25
6.2-21	Forward Internal Connectors to Spool Piece (TBC-3 Shown) . .	6-25
6.2-22	Completed ETG	6-25
6.2-23	ETG to RTG Conversion	6-27
6.3-1	E1-E2 Forward Pressure Dome Assembly	6-32
6.3-2	Converter Pressure Decay Requirements	6-34
6.3-3	RTG Gas Management Assembly	6-35
6.3-4	C-Seal Fabrication	6-38
6.3-5	T/E Attachment C-Seal Compression Test Wall Thickness vs. Compression Torque	6-41
6.3-6	Leak Rate versus Internal Pressure for Seal 47C30p581Pl . .	6-42
6.3-7	C-10 Pressure vs. Time for Argon Gas	6-44
6.3-8	C-10 Pressure Decay Testing	6-45
6.3-9	C-10 Pressure Decay Testing	6-46
6.3-10	Stacked RTG Routing Mockup	6-49
6.4-1	Module-to-Foil Resistance vs. Time	6-67
6.4-2	D5A2 (14-Couple Module) $T_{HS} \cong 900^{\circ}\text{C}$	6-75
6.4-3	90% Confidence Limit, P-Type, 78% SiGe, Electrical Resistivity	6-78
6.4-4	90% Confidence Limit, P-Type 78 At.% SiGe, Seebeck Coefficient	6-79
6.4-5	89% Confidence Limit, P-Type 78 At.% SiGe, Thermal Conductivity	6-80
6.4-6	Test Specimen Configuration	6-85
6.5.1-1	18-A TBC-1 Couples $\sim 1035^{\circ}\text{C}$ Hot Shoe	6-90
6.5.1-1A	18-A N&P Leg Dopant Precipitation (Preliminary)	6-91
6.5.1-2	18-B 1035°C Hot Shoe	6-92

LIST OF ILLUSTRATIONS (Cont)

Figure		Page
6.5.1-3	18-C TBC-3 Couples, 1035°C Hot Shoe	6-93
6.5.1-4	18-D TBC-3 Couples, 1035°C Hot Shoe	6-94
6.5.1-4A	18-D N&P Model (Preliminary)	6-95
6.5.1-5	18-E Hot Shoe 1035°C	6-96
6.5.1-6	18-F 1085°C Hot Shoe	6-97
6.5.1-7	18-G 1035°C N Model Hot Shoe	6-98
6.5.1-8	18-H Hot Shoe 985°C	6-99
6.5.1-9	18-I Hot Shoe 1085°C	6-100
6.5.1-10	18-K N&P Model	6-102
6.5.1-11	DC-1 N&P Model	6-103
6.5.1-12	DC-2 N&P Model	6-104
6.5.1-13	DC-3 N&P Model	6-105
6.5.1-14	DC-4 N&P Model	6-106
6.5.1-15	DC-5 N&P Model	6-107
6.5.1-16	DC-6 N&P Model	6-108
6.5.1-17	TBC-3 and 4 1035°C Hot Shoe.	6-109
6.5.1-18	18-Couple Module Thermal Insulation Conductance Change	6-111
6.5.1-19	Thermal Insulation Conductance - TBC-3 and TBC-4	6-112
6.5.1-20	18-Couple Modules - Module to Foil Resistance.	6-113
6.5.1-21	18-Couple Modules - Resistance to Foil vs. Silicon Mass Loss.	6-116
6.5.1-22	DC Modules - Module to Foil Resistance	6-117
6.5.1-23	TBC-1/TBC-3/TBC-4 Shunt Resistance	6-120
6.5.1-24	Thermal Cycle.	6-121
6.5.2-1	MHW Performance - 1000°C	6-123
6.5.2-2	Leg Sublimation Suppression Factors	6-124
6.5.2-3	18-F Data, K-Legs = Constant	6-126
6.5.2-4	18-F Thermal Insulation Conductance Variation Based on Constant Leg Thermal Conductivity.	6-127
6.5.2-5	MHW Performance	6-128
6.5.2-6	MHW Performance	6-129
6.5.2-7	MHW Performance	6-130
6.5.2-8	MHW Performance	6-131
6.5.2-9	MHW Performance	6-133
6.5.2-10	Module to Foil Resistance	6-135
6.5.2-11	Module to Foil Resistance vs. Silicon Mass Sublimed from Hot Shoe.	6-136
6.5.2-12	TBC-4 Conductance vs. Time.	6-139
6.5.2-13	TBC-4 Resistance to Foil	6-140
6.5.2-14	DC-4 Electrical Conductance - Circuit to Foil	6-141
6.5.2-15	DC Modules - Module to Foil Resistance	6-142
6.5.2-16	Al ₂ O ₃ Mass Loss Rate	6-145
6.5.2-17	Al ₂ O ₃ Barrier	6-147

LIST OF ILLUSTRATIONS (Cont)

Figure		Page
6.5.1-18	Thermo Element Conductance Change	6-149
6.5.1-19	MHW Performance	6-150
6.5.1-20	MHW Performance	6-151
6.5.1-21	MHW Performance	6-152
6.6-1	Fin Stress vs. Axial Temperature Gradient	6-155
6.6-2	Lower Ring of RTG Stress vs. Circumferential Linear Gradient .	6-156
6.6-3	MHW-RTG Outer Shell Allowable Temperature Gradients for Pre-Launch and Launch Conditions	6-157
6.6-4	Stacked Configuration	6-158
6.6-5	RTG-Stacking Adapter	6-159
6.6-6	MHW-Mount Configuration	6-160
6.8-1	Heat Source Support System	6-164
6.8-2	Heat Source Support Cap Preassembly	6-165
6.8-3	Aft End of EHS with New Zirconia Bearing Disk and Retention Washer	6-166
6.8-4	Forward End of EHS with Vent Tube Installed Prior to Bending. .	6-166
6.8-5	C11 Fixture Including Interconnections and Vent Tube Routing .	6-167
6.8-6	Random Vibration Acceptance Test	6-170
6.8-7	Random Vibration Qualification Test	6-171
6.8-8	Equivalent Shock Response Spectrum for 775 g's (Peak), 0.2 $\pm$ 0.1 msec., Half-sine Wave Pulse	6-172
6.8-9	Aft Astroquartz Pad After Test	6-173
6.8-10	Dynamically-Simulated RTG Models on LES 9 Truss	6-176
6.8-11	C-10 Test Assembly	6-178
6.8-12	C-10 System Layout	6-179
6.8-13	C-10 Phase I Test Results	6-180
6.8-14	C-10 Phase II Test Results	6-181
6.8-15	D5A3 (14-Couple Module) - $T_{HS} \approx 900^{\circ}\text{C}$	6-182
6.8-16	D53 Dynamic Test Unit	6-183
6.9-1	+Y, +Z Axes Acceleration Test Set-up (Shown Without the Protective Shroud Covering the Converter)	6-186
6.9-2	Acceleration Test Orientations	6-186
6.9-3	TBC-3 Post-Dynamic Thermal-Vacuum I-V Map	6-189
6.9-4	LES 8/9 Truss Mounted to TBC-3	6-191
6.9-5	Truss Thermal Instrumentation	6-192
6.9-6	Dome-to-Dome Thermal Test Set-up	6-193
6.9-7	Thermal Profile During Dome-to-Dome Test.	6-193
6.9-8	Output Power During Transient Vending	6-196
6.9-9	Aft Preload Screw Showing Deformation Due to Overtemperature .	6-198
6.9-10	Magnetic Signature Mapping Test Set-up	6-199
6.9-11	Reference Coordinates, Coil System vs. Converter Axes . . .	6-203
6.9-12	Argon Transient Venting Flow Rate and Internal Pressure Decay .	6-206

LIST OF ILLUSTRATIONS (Cont)

Figure		Page
6.9-13	Argon Transient Test No. 5 - Output Power, H.S. and C.S. Temperatures	6-207
6.9-14	Argon Transient Test No. 5 - Chamber and Converter Pressures. .	6-208
6.9-15	Life Test Major Parameter History.	6-210
6.9-16	Converter Terminal Voltage and Shunt Regulator Voltage During Transient Venting	6-211
6.9-17	Argon Transient Test with Shunt Regulator - Output Powers at Converter Terminals and at Shunt Regulator; Hot Shoe and Cold Strap Temperatures	6-212
6.9-18	TBC-3 Second Stability Test Parameters History	6-216
6.10-1	TBC-4 I. V. P. Characteristics	6-221
6.10-2	TBC-4 Stability Test Parameter History	6-225
11-1	Q-1 ETG I. V. P. Characteristics	11-8
11-2	Q-1 ETG Stability Test Parameter History	11-10
13-1	MHW Graphite Dynamic Electrics Heat Source	13-2
13-2	EHS Aft End Showing Bearing Disc, Vent Holes and Cutouts for Retention Washer	13-3
13-3	Heater Element Assembly Tool Being Used to Insert Ceramic Support Tube into Heater Element	13-3
13-4	GDEHS - 24 in Vibration Fixture on C-150 Slip Table Adapter Plate .	13-7
13-5	Electric Heat Source Dynamic Acceptance Test Fixture	13-9
13-6	EHS Vibration Acceptance Test	13-10
13-7	Random Spectrum EHS Longitudinal Axis	13-11
13-8	Random Spectrum EHS Lateral Axis	13-12
13-9	EHS Processing Furnace	13-13
13-10	RTG Shipping Container	13-19
13-11	VFSC-Building 800 RTG Assembly and Storage Facility	13-21
13-12	LAS Chambers.	13-23
13-13	LAS #2-195 ⁰ Station	13-24
13-14	0 ⁰ Work Station	13-25
13-15	LAS #1 Internal View	13-26
13-16	View of Load Lock 0 ⁰ Station	13-27
13-17	Internal View of Load Lock	13-28
13-18	Portable Tool Lock	13-29
13-19	Gas Purification System	13-31
13-20	LAS Controls System.	13-33
13-21	LAS Controls System Vacuum Monitoring and Lighting Control Panel.	13-34
13-22	LAS Main Control Panel	13-35
13-23	Operating Consoles and Slide Trays.	13-37
13-24	Radiological Shields at LAS #2 13-32	13-32

LIST OF ILLUSTRATIONS (Cont)

Figure		Page
13-25	Special Tools	13-40
13-26	Storage Protection Container	13-48
13-27	SPC Installed in Isotope Heat Source Shipping Container	13-49
13-28	Spring Load Conditions	13-50
13-29	Spring Load/Deflection Curve.	13-50
13-30	SPC Temperature Locations	13-51

BLANK

LIST OF TABLES

Table	PART I	Page
3-1	SST Test Conditions and Results	3-23
3-2	Results of Nuclear Criticality Analysis	3-30
3-3	Blast Wave Overpressures	3-32
3-4	Spacecraft Structural Characteristics	3-34
3-5	RTG Blast Wave Response	3-35
3-6	Transtage Abort Modes and Probabilities	3-39
3-7	First Burn V, γ Results (From Park Orbit)	3-41
3-8	Second Burn V, γ Results (From Transfer Orbit)	3-41
3-9	First Burn V, γ Results (From Park Orbit)	3-42
3-10	Second Burn V, γ Results (From Transfer Orbit)	3-43
3-11	P74-1 Mission Re-entry Summary-Post Earth Orbit	3-44
3-12	LES 8/9 Breakup Analysis Cases (Initial Conditions)	3-45
3-13	Conditions at RTG Release	3-46
4-1	LES 8/9 Orbit Hot Shoe Temperatures	4-5
4-2	Axial Wall Jet Cooling	4-14
4-3	Radial Impingement Cooling	4-16
4-4	Effect of Shroud on Free Convective Cooling of the MHW Generator.	4-17
5.1-1	MHW-HSA Design Specifications	5-1
5.3-1	MHW Fuel Characterization	5-17
5.3-2	Fuel Characterization-1	5-18
5.3-3	Fuel Characterization-2	5-19
5.3-4	MHFT-1 Thru MHFT-14	5-27
5.3-5	MHW Fuel Sphere Assembly Compatibility Tests at LASL.	5-31
5.3-6	MHFT-15 Thru MHFT-33	5-33
5.3-7	Test Matrix	5-39
5.3-8	Diagnostic Tests	5-40
5.3-9	Summary of LASL Test Results	5-41
5.3-10	DWDL Mockup Compatibility Test Matrix	5-45
5.3-11	Summary of DWDL Test Results	5-46
5.3-12	Iridium Short Term Exposure Tensile Test Data.	5-47
5.3-13	Iridium Long Term Exposure Tensile Test Data	5-48
5.3-14	Iridium Tensile Specimen Group Averages.	5-49
5.3-15	Summary of Micro Tensile Tests at DWDL.	5-50
5.4.2-1	MHW Fuel Sphere Assembly Compatibility Tests at LASL	5-61
5.4.2-2	MHFT-15 Thru MHFT-33-Vent Condition	5-62

LIST OF TABLES (Cont'd)

Table		Page
5.4.3-1	FSA Test Summaries-Impacted at-GE-VF (Bldg. 8)	5-68
5.4.3-2	FSA Test Summaries-Impacted at-GE-VF (Bldg. 8)	5-69
5.4.3-3	FSA Test Summaries-Impacted at-GE-VF (Bldg. 8)	5-70
5.4.3-4	FSA Test Summaries-Impacted at-GE-VF (Bldg. 8)	5-71
5.4.3-5	FSA Test Summaries-Impacted at-GE-VF (Bldg. 8)	5-72
5.4.4-1	FSA Test Summaries-Impacted at-GE-Vf LASL	5-75
5.4.4-2	Impact Tests at LASL (Simulant)	5-76
5.4.4-3	FSA Test Summaries-Impacted at LASL	5-77
5.4.4-4	Impact Tests at LASL (Fuel).	5-78
5.5-1	Clearance Between Iridium PICS and Graphite Impact Shell.	5-82
5.5-2	Graphite Impact Shell Final Product Purity	5-83
5.7-1	Mechanical Properties	5-88
5.7-2	AXF 50 Impurity Limits	5-88
5.11.1.1-1	Summary of Steady State Temperatures	5-110
5.11.1.1-2	Summary of Gap Temperature Drops	5-111
5.11.1.1-3	PICS and Emissivity and Clad Coatings	5-113
5.11.1.3-1	MHW HSA Temperatures During Assembly Test and Handling	5-128
5.11.2-1	Summary of Liquid Fire Results	5-135
5.11.3-1	Burial Analysis Results	5-154
5.11.4.5-1	Re-entry Cases Analyzed	5-234
5.11.4.5-2	Conditions at RTG Release.	5-237
5.11.4.5-3	Analysis Summary	5-253
5.11.5.1-1	RTG Aerodynamic Characteristics	5-256

PART II

6.2-1	Converter Primary Components and Material Summary Flight Configuration.	6-29
6.2-2	MHW-RTG Flight Converter Weight Breakdown	6-31
6.3-1	Allowable C-Seal Leak Rates.	6-39
6.4-1	Air-Vac Life Test Status	6-51
6.4-2	RCA SiGe Thermoelectric Module Life Tests Snap 17A Module.	6-53
6.4-3	Computer Simulation of Foil System Degradation	6-57
6.4-4	Design Correction Program-18 CPL Converter Status as of 12/11/73	6-66
6.4-5	Configuration Evaluations of Quartz Sleeves	6-71
6.4-6	Thermal Comparison of DC-4 to 18J and 18K	6-73
6.4-7	Mechanical Properties of Thermoelectric Materials with Simulated Production Thermal Cycles	6-81
6.4-8	Weight of Silicon Nitride Coatings	6-82

LIST OF TABLES (Cont'd)

Table		Page
6.4-9	Sublimation Weight Loss	6-85
6.4-10	Test Results for Si ₃ H ₄ Coated Couples and Legs	6-86
6.5.1-1	18 Couple Module Characteristics and Test Conditions	6-89
6.5.2-1	Initial and 5-year Power as a Function of Fuel Loading	6-134
6.5.2-2	Extrapolated 5-year Values of Resistance Between Module and Foil in a Full Scale Converter	6-138
6.5.2-3	BOL Hot Shoe Temperatures for Present Design	6-144
6.5.2-4	BOL Hot Shoe Temperatures for SiO ₂ Plates (.58"L) around Legs	6-144
6.5.2-5	Present Design - Al ₂ O ₃ Recession	6-146
6.5.2-6	Proposed Design with SiO ₂ Plates and 2430 W Fuel Loading - Al ₂ O ₃ ² Recession	6-146
6.6-1	Stress Summary	6-162
6.8-1	Temperature Distribution - C-11G Test.	6-169
6.8-2	Temperature Distribution - C-11G Retest	6-175
6.8-3	C-10 Phase I Back Diffusion Test Results	6-179
6.8-4	C-10 Phase II Test Results	6-180
6.9-1	TBC-3 Performance Before and After Acceleration Tests.	6-187
6.9-2	TBC-3 Post Dynamic Thermal Vacuum Performance Characteristics.	6-188
6.9-3	TBC-3 Thermal Vacuum Performance Before and After Dynamic Tests	6-190
6.9-4	Air Performance Characteristics.	6-191
6.9-5	Transient Venting Parameters.	6-195
6.9-6	TBC-3 "Sniffing" Leak Test Data Summary	6-197
6.9-7	TBC-3 Magnetic Dipole Moments and Axial Fields at 3 Meters.	6-202
6.9-8	TBC-3 Life Test Performance.	6-209
6.9-9	TBC-3 "Cold" Resistance History.	6-213
6.9-10	TBC-3A Second Life Test Performance Parameters	6-215
6.10-1	TBC-4 Thermal Vacuum Performance Characteristics (Q = 2380 W)	6-220
6.10-2	TBC-4 First Phase Stability Test Major Parameter History (Q = 2380 W)	6-222
6.10-3	TBC-4 Shunt Resistance Values as a Function of Hot Shoe Temperature	6-223
6.10-4	TBC Second Phase Stability Test Major Parameter History (Q = 2380 W)	6-224

LIST OF TABLES (Cont'd)

Table		Page
8-1	FMECA Data Summary.	8-3
8-2	MHW-RTG Critical Single Point Failure Listing	8-4
8-3	Status of SPF's	8-9
8-4	RCA Reliability Documentation	8-13
8-5	Summary of Nonconformance Reports.	8-19
11-1	Cable Tested and Verified for MHW	11-5
11-2	Q-1 ETG Thermal-Vacuum Performance Characteristics (Q = 2400 W).	11-7
11-3	Q-1 ETG Stability Test Major Parameter History.	11-7
13.9-1	Stress Summary	13-53
13.9-2	Thermocouple Locations for SPC Thermal Certification Test	13-55

SECTION 1

INTRODUCTION AND SUMMARY

1.1 INTRODUCTION

This report summarizes significant events, activities and achievements on the MHW-RTG Program during the calendar year 1 January to 31 December 1973. It covers the continuation of work for the Atomic Energy Commission on the Lincoln Laboratory's LES 8/9 Program and the initiation of effort for the AEC on the Jet Propulsion Laboratory's Mariner, Jupiter and Saturn Program.

1.2 SUMMARY

The year 1973 saw a transition from the design and development phases of the program into the fabrication and test of full-scale qualification and flight hardware and ground support equipment.

The following paragraphs summarize some of the major activities carried out during the year, with further details being presented in subsequent sections of this report.

1. The following preliminary tests in the MHW Safety Test Program were conducted: S-1, Explosion; S-2, Liquid Propellant Fire; S-3, Solid Propellant Fire; FSA/PISA Water Immersion Tests; and SST, Safety Sequential Tests.
2. Operational safety effort included issuance of the MHW Radiological Protection Program and Safety Analysis Report documents, radiological safety training classes for 85 persons, the conduct of a walkthrough demonstration of the RTG Loading and Assembly Station and participation in the Operational Analysis Task Group.
3. Systems integration and interface activity was concentrated in the areas of requirements and procedural documentation and the development of performance requirements and data through analyses and tests.
4. Fuel form characterization and compatibility test programs were carried out at LASL, MRC/ML and DWDL to establish or verify high temperature properties and behavior of candidate materials.

5. ORNL/MRC became the prime source for PICS iridium hemispheres.
6. Analyses and tests were carried out on the Heat Source Assembly to investigate the areas of operational assembly and launch temperatures, pad abort thermal responses, RTG breakup and HSA thermal response during reentry, thermal stress in the aeroshell and materials properties and aerodynamic characteristics.
7. E-1 and E-2 ETG design was completed and all hardware for these converters was fabricated. The major configuration change from standard ETG's was the elimination of the spool piece.
8. Pressure decay testing was carried out on the C-10 module to measure oxygen back-diffusion rates and verify design of the converter C-seals and gas management system.
9. Design and development effort at RCA included long-term performance analysis and life testing of modules and converters to verify the design and performance capability of the qualification and flight units. A design correction engineering program was initiated to investigate possible design changes that could be incorporated to eliminate or reduce the impact of shunt resistance degradation on converter power output.
10. Component and module test programs were carried out to evaluate the redesigned heat source support system, to verify two dynamically-simulated RTG models for Lincoln Laboratory, and to investigate the dynamic capabilities of several candidate thermopile design corrections.
11. Extensive testing was carried out on TBC-3 and TBC-4 Test Bed Converters, consisting of acceleration, post-dynamic thermal vacuum and air performance, transient venting, magnetic signature evaluation and life/stability tests on TBC-3, and mass properties, thermal vacuum and air performance tests on TBC-4. A programmatic change directed that these two units will be converted to E-1 and E-2 ETG's, respectively.
12. The Q-1 Qualification ETG was shipped from RCA to GE on November 6, 1973 and the planned qualification test program was started on this unit. The following tests had been completed by the end of the year: Receiving inspection, cold and hot pressure decay, thermal vacuum performance and air performance. This unit was destined to be the first to receive a live isotope heat source, an event scheduled for early 1974.

13. A major portion of the prime Ground Support Equipment was fabricated, tested and put into service. A walk-through exercise was conducted in December to demonstrate operation and use of the Loading and Assembly Station, RTG Shipping Container, IHS Shipping Container, SPC, RTG Loading Fixture, RTG and SPC Slide Tray Assemblies and various other items of handling and monitoring equipment.
14. Building 800 at GE-VFSC, originally constructed primarily for storage of RTG's for the SNAP-19 and SNAP-27 programs, was expanded in size and equipped to handle the added requirements of the MHW program. This facility became operational during the year and was utilized for the operational demonstration walk-through in December.

BLANK

SECTION 3

SAFETY

3.1 SAFETY TEST PROGRAM

The preliminary tests in the MHW Safety Test Program were conducted over a period from January through April, 1973. These tests involved the use of test specimens consisting of individual FSA's and were intended to gain an early indication of the capability of the HSA design. Later safety qualification or proof tests were to be conducted on the total heat source. Each of the preliminary tests is summarized in the following paragraphs.

3.1.1 TEST S-1, EXPLOSION

The objective of this test was to subject eight (8) individual Fuel Sphere Assemblies (FSA) of the Heat Source preliminary design to the overpressure and impulse characteristics of a five (5) percent TNT equivalent yield from an explosion occurring in the Transtage of the Titan IIC launch vehicle undergoing a catastrophic launch pad accident. The purpose of the test was to evaluate the capability of the FSA's to withstand the pad abort and early launch blast environment from the Transtage.

Safety Test S-1 was conducted in the six-foot diameter shock tube facility at Kirtland Air Force Base on January 19, 1973. An 8-pack assembly of FSA's containing thoria simulant was preheated to approximately 2000°F (1093°C) in an argon atmosphere by a remotely controlled oven. The oven was opened and retracted from the shock tube, and the shock wave was initiated by detonating 26 strands of Primacord, 20 feet long having 400 grains per foot. Peak overpressures and impulses attained were as follows:

Peak Static Overpressure	= 57 psi
Peak Stagnation Overpressure	= 133 psi
Static Impulse	= 1.31 psi-sec
Stagnation Impulse	= 3.2 psi-sec

The velocity of the FSA's resulting from the shock wave interaction was determined to be about 160 fps. One of the FSA's was found as far as 200 yards from the end of the shock tube. None

of the eight (8) Post Impact Containment Shells (PICS) were breached, and no external radioactive contamination was found. One graphite impact shell was gouged, and an older type, non-threaded cap of another shell loosened and delaminated due to secondary impacts. Three (3) PICS's had slight to moderate deformations.

Figure 3-1 shows the 8-pack as installed on its support pedestal in the end of the shock tube. The preheat oven on its support pedestal and transfer arm is shown in Figure 3-2 enclosing the test specimen in the shock tube. Figure 3-3 is a photo of the eight individual FSA's recovered after the shot.

3.1.2 TEST S-2, LIQUID PROPELLANT FIRE

The objective of the S-2 test was to subject two (2) Fuel Sphere Assemblies (FSA's) and two (2) Post Impact Shell Assemblies (PISA's) to an Aerozine-50 (50% N_2H_4 and 50% UDMH) fuel fire to see what effects the combustion products and the thermal environment have on the specimens. The fire would simulate conditions resulting from a Titan III-C Aerozine-50 fuel tank rupture and the subsequent fire which could occur during a launch pad or early flight accident/abort.

Safety test S-2 was conducted at the Engineering Propulsion Laboratory, Martin Marietta Corporation, Denver Division, Denver, Colorado, on April 5, 1973. The two (2) FSA's and two (2) PISA's containing thoria simulant were preheated to approximately 2000^oF (1093^oC) in an argon atmosphere by a remotely controlled oven. The heated specimens were removed from the furnace and dropped into a tube containing approximately one (1) inch of Aerozine-50, igniting the fuel. The fuel burned for approximately twenty (20) minutes at temperatures ranging from 1500-1700^oF (815.5 - 926.6^oC). After the fire subsided, the FSA's were slightly charred, with some small degradation in the outer layer of the Thorneel-50 weave, while the PISA's were slightly discolored. There were no breaches and there was no detectable radioactive contamination, thus demonstrating the safety capabilities of the specimens.

Figure 3-4 shows the test setup with the four test specimens being transferred to the knockoff arms over the tub containing Aerozine-50. Photos of the test specimens recovered after the test are shown in Figure 3-5.

Figure 3-1. 8-Pack Installed on Support Pedestal

Figure 3-2. Oven Closed Around Specimen in Shock Tube

Figure 3-3. S-1 FSA's Following Test

Figure 3-4. S-2 Test Setup Prior to Live-Simulant Test (Specimens Near Drop Point)

MHT 16

MHT 17

MHT 18

MHT 19

Figure 3-5. MHT 16, 17, 18 and 19 Subsequent to Test S-2

3.1.3 TEST S-3, SOLID PROPELLANT FIRE

The objective of this test was to subject the eight (8) individual Fuel Sphere Assemblies (FSA) of the Heat Source preliminary design that had gone through Test S-1, Explosion, to the combustion environment of the solid propellant used in the Stage 0 strap-on rocket engines for the Titan IIIC launch vehicle. Parts of the test were configured to represent two situations: 1) FSA's falling onto a large segment of solid propellant either directly or as a result of rebound after impact in the pad area, and 2) FSA's falling between two large burning segments of the solid propellant. The purpose of the test was to evaluate the capability of the FSA's to withstand the combustion environment of the UTP-3001 propellant.

Safety Test S-3 was conducted in two parts at the Keller Road Test Range of the NASA Mississippi Test Facility on March 9 and 15, 1973. In the first part, two FSA's and two PISA's (containing thoria simulant) from Test S-1, Explosion, were preheated to approximately 2000⁰F (1093⁰C) and were then transferred remotely and dropped about six (6) inches onto the top of a single three-foot cube of solid propellant, causing it to ignite. Thirty seconds elapsed before the flame spread over the entire top surface of the propellant, and from that time, the fire lasted nine (9) minutes.

The two PISA's bounced on the somewhat resilient propellant and rolled off the top and onto the surrounding sand bed; they did not experience any appreciable heating. Recovery of the two FSA's, found in the middle of the pallet where the propellant had been, revealed that the graphite impact shells had flat spots about an inch in diameter on the surfaces that had been in contact with the propellant. Otherwise, the graphite shells were essentially intact with only minor loss of material having occurred from the surface. The PICS were not exposed, and there was no detectable radioactive contamination (also true of the two PISA's recovered from the sand bed).

Part two, or the dual proximity test, was conducted in the same manner except that two three-foot cubes of UTP-3001 were spaced twelve (12) inches apart with their uninhibited surfaces facing each other in parallel vertical planes. The four test specimens (two FSA's and two PISA's being the remainder of the specimens from Test S-1), preheated to 1980⁰F (1082⁰C), were

dropped in between the propellant cubes, one FSA and one PISA in the center, and the other two specimens toward the outer edge of the propellant cubes. The propellant was ignited with squib fired detonators, and the fire lasted 350 seconds (5 minutes, 50 seconds). Thermo-couples used in this part of the test indicated a peak temperature of about 3900⁰F for the fire.

After the dual proximity test, both PISA's were found to have holes in the PICS, one hole being roughly 1/16 square inch in area and the other being about 1/4 square inch in area; both holes were located in the uppermost portion of the PISA's as recovered. The FSA in the center position was found to be breached through the PICS with roughly a 1/4 square inch hole; the thorium sphere was plainly visible. This hole was found on the side of the FSA resting against the sand. The other FSA, found about eight (8) inches outside the line of the propellant cubes, was essentially intact. No detectable radioactive contamination could be found in the immediate area around the test specimens after the test.

Post test examinations of one of the FSA's from the contact test revealed that localized surface melting occurred on the PICS. Elements found in the affected area were mainly aluminum, iron, and silicon. These elements evidently came from the solid propellant, the steel test stands, and the sand bed, respectively. A leak test of the PISA showed that the PICS was not breached and that the localized surface attack on the iridium had not affected the integrity of the shell. The other FSA from the contact test was sent to LASL for analysis.

One of the bare PISA's from the dual proximity test was sectioned through the breached area and analyzed for foreign elements. The analysis revealed a high concentration of iron with lower concentrations of chromium, nickel, manganese, and silicon in the immediate vicinity of the breached area of the PICS. Aluminum and silicon were found distributed generally over the inner and outer surfaces of the PICS, these materials appearing in the form of mullite ($3 \text{ Al}_2\text{O}_3 \cdot 2 \text{ SiO}_2$).

Portions of the graphite shell from the FSA breached in the dual proximity test were analyzed also. The same elements (Fe, Cr, Ni, Mn, Si, Al) were found essentially in the same relative concentrations as those for the breached PISA described previously. In addition, tantalum was

discovered. The breached area of the PICS from this FSA revealed the same elements, but without tantalum. The tantalum most probably came from the sheaths around the W-Re thermocouples used in the test; one of the thermocouples was in the immediate vicinity of the breached FSA.

The other PISA from the dual proximity test was sent to LASL for analysis. Reports on this specimen and the other FSA from the contact test have appeared in the LASL CMB monthly reports.

Figure 3-6 shows the test arrangement for the dual proximity test showing the UTP-3001 propellant cubes and the preheat oven partially closed. A closeup of the oven and specimen transfer arm is shown with the specimens in place in Figure 3-7. Figure 3-8 shows the propellant block used for the contact test. The preheat oven and specimen transfer mechanism were the same for this part of the test as that shown for the dual proximity test in Figure 3-6.

Figures 3-9 and 3-10 show the condition of the test specimens recovered immediately after the contact test and dual proximity test, respectively. Post-test views of the PISA from MHT-5 (graphite shell removed) are shown in Figure 3-11. FSA MHT-9, Figure 3-12, shows the large breached area that developed in the dual proximity test.

3.1.4 TEST PLANNING

The course of planning for the safety qualification or proof tests changed considerably during the year, especially with regard to Test S-6, Pad Abort Sequential. A decision was made early in the year by the AEC/SNS Program Office to continue long term operational testing of TBC-1 at JPL; therefore, TBC-1 was no longer available to be used for the converter in S-6. SNS Safety and Reliability then asked LASL to investigate conceptual designs and projected costs for building a simulated converter for use in S-6.

At a meeting held in February, an agreement was reached among those present (LASL, SNS/S&R, AFWL, APL, GE/IASO) to delete Test S-6 entirely since a reasonable simulated converter could not be built with the potentially available funding. Based on this decision, the

Figure 3-6. Solid Propellant Fire Test-Dual Proximity Test Setup

Figure 3-7. Pre-Heat Oven and Test Specimen Holder

Figure 3-8. Propellant Block for Contact Tests

Figure 3-9. Recovered Contact Test Specimens

Figure 3-10. Recovered Dual Proximity Test Specimens

Figure 3-11. PISA From MHT-5 After Contact Test

Figure 3-12. FSA MHT-9 After Dual Proximity Test

stipulation was made that additional tests may be required later in the safety review cycle if the results of the Solid Fire Test, S-3, indicate a breach of the PISA's. Since either FSA's or an entire HSA could be involved, the consensus was to maintain the S-6 hardware for this purpose and this recommendation was made to the AEC/SNS Program Office. Other agreements reached at the February meeting were:

1. A projectile test would not be performed, especially on individual FSA's. AFWL representatives stated that projectiles would either not exist or be negligible at the RTG location.
2. All eight (8) FSA's from Test S-1, Explosion, would be used in Test S-3, Solid Fire, contingent upon LASL's inspection of the S-1 hardware (post S-1). This inspection was accomplished at GE and the decision to use the hardware was made. Also pairs of specimens containing a PISA and a FSA would be tested in the two parts of the test (contact and double proximity).

After the decision to delete Test S-6 was made, the AEC/SNS Program Office removed the S-6 hardware (one complete Heat Source Assembly) from the Safety Test Program. Subsequent development from the outcome of the Solid Fire Test, S-3, and the inability of the Air Force to provide facilities for Test S-8 necessitated a reconsideration of the remainder of the Safety Test Program by SNS. As a result, the S-6 test hardware was reinstated.

Testing of the Heat Source in a pad abort sequential type of environment (S-6 type of test) was to be contingent upon SNS Safety and Reliability providing the simulated converter. Following this event, SNS/S&R decided to support S-6 directly, with LASL being responsible for having the test performed. LASL was to build the simulated converter and be responsible for test planning, preparation, liaison with the test agency (Sandia) and reporting. Sandia would perform the test.

Along with the developments of S-6, the decision was made to delete Test S-8, the air drop of the preheated HSA. With the demise of S-8, test results from the Systems Verification Test (end-on and side-on impact of two HSA's using a rocket sled track) were to be used as input for the Updated Safety Analysis Report.

As previously mentioned, LASL investigated means of building a simulated converter for S-6. Toward this end, GE sent them six reject first bond couples (Converter thermocouples) for purposes of running tests and sets of drawings for the test bed and flight converters.

A meeting of the MHW INSRP was held in June at AEC Headquarters to review Test S-6 along with other test options. The objective of the meeting was to obtain the INSRP opinion on the best method of performing the pad abort test so as to yield maximum data for the safety review. At this meeting, GE reviewed the initial results of the pad abort blast analysis. LASL presented several test options both with and without the use of a simulated converter. They stated that there were many problems in arriving at the design of a good simulated converter as well as its use (manipulation) in the various parts of a sequential type of test. A good simulation would involve a considerable sum of money and there would be no assurance that the simulated converter could be used for more than one part of the test because of damage it might receive plus problems of preheat of the HSA required in other parts of the test. Results of the blast analysis showed that, even for a non-representative (conservative) yield explosion of five percent, the HSA would not experience the direct effects of the pressure wave, and only minor damage would be expected to the outer can and aeroshell arising potentially from the RTG's impacting together as a result of an unsymmetrical pressure loading. The converter would essentially be gone, especially after impacting in the pad area, and could be removed earlier due either to blast damage or to the fireball.

As a consequence of the test aspects and blast analysis results discussed in the meeting, the panel agreed to the test approach of eliminating the converter and testing only the HSA for an S-6 type of test. LASL was asked to proceed with the development of a test plan for an HSA test; the plan was distributed to INSRP members for comment.

The revised plan involves testing a full HSA with fuel simulant starting with a low velocity impact representative of fallback onto the launch pad from the height of the launch vehicle. This will be followed by subjecting the remains of the HSA to a solid propellant (UTP-3001) fire. The exact details of the arrangements for the fire test have not yet been selected due to the condition of the HSA components following the impact being unknown. However, LASL has

considered removing some of the iridium FSA's (all T-111 FSA's would be removed from the aeroshell following the impact) regardless of their condition and subjecting these to the solid fire both in contact and in proximity to the burning propellant.

In order to maximize the utilization of the test hardware to broaden the data base for the MHW safety review, LASL proposed a third phase of testing for the S-6 hardware. This phase would involve FSA impacts under various conditions not previously tested. LASL's recommendation was to remove the four least-damaged iridium FSA's following the impact test (Phase I) of S-6 for use in this third phase of testing. However, AEC/SNS has provided 16 T-111 FSA's instead of the 12 originally designated for the S-6 hardware. Four (4) of the original iridium FSA's will be held aside for these additional tests, leaving only eight iridium FSA's in the test HSA. In this manner, the FSA's will not have been subjected to the simulated pad impact prior to the individual FSA impacts in Phase III. Test conditions being considered for Phase III are: (1) 120 fps impact on concrete; (2) 120 fps impact on angle iron; (3) 285 fps impact on soft dirt; and (4) undesignated.

As mentioned previously, the makeup of the S-6 test hardware has changed by direction of AEC/SNS. In addition to the reduction of iridium FSA's in the HSA test specimen (from 12 to 8), the following constitutes the total FSA inventory to be used in the three test phases:

- 16 T-111's simulated PISA's
- 2 Type III (high W) EI iridium PISA's
- 6 Type II (low W) EI iridium PISA's
- 4 Type WC (high W) ML/ORNL iridium PISA's
- 12 prime graphite impact shells
- 16 1969 vintage graphite impact shells

LASL has proposed (GE has concurred) to use these spheres in the following manner:

	<u>Type II</u>	<u>Type III</u>	<u>Type WC</u>	<u>T-111</u>
Layer #1 (impact end) of HSA	-	2	2	-
Layer #2 of HSA	3	-	1	-
Remainder of HSA	-	-	-	16
Other impacts (single FSA's)	3	-	1	-

Projected costs for performing the S-6 test at Sandia were higher than anticipated. Because of budgetary restrictions, SNS Safety and Reliability decided to have the test performed at LASL facilities and by LASL personnel. The impact portion of the test will be conducted by dropping the HSA test specimen from a 150 foot tower after the HSA is preheated to launch temperatures.

3.1.5 OTHER FSA/PISA TESTS

In January, an FSA with thoria simulant was heated to 2300°F (1260°C) and immediately immersed in ambient (40°F) water. The FSA specimen was made up with a 0.410 inch concentric graphite impact shell and PISA 550, having iridium hemispheres S/N 435 and 400 from Lots 24 and 25. After being heated up, the FSA was pushed from the furnace onto a graphite ramp (about 18 inches long), from which it dropped into a tank of water. There was no steam or eruption of water from the tank; the sphere evidently quenched immediately. The FSA was examined after the test and no damage was found either to the graphite shell or to the PISA.

A similar test was conducted on a bare PISA (MHT-14). The PISA was heated to 2670°F (1465°C) and then immersed in 50 to 60°F water. Again, no steam or water eruption occurred. The test was then immediately repeated on the same PISA with the same response. The specimen was examined under a microscope at 60X and no defects were found in the PICS or its weld joint. Both water immersion tests (FSA, PISA) were conducted to determine if the PISA's were prone to failure from thermal shock.

3.1.6 SAFETY SEQUENTIAL TESTS (SST'S)

A test readiness review was held at LASL in April for the Safety Sequential Tests. These tests, run with live fueled FSA's, had been postponed during the previous 12 months due to changes in program emphasis and limited availability of hardware. The hardware was delivered by ML/MRC and testing commenced during the month of July.

The test philosophy was essentially to test the worst case, simulating a launch vibration, short operating life and minimum angle re-entry (at earth escape velocity), but with nominal impact. This series is not completely self-consistent since escape velocity ($\sim 36,000$ fps), minimum

angle re-entries and 30-day space operation (low earth orbit) are mutually exclusive conditions, but the technical difference to the test hardware is probably minimal.

Aging of the first four FSA's (MHFT-25, -26, -27, -28) as a group was completed at LASL in August. The aging temperature for these spheres was in the 1260-1270⁰C range on the exterior of the GIS's. After 738 hours of aging, MHFT-28 was taken out for inspection. Both vents were found to be plugged; elements found were calcium, plutonium, oxygen, and silicon. After this outcome, the decision was made to determine the condition of the vents in the other three FSA's before they were impacted.

MHFT-25, -26, -27 were pulled out after aging for 857 hours, and their graphite impact shells were removed. The vents in all three were found to be open, with dark spots around the vents in MHFT-25 and -26. There were no deposits found around the vents in MHFT-27, but the graphite shell was found to be pitted as deep as had been seen in any previous test.

The second group of three FSA's, MHFT-29, -30, -31, was aged for 847 hours at the 1260-1270⁰C temperature, and these completed aging during November. The vents in MHFT-29 and -31 were found to be open after aging, but the vents in MHFT-30 were found to be closed as determined by He leak detection in vacuum. Table 3-1 is a summary of the test results as of the end of the year.

3.1.7 SAFETY VERIFICATION TESTS (SVT'S)

A tentative plan and an alternate have been proposed by LASL for the Safety Verification Tests to be performed for SNS Safety and Reliability. The two plans are outlined in Figures 3-13 and 3-14. Figure 3-13 shows the plan based on the use of six FSA's from one iridium source and four FSA's from another source. The plan in Figure 3-14 is based on the use of six FSA's from a single source. By direction of the AEC/SNS, there will be six FSA's having the WC type (high tungsten) iridium (EI). Consequently the proposed SVT plan will have to be revised. As supporting data for these tests, SNS has requested that GE furnish operational, re-entry, and impact temperatures for the FSA's within the HSA having an internal helium atmosphere as now proposed. This data is currently being generated.

Table 3-1. SST Test Conditions and Results

FSA No.	Aging History		Sequential Tests	Results
	Temp °C	Time Hours		
MHFT-25	1268	857	1) Launch Vibration 2) Max. Re-entry Pulse* 5 min. @ 1850 °C	Both vents plugged to He @ 25 psi after 2)
MHFT-26	1268	857	1) Launch Vibration 2) Max. Re-entry Pulse* 5 min @ 1850 °C 3) Impacted @ 246.5 fps, 1370 °C, 45° from cap.	Passed impact. One vent plugged by post test inspection
MHFT-27	1268	857	1) Launch Vibration 2) Max. Re-entry Pulse* 5 min. @ 1850 °C 3) Impacted @ 249 fps, 1370 °C, 180° from cap.	Failed impact. Minor cracks Ir stretched over large fuel fragments
MHFT-28	1268	857	1) Launch Vibration	Both vents plugged by post test inspection
MHFT-29	1268	847	1) Launch Vibration 2) To get re-entry pulse* and to be impacted	
MHFT-30	1268	847	1) Launch Vibration 2) Max. Re-entry Pulse* 5 min. @ 1850 °C 3) Impacted @ 250 fps, 1550 °C, 180° from cap.	Failed impact Fingerprint and hoop stress cracks
MHFT-31	1268	847	1) Launch Vibration 2) Max. Re-entry Pulse* 5 min @ 1850 °C 3) Impacted @ 243 fps, 1550 °C, 180° from cap	Failed impact

* Thermal only - no dynamic heating

Figure 3-13. SVT Proposed Plan

Figure 3-14. Alternate SVT Plan

3.2 SAFETY ANALYSIS AND REPORTS

3.2.1 EXPLOSION ENVIRONMENT

Parametric data for the launch pad explosion environment was generated for use in the Updated Safety Analysis Report and accompanying analyses. The data were generated in a manner similar to that used in the PSAR (Document Nos. GESP-7075, GESP-7076, dated July 1972) with the exception that the shock wave duration was deduced from the static impulse on the basis of a triangular shock pulse. Typical results are shown in Figure 3-15 for the Transtage only.

Figure 3-16 shows the locations of the RTG's and satellites with respect to the explosion center for the Transtage. These locations are based on the current Martin-Marietta Document MGR-72-260 and included as Figure 3-17 herein for reference.

Similar data for an explosion occurring in the launch vehicle as a function of time after liftoff (altitude) were generated. Figures 3-18 and 3-19 show curves of static overpressure and static impulse respectively. These curves were generated on the basis of high explosive data variation with altitude and then were normalized to hypergolic data at sea level. If a comparison is made of these curves with Figures 3-16 and 3-20, one can see that the overpressures and impulses never exceed the sea level values. Only curves of blast wave duration increase with altitude as shown in Figure 3-21 which condition is necessary to fit the slower decaying values of impulse compared with those of overpressure.

Figure 3-15. Shock Wave and Particle Velocity vs Distance from Explosion Center Transtage

Figure 3-16. Static Overpressure vs Distance from Explosion Center Transtage

3.2.2 NUCLEAR CRITICALITY

LASL completed a series of calculations to determine the nuclear criticality of the MHW Heat Source Assembly (HSA), its components (FSA's, PISA's) and the complete RTG in various configurations. The study included calculations for the RTG shipping container also. Results were obtained by the use of ANDY and KENO Monte Carlo codes with verification calculations being made using the DTF and TWOTRAN transport codes and the MCN Monte Carlo code. The results are presented in Table 3-2.

3.2.3 LIQUID PROPELLANT FIREBALL/AFTERFIRE

Sandia updated the liquid propellant fireball/afterfire model to include hypergolic fuels. Figure 3-22 is included to show the effect on the Titan IIIC liquid propellant fire environment used in MHW analyses. The solid curves in the figure are based on the 1970 report (Bader, B. E.; Donalson, A. B. and Hardee, H. C., "Liquid-Propellant Rocket Abort Fire Model". SC-RR-70-454, October 1970) and the dashed curves on the updated analysis. The overall differences are slightly lower temperatures and heat fluxes for the updated model.

Figure 3-17. Payload Configuration

Figure 3-18. Static Overpressure vs Time After Liftoff Transtage

Figure 3-19. Static Impulse vs Time After Liftoff Transtage

3.2.4 ABORT ENVIRONMENTS - GENERAL

The AFWL summarized their assessment of the launch pad abort environments for the LES 8/9 mission in a letter to Air Force Systems Command. Each of the following environments was addressed:

Figure 3-20. Static Impulse and Positive Phase Duration vs Distance from Explosion Center Transtage

Figure 3-21. Blast Wave Duration vs Time After Liftoff Transtage

1. Overpressure

Stage	Propellant Weight (lb)	Yield (%)	Distance (ft)	Scaled Distance	Static Overpressure	Static Impulse
1	259,530	3	79.7	4.016	8 psi	0.13 psi-sec
2	67,322	5	32.72	2.18	13 psi	0.22 psi-sec
3	23,477	5	13.88	1.32	20 psi	0.35 psi-sec

Table 3-2. Results of Nuclear Criticality Analysis

Configuration	k_{eff}	M
Single HSA	0.173	1.21
2 HSA's end-to-end	0.187	1.23
Single FSA	0.152	1.18
Grouped HSA's
0.265	1.36

Grouped RTG's	0.248	1.33
0.173	1.21
Grouped FSA's		
5 x 5 x 5 array*		
unreflected	0.35	1.54
steel reflected	0.49	1.96
water flooded and reflected	0.54	2.17
concrete reflected	0.51	2.04
Grouped PISA's		
5 x 5 x 5 array*		
unreflected	0.61	2.56
steel reflected	0.82	5.55
water flooded and reflected	0.84	6.25
concrete reflected	0.80	5.0
RTG Shipping Container		
single RTG-SC	0.25	1.33
infinite array		
with water shielding (in SC)	~ 0.25	1.33
without water shielding (in SC)	≤ 0.32	≤ 1.47

*Although there are only 96 FSA's or PISA's available on the LES 8/9 mission (4 HSA's of 24 FSA's each), the 5x5x5 (125 FSA's or PISA's) array was easier to run using the KENO program.

Stagnation and reflected overpressure, blast wave velocity particle velocity, and positive phase duration are stated as being derivable from the static overpressure and impulse by well defined and accepted techniques.

1. Fragments (Projectiles) - These are discounted at the RTG position.
2. Fireball - Use of the Sandia constant fuel addition rate model (SC-RR-70-454) is recommended until Sandia is able to modify it based on additional data.
3. Liquid Propellant Afterfire - Actual thermal model not given. Two environments are postulated:
 - a. Fuel (N_2H_4 - UDMH) burning in air.
 - b. Oxidizer (N_2O_4) reaction with water to form nitric acid. Situation of fuel and oxidizer reacting together on the launch pad in a flow controlled manner to produce a more severe environment is discounted.
4. Solid Propellant Fire - Flame temperature of 4800 - 4900^oF. No heat flux or duration given, but statement is made that duration would be a function of the smallest dimension (or 36 inches for the largest conceivable piece of propellant; i.e., a full web thickness).
5. Impact - Freefall from 120 feet listed as 90 ft/sec.

Figure 3-22. Fireball and Afterfire Model Titan IIIC Liquid Propellants

Fallback or tipover as a result of thrust termination is discounted because of the use of the solid rocket motors. The only way their thrust can be terminated is by vehicle destruct, either by the vehicle destruct system or by command destruct.

3.2.5 PAD ABORT BLAST ANALYSIS

A study was completed to determine the response of the LES spacecraft and MHW-RTG's to blast environment associated with a launch pad abort, assuming an explosion of the propellants in the Transtage to occur. The configuration (spacing) of the spacecraft on the launch vehicle is that shown in Figure 3-17. Blast wave components used in the calculations were based on this configuration and were determined from the curves. Table 3-3 summarizes those overpressure components acting on the spacecraft and RTG's for a 0.34 percent yield explosion, the 99.5% confidence interval.

Blast loading of the structures (SOLRAD, LES, RTG's) was based on the technique given in "Explosive Shocks in Air" by G. F. Kinney (New York: Macmillan Company, January 1962) and used composite triangular shaped pulses for the static, stagnation, reflected, and drag pressure components. Due to the configuration arrangement, the RTG end (of each stacked pair) nearest the LES does not experience reflective shock resulting from direct impingement of the shock wave due to interference of the spacecraft structure.

Table 3-3. Blast Wave Overpressures
(0.34% Yield Explosion)

Overpressure - psi				
Location	Static	Stagnation	Reflected	Drag
SOLRAD Sta. + 57	32	55	120	19.5
LES 8/9 Sta. - 12	16	22	45	14.5
RTG Sta. - 70	11.5	14	30	13

The LES 8/9 spacecraft response to the blast environment was examined for transitional failure states at explosive yields between 0.3 percent and 0.5 percent. Table 3-4 summarizes the design capabilities and response of major sections of the LES spacecraft structure. Based on these results, the residual spacecraft structure remaining after the blast wave

has passed will consist of the inner decagon structure, support keels and bottom panel tied down to the payload support truss. The RTG's will be attached to the decagon structure.

The RTG's were analyzed for two specific conditions of loading resulting from the blast overpressures:

1. Symmetrical blast wave engulfing the RTG's, either attached or freed from the residual spacecraft structure.
2. Unsymmetrical loading of the RTG's when mounted to the truss structure.

The second case is probably the most realistic due to the intervening structures of the SOLRAD and LES spacecraft.

Table 3-5 presents the stresses anticipated for the RTG and HSA supports for the 0.34 percent yield case. Results are shown for both the RTG's attached to the truss and free of the spacecraft. The effect of the preload support of the HSA by the end closure member has been conservatively neglected. Based on these results, the analysis shows no anticipated failure modes which would free the HSA from the Converter for a symmetrical loading by the shock wave.

An unsymmetrical pressure wave acting on the RTG's is anticipated from the blast wave due to blockage and attenuation of the primary wave front expanding around the SOLRAD and LES spacecraft. The unsymmetrical effects have been estimated to be as high as 20 to 50 percent of the static or side-on pressure. These effects, acting on the outboard side of the RTG's will tend to force the pairs of RTG stacking adapters failure, at 18.7 g's, the transition failure can be either a single RTG or a tandem pair because of the close failure levels. For purpose of the analyses a single RTG was chosen as being representative of either case.

Table 3-4. Spacecraft Structural Characteristics

STRUCTURE	CONSTRUCTION	f_N CPS	P_{ALLOW} (PSI)	(1) P_{MAX} EQUIVALENT PRESSURE (PSI)		Q (2)	FAILURE MODE
				0.5%	0.3%		
N/S PANELS S60010	6061-T6 HONEY COMB $t_F = 0.005$	425	5.5	27	23	1.90	PANEL BENDING
N/S CENTER PANEL - S60013	6061-T6 HONEY COMB $t_F = 0.005$	1020	15.3	27	23	1.90	PANEL BENDING
E/W PANEL S60012	6061-T6 HONEY COMB $t_F = 0.005$	250	5.2	27	23	1.90	PANEL BENDING
PLATFORM PANEL S-61214	6061-T6 HONEY COMB $t_F = 0.032$	290	84	92	70	1.68	PANEL BENDING
S/C RADIAL KEEL S61165	7075-T6 $t = 0.072$	40	150 ⁽³⁾	44	32	0.80	PANEL SHEAR
DECAGON PANEL STIFFNER	7075-T6 $t = 0.062$	-	136 ⁽³⁾	54	42	1.00	COLUMN INSTABILITY
PLATFORM PANEL SUPPORT TUBE	6061-T6 5/8 DIA X 0.035	-	24 ⁽³⁾	54	42	1.00	COLUMN INSTABILITY

NOTES:

- (1) STATIC EQUIVALENT PRESSURE
BASED ON DYNAMIC RESPONSE CHARACTERISTICS
- (2) DYNAMIC RESPONSE FACTOR
- (3) REF TO EQUIVALENT PRESSURE
APPLIED TO PLATFORM PANEL (BOTTOM)

Table 3-5. RTG Blast Wave Response

SYMMETRICAL LOADING COMPONENT RESPONSE - 0.34% YIELD

COMPONENT	f_N (cps)	RESPONSE STRESS (PSI)	DESIGN ALLOWABLE (PSI)	FAILURE MODE
END DOME	378	5700	26000	BUCKLING
Be CASE		810	24000	LONGITUDINAL COMPRESSION
		1380	24200	CIRCUMFERENTIAL BUCKLING

INERTIAL RESPONSE - 0.34% YIELD

CONDITION	HSA SUPPORT LBS	RTG MOUNT LBS	THERMOPILE LBS
I	2430	3820	725
II	1580	3570	360
III	1180	-	-
DESIGN ALLOWABLE	6500	23400	4670 TEST REF (b)
FAILURE MODE	PRELOAD CAP BENDING	FLANGE TENSION AT INTERFACE ATTACHMENT	

- I SINGLE RTG - TRUSS MOUNTED
- II TANDEM RTG'S - TRUSS MOUNTED
- III SINGLE RTG FREE OF TRUSS

NOTES:

1. RIGID HSA
2. SC/RTG TRUSS $f_N = 40$ cps
3. AMPLIFICATION FACTOR OF 2.0 APPLIED TO INDUCED INERTIAL RESPONSE - f_N UNKNOWN
4. HSA RESPONSE NEGLECTS PRELOAD

Failure of the mounts (truss or stacking adapter) will result in collision between adjacent RTG's (i.e., from each spacecraft). Figure 3-23 shows the sensitivity of the lateral mode failure response as a function of blast yield. At the 0.34 percent maximum yield explosion for a pad abort involving hypergolic fuels, the RTG's would experience a lateral load of about 16 g's for a 50 percent unsymmetrical blast loading effect. This would not result in release of the RTG's (17.5 g capability of RTG-to-LES truss). Thus, even a maximum unsymmetrical blast loading is not expected to release the RTG's from the spacecraft. The RTG's will still be intact after the explosion.

Figure 3-23. RTG Lateral Impact Effect of Lateral Pressure Ratio on Yield

3.2.6 FLIGHT P74-1 MISSION REQUIREMENTS/CHARACTERISTICS

In April, a meeting was held at the AEC/SNS with the Air Force (SAMSO) to discuss the mission requirements for Flight P74-1 (LES 8, 9-SOLRAD). The main topic of the meeting involved the guidance backup modes to be used in the event of anomalous performance in any of the launch vehicle stages. With regard to malfunctions occurring in the Transtage, the philosophy expressed jointly by GE/SNS/APL was as follows:

1. First Transtage Burn Malfunctions (from earth parking orbit):

- a. Malfunctions which result in the RTG's remaining in the parking orbit are undesirable because of the short (orbital) lifetime and random re-entry aspects. Conversely, contingency maneuvers (i.e., restart, longer burns, etc.) that can place the RTG's into orbits with much longer lifetimes are preferable.
- b. Powered re-entries should be avoided.
- c. Release of the LES 8/9 spacecraft is desirable for all contingencies. The spacecraft should be released especially if an immediate re-entry is imminent or if the vehicle remains in the parking orbit, these resulting from any malfunctions. The use of the standard LES 8/9 separation sequence (as was being considered by the Air Force) was questioned as not being expeditious because of insufficient time to accomplish these maneuvers prior to re-entry.

2. Second Transtage Burn Malfunctions (to circularize at synchronous altitude):

- a. Release of the LES 8/9 spacecraft is desirable for all contingencies. If the first burn of the Transtage has been successful, the lifetime of the transfer orbit is estimated to be greater than 1000 years, which is desirable. If the second burn cannot be effected for any reason, ejection of the spacecraft in the transfer orbit removes any doubt about the Transtage firing at some later time to produce undesirable re-entry conditions (considerably shorter orbit lifetimes, immediate re-entry, powered re-entry).

One of the backup modes for the mission under consideration by the Air Force was to require that the Transtage engines be burned to produce an orbit of at least 80 X 95 nm. in the event of anomalous performance of Stage O, I, or II, resulting in Stage II shutdown with a resulting parking orbit of less than 80 X 95 nm. GE/AEC's position, as expressed in the meeting, was to prefer an immediate ballistic re-entry (ocean impact) rather than attaining a parking orbit that would yield an orbit with a few days duration and with a random re-entry.

The Air Force (SAMSO) subsequently incorporated a backup mode in the specification for the Program 624A (LES 8/9) mission designed to boost the LES 8/9 spacecraft into an orbit having a lifetime of at least 100 years. This backup guidance mode is included in the event of anomalous performance of Stage O, I, or II resulting in Stage II shutdown prior to attaining parking orbit. If a parking orbit of less than 80 X 95 nm. results from the Stage II burn, the Transtage engines will be burned to produce an orbit of at least this magnitude. However,

the backup mode is not to be initiated at such a point (prior to attaining the nominal parking orbit) that the 100-year lifetime orbit cannot be reached.

A meeting was held at SAMSO in June to familiarize members of the INSRP with the guidance system and trajectory for the LES 8/9 mission (Flight P74-1). Items of general interest for the mission are:

1. Transtage burns are based on vehicle position and not on time after launch.
2. Payload separation is controlled by a timer which is activated after the end of the Transtage second burn.
3. The Standard Orbital Orientation (SOO), in which the roll axis of the Transtage is perpendicular to the radius vector, is maintained throughout the parking orbit until the Transtage is reoriented in preparation for the first burn.
4. Release of the LES 8/9 spacecraft will occur after the Transtage second burn (action of timer) even if the first burn is incorrect. The missile guidance computer may think that the first burn is correct if there is an error in the computer such as a bit failure or loss of a word.
5. If an early burn of the Transtage occurs, actuator (motor gimbal control) hydraulic pressure will not be available, and the engines will move about (flop) without control. This will probably result in a tumbling vehicle with breakup occurring.
6. Most likely the payload will not be released after a parking orbit malfunction because the inertial guidance system will probably not know that anything is wrong (timing action for payload release occurs only after second burn).

3.2.7 SUPPORTING DATA FOR SAFETY REPORTS

The Air Force supplied additional information for use as input to the safety reports (SAR's). This information includes the "P74-1 Mission Planning Trajectory" (Document No. MCR-73-204, Preliminary plus Revision 1), and the "Transtage Orbital Anomaly Probabilities Report" (Document MCR-73-192, Revisions 1 and 2). The latter document includes the potential failures that can occur in the Transtage as used in the P74-1 mission. A summary of the failure combinations and associated probabilities from the last revision to that study is presented in Table 3-6. As seen from this table, the modes of failure that have been considered are velocity magnitude errors (ΔV addition) errors in the position in orbit at which the burn

Table 3-6. Transtage Abort Modes and Probabilities

ORBIT	ΔV	ORIENTATION	θ - TRUE ANOMALY		
			EARLY	ON-TIME	LATE
PARK	NOMINAL	RANDOM	$<10^{-10}$ (2)	3.3×10^{-4}	$<10^{-10}$ (2)
		CORRECT	///	SUCCESS	///
	NONE	2.7×10^{-3}			
	DEPLETION	RANDOM	7×10^{-6} (1)	2.3×10^{-9} (3)	$<10^{-10}$ (2)
		CORRECT	///	7×10^{-6} (1)	///
TRANSFER	NOMINAL	RANDOM	$<10^{-10}$ (2)	1.65×10^{-3}	$<10^{-10}$ (2)
		CORRECT	///	SUCCESS	///
	NONE	4.3×10^{-3}			
	DEPLETION	RANDOM	1.05×10^{-5} (1)	1.73×10^{-8} (3)	$<10^{-10}$ (2)
		CORRECT	///	1.05×10^{-5} (1)	///

(1) FAILURE OF DISCRETE IN MGC

(2) COMPUTATION ERROR BY MGC. LATE BURNS WOULD NOT OCCUR DUE TO A DISCRETE FAILURE SINCE MGC WOULD SHUT EVERYTHING DOWN IF BURN DID NOT OCCUR ON TIME (I.E., WITH CORRECT TRAJECTORY COMPUTATION).

(3) DOUBLE FAILURES.

occurs (Time of Burn), and spatial misalignment of the velocity additional vector (Orientation). Combinations of these failure modes yield the probabilities shown in the table.

3.2.8 V, γ, PROBABILITY STUDY

After the preliminary V, γ, study was completed by the Aerospace Corporation (see GESP-7097, "Multi-Hundred Watt Radioisotope Thermoelectric Generator Program." Part I Annual Report. 1 July 1971 - 31 December 1972), the AFWL undertook a study to develop a more detailed mapping of the initial re-entry conditions associated with the potential Transtage malfunctions. This study included the probabilities of malfunctions related to the LES 8/9 mission as translated to the initial re-entry conditions. Probabilities for the potential malfunctions were obtained from the Martin-Marietta/Denver study reported in Document MRC-73-192 plus revisions (refer to Section 3.2.7 of this annual report).

The first part of the study was based on a preliminary reference trajectory for the LES 8/9 mission (prior to the issuing of the P74-1 Mission Planning Trajectory, MRC-73-204) plus associated ground rules and mission specifications. Since the orientation of the Transtage

heading vector was undefined at that time for the transfer orbit, five potential direction cases were defined so that the combined results would bracket the finalized orientation. A later revision to the study eliminated the five leading cases; the category of random orientations for the second burn was used instead, since this would effectively cover the conditions applicable to the previous five cases. Tables 3-7 and 3-8 present the results of the latter V, γ study for the first burn and second burn failures, respectively. In these tables the types of resulting orbits and re-entries are listed also for each of the cases based on the malfunctions investigated. A re-entry event is defined to occur when the geodetic radius is equal to 400,000 feet at any point in the resulting orbit (based on a given set of initial conditions and one or more malfunctions). A prompt re-entry is one in which the body in question is captured by the atmosphere and never again exceeds the 400,000 feet altitude and with eventual impact on Earth. Powered re-entries are those prompt re-entries having burn durations longer than the elapsed time from malfunction to the 400,000 feet altitude. A decay re-entry is defined to occur when the perigee of the resulting orbit is not less than 400,000 feet on the first circuit, but, due mainly to atmospheric drag, will decay in speed and altitude such that the body will be eventually captured by the atmosphere, with a characteristic final re-entry speed of approximately 25,700 fps and a very small re-entry angle. An escape event is defined to occur when the resulting orbit is hyperbolic and the body escapes the Earth's gravitational pull, even though the initial part of the trajectory may have an altitude lower than 400,000 feet.

The final revision to the study was based on the P74-1 Mission Planning Trajectory (MRC-73-204, Rev. 1) and the updated Transtage failure analysis (MRC-73-192, Rev. 1) as shown in Table 3-5. Tables 3-9 and 3-10 give the corresponding results for the first burn and second burn failures, respectively. Failures that have a probability of less than 10^{-9} were not included in these results, thus the reason for the elimination of the failure mode case having a nominal V, random orientation, and early burn. A comparison of the overall range of potential V, conditions based on the mission planning trajectory and the preliminary mission trajectory is given in Table 3-11.

Table 3-7. First Burn V, γ Results (from park orbit)

ΔV α θ	NOM. RANDOM EARLY	NOM. RANDOM ON-TIME	NONE	TOTAL RANDOM EARLY	TOTAL CORRECT ON-TIME
V_{MAX} (FPS)	36,299	34,035	33,080*	36,360	--
V_{MIN} (FPS)	29,517	33,448		28,898	--
γ_{MAX} ($^{\circ}$)	3.14	83.4	< 1	89.4	--
γ_{MIN} ($^{\circ}$)	0.41	11.1		0.79	--
RE-ENTRY					
PROMPT	0.28	0.30	--	0.28	--
POWERED	--	--	--	--	--
ESCAPE	0.11	--	--	0.14	--
DECAY	0.61	0.70	1.00	0.58	1.00
PROMPT RE-ENTRY					
T_{MAX}	3.5 MO.	8.0 HR.	--	45 YR.	
T_{MIN}	2.3 HR.	3.1 HR.	--	1.9 HR.	
DECAY					
$T_{MAX}^{1/\beta}$	10^{145} CEN.	10^{145} CEN.	1.6 CEN.	10^{147} CEN.	10^{147} CEN.
$T_{MIN}^{1/\beta}$	21 DAY	2.5 YR.	1.6 CEN.	10 DAY	10^{147} CEN.

* FIRST PERIGEE - NOT AT 400,000 FT.

Table 3-8. Second Burn V, γ Results (from transfer orbit)

ΔV α θ	NOM. RANDOM EARLY	NOM. RANDOM ON-TIME	NONE	TOTAL RANDOM EARLY	TOTAL CORRECT ON-TIME
V_{MAX} (FPS)	33,439	31,824	25,700	36,751	---
V_{MIN} ($^{\circ}$)	18,325	18,828		13,507	---
γ_{MAX} ($^{\circ}$)	22.8	22.2	0.2	37.7	---
γ_{MIN} ($^{\circ}$)	0.47	1.5		0.35	---
RE-ENTRY					
PROMPT	0.58	0.62	--	0.45	---
POWERED	0.083	0.018	--	0.20	---
ESCAPE	--	--	--	0.11	1.0
DECAY	0.34	0.36	1.00	0.24	---
PROMPT RE-ENTRY					
T_{MAX}	8.4 HR	3.6 HR.	--	3.8 YR.	---
T_{MIN}	1.9 MIN.	5.3 MIN.	--	2.0 MIN.	---
POWERED RE-ENTRY					
T_{MAX}	3.5 MIN.	7.2 SEC.	--	5.3 MIN.	---
T_{MIN}	0.64 SEC.	0.69 SEC.	--	1.1 SEC.	---
DECAY					
$T_{MAX}^{1/\beta}$	5.0 CEN.	1.3 CEN.	5.0 HR.	100 CEN.	---
$T_{MIN}^{1/\beta}$	4.0 HR.	2.6 DAY	5.0 HR.	4.5 HR.	---

Table 3-9. First Burn V, γ Results (from park orbit)

ΔV & θ	TOTAL RANDOM ON-TIME	NOM. RANDOM ON-TIME	NONE	TOTAL RANDOM EARLY	TOTAL CORRECT ON-TIME
V_{MAX} (FPS)	38,054	32,030		39,195	28,964
V_{MIN} (FPS)	13,594	18,586		12,790	
γ_{MAX} ($^{\circ}$)	35.1	20.7		36.4	26.1
γ_{MIN} ($^{\circ}$)	1.38	1.41		0.65	
RE-ENTRY					
PROMPT	0.29	0.34	--	0.32	1.0
POWERED	0.40	0.34	--	0.37	--
ESCAPE	0.12	--	--	0.12	--
DECAY	0.19	0.32	1.0	0.19	--
PROMPT RE-ENTRY					
T_{MAX}	3.3 DAY	3.1 HR.	--	9.9 MO	1.9 HR.
T_{MIN}	54 SEC.	1.5 MIN.	--	6.4 SEC.	1.9 HR.
POWERED RE-ENTRY					
T_{MAX}	6.4 MIN.	4.0 MIN.	--	7.2 MIN	--
T_{MIN}	44 SEC.	1.8 SEC.	--	2.4 SEC	--
DECAY					
T_{MAX}/β	100 CEN.	25 YR.	2.9 HR.	100 CEN.	--
T_{MIN}/β	19 HR.	4.3 HR.	2.9 HR.	2.3 HR.	--

REF: "P74-1 MISSION PLANNING TRAJECTORY," MCR-73-204, REV. 1

3.2.9 RE-ENTRY BREAKUP STUDY

In order to determine the complete response of the RTG's during potential re-entry situations, the effects of the launch vehicle stages (where applicable) and the spacecraft must be assessed. The Air Force (SAMSO/AFWL) funded TRW to undertake a study to define environments imposed on the RTG's until their eventual release from the LES spacecraft occurred. Representative re-entry cases were chosen by the INSRP re-entry subpanel based on the V, γ , probability study. Results that were used are those summarized previously in Tables 3-7 and 3-8.

Table 3-12 lists the cases chosen by the re-entry subpanel to be analyzed. For a meaningful tumble rate to be used for Cases 5 and 7, SAMSO furnished to TRW the maximum pitch rate

Table 3-10. Second Burn V, γ Results (from transfer orbit)

ΔV α θ	TOTAL RANDOM ON-TIME	NOM. RANDOM ON-TIME	NONE	TOTAL RANDOM EARLY	TOTAL CORRECT ON-TIME
V_{MAX} (FPS)	34,111	34,030		36,347	
V_{MIN} (FPS)	33,443	33,446		28,633	
$\gamma_{MAX} (^{\circ})$	89.9	84.4		89.9	
$\gamma_{MIN} (^{\circ})$	11.8	11.6		1.66	
RE-ENTRY					
PROMPT	0.29	0.31	--	0.32	--
POWERED	--	--	--	--	--
ESCAPE	--	--	--	0.14	--
DECAY	0.71	0.69	1.0	0.54	1.0
PROMPT RE-ENTRY					
T_{MAX}	8.8 HR	8.0 HR		2.4 YR	
T_{MIN}	2.9 HR	3.1 HR		1.8 HR	
DECAY					
T_{MAX}^{β}	10^{147} CEN.	10^{145} CEN.	26 YR.	10^{147} CEN.	4.3 MO.
T_{MIN}^{β}	4.3 MO.	1.1 YR.	26 YR.	7.7 DAY	4.3 MO.

REF: "P74-1 MISSION PLANNING TRAJECTORY" MCR-73-204, REV. 1

Table 3-11. P74-1 Mission Re-Entry Summary - Post Earth Orbit

	AEROSPACE 73-2620-50-1	MARTIN-MARIETTA MRC-73-204, REV. 1
V_{MAX} (FPS)	36,751	39,195
$\gamma @ V_{MAX}$ ($^{\circ}$)	4.54	3.32
V_{MIN} (FPS)	13,507	12,790
$\gamma @ V_{MIN}$ ($^{\circ}$)	20.4	9.00
γ_{MAX} ($^{\circ}$)	89.4	89.9
$V @ \gamma_{MAX}$ (FPS)	33,442	33,472
γ_{MIN} ($^{\circ}$)	0.349	0.647
$V @ \gamma_{MIN}$ (FPS)	30,974	26,364
RE-ENTRY		
PROMPT	.42	0.32
POWERED	.077	0.21
ESCAPE	.083	0.12
DECAY	.42	0.35
PROMPT RE-ENTRY		
T_{MAX}	45.4 YR.	2.39 YR.
T_{MIN}	1.93 MIN.	6.37 SEC
POWERED RE-ENTRY		
T_{MAX}	5.32 MIN.	7.23 MIN.
T_{MIN}	0.64 SEC.	1.84 SEC.
DECAY		
$T_{MAX}^{1/\beta}$	1.9×10^{147} CEN.	2.5×10^{147} CEN.
$T_{MIN}^{1/\beta}$	4.0 HR.	2.3 HR.

Table 3-12. LES 8/9 Breakup Analysis Cases (Initial Conditions)

Case No.	Velocity (Kfps)	γ , Degree	α , Degree	Configuration $\neq$
1	28.0	- 16.0	0.0	TS/SC/RTG's (post 1st burn)
2	28.0	- 16.0	- 15.0	TS/SC/RTG's (155 sec burn left)
3	25.7	- 0.2	165.0	TS/SC/RTG's (pre 1st burn)
4	33.5	- 82.0	90.0	SC/RTG's (post 2nd burn)
5	33.5	- 82.0	tumbling	SC/RTG's (post 2nd burn)
6	33.5	- 40.0	90.0	SC/RTG's (post 2nd burn)
7	33.5	- 40.0	tumbling	SC/RTG's (post 2nd burn)
8	33.2	- 3.1	12.5	TS/SC/RTG's (post 1st burn)
9	33.5	- 60.0	90.0	SC/RTG's (post 2nd burn)
10	33.5	- 82.0	90.0	TS/SC/RTG's (post 2nd burn)
11	33.2	γ_{min}^*		TS/SC/RTG's (post 1st burn)
12	36.0	- 40.0	90.0	SC/RTG's (post 2nd burn)

* The γ_{min} value was subsequently determined by TRW with a slight adjustment in the velocity for ease of computation.

TS = Transtage
 $\neq$ SC = Spacecraft (LES 8 or 9)
 RTG = MHW Radioisotope Thermoelectric Generator

which the TS/SC/RTG configuration could withstand structurally without breaking up. This value was 98.5 rpm, at which value the spacecraft truss structure was predicted to fail at Station 77. TRW then performed 6 degree-of-freedom studies of the Transtage/Spacecraft configuration at various initial tumble rates and found that damping occurred relatively quickly in all cases. Also the stable trim altitude was not appreciably affected by the tumble rate. Consequently, Cases 5 and 7 were eliminated because of their reverting essentially to 4 and 6, respectively, based on those results. The larger number of cases at the 33,500 ft/sec velocity and -40 to -82 degrees re-entry angles were selected because of the emphasis in the potentially high stress situations for the HSA aeroshell.

Table 3-13 summarizes the results of the breakup study in terms of the conditions existing at the time that the RTG's become free of the spacecraft. Where two release altitudes are given, the top number refers to the outboard RTG, and the bottom number, to the inboard or nearest RTG to the LES. The column on the left shows the original case number established by the re-entry subpanel and as given in Table 3-12.

Table 3-13. Conditions at RTG Release

CASE IDENTIFICATION						RELEASE CONDITIONS			
ORIG. NO.	TRW NO.	CONFIG.	V - FPS	γ - DEG	α - DEG	ALTITUDE - ft	V - FPS	γ - DEG	α - DEG
3	1	TS/SC	25,700	- 0.2	+ 165	266,000	25,770	- 0.17	+ 152
2	2	TS/SC	28,000	- 16.0	- 15	230,000	28,640	- 16.0	+ 15
1	3	TS/SC	28,000	- 16.0	0	190,000	27,750	- 15.7	+ 27 °
11	4	TS/SC	32,660	- 5.1		277,000	32,750	- 3.39	+ 152
						262,000	32,730	- 3.11	+ 138
8	5	TS/SC	32,762	- 4.2	+ 12.5	285,000	32,840	- 2.05	+ 152
						277,000	32,830	- 1.81	+ 138
10	6	TS/SC	33,500	- 82.0	+ 90.0	197,000	33,540	- 8.2	+ 52 T
6	7	LES 9	33,500	- 40.0	+ 90.0	130,000	31,760	- 82.2	+ 150
9	8	LES 9	33,500	- 60.0	+ 90.0	140,000	31,840	- 39.9	+ 158
4	9	LES 9	33,500	- 82.0	+ 90.0	130,000	31,460	- 59.2	+ 150
12	10	LES 9	36,000	- 40.0	+ 90.0	155,000	33,910	- 39.8	+ 158

* BASED ON PRELIMINARY C_m VS α DATA - REVISED VALUE IS +15 (SAME AS CASE 2)

NOTES: CASE NO. 4 IS A γ_{MIN} RE-ENTRY

CASE NO. 5 IS A MULTIPLE SKIP RE-ENTRY

3.2.10 UPDATED SAFETY ANALYSIS REPORT

Deletion of the USAR as a formal document was the decision made by the INSRP in a meeting of the panel coordinators held in May. Instead, a vu-graph presentation was scheduled to be given to the panel around mid-November in a meeting to be held at Kirtland AFB, New Mexico. The meeting was later rescheduled for the end of January, 1974 to allow time to analyze the effects of the TRW re-entry breakup study results on the re-entry capability of the RTG/HSA.

3.3 OPERATIONAL SAFETY

The operational safety task includes the training procedures, facilities and equipment required for the safe handling of MHW Isotope Heat Source (IHS's) and RTG's.

3.3.1 RADIOLOGICAL PROTECTION PROGRAM

The MHW Radiological Protection Program, GE Document GEMS-409, was prepared to document the program that would be utilized at GE/VFSC to minimize radiation exposure while handling IHS's and RTG's. The contents of this document were reviewed in the MHW Radiological Safety Training Courses.

The initial submittal of GEMS-409 for review and approval was on May 14, 1973. Revision A, distributed on August 13, 1973, incorporated comments suggested in AEC/SNS letter (G. L. Kimball to A. A. Gayuski) dated July 3, 1973.

3.3.2 RADIOLOGICAL SAFETY TRAINING

Radiological Safety Training classes were conducted on July 9 and 23, August 6, and October 16 and 30, 1973 for a total of 85 test and engineering personnel. Each attendee received a copy of the MHW Radiological Protection Program, GEMS-409, and was instructed in the contents of this document.

In addition to radiological safety classes, dry runs (walk throughs) were performed with a heat source and converter mockup. During the dry runs, the test procedures were checked out and evaluated to assure that radiation exposures are minimized during the operations. Time and distance studies were conducted during the dry runs to obtain estimates of personnel radiation exposure. The dry runs provided a good opportunity for a shakedown of handling equipment and facilities.

AEC/SNS personnel witnessed the RTG walkthrough demonstration in the RTG Assembly and Storage Facility (Building 800) on December 17-20, 1973. The walkthrough consisted of removal of the Storage Protection Container (SPC) from a Heat Source Shipping Container, loading of the SPC into the Loading and Assembly Station (LAS), removal of the IHS from the SPC, installation of an emissivity sleeve on the IHS, loading of the IHS into the converter, and removal of the RTG from the LAS. The walkthrough also demonstrated the unloading of an IHS from the converter and back through the procedures to the loading of the SPC into the Heat Source Shipping Container. A second walkthrough for the AEC will be conducted in January 1974.

3.3.3 SAFETY ANALYSIS REPORT OF MHW ISOTOPE HEAT SOURCE OPERATIONS AT GE-VALLEY FORGE SPACE CENTER

A Safety Analysis Report was prepared for the MHW Isotope Heat Source Operations at GE Valley Forge Space Center as requested by the AEC MHW Contract Administration Office

(letter to A. A. Gayuski from G. L. Kimball dated May 1, 1973). GE Document GEMS-412, dated June 25, 1973 was submitted for review and approval. Revision A of this document will be issued early in January 1974 to reflect the suggested revisions discussed at the AEC/GE/MRC meeting held July 31, 1973 - August 2, 1973 at Mound Laboratory, and recent modifications in IHS design and qualification and acceptance test plans and procedures.

The Safety Analysis Report includes a description of the IHS, facilities, equipment and procedures that will be utilized for the safe performance of the test program at GE. The entire handling and test sequence involving the IHS has been reviewed for the identification of possible credible accidents. These credible accidents have been evaluated to determine the IHS response and possible consequences resulting from the accident environments. Safety plans and safeguards for the prevention of accidents and for the minimization of consequences in the event that an accident occurs are discussed.

Appendix A of the document describes the MHW IHS containment capability which is based on multi-layers of barriers restricting release of plutonium under normal operations and potential accident situations.

3.3.4 SAFETY ANALYSIS REPORT FOR PACKAGING (SARP)

Safety Analysis Report for Packaging (SARP) for MHW-RTG Shipping Container, GE Document GEMS-408, was submitted for approval on April 24, 1973. The RTG Shipping Container has been designed to meet the packaging standards contained in AEC Manual Chapter 0529, "Safety Standards for the Packaging of Radioactive and Fissile Materials", and conforms with AEC immediate action directive IAD No. 5201-3, "Review and Evaluation Procedures for Design Approvals for Radioactive Material Packages." GEMS-408 is being revised to incorporate AEC comments.

3.3.5 TEST PROCEDURES

The following test procedures were reviewed and signed off for radiological safety prior to submittal to AEC/ALO Operational Safety for review and approval:

<u>Procedure</u>	<u>Number</u>
IHS Receiving, Storage, and Shipping Procedure	SI249234
IHS Acceptance Test Procedure	SI249237
RTG Qualification Test Procedure	SI249240
RTG Acceptance Test Procedure	SI249239
RTG Interplant Transfer, Storage, Shipping, and Receiving Procedure	SI249238
RTG/ETG Mass Properties Measurement Procedure	SI249631
RTG Assembly and Disassembly Procedure	SI249235
Portable GSC Operating Instructions	SI249229
RTG Shipping Container Instructions	SI249246
IHS Shipping Container Instructions	1-14898

3.3.6 IHS TEST FACILITIES

While at GE/VFSC the IHS's will be handled and tested in the following three facilities:

Building 800 - RTG Assembly and Storage

Building 400 - RTG Shock and Vibration

Building 300 - RTG Thermal Vacuum

3.3.6.1 Building 800

Building construction to provide additional space for the MHW program was completed. Two Loading and Assembly Stations were installed and acceptance tested. The following functions will be performed in a safe manner in Building 800:

1. Receipt of IHS
2. Acceptance testing of IHS
3. Loading IHS into converter for RTG assembly
4. RTG air performance
5. RTG weight and center of gravity

6. IHS and RTG storage

7. Qualification IHS shipment to Mound

Figure 3-24 indicates the floor plan for the RTG Storage and Assembly Building. Figure 3-25 shows the two Loading and Assembly Stations with the Gas Management System cabinets in front of the left Loading and Assembly Station.

3.3.6.2 Building 400

Construction of a 16-by-30 foot test cell addition to Building 400 for MHW-RTG shock and vibration testing was completed. The doors, roof, and penetrations are sealed and a ventilation system with a HEPA filter provides a negative pressure (~ 0.1 inch water) to contain plutonium contamination inside the facility in the event of a release of plutonium.

Figure 3-26 indicates the floor plan for the facility. Figure 3-27 shows the vibration equipment (C-150 vibration exciter, Team table and shaker isolation system) which have been installed and acceptance tested in the MHW-RTG facility. Figure 3-28 indicates the facility access doors for RTG shipping containers and other test equipment.

3.3.6.3 Building 300

The RTG thermal vacuum test facility floor plan is shown in Figure 3-29. The three vacuum chambers, Nos. 2, 3 and 4 can be used for thermal vacuum testing of qualification and flight RTG's. These chambers are shielded with 4 inches of Benelex walls and have HEPA filters installed in the exhaust lines. Figure 3-30 indicates one of the vacuum chambers with a Gas Servicing Cart connected to the chamber for gas sampling for radioactive contamination or argon gas management of the converter void volume.

3.3.7 OPERATIONAL SAFETY ANALYSIS

The Operational Analyses Task Group for the MHW/LES 8/9 met at Lincoln Laboratory on January 19, 1973 and at Cape Kennedy Air Force Station on February 7-8, 1973. Prior meetings of the Operational Analysis Task Group were held at Mound Laboratory and GE in October and November 1972. The operational analysis assessed the ground operations involving IHS's

Figure 3-24. RTG Storage and Assembly Building No. 800

Figure 3-25. Loading and Assembly Stations in Building 800

Figure 3-26. MHW-RTG Vibration Facility - Building 400

Figure 3-27. Dynamic Test Equipment in Building 400 - MHW-RTG Facility

primarily from the safety point of view, including all phases of transportation, handling, test, and storage from "factory-to-flight". The purpose of the analysis is to assure the preparedness and continuity of operations affecting nuclear safety, to evaluate plans and procedures, and to provide guidelines on potential problem areas.

Functional block flow diagrams were prepared to form the basis for the analysis. These diagrams indicate the sequence of events, various responsibilities, organization, facilities, operating temperatures and environments for each IHS handling operation.

The Operational Analysis Task Group met at Mound Laboratory on November 8-9, 1973 to investigate the implications of handling the MHW/LES 8/9 IHS without an iridium clad on the IHS. In order of preference the following conclusions were made:

1. It was recommended that the outer clad be retained from an overall program standpoint.
2. Elimination of the outer clad may be feasible provided the following disadvantages can be tolerated.
 - a. Increased safety risk (ground containment, re-entry, etc.)
 - b. Loss of performance due to helium in converter voids.
 - c. Added costs and schedule delays due to modifications to equipment tools and procedures.
 - d. Unknown aspects of converter contaminants, e.g., plutonium, graphite, oxygen, etc.
3. Selection of an alternate material to replace the iridium outer clad was not recommended for LES 8/9 due to unreasonable time factors.

Figure 3-28. MHW-RTG Shock and Vibration Test Cell

Figure 3-29. Thermal-Vacuum Facility - Building 300

Figure 3-30. Thermal-Vacuum Chamber for MHW-RTG Testing

BLANK

SECTION 4
RTG SYSTEMS

4.1 SYSTEM INTERFACE, ENVIRONMENTAL AND PERFORMANCE REQUIREMENTS

During the reporting period activity was concentrated in the areas of requirements documentation and the development of thermal and dynamics requirements and data through analyses and test.

4.1.1 REQUIREMENTS DOCUMENTATION AND DATA

1. The Product Specification for Prime ETG's and RTG's was completed.
2. The latest revisions for both the Environmental and Systems Specifications, incorporating SNS comments, were forwarded to SNS.
3. The IHS Specification, Revision A, which incorporates SNS comments, was forwarded to SNS for review and approval.
4. A study was completed to evaluate methods and equipment required for RCA to adapt their plant wiring to meet the EHS power supply console electric power requirements.
5. The following specifications were reviewed with SNS and were updated:
 - a. CP47A14609 - SPC Handling Tool Product Specification
 - b. CP47A14614 - RTG, ETG, E1 and E2 Product Specification
 - c. CP47A14615 - Storage Protection Container Product Specification
 - d. CP47A14604 - RTG Shipping Container Product Specification
 - e. NS 0010-13-07 - Electrical Receptacle Product Specification
 - f. NS 0080-07-04 - IHS Acceptance Test Plan
6. As a result of the above revisions, the following documents were also updated:
 - a. NS 0010-13-03 - TBC #3 Product Specification
 - b. SS 47A14603 - Systems Specification

c. NS 0020-05-02 - Environmental Criteria and Test Requirements Specification

d. CP47A14611 - TBC #4 Product Specification

7. Information related to the GE fill and purge valve, vent tube geometry and material was forwarded to Lincoln Laboratories.
8. A program was assembled which defines the testing and development required to bring the JPL RTG internal instrumentation up to flight status.
9. The TBC-3 transient power data were submitted to LL. This test verified that the MHW-RTG exceeds the systems requirement for BOM power (130 watts) and T + 48 hours power (145 watts).
10. JPL was contacted in regard to micrometeoroid requirements in the MJS specification; these levels appeared to be 10 to 100 times higher than current LES 8/9 values. JPL indicated that the micrometeoroid requirements in their specification are correct; this means a factor of 10 increase in intensity over the LES 8/9 environments.

4.1.2 THERMAL ANALYSES AND TEST

1. Thermal conductance across the RTG self-aligning bearing was determined to be seven watts per bearing on the launch pad and about 11 watts per bearing in orbit.
2. An evaluation comparing argon versus xenon as the RTG fill gas for the LES 8/9 mission was completed.
3. Venting of MHW-RTG at (t + 280) seconds into the LES 8/9 mission was analyzed, and no detrimental effects on either the converter or IHS assemblies were identified.
4. The results of the 26-node thermal model and revised 66-node thermal model were presented at the P74-1 Mission Thermal Interface Meeting on October 23 at LL.
5. A preliminary assessment of the thermal stresses in the RTG cases, resulting from the temperature distributions provided by TRW on October 9, 1973, was completed, based on load factors of 5 g's lateral and 8 g's longitudinal.
6. The distribution of heat input for the 26-node model was reviewed per Lincoln Lab's request. This model has no heat input to the domes. It was concluded that the distribution should not be changed and that the model should be considered conservative.
7. TRW's analysis of the RTG, using a 66-node model, was received and reviewed. Several comparative runs were made to determine sensitivity to various important parameters.

8. The 26-node model of the generator outer case was completed. An error was uncovered in the thermal mass of the thermopile nodes in the 66-node model. This error was corrected and a revised 66-node model with some other refinements was developed.
9. The TRW launch analysis was duplicated using the 26-node thermal model and the revised 66-node model. Using the TRW assumptions, there was good agreement between the GE and TRW results.
10. A preliminary assessment of the consequences of increasing the on-pad and launch temperature of the generator case from 385°F to 520°F in response to direction from AEC was completed.
11. Previous uncertainty regarding on-pad generator outer case temperature and gradient has been resolved. TRW has rerun the thermal analysis of the payload system using the latest 26-node RTG thermal models and the results of MMC's latest air conditioning development testing. The predicted RTG temperatures based on test data for ground conditioning are:

- Maximum Case Temperature 364°F
- Maximum Circumferential Gradient < 100°F
- Maximum Axial Gradient < 160°F

The predicted RTG temperatures based on design requirements for Ground Conditioning are:

- Maximum Case Temperature 386°F
- Maximum Circumferential Gradient < 100°F
- Maximum Axial Gradient < 160°F

4.1.3 DYNAMIC ANALYSES AND TEST

1. The dynamic simulated models fabricated by GE/Evendale for Lincoln Labs were dynamically tested. The test consisted of 0.5 g sine survey, 7.5 g (rms) random equalization and 15.1 g (rms) random vibration for 20 seconds. The configurations were, 1) a single unit mounted to the clevis and shaker, 2) tandem mounted units mounted to the LES 9 truss.
2. Qualification level vibration testing of the stacked RTG Dynamic Simulator Models mounted on the LL (LES 9) truss was completed. The stacked RTG models, without

the truss, also completed vibration testing to assess the effect of the truss on response. The results indicate that the stacking adapters are relatively stiff and the low frequency is due primarily to the LL truss. A formal report is in preparation.

3. The dynamically-simulated RTG models, mounted on the LES 9 spacecraft model, were subjected to acoustic and vibration tests at TRW. Review of responses indicated no excessive vibration of RTG's.
4. The load factors at the centroids of each RTG in the LES 8 and 9 configuration were reevaluated based on the linear and angular accelerations submitted by TRW. A negative margin of 0.20 at the stacking adapter exists.
5. Approval was received from Lincoln Labs to revise the shock spectrum and the acceptance vibration level, to minimize IHS fuel fines.
6. Lincoln Laboratories calculated the ball joint/truss interface loads for the short truss, for two lower ring boundary conditions, per GE request and the results were sent to GE for use in the beryllium case loads analysis.
7. The RTG shock and acceptance vibration spectra have been revised and negotiated with LL for acceptance. The shock spectrum was lowered and the acceptance vibration spectrum will be notched in the region of the IHS Fundamental Frequency $\cong 160$ Hz.

4.2 SYSTEM ANALYSES

4.2.1 ORBITAL THERMAL ANALYSIS

The LES 8/9 spacecraft designs result in significant blockage to the RTG waste heat rejection. RTG case gradients and hot shoe temperature levels are affected by this blockage. Lincoln Laboratories conducted an orbital analysis using a thermal model of the entire spacecraft which approximated each RTG as a single node. To determine the RTG circumferential temperature gradients and internal temperatures, a 36-node RTG model was used with boundary conditions derived from the spacecraft temperatures and view factors of the Lincoln analysis.

The orbital analysis was approximated by determining the steady state temperatures at the four orbital positions shown in Figure 4-1. Position 4 is a condition of no solar heating to the spacecraft. Position 2 has solar heating on the top of the spacecraft and forward dome of the top RTG. Positions 1 and 3 have maximum solar heating on the +X and -X sides of the spacecraft respectively. Although the RTG temperatures calculated at each position are steady

state, the boundary temperatures from the Lincoln analysis were determined for transient conditions.

Table 4-1 lists the maximum hot shoe node temperatures for the four RTGs at the four orbital positions. Since the model does not account for axial gradients, these are not the maximum individual hot shoe temperatures but are the maximum circumferential row average hot shoe temperatures. The arithmetic average orbital temperatures are also shown. Since these results are steady state, the actual orbital average temperature may be slightly less than the arithmetic average due to transient effects, especially for LES 8 where position 3 temperatures are significantly higher than at the other three positions.

Figure 4-1. LES 8/9 Orbital Positions for Thermal Analysis

Table 4-1. LES 8/9 Orbit Hot Shoe Temperatures

Maximum Circumferential Row Average Temp.				
Position	LES 8		LES 9	
	Top RTG	Lower RTG	Top RTG	Lower RTG
1	1037°C	1042°C	1040°C	1044°C
2	1039°C	1041°C	1029°C	1031°C
3	1048°C	1057°C	1040°C	1044°C
4	1035°C	1040°C	1026°C	1030°C
Arith. Avg.	1040°C	1045°C	1034°C	1037°C
Temp. includes effects of: 1 - Spacecraft blockage 2 - Solar heating (sides or dome) 3 - RTG stacking				

The results show that the LES 9 hot shoe temperatures are near the 1035°C design value, but the LES 8 RTGs are 5°C and 10°C higher due to greater spacecraft blockage. Figures 4-2 thru 4-5 show the case and hot shoe circumferential temperature distributions for 4 RTGs. The circumferential distributions are all symmetrical about the X spacecraft axis. The hot shoe temperatures show little variation because the heat source distributes the heat fairly uniformly. The case gradients are caused more by the solar heating than by spacecraft blockage.

4.2.2 LES 8 & 9/MHW THERMAL INTEGRATION ANALYSES

Lincoln Laboratories asked GE to perform several tasks related to RTG/Spacecraft thermal integration. The specific questions were:

1. Can you suggest some approaches to the cooling of the four RTG's on pad under the fairing? Give an estimate of flow rate and air velocities keeping in mind that there exists a practical limit of 125 lb/min currently supplied through the umbilical and payload fairing duct. Are any problems anticipated with downward forced convective cooling? What air temperature do you anticipate around the spacecraft?
2. What is the effect on the RTG's of free convective cooling inside a fairing? What is the effect of the fairing emissivity?
3. What is the thermal resistance through the monoball connectors? Under load? In orbit?
4. Please provide a coarse, transient, 3D model of the RTG's for the integrating contractor.
5. What and where will be the hot spot in the tandem configuration on pad?
6. What will be the time constant for the RTG's after liftoff in a fairing which is being aerodynamically heated?
7. What will be the axial temperature distribution on the RTG outer surface?

4.2.2.1 Question 1

This question addresses itself to the problem of how, if required, might GE cool the MHW RTG's in the on-pad, LES 8&9 configuration. The infeasibility of cooling the RTG's by dumping air in the top of the shroud and venting it at the transtage interface was previously

Figure 4-2. Orbital Temperatures - Lower LES 8 RTG Steady State Temperatures at Each Position

Figure 4-3. Orbital Temperatures for Top LES 8 RTG

Figure 4-4. Orbital Temperatures for Lower LES 9 RTG

Figure 4-5. Orbital Temperatures for Top LES 9 RTG

shown.* More than 2500 lb/min of cooling air are required to obtain an average RTG temperature of 250⁰F; at present only 125 lb/min are available, but the flow rate could be increased to 250 lb/min by enlarging the flow ducts.

In order to further reduce the RTG temperatures, direct cooling of the RTG's will be required. The most obvious and direct approach would be to channel the cooling flow around each RTG by employing a duct. To obtain an average RTG radiator temperature of 350⁰F, an air velocity of 8.0 ft/sec is required. Assuming a duct diameter of 16 inches, an air flow rate of 28 lb/min/RTG or a total flow rate of 112 lb/min for four RTG's would be required. The problem with this duct concept or any other channelling device is that it must be removed prior to the operational phase of the mission, since it would obscure the RTG's view to space and hence, hinder the RTG's capability to reject its waste heat.

Removal of any channelling device is complicated by the proximity and vulnerability of the spacecraft. Consequently, any pad cooling concept not only has to meet any thermal and hydrodynamic constraints but also any geometry constraints; this entails integrating the efforts of the S/C contractor, the RTG contractor and the payload fairing contractor. One approach that appears feasible at this time is wall jet cooling of the RTGs.

Cooling a pair of tandem-mounted RTG's utilizing the axial wall jet concept was investigated. Several questions would have to be answered before this concept could be employed. These being:

1. Does the axial temperature distribution along the two or three RTG's become prohibitive?
2. What is the effect of the surface discontinuity between RTG's on the wall jet flow?
3. What is the effect of the high air velocities on the RTG and on the spacecraft?

* Kelly, C. E., "TBC-3 Cooling Requirements", GE PIR 4124, 8/22/72.

The first question can probably be answered through further analysis, but the other two questions will have to be answered through testing. The cooling requirements for a pair of tandem-mounted RTG's can be calculated to give a feel for the cooling requirements. An average surface temperature of 275°F and a nozzle width of one inch was assumed, with an overall plate length of 45 inches. (A five-inch gap was assumed to exist between the RTG's.)

The required velocity is 50 fps and the volumetric flow rate is 830 cfm. The cooling requirements are not significantly greater than those for a single RTG.

Figure 4-6 shows a schematic of a possible cooling concept. The nozzles and plenum are located on top of the tandem-mounted RTG's which permit easy removal during shroud separation. This is an important requirement for the LES 8&9 mission. Another concept for other missions would be to place the plenum between the two RTG's and have the nozzles pointed in the opposite directions across each RTG.

A summary of the advantages, disadvantages, and areas of further investigation are shown in Table 4-2 for the axial wall jet concept.

The axial jet directs the flow parallel to the surface to be cooled. In contrast to this, the jet can be directed perpendicular to the surface to be cooled; here the flow will spread out radially from the point of impingement. This type of flow is known as a radial wall jet. Figure 4-7 shows a comparison of two wall jet concepts; also shown is a variation of the radial jet - the slot jet.

Spacing of the jets over the heat transfer surface is critical in order to obtain a uniform heat transfer coefficient and to minimize interference among jets. This can be accomplished with a S/D ratio greater than 20 and a Y/D ratio greater than 8, where

D is the jet diameter.

Y distance between centerline of jets

S perpendicular distance from the heat transfer surface to the jet opening.

PLENUM AND
FLOW NOZZLES

INLET AIR

LAUNCH
DIRECTION

Figure 4-6. Pad Cooling Concept

Table 4-2. Axial Wall Jet Cooling

<u>Advantages</u>	<u>Disadvantages</u>
<ul style="list-style-type: none"> • Low flow rate, < 60 lbs/min/pair of RTG's • Adaptable to easy removal of ducting at time of shroud separation • Concept has no effect on operational RTG temperatures • No additional weight penalty on RTG system • Small circumferential temperature gradients. 	<ul style="list-style-type: none"> • Pressure requirements greater than that required for ducted flow • Exit air velocities are high; approximately 40 fps • Possible temperature gradients on converter outer case in axial direction
<p><u>Areas for Further Investigation</u></p> <ul style="list-style-type: none"> • Determine axial temperature gradient along tandem-mounted pair of RTG's. • Determine effect of surface discontinuity between tandem-mounted pair of RTG's on wall jet cooling approach. • Determine effect of high exit air velocities on the spacecraft.

Figure 4-7. Two Wall Jet Concepts

The procedure for jet sizing and spacing must be to first determine an acceptable temperature rise. The air flow rate can then be established. If a 50^oF temperature rise is acceptable, a 160 cfm flow rate per RTG is required. This can be achieved if there are 12 tubes and 20, 1/4 inch diameter holes/tubes; the critical design ratios are:

$$Y/D = 4$$

$$S/D = 4$$

This spacing will produce a non-uniform film coefficient over the surface and there will be significant interference between jets. In view of these facts, a slot jet would probably be more suitable for the MHW application.

The slot jet variation to the impingement cooling concept is now pursued. Assuming that two headers are placed between each fin on the MHW (for a total of 12 tubes/RTG) and assuming an allowable increase in the air flow temperature of 50^oF, the velocity requirements are:

<u>Slot Width Inches</u>	<u>Velocity Requirement</u>
1/16	51 fps
3/32	34 fps
1/8	26 fps

There are several disadvantages with the radial jet impingement cooling concept. The primary disadvantage is that the headers between fins restrict the RTG's capability to reject its waste heat. To determine the effect that these tubes will have on RTG temperature is difficult due to the complexity of the geometry in determining the radiation interchange factors.

Other advantages and disadvantages are given in Table 4-3 along with areas in which further investigation is required.

Table 4-3. Radial Impingement Cooling

<u>Advantages</u>	<u>Disadvantages</u>
<ul style="list-style-type: none"> • Low flow rates, $< 30 \frac{\text{lbs}}{\text{min}}/\text{RTG}$ • Small temperature gradients, $< 20^{\circ}\text{F}$ • Adaptable to easy removal of ducting at time of shroud separation. 	<ul style="list-style-type: none"> • Slightly increased operational cold side temperature • Added weight to RTG system. • Pressure requirements greater than required for ducted flow.
<u>Areas for Further Investigation</u>	
<ul style="list-style-type: none"> • Determine uniformity of discharge flows as a function of header axial length. • Perform structural analysis to determine acceptable circumferential temperature gradients on MHW converter outer case. • Determine the effect that the header tubes has on average RTG temperature. 	

4.2.2.2 Question 2

An analysis was performed to determine the steady state temperatures of four MHW-RTG's in a free convection environment inside the TITAN III-C universal payload fairing. This was not a detailed thermal analysis, but was meant only to give a feel for the on-pad temperature problem. The analysis assumed that the 4 RTG's were canned in a 10-foot diameter cylinder, six feet high. The RTG waste heat was transferred to the pay-load fairing by a combination of radiation and free convection.

The free convection contribution was assumed to be a "cellular" type of flow; a film coefficient was calculated from the RTG to the bulk fluid and from the bulk fluid to the payload fairing. These conductances were assumed to be in series.

The results of the analysis are summarized in Table 4-4 which shows that:

- The internal shroud film coefficient does not significantly affect RTG surface temperature. A more significant variation in the bulk air temperature is seen, however.
- Solar heating has an insignificant effect on temperatures.
- Improving the shroud emissivity would reduce RTG temperature only 30°F.

Table 4-4. Effect of Shroud on Free Convective Cooling of the MHW Generator

	RTG Surface	Bulk Air	Shroud	Environment
1. Single RTG No Shroud	410°F	--	--	100°F
2. 4 RTG's Inside Shroud • Shroud $\epsilon = 0.1$ • $\bar{h}_c \text{ shroud} = 1/2 \bar{h}_{c \text{ ref}}^*$	494	288	188	100
3. Same as 2, but $\bar{h}_c \text{ shroud} = 1/3 \bar{h}_{c \text{ ref}}$	502	319	188	100
4. Same as 2, but $\bar{h}_c \text{ shroud} = \bar{h}_{c \text{ ref}}$	483	247	188	100
5. Same as 2, but $\epsilon_{\text{shroud}} = 0.85$	459	277	188	100
6. Same as 2, but no solar heating	492	284	183	100

* $\bar{h}_{c \text{ ref}} = 0.86$

4.2.2.3 Question 3

One of the problems involved in spacecraft integration is determining the amount of heat transferred from the RTG's to the surface to which they are mounted. The modes of heat transfer are direct radiation and conduction through the mounting assembly. The thermal conductance of the self-aligning bearing and clevis interfaces is an important parameter in the integration analysis.

In order to determine the thermal conductance of the mounting configuration, two of the mounts were instrumented for the TBC-3 vibration tests. During these tests, the converter was setting vertically in air with no active cooling. The temperature profiles of the mounts were similar. The arrangement of the four thermocouples on the clevis enables the heat flow to be calculated. The temperatures of the race of the self-aligning bearing and of the aluminum baseplate were also measured.

Using the temperature profile of clevis #1, the heat flow per mount was calculated to be 7.43 watts. Net convection and radiation transfer was less than 4% of the total heat flow and was neglected. Assuming the total 7.43 watts to be transferred through the mount, the overall case to baseplate conductance was 0.0364 w/^oF. The case to clevis conductance was 0.0594 w/^oF. The 290^oF case temperature was measured at the base of the support ring. In orbit, this case temperature will be in the order of 400^oF. If the surface to which the RTG's are mounted is of the order of 100^oF, the heat flow will be 0.0364 (400-100) = 11 watts per mount during orbital operation.

4.2.2.4 Question 4

GE was asked to provide a three-dimensional transient thermal model of the MHW-RTG for use by the integrating contractor for system thermal analyses. A comprehensive model to determine gradients and transient characteristics requires a substantial number of nodes. Initially, a 98-node model was prepared but was reduced to a 66-node model at the request of Lincoln Laboratories. This 66-node model, shown in Figure 4-8, is the minimum size for general use. A 26-node model of the outer case and domes was prepared, again at Lincoln's request, for use in on-pad cooling steady state analyses. This model is shown in Figure 4-9. It was demonstrated that this model could also be used for the transient launch analysis in the time period from lift-off to payload fairing separation.

4.2.2.5 Question 5

An analytical study was performed to determine where on a tandem-mounted pair of RTG's the hot spot occurs during the on-pad phase of the mission. It was felt that the hot spot might occur on the beryllium dome sections between the two RTG's as a result of the restricted view factor and heat transfer film coefficient.

Figure 4-8. MHW 66-Node Thermal Model

Figure 4-9. MHW 26-Node Thermal Model

The view factors from the beryllium dome to the sink were reduced to take into account the restricted view to space when the RTG's are tandemly mounted. The effect on the free convection film coefficient as a result of the tandem mounting configuration is impossible to empirically predict. The film coefficient will be reduced compared to the undisturbed boundary layer value, but how much? Consequently, three different values of the film coefficient were used in the analysis, 1.25, 0.5 and 0.0 Btu/hr.ft².

The conclusion is: the peak RTG surface temperature will occur on the RTG lateral surface, not on the dome. When the heat rejection capability of the dome is unrestricted, it acts as a fin for the outer case, i.e., the dome rejects more than just the end shunt losses of the RTG. As the heat rejection capability of the dome is reduced, the dome becomes less and less of a fin for the case.

It is reasonable, then, that the dome temperature will not exceed the peak RTG lateral surface temperature.

4.2.2.6 Question 6

Two approaches were investigated in an attempt to define a time constant for the MHW-RTG. One is the classic solution for a solid, homogeneous mass with an infinite conductivity rejecting or accepting energy via convection. The solution is:

$$\frac{T - T_{\infty}}{T_e - T_{\infty}} = e^{\frac{\bar{h}A}{CW}} \Theta$$

where,

- T_e is the steady state temperature
- T_o is the initial temperature
- T_{∞} is the sink temperature
- T is the temperature at time Θ
- CW is the thermal capacitance of the solid
- A is the heat transfer area
- $\bar{h}$ is the h.t. coefficient.

In this case the time constant is defined as:

$$\tau = \frac{\bar{h}A}{CW}$$

and the radiation heat transfer coefficients are:

$$\bar{h} = \frac{\sigma \tau (T^4 - T_{\infty}^4)}{(T - T_{\infty})}$$

The time constant would be changing, then, during the transient response of the solid.

Figure 4-10 compares this solution with the TBC-1 transient test. The time constant was evaluated at two different temperatures - at the initial temperature and at the equilibrium temperature. As can be seen, using the equilibrium temperature in the time constant provides a better solution.

Figure 4-10. TBC-1 Transient Test

An RTG is far from a solid, homogeneous mass with an infinite conductivity. Consequently, in these and any other calculations to be made, only the thermal capacitance of the converter was used in the time constant calculation. The solution, therefore, is not expected to match exactly, especially at the longer time periods.

The other approach handles the radiation properly, but is a more complicated solution. This transient solution is:

$$\Theta / \tau + C = 2 \left[\tanh^{-1} (T/T_e) + \tan^{-1} (T/T_e) \right]$$

Where C is the constant of integration and τ is the time constant =

$$\tau = \frac{WC}{4\epsilon A \sigma T_e^3}$$

Expanding the terms in brackets and defining $\mu = T/T_e$

$$\Theta / \tau + C = 4 \left[\mu + \frac{\mu^5}{5} + \frac{\mu^9}{9} + \frac{\mu^{13}}{13} + \dots \right]$$

A time constant of .311 hrs was calculated for the TBC-1 test conditions. The resulting transient response is shown in Figure 4-11; the match is excellent during the first half hour.

If the transient response of the RTG is designed for a duration less than 1/2 hour, the infinite series, rather than the exponential, time constant is recommended.

4.2.2.7 Question 7

The axial temperatures distribution for the MHW-RTG are shown in Figures 4-12 and 4-13 for air and vacuum environments respectively. These distributions were test data for TBC-3. The significant temperatures for integration considerations are the beryllium shell and fin temperature distribution.

Figure 4-11. TBC-1 Transient Test

Figure 4-12. TBC-3 Axial Temperature Profiles in Air

Figure 4-13. TBC-3 Axial Temperature Profiles in Vacuum

The in-vacuum outer case temperature distributions are 20° to 30° F lower than predicted. This is attributable to the conservatism that was employed in the radiation factors used in the analysis.

The case temperatures shown in Figure 4-12 will be low for the on-pad configuration where there are 4 RTG's inside the payload fairing. This is due to:

1. The effective sink temperature inside the shroud will not be 70° F; the shroud temperature on-pad could run as high as 200° F, for a non-cooled on-pad condition.
2. The free convective film coefficient during TBC-3 testing was enhanced by some forced convection as is evidenced by the 1.7 Btu/hr-ft^2 $^{\circ}$ F average film coefficient.

4.3 SYSTEMS INTEGRATION

During the reporting period planning and design efforts were devoted to the areas of thermal, mechanical, electrical and procedural integration.

4.3.1 THERMAL INTEGRATION

1. A meeting was held at Cape Kennedy to discuss thermal integration of the RTG's with the LES 8 and 9 satellites and the launch vehicle. Attendees included US Air Force (SAMSO), ETR engineering personnel, Martin-Marietta, Aerospace Corp. and TRW. GE presented the results of a study to evaluate the RTG thermal integration with the payload, as well as a number of possible solutions to the pad cooling problem. SAMSO and ETR were assigned an action item to determine the auto-ignition temperature of the hypergolic fuel. The question of whether or not RTG pad cooling is required is contingent upon the auto-ignition temperature of the hydrazine fuel and cooling system flow direction and configuration.
2. A revised thermal model for LES 8/9 MHW-Thermal Integration was transmitted to Lincoln Laboratories. Discussions were held at LL concerning thermal blockage of the RTG's caused by the satellites. One of the two satellites causes considerable blockage and a detailed orbital analysis is required to define the converter hot junction temperature profile.
3. The P74-1 Definition Phase Design Review was held at TRW on September 19. Principal results were that TRW was designing air conditioning for on-pad cooling to provide 547° F maximum on the RTG outer shell, and that the Stage II shut down loads are quite severe; MMC and TRW are reevaluating these loads.
4. The P74-1 thermal control meeting was held at TRW on September 20. TRW performed a reevaluation of RTG with more complete models and realistic boundary conditions.
5. A thermal integration meeting was held at LL on October 18. Some pertinent results of the meetings were:
 - a. USAF/LL have a feeling of confidence about the GE approach to the shunt resistance problem.
 - b. LL has requested that GE define the lead time necessary to fix up TBC-3A for delivery to LL as E-1.
 - c. LL is agreeable to notching the RTG acceptance vibration level at the fundamental frequency of the IHS ($\cong$ 160 Hz).
 - d. LL will entertain a request from GE for the lowering of the present shock spectrum below the present 200 Hz.
6. Revised thermal models (66 and 26 node) were transmitted to LL to aid in the TRW System Thermal Analyses.

7. The design of the LL thermal simulator models was initiated. These models will be used to verify the on-pad temperature profile using the MMC on-pad air conditioning system.

4.3.2 MECHANICAL INTEGRATION

1. A Lincoln Laboratories truss was successfully mated to TBC-3. The total operation lasted approximately 10-15 minutes, but it would appear this could probably be reduced to a maximum of five minutes after installation personnel gain experience.
2. Lincoln Laboratories will fund GE to design and fabricate two RTG mockups for use in their system structural dynamic test. A meeting was held with GE-ESP-Evendale to review the design, analysis and fabrication requirements for these mockups.
3. The RTG Loading Interfaces were reviewed and the following items resolved:
 - a. Method of installation of the graphite emissivity sleeve over the Isotope Heat Source.
 - b. Method of attaching the loading fixture to the heat source and installing it in the converter.
 - c. Method for installation of end enclosures and application of preload.

4.3.3 ELECTRICAL INTEGRATION

1. A design concept for the RTG cable shorting plug was completed. The shorting plug will be used to maintain the RTG under short-circuited conditions during transit and storage.
2. Lincoln Laboratories agreed to accept one shield around the power and instrument cables rather than individually shielding each twisted pair of wires. This revision allows for a lighter weight and more flexible cable.
3. A wiring diagram defining all the cables and connectors required at Lincoln Labs for the operation of ETG's and RTG's was completed and forwarded to Lincoln Labs.
4. Cable drawings were completed depicting stacked ETG's/RTG's in the LL T/V Chamber. Drawings show power, instrumentation, console and test (air) cables. These drawings were sent to LL for their approval.

4.3.4 PROCEDURAL INTEGRATION

1. RTG and LES 8/9 Operational Procedures were reviewed with Lincoln Labs for those RTG operations to be performed at Lincoln and the launch site. Significant highlights of this meeting were:
 - a. RTG's will always be loaded onto the satellite in the vertical position.
 - b. If the RTG's are evacuated at Lincoln during thermal vacuum testing, pumpdown will be accomplished by a turbomolecular pump.
 - c. Some type of nuclear radiation shielding will be provided for the RTG integration operations.
 - d. A gas management service cart will be required both at Lincoln Labs and at the launch site.
 - e. The IHS will not be vented (if possible) during thermal vacuum operations at Lincoln Labs.
2. An Operational Safety Review meeting was held at Lincoln Laboratories. The safety review panel was pleased with Lincoln's presentation. Lincoln is considering mating the F-3 and F-4, as well as F-1 and F-2, RTG's with the LES 9 satellite if the schedule permits. This represents a possible program change.
3. An interface meeting was held at Lincoln Laboratories on March 9. The highlights are summarized below:
 - a. The interface control drawing was reviewed.
 - b. Lincoln will size their truss for the installation of an RTG at an average outer case temperature of 356°F.
 - c. Lincoln will supply all the connectors that will mate with the satellite.
 - d. GE will send a representative to the Lincoln structural tests to be performed at TRW.
 - e. All RTG and ETG cable lengths and configurations were agreed upon.
 - f. Lincoln's on-pad power requirements will be 70 watts per RTG.
4. A flow chart, of RTG integration at Lincoln Labs, was received from LL. The chart depicts operations that LL wants GE to perform, or assist in performing.

-
5. A letter was sent to LL to initiate resolution of GE support at LL and CKAFS.
 6. Procedures and operating manuals were transmitted to LL to aid them in assessing the documentation required for MHW-RTG integration at LL and at the Cape.

BLANK

SECTION 5

HEAT SOURCE

5.1 HEAT SOURCE DESIGN REQUIREMENTS

The MHW Heat Source Assembly (HSA) must:

1. Provide the thermal input of 2400 watts to the thermoelectric converter.
2. Create no hazard to man, either directly or indirectly, under all expected operational and abort conditions.

The details of the design requirements are set forth in GE-ISA0 Specification SS47A14603. The heat source is referred to as the Isotope Heat Source (IHS).

The functional requirements imposed by the mission on the MHW-RTG power system set certain interfaces with which the IHS must be integrated, and establish conditions under which the IHS must provide thermal energy to the converter. These requirements influence the design of the IHS, but are generally less stringent than the requirements for safe behavior under all predicted operational and credible accident conditions. These requirements categories are tabulated in Table 5.1-1 and are briefly noted in the succeeding paragraphs.

Table 5.1-1. MHW-HSA Design Specifications

1. <u>Operational</u>		2. <u>Safety</u>
Vibration	Thermal Power	Criticality
Venting	Temperatures	Neutron Emission
Pad Access	Mounting Configuration	Subcritical Multiplication
Operating		
Life	Envelope	Venting Frequency
Storage Life	Weight	Emergencies
Fuel	Center of Gravity	Reentry
Mounting	Venting Connections	Impact/Post-Impact
		Pad abort
		Hazard Reduction

5.1.1 OPERATIONAL REQUIREMENTS AND CONSTRAINTS

5.1.1.1 Isotopic Fuel

The fuel form shall be $^{238}\text{PuO}_2$ ceramic compact spheres as specified in GE Drawing, Fuel Sphere, No. 47B301663 and Mound Laboratories MHW Fuel Specification - 1-14813.

5.1.1.2 Thermal Inventory

The thermal loading of the IHS at the time of assembly shall be as required to satisfy the requirements of the mission, 2350 to 2410 watts.

5.1.1.3 Temperatures

Ground and Launch - The IHS while stored in the Converter, RTG Shipping Container or in the IHS Shipping Cask shall be capable of operating in a gas atmosphere of argon (1% He) or xenon (1% He) or Helium (100%) and its cylindrical surface temperature shall not exceed 1100°C .

Orbit - The IHS shall be capable of operating in the MHW Thermal Converter in a space vacuum $\cong 10^{-10}$ torr for a period of five years without exceeding an exterior cylindrical surface temperature of 1100°C based on an effective sink temperature of 1045°C and an effective radiation interchange factor of 0.46. Maximum temperature of any iridium surface shall not exceed 1500°C .

5.1.1.4 Operating Life

The maximum period of operation of the IHS shall be six (6) years including: (1) one year storage and test commencing with fuel encapsulation and ending at Beginning of Mission (BOM) and (2) five years orbital operation commencing at BOM and ending five years later.

5.1.1.5 Storage Life

The IHS shall perform satisfactorily after one year earth storage and test. Storage and test time is defined as commencing with fuel encapsulation by the fueling agency and ending at the BOM. Storage can be either in the IHS Shipping Container (GE Specification CR47A14613) or integrated as part of an RTG. During storage the IHS outer clad shall not exceed a temperature of 1100°C .

5.1.1.6 Envelope

The IHS design envelope shall be cylindrical, with a maximum length of 16.90 inches and a maximum diameter of 7.50 inches at room temperature, excluding the Gas Management System. The interface length with the converter support shall be 16.22 inches, excluding the GMS.

5.1.1.7 Weight Goal

The IHS weight goal shall not be greater than 43 pounds, including the Gas Management System.

5.1.1.8 Center of Gravity

The center of gravity of the IHS shall be within a 0.25 inch diameter sphere whose center is at the geometric center of the IHS excluding the Gas Management System.

5.1.1.9 IHS Vent

The IHS shall be equipped with a Gas Management System which permits pressure relief of the IHS as required during each storage period and which shall be designed so that venting to space occurs or internal IHS helium environment is maintained as required by the RTG/Spacecraft Interface Control Document.

5.1.1.10 Vent Connection

The IHS vent shall be equipped with a connecting tube which passes through the RTG outer case to provide for venting the IHS outside the RTG cavity.

5.1.1.11 Mounting

The IHS shall be supported on its ends in the MHW Thermal Converter by the converter support structure, with a preload which is applied during installation and maintained before BOM. Engagement throughout life is a goal.

5.1.1.12 Loading

The IHS shall be capable of withstanding a combined preload and dynamic load of 3600 pounds.

5.1.2 SAFETY CONSTRAINTS

5.1.2.1 Criticality

For any single RTG application (Mission), the total number of Heat Sources, either in their as-designed and assembled shape and configuration or in possible configurations resulting from credible accidents, aborts, and post-abort environments, shall be incapable of a self-sustaining fission reaction in any arrangement and/or combination with reflecting materials.

5.1.2.2 Neutron Emission Rate

The specific neutron emission rate from the unshielded IHS shall not exceed 6.4 times 10^3 neutrons/sec/gr of plutonium, including subcritical multiplication.

5.1.2.3 Subcritical Multiplication

The subcritical multiplication for the Heat Source shall not exceed 1.25, i.e., $K_{\text{eff}} 0.2$.

5.1.2.4 Venting

The IHS and the Gas Management System (GMS) shall be capable of supporting the buildup in helium pressure for a period of two months during transit and storage. The IHS vent system downstream shall be configured such that the IHS can be sealed, opened (both as a separate unit and as installed in the converter) to vent the helium gas and then to reseal for ground handling. A minimum capability of twelve of these cycles shall be provided during the storage life of the IHS.

5.1.2.5 Normal and Abnormal Missions

The design of the RTG shall minimize all safety hazards to the maximum extent possible consistent with the following mission abort design requirements:

1. Specific Re-entry Requirements. No melting of the post-impact shell or failure of the Heat Source structural elements for the probable worst case configuration, i.e., a heat source under the following initial re-entry conditions:
 - a. 400,000 feet - Altitude

- b. 24-36,330 fps - Velocity
- c. 0 to 90⁰ from horizontal - Path Angle

Satisfaction of these requirements shall be demonstrated through analyses which shall be agreed upon by the AEC and the Contractor.

2. Impact/Post Impact

- a. Impact velocity to be maximum sea level velocity observed during the June 1972 "MHW Hot Drop Tests" (256 ft/sec) plus 10 percent, or 281 ft/sec.
- b. Impact on a granite surface at the velocity defined in Item a with no breach of post impact shell as demonstrated by systems verification as follows:
 - (1) Demonstrate with thoria fuel simulant, with characteristics of thoria simulant to be mutually agreed to between AEC/Contractor.
 - (2) See Subsection 5.1, Appendix A-5 of AEC-GE Contract AT(29-2)-2831. Post-Impact shall result in fuel immobilization for one year in the following environment:
 - (a) Land surface $\leq$ 50% burial of a fueled PISA
 - (b) Water $\leq$ 600 feet deep

3. Pad Abort. Fuel shall be immobilized for thirty days after experiencing the following Pad Abort environments:

- a. Explosion: 5% yield
54 psi static over-pressure, 0.77 psi-sec static impulse.
- b. Liquid Fire: 7.5 sec. fireball; fireball emissivity 1.0; after-fire temperature 1875⁰F; after-fire heat flux 13 Btu/sec-ft² ; after-fire duration 30 minutes; after-fire emissivity 0.9.
- c. Impact: Velocity associated with pad abort conditions.
- d. Solid Fire: 10 minutes, 4250⁰F or as experienced in testing.

5.1.2.6 Hazards Proofing

The design of all items shall minimize the hazards of fire, explosion and toxicity to launch area personnel and facilities.

5.1.2.7 Margin of Safety

The design Margin of Safety (M.S.) as defined below shall be greater than zero.

$$\text{M.S. equals } \frac{1}{R} - 1$$

where

$$R \text{ equals } \frac{\text{Applied Stress}}{\text{Allowable Stress}}$$

The following factors of safety shall be applied:

Yield Stress 1.0

Ultimate Stress 1.25

5.2 REFERENCE DESIGN

The design of space-borne radioisotope heat sources has progressed in the past years, from an initial philosophy of high altitude dispersal to the current approach of maximum immobilization in the event of an abort.

The MHW heat source concept represents a significant improvement in safety over previous designs and yet achieves greatly increased temperature capability and attractive specific performance parameters. Rather than encapsulate a large fuel inventory within a single protective shell, it subdivides the fuel inventory into small modules having inherently strong shapes (spherical) which are each individually protected at impact. The fuel spheres are arrayed in a manner such that heat source impact from any angle will not result in an in-line impact of more than two fuel spheres. The array assembly is protected from severe thermal environments; i.e., reentry, pad fire, etc., by a graphitic ablator. In addition, the MHW heat source replaces the "conventional", fuel-encapsulating, refractory metal strength member with a thin, deformable, noble metal shell surrounded by a protective impact shell made of a graphitic crush-up material. This nearly inert metal shell will provide immobilization of the fuel after an accident for an indefinitely long period of time. In the original concept, the spheres were arrayed in a helix which gave rise to the concept name "Helipak". Later trade-off studies made it apparent that multiple planes of spheres, each rotated with respect to the adjacent planes, offered the same ability to eliminate in-line impacts with greatly reduced heat source weight and volume.

In addition, the Helipak concept incorporates a noble metal outer clad over the ablator. This outer clad has four primary functions:

1. Provides pneumatic insulation between heat source and converter.
2. Provides oxidation protection to the graphite aeroshell.
3. Provides skip reentry capability.
4. Provides the capability of maintaining a helium environment within the IHS for heat transfer purposes.

Operationally, the heat source is vented to release the helium generated within the fuel; however, under development is a pressure maintenance device (PMD) which will maintain a slight helium pressure within the IHS during operation if lower IHS internal temperatures are required. The present concept potentially offers the highest degree of containment of any vented heat source designed thus far. Effectively, there are four filters in series through which fuel particles would have to pass in order to escape the heat source: the vents within the fuel encapsulating shells; the semi-porous graphitic impact shells, the aeroshell, and the relatively tortuous path through the gas management system.

As a consequence of these features, the MHW heat source concept:

1. Maximizes the likelihood of very long term fuel containment within the noble metal encapsulation shell both operationally and in the event of an accident.
2. Minimizes the likelihood of a total fuel inventory release in the event of an exceptional accident.
3. Provides an effective, light weight, omni-directional impact protection system.

The current Heat Source design concept is shown in Figure 5.2-1.

5.2.1 DESIGN DESCRIPTION

The MHW heat source utilizes twenty-four spheres of plutonium dioxide to make up the 2400-watt (thermal) inventory. Each sphere is encapsulated in a noble metal (iridium) post-impact containment shell and is provided with its own impact protection. This assembly is referred to as a Fuel Sphere Assembly (FSA). The FSA's are arrayed in six planes along the length of the heat source, four FSA's to a plane, with each plane rotated 45 degrees with respect to the planes adjacent to it. The FSA array is contained within a POCO graphite aeroshell for reentry protection, which is also the basic structure of the heat source. The aeroshell is enclosed by a noble metal (iridium) outer clad which provides the mechanical and thermal interface of the heat source, pneumatic isolation of the heat source from its environment, protection during the multiple-skip class of reentries, and gas management for the helium generated by the fuel.

Figure 5.2-1. MHW Heat Source

5.2.1.1 Fuel

The MHW heat source fuel is a stable ceramic compound, plutonium dioxide, which produces heat as a result of the alpha decay of plutonium-238. The total thermal inventory of 2400 watts is provided by 24 solid ceramic spheres of fuel, nominally 100 watts each. The function of the fuel is to provide the heat energy which is utilized by the converter for direct conversion into electricity, the principal output of the MHW-RTG.

The alpha radiation produced by the fuel, because of its lack of penetration, does not constitute a hazard to human life unless the fuel is inhaled or ingested. The probability of inhalation or ingestion becomes non-zero as the result of an abort accident/reentry, and this is investigated in the MHW safety analysis task. However, the alpha particle reaction with light elements and spontaneous fission of plutonium combine to produce neutrons and gamma rays which can effect sensitive electronic equipment or human life if exposed at close proximities for long periods of time. Typical radiation dose rates at 1 meter for one heat source would be:

1. Neutron dose rate - 30 mrem/hr
2. Gamma dose rate - 15 mrem/hr

These values are estimates based on typical source values of $4(10)^7$ neutrons/sec and $(10)^{12}$ photons/sec for one 2400 watt MHW heat source.

As the alpha particles lose energy, they capture electrons and result in the production of helium molecules. This necessitates the incorporation of gas management considerations into the heat source design.

5.2.1.2 Post Impact Containment Shell (PICS)

The Post Impact Containment Shell (PICS), is a metal cladding which encapsulates each fuel sphere. It is designed to remain intact after exposure to all credible accidents, atmospheric reentries and impacts. The material of the PICS is a noble metal, iridium, which is one of the most non-reactive of materials. As a consequence, it is expected that the basically stable ceramic fuel will be contained for a long period of time within the iridium metal clad following

any accident or reentry. The design requirement for the post-impact containment period is one year after impact.

In addition to its role as a 'primary containment', the PICS provides a surface around the fuel which can be decontaminated after fuel encapsulation and which can be controlled dimensionally to a finer tolerance than the fuel sphere. Release of helium generated by the fuel is provided for by two vent holes in the PICS which are uncapped before final heat source assembly but after PICS decontamination. Retention of any fine particles of fuel which may exist within the PICS is accomplished by a particulate filter (a sintered iridium frit) within the PICS and underneath each vent hole. The total fuel plus PICS assembly is referred to as a Post-Impact Shell Assembly (PISA).

5.2.1.3 Graphite Impact Shell

The Graphite Impact Shell, (GIS), surrounds the PISA and is essentially a crushup structure which minimizes the deceleration loads during earth impact, so that the PISA can survive intact. Trade-off studies conducted earlier in the program determined that it was more advantageous to modularize the fuel and provide individual impact protection for each module rather than to provide one impact structure which protected the entire fuel inventory. This also had the effect of divorcing the design requirements for the impact shell from those of the aeroshell and allowed the choice of optimum materials and design for each component rather than a compromise. Consequently, the aeroshell's function ends at earth impact, at which point its size and shape determine the terminal velocity of the total heat source assembly. Thus, the impact shell is designed to protect the PISA during the impact at the heat source's terminal velocity (plus a 10 percent margin for uncertainties). The assembly of the PISA and its impact shell is referred to as a Fuel Sphere Assembly (FSA).

5.2.1.4 Aeroshell

The aeroshell is an assembly consisting of an ablator cylinder with end caps. The aeroshell is the basic structural member of the heat source assembly and contains all of the other heat source components with the sole exception of the outer clad/gas management assembly. The ablator is designed to maintain the structural integrity of the heat source from assembly

through all credible events (with the exception of some booster explosion situations) to the moment of earth impact following a reentry or an abort. The ablator also provides the heat source aerodynamic shape and thus defines its reentry behavior (heating, ablation, terminal velocity, etc.).

5.2.1.5 Outer Clad

The outer clad is a metal envelope which surrounds the heat source aeroshell. As mentioned earlier, the four primary functions of the outer clad are:

1. Provides pneumatic insulation between heat source and converter.
2. Provides oxidation protection to the graphite aeroshell.
3. Provides skip reentry capability.
4. Provides the capability of maintaining a helium environment within the IHS for heat transfer purposes.

The first function of pneumatic isolation precludes any interactions which could arise due to gaseous species produced by either device.

The second function protects the graphite aeroshell and other IHS internal components from oxidation and any other deleterious effects from ground handling of the IHS in air environment prior to installation into the converter.

The third function protects the ablator from the large surface recessions inherent in the multiple skip class of reentries (those having either too high a velocity or too shallow an entry angle to be captured on the first entry). The ability of this outer clad to remain intact from 3500° to 4000°F eliminates the need for a 0.2 to 0.5 inch of extra ablator thickness.

The fourth function, coincident with the gas management system, provides a capability to utilize the helium generated by the fuel to reduce internal temperatures within the heat source.

The clad also provides the mechanical interface to the heat source.

The outer clad material which satisfies these functions is a noble metal, iridium.

The outer clad is encased in an "emissivity sleeve", a tight-fitting graphite sleeve which provides some structural support for the thin metal and enhances the radiative heat transfer to the thermopile hot junctions. The outer clad end caps furnish the bearing surfaces for support of the IHS within the converter.

5.2.1.6 Gas Management System

The gas management system (GMS) vents the fuel-generated helium from the heat source outboard of the converter. The GMS is comprised of tubulation from the IHS outer clad through the converter outer case, a ground handling vent valve, and either a Pressure Relief Device (PRD) or a Pressure Maintenance Device (PMD). The PRD is a barometrically-operated device which relieves essentially all helium pressure within the IHS during space operation. The PMD maintains slight helium pressure (1 or 2 psi) within the IHS throughout operation to minimize internal temperatures.

At the end of this reporting period, tradeoff studies were being made to select between PRD and PMD.

5.2.2 ASSEMBLY PROCEDURE

A pictorial sequence representing the assembly operations from fuel sphere to clad heat source assembly (without emissivity sleeve) is shown in Figures 5.2-2, 5.2-3 and 5.2-4. The operations shown are those which involve operation in a manipulator cell at Mound Laboratory, the fueling agency.

The sequence shown in Figure 5.2-2 results in twenty-four (24) FSA's ready for installation into the retainer ring subassembly. Figure 5.2-3 illustrates the buildup of the modular, graphitic retainer ring subassemblies (8 FSA's each) and also shows the installation of the non-rigid graphite sphere locks, tie bolts, etc., which position the FSA's against the retaining ring.

Figure 5.2-2. FSA Sequence

Figure 5.2-3. Retainer Ring Assembly

Figure 5.2-4. Final Assembly Sequence

The final Mound Laboratory assembly sequence is shown in Figure 5.2-4. The outer clad-ablator subassembly is open on one end and receives three retaining ring subassemblies. Then a dimensionally-selected end spring is installed, followed by the installation of the ablator end cap which is screwed down to a positive shoulder stop for both dimensional and pre-load control. The heat source assembly operations at Mound are then finalized by effecting the closure weld around the outer clad end cap, thereby joining it to the cylindrical clad section and sealing the assembly.

The final heat source assembly is effected at GE, Valley Forge, in the RTG Loading and Assembly Station (LAS), where the emissivity sleeve is slid over the outer clad, just prior to IHS installation into the converter. This assembly is reserved for LAS operation to assure a high degree of cleanliness of the emissivity sleeve which, when installed, is in close proximity to the converter thermoelectrics and insulation. This assembly sequence precludes the hazards of air environment and other ground handling and transportation considerations.

5.3 ISOTOPIC FUEL

The thermal energy for the MHW-RTG is produced by the radioactive decay of the isotope $^{238}_{94}\text{Pu}$. The isotopic fuel form, its characterization and its behavior under operational and certain simulated accident conditions, are reviewed here. An extended discussion of the MHW-RTG isotopic fuel history is given in the preceding annual reports for this contract, GESP-7071 and GESP-7097.

5.3.1 FUEL FORM

The ^{238}Pu isotope fuel is used in a stable chemical form, the dioxide, PuO_2 , and in the physical form of spheres with diameters of 1.465 ± 0.015 inches. These chemical and physical forms are adequate to satisfy the basic operational and safety requirements of the MHW system.

5.3.2 BASIC REQUIREMENTS

The MHW-RTG fuel form is required to provide the thermal energy of 2400 watts, at a heat source surface temperature of around 1100°C , which gives a fuel/cladding interface maximum temperature around 1450°C . It must be fabricable to the desired physical shape and size. It must be stable during and after encapsulation in all of the expected operational and credible accident conditions to which it may be exposed.

5.3.3 FUEL CHARACTERIZATION

Fuel form characterization and compatibility test programs were carried out during the period January 1973 through December 1973 under AEC-SNS direction at University of California/Los Alamos Scientific Laboratory (LASL), Mound Laboratory/Monsanto Research Corporation (ML/MRC) and the McDonnell Douglas Corp/the Donald W. Douglas Laboratories (DWDL). These programs were designed to establish or verify high temperature properties and behavior of the candidate materials.

The basic properties of the fuel form affect its behavior/response to a variety of thermal, physical, chemical, and mechanical conditions. These are separated somewhat arbitrarily as shown in Table 5.3-1. Specific tests associated with these categories are summarized as to status and results in Tables 5.3-2 and 5.3-3. In briefly reviewing the results here, the

Table 5.3-1. MHW Fuel Characterization

Properties	Behavior/Response
<ul style="list-style-type: none"> • Density • Mechanical <ul style="list-style-type: none"> • Compressive • 4 Point Bend • Youngs Modulus • Poisson's Ratio • Conductivity • Specific Heat • Melting Point <ul style="list-style-type: none"> • Pure Fuel • PPO + Ir + C • Thermal Expansion • Emittance • Solubility • Vapor Pressure • Radiation • Sinterability 	<ul style="list-style-type: none"> • Vibration <ul style="list-style-type: none"> • FSA • HSA • Impact <ul style="list-style-type: none"> • Pellets • Hot Bare Spheres • FSA • HSA • Oxidation • Helium Retention <ul style="list-style-type: none"> • Pellets • Large Samples • Spheres • Radiation Damage <ul style="list-style-type: none"> • Pellets • Spheres • Deoxygenation • Pu Release <ul style="list-style-type: none"> • Vapor • Alpha Knock on • Particulates • Ablation Vaporization • Compatibility • Temperature Ramping

Table 5.3-2. Fuel Characterization - 1

1.0 Fuel Properties (Status - as of 31 December 1973)

Category	Status	Results/Comments
1.1 Density	Continuing	Diametral sectioning in progress to define porosity distribution; range for spheres: 80% TD to 90% TD; nominal MRC/LM 82% TD density variation within spheres = 15%; center-to-surface
1.2 Mechanical <ul style="list-style-type: none"> • Compressive • 4-point bend • Young's Modulus • Poisson's Ratio • Shear • Hardness 	In process Data due '74 Data due '74 Data due '74 Data due '74 Cancelled Cancelled	Room temp. data on LASL and ML/MRC material completed; high temp. data in process Obtainable by LASL Obtainable by LASL No work planned No work planned
1.3 Thermal Conductivity	Partial data Deferred	Complete to 1350°C; high temperature data, to 2200°C, waiting on installation and checkout of apparatus; data needed but no work is planned
1.4 Specific Heat	Complete	Data available from studies on Pu reactor fuels ²³⁹ Pu
1.5 Melting Point <ul style="list-style-type: none"> • Pure Fuel • Fuel + Ir + Carbon 	Complete Complete	Data on production fuel shows effects of impurities Needed to determine synergistic MP depression, no enhanced reaction below Ir + C eutectic at 2200°C. (data by LASL)
1.6 Thermal Expansion	Partial data Deferred	Complete to 1400°C; verifies previous work; requires high temperature apparatus; data required but no work is now planned
1.7 Emittance	Partial data Cancelled	LASL reported data through 1400°C; to 2000°C later; no work planned
1.8 Solubility	Continuing	Solubility in fresh and salt water continuing under study; encapsulated sphere test started Mar 73 (MHFT-13) sea water
1.9 Vapor Pressure	Available	Data are for "Pure" ²³⁹ PuO ₂ ; data checks on production material are required.
1.10 Radiation	Continuing	Data are checked on production spheres; neutron levels dependent on light element impurities content and effectiveness of ¹⁶ O exchange
1.11 Sinterability	Continuing	See LASL reports for prime fuel development studies 10,000 hr. data desired at operational temperatures

Table 5.3-3. Fuel Characterization - 2

2.0 Fuel Behavior/Response (Status as of 31 December 1973)

Category	Status	Results/Comments
2.1 Vibration <ul style="list-style-type: none"> FSA at MRC/LL HSA at GE-ISA0 	In process In process	MHFT-11 vibrated; examined at LASL; sphere broken in 5 pieces; fines at 0.6% total fuel. 7 additional tests = broken spheres Exposure planned on Q-1 IHS; post-mortem due in July 1974
2.2 Impact <ul style="list-style-type: none"> Pellets at LASL Hot bare spheres at ML FSA at LASL HSA at Sandia 	Partial Complete In process In process	Low temperature pellet impact data; need high temperature data Two spheres dropped at 1000°C; broken in large pieces 10 fueled FSAs impacted; (see Impact Test Section 5.2) 3 with ORNL/ML Ir 12-pack impact No. 1 good results; No. 2 with shaved GIS and "bad" Ir poor results; additional tests planned with 24-pack in July 1974
2.3 Aging Effects		
2.3.1 He Retention/Release <ul style="list-style-type: none"> Small pellets at LASL Large samples at LASL Stored spheres at LASL 	Complete In process In process	Data available indicate no problem at 1300°C for small pieces Quantitative retention data at 900, 1150, 1250, 1370 and 1450 obtained On test (compatibility test spheres) release continuous at 1100°C
2.3.2 Radiation Damage <ul style="list-style-type: none"> Small pellets Stored spheres 	In process In process	Change in mechanical properties and microstructure (see Section 2.3.1) Dimensional changes of critical interest (see Section 2.3.1)
2.3.3 Deoxygenation at LASL at GEVNC	Complete Proposed	Exposure to even low levels of oxygen restores O/Pu ratio = 2.00 Determine oxygen potential at O/Pu = 2.000 and 1.995; EMF also
2.4 Oxidation at LASL & ML	In process Due Mar 74	Reduction and reoxidation showed small dimensional changes; Condition of as-finished spheres under review; reduced material observed
2.5 Pu Release <ul style="list-style-type: none"> Vapor Alpha Knock-on Particulates 	Calculated Calculated Under Study	Experiment at ML/MRC delayed by equipment problems
2.6 Ablation/Vaporization	Cancelled	Facilities problems in the use of live fuel
2.7 Compatibility at LASL at DWDL	Tests in Process Test Complete	See detailed breakdown on completed and planned tests - Section 5.3.4 Tests on full-up FSAs started April 1973 5000 hours in gradient, isothermal examined; tensile tests reported
2.8 Thermal Response <ul style="list-style-type: none"> Reentry Thermal Ramp Asymmetrical Thermal Ramp Symmetrical Ramp to Operational - 1450°C 	Complete Complete Complete	Included in the SSST Spheres cracked but remained integral; LASL & MRC/ML spheres run LASL spheres stored 1 year showed no damage or change when ramped to 1450°C at MRC/ML: 1 sphere after 6 months storage OK when ramped to 1250°C

properties and behavior/response are considered as they influence function and not necessarily in the discrete categories listed. Impact testing is discussed in Section 5.4.

5.3.3.1 Density and Mechanical Properties

The fuel spheres fabricated in the production facilities at ML/MRC are routinely checked for density by geometry versus weight. The density currently runs in the range 82 to 85 percent of theoretical density. It has been shown by destructive examination that the sphere density varies from a dense region at the equator and elsewhere near the surface, to a lightly consolidated structure in the center. The variation from equatorial surface to center may range from 90 to 70 percent of theoretical density. There is no specification on the density or the density distribution. It is recognized that the physical dimensions and the thermal inventory of the fuel spheres are of more direct consequence in the operational modes than the geometric density. However, it appears that the density distribution in a sphere has a significant effect on its mechanical behavior under vibration and impact. Studies at LASL are continuing to identify the variations in mechanical strength resulting from differences in microstructure. A homogeneous distribution of the voids seems to produce a stronger material than the discrete void distribution and heterogeneous densification in the fuel currently fabricated by ML/MRC. Whether this more homogeneous fuel structure will survive such abort modes as impact better than the heterogeneous structure has not been established.

Formal measurements of mechanical properties have been delayed by problems of various kinds at LASL. No data at high temperature were obtained during the reporting period.

5.3.3.2 Thermal Properties

There has been no significant change in the status of the thermal properties data during 1973. Determination of the thermal conductivity and thermal expansivity data beyond the current maximum level of $\sim 1400^{\circ}\text{C}$, has been indefinitely deferred. Melting point and specific heat data are adequately defined. No further studies are planned on emittance.

5.3.3.3 Vapor Pressure

A detailed topical report on the various thermodynamic aspects of PuO_2 as the MHW-RTG fuel was prepared by R. A. Kent and issued by LASA as LA-5202-MS, March 1973. This document presents the best current data from both theoretical and empirical sources. Certain questions still remain regarding oxygen losses above 1200°C (see Paragraph 5.3.3.9).

5.3.3.4 Radiation

The radiation from the individual fuel spheres is routinely checked as part of the quality assurance in fabrication by ML/MRC. The primary concern is to insure exclusion of light element impurities and the adequacy of the oxygen-16 exchange.

5.3.3.5 Sinterability

A continuing study is being carried out at LASL for the determination of the effects of various fuel process parameters on the fuel microstructure and the fuel behavior. The size of the starting granules, the heat treatments to which they are subjected before hot pressing, hot press temperature and pressure cycles, and post-press heat treatment of the ceramic compacts are all under investigation.

Sintering or expansion at operational temperature exposures is clearly of concern. The data currently available show negligible dimensional changes in fuel spheres after storage for 6 months followed by temperature excursions to greater than 1450°C . These results have been verified both at LASL and at ML/MRC. In addition, prolonged exposure at 1450°C has not produced significant dimensional changes although a number of spheres have cracked. Exposures for as little as 18 hours at temperatures as high as 1600°C , about 150°C above the fabrication temperatures, however, have resulted in significant additional densification from sintering. (See Paragraph 5.3.3.9.)

5.3.3.6 Solubility and Soil Interaction

Studies continue at LASL on the solubility of bare PuO_2 (both ^{238}Pu and ^{239}Pu) in fresh and salt waters. At the end of this reporting period, (December 1973), a full-size fuel sphere in its vented iridium containment had been in sea water for ~ 275 days without detectable release of Pu. Examination of this sphere is planned after an exposure of at least one year.

Other solubility studies, documented in the LASL monthly reports, cover exposure of bare PuO_2 in the form of debris from impacted fuel spheres, in contact with various soils, to simulated climatic phenomena such as rain and humidity. These tests are in progress, and results are noted in the LASL monthly report.

5.3.3.7 Vibration and Impact

The vibration of the single sphere MHFT-11 produced about one percent fines, (particles ≤ 10 micrometers dia.). The vibration levels used were about 3 times the maximum expected to be found in the mission and thus this result is a worst case. Vibration of an IHS is scheduled as part of the qualification testing of the Q-1 IHS at GE-VF during 1974. Data on fuel behavior from these tests must wait for the post mortem of the FSAs sometime toward the end of 1974. The impact tests of FSAs using both fuel and fuel simulant are discussed in Section 5.4.

5.3.3.8 Helium Release/Retention

The production of He is inherent in the decay of $^{238}\text{PuO}_2$. The range of the alpha particles, α , is of the order of 15 to 20 μm in a solid with the apparent density of the PuO_2 ceramic. Thus in a practical fuel form, the α particles are neutralized within the structure of the ceramic, with the exception of those relatively small numbers which are actually emitted from within $\sim 20\ \mu\text{m}$ of the surface.

LASL has issued a topical report, CMB-5-C-362, by B. A. Mueller, dated 14 March 1973, covering the results to date of the continuing MHW He release/retention studies.* The quantitative retention data - cm^3 per gram of PuO_2 - are plotted versus temperature in Figure 5.3-1. These confirm earlier studies at LASL, ORNL, and MRC/ML on PuO_2 and at other sites on thorium minerals and reactor fuels. These studies showed retention of more than half of the decay product He at temperatures of 900°C . At any given temperature He apparently reaches saturation since the quantity of He retained is significantly less than the amount generated by decay. This assumes that sufficient time has elapsed to generate the quantity required for saturation. The cumulative quantity of He generated by components of the MHW are plotted versus time in Figure 5.3-2.

*See also: Mulford, R.N.R. and Mueller, B.A. "Measurements of Helium Release from materials containing $^{238}\text{PuO}_2$," LASL Report LA-5215, July, 1973.

Figure 5.3-1. Saturation Concentration of Helium in Solid PuO_2 (From LASL Data)

Figure 5.3-2. Helium Generation vs. Time for Various MHW Components

At one time there was concern that the abrupt release of He as noted repeatedly in the literature might occur so rapidly in a real clad fuel sphere that excessive pressures might result because the vents were a significant resistance for the He release. This concern has been dispelled by three factors:

1. The He release occurs rapidly but over a period of tens of seconds rather than a few seconds, allowing time for He to escape without pressure buildup.
2. The vent hole size was increased from 0.0008 inch dia. to 0.005 inch providing a much larger channel for He release.
3. The temperature of the fuel spheres is raised in several steps to the final operational temperature; after assembly into the IHS, the temperature increase is gradual; at the higher temperature, the quantity of He retained at saturation per gram of fuel is lower by as much as a decade.

5.3.3.9 PuO₂ Behavior at High Temperature

5.3.3.9.1 PuO₂ Vaporization,, Oxygen Loss, and Vent Plugging

Vent plugging has occurred from two sources:

1. IrO₂ vapor transport and decomposition to metallic Ir
2. Vapor transport of PuO₂ and other oxides in the fuel into the vent.

5.3.3.9.1.1 Ir Transport. The transport of Ir within the PISA was noted first in the "Goldenberg Test" (reported by ML); later in MHFT-5, -6, -9, examined at ML/MRC; and in MHFT-1, -8, examined at LASL. The vapor phase transport occurred at temperatures of 1450°C where the vapor pressure of pure elemental Ir is 10⁻¹² Atm. Thus, it was concluded that the transport was effected by a volatile compound, almost certainly the oxide, IrO_x (1.5 ≥ x ≤ 2.0). The oxide dissociates above 1100°C in the absence of significant oxygen pressure according to IrO₂ (s) ⇌ Ir(s) + O₂ (g). When oxygen is available, gaseous oxides of Ir can be formed.

This permits Ir to be moved from a hot partially oxidized surface to a cooler surface. These conditions can exist in a fueled PISA if it is assumed that some Ir part such as the weld shield is:

1. Above the decomposition temperature $\cong 1100^{\circ}\text{C}$.
2. Is itself partially oxidized, or
3. Contains oxygen-containing impurities.

Confirmation of this supposition was obtained from examination of MHFT-11 which was recrystallized by heating under vacuum at 1500°C for one hour before assembly. Although this treatment was not intended for this purpose, it is assumed that any IrO_2 and other volatile oxides present were decomposed during the recrystallization. MHFT-9, which was treated similarly, showed minor and questionable indications of Ir transport. A thermal scouring of Ir at 1500°C is now required, in addition to the chemical cleaning used to remove surface impurities.

The question of whether IrO_2 can be formed by reactions between oxygen released by disproportionation of stoichiometric PuO_2 has been examined in some detail (see Paragraph 5.3.3.9.1.2) and is discussed in the LASL Monthly Report for March 1973 (CMB-1954, dated 9 April 73, Pp. 23 and 24). This discussion notes the total absence of Ir transport in MHFT-11 (Table 5.3-4) and in the LASL compatibility tests (to which must be added the DWDL in-gradient and isothermal compatibility tests) and concludes that the presence of IrO_2 on parts prior to assembly adequately accounts for the observed results whereas the presence of sufficient oxygen from the fuel is questionable.

Irrespective of the source of the Ir transport, the design and process changes appear sufficient to minimize the probability of further occurrence. These changes include:

- Thermal scouring of Ir parts at 1500°C in vacuum,
- Increase of the He vent hole size to 0.005 inch diameter,

Table 5.3-4. MHFT-1 thru MHFT-14

FUEL SPHERE ASSEMBLY DESIGNATION	STATUS	EXPOSURE CONDITIONS			VENT HOLE SIZE INCH	DIMENSIONAL CHANGE REPORTED	He RELEASE CHECK METHOD AND RESULT				METALLOGRAPHIC EXAMINATION AND RESULTS
		TEMP. °C	TIME HOURS	ENVIRONMENT			MASS SPECTROMETER He LEAK DETECTOR		BUBBLE TEST HE@20 PSIG		
							TEST FURNACE	VACUUM FIXTURE	VENT 1	VENT 2	
MHFT-1	C&E	1440	2150	VACUUM	0.0008	YES, N/A SWOLLEN	N/A	OPEN BREACHED	CLOSED	CLOSED	VAPOR DEPOSITED IR IN VENTS. BREACHED IN HAZ SWELLED INTO GIS
MHFT-2	C&E	1500	18	VACUUM	0.0008	N/A	N/A	N/A	N/A	CLOSED	IMPACTED. IMPACT FACE VENT CLOSED
MHFT-3	C&E	1500	18	VACUUM	0.0008	N/A	N/A	N/A	N/A	N/A	IMPACTED. VENT DESTROYED
MHFT-4	C&E	1500	18	VACUUM	0.0008	N/A	N/A	N/A	OPEN	CLOSED	IMPACTED 1/F VENT CLOSED
MHFT-5 (1)	C&E	1500	18	VACUUM	0.0008	YES = 0.050	N/A	N/A	CLOSED	N/A	VAPOR DEPOSITED IR (V/D IR) IN VENTS CLAD BREACHED IN HAZ
MHFT-6 (1)	C&E	1500	18	VACUUM	0.0008	YES = 0.060	N/A	N/A	N/A	N/A	V/D IR ON INTERIOR. VENT PROBABLY CLEAR DECON COVER RUPTURED
MHFT-7	C&E	1325	18	VACUUM	0.0008	N/A	N/A	N/A	N/A	N/A	V/D IR ON INTERIOR. N/A ON VENTS
MHFT-8	C&E	1325 700	18 8000	VACUUM HE (STORAGE)	0.0008	N/A	N/A	CLOSED	CLOSED	CLOSED	V/D IR INTERNALLY. N/A ON VENT STORED 11 MONTHS
MHFT-9 (2)	C&E	1500	18	VACUUM	0.0008	N/A	N/A	N/A	N/A	N/A	V/D IR ON INTERIOR. N/A ON VENTS
MHFT-10 (2)	C&E	1500 1150	18 1244	VACUUM HELIUM	0.0008	YES, SHRANK	OPEN	N/A	OPEN	CLOSED	V/D IR+Pu O2 ON INTERIOR. N/A ON VENTS
MHFT-11 (2)	C&E	1500 1000	18 ?	VACUUM HELIUM	0.0008	N/A	N/A	N/A	OPEN	OPEN	NO V/D IR. SOME V/D Pu O2 ON INTERIOR. VENT AREA CLEAN
MHFT-12	C&E	1500	18	VACUUM	0.0008	N/A	N/A	OPEN	N/A	N/A	IMPACTED. PICS WAS DESTROYED BY PHOSPHOROUS CONTAMINATION. N/A ON VENTS
MHFT-13	ON TEST	1500 100	18 3000	VACUUM SEA WATER	0.0008	N/A	N/A	OPEN	N/A	N/A	ON TEST IN SEA WATER
MHFT-14	C&E	1500 1250	18 1248	VACUUM VACUUM	0.0008	YES- SHRANK	CLOSED	N/A	CLOSED	CLOSED	V/D IR+Pu O2 ON INTERIOR. N/A ON VENTS

C&E = TEST EXPOSURE COMPLETE AND SPECIMEN EXAMINED
 N/A = NOT AVAILABLE (NOT DONE OR NOT YET REPORTED)
 (1) = DECONTAMINATION COVERS LEFT INTACT DURING HEAT TREATMENT
 (2) = PARTIALLY ASSEMBLED HEMISPHERES HEAT TREATED ONE HOUR AT 1500 C BEFORE FINAL ASSEMBLY

- Bakeout of the PuO_2 sphere in vacuum at 1500°C and subsequent handling in an inert atmosphere.

The thermal scouring has been discussed. The increase in vent hole size is shown analytically to reduce the possible oxygen partial pressure by several decades in around 40 seconds. The high temperature vacuum bakeout will tend to force PuO_2 from the nominal O/Pu ratio - 2.000 to some value O/Pu - 2.000 - x; $x \geq 0.001$.

The oxygen loss from PuO_2 of exact stoichiometric composition, with the O/Pu ratio = 2.000 was a matter of concern especially because of the potentially high partial pressure of oxygen at temperatures of 1500°C and higher. In the context of the MHW Heat Source the interactions among the constituent materials, PuO_2 -Ir-C, are of direct practical importance. R. A. Kent of LASL prepared an exhaustive survey of this area, issued as a LASL topical report, LA-5202-MS, March 1973.

As a consequence of the concern with Ir transport in the MHW PISA and the possibility of the PuO_2 being the source of oxygen, a meeting was held at Argonne National Laboratory on 22 March 1973, to review the current data on the high temperature vaporization of stoichiometric PuO_2 . It was generally agreed that the calculated values for the oxygen partial pressure above stoichiometric PuO_2 are maximum values and may be high by as much as 2 decades. The meeting also addressed the specific question of Ir transport concluding that while the fuel could not be absolutely excluded as a source of oxygen, it appeared more probable that Ir transport resulting from IrO_2 on and in Ir piece parts, from oxide impurities in the Ir, and/or from unspecified impurities in the fuel. Several experiments were proposed to clarify the behavior of the Pu-O and the Ir-O systems, including:

1. determination of the oxygen pressure/oxygen potential by mass spectrometer, EMF cell, and carbon conversion.
2. the specific effects of IrO_2 on and/or in Ir metal.
3. the specific effects of impurities in both the Ir and in the fuel.

These experiments were in addition to the planned compatibility tests on full-size FSAs done at LASL. (See Figures 5.3-3 and 5.3-4.) In fact, the results of the FSA tests using Ir material subjected to thermal scouring supported the position that Ir transport was the result of IrO_2 on the Ir. See Table 5.3-4.

5.3.3.9.1.2 PuO_2 Vapor Transport. The LASL tests on spheres (Table 5.3-5) uncovered a further problem of He vent plugging.

The compatibility test spheres, MHFT-18 and MHFT-19, were removed from test after approximately 1000 hours and examined. MHFT-18 was exposed at a nominal Ir/ PuO_2 temperature of 1250°C . Examination showed no evidence of Ir transport in any area of the sphere. The vents were both open post-test and in metallographic examination indicated no ceramic material in the vent filters. There was no evidence of any interaction between the Ir and the fuel.

Figure 5.3-3. MHW Fuel Sphere Assembly - FSA: Post Impact Shell Assembly - PISA, in Graphite Impact Shell - GIS

Figure 5.3-4. LASL Compatibility Sphere Test Configuration

MHFT-19 was exposed at a nominal Ir/PuO₂ temperature of 1450°C. Post-test examination showed no evidence in the vent areas of Ir transport. The condition of the Ir under the displaced weld-shield was indeterminate and the roughening of the surface may be a result of Ir transport but this does not appear likely. One of the vents was open post-test and some ceramic material, identified as PuO₂ and CaSiAl + O₂ was found vapor deposited around the Ir granules on the vent filter. The second vent filter did not pass He post-test. Metallographic examination of the vent filter showed what may be a solid plug of ceramic material of the composition noted. The vapor-deposited ceramic on the surface toward the PISA was predominately PuO₂.

LASL uses two different methods to determine whether He is passing through the PISA vents:

1. The mass spectrometer He leak detector which is used on the furnaces containing the compatibility test spheres and on the laboratory vacuum can.

Table 5.3-5. MHW Fuel Sphere Assembly Compatibility Tests at LASL

SPECIMEN NUMBER	MHW OPERATIONAL MODE-GAS IN IR/FUEL GAP	HELIUM VENT HOLE DIAMETER-INCH	EXPOSURE TEMPERATURE DEGREES CELSIUS		TEST START DATE	EXPOSURE TIME HOURS		STATUS (AS OF 31 DECEMBER 1973)
			Q.D. GIS MEASURED	IR/FUEL CALCULATED		PLANNED	ACTUAL	
MHFT-1	VACUUM	0.0008	1340	1440	AUG. 1972	2150	2150	EXPOSURE COMPLETE. DATA REPORTED
MHFT-19	VACUUM	0.005	1340	1440	18 MAY 1973	1460	1105	EXPOSURE COMPLETE. DATA REPORTED
MHFT-15	VACUUM	0.0015	1340	1440	25 MAY 1973	3300	5200	ON TEST
MHFT-22	VACUUM	0.005	1340	1440	MAY 1973	3300	3300	EXPOSURE COMPLETE. DATA REPORTED
MHFT-20	VACUUM	0.005	1340	1440	MAY 1973	8760	4500	ON TEST
MHFT-14	HELIUM	0.0008	1180	1250	MAR. 1973	2150	1248	EXPOSURE COMPLETE. DATA REPORTED
MHFT-18	HELIUM	0.005	1180	1250	21 MAY 1973	1460	1365	EXPOSURE COMPLETE. DATA REPORTED
MHFT-21	HELIUM	0.005	1180	1250	1 JUN. 1973	8760	5200	ON TEST
MHFT-10	HELIUM	0.0008	1018	1150	MAR. 1973	4380	1244	EXPOSURE COMPLETE. DATA REPORTED
MHFT-17	HELIUM	0.005	1018	1150	7 MAY 1973	1460/3300	5700	ON TEST
MHFT-16	HELIUM	0.0015	1018	1150	6 JUNE 1973	3300	5100	ON TEST
MHFT-23	HELIUM	0.005	1018	1150	MAY 1973	3300, 8760	-0-	NOT STARTED

2. A bubble test in which the vent is subjected to 15 - 20 psig He pressure and the presence or absence of bubbles in a cover liquid is observed.

The detection limits of the mass spectrometer He leak detector is around $10^{-11} \text{ cm}^3 - \text{sec}^{-1}$ (STP). It must be noted that the He generation rate of a single MHW fuel sphere is approximately $4.5 \times 10^{-7} \text{ cm}^3 - \text{sec}^{-1}$. Although the bubble test will certainly indicate if a vent is open, it is not sufficiently sensitive to determine whether a vent is closed to the passage of He at the generation rate. Consequently, some question must remain when the ability of a vent to pass He at the generation rate is determined by any technique except the mass spectrometer He leak detector.

Table 5.3-6 shows the conditions of the PISA vents as reported by LASL. "Erratic" release ranges from undetectable levels below $10^{-12} \text{ cm}^3 - \text{sec}^{-1}$ (STP) to $2 \times 10^{-4} \text{ cm}^3 - \text{sec}^{-1}$ (STP). This indicates that the vent ranges from temporary closure to full open, dumping accumulated He at that time. The mechanism of closure appears to be vapor deposition of ceramic material, mostly PuO_2 , around the Ir granules of the vent filter. At this time, it is not clear that a vent has been permanently closed under operational conditions. The He release behavior of the compatibility test spheres MHFT-15, -20, and -22 is being closely monitored by LASL. As shown in Table 5.3-6, these specimens have all been at temperature for more than 3000 hours, at 1440°C , without permanent closure of both vents. The dimensions of the Ir will be carefully checked post-test to determine if any swelling from pressure has occurred.

It was agreed between LASL and GE that He release checks on all spheres will not be made with the mass spec instrument prior to destructive analysis.

5.3.3.9.2 Pu Release

The mechanism of release of Pu species from the MHW PICS vents was discussed at length in the MHW Annual Report GESP-7097 (Paragraph 4.4.4.3 pp 4-79 thru 4-87). Experimental verification of the calculations may be obtained when destructive examination of the GIS used in the FSA tests is performed at LASL, although surface contamination of the PICS during assembly may interfere with an accurate assessment.

Table 5.3-6. MHFT-15 Thru MHFT-33

FUEL SPHERE ASSEMBLY DESIGNATION	STATUS	EXPOSURE CONDITIONS			VENT HOLE SIZE - INCH	DIMENSIONAL CHANGE REPORTED	He RELEASE CHECK METHOD AND RESULT				METALLOGRAPHIC EXAMINATION AND RESULTS
		TEMP. °C	TIME -HOURS	ENVIRONMENT			MASS SPECTROMETER - He LEAK DETECTOR		BUBBLE TEST - He @ 20 PSIG		
							TEST FURNACE	VACUUM FIXTURE	VENT 1	VENT 2	
MHFT-15	ON TEST	1440	2400 (8-31)	VACUUM	0,0015	NYA	ERRATIC	NYA	NYA	NYA	ON TEST
MHFT-16 (1)	ON TEST	1150	2100 (8-31)	HELIUM	0,0015	NYA	N/A	NYA	NYA	NYA	ON TEST IN He
MHFT-17 (1)	ON TEST	1150	2700 (8-31)	HELIUM	0,005	NYA	N/A	NYA	NYA	NYA	ON TEST IN He
MHFT-18 (1)	C&E	1250	1365	HELIUM	0,005	NO	N/A	N/A	OPEN	OPEN	CLEAN, VENT AREA FREE OF ALL V/D MATERIAL, MAY BE SAME V/D Ir ON INTERIOR
MHFT-19	C&E	1440	1105	VACUUM	0,005	NO	ERRATIC	N/A	OPEN	CLOSED	V/D CERAMIC AROUND Ir GRANULES IN VENT FILTER: PuO ₂ + Ca + Si + Al + O
MHFT-20	ON TEST	1440	1500 (8-31)	VACUUM	0,005	NYA	ERRATIC	NYA	NYA	NYA	ON TEST IN VACUUM
MHFT-21 (1)	ON TEST	1250	2200 (8-31)	HELIUM	0,005	NYA	N/A	NYA	NYA	NYA	ON TEST IN He
MHFT-22	ON TEST	1440	1800	VACUUM	0,005	E-1.6, P=1.6 45° ± 1.604MC	ERRATIC	NYA	CLOSED	CLOSED	V/D CERAMIC DEPOSITED IN VENT FILTER SOME EFFUSION THRU VENT TO SURFACE
MHFT-23 (1)	ON TEST	1150	N/A	HELIUM	0,005	NYA	N/A	NYA	NYA	NYA	ON TEST IN He
MHFT-24 (2)	—	—	—	—	—	—	—	—	—	—	DISPOSAL NOT REPORTED (2)
MHFT-25 (3)	C&E (PART)	1440 1850 ²	857 5 m ²	VACUUM	0,005	N/A	N/A	OPEN CLOSED (AFTER)	CLOSED	CLOSED	VENT(S) SHOWED OPEN BEFORE REENTRY THERMAL EXCURSION; VENT AREA NYA BY METALLOGRAPHIC EXAMINATION
MHFT-26 (3)	ON TEST	1440 700 ²	857 720 ²	VACUUM AIR	0,005	N/A	N/A	OPEN PRE-POST	CLOSED	CLOSED	IMPACTED, SHOWS OPEN BY He DETECTOR BUT PISA IS CRACKED, V/D CERAMIC IN VENT AREA
MHFT-27 (3)	ON TEST	1440 700 ²	857 720 ²	VACUUM AIR	0,005	N/A	N/A	OPEN	NYA	NYA	IMPACTED, NOT YET DISSECTED, ON TEST IN AIR
MHFT-28 (3)	C&E (PART)	1440	744	VACUUM	0,005	NO SWELLING, DIMENSIONS N/A	N/A	N/A	CLOSED	CLOSED	V/D CERAMIC AROUND Ir GRANULES IN VENT FILTER: PuO ₂ + Ca + Si + Al + O + W
MHFT-29 (3)	ON TEST	1440	721 11-07	VACUUM	0,005	NYA (DISTORTED)	ERRATIC	OPEN	NYA	NYA	ON TEST, PISA DISTORTED PROBABLY IN VIBRATION TEST, MAY NOT BE IMPACTED
MHFT-30 (3)	ON TEST	1440	721 (11/7)	VACUUM	0,005	NYA	CLOSED	CLOSED	NYA	NYA	ON TEST, TO BE IMPACTED
MHFT-31 (3)	ON TEST	1440	721 (11/7)	VACUUM	0,005	NYA	ERRATIC	OPEN	NYA	NYA	ON TEST, TO BE IMPACTED
MHFT-32	C&E	1370	30 m ²	VACUUM	0,005	N/A	N/A	N/A	OPEN	OPEN	IMPACTED, VENT AREA CLEAN
MHFT-33	C&E	1370	30 m ²	VACUUM	0,005	N/A	N/A	N/A	CLOSED	CLOSED	IMPACTED, VENT AREA CLEAN; METALLOGRAPHY SHOWS NO APPARENT VENT FILTER OR VENT BLOCKAGE

NOTES

NYA = NOT YET AVAILABLE

N/A = NOT AVAILABLE (NOT DONE, NOT REPORTED, OR NOT FEASIBLE)

(1) = SPECIMENS ON TEST IN HELIUM-FILLED FURNACE. He MASS SPEC. DETECTOR CANNOT BE USED DURING TESTING

(2) = SPECIMEN REPORTED CONTAMINATED ABOVE ALLOWABLE LIMITS BY ML / MRC, NOT SHIPPED TO LASL

NC = NO CHANGE

(3) SST - SAFETY SEQUENTIAL TESTS. EXPOSURE FOR 30 DAYS AT OPERATIONAL + REENTRY PULSE + IMPACT + POST-IMPACT EXPOSURE IN AIR

C&E TEST EXPOSURE COMPLETE AND SPECIMEN EXAMINED

ERRATIC - He SIGNAL VARIES FROM BELOW DETECTION LIMIT OF $\geq 10^{-12} \text{cm}^3\text{-SEC}^{-1}$ (STP) TO $\geq 50 \times$ GENERATION RATE = $4 \times 10^{-6} \text{cm}^3\text{-SEC}^{-1}$ (STP) FOR EXTENDED PERIODS

5.3.3.9.3 Fuel Anomaly

Metallographic examination of the fuel from MHFT-34, -35, and -36 showed microstructures different from those previously observed by LASL in ML/MRC fuel. The fuel spheres, PPS-34, -35, and -37--respectively--were the first production run spheres made in the ML/MRC 'A' line of the Plutonium Production, 'PP' Bldg., to be examined at LASL. These microstructures suggested that the fuel (34 and 35 only) had zones in the sphere core where the composition was substantially reduced below stoichiometric, $O/Pu \cong 2.00$, and that the equatorial region and some other surface areas had zones of significantly lower porosity (high density), in -35 and -37 only.

5.3.3.9.3.1 Substoichiometry. It is recognized by LASL, ML/MRC, SRL, and GE that reduction of $PuO_{2.00}$ feed powder occurs in hot pressing of compacts in graphite dies in vacuum. The degree of reduction is not constant, but the resultant compact may be reduced to an overall O/Pu ratio $\cong 1.85$. If the total oxygen available in subsequent carbon removal/reoxidation is insufficient to restore the O/Pu to 2.00, some residual substoichiometric zones can persist. The apparently reduced, two-phase structure seen metallographically in PPS-34 and -35 (but not in PPS-37) was found to be single-phase by X-ray and to have $O/Pu = 2.00$ by TGA. Persistence of the structure suggests that the area examined was initially substoichiometric but was reoxidized in post-test handling at a temperature too low to permit the structure to revert to single-phase structure.

An intensive review of production process equipment and procedures at ML/MRC showed no significant changes over what was previously used to fabricate development spheres in the 'R' Bldg. with one exception. It was found that a change in the $^{16}O_2$ flow rate had been made which under extreme conditions of reduction in the pressing process might have been insufficient to provide enough oxygen to restore the material to $O/Pu \cong 2.0$. Of the spheres made in the ML/MRC 'A' line, about 12 were found to be around $O/Pu = 1.96$ based on the measured weight differences between the stoichiometric feed material and the as-finished sphere. The $^{16}O_2$ flow rate was increased to $80 \text{ cm}^3\text{-min}^{-1}$ (STP) and the 12 suspect spheres were re-treated.

5.3.3.9.3.2 Densification. It was found that the fuel spheres PPS-35 and -37 were heat treated with the Ir PICS in vacuum for 18 hours at 1500°C . This subjected the fuel to temperatures significantly above maximum operational temperatures. These are estimated at 1600°C on the sphere surface and approximately 1775°C at the sphere center. As a result of this heat treatment, entirely outside the normal production processes at ML/MRC, densification of PPS-35 and -37 appears to have occurred. This decrease in porosity/densification was substantiated by Quantimat scanning at LASL. A test with a bare sphere, PPS-90, exposed for 18 hours at 1600°C , produced additional sintering as shown by a decrease of around 0.006 inches on the diameter. It thus appears that significant densification did occur in PPS-35 and -37.

A topical report by the Fuel Task Force, which includes representatives of LASL, ML/MRC and GE-ESP, is in progress.

5.3.4 COMPATIBILITY TESTS

A series of compatibility tests was carried out at LASL and DWDL involving the three MHW-IHS constituents:

1. $^{238}\text{PuO}_2$ Fuel
2. Iridium
3. Carbon (Graphite)

The results from all tests examined to date show that PuO_2 is compatible with Ir for periods of at least 10,000 hours in vacuum at temperatures of 1450°C and in helium at temperatures of 1500°C in the presence of carbon directly contacting the Ir. In addition, compatibility has been similarly demonstrated for up to one hour at 1950°C . The long-term, ≈ 8000 hours, contact of Ir tensile specimens with PuO_2 fuel at 1500°C , produced no significant changes in either ultimate strength or percent elongation.

The tests are briefly discussed in the succeeding paragraphs. Full details may be found in the LASL monthly reports and in the DWDL monthly, quarterly and topical reports. The previous compatibility test series was discussed in GESP-7097.

5.3.4.1 Tests at LASL

Compatibility tests at LASL included capsule tests using the specimen configurations shown in Figures 5.3-5 and 5.3-6. In addition, tests on complete FSAs are in progress in the test configuration shown in Figure 5.3-4. The test matrix for the capsule tests is shown in Table 5.3-7 for the design verification test and in Table 5.3-8 for the diagnostic tests. The test matrix for the FSA tests is given in Table 5.3-5. Table 5.3-9 summarizes the LASL capsule tests through 31 December 1973. The test exposures are completed and detailed examinations are in progress.

5.3.4.2 Tests at DWDL

The compatibility tests at DWDL included capsule tests using the test configurations as shown in Figure 5.3-7 for thermal-gradient tests, and as shown in Figure 5.3-8 for isothermal tests. Tests on tensile specimens as shown in Figure 5.3-9 were also included. The test matrix for the DWDL tests is shown in Table 5.3-10. The test results are summarized in Table 5.3-11. Exposure times for the DWDL tensile specimens were 1250 hours and 8000 hours with selected specimens being subjected to an exposure for five minutes at 1950⁰C to simulate reentry heating. The test data for the 1250-hour tests are given on Table 5.3-12 and the test data for the 8000 hour tests in Table 5.3-13. The statistical data for these tests are listed in Table 5.3-14. A summary of the tensile test results is given in Table 5.3-15. Note that the standard subsize (GE) tensile specimens have been exposed but have not yet been tensile tested.

5.3.5 FUEL FABRICATION

Fuel sphere fabrication process development has proceeded at LASL. Fabrication of test and flight-level fuel spheres on a production basis is underway at ML/MRC.

ASSEMBLY - TEST CAPSULE - UNVENTED

Figure 5.3-5. PuO₂/Ir Compatibility Test Assembly

Figure 5.3-6. LASL-Vented Test Capsule

Table 5.3-7. Test Matrix

Specimen			Fuel			GIS Crush-Up			Exposure Temperature				Exposure Time		Other Conditions		Test Status				
	LASL No.	GE No.	PuO ₂ - ML/MRC	PuO ₂ - SRP	PuO ₂ - ML/MRC + 200%	Impregnated Carbon (RI)	CVD Carbon	Other	Storage - 1150 ^o C	Operational - 1450 ^o C	Option - 1300 ^o C	Reentry - 1850 ^o C - 5 min.	2000 - 3000 hours	Optional/Long	Vented	Unvented - He	Exposure Complete	Accumulated Exposure Hours at 31 December 1973	Examination Complete	Comment	
System Design Verification Tests	14	1	x			x			x					x	x		x	6100	Yes	N/R	
	15	2	x			x			x					x	x		x	10600	(1)		
	16	3	x			x			x					x	x		x	10600	(1)		
	17	4	x			x					x			x	x		x	10600	(1)		
	18	5	x			x					x			x	x		x	10600	(1)		
	19	6	x			x					x			x	x		x	10600	(1)		
	1	7	x			x				x			x		x		x	2333	Yes	N/R	
	2	8	x			x				x			x		x		x	2333	Yes	N/R	
	3	9	x			x									x		x	2333	Yes	N/R	
	5	10	x			x				x				x	x		x	11260	(1)		
	6	11	x			x								x	x		x	11260	(1)		
	7	12	x			x				x				x	x		x	11260	(1)		
	8	13	x			x				x		x		x	x		x	11260	(1)		
	9	14	x			x						x		x	x		x	11260	(1)		
	10	15	x			x					x	x		x	x		x	6000	Yes	N/R	
	11	16	x			x				x				x		x	x	10600	(1)		
	12	17	x			x				x				x		x	x	10600	(1)		
	13	18	x			x				x				x		x	x	10600	(1)		
	20	19	x			x					x			x		x	x	10600	(1)		
	21	20	x			x					x			x		x	x	10600	(1)		
	22	21	x			x					x			x		x	x	10600	(1)		
	23	22	x					x			x			x		x		x	6100	Yes	N/R
	24	23	x					x			x			x		x		x	6100	Yes	N/R
	25	24	x					x			x			x		x		x	10600	(1)	
	26	25		x				x		x				x		x		x	10600	(1)	
	27	26		x				x		x				x		x		x	10600	(1)	
	31	27		x				x		x				x		x		x	10600	(1)	
	28	28		x			x			x			x	x		x	x	x	1662	Yes	N/R
	29	29		x			x			x				x		x		x	6100	Yes	N/R
	30	30		x			x			x				x		x		x	10600	(1)	

(1) Examination in Progress

N/R = No reaction between Ir and PuO₂
 RI - Resin-impregnated Carbon } ThorneI-50 Base
 CVD - Vapor-Deposited Carbon }

ML/MRC - Mound Laboratory Production Grade

SRP - Savannah River Plant - As Received

ML/MRC + 200% - Impurities added to ML/MRC Fuel

Table 5.3-8. Diagnostic Tests

Diagnostic Tests																							
Specimen			Fuel			GIS Crush-up			Exposure Temperature			Exposure Temperature		Other Conditions				Test Status					
		LASL No.	GE No.	PuO ₂ - ML/MRC	PuO ₂ - SRP	PuO ₂ - ML/MRC + 200%	Impregnated Carbon (RI)	CVD Carbon	Spectrographic Carbon	Storage - 1150 °C	Operational - 1450 °C	Option - 1300 °C	Reentry - 1850 °C - 5 min.	2000-3000 hours	Optional/Long	Vented	Unvented - He	CO Gas Flow	Solid Carbon Addition	Exposure Complete	Accumulated Exposure - Hours at 31 December 1973	Examination Complete	Comment
Impurity			31			x			x		x		x	x		x				x	1400	Yes	N/R
			32			x			x		x				x	x				x	4121	(4)	
			33			x			x		x				x	x				x	4121	(4)	
Reentry			34	x			x						x			x				x	60 min	Yes	N/R
			35	x			x						x			x				x	30 min	Yes	N/R
			36	x			x						x			x					(1)		
			37		x		x						x			x					(1)		
			38			x	x						x			x					(1)		
Carbon Reactions With PuO ₂ Fuel			39	x			x				x					x		x			(2)		
			40	x			x				x					x		x			(2)		
			41	x			x				x					x		x			(2)		
			42	x			x				x			x		x			x	x	2300 ?	Yes	N/R
			43	x			x				x				x	x			x		(3)		
			44	x			x				x				x	x			x		(3)		
			45																				
			46																				
			47																				
			47																				
			49																				
			50																				

- (1) Test not yet performed
 (2) Testing in progress on bare fuel
 (3) Test not committed; alternative approach under consideration
 (4) Examination in process
 N/R = No reaction between Ir and PuO₂

Table 5.3-9. Summary of LASL Test Results

NUMBER REPORTED:	NINE OUT OF THIRTY
EXPOSURE TIMES:	1 @ 1600 HOURS 3 @ 2300 HOURS 5 @ 6000 HOURS
TEST RESULTS:	IRIDIUM VERSUS FUEL - NO REACTIONS OBSERVED VAPOR DEPOSITED PuO_2 COATS FUEL SIDE OF VENT COVER SLIGHT PITTING OF GRAPHITE ADJACENT TO VENT HOLE ANOMALOUS PLUGGING OF VENT IN ONE CASE

Figure 5.3-7. Thermal Gradient Test Configuration

Figure 5.3-8. Uniform Temperature Test Configuration

DWDL MICRO-SIZE Ir TENSILE SPECIMEN

GE Iridium Tensile Test Specimen

Figure 5.3-9. Tensile Test Specimens

Table 5.3-10. DWDL Mockup Compatibility Test Matrix

TEST NUMBER	SPECIMEN NUMBER	I _r INNER CAPSULE	I _r OUTER CAPSULE	P _u O ₂ PELLET LASL/ML/MRC	P _u O ₂ PARTICLES ML/MRC	MICROTENSILE - INNER	MICROTENSILE - GRAPHITE	STD. SUBSIZE (GE)	VENT COVER	GRAPHITE CRUSHUP	AEROSHELL GRAPHITE	1500/3000 HOURS	OPTIONAL TIME	OPERATIONAL 1370 C	REENTRY 1850 C (5 MIN)	EXPOSURE COMPLETE	ACCUMULATED EXPOSURE HOURS@31 OCTOBER 1973	EXAMINATION COMPLETE	COMMENT
T-1	1	X	X	X		4	4		X	X	X	X		X		X	1250	YES	N/R
T-2	2	X	X	X		4	4		X	X	X		X	X		X	7800	YES	N/R
T-3	3	X	X	X		4	4		X	X	X	X		X	X	X	1250	YES	N/R
T-4	4	X	X	X		4	4		X	X	X		X	X	X	X	7800	YES	N/R
C-1*	5	X	X			4	4			X	X	X		X		X	1250	YES	N/R
C-3*	6	X	X			4	4			X	X	X		X	X	X	1250	YES	N/R
C-4*	7	X	X			4	4			X	X		X	X	X	X	7800	YES	N/R
M-1	8	X	X		X			6	X	X		X		X		X	1250	YES	N/R
M-3	9	X	X		X			6	X	X		X		X	X	X	7800	YES	N/R
M-4	10	X	X		X			6	X	X			X	X	X	X	7800	YES	N/R
MC-1*	11	X	X					6		X		X		X		X	1250	YES	N/R
MC-3*	12	X	X					6		X		X		X	X	X	7800	YES	N/R
MC-4*	13	X	X					6		X			X	X	X	X	7800	YES	N/R

UNFUELED CONTROL CAPSULES

Table 5.3-11. Summary of DWDL Test Results

NUMBER REPORTED: ALL COMPLETE: 7 FUELED AND 6 CONTROLS

EXPOSURE TIMES: 6 @ 1250 HOURS
7 @ 8000 HOURS

TEST RESULTS: IRIDIUM VS. FUEL - NO REACTIONS OBSERVED
VAPOR DEPOSITED PuO_2 OBSERVED ON INSIDE OF CAPSULE
VENT FLOW RATE REDUCED IN SOME CASES

Table 5.3-12. Iridium Short Term Exposure Tensile Test Data

SPECIMEN NO. A	TEMP (°C)	TENSILE STRENGTH (KSI)	ELONGATION (B) (%)	AVERAGE STRAIN RATE (C) (10 ³ IN./IN.-MIN)	SPECIMEN NO. (A)	TEMP (°C)	TENSILE STRENGTH (KSI)	ELONGATION (B) (%)	AVERAGE STRAIN RATE (C) (10 ³ IN./IN.-MIN)
PRE-TEST									
1 NOTCH	1320	44	52	22					
2 NOTCH	1320	46	57	22					
P-1	1390	34	57	19					
P-2	1400	42	63	23					
P-3	1400	40	55	21					
P-4	1395	42	55	21					
P-5	1350 (G)	47	58	21					
POST-TEST					POST-TEST				
C-1-1-1	1365	37	47	19	T-1-1-1	1405	47	48	11 (F)
C-1-1-2	1395	39	49	20	T-1-1-2	1400	43	51	20
C-1-1-3	1400	38	54	20	T-1-1-3	1400	39	47	21
C-1-1-4	1400	37	46	24	T-1-1-4	1395	45	50	25
C-1-HG-1	1400	45	47	18	T-1-HG-1	1395	37	56	20
C-1-HG-2	1390	38	46	19	T-1-HG-2	1395	39	51	18
C-1-HG-3	1400	36	44	16	T-1-HG-3	1405	36	43	18
C-1-HG-4	1395	40	47	26	T-1-HG-4	1400	41	51	26
C-1-CG-1	1395	44	45	25	T-1-CG-1	1405	39	58	23
C-1-CG-2	1400	40	33 (E)	14	T-1-CG-2	1395	46	54	22
C-3-1-1	1385	36	49	19	T-3-1-1	1405	38	53	20
C-3-1-2	1385	36	47	20	T-3-1-2	1395	44	48	21
C-3-1-3	1405	31	53	19	T-3-1-3	1395	44	54	22
C-3-1-4	1395	34	48	24	T-3-1-4	1400	42	40	23
C-3-HG-1	1400	38	46	20	T-3-HG-1	1395	40	51	19
C-3-HG-2	1400	33	32	14	T-3-HG-2	1410	38	49	20
C-3-HG-3	1385	35	51	19	T-3-HG-3	1390	41	52	22
C-3-HG-4	1395	32	28	20	T-3-HG-4	1405	42	45	24
C-3-CG-1	1410	32	38	21	T-3-CG-1	1400	46	53	13 (F)
C-3-CG-2	1395	37	24 (D)	13	T-3-CG-2	1405	35	24 (D)	6

- A. SPECIMEN DESIGNATIONS ARE T-1 FUELED, 1250 HR C-1 CONTROL, 1250 HR
T-3 FUELED, 1250 HR + 5 MIN AT 1950°C, C-3 CONTROL 1250 HR + 5 MIN AT 1950°C
I, INSIDE IRIIDIUM CAPSULE AT 1370°C
HG, OUTSIDE IRIIDIUM CAPSULE ADJACENT TO ATJ GRAPHITE AT 1400°C
CG, IN POCO GRAPHITE AT 1200°C
P, PRETEST - 1500°C, 1 HR
- B. ESTIMATED EFFECTIVE GAGE LENGTH OF 0.190 IN. USED FOR ALL ELONGATION MEASUREMENTS.
- C. AVERAGE STRAIN RATE BASED ON SPECIMEN ELONGATION, EFFECTIVE GAGE LENGTH, AND TIME TO BREAK.
- D. BROKE NEAR END OF GAGE SECTION.
- E. BROKE AT PIN.
- F. RUBBER BUMPER STUCK TO BOTTOM OF WEIGHT CAUSING SLOWER STRAIN RATE BUILDUP.
- G. SPECIMEN HELD 3 MIN AT TEMPERATURE ALL OTHER SPECIMENS HELD 10 TO 30 SEC.

Table 5.3-13. Iridium Long Term Exposure Tensile Test Data

SPECIMEN NO.(A)	TEMP (°C)	TENSILE STRENGTH (KSI)	ELONGATION (B) (%)	AVERAGE STRAIN RATE (C) (10 ³ /MIN)	SPECIMEN NO.	TEMP (°C)	TENSILE STRENGTH (KSI)	ELONGATION (B) (%)	AVERAGE STRAIN RATE (C) (10 ³ /MIN)
PRE-TEST									
P-6	1400	38	48	22					
P-7	1400	36	35	22					
POST-TEST					POST-TEST				
C-4-I-1	1400	39	51	26	T-2-I-1	1400	42	49	25
C-4-I-2	1410	36	48	25	T-2-I-2	1395	42	46	24
C-4-I-3	1400	40	48	25	T-2-I-3	1400	40	49	25
C-4-I-4	1395	36	50	25	T-2-I-4	1410	50	51	25
C-4-HG-1	1410	37	53	25	T-2-HG-1	1400	42	44	23
C-4-HG-2	1415	32	50	25	T-2-HG-2	1405	42	47	25
C-4-HG-3	1395	39	43	22	T-2-HG-3	1400	41	49	25
C-4-HG-4	1395	37	40	24	T-2-HG-4	1395	43	47	25
C-4-CG-1	1405	36	48	24	T-2-CG-1	1405	42	45	23
C-4-CG-2	1395	38	51	25	T-2-CG-2	1400	43	65	29
					T-4-I-1	1405	42	46	23
					T-4-I-2	1395	43	48	25
					T-4-I-3	1390	48	55	25
					T-4-I-4	1395	43	60	26
					T-4-HG-1	1400	38	46	25
					T-4-HG-2	1385	35	48	25
					T-4-HG-3	1395	36	41	22
					T-4-HG-4	1395	42	44	23
					T-4-CG-1	1400	42	— (D)	—
					T-4-CG-2	1395	42	— (D)	—

A. SPECIMEN DESIGNATIONS ARE: T-4 FUELED, 8000 HR, C-4 CONTROL, 8000 HR
T-4 FUELED, 8000 HR + 5 MIN AT 1950°C

B. ESTIMATED EFFECTIVE GAGE LENGTH OF 0.190 IN. USED FOR ALL ELONGATION MEASUREMENTS.

C. AVERAGE STRAIN RATE BASED ON SPECIMEN ELONGATION, EFFECTIVE GAGE LENGTH, AND TIME TO BREAK.

D. UNABLE TO MEASURE ELONGATION, BROKE NEAR END OF GAGE AREA.

Table 5.3-14. Iridium Tensile Specimen Group Averages

SPECIMEN GROUP	NUMBER OF SPECIMENS	TENSILE STRENGTH (KSI)		ELONGATION (%)	
		MEAN	S	MEAN	S
SHORT TERM EXPOSURE					
T-1	10	41	4.0	51	4.4
C-1	10	39	3.0	46	5.2
C-1	9 (B)			47	2.8
T-3	10	41	3.4	47	9.0
T-3	9 (B)			50	4.5
C-3	10	34	2.3	42	10.4
C-3	8 (B)			45	7.1
I	16	39	4.5	49	3.4
HG	16	38	3.4	46	7.5
HG	15 (B)			48	5.9
CG	8	40	5.1	41	13.5
CG	5 (B)			50	8.2
JANUARY TESTS	30	39	4.0	47	8.4
JANUARY TESTS	27 (B)			49	5.1
NOVEMBER TESTS	10	39	4.9	44	6.9
NOVEMBER TESTS	9 (B)			46	4.3
ALL	40	39	4.2	46	8.1
ALL	36 (B)			48	6.1
LONG TERM EXPOSURE					
T-2	10	43	2.8	49	5.8
C-4	10	37	2.2	48	3.9
T-4	10	41	3.9		
T-4	8 (A)			48	6.3
I	12	42	4.1	50	4.0
HG	12	38	3.3	46	3.9
CG	6	41	2.6		
CG	4 (A)			52	8.7
ALL	30	40	3.7		
ALL	28 (A)			49	5.2
PRETEST					
JANUARY TESTS	4	40	3.5	58	3.6
NOVEMBER TESTS	2	37	0.8	41	9.5
ALL	6	39	3.0	52	9.8

- NOTES A. UNABLE TO MEASURE ELONGATION FOR 2 SPECIMENS.
- B. SPECIMENS HAVING LOW ELONGATION ELIMINATED FROM AVERAGING. SPECIMEN ELIMINATION DID NOT AFFECT TENSILE STRENGTH.

Table 5.3-15. Summary of Microtensile Tests at DWDL

NUMBER OF SPECIMENS	TEMPERATURE OC	TIME HOURS	MEAN TENSILE STRENGTH KPSI	MEAN ELONGATION PERCENT
40	1500	1250	39	48
4	1500	1250	41	51
(FUELED)				
4	1500+	1250+	41	50
(FUELED)	1950	5 MIN.		
30	1500	8000	40	49
4	1500	8000	43	49
(FUELED)				
4	1500+	8000+	41	48
(FUELED)	1950	5 MIN.		
6	1500	1.0	39	52
(UNEXPOSED)				

5.3.5.1 LASL Process Development

The LASL fuel process development effort has concentrated on evaluating the effects of process variables on the finished compact by varying the parameters in a consistent fashion. Details of the work are given in the LASL monthly reports (see also Paragraph 5.3.3.5).

The LASL process development data are transmitted to DuPont-Savannah River Plant to provide a technology base for the planned shift of fuel form production from ML/MRC to SRP in CY 1975.

5.3.5.2 ML/MRC Fuel Form Production

Up until March of 1973, MHW fuel spheres were made by ML/MRC in the "R" building glove box lines. Since that time fabrication of fuel spheres both for testing and flight has been done in the Plutonium Process Building Hot Cell "A" Line.

Although the equipments of the "A" line are somewhat different, the hot pressing processes and the quality assurance procedures supporting them are substantially unchanged (see also Paragraph 5.3.3.9.3). The processes and the product were described in detail in GESP-7097 (pp 4-43 to 4-45).

5.3.5.3 Fuel Specification

The ML/MRC Fuel Specification ML-1-14813 was issued in final form. It has been changed on several occasions to conform to the realities of production processing and accommodate changes in systems requirements.

5.3.5.4 Feed Material

The feed material to ML/MRC comes from the SRP process. On several occasions during the year, levels of Ca, Na, Si and some other impurities have exceeded the limits of the ML/MRC Feed Material Specification. The impurity levels in the final product have been kept within the MHW specification limits by blending of batches. Although the LASL and DWDL tests have shown conclusively that there is no chemical incompatibility resulting from impurities, they have also shown that a substantial part of the material vapor-deposited in and tending to plug the PICS vent filters does consist of impurities such as Ca and Si. Study of the fuel feed impurity problem is continuing.

5.3.6 FUEL SIMULANT

Because of the high cost of the $^{238}\text{PuO}_2$ fuel and the problems inherent in handling it, a simulant for the fuel was needed to perform a large number of mechanical tests. These tests were aimed principally at verifying the ability of Ir-encapsulated graphite-encased fuel to survive impact without release. Initially, it was considered sufficient to simulate the size and weight of the PuO_2 . Subsequently, it was found necessary to attempt the simulation of the mechanical behavior of the fuel, first, by matching its density and later by matching its thermal response and microstructure. Thorium dioxide was chosen as the simulant material. A full discussion of the evolution of the thoria simulant was given in GESP-7097 (pp 4-45 to 4-49).

The thoria simulant fabricated by hot pressing has been used for the impact tests discussed in Section 5.4. The original process specification was amended to include heat treatment at 1500°C successively for four hours under each condition in:

1. Oxygen - to remove carbon,
2. Hydrogen - to scavenge sulfur,
3. Vacuum - to remove absorbed gases.

An additional change was made in the pressing parameters to effect a more coherent outer surface. The microstructure remains essentially unchanged and is deliberately patterned after the ML/MRC PuO_2 fuel microstructure.

To confirm the dimensional stability of the ThO_2 spheres two spheres, one from the group made prior to the change in pressing parameters and one made subsequent to the change, were held for > 24 hours in vacuum at 1500°C . Within the limits of the measuring devices, ± 0.002 inches on the sphere diameter, no significant dimensional changes were observed on either sphere.

5.4 POST IMPACT SPHERE ASSEMBLY (PISA)

The helipak concept divides the total fuel inventory into 24 equal parts, each of which is provided with its own impact protection. Each of the 24 fuel spheres is individually encapsulated within iridium and graphite; this is labeled an FSA, fuel sphere assembly. The graphite is specially selected for impact absorption and is discussed in Section 5.5. The iridium is required for post-impact containment and to isolate the fuel from interaction with the graphite impact-absorbing member of the FSA.

A PISA, comprised of a fuel sphere and eight iridium components, is shown in Figure 5.4-1. Two matched hemispheres are the prime containment. Each hemisphere is provided with a 0.005-inch diameter electrical-discharge-machined (EDM) hole, in the polar region, to vent helium produced by the fuel during operation. Each vent hole is covered on the I.D. by an iridium, frit-type, vent assembly, which forces the helium through a tortuous path to the outside of the PISA. The vent assembly is discussed in Section 5.4.2. An iridium decontamination cover is welded to the O.D. of each hemisphere to prevent liquids from entering

Figure 5.4-1. Post-Impact Sphere Assembly

the assembly during decontamination of the PISA after closure welding. The last item in the assembly is an 0.005-inch thick by 3/8 inch wide weld-shield that is positioned between the fuel and the hemisphere to minimize weld-heat at the surface of the fuel during closure welding. The minimum I. D. of the iridium hemisphere and the maximum O. D. of the fuel sphere are sized to provide a gap through all phases of flight and reentry.

5.4.1 DESIGN OBJECTIVES AND CHANGES/FABRICATION EXPERIENCE

5.4.1.1 Design Objectives

PISA design objectives are as itemized below:

1. Contain fuel with iridium during:
 - a. Ground handling @ 300°C
 - b. Operation (He) @ 1220°C
 - c. Post-Impact @ 800°C
2. Isolate liquids from fuel during decontamination.
3. Allow helium to escape; clad loading until after impact.
4. Remain essentially unchanged after exposure to MHW environment; Fuel & Graphite @ 1335°C in vacuum (Gaps)

Development, covered in the 1972 Annual Report, has demonstrated acceptable iridium performance in 800°C air. Development covered in the 1973 report shows that iridium behaves acceptably with both fuel and carbon in all anticipated MHW environments. The response of iridium on impact is discussed in Section 5.4.3 and PISA venting performance is covered in Section 5.4.2.

5.4.1.2 Fabrication Experience

Until this year, fabrication of iridium components has been the sole responsibility of Engelhard Industries (EI), Carteret, New Jersey. All phases of fabrication are considered pro-

proprietary information and cannot be discussed here. Early this year, a back-up effort was undertaken under AEC direction. Ir sheet is produced by ORNL (See ORNL Metals and Ceramic Division Monthly and Quarterly Reports) and ML/MRC forms the hemispheres. Toward the latter part of the year it was learned that Engelhard could not meet the required schedule commitments and ORNL/MRC became the prime source for the hemispheres for the Program. Background concerning this change in suppliers is discussed briefly below.

After a series of successful impacts (See Section 5.5), GE began experiencing a number of failures indicating possible reduced quality in hemispheres produced at EI. GE signed a proprietary agreement that provided full access to the EI fabrication/production processes. Significant changes in the PICS processes occurred between Lot 32, which was used to finish single ball reference design testing, and Lots 36 and 38 which were in the eccentric/shaved, 12-pack. Any of these changes could have produced the abrupt loss of apparent toughness evidenced in single ball impacts S/N 565, 568 and 581 and the 12-pack PISA's. It was concluded, from a review of EI procedures, that the hemisphere fabrication process was out of control. A significant effort was made to upgrade processing and to monitor and control material flow through EI. The first fully-controlled Lot emanating from this Program was Lot #80. This lot was alloyed with ~ 2500 ppm tungsten to improve the impact toughness of the end-product. All material fabricated prior to this Lot is designated Type II and Lots subsequent to #80 are identified as Type III. (Type I is a particular grade of ultra pure Ir, Lot #57, that was tested experimentally but never became a part of the hardware scheduled for PISA fabrication).

GE determined that Lot 80 type material fulfilled the impact needs of the MHW Program (See Section 5.5). EI experienced difficulty in shipping finished Type II and Type III PISA's because of dye-penetrant indications detected on the surfaces of the majority of hemispheres and because of deviations in geometry (especially in male and female step-joint preparations) identified by dimensional inspection. Delay in program test objectives were experienced and flight commitments were jeopardized. As a result, the AEC decided the EI would no longer produce hemispheres for the MHW Program. Instead, ORNL would produce Ir sheet and blanks and ML/MRC would hydroform hemispheres from the blanks provided by ORNL.

ORNL - ML/MRC impacted PISA's (Section 5.4.3) failed to perform as expected and a special team with representatives from ML/MRC, ORNL, SNS, and GE was formed to study the problem. The ORNL-produced sheet was found, by use of ion mass analyses (IMA) to have high levels of surface impurities including Al, Ti, and Si, (See Figures 5.4.1-1, 2, and 3). At the MHW operational temperature the impurities tend to diffuse into the body of the material preferentially along grain boundaries. These impurities are believed to originate in the grinding materials used to bring the sheet to the final desired thickness. The processes initially used to remove these grinding residues were ineffective. Subsequent improvements in the cleaning process have reduced the amounts of surface contaminants. Details of this work are documented in ORNL and ML/MRC monthly reports.

EI fabricated sufficient 0.005-inch thick Ir sheet to supply all the projected program needs for vent components, decontamination covers, and weld-shields. EI shipped the finished decontamination covers and weld-shields directly to ML/MRC but shipped the vent components to GE for subsequent fabrication into finished vent assemblies.

5.4.2 PISA VENTING

At the start of the year the prime vent assembly was configured as shown in Figure 5.4.2-1 except that each hemisphere is provided with a 0.0005 inch diameter laser-drilled hole in the polar region to vent helium produced during operation. Each vent-hole is covered on the inside surface of the hemisphere, by two perforated (4-holes) disks which force the helium through a tortuous path in its passage to the outside of the assembly. The innermost disk is coated on both sides with pure iridium powder to a thickness of about 0.002 inch. The powder prevents self-welding during operation and captures solid fuel particles that may be entrained in the helium stream.

Plugging of vents became of concern as a result of fuel-test experiments performed at LASL. Testing data is shown in Tables 5.4.2-1 & 2. The type of vent assembly on test at the time is shown in Figure 5.4.2-1 except that the vent holes were 0.0005 inches in diameter (instead of 0.005 inches diameter as per a later redesign). PISA's behaving in an erratic manner indicate that the vents were operating over a range of temporary closure to full open. The

Figure 5.4.1-1. Sample WC28-4 (with KCN). Positive ion IMA-images prepared from cross-section specimen. Sample had been ground with alumina wheel, then stress relieved at 850°C for 15 minutes. It was then cleaned in KCN for 15 minutes at 5 volts. Magnification = 240X.

Figure 5.4.1-2. Sample WC 14 (cross-section) before long high temperature treatment. Sample was first heated to 1050°C for one hour before ORNL-super-cleaning. Cleaning procedure used: 10 minutes in hot (90°C) 70% HNO_3 + 30% HF ; then 2 minutes in hot (90°C) HNO_3 ; then two distilled water rinses; then hot (90°C) distilled water rinse; then air dry. Magnification = 240X.

Figure 5.4.1-3. Sample WC 14 (cross-section), same as sample in Figure 5.4.1-2 except long heat treatment after cleaning. Heat treatment: 1500°C for 18 hours. Magnification = 240X.

VENT FILTER - ASSEMBLED VIEW

VENT FILTER - EXPLODED VIEW

Figure 5.4.2-1. MHW Vent Filter - Current Design

Table 5.4.2-1. MHW Fuel Sphere Assembly Compatibility Tests at LASL

SPECIMEN NUMBER	MHW OPERATIONAL MODE - GAS IN Ir/FUEL GAP	HELIUM VENT HOLE DIAMETER - INCH	EXPOSURE TEMPERATURE - DEGREES CELSIUS		TEST START - DATE	EXPOSURE TIME - HOURS		STATUS (as of 31 October 1973)
			O. D. GES -MEASURED	Ir/FUEL CALCULATED		PLANNED	ACTUAL	
MHFT-1	vacuum	0.0008	1340	1440	Aug. 1972	2150	2150	Exposure complete; data reported
MHFT-19	"	0.005	1340	1440	18 May 1973	1400	1105	Exposure complete; data reported
MHFT-15	"	0.0015	1340	1440	25 May 1973	3300	3700	On test
MHFT-22	"	0.005	1340	1440	May 1973	3300	3300	On test
MHFT-20	"	0.005	1340	1440	May 1973	8760	3000	On test
MHFT-14	helium	0.0008	1180	1250	Mar. 1973	2150	1248	Exposure complete; data reported
MHFT-18	"	0.005	1180	1250	21 May 1973	1400	1365	Exposure complete; data reported
MHFT-21	"	0.005	1180	1250	1 June 1973	8760	3700	On test
MHFT-10	helium	0.0008	1018	1150	Mar. 1973	4380	1244	Exposure complete; data reported
MHFT-17	"	0.005	1018	1150	7 May 1973	1460/ 3300	4200	On test
MHFT-16	"	0.0015	1018	1150	6 June 1973	3300	3600	On test
MHFT-23	"	0.005	1018	1150	May 1973	3300/ 8760	-0-	Not started

Table 5.3-6. MHFT-15 Thru MHFT-33

FUEL SPHERE ASSEMBLY DESIGNATION	STATUS	EXPOSURE CONDITIONS			VENT HOLE SIZE - INCH	DIMENSIONAL CHANGE REPORTED	He RELEASE CHECK METHOD AND RESULT				METALLOGRAPHIC EXAMINATION AND RESULTS
		TEMP. - °C	TIME - HOURS	ENVIRONMENT			MASS SPECTROMETER He LEAK DETECTOR		BUBBLE TEST - He @ 20 PSIG		
							TEST FURNACE	VACUUM FIXTURE	VENT 1	VENT 2	
MHFT-15	ON TEST	1440	2600 (8-31)	VACUUM	0.0015	NYA	ERRATIC	NYA	NYA	NYA	ON TEST
MHFT-16 (1)	ON TEST	1150	2100 (8-31)	HELIUM	0.0015	NYA	N/A	NYA	NYA	NYA	ON TEST IN He
MHFT-17 (1)	ON TEST	1150	2700 (8-31)	HELIUM	0.005	NYA	N/A	NYA	NYA	NYA	ON TEST IN He
MHFT-18 (1)	CBE	1250	1365	HELIUM	0.005	NO	N/A	N/A	OPEN	OPEN	CLEAN; VENT AREA FREE OF ALL V/D MATERIAL; MAY BE SAME V/D Ir ON INTERIOR
MHFT-19	CBE	1440	1105	VACUUM	0.005	NO	ERRATIC	N/A	OPEN	CLOSED	V/D CERAMIC AROUND Ir GRANULES IN VENT FILTER: PuO ₂ + Ca + Si + Al + O
MHFT-20	ON TEST	1440	1500 (8-31)	VACUUM	0.005	NYA	ERRATIC	NYA	NYA	NYA	ON TEST IN VACUUM
MHFT-21 (1)	ON TEST	1250	2200 (8-31)	HELIUM	0.005	NYA	N/A	NYA	NYA	NYA	ON TEST IN He
MHFT-22	ON TEST	1440	1800	VACUUM	0.005	E=1.6; P=1.6 45° = 1.604NC	ERRATIC	NYA	CLOSED	CLOSED	V/D CERAMIC DEPOSITED IN VENT FILTER SOME EFFUSION THRU VENT TO SURFACE
MHFT-23 (1)	ON TEST	1150	N/A	HELIUM	0.005	NYA	N/A	NYA	NYA	NYA	ON TEST IN He
MHFT-24 (2)	—	—	—	—	—	—	—	—	—	—	DISPOSAL NOT REPORTED (2)
MHFT-25 (3)	CBE (PART)	1440 1850?	857 8 m?	VACUUM	0.005	N/A	N/A	OPEN CLOSED (AFTER)	CLOSED	CLOSED	VENT(S) SHOWED OPEN BEFORE REENTRY THERMAL EXCURSION; VENT AREA NYA BY METALLOGRAPHIC EXAMINATION
MHFT-26 (3)	ON TEST	1440 700?	857 720?	VACUUM AIR	0.005	N/A	N/A	OPEN PRE-POST	CLOSED	CLOSED	IMPACTED; SHOWS OPEN BY He DETECTOR BUT PISA IS CRACKED; V/D CERAMIC IN VENT AREA
MHFT-27 (3)	ON TEST	1440 700?	857 720?	VACUUM AIR	0.005	N/A	N/A	OPEN	NYA	NYA	IMPACTED; NOT YET DISSECTED; ON TEST IN AIR
MHFT-28 (3)	CBE (PART)	1440	744	VACUUM	0.005	NO SWELLING; DIMENSIONS N/A	N/A	N/A	CLOSED	CLOSED	V/D CERAMIC AROUND Ir GRANULES IN VENT FILTER: PuO ₂ + Ca + Si + Al + O + W
MHFT-29 (3)	ON TEST	1440	721 11-07	VACUUM	0.005	NYA (DISTORTED)	ERRATIC	OPEN	NYA	NYA	ON TEST; PISA DISTORTED PROBABLY IN VIBRATION TEST; MAY NOT BE IMPACTED
MHFT-30 (3)	ON TEST	1440	721 (11/7)	VACUUM	0.005	NYA	CLOSED	CLOSED	NYA	NYA	ON TEST; TO BE IMPACTED
MHFT-31 (3)	ON TEST	1440	721 (11/7)	VACUUM	0.005	NYA	ERRATIC	OPEN	NYA	NYA	ON TEST; TO BE IMPACTED
MHFT-32	CBE	1370	30 m?	VACUUM	0.005	N/A	N/A	N/A	OPEN	OPEN	IMPACTED; VENT AREA CLEAN
MHFT-33	CBE	1370	30 m?	VACUUM	0.005	N/A	N/A	N/A	CLOSED	CLOSED	IMPACTED; VENT AREA CLEAN; METALLOGRAPHY SHOWS NO APPARENT VENT FILTER OR VENT BLOCKAGE

NOTES

NYA - NOT YET AVAILABLE

N/A - NOT AVAILABLE (NOT DONE, NOT REPORTED, OR NOT FEASIBLE)

(1) - SPECIMENS ON TEST IN HELIUM-FILLED FURNACE. He MASS SPEC. DETECTOR CANNOT BE USED DURING TESTING

(2) - SPECIMEN REPORTED CONTAMINATED ABOVE ALLOWABLE LIMITS BY ML / MRC; NOT SHIPPED TO LASL

NC - NO CHANGE

(3) - SST = SAFETY SEQUENTIAL TESTS: EXPOSURE FOR 30 DAYS AT OPERATIONAL + REENTRY PULSE + IMPACT + POST-IMPACT EXPOSURE IN AIR

CBE - TEST EXPOSURE COMPLETE AND SPECIMEN EXAMINED

ERRATIC - He SIGNAL VARIES FROM BELOW DETECTION LIMIT OF $\geq 10^{-12} \text{cm}^3 \cdot \text{SEC}^{-1}$ (STP) TO $\geq 50 \times$ GENERATION RATE, $= 4 \times 10^{-6} \text{cm}^3 \cdot \text{SEC}^{-1}$ (STP) FOR EXTENDED PERIODS

mechanism of closure appears to be vapor deposition of ceramic material, mostly PuO_2 , around the Ir granules of the vent filter. At the time of this writing, it is not certain that a vent has been permanently closed under operating conditions. The He release behavior of test spheres MHFT-15, 20, and 22 is being closely monitored by LASL. As shown in Table 5.4.2-1, these specimens have been at temperature (1440°C) for more than 3000 hours. The dimensions of the Ir will be carefully checked after-test to determine if any swelling has occurred from applied pressure.

The theories concerning the nature of vent plugging are too numerous to discuss each one here, but the three major considerations are as follows:

1. Contaminants in the fuel and/or on the internal surfaces of the PISA become gaseous during operation and deposit in vent openings.
2. Oxygen is released from the fuel due to weld heat and available oxygen combines with Ir to form an unstable oxide. The volatile Ir oxide flows to the cooler regions of the vent and deposits Ir metal in available openings.
3. Oxygen and PuO_2 are released from the fuel during operation and Ir metal and PuO_2 are transported to vent openings.

A number of experiments were conducted to identify the cause of the observed plugging and back-up filter designs were investigated. The more important experimentation is discussed briefly below.

5.4.2.1 Hemisphere Cleanliness

The surfaces of MHW PISA's were cleaned by different treatments and examined for the presence of impurities that might be volatile in a MHW environment. It was determined that the surfaces of the MHFT series of PISA's (fuel spheres having vent clogging problems) did have contaminants (Si, Fe, Al, etc.) on their surfaces prior to closure welding; Aqua-Regia cleaning was proven to be inadequate.

It was learned that HF, used after Aqua-Regia removal residual volatiles (principally Si) but did not remove tenacious oxides; ie, IrO_2 . Heat treatment for 1 hour at 1500°C in 10^{-6}

torr range vacuum, removed most oxides except Al_2O_3 . As a result of this work HF and high temperature vacuum firing were introduced into cleaning requirements of all hemisphere piece parts.

5.4.2.2 Hole Drilling

It was demonstrated that smaller holes ($\sim .001$ inch dia.) can be drilled by Laser than by Electron Beam (EB) ($\sim .002$ inch min.) and that the Laser-drilled hole is more apt to be uniform in cross section than the EB hole which tends to be conical in shape (small at beam entrance and larger at beam exit). Both type holes tend to be rough and difficult to clean of residual splatter and tramp oxides. EDM machining produces the most uniform bore but the largest hole ($\sim .004$ inch min.). In addition, the larger EDM hole is easiest to clean chemically.

5.4.2.3 PIC Cleaning

Two differently cleaned PISA's were brought to 1500°C , in vacuum, to determine if surface contaminants can produce volatile constituents capable of plugging vent assemblies. The first PISA used MHFT type hardware (Type II) cleaned only by Aqua-Regia. The second PISA used type III hardware cleaned by Aqua-Regia and HF. PuO_2 was encapsulated, rather than fuel, to demonstrate cloggability without the presence of fuel. The vents in the MHFT PISA clogged shut while those in the Type III PISA remained open. A third, PISA (Type III) was cleaned with Aqua-Regia, HF, and by 1500°C in vacuum prior to closure welding, and was then exposed to 1500°C to test for clogging. The vents did not clog and all internal surfaces were clean after test.

5.4.2.4 Particulate Flow Testing

A mock-up of a prime vent was fabricated and tested with fuel at MRC but helium flow degradation was experienced early in test. The decrease in flow was considered due to improper cleaning of Ir components prior to test. A new mock-up was made changing the 0.001 inch diameter opening in the hemisphere to 0.005 in. and all components were cleaned with HF and high-temperature (1500°C) vacuum. Testing at high temperature for 50 hours showed no change in Helium flow-rate. A total of 800 cm^3 of He was passed through the filter during this test, equivalent to roughly six-years of He generation for a PISA.

5.4.2.5 Destructive Evaluation

Metallographic examination of sections taken through MHFT-19 (Table 5.4.2-2) indicated that one of the vents was plugged. A similar examination of MHFT-28 showed that both of the vents were plugged. The vent condition was verified by applying He at 15 psig and observing for bubble formations at the vent region of the hemisphere while immersed in liquid. Metallographic examinations showed presence of ceramic material infiltrating and surrounding the Ir granules used to coat the vent filter disks. This material was primarily PuO_2 although some materials such as Cu, Si, and Al were also observed. It is significant that no vapor transport was observed.

5.4.2.6 Welding

In order to determine if the heat of closure welding is sufficient to release oxygen from the fuel, experiments were conducted at MRC using established welding procedures to close a PISA over a ThO_2 ball. Low melting point materials (Sn, Pb, Au, Al, etc.) were pinned to the ball in the region of the weld. Sectioning after welding proved that the weld-heat, at the surface of the sphere, did not exceed 700°F . It was demonstrated, therefore, that the heat of welding cannot cause a change in the stoichiometry of the fuel. GE calculations confirm that temperature rise due to welding would be no greater than 1000°F . It is concluded that closure welding is not a cause of vent hole plugging.

5.4.2.7 Vent Redesign

A number of vent redesigns were evaluated as possible back-ups for the prime vents reportedly being clogged during testing at LASL. Various designs considering staggered holes and slots were discarded when MRC demonstrated that promising concepts passed solid particles similar in size to particles found on MHW fuel surfaces.

A frit-type filter assembly, shown in Figure 5.4.2-2, did not pass solid particles when tested by MRC and appears to have sufficient passages to resist clogging in MHW environments. Two Ir disks and a frit-type powder compact comprise the assembly. The cover disk faces the fuel and acts as a shield to line-of-sight vapor deposits on the face of the frit. The weld disk faces the hemisphere and has a $1/8$ inch diameter hole, centrally located, to

Figure 5.4.2-2. MHW Vent Filter - Frit Type - Backup Design

pass helium through a 0.005 inch hole that is EDM-machined into the hemisphere. Since the weld disk is welded 360°, to the hemisphere, the internal gases are forced to travel through the frit (0.005 inch thick Ir powder) which is bonded at 1900°C to the cover and weld-disk components that complete the assembly. A fourth component, a weld washer (0.005 inches thick) was added to the system at the request of MRC. The washer is about 1/16 inch wide and rests on the weld washer to provide additional filler material during EB welding of the weld-disk to the ID of the hemisphere.

5.4.2.8 Reduction in Operating Temperature

Reduction in the fuel operating temperature will moderate the tendency of the fuel to produce vapor species which can migrate to and solidify in vent openings. To reduce the fuel temperature, the emissivity of the PISA surface has been increased. A grit blasting technique, using tungsten carbide shot, has been developed at MRC which permanently increases the sphere surface emissivity to around $E = 0.3$. This operation reduces the operating temperature of the fuel by 20°C minimum. In addition, the roughened surface of the Ir is subject

to discoloration when put in contact with the graphite impact absorbing member (GIS) of the system. Further enhancement of emissivity is anticipated as a result of this effect.

Additional reduction in fuel operating temperature is possible if a small helium back-pressure can be maintained within each fuel sphere, as well as in the gaps of the total heat source system. Pressure maintenance devices are currently being explored.

5.4.3 IMPACT TESTING

5.4.3.1 Simulant

A drop test of a bare, heated (2105°F) ThO₂ sphere (#587), from a height of 35 inches was conducted to serve as a comparison with the Mound, live fuel drop test. This simulant broke in a manner very similar to the fuel.

5.4.3.2 Impact Tests at GE-VF

Twenty-three impact tests, with simulant and a variety of Ir materials, were carried out at GE during the year. Data from these tests are summarized in Tables 5.4.3-1 thru 5.4.3-2.

The 14 impact tests summarized in Tables 5.4.3-1 through 5.4.3-3 were aimed at establishing the impact capability of the Ir compositional and fabrication variations exemplified in the E.I. PISA's used. As nearly as can be determined the characteristics of the simulant and of the GIS and the impact temperature were maintained without change from previous impact series (see bimonthly MHW reports 1972 and Jan./Feb. 1973). The results show:

1. Failure of Ir containment made with E-beam melted material and of especially high quality. Failure occurred with a range of deformation $\frac{D-Do}{Do}$ ranging from 5.8 to 7.4% and at all the impact velocities used, i.e., 319, 299, and 283 FPS.
2. Ir PISA's containing both moderate amounts of W, < 1000 PPM, and relatively large amounts of W, ~2500 PPM, survived impact velocities up to 319 FPS but failed at 317, 335, and 339 FPS.

Table 5.4.3-1. FSA Test Summaries - Impacted at - GE-VF (Bldg. 8)

DATE AND TEST NUMBER	IMPACT SPECIMEN DATA				IMPACT CONDITIONS				RESULTS
	GRAPHITE IMPACT SHELL No. Density = ρ - g-cm ⁻³ Thickness = t_c - inch Type	IRIDIUM CONTAINMENT: No. Thickness Heat treatment + date Other; Welded by:	SPHERE MATERIAL: Fuel: PuO ₂ Simulant: ThO ₂ No.; Dia.; % TD; Heat treat.; other.		Specimen weight total - grams	Velocity - fps	Energy - ft-lbs	Temperature - °F	
1 Mar. 73 # 709	861571 (HITCO) ρ = 1.21 t_c = 0.460 1972 graphite yarn	941/946 0.020" thk E-Beam melt ML butt weld	ThO ₂ GE-709 Hot pressed 76% TD; HT after weld		396.2	319	1390	2500	FAILED: H/D = not available S(%) = 6.8 GIS rotation = 20°
5 Mar 73 #708	861568 ρ = 1.21 t_c = 0.460 1972 graphite yarn	937/938 0.020" thk. E-beam melt ML/butt HT postweld	ThO ₂ GE-708 H/P 76% TD HT post-weld		394	299	1221	2500	FAILED: H/D = not available S(%) = 7.4 GIS rotation = 32°
6 Mar. 73 #707	861576 ρ = 1.20 t_c = 0.460 1972 graphite yarn	934/935 0.020 thk. E-beam melt ML/butt weld HT postweld	ThO ₂ GE-707 H/P 76% HT post-weld		395	283	1090	2500	FAILED: H/D = 1.180/1.710" S(%) = GIS rotation = 39° Showed "fingerprint"
13 Mar. 73 # 711	865681 ρ = 1.20 t_c = 0.460" 1968 graphite yarn	1021/1024 0.020" thk. Mod. W (1) ML/butt; RC Weld cracked Lot 74; ML	ThO ₂ GE-711 H/P 76% TD		393	283	1082	2500	PASSED: H/D = 1.201"/1.716" S(%) = 7.9 GIS rotation = 0° Weld crack existing before impact did not open up
20 Mar. 73 # 716	861577 ρ = 1.20 t_c = 0.460" 1972 graphite yarn	1069/1072 Mod. W add(1) Lot 73 ML/butt; RC	ThO ₂ GE-716 H/P 76% TD		393	294	1170	2500	PASSED: H/D = 1.172"/1.708" S(%) = 7.6 GIS rotation = 0°
21 Mar. 73 # 718	861564 ρ = 1.18 t_c = 0.460 1972 graphite yarn	1045/1041 Mod. W add(1) Lot 75 RC; ML/butt	ThO ₂ GE-718 H/P 76% TD		394	316	1360	2500	PASSED: H/D = 1.181"/1.718" S(%) = 7.9 GIS rotation = 0°

(1) MODERATE TUNGSTEN, W, = 1000 ppm

RC = Recrystallized before welding; ML = Mound Lab weld

Table 5.4.3-2. FSA Test Summaries - Impacted at - GE-VF (Bldg. 8)

DATE AND TEST NUMBER	IMPACT SPECIMEN DATA			IMPACT CONDITIONS				RESULTS
	GRAPHITE IMPACT SHELL No. Density = ρ - g-cm ⁻³ Thickness = t_c - inch Type	IRIDIUM CONTAINMENT: No. Thickness Heat treatment + date Other, welded by:	SPHERE MATERIAL: Fuel: PuO_2 Simulant: ThO_2 No.; D.A.; Z ID; Heat treat.; other	Specimen weight total - grams	Velocity - fps	Energy - ft-lbs	Temperature - °F	
22 Mar.73 # 717	861627 ρ = 1.18 t_c = 0.460" 1968 graphite yarn	1091/1092 0.020" thk. High W add. (2) Lot 80 RC; ML/butt	ThO_2 GE-717 H/P 76% TD no HT in Ir shell	391	315	1350	2500	PASSED: H/D = 1.101"/1.790 S(%) = 12.9 GIS rotation = 56°
27 Mar.73 # 719	861260 ρ = 1.17 t_c = 0.460" thk. 1968 graphite yarn	1094/1095 0.020" thk. High W (2) Lot 80; RC ML/butt	ThO_2 GE-719 H/P; 76% TD no HT in Ir	390	335	1520	2500	FAILED: H/D = 0.999"/1.897 S(%) = 19.5 GIS rotation = 99°
29 Mar.73 # 720	861631 ρ = 1.19 t_c = 0.460" 1968 graphite yarn	1099/1093 0.020" thk. High W (2) Lot 80;; RC ML/butt	ThO_2 GE-720 H/P; 76% TD no HT in Ir shell	393	317	1360	2500	PASSED: H/D = 1.080"/1.771 S(%) = 11.6 GIS rotation = 0°
30 Mar.73 # 721	861562 ρ = 1.20 t_c = 0.460" 1972 (?) yarn	1096/1097 0.020" thk. Hi W add(2) Lot 80; RC ML/butt	ThO_2 GE-721 H/P; 76% TD no HT in Ir shell	393	339	1560	2500	FAILED: H/D = 1.092"/1.796" S(%) = 12.4 GIS rotation = 68° Weld failure; poor weld before impact
3 Apr.73 #1 Broken	865684 ρ = 1.19 t_c = 0.460" 1968 graphite yarn	1098/1100 0.020" thk. Hi W add(2) Lot 80; RC ML/butt	ThO_2 sphere sectioned into 8 equal segments H/P; 76% TD no HT in Ir	392	314	1330	2500	FAILED: H/D = 1.015"/1.847" S(%) = 16.4 GIS rotation = 80° Poor weld; failed on impact
12 Apr.73 #2 Broken	868128 ρ = 1.20 t_c = 0.460" 1972 graphite yarn	1038/1039 0.020" thk Mod. W add(1) Lot 75; RC ML/butt	ThO_2 sphere sectioned into 8 equal segments no HT in Ir	392	282	1075	2500	PASSED: H/D = 1.109"/1.713" S(%) = 7.8 GIS rotation = 7°

(1) Moderate W content ≤ 1000 ppm (2) High W addition = 2000 ppm MIN 4000 ppm MAX RC= recrystallized before welding

Table 5.4.3-3. FSA Test Summaries - Impacted at - GE-VF (Bldg. 8)

DATE AND TEST NUMBER	IMPACT SPECIMEN DATA				IMPACT CONDITIONS				RESULTS
	GRAPHITE IMPACT SHELL No. Density = ρ - g-cm ⁻³ Thickness = t_c - inch Type	TRIDIUM CONTAINMENT: No. Thickness Heat treatment + date Other; Welded by:	SPHERE MATERIAL: Fuel: PuO ₂ Simulant: ThO ₂ No.; Dia.; % TD; Heat treat.; other.		Specimen weight total - grams	Velocity - fps	Energy - ft-lbs	Temperature - °F	
13 Apr.73 # 713	868125 ρ = 1.20 t_c = 0.460" 1972 graphite yarn	1103/1104 0.020" thk. H1 W (2) Lot 80; ML/b HT post-weld	ThO ₂ GE-713 H/P; 76% TD HT in Ir post-weld		392	317	1360	2500	FAILED: H/D = not available S(Z) = 7.6 GIS rotation = 15° Showed "fingerprint"; breach associated with secondary impact
28 Apr.73 # 712	868122 ρ = 1.20 t_c = 0.460" 1972 graphite yarn	1020/1026 0.020" thk Mod. W (1) Lot 74; RC With vents ML/butt	ThO ₂ GE-712 H/P; 76% TD no HT in Ir shell		392	303	1269	2500	PASSED: H/D = 1.141"/1.730" S(Z) = 9.0 GIS rotation = 25°

- (1) Moderate W content = 1000 ppm
 (2) High W addition = 2000 ppm MIN. = 4000 MAX.
 RC = recrystallized prior to welding
 ML/b = Mound Lab. - butt weld

Table 5.4.3-4. FSA Test Summaries - Impacted at - GE-VF(Bldg. 8)

DATE AND TEST NUMBER	IMPACT SPECIMEN DATA			IMPACT CONDITIONS				RESULTS
	GRAPHITE IMPACT SHELL No. Density = ρ - g-cm ⁻³ Thickness = t_c - inch Type	IRIDIUM CONTAINMENT: No. Thickness Heat treatment + date Other; Welded by:	SPHERE MATERIAL: Fuel: PuO ₂ Simulant: ThO ₂ No.; Dia.; Z ID; Heat treat.; other	Specimen weight total - grams	Velocity - fps	Energy - ft-lbs	Temperature - °F	
13 Jun.73 MRC #1	868126 = 1.20 t_c = 0.447 1972 graphite yarn	WC1-2/WC1-1 ML weld;step 0.025/0.025 18hr.@1500 before weld	ThO ₂ :GE601 H/P; 76%TD not heated with Ir	398.6	305	1273	2500	FAILED: H/D = 1.209/1.746 S(%) = 9.7 GIS rotation = 0° Weld failed; large crack in parent metal;some secondary impact deformation
13 Jun.73 MRC #2	868132 = 1.22 t_c = 0.445 1972 graphite yarn	WC3-2/WC3-3 ML weld;step 0.024/0.025 18hr.@1500C before weld	ThO ₂ :GE602 H/P; 76%TD not heated with Ir	400.4	303	1262	2500	FAILED: H/D = 1.203/1.707 S(%) = 7.3 GIS rotation = 10° Weld cracks perpendicular to the weld; some secondary impact deformation. Weld crack noted
28 Jun.73 MRC #3	868124 = 1.20 t_c = 0.451 1972 graphite yarn	WC4-2/WC6-1 ML weld;step 0.024/0.025 18hr.@1500 C before weld	ThO ₂ :GE-605 H/P; 76%TD not heat treated with Ir	397.9	300		2500	FAILED: H/D = 1.153/1.742 S(%) = 9.1 GIS rotation = 35° Weld failure; parent metal not cracked
29 Jun.73 MRC #4	868134 = 1.20 t_c = 0.450. 1972 graphite yarn	WC6-2/WC7-1 ML weld;step	ThO ₂ :GE-606 H/P; 76%TD not heat treated with Ir	399.0	300		2500	FAILED: H/D = 1.283/1.732 S(%) = 9.0 GIS rotation = 20° Weld failure + large breach in parent metal Specimen was heated to 2500F;aborted; refitted with new GIS and impacted.

Table 5.4.3-5. FSA Test Summaries - Impacted at GE/VF (Building 8)

DATE AND TEST NUMBER	IMPACT SPECIMEN DATA			IMPACT CONDITIONS				RESULTS
	GRAPHITE IMPACT SHELL No. Density = ρ - g-cm ⁻³ Thickness = t_c - inch Type	IRIDIUM CONTAINMENT: No. Thickness Heat treatment + date Other; Welded by:	SPHERE MATERIAL: Fuel: PuO ₂ Simulant: ThO ₂ No.; Dia.; % TD; Heat treat.; other.	Specimen weight total - grams	Velocity - fps	Energy - ft-lbs	Temperature - °F	
1 Aug. 73 No. 629 MRC #5	875999-2 HITCO ρ = t_c = 0.460" 1972 graphite yarn	WC15-2/WC15-3 ML weld HT 18 hrs @ 1500C before weld	ThO ₂ :GE629 h/p: 76% TD not HT		296		2500	PASSED : H/D = S(%) = 8.25 GIS rotation = 0° Good hit
8 Aug. 73 # 631 MRC #6	875995-1 HITCO ρ = t_c = 1972 graphite yarn	EB-WC17-3/ EB-WC16-1 ML weld HT 18 hrs @ 1500 C before weld	ThO ₂ :GE631 h/p: 76% TD not HT with Ir		310		2500	FAILED: H/D = 1.11/2.08 S(%) = GIS struck gun was stripped off before impact; PISA struck granite bare
14 Aug. 73 #630 MRC #7	875995-2 ρ = t_c = 1972 graphite yarn	EB-WC14-3/ EB-WC15-1 ML weld 18hrs @1500C before weld	TnO ₂ :GE630 h/p:76% TD not HT with Ir		303		2500	FAILED: H/D = S(%) = GIS rotation = 0° Fingerprint failure
15 Aug. 73 # 636 MRC #8	875994-2 ρ = t_c = 1972 graphite yarn	EB-WC9-2/ EB-WC9-1 ML weld 18 hrs@1500C before weld	ThO ₂ :GE636 h/p: 76% TD not HT with Ir		291		2500	FAILED: H/D = not available S(%) = not available GIS rotation = 90° Ir failure catastrophic
28 Aug. 73 # 637 MRC #9	875978-2 ρ = t_c = 1972 graphite yarn	EB-WC18-2/ EB-WC18-3 ML weld 18hrs@1500C before weld	ThO ₂ :GE637 h/p 76% TD not HT with Ir		291		2500	FAILED: H/D = not available S(%) = not available GIS rotation = 110° Catastrophic failure; PISA broke into 2 pieces, signs of fingerprint

Three failures (out of 9 impacts) showed $\frac{D-D_0}{D_0}$ respectively of 12.4% at 339 FPS, 7.6% at 317 FPS, and 19.5% at 335 FPS. (It should be noted that a success occurred with a deformation of 12.9% at 315 FPS).

3. The capability of the Ir to survive impact, with simulant broken before impact, was tested with two units, one of which failed at 314 FPS with a deformation of 16.4%. The second survived at 282 FPS with a deformation of 7.8%.

Results from the impaction of 9 ORNL-ML/MRC PISA's are summarized in Tables 5.4.3-4 and -5. There were no operational problems associated with 6 of the impacts and 5 out of the 6 resulted in the production of breaches in the Ir base material and welds. Rotation of 90° or more occurred with 2 of the shots and the graphite impact absorbing material (GIS) was stripped from the simulant prior to the impaction of another PISA. Failures in the Ir occurred in all 3 instances.

5.4.4 IMPACTS AT LASL

In January, a matched pair of 285 fps impacts (fueled and simulant) at 2500°F, with re-entry, were done at LASL. The results indicate both breached badly. The impact can has been identified as a source of major disparity between GE and LASL impacts. However, the simulant appeared comparable to the fuel in that it distorts the iridium at least as much (or more) than the fuel.

Results of two additional impacts are shown in Table 5.4.4-1. The tests were intended to be substantially similar in all respects except that No. 561 was contained in an outer can of 0.005" thick tantalum with an unwelded cover and No. 564 was contained in an outer can of 0.020" thick tantalum with a welded cover. The use of the outer can constitutes a major difference between the GE and LASL impact systems.

The detailed results of these tests have been reported by LASL. These results appear to confirm the GE position that the crumpling of the heavy-walled can forms a very strong and unyielding circumferential containment which restrains the graphite of the GIS. This produces an artificially decreased stroke length for the decelerating PISA and forces the cored-out graphite piece back into the PISA like a punch in a die body. An additional complication

appears to be the very short rebound onto the sabot. LASL plans to investigate this problem further by installing flash X-ray equipment for monitoring of impact.

A summary of four simulant impacts utilizing ORNL-ML/MRC Ir is shown in Table 5.4.4-2. LASL conducted these tests to indicate the influence of the Ta can used by LASL as an operational safety requirement on the fueled tests. In these tests two FSA's were impacted without the can. The Ta can modifies the impact behavior, but whether it increases or decreases the severity of the Ir response is still subject to discussion. The Ir in all 4 spheres was breached.

Results of impacting six fueled PISA's are summarized in Tables 5.4.4-3 and -4. All specimens used E.I. hardware. Three of the PISA's came through with no failure but the Ir in the remaining three PISA's was breached.

Table 5.4.4-1. FSA Test Summaries - Impacted at - LASL

DATE AND TEST NUMBER	IMPACT SPECIMEN DATA				IMPACT CONDITIONS				RESULTS
	GRAPHITE IMPACT SHELL No. Density = ρ - g-cm ⁻³ Thickness = t_c - inch Type	IRIDIUM CONTAINMENT: No. Thickness Heat treatment + date Other; welded by:	SPHERE MATERIAL: Fuel: PuO ₂ Simulant: ThO ₂ No.; Dia.; % TD; Heat treat.; other		Specimen weight total - grams	Velocity - fps	Energy - ft-lbs	Temperature - °F	
1 Mar. 73 # 564	861573 = 1.20 t_c = 0.450" Cap - 861515-2	555/558 0.020" thk. Lot 37 RC; Evendale step-joint	ThO ₂ GE-564 H/P; 76% TD HT in Ir		n/a	290		2500	FAILED: H/D = 1.184"/1.673" S(Z) = 4.5 GIS rotation = 0°(?) Two fractures, one rupture in strain area "Fingerprint" on impact face Enclosed in 0.020" thick Ta can with welded closure. Graphite plug from GIS formed on impact surface and remained in place when can was disassembled. Can did <u>not</u> rupture.
5 Mar. 73 # 561	861572 = 1.20 t_c = 0.455" Cap 861515-2	559/560 0.020" thk. Lot 37 RC; MRC/ML step-joint	ThO ₂ GE-561 H/P; 76% TD; HT in Ir		n/a	287		2500	PASSED: H/D = 1.154"/1.717" S(Z) = 6.6 GIS rotation = 0°(?) Enclosed in 0.005" thick Ta can with unwelded closure. Graphite split and extensively fractured. No "fingerprint".

Table 5.4.4-2. Impact Tests at LASL (Simulant)

DATE AND TEST NUMBER	IMPACT SPECIMEN DATA				IMPACT CONDITIONS				RESULTS
	GRAPHITE IMPACT SHELL No. Density = ρ - g-cm ⁻³ Thickness = t_c - inch Type	IRIDIUM CONTAINMENT: No. Thickness Heat treatment + date Other, Welded by:	SPHERE MATERIAL: Fuel: PuO ₂ Simulant: ThO ₂ No.: Dia.: % TD: Heat treat.: other:		Specimen weight total - grams	Velocity - f/s	Energy - ft-lbs	Temperature - °C (°F)	
17 DEC.73 MHT-60	876003-2 $\rho = 1.19$ $t_c = 0.46$	WC-35-3 WC-37-1 Butt weld by ML. H/T ? Grit blast ?	ThO ₂ : ML-53 77% TD; 1.46 dia. Not X-rayed	?	280			1370 (2500)	FAILED: H/D = 1.22/1.67 S(Z) = 4.7 GIS rotation = ° Impacted in 0.030 thk. Ta can. ORNL-ML/MRC Ir. Fingerprint failure plus hoop stress cracks.
19 Dec.73 MHT-61	875990-2 $\rho = 1.19$ $t_c = 0.46$	WC-36-3 WC-38-1 Butt weld by ML. H/T ? Grit blast ?	ThO ₂ : ML-54 77%TD; 1.46" dia. Not x-rayed	?	283			1370 (2500)	FAILED: H/D = 1.17/1.71 S(Z) = 7.6 GIS rotation = ° Hoop stress cracks, no fingerprint. Impacted w/o can. ORNL-ML/MRC Ir
20 Dec.73 MHT-62	876002-1 $\rho = 1.19$ $t_c = 0.46$	WC-37-2 WC-40-1 Butt weld by ML. H/T ? Grit blast ?	ThO ₂ : ML-55 77%TD; 1.46" dia. not x-rayed		268			1370 (2500)	FAILED: H/D = 1.22/1.67 S(Z) = 4.7 GIS rotation = ° Impacted in 0.030"thk Ta can. ORNL-ML/MRC Ir. Fingerprint failure plus hoop stress cracks.
26 Dec.73 MHT-63	876001-2 $\rho = 1.19$ $t_c = 0.46$	WC-39-1 WC-39-2 Butt weld by ML. H/T ? Grit blast ?	ThO ₂ : ML-56 77%TD; 1.46" dia. Not x-rayed		232			1370 (2500)	FAILED: H/D = 1.18/1.71 S(Z) = 6.9 GIS rotation = ° Impacted w/o can. ORNL-ML/MRC Ir. Hoop stress cracks; no fingerprint.

Table 5.4.4-3. FSA Test Summaries - Impacted at LASL

DATE AND TEST NUMBER	IMPACT SPECIMEN DATA			IMPACT CONDITIONS				RESULTS
	GRAPHITE IMPACT SHELL No. Density = ρ - g-cm ⁻³ Thickness = t_c - inch Type	IRIDIUM CONTAINMENT: No. Thickness Heat treatment + date Other; Welded by:	SPHERE MATERIAL: Fuel: PuO ₂ Simulant: ThO ₂ No.; Dia.; Z ID; Heat treat.; other	Specimen weight total - grams	Velocity - fps	Energy - ft-lbs	Temperature - °F	
5 Jul.73 MHFT-32	868131 ρ = 1.20 t_c = 0.46 1972 graphite yarn	1112/1118 High W Ir Lot 80 ML weld	PuO ₂ ; ML SPO-61 Cracked but intact prior to impact	253.9	284.5		2500 1370C	PASSED: H/D = 1.175/1.739 S(%) = GIS rotation = 0° Impacted in 0.030" thick Ta can
10 Jul.73 MHFT-33	868133 ρ = 1.20 t_c = 0.46	SN1107/1114 High W Ir Lot 80 ML weld	PuO ₂ ML - SPO-62 Cracked but intact prior to impact	253.8	281.9		2500 1370C	PASSED: H/D = 1.144/1.717 S(%) = GIS rotation = 0° Impacted in 0.030" thick Ta can; vents did not pass He bubbles post-impact
27 Aug.73 MHFT-26	861563 ρ = 1.20 t_c = 0.46	GN350/830 Lot 50/51 ML weld Aged 867 hr @ 1450C before impact	PuO ₂ ML - SPO-51 Cracked but intact prior to impact	253.9	246.5		2500 1370C	PASSED: H/D = S(%) = GIS rotation = 90° Impacted in 0.030" thk. Ta can
30 Aug.73 MHFT-27	861574 ρ = 1.20 t_c = 0.46 1972 graphite yarn	GN341/831 Lot 50 ML weld; Aged 867 hr @ 1450C before impact	PuO ₂ ML - SPO-52 Sphere - broken prior to impact	253.9	249.0		2500 1370C	FAILED: H/D = na S(%) = na GIS rotation = na Ir conformed to broken fuel; very small tear at bottom of crease

Table 5.4.4-4. Impact Tests at LASL- (Fuel)

DATE AND TEST NUMBER	IMPACT SPECIMEN DATA			IMPACT CONDITIONS				RESULTS
	GRANITE IMPACT SHELL No. Density = ρ - g-cm ⁻³ Thickness = t_c - inch Type	IRIDIUM CONTAINMENT: No. Thickness Heat treatment + date Other; Welded by:	SPHERE MATERIAL: Fuel: PuO ₂ Simulant: ThO ₂ No.; Dia.; Z TD; Heat treat.; other	Specimen weight total - grams	Velocity - fps	Energy - ft-lbs	Temperature - °C (°F)	
13 Nov.73 MHFT-30	868127 $\rho = 1.2$ $t_c = 0.46$	SN848/853; lots 51/52 EI; 847hrs @ 1440 C; ML weld; step SST specimen 5 min @ 1850C	PuO ₂ ; SPO-58 847 hrs @ 1440C + 5 min. @ 1850C		250		1550	FAILED: H/D = 1.21/1.67 S(Z) = 5 GIS rotation = 0° Impacted in 0.030" thk. Ta can; given reentry heat pulse and cooled before impact; 'fingerprint' and hoop stress cracks
15 Nov.73 MHFT-31	868129 $\rho = 1.19$ $t_c = 0.46$	SN695/690; Lots EI 44/44; 847Hr. @ 1440C; ML step weld; 5 min @ 1850 C SST specimen	PuO ₂ ; SPO-60 847 hrs @ 1440 C + 5min. @ 1850C		243		1550	FAILED: H/D = 1.23/1.66 S(Z) = 4.6 GIS rotation = 0° Impacted in 0.030" thk. Ta can; given reentry heat pulse and cooled to 1550C before impact; hoop stress cracks

5.5 FUEL SPHERE ASSEMBLY (FSA)

5.5.1 DESCRIPTION

The FSA consists of three components assembled, one inside the other. The innermost component is the fuel sphere. The fuel sphere is plutonium dioxide (see Section 5.3) and is inserted into the protective iridium shell (PICS) (see Section 5.4). The iridium shell in turn is inserted into a hollow spherical impact shell composed of a carbon-carbon composite.

Figure 5.5-1 shows an exploded view of the FSA.

Figure 5.5-1. Fuel Sphere Assembly Exploded View

5.5.1.1 Impact Shell Function

The impact shell surrounds the PISA and is essentially a crush-up structure which reduces the deceleration forces during earth impact such that the PISA can survive intact. Thus, the impact shell is designed to protect the PISA during an impact at the heat source's terminal velocity (plus a 10 percent margin for uncertainties). The assembly of the PISA and its impact shell is referred to as a Fuel Sphere Assembly (FSA).

Figure 5.5-2. Impact Shell Dimensions

5.5.1.2 Design

The design and fabrication of the fuel and iridium (PICS) are described in Sections 5.3 and 5.4, respectively. This section will be restricted to the impact shell exclusively.

The impact shell is a carbon-carbon composite composed of two parts, a spherical body and cap. The outside diameter is 2.554 ± 0.010 inches. The inside diameter is 1.634 ± 0.005 inches (0.460 inch thick). The inside diameter is dimensioned such that the maximum circumferential weld diameter of the PICS has 0.005 inch clearance at 1400°C (2552°F) (operating temperatures) (see Figure 5.5-2). At the maximum re-entry temperature of 1870°C , there is a diameter clearance of 0.006 inch between the PICS and the shell at all locations remote from the weld and 0.0012 inch interference at the weld zone line. A summary of clearances is given in Table 5.5-1. After the PISA is inserted into the body at temperatures between 1100 to 1400°F , the cap is screwed down and locked with the index hole aligned.

During the reporting period, two experimental graphite impact shell models were fabricated for testing. One features a hoop-wound band designed to maintain the spherical configuration near the cap area during impact. The second model features a densification of the T-50 yarn at the outermost surface, obtained by feeding on two yarns for the outer 0.05 inches of surface winding. These two experimental models will be impact tested early in 1974.

5.5.1.3 Fabrication

All impact shells are fabricated using Thornel T-50 yarn as the base material. The yarn is ordered from Union Carbide per their specification for Thornel-50 (wyg 130 - 1/2). The yarn is randomly wound into spheres on a graphite mandrel using a golf ball winding machine (see Figure 5.5-3). During winding the yarn is continually impregnated with SC-1008 resin per MIL-R-9299. After winding and initial set, the spheres are alternately impregnated and cured through several cycles until a density of 1.2 gms/cc is obtained. The shell is then cured and final pyrolyzed at 2100°C . The part is then machined to size and inspected.

Table 5.5-1. Clearance Between Iridium PICS and Graphite Impact Shell

Condition		R.T.	2500°F	3300°F	Assembly (11-1400°F)
Min Clearance	Ir dia	1.604	1.6219	1.630	1.6128
	T-50 dia	1.634	1.6347	1.6357	1.634
	Clearance	0.030	0.0123	0.0057	0.0211
Max. Clearance	Ir dia	1.576	1.594	1.6013	1.5826
	T-50 dia	1.639	1.6347	1.6403	1.639
	Clearance	0.063	0.041	0.0395	0.0564
Normal Clearance	Ir dia	1.59	1.6077	1.6156	1.5977
	T-50 dia	1.634	1.6347	1.6357	1.634
	Clearance	0.044	0.027	0.02	0.0363
Minimum Clearance Near Weld	Ir dia	1.611	1.6289	1.6369	1.6196
	T-50 dia	1.634	1.6347	1.6357	1.634
	Clearance	0.023	0.0057	0.0012*	0.0141
Nominal Clearance Near Weld	Ir dia	1.60	1.6178	1.6258	1.6078
	T-50 dia	1.634	1.6347	1.6357	1.634
	Clearance	0.034	0.0169	0.0099	0.0262

* Interference

Figure 5.5-3. Impact Shell Random Winding Device

5.5.1.4 Main Points of Specification

The specification Dwg No. NS0060-03-49 defines all base materials, manufacturing procedures, test methods, final parts, requirements and shipping procedures. Recent changes to the specification have been made to predict product quality. These changes include chemical purity (see Table 5.5-2), compression and shear strength, limits, and strain rate requirements. In addition, a lot of impact bodies and caps was defined as being composed of a maximum of 30 finished bodies and caps identified and maintained throughout the entire manufacturing process. The sample size required for testing for different lot sizes is defined as:

<u>Number of finished impact bodies and caps processed in a lot</u>	<u>Number of randomly selected samples required for testing per lot</u>
1 to 5	1
6 to 15	2
16 to 30	3

Chemical purity will eliminate elements which may be detrimental to the iridium (PICS). The physical requirements were imposed because tests show a relationship between these properties and impact capability. The compressive strength, a strain to failure rate are indicative of energy absorption capability. The shear stress levels indicate the degree of bonding between the random fibers. Specimens with shear stress levels between 200 to 600 psi combined with strain rates above 22 percent with minimum compressive strength of 13,000 are considered of optimum quality.

Table 5.5-2. Graphite Impact Shell Final Product Purity

Element	Acceptable Level (PPM)
Phosphorus	100
Fe, Ca, Si	50 each
Cl, S, Al	30 each
Na, K, Mg, Sr, Ba, Ti, V, Mn B, Cr, Cu, Mo, Co, Ni, Zn, Pb	10 each

All material in the finished body and cap shall be no less than 99.94 percent carbon. Maximum levels of impurities measured by weight in PPM shall be as shown above.

5.6 FSA INTEGRATION HARDWARE

There are three FSA retaining tray assemblies within each heat-source assembly. Each tray assembly has two planes of FSA's, four FSA's to a plane, with each adjacent, plane rotated 45 degrees. The eight FSA's within each tray are held together by two sphere locks and a tie-bolt. Compliance pads between trays and graphite springs provide flexibility for the distribution of 500 to 800 pounds of pre-load force exerted at the time of assembly. Last year's report describes the function of each part in detail, while Figures 5.6-1 through 5.6-3 show the general arrangement and configuration of the required hardware.

The assembly of the top end cap to the aeroshell body, with the fuel inventory of 24 simulant FSA's was accomplished several times at room temperature in an ambient air atmosphere. However, the end-cap could not be screwed down to seat when the assembly was attempted during a "walk-thru" exercise at ML/MRC. Upon investigation, it was determined that mechanical interference between adjacent sphere locks occurred at temperature and the design was revised to stipulate shorter components. In addition, the possibility of an unanticipated, large increase in friction between threaded graphite surfaces, due to the dry helium atmosphere (dew point $< -40^{\circ}\text{C}$) in the assembly line was considered a possible detriment during HS assembly operations. The problem was solved by assembling the end-cap with a rod extending through a hole in the top end-cap and compressing the top compliance spring sufficiently to prevent it from exerting its force on the end-cap during the seating operation. The hole in the end-cap is plugged after assembly is completed.

No other major changes in FSA integration hardware were made during this report year, but minor modifications were made to component specifications altering the type of analytical techniques used to determine product purity.

Figure 5.6-1. Retaining Tray Assembly

- APPLIES PRELOAD ON FSA's IN HSA
- COMPENSATES FOR THERMAL GROWTH OF HSA HARDWARE
- COMPENSATES FOR TOLERANCE STACKUP IN BOTH AXIAL AND RADIAL DIRECTION

1. SEATING PLATE
2. CRUSH-UP DISKS (SPRINGS)
3. BOLT-END CRUSH-UP
4. NUT

Figure 5.6-2. Tap End Spring Crushup

Figure 5.6-3. Bottom Tray and Laminated Spring Assembly

5.7 AEROSHELL

The aeroshell is an assembly consisting of an ablator cylinder which has two threaded end caps. The aeroshell is the basic structural member of the heat source assembly and contains all of the other heat source components with the exception of the iridium outer clad and emissivity sleeve. The ablator cylinder is designed to maintain the structural integrity of the heat source from assembly through powered flight and re-entry to the moment of impact. The ablator cylinder also provides the heat source aerodynamic shape and thus defines the re-entry heating, ablation, terminal velocity, etc. The aeroshell assembly is shown with the retaining assemblies in Figure 5.7-1. The ablator cylinder is machined from a solid, hot-pressed graphite billet, while the end-caps are machined from hot-pressed graphite plates. The ablator material is a polycrystalline graphite AXF-5Q. The inside and outside diameters are $6.356 \pm .003$ and $7.110 \pm .003$ inches, respectively. The overall assembled length (with end caps) is $16.420 \pm .008$ inches.

Figure 5.7-1. Aeroshell Assembly

The critical mechanical properties of ultimate strength and modulus of elasticity and purification have specified levels for flight hardware acceptance. Ultrasonic inspections are also used to eliminate hardware that has large defects or voids and hence unacceptable mechanical properties. The mechanical and chemical specification requirements are given in Table 5.7-1 and Table 5.7-2, respectively.

Table 5.7-1. Mechanical Properties

A. DENSITY (R.T.)

AEROSHELL - ONE TEST - 1.76 g/cc MIN

PLATE - ONE TEST - 1.79 - 1.83 g/cc CLASS I

- 1.85 g/cc CLASS II

B. MINIMUM STRENGTH AT ROOM TEMPERATURE

TENSION 7250 PSI - BILLET - 3 TEST SAMPLES

TENSION 8500 PSI

COMPRESSION 17500 PSI

} CLASS II PLATE - 2 TEST SAMPLES EACH

C. MAXIMUM MODULUS OF ELASTICITY

TENSION - 1.75×10^6 - BILLET, 1.90×10^6 CLASS II PLATE

COMPRESSION - 2.00×10^6 - CLASS II PLATE

DATA TAKEN FROM TEST SAMPLES OF SECTION B

D. MINIMUM STRENGTH AT ELEVATED TEMPERATURE

TENSION AT 3000°F - 9400 psi - BILLET

3 TEST SAMPLES

COMPRESSION AT 3500°F - 16,500 psi BILLET

17,500 psi CLASS I PLATE

2 TEST SAMPLES EACH

E. MAXIMUM MODULUS OF ELASTICITY AT ELEVATED TEMP.

TENSION AT 3000°F 2.10×10^6

COMPRESSION AT 3500°F 2.15×10^6

DATA TAKEN FROM TEST SAMPLES OF SEC. D.

Table 5.7-2. AXF-50 Impurity Limits

ELEMENTS	ACCEPTABLE LIMIT (MAX PPM)*
Cl, Na, K, P, S	50 EACH
Si, Al, Mg, Ni, Cr, Cu B, Mn, Li, Cd, Mo, Pb, Zn	100 EACH
Fe, V, Ca, Ti	200 EACH

* The sum total of all listed impurities shall not exceed 900 PPM. Testing for unlisted impurities is not required for conformance to this specification

The following two design changes were made during the reporting period: (1) a hole was machined into one end-cap to allow insertion of a rod to compress the compliance springs, thus reducing the torque required to seat the end-cap; (2) all thread roots in the cylinder and end-caps were rounded to decrease stress concentrations. These changes were incorporated in Q-1 and subsequent heat sources.

5.8 OUTER CLAD

5.8.1 DESIGN

The design objectives of the MHW Ir clad are covered in the 1972 Annual Report. In brief, the outer clad isolates the heat source from the converter and allows maintenance of a He back pressure which promotes heat flow by improving conductivity in voids. In addition, the clad permits safe handling of a fueled source and protects the POCO reentry shell from attack by short time exposure to high temperature air.

The cylinder is shipped open-ended to ML-MRC, where the bottom lid is welded into the cylinder in the absence of fuel. After fueling, the top end-cap is welded into place.

The outer clad is fabricated from many piece parts (Figure 5.8-1) but is comprised of 3 basic components: top end-cap, cylinder, and bottom end-cap. A brief description of the type and function of individual piece parts and basic components follows.

5.8.1.1 Cylinder

The overall length of the iridium cylinder is 16.610 inches long with an outside diameter of 7.197 inches. It is composed of a cylinder 15.410 inches long with a wall thickness of 0.010 inches to which two short cylindrical rings are welded, one at each end. The cylindrical end rings are 0.600 inches long with a wall thickness of 0.030 inches. Thus, when assembled, a 0.020 inch step exists between the inside diameter of the long cylinder and inside diameter of the two end rings.

Figure 5.8-1. Iridium Clad Parts in Fabrication

The long cylinder is composed of 3 or 4 longitudinal segments, TIG butt welded together. The two end rings are then joined to the cylinder with circumferential TIG welds. Three lifting lugs are EB welded to the outside diameter of the top end ring, 120° apart.

5.8.1.2 Bottom End Cap

The end cap is composed of the following three elements welded together (Figure 5.8-1): (1) A flat washer-shaped outer flange with the outside edge rolled 90° to provide a 1/16 flange to match the inside diameter of the cylinder; (2) A ring having a 1/8 inch square cross-section; (3) A flat disk to complete the end enclosure. The inside surface of the end-cap has 2 torque tabs welded to it, approximately 180° apart, which fall into recesses provided in the bottom POCO end cap and permit the graphite spring to be torqued to its correct pressure (500 to 800 ft.-lbs.) without slipping. Guide segments are welded to the outside surface of the end-cap to permit the assembly to be indexed in the ML/MRC assembly fixture, thus preventing rotation during assembly of the top POCO end-cap. The ring portion of the end-cap provides a recess to support the converter. The outside diameter of the ring fits into a similar recess in the POCO end cap and prevents relative motion between components due to powered flight vibrations.

5.8.1.3 Top End-Cap

The top end cap is similar to the bottom end cap except that a vent tube and vent-tube bracket are welded to the face of the recessed portion of the cap as shown in Figure 5.8-2. The tube is approximately 16 1/2 inches long and its outboard end is attached to the converter feed-thru fitting. The tube is constructed from 3 segments along its length, utilizing different materials. Starting from the inboard side these segments are as follows: (1) A 2-inch length of iridium tubing, (2) A 12-inch length of Pt 10 Rh tubing and (3) A 2 1/2-inch length of 304 stainless steel tubing. All three segments are joined by TIG butt welds.

Figure 5.8-2. Schematic of Gas Management Subsystem

5.8.2 FABRICATION

1. Clad No. 1

This clad was designated as a fabrication and weld development item; information obtained would prove design feasibility and establish procedures and specifications guiding the operations and quality of subsequent clads. No usable end-caps resulted from this activity because of cracking and distortion problems encountered along the way. The finished cylinder was used to develop nondestructive testing capability and sections taken from it were bend tested to obtain information concerning the handability of structure at ML/MRC. It was established that the welds behaved adequately to permit fuel loading at MRC and subsequent handling at GE.

2. Clad No. 2

This clad had some development problems which prevented its completion. Radiography indicated that the welded seams were porous and end-caps could not be produced that mated properly with the cylinder; gaps as wide as 0.020 inch caused weld cracking during closure welding at MRC. However, a practice end-ring to simulated end-cap weld was completed successfully by MRC.

It was during this interval of time that EI became aware of a raw material acquisition problem. There was insufficient virgin powder available for completion of the end-products required for the Program. GE accepted the use of melted solids for ingot fabrication, but EI soon demonstrated that remelting resulted in incremental build-up of contaminants that produced cracking of the finished items. EI then received permission to EB melt solid material so as to partially eliminate some of the lighter materials that had been accumulating in the initial material. Arc-melting was instituted, after EB melting, to improve homogenization.

3. Clad No. 3

This clad was scheduled to house an electric heater to permit creep and pressure burst-testing at GE. MRC was unable to complete the welding due to gapping between end-cap and cylinder ranging as high as 0.020 inch. The piece-parts were returned to GE for storage. It appeared that distortion of the end-caps was due to the welded construction. Attempts to straighten the warped lids at room temperature resulted in cracking of the welds. Straightening at elevated temperature provided some improvement but distortion during cooling resulted in a cap that was dimensionally unacceptable for cylinder closure welding.

Joining of end-cap components by EB, rather than TIG, welding reduced total distortion such that the design was considered feasible. EB welding of guide segments and torque-tabs did not add to overall distortion problems and the resulting joints resisted 120 ft/lb torques without failure.

A simulation pressure burst test of a low-carbon steel iridium "can" was conducted during this stage of the development. The test was conducted with the end-caps restrained, simulating actual converter end conditions. The burst pressure was 205 psi. Strain gages indicated creep commencing at about 185 psi. Failure occurred at the circumferential weld between the end-ring and the 0.010 inch thick cylinder and not at the closure joint between the end cap and cylinder.

4. Clad No. 4

The same type of fabrication problems were experienced with Clad No. 4 as with Clad No. 3. The hardware was considered acceptable for MRC weld closure development. Sections taken through the weld showed acceptable weld penetrations but poor uniformity in the thickness of the end cap joint material.

5. Clad No. 5

This clad was scheduled for in-house pressure burst testing using an electric heat-source to supply the required temperature. Although a new improved fixture was used to promote acceptable tube butt-welds, two of the seams had defects sufficient to cause He leakage in the 10^{-4} torr range; acceptable limit is 1×10^{-6} torr. In addition, it was at this time that the welded construction of the end-cap was being considered unacceptable and No. 5 was not tested.

Evendale was asked to consider fabrication of an end-cap by machining from solid material. MRC was requested to determine the feasibility of hydro-forming an end-cap from a 0.030 inch thick blank and ORNL was asked to look into hot-pressing using graphite dies. EI began to investigate spinning.

Machining of end-caps from a solid was dropped because spinning and hot-pressing were proven more feasible.

6. Clad No. 6

This clad was scheduled for an MRC fueled walk-through. The cylinder was determined to be leak-tight by He leak-testing and radiography. End caps were fabricated by welded construction. MRC welded the bottom end-cap into the cylinder successfully and fueled the unit, but closure welding was called off when it was decided that this clad should be upgraded for service as the Q-1 unit. The bottom end cap was cut from the clad and a new end-ring was welded to the tube; the original end ring was destroyed during removal of the bottom end-cap. The first spun end-rings were fabricated to close the ends of cylinder No. 6 and mating between cylinder and end-caps was within design requirements.

A total of 5 spun end-caps were produced during this year but no acceptable ORNL-pressed end-caps were received although progress was being made.

5.9 EMISSIVITY SLEEVE

5.9.1 DESCRIPTION

The emissivity sleeve is a thin-walled cylinder made from a graphitic yarn (Thornel-50S) which is wound on a mandrel in a 60° - 30° weave. The yarn is then resin impregnated and infiltrated. When completed, the sleeve is 16.69 inches long with a 7.232 inch I.D. and a 0.050 inch wall. The fabrication process is shown schematically in Figure 5.9-1. A final bake-out at 2100°C effectively removes residual volatiles and impurities. Weaves other than the 60° - 30° configuration have been tried, but this orientation appears to be optimum and is specified in Document NS 0060-02-64, Acceptance Requirements for the Heat Source Emissivity Sleeve.

5.9.2 FUNCTION

The emissivity sleeve serves three basic functions:

1. Increases the emissivity of the HSA from ~ 0.2 to ~ 0.8 .
2. Increases the hoop strength of the Iridium Outer Clad.
3. Increases the margin of reentry protection.

5.9.3 ASSEMBLY

Fitting of the sleeve to the iridium outer clad can only be accomplished cold. Figure 5.9-2 shows the minimum-maximum fit-up tolerances at various temperatures. The heat source is assembled to the sleeve at GE and is locked in place with tabs spaced at 120 degrees apart around the iridium clad.

5.10 GAS MANAGEMENT SUBSYSTEM

The helium generated by the radioactive decay of ^{238}Pu is vented from the individual PISA's and discharges into the void space of the IHS. Its subsequent disposition is controlled by the Gas Management Subsystem (GMS).

Figure 5.9-1. Emissivity Sleeves Fabrication

Figure 5.9-2. Emissivity Sleeve/Clad Fit Up

5.10.1 SUBSYSTEM REQUIREMENTS

The principle requirements imposed on the GMS are:

1. To prevent damage to the IHS and to the converter by pressures generated through He build-up or by in-leakage of oxygen from the atmosphere.
2. To prevent gas phase intercommunication between the interior of the IHS and the interior of the converter and thus to avoid potential chemical interactions.
3. To provide the capability, if required, to maintain some inventory of helium within the IHS throughout operation, and thus decrease IHS internal temperatures.

The GMS must, therefore, bleed off the generated He, outside the converter. The He control is exercised from the time of IHS closure throughout normal handling, storage, launch, and mission operation. This function is not expected to be effective under abort conditions.

5.10.2 GMS DESIGN

The IHS void volume was analyzed with respect to porosity and helium absorption characteristics of the large amount of graphite in the IHS. The results revealed a much larger effective void volume when these graphite characteristics are considered. Design modifications also increased the free void volume over that which was initially calculated. The IHS void volume is as follows:

Free Void Volume (without graphite porosity) = 252 in³

Effective Void Volume (with graphite porosity) = 379 in³

Helium generation rate within the IHS is 1×10^{-4} scc/sec. Therefore, the potential pressure increase within the IHS for various conditions is as follows:

<u>Condition</u>	<u>IHS Pressure Increase</u>	
	<u>IHS Temperature (°F)</u>	<u>Pressure Increases (psi/mo)</u>
In SPC	1000	2
In RTG Shipping Cask	1650	2.7
Operation	2000	3

The pressure increase is based on a "bulk helium" temperature within the IHS, and the "effective" void volume (i.e., including graphite porosity). The design requirements are affected by the following components:

1. Vent tube
2. Pressure Relief Device (PRD)*
3. Pressure Maintenance Device (PMD)*
4. Ground handling vent valve

* Either the PRD or the PMD will be selected on the basis of the need for retaining some helium within the IHS. These two components are alternatives, and only one will be selected for incorporation into the flight RTG configuration.

The relationship of these devices to one another and to the HSA and converter is shown schematically in Figure 5.10-1.

Figure 5.10-1. Schematic of Gas Management Subsystem

5.10.2.1 Vent Tube

The vent tube connects the IHS interior to the converter outer wall feed-through, by-passing the converter plenum and allowing the IHS to vent outside the system. The vent tube consists of three different materials (iridium, platinum-rhodium, and 304SST) to match the thermal and mechanical requirements of the GMS subsystem. The iridium end of the vent tube is welded to the clad forward end cap, and stability is increased by a support bracket welded to the end cap at the tube juncture.

The original weld boss fabrication was unsuccessful. The relatively massive arc-cast cylindrical boss, which had significant dimensions ($\sim 1/2$ inch diameter by $1/2$ inch long) had

much porosity and laminations in it, resulting in questionable integrity for leak-tight requirements. Also, much distortion of the end cap, and resulting cracking occurred during the weld-boss-to-end-cap welding.

A design change resulted in successful fabrication and was demonstrated to be effective as a structural support. The new design is a simple tripod-shaped bracket, of 0.02 inch thickness. The decreased mass, and discontinuous weld (short weld at each of three bracket feet) minimize end cap distortion. Leak tightness is maintained by a circumferential seal weld of vent tube to end cap.

A 3-mil thick tungsten shield was incorporated as a barrier between the silicon in the micro-quartz insulation (in the converter preload screw) and the iridium length of vent tube. The tungsten shield is formed into a free-standing cylinder of two revolutions resulting in 6-mils of barrier thickness. The tube is not welded, but simply slid over the vent tube during end cap assembly (room temperature conditions) prior to IHS assembly.

The vent tube connection to the converter feed-through was evaluated against several considerations. The mechanical fitting used at this interface was a flareless compression fitting. Leak tightness had been amply demonstrated for this fitting. However, the vent tube ground handling configuration was a weld joint between the stainless length of tube and the ground handling vent valve in the SPC. The weld joint required cutting of the vent tube in the LAS. Disadvantages of cutting the vent tube and associated procedures were eliminated by incorporating a mechanical fitting on the valve/vent tube ground configuration.

However, flareless fittings did not demonstrate reliable remake characteristics. Consequently, the vent tube flareless joint made for ground handling configuration in the SPC did not reliably remake with the converter feed-through, and seizing of the ferrule occurred during engineering tests. Silver plating and other anti-seize conditions such as aquadag and preferential oxidation did not solve the flareless fitting seizing problem.

The design was subsequently modified to incorporate flared fittings, which have the characteristic of remaking without leaking.

The platinum-rhodium length of vent tube provides ductility for maneuvering the vent tube from its coiled storage configuration to its RTG configuration. Parameters which were evaluated and optimized are: (1) rhodium content in the Pt-Rh alloy; (2) annealing temperature; (3) diameter/thickness relationship. Ductility is indirectly proportional to rhodium content in Pt-Rh. Some rhodium is desirable for machineability and toughness. The alloy selected was Pt-10Rh.

Anneal temperature is very important and sensitive. Ductility and toughness are optimized at 1000°C anneal. Annealing time is limited to 15 minutes to preclude excessive re-crystallization.

The Pt-Rh tube wall thickness and diameter optimization is limited by very tight clearances and sharp bending radii forced by converter dome head room. Although maximum internal diameter is desirable for helium flow conditions, relative wall thickness must not decrease to a value where kinking is a problem. The most functional Pt-Rh tube dimensions were determined to be 0.125 ID x 0.02 wall thickness.

Vent tube vibration tests on the C-11 converter test were successful with no leakage detected prior to or after vibration testing.

The transition welds of the three vent tube materials required welding development. At first there was not sufficient consistency of parameters to produce reliability and repeatability. The weld development program succeeded, and the transition welds show satisfactory integrity. Bend tests demonstrate the Pt-Rh/iridium joint to flex through 90 degrees over ten times without failure. Procedures in the LAS are scrupulous to minimize bending near the transition welds, and the design provides reinforcement in the transition area to minimize bending stresses in the transition welds.

5.10.2.2 Pressure Relief Device (PRD)

The PRD was designed and partially tested for the application of venting the IHS completely to space vacuum. (As reported in the next section, the PMD was evaluated as an alternative in the second half of 1973.)

The PRD is a barometrically-actuated spring/bellows device which operates when the system is exposed to space vacuum. When atmospheric pressure is removed from the evacuated bellows, the unbalanced force across the spring/bellows drives a lance through an aluminum diaphragm allowing permanent access for the He to reach space.

The PRD bellows spring force which is used to open (pierce) the diaphragm is limited by the envelope between the RTG fins where the PRD is mounted. The maximum desirable effective diameter of the bellows is 1-1/2 inches, which provides about 26 pounds of force. This force was demonstrated to be sufficient to pierce an aluminum rupture diaphragm with a margin of safety of at least 2. Several lance designs were tested to assure adequate penetration of the rupture diaphragm.

The PRD temperature is 350°F on the launch pad and 450°F during mission operation. Aluminum was selected for the diaphragm material because of its strength sensitivity in the 300°F to 500°F range. The sharp decrease of strength from launch to operational temperature enhances the diaphragm's characteristic of providing a sealed system in air atmosphere and venting in vacuum.

The PRD is a "fail-safe" system in that the rupture diaphragm will open from creep stresses in the unlikely event the lance does not activate it immediately after launch. Increasing helium pressure within the IHS will creep rupture the aluminum diaphragm before critical stresses are reached in the iridium clad.

Creep tests were performed on the diaphragms as a function of pressure and thickness for 350°F and 450°F, and criteria were established as follows:

At 350°F: creep rupture life > 2 months.

At 450°F: creep rupture life ≤ 1 month.

These criteria are nicely met by the assisting behavior of:

1. Decreasing strength from 350°F to 450°F.
2. Immediate increase of 1 atmosphere of differential pressure across the diaphragm after launch.

As seen in Figure 5.10-2, a 6-mil diaphragm will survive several months at launch with 10 psig helium in the IHS. In vacuum, the 10 psig becomes ~ 25 psi differential across the diaphragm, resulting in creep rupture in less than 100 hours.

Creep test results demonstrated the phenomena of Figure 5.10-2, but final definition of diaphragm thickness has not been accomplished.

Aluminum 1100, a highly annealed pure aluminum, was selected for the diaphragm material. The rest of the PRD is aluminum alloy 6061.

5.10.2.3 Pressure Maintenance Device (PMD)

Development of a PMD was initiated as a result of high fuel temperature. Potentially adverse affects of this high fuel temperature are discussed in Section 5.3. The fuel temperature can be reduced by approximately 100°C by maintaining helium in the IHS gaps instead of vacuum. Unlike the full venting of the PRD, the PMD will maintain about 2 psi differential helium pressure in the IHS.

The approach and sequence of effort to PMD design, selection and definition is as follows:

Figure 5.10-2. PRD Diaphragm Rupture Characteristics

1. PMD Configuration Evaluation
2. Failure Modes and Effects Analysis
3. Selection of most Feasible Configuration
4. Analysis
5. Component Test
6. Data Evaluation
7. Configuration Selection Verification
8. Fabrication Development
9. System Test

The configuration evaluation and failure modes/effects analysis resulted in selection of a PMD system consisting of the following components:

1. Check Valve
2. Capillary Tube
3. Filter

The schematic of this proposed PMD is shown in Figure 5.10-3. The parallel path of the check valve and capillary tube provide protection against both failure modes, open and closed. The check valve has a predominant "open" failure mode, and the capillary tube has a predominant "closed" failure mode. A filter is placed upstream of the capillary tube to prevent the possibility of a single large particle clogging it.

Figure 5.10-3. PMD System Schematics
(Initial Configuration)

NUPRO check valves were selected for flow testing. The NUPRO Company, Cleveland, Ohio, has consistently been chosen as the supplier for highest reliability in high temperature, light weight, and low leak rate valves. The heat source ground configuration (SPC) utilizes NUPRO valves, and the RTG gas management valves (converter argon and IHS helium) are also NUPRO valves, certified for flight.

The PMD NUPRO check valves flow tested were 1, 5 and 10 psi rated. These pressure ratings are cracking pressures at which it was anticipated that equilibrium helium flow conditions would be established at the generation rate (1×10^{-4} scc/sec). However, 1×10^{-4} scc/sec helium flow occurs through the check valve below the rated cracking pressure. In other words, although the valves have a 10^{-6} scc/sec or better leak rate at very low pressure, the leak rate reaches the system equilibrium flow rate before the valve "opens" as defined in the normal use of the valve. Therefore, flow tests were performed to determine pressure vs flow rate characteristics of the valves for 350°F (launch pad) and 450°F operation.

The highest temperature seal that provides a sufficiently soft seat for the low pressure/flow rate requirements is a Viton elastomer. Literature and tests showed that Viton seals remained sufficiently pliable for hours at 450°F, but degraded over long periods of time > 100 hours to unacceptable leak rates.

The reliability of check valves were consequently concluded to be unacceptably low (primarily based on long life seal requirement), and a reselection of the most feasible PMD configuration was required.

Data evaluation forced redefinition of the PMD system. The PMD was redefined as a capillary tube with filter upstream only. The bases for this selection were primarily on the satisfactory flow test results of the capillary tube, the protection from clogging afforded by the filter, and the reliability advantage of having a passive system with no seals.

A test plan was initiated to parametrically measure flow rate as a function of tube diameter, length, and temperature. Capillary tube dimensions have not yet been defined, but will probably be in the range of 0.020 ID x 25 feet long.

5.10.2.4 Ground Handling Valve

The ground handling valve is a bleed valve which is mounted on the converter fin. This is used to bleed off He periodically during the storage period. At that time, the discharge side is attached to a vacuum system to preclude oxygen backflow. The pumped out He is monitored for radioactivity (some ^{220}Rn may be detectable). This valve may or may not be removed immediately prior to integration of the RTG with the vehicle. The PMD or PRD, whichever is used, may be attached downstream of this valve or may be attached in place of the valve, connecting directly to the converter feed through. In Figure 5.10-1 the PRD (or PMD) is shown attached directly to the feed-through. Two additional holes are drilled into the converter fin for this option.

The ground handling valve is of identical concept and construction as the RTG gas management valve. The RTG gas management valve was sectioned to inspect the internal workmanship of the valve. The valve is manufactured by Nuclear Products Company and is an off-the-shelf item. The cross-section of the valve is shown in Figure 5.10-4. In addition, all internal welds were etched and inspected. The workmanship of the valve in all areas was acceptable.

5.10.3 SEQUENCE OF GMS OPERATIONS

The GMS becomes operative when the IHS is welded closed. The sequence of events is shown in Figure 5.10-5.

Note that this operational series does not explicitly identify the points at which He is released from the IHS. The need to perform this operation is contingent upon the actual time/temperature profiles for a given RTG. The operation of the GMS is tied directly to the time/temperature profiles of the IHS fabrication and handling procedures, and the resulting helium pressure profiles. The temperature sensitivity of helium release was discussed in Section 5.

Figure 5.10-4. Gas Management Valve Cross Section

Figure 5.10-5. Gas Management Subsystem Sequence of Events

BLANK

5.11 HSA ANALYSIS AND TEST

This section will describe the analysis and testing performed for the HSA during 1973. The discussion will review the areas of operational assembly and launch temperatures, pad abort thermal response, RTG generator breakup and HSA thermal response during reentry, thermal stress in the aeroshell, and aerodynamic testing.

5.11.1 OPERATIONAL, ASSEMBLY AND LAUNCH TEMPERATURES

Calculations have been made to determine the effect on operational temperatures of various combinations of PICS and outer clad surface finish treatments. Calculations were also made in support of a ΔT test conducted at MRC with the walk-through heat source.

5.11.1.1 Operational Temperature Distribution

A series of calculations was run (Ref. 1) with the three dimensional 1/4 sphere THT-D model to determine the effect of PICS emissivity on temperatures. Calculations were also made with vacuum gaps, with helium inside the PICS and with helium throughout the heat source, and with various values of preload conductance.

The results given in Tables 5.11.1.1-1 and 5.11.1.1-2 were obtained for the parameters:

1. PICS inside diameter emissivity from "clean" iridium ($E \approx .186$) to the emissivity of fuel ($E \approx .83$). This variation will be obtained naturally due to vaporization of the fuel and vapor deposition on the PICS ID.
2. PICS outside diameter emissivity from "clean" iridium to $E = .4$ which is estimated to be the maximum achievable with grit blasting.
3. Measured GIS thermal conductivity.
4. Helium gap conductances are 100% of the continuum values i.e. helium pressure is 1 to 2 psia with nominal gap dimensions.

Table 5.11.1.1-1. Summary of Steady State Temperatures

GAPS		PICS Emissivity		Temperatures °C		ΔT °C
PICS/Fuel	Others	OD	ID	Fuel OD	PICS	
Vac.	Vac.	Clean	Clean	1528	1428	100
Vac.	Vac.	.4	Clean	1486	1378	90
Vac.	Vac.	Clean	.83	1455	1428	27
Helium	Vac.	Clean	Clean	1448	1428	20
Helium	Vac.	Clean	.83	~1438	1428	~10
Vac.	Vac.	.3	.83	1423	1395	28
Helium	Vac.	.3	.83	~1405	~1395	~10
Helium	Vac.	.4	Clean	~1398	1378	~20
Helium	Helium	Clean	Clean	1243	1217	26
Helium	Helium	.4	Clean	~1226	~1200	~26
Helium	Helium	Clean	.83	~1230	1217	~13
Helium	Helium	.3	.83	~1213	~1200	~13

Table 5.11.1.1-2. Summary of GAP Temperature Drops

	<ul style="list-style-type: none"> • Nominal gaps • Nominal emissivities • No contact conductance except emissivity sleeve (E.S.) to clad $h_c = 100 \text{ BTU/hr ft}^2 \text{ } ^\circ\text{F}$ 	
	<u>Vacuum Gaps</u>	<u>Helium Filled</u>
E.S./Clad	21 [°] C	21 [°] C
Clad/Aeroshell	119	17
Aeroshell/Retainer Ring	43**	28
Retainer Ring/GIS OD	28*	28
GIS ID/PICS	115	15
PICS/Fuel OD	100	26

*This value is 11[°]C with $h_c = 1000 \text{ BTU/hrft}^2 \text{ } ^\circ\text{F}$ for the retainer ring/GIS contact

**This value is 20[°]C with $h_c = 100 \text{ BTU/hrft}^2 \text{ } ^\circ\text{F}$ for the retainer ring/aeroshell contact (Ref. 6).

The analyses were made using the THT-D code with the 1/4 sphere model. Pertinent features of the node include:

	GAP, mils	Helium Conductance BTU/ft - hr °F
Fuel/Pics	39.5	85
PIC/GIS	13.5	250
GIS/Retainer Ring	0	0
Aeroshell/Clad	22	150
Clad/Emissivity sleeve (E.S.)	0	0
Retainer Ring/Aeroshell*	51.5	60

*These gap values assume no deformation of the retainer ring. In reality, the preloading will force local contact areas between the retainer ring/aeroshell.

All runs were made with a clad/emissivity sleeve (E.S.) contact conductance of $h_c = 100$ BTU/ft² hr. °F. The GIS/retainer ring contact was treated as a radiation gap in most runs although runs were made with a contact conductance $h_c = 100$ BTU/hr ft² °F and also with $h_c = 1000$ BTU/hr ft² °F. One run was made assuming contact between the retainer ring and aeroshell.

Additional results have been obtained (Ref. 2) for PICS OD emissivity up to .8 and for clad ID coatings. The results, summarized in Table 5.11.1.1-3 are for a THT-D model including:

Vacuum gaps

Clad/E.S. Contact Conductance = $100 \text{ BTU/Ft}^2 \text{ Hr. } ^\circ\text{F}$

Other Contact Conductances = 0

Clean Iridium on the PICS ID

Table 5.11.1.1-3. PICS and Emissivity and Clad Coatings

Emissivities		Temperatures °C	
PICS OD	CLAD ID	Fuel OD	PICS
.18	.18	1528	1428
.3	.18	1500	1395
.4	.18	1486	1378
.8	.18	1469	1362
.4	.18	1486	1378
.4	.4	1449	1337
.4	.8	1433	1315

The temperature distribution for the nominal case - vacuum gaps, clean iridium - is shown in Figure 5.11.1.1-1.

Figure 5.11.1.1-2 gives the temperature distribution with helium in all the gaps of the HSA. Helium in the gaps lowers the PICS temperature by 285° C and the fuel surface temperature by 311° C and causes the PICS front-to-back ΔT to increase to 10° C.

Figure 5.11.1.1-3 shows the temperature distribution including a preload contact conductance of 1000 BTU/ft² Hr °F between the GIS and the retainer ring. The gap ΔT is 11° C compared to 28° C for a radiation gap. The PICS front-to-back temperature gradient is 5° C.

Steady state temperature distributions for other cases are similar to those in Figures 5.11.1.1-1, 2, and 3. The PICS and fuel OD temperatures are shown in Figures 5.11.1.1-4, 5 and 6 as functions of emissivities and gap conditions.

5.11.1.2 ΔT Test

During assembly of the walk-through heat source at MRC, an instrumented FSA was used in order to obtain a temperature measurement. Analyses were made (Ref. 3) prior to the test to determine the expected transient and steady state temperatures. The 1/4 sphere, three dimensional THT-D thermal model was used for the analyses. The HSA configuration

was the assembled HSA except no emissivity sleeve was included. A preliminary steady state run (Ref. 3) was made in order to obtain initial temperatures for the transient calculation.

This steady state run assumed the outer clad to be held at 140°F by the chill blocks and resulted in the PICS maximum temperature of 1307°F (708°C).

The transient analysis (Ref. 3) assumed that the HSA was lifted out of the chill blocks and suspended in a 100°F helium environment. The response of the HSA is shown in Figure 5.11.1.2-1. At the end of 1800 seconds (30 Minutes) the clad has reached 432°C (810°F) while the PICS had reached 757°C (1395°F). The PICS therefore shows a rise of 68°C (123°F), which would be a reasonable rise to confirm the thermal model.

A detailed thermal analysis was performed (Ref. 4) for the steady state temperature profile assuming the outer clad was connected to the 100°F chill block boundary condition by a thermal contact coefficient of $1000 \text{ BTU}/\text{Hr}\cdot\text{Ft}^2\cdot^{\circ}\text{F}$ which represents a good fit under moderate pressure.

The model cross-section is shown in Figure 5.11.1.2-2. The gaps indicated are the nominal room temperature (70°F , 21°C) values. The gap thicknesses were recalculated for the actual temperatures and effective gap conductances developed using the helium gas conductivity vs. temperature and the equation $hc = k/\Delta X$, where hc - helium thermal conductivity ($\text{BTU}/\text{hr}\cdot\text{ft}\cdot^{\circ}\text{F}$) and ΔX = gap thickness (ft.). The conductances used in the analyses are shown in Figure 5.11.1.2-2.

Figure 5.11.1.2-3 shows the results of the analysis. The fuel centerline value is 1355°F (736°C) with a $\Delta T = 1255^{\circ}\text{F}$ (696°C). The T/C will be located 30-45 off the cap center and therefore will measure a PICS value of 1086°F (586°C). However, the T/C will actually be measuring a value between the GIS ID and PICS or between 1086° and 1041°F or an average of 1063°F (573°C).

Figure 5.11.1.1-1. MHW Helipak Steady State Temperatures

Figure 5.11.1.1-2. MHW Helipak Steady State Temperatures

Figure 5.11.1.1-3. MHW Helipak Steady State Temperatures

Figure 5.11.1.1-4. Fuel/PICS Temperature Variation with PICS ID Emissivity

Figure 5.11.1.1-5. Fuel/PICS Temperature Variation with PICS OD Emissivity

Figure 5.11.1.1-6. Fuel/PICS Temperature Variation with Gap Condition

Figure 5.11.1.2-1. Response of MHW HSA after Removal from MRC Chill Blocks

Figure 5.11.1.2-2. Thermal Model of MHW HSA Steady State

Figure 5.11.1.2-3. MHW HSA Temperature Profile-MRC Test Conditions

The ΔT test at MRC consisted of a temperature measurement of the PICS/GIS interface of an FSA in the heat source while the heat source was in chill blocks in a helium environment. The top end cap of the HSA was not in place so there was no preloading. The test set up was completed by stacking graphite pads on top of the top eight pack.

The temperature measurement was made with a thermocouple mounted on a 20-mil foil placed between the PICS and GIS of an FSA on the lower level of the top eight pack. The thermocouple leads were brought through the GIS at the cap threads, so the thermocouple was located as shown:

Another measurement was made on the clad with a hand-held thermocouple.

The PICS/GIS measurement was 413°C while the clad measurements were 95°F to 105°F . Prior calculations (Ref. 4) predicted higher temperatures. New calculations (Ref. 5) were made with:

- New impact shell thermal conductivity
- No preload contact conductance between retainer ring and aeroshell
- Nominal gaps as in Ref. 4 and minimum gaps as a limiting case

The results shown in Figures 5, 11.1.2-4 and 5 indicate predictions of 512°C or 486°C vs. the measured 413°C . Comparison of the new calculations to those of Ref. 4 show a more uniform GIS and PICS temperature due to the differences in the thermal model, primarily the GIS conductivity, and therefore better agreement to test. The effect of gap thickness, 26°C , cannot account for the lower test temperature.

There are two effects which would make it impossible to achieve the calculated temperatures in test:

Figure 5.11.1.2-4. Temperature Profile MRC ΔT Test Steady State

Figure 5.11.1.2-5. Temperature Profile MRC ΔT Test Steady State

1. Free convection in the HSA (between the eight packs and retainer ring) would cause lower test temperature.
2. End losses from the HSA would cause lower test temperatures

A crude evaluation of the effect of free convection indicates that the predictions should be dropped as much as 40°C . This effect is important only for the ΔT test due to the low temperature level and therefore reduced radiation heat transfer in the ΔT test. No evaluation of end losses has been made.

Steady state temperatures including these free convection values give an interface temperature of 472°C or 449°C . The measured temperature, 413°C is close enough to the calculated values, $472^{\circ}\text{C} - 449^{\circ}\text{C}$, (known to be high) to assume that the calculations are reasonable and that there is no unknown conservatism in the calculation model and techniques.

5.11.1.3 Assembly/Launch Temperatures

Calculations have been made to determine the heat source temperatures for various assembly conditions and the thermal behavior at launch, including the transient due to RTG venting. Results are also given to show the effect of the emissivity sleeve/clad gap on heat source steady state temperatures.

5.11.1.3.1 Assembly Steady State

Assembly temperatures have been presented previously. A reanalysis was made (Ref. 8) using recent input data including the LASL iridium emissivity data and TBC - 3 test data. The results are summarized in Table 5.11.1.3-1.

5.11.1.3.2 Launch Steady State

This analysis was performed (Ref. 7) using the TBC-3 test results as a base for the sink environment. This led to the use of a hot shoe average temperature of 1465°F (796°C) which was the measured value, but a net radiation interchange factor of 1.12 was required from the HSA surface to the hot shoe average to match the test values of electrical heater/hot shoe ΔT . The RTG cover gas is argon, which means the emissivity sleeve/outer clad

Table 5.11.1.3-1. MHW HSA Temperatures During Assembly Test and Handling

HSA Condition	Gas in HSA	HSA Sink Temp.	Gas in Sink HSA Gap	HSA External Surface	HSA Surface Temp.	HSA Outer Clad Temp.	PICS Avg. Temp.	Fuel Surface Temp. (Min)	Fuel Center-line Temp.
HSA in shipping protection can	He	~700 °F	He	Clad	1275 °F	1275 °F	1765 °F	1850 ⁽²⁾	1955 ⁽²⁾
HSA in assembly station to without E.S.	He	~70 °F	Argon	Clad	~950 °F	950 °F	1535 °F	1625 °F ⁽²⁾	1825 °F ⁽²⁾
HSA in assembly station with E.S.	He	~70 °F	Argon	E.S.	~710 °F	~930 °F ⁽¹⁾	1515 °F	1605 °F ⁽²⁾	1805 °F ⁽²⁾
HSA in Conv., Conv. in Air, Ar in Conv.	He	-	Argon	E.S.	1530 ⁽⁸⁾	1555 ⁽¹⁾	1925 °F	2005 ⁽²⁾	2215 ⁽²⁾
HSA in Conv., Conv. in Air, Xe in Conv.	He	-	Xenon	E.S.	1675	1700 ⁽¹⁾	2035	2110	2320
HSA in Conv, RTG in Ship. Cont., Ar in RTG	He	-	Argon	E.S.	1578 ⁽⁴⁾ 1684 ⁽⁵⁾	1605 ⁽¹⁾ 1705 ⁽¹⁾	1970 2070	2050 ⁽²⁾ 2150 ⁽²⁾	2260 ⁽²⁾ 2360 ⁽²⁾

- (1) Based on nominal clad/emis. sleeve gaps
 (2) Uses old fuel emis. data, error $\leq 20^{\circ}\text{F}$ (high)
 (3) Assumes emis. sleeve/clad conductance $\geq 100 \text{ Btu/hr-ft}^2\text{-}^{\circ}\text{F}$
 (4) Short-circuited RTG, 520 °F Be case
 (5) 30V operation of RTG, 520 °F Be case
 (6) TBC-3 data gives 1925 °F (avg)
 (7) Using LASL fuel emis. values
 (8) Based on avg. TBC-3 data.

gas is argon filled. The HSA gaps are assumed filled with helium of a pressure greater than 0.1 atm. The gaps and conductances used are listed below:

	Nominal Thickness ~ mils.	Effective Conductance ~ BTU/Hr-ft ² -°F
Emissivity sleeve/outer clad	3.5	93.3
Outer Clad/aeroshell	1.5	1440
Aeroshell/retaining ring	45.0	48
Retaining ring/GIS	< 1.0	100 [*]
GIS/PICS	15.0	160
PICS/Fuel	40.0	63

*Used only 100 because of uncertainty of contact area, etc.

The analyses results are shown in Figure 5.11.1.3-1.

POST LAUNCH TRANSIENT

There are two separate venting conditions which determine the HSA thermal behavior subsequent to launch: the HSA venting, and the RTG venting. The former event begins when the local external pressure reaches 6 psia which corresponds to an altitude of 25,000 ft MSL. This should occur 60 seconds after launch. The presence of the fairing until 305 seconds may slightly delay this event. The second venting takes place 380 seconds after launch and the argon RTG cover gas begins to vent. The HSA venting primarily affects the HSA internals since the high conductivity helium gas is being replaced by vacuum thereby greatly increasing the resistivity across the multiple gaps. However, the RTG venting affects the hot shoe temperature and since this is the HSA sink, it affects the HSA surface temperature.

The transient response of the hot shoe temperature has been estimated by Wiley and was used as the transient boundary condition for the HSA response.

Figure 5.11.1.3-1. MHW HSA Temperature Profile-On Pad Conditions

The HSA was assumed to be evacuated instantaneously at time of launch and the gap conductances were instantaneously reduced to zero, except the pressure contacts, namely, the emissivity sleeve/outer clad, and the retaining ring/GIS. The first gap gets smaller as temperatures increase and therefore an interference fit occurs. The second gap is generated when the sphere locks are torqued into place. An h_c value of 100 BTU/hr-ft² was used which is a low (conservative) value when the meager data is reviewed.

The results of the analyses are shown in Figure 5.11.1.3-2. The transient response of the fuel center node, the outer PICS node and the emissivity sleeve node are shown compared to the sink environment transient. The HSA fuel node is at 2950°F (1621°C) at 1000 minutes which compares quite well with the 2943°F (1617°C) value in the operational steady state. The PICS value of 2590°F (1421°C) also compares well with the steady state value of 2580°F (1416°C). The transient values are slightly greater than the steady state values owing to small differences in the model inputs. Therefore, the HSA can be assumed to be at steady state within 1000 minutes (60,000 sec) of launch provided the venting systems operate.

Also of interest is the fact that at 500 minutes (30,000 sec) after launch the HSA fuel and PICS nodes have achieved > 80% of the total 1000 minute temperature rise.

5.11.1.3.3 Effect of Emissivity Sleeve/Clad Gap

An analysis (Ref. 6) was made to evaluate the emissivity sleeve/outer clad temperature drop since this interface is a significant design problem. If the gap is too large operationally, then the outer clad and hence fuel will be too "hot", but if the gap is too small the girdle cannot be fitted in place over the clad. Ref. 6 evaluated the temperature penalties occurring in the gap for the gap thickness range in the present design.

The analysis was based on a one-dimensional hand calculation which assumes the POCO aeroshell will distribute the heat uniformly over its surface since it is a good conductor—a reasonable assumption based on previous test and analyses. The gap temperature drop is critical when the gap gas is argon, which has relatively low conductivity—rather than helium which has an order of magnitude higher conductivity. The clad emissivity is taken as 0.18 (a minimum value) and the sleeve emissivity as 0.8.

MHW HSA TRANSIENT
POST LAUNCH RESPONSE

• USING 1/4 SPHERE 3-D MODEL

Figure 5.11.1.3-2. MHW HSA Transient Post Launch Response Using 1/4 Sphere 3-D Model

The conduction and radiation heat fluxes were calculated as a function of clad and emissivity temperatures for the nominal, maximum and minimum gap thicknesses with the thickness referenced to the clad temperature, since the expansion of the E.S. is nil. An assumption is made that even if the gap thickness goes to zero, the maximum conductance value is 100 BTU/Hr- H^2 - $^{\circ}$ F. This would result in a minimum ΔT of 20° F across the gap as shown by the results in Figure 5.11.1.3-3.

Other analyses have shown (Table 5.11.1.3-1) that a free HSA in the assembly station will have the emissivity sleeve at about 710° F. Extrapolating the results in the Figure, we see that the clad would be $\sim 820^{\circ}$ F if minimum gaps existed, $\sim 870^{\circ}$ F for nominal gaps and $\sim 900^{\circ}$ F for maximum gaps. (Note that Table 5.11.1.3-1 indicated 930° F for nominal gaps--this was based on early numbers with conduction effects only).

A critical case is the launch temperatures which have the sleeve running at 1530° F for nominal gaps, 1550° F for minimum and 1590° F for maximum gaps.

The normal operation case should have a 20° F ΔT for all cases and the ΔT is based on a 100 BTU/Hr-ft 2 - $^{\circ}$ F contact coefficient generated by the interference fit.

5.11.2 PAD ABORT THERMAL RESPONSE

A launch phase abort may subject the RTG to a fire environment if the abort results in a solid or liquid propellant fire. Analyses have been made to determine the thermal response of the RTG and HSA including the effects of oxidation reactions. A summary of the results is given in Table 5.11.2-1.

Two basic models have been used to calculate thermal response with the THT-D computer program. These are:

1. RTG Model (Generator and HSA combined)
2. HSA Model

Figure 5.11.1.3-3. MHW HSA Emissivity Sleeve/Outer Clad Gap Temperature Penalty

Table 5.11.2-1. Summary of Liquid Fire Results

Initial Configuration	Environment	Analysis Results/Conclusions
RTG/HSA	Fireball	Case may not melt.
	After fire with case	PICS at 2310 ^o F at conclusion of fire.
	After fire without case	PICS at 2320 ^o F at conclusion of fire.
	After fire + oxidation	PICS at 2465 ^o F or less when foil is removed.
Bare HSA	Fireball Afterfire	Clad does not melt PICS reaches 2000 ^o F
Summary of Solid Fire Results		
Initial Configuration	Environment	Analysis Results/Conclusions
HSA/RTG	Contact	Case melts after 6 sec. PICS @ 2200 ^o F at conclusion of fire.
	Contact & oxidation	Foil is removed between 200 to 410 sec after fire. PICS @ 2300 ^o F to 2500 ^o F when foil is removed.
	Proximity	Case melts after 6 sec. PICS @ 2175 ^o F at conclusion.
	Proximity & oxidation	Same as for Contact
Bare HSA	Contact	Clad doesn't melt. PICS @ 3230 ^o F
	Proximity	Maximum PICS temperature 2475 ^o F

The RTG model represents a 45° circumferential section, 1.08 inches in length from the center section of the RTG. This model incorporates the 3-D, $1/4$ sphere model for the HSA, combined with a generator model. Due to symmetry, the model represents the entire RTG if it is restricted to calculations with a circumferentially uniform environment.

The bare HSA model is two-dimensional, representing 180° of the center section of the heat source, and due to symmetry, models the entire HSA.

Three different fire environments were used:

1. Liquid Propellant

The temperature and heat flux model for the liquid propellant fire resulting from a Titan III C abort on the launch pad is shown in Figure 5.11.2-1, (1973 model). This environment which results from a catastrophic failure of the Titan III C, consists of a fireball and afterfire. The fireball environment was revised in 1973, while the 1500°F afterfire temperature was obtained during the S-2 test (Ref. 3).

2. Solid propellant contact (Ref. 10)
3. Solid propellant proximity (Ref. 10)

Figure 5.11.2-1. Titan III C Liquid Fire Environment

5.11.2.1 Oxidation

The molybdenum foil and the graphite emissivity sleeve will oxidize when exposed at high temperature to air or oxygen. These reactions are exothermic and must therefore be accounted for in the analysis of RTG/HSA thermal response. Oxidation of the molybdenum is particularly significant since removal of the multi-foil insulation has a major effect on the HSA thermal response.

In the presence of oxygen, the molybdenum reaction is

where Q is the heat of combustion, 2120 BTU/lb. MoO_3 .

Molybdenum oxidation rate data shown in Figure 5.11.2.1-1 for reaction rate limited has been taken from Ref. 39. This is the rate of molybdenum oxidation which would occur in the presence of a plentiful supply of oxygen so that the controlling factor is the reaction. In reality, the oxidation rate may be less due to a restricted supply of oxygen. For example, on both the liquid and solid fire environments, there should be no free oxygen so it was assumed that the molybdenum would not oxidize.

After the fire, the molybdenum foil outside surface was assumed to be exposed to free convection air currents which supply the oxygen for reaction. The corresponding molybdenum oxidation rate, shown in Figure 5.11.2.1-1, assumes all the air convected to the surface is consumed in the reaction.

For the inside surface of the molybdenum foil insulation, there are two extremes which have been considered; no oxygen penetrates the outer foils and therefore there is no oxidation at the inside surface, or the inside surface oxidation rate is equal to the rate for the outside surface. Realistically, air will reach the inside surface by leaking in around the seams and thermoelectric elements and the inside surface oxidation rate will be between these extremes.

Figure 5.11.2.1-1. Molybdenum Oxidation Rate

A surface energy balance on the foil consists of three terms, an exothermic heating, a convection cooling and a radiation interchange. The exothermic heating due to oxidation of the foil is 2120 BTU/lb of MoO_3 or 169 BTU/ft² per mil of molybdenum foil. Using the molybdenum rates already given, results in the exothermic heating term shown in Figure 5.11.2.1-2. The convection cooling accounts for the fact that the air is at room temperature when it contacts the foil and the reaction products are assumed to be at the surface temperature. The values shown in Figure 5.11.2.1-2 are for laminar free convection over a vertical surface. Figure 5.11.2.1-2 also shows for comparison of terms the radiation cooling from the outside of the foil insulation of a 70°F environment.

Air which leaks inside the foil insulation will also oxidize the graphite emissivity sleeve. The reaction rate data for AVCO 3D Orthogonal Carbon/Carbon Composite (Ref. 22) has been used along with the laminar free convection air supply to determine the rate of graphite oxidation as a function of temperature. The graphite has been assumed to oxidize via the reaction

where the heat of combustion, Q , is 10,500 BTU/lb. C. Figure 5.11.2.1-3 shows the graphite surface energy terms.

5.11.2.2 Solid Propellant Fire Analysis

The response of the generator and HSA has been evaluated using both the contact fire model and the proximity fire model assuming as a worst case condition, that the fire environment is uniform over the entire generator surface. The post fire response of the HSA has also been evaluated for various assumptions regarding oxidation effects. The initial temperatures for the generator and HSA are those for the on-pad steady state condition calculated with argon in the generator and a 350°F case temperature.

Figure 5.11.2.1-2. Molybdenum Surface Energy Balance Terms

Figure 5.11.2.1-3. Graphite Surface Energy Balance Terms

5.11.2.2.1 RTG Model Contact

Calculations were made of the generator and HSA thermal response during the 600 second solid fire assuming instantaneous removal of the case. An auxiliary calculation showed that the case would melt after only about six seconds and would therefore have little effect on the thermal response. The calculations show that the most severe possibility for oxidation inside the HSA occurs with insulation oxidation on inside and outside and emissivity sleeve oxidation. The PICS reaches 2450°F for this condition.

5.11.2.2.2 RTG Model Proximity

These calculations were also made neglecting the effect of the case since it will be melted after only about six seconds. The results are shown in Figure 5.11.2.2-1 for the 600 second duration of the solid fire. The maximum PICS temperature is 2175°F only 25°F below that for the contact fire. The post fire behavior and final PICS temperatures should be approximately the same as those given in the previous section.

5.11.2.2.3 Bare HSA Contact

The thermal response of the HSA in the contact solid fire is shown in Figure 5.11.2.2-2. The maximum clad temperature, 3770°F , is well below the iridium carbon eutectic so the clad will remain in place. The maximum PICS temperature is 3230°F . Post fire oxidation of the emissivity sleeve will occur but will not prevent the HSA from cooling rapidly.

5.11.2.2.4 Bare HSA Proximity

The bare HSA model has also been run in the solid fire proximity environment. Since the HSA model can be used with a circumferential variation in boundary conditions, this calculation is more realistic than that with the combined HSA/RTG model. The circumferential variation in boundary conditions was treated by using view factors between HSA surface nodes and the solid fire environment. The results are shown in Figure 5.11.2.2-3.

5.11.2.3 Liquid Propellant Fire Ball/After Fire Analysis

The initial temperatures for the generator and HSA are those for the on pad steady state condition calculated with argon in the generator and a 350°F case temperature.

Figure 5.11.2.2-1. RTG Model Proximity

Figure 5.11.2.2-2. Bare HSA in Solid Fire - Contact

Figure 5.11.2.2-3. Bare HSA in Solid Fire

5.11.2.3.1 Generator Response in Fire Ball/After Fire

Figure 5.11.2.3-1, 2 and 3 show the thermal response of the generator and HSA to the liquid fire environment. Calculations were made both with and without the case in the after fire. There was no appreciable difference in the HSA thermal response.

Only the worst oxidation condition has been considered for evaluating the HSA post-fire condition; foil oxidation inside and outside and oxidation of the emissivity sleeve. The results are shown in Figure 5.11.2.3-3 until the foil burns off, about 200 seconds after oxidation begins. The maximum PICS temperature is 2465⁰F.

5.11.2.3.2 Bare HSA Thermal Response

Figure 5.11.2.3-4 and 5 show the HSA thermal response during the liquid fireball and after fire. The HSA is near equilibrium with the liquid fire after about 200 seconds with a PICS temperature of 2000⁰F.

5.11.3 POST IMPACT TEMPERATURES

A thermal analysis (Ref. 11) was performed to determine the temperature profiles of the FSA and PISA in several post-impact burial configurations at steady state. The analysis included an FSA and PISA on the surface buried to half diameter, buried to full diameter, and buried to a distance of at least 6 inches below the surface. The foils used were Ottawa Sand and Diatomaceous Earth, (Conductivity shown in Figure 5.11.3-1).

Because of the symmetry in the circumferential direction the analysis was two dimensional. The FSA Model is a 36-node model in which all the nodes of the same radial distance from the center have approximately the same volume. The PISA model is made by simply removing the Graphite Impact Shell from the FSA. Figure 5.11.3-2 shows a typical model including the model for the earth nodes.

The GIS-PICS and PICS-fuel gaps in the FSA were assumed to be of nominal size and filled with air. To account for this a contact coefficient of 21.3 BTU/Hr-Ft²-⁰F was input for

Figure 5.11.2.3-1. Temperature Response During Liquid Fireball

Figure 5.11.2.3-2. Temperature Response of RGT w/Case During Liquid Fire

Figure 5.11.2.3-3. Temperature Response of RTG W/Case During Liquid Fire

Figure 5.11.2.3-4. Bare HSA Temperature Response During Liquid Fireball

Figure 5.11.2.3-5. Bare HSA Temperature Response During Liquid Fireball

Figure 5.11.3-1. Thermal Conductivities Burial Analysis Materials

.Nodes 1, 2, 11, 12, 21, 22 are deleted for PISA

the GIS-PICS gap and $9.35 \text{ BTU/Hr-Ft}^2\text{-}^\circ\text{F}$ for the PICS-fuel gap. This would be conservative at best because air has a higher thermal resistance than helium.

No oxidation of either the graphite or the iridium was assumed since this would be conservative because oxidation would raise the PICS emissivity and thus lower the temperature of the FSA or PISA.

Free convection and radiation to 100° air are the heat transfer modes used to cool the FSA or PISA when above ground, while conduction (no contact resistance) to the soil and soil radiation and free convection to 100° air dumps the heat when buried. The results are given in Table 5.11.3-1.

As noted, the calculations were made without accounting for oxidation of the graphite or PICS. Many of the temperatures listed in Table 5.11.3-1 are high enough for oxidation to occur. The oxidation threshold of graphite (1% weight loss in 24 hours) is generally given as 750°F .

The GIS on an FSA above ground or half buried in Ottawa Sand is below the threshold so oxidation would be very slow. Removal of the GIS would take in excess of 100 days so there should be no adverse effects due to graphite oxidation. For an FSA half buried in Diatomaceous Earth, the GIS is near the oxidation threshold so removal should occur in about 100 days. Eventually the GIS would be removed and the PICS temperatures would increase. The maximum PICS temperature, half buried without the GIS, is 1447°F . Laboratory tests have shown that at 1500°F , iridium forms a stable oxide which causes the emissivity to increase. The PICS above ground or half buried will eventually reach equilibrium at less than 1447°F with a stable oxide coating. The PICS should provide containment indefinitely.

In the full and 6" burial cases in Ottawa Sand, the GIS temperatures are near the oxidation threshold so the oxidation rate will probably not be affected by the fact that the oxygen must diffuse through the sand to reach the GIS. GIS removal should require about 100 days

after which the PICS temperature will decrease slightly due to the fact that the thermal conductivity of Ottawa Sand is higher than that of the GIS. The PICS temperature in Ottawa Sand will not exceed 1043°F.

FSA's in Diatomaceous Earth, full burial or 6" burial, have GIS temperatures significantly above the oxidation threshold so the GIS oxidation rate should be controlled by the rate at which oxygen can travel through the soil. The time for GIS removal has not been evaluated. PICS temperatures after GIS removal are 2128°F and 3312°F for full burial and 6" burial. These are well above the 1960°F Temperature level at which iridium oxide becomes volatile.

Table 5.11.3-1. Burial Analysis Results

Case Identification	Max. GIS Sur. Temp. Above Ground ~°F	Max. GIS Inner Surface Temp. ~°F	Max. PICS Temp. ~°F	Fuel Center Temp. ~°F
FSA Above Ground	615	702	953	1538
PISA Above Ground	—	—	1417	1908
FSA Half Buried - Insulated	806	946	1157	1700
FSA Half Buried in Ottawa Sand	506	590	848	1455
FSA Half Buried in Diatomaceous	707	807	1059	1620
PISA Half Buried in Ottawa Sand	—	—	909	1477
PISA Half Buried in Diatomaceous	—	—	1447	1906
FSA Full Burial Surface; Ottawa	—	595	894	1492
FSA Full Burial Surface; Diatomaceous	—	1650	1817	2248
PISA Full Burial Surface; Ottawa	—	—	772	1391
PISA Full Burial Surface; Diatomaceous	—	—	2128	2514
FSA Full Burial 6" Ottawa	—	808	1043	1610
FSA Full Burial 6" Diatomaceous	—	2561	2657	3030
PISA Full Burial 6" Ottawa	—	—	934	1525
PISA Full Burial 6" Diatomaceous	—	—	3312	3650

5.11.4 REENTRY AEROTHERMAL ANALYSES

Aerothermal analyses were performed for several different reentry conditions with the emphasis on evaluating the effects of the emissivity sleeve, and the RTG on aeroshell temperature gradients and calculating aeroshell gradients for the safety cases selected as representative of the LES 8/9 flights. The analyses included RTG breakup, HSA thermal response and aeroshell thermal stress. An analysis was also made to evaluate an FSA's reentry capability. Some 6-degree-of-freedom trajectory simulation studies were also performed.

5.11.4.1 Minimum Gamma Trajectory

5.11.4.1.1 Thermal Analysis of Free HSA

Calculations were made (Ref. 12) to determine the effects of iridium coatings (Clad ID $\epsilon = .8$, PICS OD $\epsilon = .3$) on HSA temperatures in a minimum gamma reentry with nominal and increased heating* for the design case of a free HSA from 400,000 feet. The analyses were made with the 2-D THT-D model with the revised GIS thermal conductivity and fuel emissivity data. The PICS was assumed to have an emissivity of .3 on the outside and the emissivity of clean iridium on the inside. The outer clad was assumed to have an emissivity of .8 on the inside and the outside was assumed to be clean iridium.

All gaps were vacuum with deceleration contact coefficients as follows:

	hc BTU/ft ² hr °F
Emissivity Sleeve/Outer Clad	1000 ($\theta = 0^\circ$ to 30°) 500 ($\theta = 30^\circ$ to 55°) 100 ($\theta = 55^\circ$ to 180°)
Outer Clad/Aeroshell	100 ($\theta = 0^\circ$ to 30°) 50 ($\theta = 30^\circ$ to 55°) 0 ($\theta = 55^\circ$ to 180°)
Aeroshell/Retainer Ring	100 ($\theta = 0^\circ$ to 30°) 50 ($\theta = 30^\circ$ to 55°) 0 ($\theta = 55^\circ$ to 180°)
Retainer Ring/GIS	100

Figure 5.11.4.1-1 gives the heating rate and recession based on a 20% increase in the heat transfer rate above the theoretical value. Emissivity sleeve removal occurs at 127 sec. compared to removal at 150 sec. for the theoretical heating case. Total aeroshell recession to impact is 203 mils compared to 158 mils for the theoretical heating case.

*The Langley aerodynamic heating tests indicated that stagnation point heating side-on is higher than theoretical. The minimum gamma runs documented here were run with the heating 20% higher than theoretical.

Figure 5.11.4.1-2 shows the temperatures for the nominal heating case with coatings. The maximum PICS temperature of 3080°F is lower than the 3101°F value of previous work due to the reduced initial temperatures.

Figure 5.11.4.1-3 shows the HSA temperatures corresponding to the increased heating rate of Figure 5.11.4.1.1-1 with coatings. The maximum PICS temperatures is 3230°F compared to a maximum of 3080°F with nominal heating.

5.11.4.2 Reference Design Steep Angle Trajectories

Calculations were made for a free HSA at 400,000 feet with and without the clad and emissivity sleeve. These analyses consisted of aeroshell thermal response calculations and thermal stress calculations.

5.11.4.2.1 Without Clad and Emissivity Sleeve

5.11.4.2.1.1 Center Section Analyses

SIDE-ON THERMAL ANALYSIS

Analyses of the MHW HSA aeroshell center-section thermal response during a $\alpha = -90^\circ$ re-entry were performed and the results are presented in a series of figures. HSA design is a 2400 W_{th} 4 x 6 Helipak using 0.425" impact covers and a concentric FSA design. The thermal model used in the analysis is shown in Figure 5.11.4.2-1. Only the aeroshell and retaining ring were modeled. A total of 160 nodes were used in the analysis, and the THT-DA computer code was employed. The backface of the retaining ring radiated to

Figure 5.11.4.1-1. Heating Rate and Recession - Heating Rate Increased 20%

Figure 5.11.4.1-2. Minimum Gamma Reentry Temperatures - Nominal Heating

Figure 5.11.4.1-3. Minimum Gamma Reentry Temperature - Heating Rate Increased 20%

Figure 5.11.4.2-1. MHW HSA Aeroshell Thermal Gradient Model

sink temperatures to simulate the fuel. The aeroshell OD = 7.186" and the thickness is .386 calculated at 2000°F.

The trajectory for this analysis was calculated previously for a $B = 41.4$ psf HSA, and this was used for the present HSA which has a $B = 42.4$ psf. Since the analysis is to be conservative, the outer clad was assumed removed at 400,000 ft. The altitude and velocity profile for the first 20 seconds of re-entry are shown in Figure 5.11.4.2-2. The ABTON program was used to calculate the heating environment; enthalpies and pressures - for the HSA, as well as stagnation pressure and reference cold wall heating rates. These latter parameters are plotted in Figure 5.11.4.2-3. The stagnation pressure can be used to calculate the mechanical load on the aeroshell during re-entry which is then superimposed on the thermal loads to calculate the total structural load and net margin of safety. The output from ABTON was then used in conjunction with the thermal model as input to the THT-DA computer code to calculate temperature gradients.

Figure 5.11.4.2-4 shows the response of the aeroshell at the stagnation line as a function of time. The distances from the surface, (except the surface plot) are original distances neglecting recession, which is a small effect, being only 15 mils at 10 seconds. From this plot it is apparent peak surface temperatures and temperature gradients occur at 8.2 sec, the time of peak heating.

CENTER SECTION-THERMAL STRESS ANALYSIS

The thermostructural analysis of the center of the graphite aeroshell during side-on re-entry was performed (Ref. 16) using the GE ORTHOSAFE computer program. The program uses the finite element, direct stiffness solution for orthotropic elastic bodies. The program has the capability to incorporate up to six orthotropic materials with bi-linear stress-strain relations for an elastoplastic solution. The program is limited to thermal gradients in two directions and therefore, only the stresses in those two directions are computed.

Figure 5.11.4.2-2. MHW HSA Steep Angle Reentry Trajectory

Figure 5.11.4.2-3. MHW HSA Steep Angle Reentry Pressure and Heating History

Figure 5.11. 4.2-4. MHW Aeroshell Temperature History

The aeroshell was considered to be an annular section remotely located from the end caps. Radial and circumferential temperature gradients are programmed into the structural model of ORTHOSAFE which computes stresses, strains and displacements. Of the two program options, the plane stress solution was chosen over the plane strain solution since it produces the higher hoop stresses.

The aeroshell model was cut at the plane of symmetrical loading to reduce the size of the ORTHOSAFE structural model. The model consists of 17 equal radial nodes and 40 circumferential nodes. The circumferential nodes are smaller near the stagnation point and larger near the back side of the aeroshell.

Ten radial temperature profiles from the aerothermal computer analysis of the previous section at a specific time during reentry were used in the structural analysis. These were supplemented with seventeen computer extrapolated profiles to maintain the accuracy of the orthosafe analysis.

Since the AXF-5Q tension and compression properties are different, they were considered as separate materials in the orthosafe program. The AXF-5Q properties used in the computer program are described in PIR 4482.

The maximum thermal tensile stress on the inside of the aeroshell wall (at the stagnation point) was found to occur at 8.4 seconds. (See Figure 5.11.4.2-5). The stress distribution through the stagnation point is shown in Figure 5.11.4.2-6.

The additive bending stresses in the aeroshell resulting from the re-entry pressure was calculated using a fixed plate supported by the FSA's. The maximum additive tensile bending stress on the inside of the aeroshell at a peak pressure of 198 psi (PIR 3776) is 2100 psi.

The peak elastic thermal stress at 8.4 seconds is 8600 psi. The peak elastic pressure stress at max. q is 2100 psi. Reducing the combined elastic stresses to a plastic stress results in 8650 psi.

Figure 5.11.4.2-5. Aeroshell Margin of Safety

Figure 5.11. 4.2-6. Thermal Stress at Stagnation Point 8.4 Sec.

$$f_{\text{plastic}} = f_y + \frac{E_p}{E_e} ([f_t + f_{\text{press}}] - f_y)$$

$$8650 \text{ psi} = 7600 + 1/3 ([8600 + 2100] - 7600)$$

The minimum margin of safety based on minimum unflawed data and minimum acceptable specification data is:

$$\frac{9500}{8650} - 1 = 9\%$$

5.11.4.2.1.2 End Section

SIDE ON THERMAL ANALYSIS

An aerothermal analysis was performed to define the temperature levels and gradients in the MHW aeroshell in the vicinity of the threaded end cap during a $\gamma_o = -90^\circ$, $V_o = 36,333$ ft/sec, side-on re-entry. This re-entry results in high thermal stresses in the aeroshell and the MHW HSA must demonstrate a positive margin of safety during these type re-entries. The threaded section is a critical area and therefore was examined in depth.

The analysis was performed for the MHW HSA design (2400 W_{th} 4 x 6 Helipak), a concentric FSA design, and POCO 5Q aeroshell, (0.386 in. thick).

Several preliminary analyses were performed before finalizing the nodal configuration for the thermal model. These were done so that we could minimize the number of nodes required and still retain an accurate thermal representation.

To determine whether modeling in the circumferential direction was needed, a 16-node radial section at the stagnation point of the existing aeroshell thermal stress model was run with exactly the same boundary conditions as the 160-node model. The only difference being the 16-node model was adiabatic in the circumferential direction. The results showed no significant difference in the thermal response at the stagnation line between the two models. It was therefore concluded that it was not necessary to include nodes in the circumferential direction.

The effect of the internal structure (i. e. retaining ring, FSA's, etc.) was also investigated and found insignificant to the thermal response of the aeroshell.

The thermal model used in the analysis is shown in Figure 5.11.4.2-7. Only the end cap and the aeroshell in the vicinity of the threads were modeled. The analysis assumed the aeroshell and end cap adiabatic in the circumferential direction and at the internal surfaces. These assumptions were based on the preliminary analyses mentioned above. The threaded area was modeled in detail because this was one of the main points of concern in this analysis. Contact resistances and internode radiation at the threaded joint were included.

A total of 183 nodes were used in this analysis. The THT-DA computer code was employed. This code calculates the aeroshell recession and thermal response simultaneously. The aeroshell OD = 7.186", and the thickness is .386", calculated at 2000°F.

The trajectory for this analysis was calculated for a continuum ballistic coefficient $\beta_0 = 42.4$ PSF (HSA). The ABTON program was used to calculate the heating environment; (enthalpies and pressures) -- for the HSA, as well as stagnation pressure and reference cold wall heating rates. The stagnation pressure can be used to calculate the mechanical load on the aeroshell during reentry which is then superimposed on the thermal loads to calculate the total structural load and net margin of safety. The output from ABTON was then used in conjunction with the thermal model as input to the THT-DA computer code to calculate the aeroshell recession and thermal response. Figure 5.11.4.2-8 shows the temperature distribution in the aeroshell and end cap at the time (8.0 sec.) of peak temperature on the outside surface of the aeroshell.

Figure 5.11.4.2-9 shows the temperature response as a function of time at the stagnation line of the aeroshell 2.25" from the threaded joint. The stagnation line temperature/time history at the threaded joint is shown in Figure 5.11.4.2-10. The temperature drop across the threads can be seen by comparing the curve for node 510 against the curve for node 511.

Figure 5.11.4.2-7. Side-on Thermal Model of HSA End Cap Cross Section

TEMPERATURE DISTRIBUTION AT 8 SEC.

Figure 5.11.4.2-8. Side-on HSA End Cap Cross Section

Figure 5.11.4.2-9. MHW Aeroshell Temperature History 2.25" from Threaded Joint

Figure 5.11.4.2-10. MHW Aeroshell Temperature History at Threaded Joint

END-ON THERMAL ANALYSIS

An aerothermal analysis was performed to define the temperature levels and gradients in the MHW end cap and aeroshell during a $\gamma = -90^\circ$, $\gamma_o = 36,333$ ft/sec. end-on reentry.

A total of 208 nodes were used in the model which included a deceleration contact coefficient of $100 \text{ BTU/hr-ft}^2 - ^\circ\text{F}$ and thermal radiation between end cap and aeroshell (Nodes 213-214, 313-314, 413-414, 513-514, 514-614, 515-615, 516-616). The threads were not modeled in this analysis; rather a plane area interface was used between Nodes 514-614, 515-615, 516-616. This causes about 3.5 times more area to be in contact and radiating across the gap than if the threads were modeled. This results in the calculated gradient across them being less than if the threads were modeled and the contact coefficient also = $100 \text{ BTU/hr-ft}^2 - ^\circ\text{F}$. However, since the calculated gradient is small (about 200°F) this is of little concern.

The aeroshell cylinder OD = 7.186" and the cylinder thickness = .386" and the end cap thickness = .504". All dimensions are measured at 2000°F . The trajectory was calculated for a continuum ballistic coefficient $\beta = 42.4$ PSF HSA.

The ABTON program was used to calculate the heating environment (enthalpies and pressures) for the HSA as well as stagnation pressure and reference cold wall heating rates. The output from ABTON was used in conjunction with the thermal model as input to the THTDA computer code to calculate the thermal response and surface recession.

The results of this analysis are shown in the plots that follow. Figure 5.11.4.2-11 is the nodal diagram while Figure 5.11.4.2-12 shows the temperature profile at 8.1 seconds (Peak Surface Temperature and peak gradient across end cap). Figures 5.11.4.2-13 and 14 show temperature as a function of distance from stagnation surface at peak surface temperature.

Figure 5.11.4.2-11. End-on End Section MHW HSA Nodal Diagram

Figure 5.11.4.2-12. Temperatures at 8.1 Sec. (Peak Surface Temperatures)

Figure 5.11.4.2-13. End-Section In-Depth Temperatures

Figure 5.11.4.2-14. End-on End-Section In-Depth Temperatures

SIDE-ON AND END-ON THERMAL/STRESS ANALYSIS

Ref. 17 gives the results of a Thermal/Stress analysis using a three-dimensional structural model having a two-dimensional thermal gradient. Two configurations (side-on and end-on) each with a different thermal environment were analyzed.

The elastic thermal stress analysis was performed using a GE computer program called BIG 2D. The program uses finite elements for analyzing either axisymmetric or plane two-dimensional configurations. Specified temperature-dependent properties are applied to each finite element of the structural model. All nodes (four per element) are analyzed per their respective temperatures using the elemental properties. The thermal/mechanical three-dimensional octahedral and principal stresses are determined for each element as well as the two-dimensional distortions at every node. The program is capable of analyzing composite and anisotropic materials and is virtually unlimited in size.

A GE computer program called TRACE is used to determine whether a finite element from the BIG-2D program is stressed within the elastic or plastic range. The procedure is to enter the calculated octahedral elemental stresses and temperature into the TRACE program. The program determines if the allowable tensile or compressive yield strength for the element at temperature has been exceeded. If the yield strength is exceeded, the appropriate stress-strain curve is followed for the previously calculated strain and the plastic stress determined. Using the plastic stress and strain, a plastic secant modulus is calculated. The data is fed back into the BIG-2D program which assembles and re-analyzes the entire structure using the plastic properties for those elements stressed beyond their allowable yield. The procedure is continued until the structure is in equilibrium with the applied thermal environment or an element has exceeded its stress or strain capability.

The computer model for the side-on configuration is shown in Figure 5.11.4.2-15. It is axisymmetric about the y axis, representing the entire end cap and about 3 inches of the aeroshell cylinder wall. There are 230 nodes, each with its respective temperature and 172 elements with variable material properties. Due to axisymmetry, the model is

THIS PAGE LEFT BLANK INTENTIONALLY

structurally three dimensional. Since a 2D thermal profile was used, this analysis does not exactly represent the actual conditions. The end-on model (Figure 5.11.4.2-16) has 198 nodes and 143 elements. Analytically, it has the same capability and properties as previously stated for the side-on mode. Since the thermal environment is axisymmetric about the y axis, this analysis represents actual conditions. Material properties used are shown in Figures 5.11.4.2-17 through 21.

For the side-on model, the analysis showed that the critical elements were 158, 160, 169. When these elements were removed from the model due to a simulated compressive failure and re-analyzed by the computer, the minimum margin of safety was + 0.033 at element 69 which represents a screw thread.

For the end-on model, the critical elements were 14, 32, 44, 56, 62 all of which represent threads. These elements were then eliminated from the model except for the lowest stressed thread (element 56), which was kept to restrain the end cap. A reanalysis under these conditions showed elements 56, 77, and 130 to be critical all with negative margins of safety of -0.428, -0.084 and -0.121 respectively.

Computer output showing stresses and margins in each of the nodes is contained in Reference 17.

5.11.4.2.2 With Clad and Emissivity Sleeve

As a first step in evaluating the actual thermal stresses encountered by the aeroshell during the LES 8/9 steep angle reentries, the steep angle reference design thermal analysis has been performed (Ref. 13) for a re-entering RTG HSA. The following analysis presents the clad/emissivity removal times and the thermal response of the HSA in the thermal shock environment.

The study first defined the time of removal of the clad and emissivity sleeve in the actual and also a 10% increased heating environment. In both cases, the HSA was exposed at peak heating. To evaluate the worst thermal shock environment for the aeroshell, a

BLANK

BLANK

Figure 5.11. 4.2-17. Tension Modulus of Elasticity vs. Temperature AXF-5Q

Figure 5.11.4.2-18. Tensile Strength vs. Temperature AXF-5Q

Figure 5.11.4.2-19. Compressive Modulus of Elasticity vs. Temperature AXF-5Q

Figure 5.11.4.2-20. Compressive Strength vs. Temperature AXF-5Q

Figure 5.11.4.2-21. Poisson's Ratio vs. Temperature for Tension and Compression AXF-5Q

point approximately 0.5 seconds prior to peak heating, (7.5 seconds from 400 kft), was chosen to expose the aeroshell to the heating.

The trajectory was calculated using an initial velocity of 36333 ft/sec, an initial altitude of 400,000 feet and initial path angle of -90° with the HSA oriented in the side-on stable mode.

A two-dimensional thermal response/ablation model characterizing a 180° circumferential slice of the HSA including the clad, emissivity sleeve, aeroshell and retaining ring, was generated for use in the THTDA code. The 190-node model represents a center-section of the HSA and consists of 10 nodal locations about the circumference, each with 19 nodes in depth. No heat transfer is assumed in the axial direction due to the heating symmetry which exists.

Figure 5.11.4.2-22 presents the nodal breakdown and the heat transfer coefficients (h_c) used in the analysis. Heat transfer coefficients of $1000 \text{ BTU/hr-ft}^2 - ^{\circ}\text{F}$ were used in the clad/emissivity sleeve and clad/aeroshell interfaces, following calculations of contact pressure vs. decelerating force and correlating this information with data.

Aerodynamic heating shown in Figure 5.11.4.2-23 shows the recession history for the emissivity sleeve.

The total (thermochemical + mechanical) recession history of the emissivity sleeve was found using the thermal response and thermochemical ablation data obtained from THT-DA and applying it to the equation developed in Ref. 14.

For both the actual and 10% increased heating, emissivity sleeve removal times occur at the approximate HSA peak heating period. At this point it is assumed since the clad is at its melt temperature ($\sim 4020^{\circ}\text{F}$), that exposure of the aeroshell to the environment is immediate. From the results of the one-dimensional thermal stress calculations (Ref. 15),

SECTION A-A
IN-DEPTH
NODE BREAKDOWN

Figure 5.11.4.2-22. 2-D Thermal Stress Circumferential Nodal Locations

STEEN ANGLE REFERENCE DESIGN STUDY

THORNEL-50 ES REMOVAL

$V_0 = 36333 \text{ FPS}$ $\delta_0 = -90^\circ$

- ACTUAL HEATING -

Figure 5.11.4.2-23. Recession History for the Emissivity Sleeve

the most severe HSA thermal shock environment would be exposure at 7.5 seconds from 400 Kft. The free HSA thermal performance was evaluated, therefore, using exposure times of 7.5 seconds for both heating cases.

Figures 5.11.4.2-24 and 25 show the resulting indepth thermal response of the HSA using 7.5 second exposure time, for both the actual and 10% increased heating environments. There is only a slight difference in the peak temperatures of each case; i. e. E. S. maximum surface temperatures of 7100 vs. 7120°F. Figure 5.11.4.2-26 shows the indepth temperature response of the aeroshell for both heating environments at the point of maximum surface temperature and gradient. Here, too, the peak surface temperatures vary slightly, (6760° vs. 6825°F as do the gradients (4360° vs. 4415°F).

THERMAL STRESS ANALYSIS

A thermostructural analysis of the center-section of the aeroshell has been performed using the ORTHOSAFE computer program as in Section 5.11.4.2.2. The details of the structural model, calculation technique, assumptions, material properties, etc. are discussed in the following paragraphs.

Figure 5.11.4.2-27 shows the elastic stresses on the OD and ID of the aeroshell for the nominal heating case as a function of time while Figure 5.11.4.2-28 shows the corresponding margin of safety. Thermal shock produces the minimum margin of safety for the OD but the most critical region (lowest margin) is the ID and the worst time, 8.7 to 9.0 sec., is beyond the thermal shock period.

The stress distribution at the time of maximum stress in the nominal heating case is shown in Figures 5.11.4.2-29 and 30 for an elastic and plastic analysis respectively. Figures 5.11.4.2-31 and 32 give the same information for the increased heating case. A comparison of these results shows that the 10% higher heating produces the same thermal stress.

Figure 5.11.4.2-24. Steep Angle Reference Design Analysis In-Depth Temperature History at Stag Line Actual Heating

Figure 5.11.4.2-25. Steep Angle Reference Design Analysis In-Depth Temperature History Stag Line 10% Increased Heating

Figure 5.11.4.2-26. In-Depth Aeroshell Temperatures $t=8.0$ Seconds Max ΔT & Max Surface Temperature

Reference Design Case

$V_0 = 36,333$ fps $\gamma_0 = -90^\circ$

Nominal Heating

Aeroshell Thickness = .387"

Emissivity Sleeve Removal at 7.5 Sec.

Elastic Analysis

Time from 400 Kft - Sec.

Figure 5.11.4.2-27. Thermal Stress History

Time from 400 Kft - Sec.

Figure 5.11.4.2-28. Margin of Safety

- EMISSIVITY SLEEVE BURN-THRU @ 7.5 SEC.
- TEMPERATURE DISTRIBUTION @ 9.0 SEC.
- NOMINAL HEATING PROFILE
- ELASTIC ANALYSIS

Figure 5.11.4.2-29. Aeroshell Stress at Stagnation Point

- EMISSIVITY SLEEVE BURN-THRU @ 7.5 SEC.
- TEMPERATURE DISTRIBUTION @ 9.0 SEC.
- NOMINAL HEATING PROFILE
- PLASTIC ANALYSIS

Figure 5.11.4.2-30. Aeroshell Stress at Stagnation Point

Figure 5.11.4.2-31. Aeroshell Stress at Stagnation Point

- EMISSIVITY SLEEVE BURN-THRU @ 7.5 SEC.
- TEMPERATURE DISTRIBUTION @ 9.1 SEC.
- 10% HIGHER HEATING PROFILE
- ELASTIC ANALYSIS

Figure 5.11.4.2-32. Aeroshell Stress at Stagnation Point

For the nominal heating case, the plastic stress has been estimated as:

$$f_p = f_y + \frac{E_p}{E} (f_t - f_y) =$$
$$8200 + 1/3 (8390 - 8200) = 8263 \text{ psi}$$

Superimposing the pressure stress gives

$$f_p = 8200 + 1/3 [(8390 + 2100) - 8200] = 8963$$

for a margin of

$$MS = \frac{9250}{8963} - 1 = 3\%$$

Thermal stresses during cool down are shown in Figure 5.11.4.2-33. The maximum stress, about 2000 psi, occurs near 12 seconds. The margins of safety during cool down are all high.

5.11.4.3 FSA Reentry

An analysis has been made (Ref. 20) to determine the behavior of an FSA reentering as a free body after release from the HSA. The trajectory corresponds to release of an FSA from the HSA at the time of peak heating, 8.0 seconds, in an end-on; $\gamma_o = -90^\circ$, $V_o = 36,333 \text{ ft/sec.}$, reentry. Trajectories for the FSA were calculated assuming laminar and turbulent flow in the subsonic range. Heating rates, recession rates and thermal response calculations were made assuming a spinning FSA both with and without exothermic heating.

Trajectories were calculated with DYNMAS, for both laminar and turbulent flow in subsonic flight although the flow is expected to remain laminar. The initial trajectory conditions corresponded to release from the HSA at peak heating and were:

$$\begin{aligned} \text{Altitude} &= 110,460 \text{ feet} \\ \text{Velocity} &= 33,623 \text{ ft/sec. (Relative)} \\ \gamma &= -87.6^\circ \text{ (Relative)} \end{aligned}$$

Figure 5.11.4.2-33. Thermal Stress at Stagnation Point During Cool Down

The FSA was assumed to be spinning at a rate which is high enough for every location on the sphere surface to experience an average heating rate. This average was obtained by integrating the heating rate distribution over the sphere. In hypersonic and supersonic flow, 0 to 8.7 sec., Lee's distribution was used with the average obtained as (Ref. 21):

$$\dot{q}_{ave} = 26.3\% \dot{q}_{\text{stagnation point}}$$

In subsonic flow, the equations of Ref. 4 were used to obtain an average value

$$\dot{q}_{ave} = 50\% \dot{q}_{\text{stagnation point}}$$

The ratio of average to stagnation point heating should actually be a function of Reynolds number in subsonic flow but the functional dependence is weak so the value of 50% was used throughout the subsonic flight.

The influence of graphite oxidation on the heating rate was handled as detailed in Ref. 22 with the FSA Graphite Impact Shell assumed to have a reaction rate equal to that for AVCO 3D orthogonal. In the reaction rate regime, an iteration was required between the calculation of GIS surface temperature and heating rate. The resulting temperature and ablation heating terms are shown in Figures 5.11.4.3-1 and 2. Impact temperatures are:

	Turbulent Trajectory		Laminar Trajectory	
	No Ablation Heating Effect	With Ablation Heating	No Ablation Heating Effect	With Ablation Heating
GIS	1282 °F	2284 °F	742 °F	2092 °F
PICS	2532	2575	2172	2500
FUEL OD	2800	2810	2729	2783
FUEL	2946	2960	2944	2961

Figure 5.11.4.3-1. Ablation Heating - Laminar Trajectory

Figure 5.11.4.3-2. Ablation Heating - Turbulent Trajectory

The surface temperature with ablation heating remains high enough for recession to be diffusion limited to impact. For the cases with no ablation heating, recession has been evaluated using a 1900° F "cut off temperature" which is conservative. The results are:

Turbulent Trajectory	Laminar Trajectory
No Ablation Heating Effect 17.5 mils	17.6 mils
With Ablation Heating 86.1	102.3

5.11.4.4 Free RTG Reentry

In the reference design study, Section 5.11.4.2.2, the thermal response was evaluated with respect to the effects of the clad and E.S. for the steep angle superorbital trajectory ($\gamma = -90^\circ$, $V_o = 36333$ fps).

Another study was made to analyze the actual response of the aeroshell including not only the effects of the clad and E.S., but also those of the RTG converter, as the RTG reenters freely from 400 Kft at super-orbital velocities (36333 fps). The first step in this study was to determine the RTG breakup conditions.

5.11.4.4.1 RTG Breakup

The purpose of the RTG breakup study (Ref. 24) was to develop an insight into the capabilities of the MHW RTG as a re-entering body and thereby calculate the range of breakup altitudes and times to evaluate the HSA behavior when the HSA actually becomes a free re-entering body. This information is necessary to present a better understanding of the subsequent thermal shock environment, temperature response and ablation characteristics of the HSA in an actual re-entry. Previous analyses have focused on the HSA behavior as a free re-entry body at 400,000 ft.

The study defined the time and altitude bands at which breakup occurs for side-on stable, side-on spinning and tumbling RTG re-entry orientations. In the side-on stable mode, a

band about the nominal was defined, which accounts for fin-to-flow orientations and the converter foil/astoquartz insulation removal. A band was also generated for the tumbling mode due only to the varying fin-to-flow orientations, since the dynamic forces in both the tumbling and spinning situations cause almost instantaneous insulation removal. The spinning mode, however, has been shown to be impossible, but is presented for comparison purposes.

Approach

Three specific re-entry conditions were analyzed. They were:

1. The earth escape initial velocity re-entry, since this is the design trajectory ($V_0 = 36333$ fps, $\gamma = -6^\circ$ to -90° ,
2. The earth orbital decay re-entry, since this has been determined to be the most probable ($V_0 = 25708$ fps, $\gamma_0 = -0.1^\circ$, and
3. The LES 8/9 V- γ map extremity, since this is the most severe trajectory in the LES 8/9 mission ($V_0 = 34000$ fps, $\gamma_0 = -90^\circ$).

All of the above cases were analyzed for three RTG re-entry modes, namely: Side-on stable, side-on spinning and end-over-end tumbling. Nominal conditions for each orientation were used to formulate the breakup band and these conditions were the fin leading case for the side-on stable mode and the fin offset 30° case for the tumbling mode. Separate analyses were performed to define the bands which exist in both the stable and tumbling modes, since fin-to-flow orientations may vary in both cases and insulation removal time can be applied to the stable mode.

Initially, two converter/HSA heat transfer models were generated for the THT-D computer code: an 87-node model to characterize a 90° section of the assembly for the stable and tumbling cases since 90° was deemed sufficiently far away from the stagnation line so as not to affect the analysis, and a 36-node model representing 30° for the spinning case; both models taking advantage of the respective symmetries in the heating environments. Both models were 1.08 inches axially and center sections of the RTG. The heating outputs from ABTON along with the calculated distributions, were applied to these models only to find that the thermal inertial of the insulating blankets was far greater than that of the converter shell, indicating modeling of the interior components to be unnecessary for most cases. Thus,

models of the converter shell only, were developed and used to complete the analysis; a 13 node 90° section and a 6 node 30° section. See Figure 5.11.4.4-1 for a schematic of the 87 and 36 node models and note that the reduced node models are identical to these, upon removal of the thermoelectrics, composite insulation and the HSA.

The criterion used to signify loss of the converter shell varied with the specific RTG re-entry orientation. In the side-on stable orientation a converter shell melt of $\pm 20^\circ$ to 30° circumferentially from the stagnation line was considered sufficient for breakup. However, to bound the problem, the longest possible time has been calculated for melt and subsequent removal of the insulation, using a heat balance calculation which considered the aerodynamic heating, heat stored and heat re-radiated. Breakup in the tumbling case was considered to occur when ± 20 to 30° of the shell melts, at which time, due to the dynamic forces involved and the RTG motion, the insulation was assumed to be removed, thereby exposing the HSA. In the spinning case, melt of the fin root area was considered sufficient for complete RTG dissociation since this was occurring under each of the 6 fins. Once again dynamic forces in the spinning process were assumed to bring about immediate insulation removal and HSA exposure.

The results of the analysis are summarized in the curves in Figures 5.11.4.4-2 through 5. Figure 5.11.4.4-2 shows the nominal altitude and time breakup bands for the RTG orientations considered, and for the range of gamma angles in the design trajectories ($V_0 = 36333$ fps). Figure 5.11.4.4-3 presents the ratio of the instantaneous RTG stagnation line heat flux at breakup, to the peak RTG flux ($\dot{q}_{inst}/\dot{q}_{peak}$), and also the instantaneous RTG flux at breakup ($\dot{q}_{inst}$) over the range of gamma angles. Figures 5.11.4.4-4 and 5 present the bands that could exist for the stable and tumbling modes, due to variations in fin orientation for both the stable and tumbling modes and insulation melt and removal in the stable mode only. Finally, Figure 5.11.4.4-6 presents the results of the remaining two re-entry conditions (earth orbital decay and LES 8/9 V - γ map extremity) in tabular form.

Figure 5.11.4.4-1. RTG Breakup Study

Figure 5.11.4.4-2. Altitude and Time Breakup Bands

Figure 5.11.4.4-3. Ratio of the Instantaneous RTG Stagnation Line Heat Flux at Breakup

Figure 5. 11.4.4-4. Side-on Stable RTG Breakup Altitude Band and Flux Ratio-Time Bands

Figure 5. 11. 4. 4-5. Tumbling RTG Breakup Altitude Band and Flux Ratio-Time Bands

- LES 8/9 MISSION - V-GAMMA MAP EXTREMITY

- TRAJECTORY - $V_o = 34,000$ FPS @400 KFT.

$$- \gamma = -90^\circ$$

- RESULTS

ORIENTATION	BREAKUP TIME (SEC.)	BREAKUP ALT. (KFT)	INSTANT. FLUX ($\frac{\text{BTU}}{\text{SEC.} \cdot \text{FT}^2}$)	FLUX RATIO (%)
SIDE ON-STABLE	6.75	170.5	856.0	71.6
TUMBLING	7.00	161.8	966.0	89.2
SIDE ON SPIN	9.00	102.8	353.0	AFTER PEAK HEATING

- EARTH ORBITAL DECAY

- TRAJECTORY - $V_o = 25708$ FPS @400 KFT

$$- \gamma = 0.1^\circ$$

- RESULTS

ORIENTATION	BREAKUP TIME (SEC.)	BREAKUP ALT. (KFT)	INSTANT. FLUX ($\frac{\text{BTU}}{\text{SEC.} \cdot \text{FT}^2}$)	FLUX RATIO (%)
SIDE ON-STABLE	2000	268.9	36.38	51.7
TUMBLING	2340	289.8	17.53	24.4
SIDE ON-SPIN	2000	268.9	11.98	49.3

Figure 5.11.4.4-6. RTG Breakup Study

The results of the superorbital re-entries ($V_0 = 36333$ and 34000 fps) indicate that the side-on stable and spinning orientations bound the altitude and time breakup envelopes. Since these envelopes were considered quite wide, it was deemed necessary to look at the probable re-entry orientations resulting from the behavior of the RTG in hypersonic flow. Data supplied in Ref. 25 demonstrates that the trim points for a re-entering RTG in hypersonic flow are the end-on stable or end-over-end tumbling modes. Furthermore, even if the side-on re-entry occurs, a side-on spinning RTG will stabilize, making the spinning case impossible and thereby reducing the nominal breakup band to the narrow area bounded by the stable and tumbling modes (Figure 5.11.4.4-2).

The insulation removal times presented in Figure 5.11.4.4-4 indicate a wide variation from the nominal stable case. It is important to note that this insulation removal altitude is conservative due to the following reasons:

1. The insulation is a composite of 6 bundles of 10 layers of alternating astroquartz and molybdenum foil, in 180° staggered circumferential arcs. This makes it a relatively weak structure, incapable of withstanding substantial loads.
2. As the lower altitude boundary limit is approached, the dynamic pressure increases becoming considerable, thereby bringing about a loading situation in which the HSA will attempt to tear out of the foil due to the dynamic pressure forces acting upon the RTG/foil-remains acting against the inertial forces of the encapsulated HSA.

These reasons indicate that total insulation removal is likely to occur earlier, thus affecting a narrowing of the band.

Figures 5.11.4.4-4 and 5 point out the fact that the tumbling band is enclosed in the stable band for all re-entry parameters (time, altitude, and flux ratio). Therefore, it can be concluded that the stable envelope entirely bounds the situation and that HSA exposure will not occur outside of this band for the orientations analyzed thus far. Furthermore, the flux ratios will not exceed 72% of the RTG/stag max value for any orientation at any re-entry angle and 62% for any steep angle ($\gamma = 90^\circ$) re-entry.

RTG breakup during end-on reentry was evaluated in Ref. 27. The study defined the times and altitudes at which breakup, and subsequent release of the HSA, occur for the end-on orientation for the same trajectories analyzed for the RTG side-on.

A 13-node THT-D model characterizing a 30° section of the forward end portion of the RTG was used to calculate its thermal response. Figure 5.11.4.4-7 presents the model which includes the forward pressure dome flange, the RTG fin, and the forward mounting bracket for the spider structure. The converter skin aft of the mounting bracket was not modeled but the finning effect was accounted for by taking the shell as a fin and calculating its radiation effectiveness. Thermal contact coefficients (h_c) of $1000 \text{ BTU/Hr ft}^2 - ^\circ\text{F}$ were used over the entire contact areas at the dome/converter flange interface and the mounting bracket/converter shell interface, as well. In all calculations, the beryllium emissivity coating was assumed to "burn off" at 1200°F , changing the emissivity from 0.85 to 0.2.

In order to properly simulate the reentry breakup sequence, the dome melt point was determined prior to application of the THT-D model. It was calculated using the MSTTR computer code treating the dome as a thin-wall, applying the heating of body point 3 which is an average of the dome heating, and employing the emissivity change at 1200°F as mentioned above. The THT-D model was then run, starting all trajectories at 400,000 ft. and applying aerodynamic heating to the appropriately exposed surfaces and heating to the titanium mounting bracket at its pre-determined point of exposure (dome melt) to the flow. The analysis showed that the mounting bracket melted slightly before the flanges, thereby releasing the spider and insulation. At this point, it is assumed that since there no longer exists any positive containment of the HSA, and since the HSA has a higher ballistic coefficient ($\beta = 85 \text{ psf}$) than the RTG ($\beta = 59 \text{ psf}$), the HSA will separate from the converter shell and become a free reentry body. Further, it is felt that the disintegrating converter will impart a rotational motion to the HSA allowing it to seek its statically stable trim point in the side-on orientation. This requires a minimum rotational rate of approximately 1 rpm.

Figure 5.11.4.4-7. 30° Thermal Response Model End-on RTG Breakup Analysis

The results of the breakup study are summarized in Figures 5.11.4.4-8 and 9. Figure 5.11.4.4-8 shows the breakup times and altitudes for the design trajectories. It also shows the instantaneous RTG aerodynamic heat fluxes at breakup and a ratio of these fluxes to the peak trajectory fluxes had the RTG remained intact. Figure 5.11.4.4-9 is a comparison of the cold wall sphere heat fluxes at breakup for the three reentry orientations studies thus far--the side-on stable, tumbling and end-on orientations.

As seen in Figure 5.11.4.4-9, the breakup of the RTG in the end-on reentry orientation, occurs in the least severe heating environment of the three orientations. Thus from the standpoint of thermal shock one can bound the problem by the side-on stable (most severe) and end-on (least severe) orientations.

5.11.4.4.2 HSA Thermal Response After Release

Reference 23 evaluated the thermal response of the HSA at two separate release points both for a steep angle ($\gamma = -90^\circ$) and a shallow ($\gamma = -20^\circ$) trajectory. These release points characterize the converter breakup in the side-on stable and end-on reentry orientation. They were chosen for the analysis because they bracket the breakup results.

It was determined that for both the 90° and 20° path angle trajectories, the clad/emissivity sleeve was removed at the same point in each respective trajectory, regardless of the fact that they were released from the converter at different points along the trajectories. For the 90° trajectory, the clad and E.S. were removed at 106 Kft or slightly past the peak HSA heating. For the 20° trajectory, the clad and emissivity sleeve were removed at about 91 Kft, long after peak heating. In all cases the aeroshell was exposed to a much less severe thermal shock environment than that of the reference design study.

Figures 5.11.4.4-10, 11, 12, and 13 show the temperature histories for the emissivity sleeve, clad and aeroshell from the HSA release point from the RTG, well into the aeroshell cool-down phase. For cases I and II ($\gamma = -90^\circ$) the aeroshell experiences a lower maximum temperature and temperature gradient than the reference design. Cases III and IV show that the aeroshell experiences its peak temperature and peak gradient while the aeroshell

Figure 5. 11. 4. 4-8. End-On MHW RTG Breakup Altitudes & Times Design Trajectories

Figure 5.11.4.4-9. Cold Wall Heat Fluxes at Breakup for Various Re-entry Orientations

Figure 5.11.4.4-10. Case I - HSA Release from Side-on RTG

Figure 5.11.4.4-11. Case II - HSA Release from End-on RTG

$$V_{@400K} = 36333 \text{ FAS}$$

E.S. $\theta_{@400K} = -20^\circ$

Figure 5. 11. 4. 4-12. Case III - HSA Release from Side-on RTG

Figure 5.11.4.4-13. Case IV - HSA Release from End-on RTG

is still encased in the emissivity sleeve. At emissivity sleeve removal, the aeroshell cools, thus equalizing the gradients across the material.

Figures 5.11.4.4-14 and 15 show the in-depth temperatures of the aeroshell for each case studied, (the point of maximum surface temperature and gradient for cases I and II and for the maximum gradient for cases III and IV).

5.11.4.4.3 Thermal Stress

The previous section gives the results of thermal response calculations for an HSA after release due to break up of the RTG reentering as a free body. An analysis has been made (Ref. 19) of the thermal stress condition in the aeroshell for the case of an RTG reentering side-on $V_0 = 36,333$ fps, $\gamma_0 = -90^\circ$ (Case I of Ref. 23).

The analysis technique assumptions, model, etc. have been summarized in Section 5.11.4.2.1.

The Case I thermal response results show that RTG breakup occurs 5.5 seconds after the reentry (from 400 Kft) begins and that the emissivity sleeve burns off after 2.69 sec. of HSA exposure. The aeroshell experiences a thermal shock and reaches its peak temperature (5700°F), 3.3 sec. after RTG breakup.

Previous results have shown that the thermal shock is not critical and that the critical stress time occurs slightly after peak temperature. Results are tabulated below. The minimum margin of safety is 26%.

Time	ID		OD	
	Stress	Margin	Stress	Margin
3.5 sec.	6996 psi	29%	-7239	42%
3.9 sec.	7237 psi	26%	-7499	59%
4.1 sec.	7027 psi	31%	-7860	79%

Figure 5.11.4.4-14. In-Depth Aeroshell Temperatures at Max ΔT & Max Surface Temperature

Figure 5.11.4.4-15. In-Depth Aeroshell Temperatures at Max ΔT

5. 11. 4. 5 TRW Payload System Breakup Analysis

The P74-1 payload system breakup analysis was undertaken by TRW Systems, to determine the RTG environment during a simulated spacecraft reentry, Reference 35. The effort was aimed at determining the HSA performance during its most realistic reentry environment; namely with the spacecraft or spacecraft/transtage system. TRW Systems was assigned the task of analytically predicting the conditions under which the RTG becomes separated from the spacecraft and determining the heating imposed on the RTG up to the time of separation. The flow chart of Figure 5. 11. 4. 5-1 shows the analytical methodology used

Figure 5. 11. 4. 5-1. Analytical Methodology

by TRW to arrive at these results. Figures 5. 11. 4. 5-2 thru 5 show the sign convention and pitching moment variations with angle of attack generated by TRW.

A total of ten cases were analyzed, Table 5. 11. 4. 5-1; each defined by the USAF/SAMSO. Each reentry case constitutes a discrete set of conditions which includes definition of the reentering configuration and of the state vector at 400,000 ft.

It was the task of GE to determine when the HSA becomes a free body. This could occur during the RTG/Spacecraft attachment phase or free RTG phase of the reentry. The flow chart in Figure 5. 11. 4. 5-6 summarizes the overall analysis.

During the "attachment" phase, the heating was supplied by TRW in the form of a heating to the stagnation line of a side-on, 12.7" diameter cylinder undergoing the system trajectory. For all cases analyzed, except Case II, the heating was modified to account for the RTG with its fins off-set 30° to the flow, undergoing the specified angle of attack history. In Case II, the heating was modified using the heating factors to the end dome of a cylinder at angle of attack, found in Reference 36. The breakup analysis technique and criterion for the side-on and end-on cases were described in Section 5.11.4.4.1.

Figure 5.11.4.5-2. Coordinate System and Sign Convention

Figure 5.11.4.5-3. Pitching Moment Coefficient Vs. Angle of Attack of Payload System Plus Transtage

Figure 5.11.4.5-4. Pitching Moment Coefficient Vs. Angle of Attack of Payload System Plus Transtage.

Figure 5.11.4.5-5. Pitching Moment Coefficient Vs. Angle of Attack of LES 9

Table 5.11.4.5-1. Reentry Cases Analyzed

CASE	CONFIGURATION	CONDITIONS AT 400,000 FT. ALTITUDE		
		INERTIAL VELOCITY (FT/SEC)	INERTIAL FLIGHT PATH ANGLE (DEG)	ANGLE OF ATTACK (DEG.)
I	TRANSTAGE + SPACECRAFT + RTG'S (PRE-FIRST BURN)	25,700	- 0.2	+ 165.0
II	TRANSTAGE + SPACECRAFT + RTG'S (POWERED RE-ENTRY DURING FIRST BURN)*	28,000	- 16.0	+ 15.0
III	TRANSTAGE + SPACECRAFT + RTG'S (POST-FIRST BURN)	28,000	- 16.0	15 or less
IV	TRANSTAGE + SPACECRAFT + RTG'S (POST-FIRST BURN)	32,660	- 5.1 ($\gamma_{MIN.}$)	+ 165.0
V	TRANSTAGE + SPACECRAFT + RTG'S (POST-FIRST BURN)	32,762	- 4.2	+ 12.5
VI	TRANSTAGE + SPACECRAFT + RTG'S (POST-SECOND BURN)	33,500	- 82.0	+ 90.0
VII	LES 9 SPACECRAFT + RTG'S	33,500	- 40.0	+ 90.0
VIII	LES 9 SPACECRAFT + RTG'S	33,500	- 60.0	+ 90.0
IX	LES 9 SPACECRAFT + RTG'S	33,500	- 82.0	+ 90.0
X	LES 9 SPACECRAFT + RTG'S	36,000	- 40.0	+ 90.0

* 155 SEC. OF BURN REMAINING @ 400,000 FT.

Figure 5. 11.4.5-6. Analysis Flow Chart

If it was determined that the RTG remained integral until release from the spacecraft system, a new trajectory was calculated and a new heating rate was generated for the free RTG using the DYNMASS and ABTON computer codes as in Section 5.11.4.4.1, and the breakup analysis was continued. For analysis of the free RTG's, it was assumed in all cases that the RTG's would reenter side-on stable. This is the expected orientation for those cases in which the RTG's remain paired when they separate from the spacecraft since the paired RTG's are symmetrical. For the cases in which an RTG reenters singly, the analysis with a side-on orientation is conservative since this results in the earliest RTG break up and heat source exposure as shown in Section 5.11.4.4.1. The analyses were made with the fins offset 30° to the flow, another conservatism as shown in Section 5.11.4.4.1.

RTG and HSA trajectory, heating and thermal response analysis techniques and thermal models were the same as discussed in previous sections. The HSA was analyzed in the side-on orientation using the THT-DA program with the thermal model of Figure 5.11.4.2-1 (Clad, emissivity sleeve, and aeroshell with no intervals) since the first results were to be used in thermal stress calculations. The emissivity sleeve was removed by the combined effects of both mechanical and thermochemical ablation as applied in Section 5.11.4.4.2. References 28, 29 and 30 document the details of the analyses.

The thermal stress calculations are detailed in Reference 31. They were generally made at the time of maximum aeroshell ΔT since this is near the time of minimum margin of safety. In some cases, multiple thermal stress calculations were made.

The analysis techniques used for calculating the thermal stress in the aeroshell were the same as those used in Sections 5.11.4.4.2 and 5.11.4.4.3. The thermal stress analysis was a 2-D plane stress analysis using the ORTHOSAFE computer code. Stress margins of safety were calculated using the equation:

$$\text{MARGIN OF SAFETY} = \frac{\sigma_{\text{ULT}}}{\sigma_{\text{HOOP}}} - 1,$$

and expressed as percentages (X100%).

Table 5.11.4.5-2 shows the initial trajectory conditions and conditions at RTG release for the ten breakup cases:

Table 5.11.4.5-2. Conditions at RTG Release

CASE IDENTIFICATION					RELEASE CONDITIONS			
NO.	CONFIG.	V - FPS	γ - DEG	α - DEG	ALTITUDE - ft	V - FPS	γ - DEG	α - DEG
1	TS/SC	25,700	- 0.2	+ 165	266,000	25,770	- 0.17	+ 152
2	TS/SC	28,000	- 16.0	- 15	230,000	28,640	- 16.0	+ 15
3	TS/SC	28,000	- 16.0	0	190,000	27,750	- 15.7	+ 27 *
4	TS/SC	32,660	- 5.1		277,000	32,750	- 3.39	+ 152
					262,000	32,730	- 3.11	+ 138
51	TS/SC	32,762	- 4.2	+ 12.5	285,000	32,840	- 2.05	+ 152
					277,000	32,830	- 1.81	+ 138
6	TS/SC	33,500	- 82.0	+ 90.0	197,000	33,540	- 8.2	+ 52 T
7	LES 9	33,500	- 40.0	+ 90.0	130,000	31,760	- 82.2	+ 150
8	LES 9	33,500	- 60.0	+ 90.0	140,000	31,840	- 39.9	+ 158
9	LES 9	33,500	- 82.0	+ 90.0	130,000	31,460	- 59.2	+ 150
10	LES 9	36,000	- 40.0	+ 90.0	155,000	33,910	- 39.8	+ 158

* BASED ON PRELIMINARY C_m VS α DATA - REVISED VALUE IS +15 (SAME AS CASE 2)

5.11.4.5.1 Case I

Case I is a representation of a random orbital decay of the transtage plus payload system following a failure to ignite the Transtage for the first burn. The system reenters with an initial velocity of 25700 fps, path angle of -0.2° , and angle of attack of $+165.0^\circ$, (RTG's well behind parent system). Shortly after loss of the transtage tanks and engines, at an altitude of 341 Kft, the RTG's are released in pairs. RTG breakup and release of the HSA's occurred at 318 Kft followed much later by the emissivity sleeve burn-through at 205 Kft, (just prior to peak heating). Clad removal is assumed to occur instantaneously upon exposure to the aerodynamic heating since the clad temperature is near the melt temperature and would rise rapidly. Figure 5.11.4.5-7 shows the RTG and HSA heating, HSA thermal response, and the release points of the RTG and HSA.

No thermal stress calculations were made since the aeroshell ΔT is only about $150^\circ F$ and thermal stresses are negligible.

AT 400 KFT

$$V_0 = 25,700 \text{ FPS}$$

$$\gamma_0 = .2^\circ$$

$$\alpha_0 = 165^\circ$$

RTG RELEASE (PAIRED)
343 KFT

HSA RELEASE
318 KFT

CONCLUSIONS

- THERMAL STRESSES NEGLIGIBLE
MAX $\Delta T = 150^\circ\text{F}$
- INTACT IMPACT OF BARE AEROSHELL

Figure 5.11.4.5-7. RTG and HSA Heating, HSA Thermal Response, and Release Points of the RTG and HSA (Case I)

5.11.4.5.2 Case II

The transtage/spacecraft system initially reenters with a velocity of 28000 fps, (inertial), path angle of -16.0 degrees (inertial) and angle of attack of +15.0 degrees with 155 seconds of burn remaining in the transtage at 400,000 ft. The counterbalancing effect of the engine thrust and aerodynamic moment give the system its angle of attack of 15 degrees. The RTG's are released paired with the release point originally calculated to be at 191 Kft by TRW (Reference 37), and this was the point used in the GE analysis. Since then, however, the release point has been revised to 230 Kft by TRW, although no revision in the GE analysis was made since it was proven that the RTG will not expose the HSA in an end-on manner.

Figure 5.11.4.5-8 shows the RTG thermal analysis results for the RTG at an angle of attack of 15 degrees. The HSA is not released in a near end on manner even if the analysis is continued to 191 Kft. At 230 Kft, most of the end dome is still intact and the titanium mounting brackets are not exposed to the flow.

Figure 5.11.4.5-8. RTG Dome & Mounting Bracket Response
RTG/SC Phase of Reentry (Case II)

A thermal analysis was done for the side of the RTG, assuming it to be at an angle of attack of 15° down to 191 Kft then turning side on. RTG breakup occurs at 181.5 Kft exposing the heat source to near peak heating. Figure 5.11.4.5-9 shows the HSA stag line heating, the HSA thermal response, the separation points, the thermal response and stress safety margins.

5.11.4.5.3 Case III

Due to the similarity of this case to Case II, it was not analyzed in depth but was examined to confirm that the HSA is not exposed in the end-on orientation.

5.11.4.5.4 Case IV

Case IV represents a grazing reentry in which the path angle of the transtage plus payload system is slightly steeper than the minimum angle required for an earth-impacting reentry. The initial velocity is 32660 fps, path angle is -5.1° and angle of attack is $+165.0^{\circ}$ (RTG's well behind parent system). The upper RTG was released at 277 Kft and broke up at 268 Kft releasing the HSA. Long afterward, at 175 Kft, the emissivity sleeve and clad were removed and the bare HSA proceeded down to impact. Figure 5.11.4.5-10 shows the heating, thermal response histories, release points and safety margins for this case.

Breakup of the lower RTG was not determined but it should occur soon after release of the RTG at 262 Kft. The lower HSA would be released later than the upper HSA and would therefore experience less severe conditions since the emissivity sleeve remains in place until well past peak heating.

5.11.4.5.5 Case V

Case V represents a multiple skip of the Transtage plus Payload System resulting from the initial path angle being slightly less shallow than that required for an earth impacting reentry. The initial conditions of velocity, 32762 fps, path angle, -4.2° and angle of attack of $+12.5^{\circ}$ resulted in two skips followed by an earth impacting reentry of the HSA.

AT 400 KFT
 $V_0 = 28,000$ FPS
 $\gamma_0 = 16^\circ$

RTG RELEASE (PAIRED)
 191 KFT
 (LATER REVISED TO 230 KFT)

HSA RELEASE
 181.5 KFT

$\alpha_0 = 15^\circ$ CONSTANT

POWERED REENTRY

HSA - STAG LINE HEATING RATE

HSA TEMPERATURE HISTORY

Figure 5.11.4.5-9. Case II Conclusions: Intact Impact of Cladded HSA

AT 400 K FT

$$V_0 = 32,660 \text{ FPS}$$

$$\gamma_0 = 5.1^\circ \text{ (MIN } \gamma \text{)}$$

$$\alpha_0 = 165^\circ$$

CONCLUSIONS:
INTACT IMPACT
OF BARE HSA

UPPER RTG RELEASED
277 K FT

LOWER RTG RELEASED
262 K FT

UPPER HSA RELEASE
268 K FT

Figure 5.11.4.5-10. Case IV Heating, Thermal Response Histories, Release Points and Safety Margins

TRW's analysis determined that the upper RTG was released at 285 Kft and the lower RTG at 277 Kft during the first reentry. The upper RTG was selected for analysis as in Case IV and it was found to break up and release the HSA at 278 Kft during the first reentry. The HSA skipped out of the atmosphere with 0.019 inches of material remaining from the original 0.050 inches in the emissivity sleeve, and swept into an eccentric elliptical orbit. Since this orbit time was assumed to be long enough for the HSA to reach steady state with the space environment, new temperatures were calculated for the start of the second reentry. A one-dimensional temperature calculation of the partially-ablated emissivity sleeve, clad, aeroshell and retaining ring, was performed. The following table shows the temperatures calculated for use as initial conditions for the second reentry:

ORBITAL STEADY STATE TEMPERATURES AFTER FIRST SKIP OUT		
COMPONENT		TEMPERATURE - °F
Emissivity Sleeve	-	850
Clad	-	890
Aeroshell	-	1535
Retaining Ring	-	1950

The initial trajectory data used for the second reentry was the data at the point of the first skip out, (400,000 ft), with the direction of the velocity vector and the sign of the path angle appropriately reversed. The emissivity sleeve burned through and the clad was removed at an altitude of 243 Kft, during the second skip.

As in the first orbit, the steady state temperatures of the HSA (aeroshell + retaining ring) were calculated for the second orbit to be used as inputs for the third reentry.

ORIBITAL STEADY STATE TEMPERATURES AFTER SECOND SKIP OUT	
COMPONENT	TEMPERATURE °F
Aeroshell -	856
Retaining Ring -	1208

The HSA proceeded to impact on this reentry. Figures 5.11.4.5-11 through 13 show the heating, the HSA thermal response, the release points and safety margins for all reentries.

5.11.4.5.6 Case VI

Case VI represents a steep angle reentry of the Transtage plus Payload System following the second burn. The initial velocity is 33500 fps, path angle is -82° and angle of attack is 90° , (RTG's broadside to the flow). The RTG's were released paired at 197 Kft., breaking up shortly thereafter, releasing the HSA's at 180 Kft. Figure 5.11.4.5-14 shows the heating, thermal response, release points and safety margins for this case.

5.11.4.5.7 Case VII

Here the LES 9 spacecraft was chosen by TRW for analysis in the reentry simulation on the basis of its higher ballistic coefficient. The LES 9 is assumed to reenter with the same initial trajectory parameters as Case VI without the UHF and K band antennae and seek a stable trim point at an α of 152° . A 30° melt occurred on the upper RTG at 140 Kft (7.0 seconds from 400 Kft), thus affecting a converter disassembly while still attached to the spacecraft since the RTG separation point is 130 Kft, (Table 5.11.4.5-2). Since, at this angle of attack, the spacecraft/RTG ballistic coefficient (β) is 55.5 psf while the HSA β is 89.3 psf, a coupling of the aerodynamic and inertial forces tending to topple the HSA from the converter remains. Figure 5.11.4.5-15 shows the heating, thermal response, release points and safety margin for this case.

5.11.4.5.8 Case VIII

Case VIII is a reentry of the LES 9/RTG system with an initial velocity of 33500 ft./sec., inertial path angle of -60° , and trim angle of 128° . A converter breakup point was again

AT 400 KFT
 $V_0 = 32,762 \text{ FPS}$
 $\gamma_0 = -4.2$
 $\alpha_0 = 12.5^\circ$

UPPER RTG RELEASED
 285 KFT

UPPER HSA RELEASED
 278 KFT

EMISSION SLEEVE
 AND CLAD REMOVED
 243 KFT

LOWER RTG RELEASED
 277 KFT

3RD SKIP IS FINAL REENTRY OF BARE HSA

Figure 5.11.4.5-11. Case V - Release Points

Figure 5.11.4.5-12. Case V - Heating, HSA Thermal Responses & Safety Margins

CONCLUSION: IMPACT OF INTACT AEROSHELL

Figure 5.11.4.5-13. Case V - Heating, HSA Thermal Responses

AT 400 KFT

 $V_0 = 33,500 \text{ FPS}$ $\gamma_0 = -82^\circ$ $\alpha_0 = +90^\circ$ RTG RELEASE (PAIRS)
197 KFTHSA RELEASE
180 KFT

Figure 5. 11.4.5-14. Case VI - Heating, HSA Thermal Responses, Release Points & Safety Margins

AT 400 KFT
 $V_0 = 33,500$ FPS
 $\gamma_0 = 40^\circ$
 $\alpha_0 = 90^\circ$

TOP HSA RELEASE
 140 KFT

Figure 5.11.4.5-15. Case VII - Heating, HSA Thermal Responses, Release Points & Safety Margins

found using a 30° melt from the stagnation line as the criterion. Figure 5.11.4.5-16 shows the results of the analysis.

5.11.4.5.9 Case IX

Case IX very closely resembled another case (Case VII), so no formal analysis was performed.

5.11.4.5.10 Case X

Case X is a reentry of the Payload System, (LES 9 + RTG's), only, with an initial velocity of 36,000 fps, path angle of -40.0° and angle of attack of $+90.0^{\circ}$, (RTG's broadside to the flow). In this case, the upper RTG broke up at an altitude of 180 Kft before the calculated separation point of 155 Kft, due to the nature of the reentry and angle of attack history. The emissivity sleeve and clad were removed after peak heating at 87 Kft. Figure 5.11.4.5-17 shows the convective and radiative heating, the thermal response history, release points and safety margins for Case X.

5.11.4.5.11 Summary

Table 5.11.4.5-3 is a summary of the analysis performed on the breakup cases.

TOP HSA RELEASE
152 KFT.

AT 400 KFT

V_0 33,500 FPS

Y_0 -60°

90°

CONCLUSIONS:

INTACT
IMPACT OF
CLADED
HSA

Figure 5. 11.4. 5-16. Case VIII Analysis Results

AT 400 KFT

$$V_0 = 36,000 \text{ FPS}$$

$$\gamma_0 = -40^\circ$$

$$\alpha_0 = 90^\circ$$

TOP HSA RELEASE
155 KFT

CONCLUSIONS:
INTACT
IMPACT OF
BARE AEROSHELL

Figure 5.11.4.5-17. Case X - Heating, Thermal Response, Release Points & Safety Margins

Table 5.11.4.5-3. Analysis Summary

CASE	HSA RELEASE FROM	CLAD MELT	T_{OD}^{OF}	THERMAL STRESS		M.S. %	CONCLUSION
				T_{ID}^{OF}	σ_T PSI		
I	PAIRED FREE RTG'S	YES					NEGLIGIBLE σ_T
II	PAIRED FREE RTG'S	NO	2833 3292	2387 2844	2129 1625	323 478	INTACT
III	SAME CONDITIONS	AS CASE II					
IV	SINGLE FREE RTG (UPPER)	YES	4234	3939	168	7102	INTACT
V	SINGLE FREE RTG (UPPER)	YES	3324	3133	522	1777	INTACT
VI	PAIRED FREE RTG'S	YES	3165	2439	2950	189	INTACT
VII	ATTACHED UPPER RTG	NO*	3067	2394	2855	215	INTACT
VIII	ATTACHED UPPER RTG	NO	2964 3064	2397 2688	2475 1694	264 449	INTACT
IX	SAME AS CASE VIII						
X	ATTACHED UPPER RTG	YES	3678	2596	3193	188	INTACT

* TEMPERATURE APPROACHED CARBON-IRIDIUM EUTECTIC

5.11.5 AERODYNAMIC STUDIES

There were three major aerodynamic efforts.

1. Generator Aerodynamic Analysis (Ref. 32)
2. HSA 6 Degree of Freedom Trajectory Simulation (Ref. 33)
3. Aerodynamic Testing (Ref. 34)

5.11.5.1 Generator Aerodynamic Analysis

Reference 32 actually considered many different configurations including spacecraft and transtage as well as RTG's. Since the TRW report (Ref. 26) was used to determine the RTG environment until separation of the RTG from the spacecraft, only the results in Ref. 32 for a free RTG will be discussed.

A Newtonian analysis of the RTG configuration is shown in Figure 5.11.5.1-1. The effects on the end-plates, fins and shading are considered in the moment predictions using Newtonian force characteristics, which cannot be assumed as being any more than an approximation of representative stability characteristics. The predominant effect on stability is the unsymmetrical end plates at either end of the cylinder. The leeward shading effect on the larger, aft end-plate produces a stabilizing moment in the low-to-mid angle of attack range. This analysis suggests that the RTG configuration will be statically stable end-on either in the forward or backward trim orientations, with a more predominant forward trim condition, but will not be stable in the broadside orientation in hypersonic flight.

The consideration of a spin condition about the longitudinal axis for the RTG configuration is pertinent to break-up analyses, and it can be assumed that fins will tend to damp out any roll tendencies induced by asymmetric cross flow over the body. The geometric symmetry of fins about the longitudinal axis and the aerodynamic forces acting on them will provide a strong resistance to any induced spin condition.

Figure 5.11.5.1-1. MHW-Static Stability Analysis RTG Configuration

A summary of the RTG characteristics is given in Table 5. 11. 5. 1-1.

Table 5. 11. 5. 1-1. RTG Aerodynamic Characteristics

Configu- ration	Attitude	Overall Length (In.)	C. G. Lo- cation Aft of Nose(In.)	Weight (Lbs.)	Continuum C_D	Continuum B (lb. /ft ²)	Free Md C_D	Free Mol. B (lb. /ft ²)
RTG	End-on	22.9	11.45	84.9	1.7	59	4.6	18
	Side-on				1.2	28	2.0	17
	Tumbling				.7	46	1.6	22

5. 11. 5. 2 HSA Trajectory Simulations

Six degrees of freedom (6-DOF) trajectory simulation studies were conducted, Ref. 33, using updated aerodynamic inputs which are based on wind tunnel results obtained on the Heat Source Assembly. The computer simulation studies were performed using the most severe superorbital re-entry conditions to demonstrate the hypersonic motion behavior and to determine if a stable end-on or near end-on attitude is possible. Initial reentry conditions used in the hypersonic trajectory simulation studies were as follows:

Re-entry Altitude, $h = 400,000$ Ft.

Re-entry Path Angle, $\gamma = -6^\circ$.

Re-entry Velocity, $V = 35,163$ Ft. /Sec.

Re-entry Angle of Attach = $\alpha = 0^\circ$ (End on).

Pitch Rate: Variable Input: $\omega_0 = 1^\circ$ to 10° /Sec.

HSA Weight: 44.6 Lbs. (Homogeneous)

The HSA center of gravity location affects the aerodynamic response in terms of the induced trim characteristics and stability. Therefore, a worst case offset of 1/8-Inch ($C_1 = .01$ Ft.) forward on the longitudinal axis was used in most of the simulation studies. A zero c.g. offset ($C_1 = 0$ ft.) was used with a 10° /Sec. pitch rate.

5.11.5.2.1 Results

The initial computer run was for a nominal c.g. location at the centroid of the body ($C_1 = 0$) and using an initial pitch rate of 10 degrees/sec. The HSA diverged to $\alpha = 180^\circ$ and then attained a stabilized broadside ($\alpha = 90^\circ$) oscillation at about 300,000 Ft. and remained in that altitude throughout the trajectory as shown in Figure 5.11.5.2-1.

The second trajectory simulation run was made with a 1 degree-per-sec. pitch rate and with a worst case c.g. offset. The results are shown in Figure 5.11.5.2-2, and indicate that with the low pitch rate and c.g. offset the HSA will stabilize at the weak trim point of $\alpha = 21^\circ$ in hypersonic flight.

The pitch rate was then increased to 2 degrees-per-second and the results are shown in Figure 5.11.5.2-3. Again the HSA hung up at the $\alpha = 22^\circ$ trim point, however, the pitch rate was almost sufficient to drive the body past this weak stable trim point. This inferred

Figure 5.11.5.2-1. Results of Initial Trajectory Simulation

Figure 5.11.5.2-2. Results of Second Trajectory Simulation

Figure 5.11.5.2-3. Trajectory Simulation with Pitch Rate = 2 Deg/Sec

that if the HSA has an initial pitch rate greater than two (2) degrees-per-second it will not trim near end-on and will assume a broadside orientation in hypersonic flight with the "worst-case" configuration c. g. asymmetry and trajectory conditions. To verify this assumption an additional trajectory run was made with a 3 degrees-per-second pitch rate and does indicate the HSA will diverge immediately to about $\alpha = 80^\circ$ and will stabilize in the broadside orientation at about 300,000 ft., as shown in Figure 5.11.5.2-4.

The 6-DOF trajectory simulation studies indicate that with a matrix of "worst-case" conditions, it is possible for the HSA to stabilize near end-on in hypersonic flight. Realistically, the necessary combination of these conditions during re-entry is highly improbable. The assumed initial re-entry conditions of zero angle of attack with a low pitch rate (≤ 2 deg./sec.) is unlikely, especially with a longitudinal c. g. offset. Even with a low pitch rate and a longitudinal c. g. offset, the initial angle of attack would have to be within $= 0 \pm 30^\circ$ during re-entry for the HSA to stabilize at $\alpha = 22^\circ$. A pitch rate of 2 degrees-per-second (120° per minute or 0.3 RPM) is unlikely when considering the manner in which it must have separated from a parent vehicle or RTG. The pitch rates imparted at separation should continue essentially undamped until re-entry and the possibility of such a low rate is extremely remote.

5.11.5.3 Aerodynamic Testing

A series of wind tunnel tests, Ref. 34, were conducted to aid in the evaluation of the MHW-HSA re-entry aerodynamic performance. The objectives of the ground tests were to determine the hypersonic, transonic, and subsonic stability characteristics and to provide adequate performance data for trajectory simulation studies of flight motion behavior.

Tests were conducted in the NASA-Ames 3.5-ft. Hypersonic Tunnel at Mach 7.3 to evaluate the hypersonic static stability and drag over a range in angles of attack from 0° to 90° . The primary objective in the hypersonic tests was to obtain an accurate assessment of end-on and near-end-on stability characteristics. Results indicated static instability end-on, with stable trim points at angles of attack of 22° and 90° .

Figure 5.11.5.2-4. Angle of Attack vs. Altitude, Pitch Rate = 3 Deg/Sec

Transonic static stability and drag data were obtained in the NASA-Ames 6 x 6 ft. supersonic tunnel at Mach numbers from 0.6 to 1.8 and at angles of attack from 0° to 90° . The test results verified pretest predictions of end-on static stability below Mach 1.1 at small angles of attack ($\alpha = \pm 5^\circ$). Above Mach 1.1, near-end-on stability is indicated with a stable trim point converging from 18° at $M = 1.8$ to about 10° at Mach 1.2. The significance of the end-on and near-end-on stability must be evaluated in trajectory simulation studies, which also consider the accompanying dynamic instabilities and the resultant inertial characteristics to determine if these statically stable trim conditions are meaningful.

Dynamic stability tests were conducted in the AEDC 4-ft. Transonic Tunnel (PWT-4T) at Mach numbers from 0.2 to 1.3. Dynamic stability data were required primarily to support the trajectory simulation studies and to provide aerodynamic damping data over the range in angle of attack covering statically stable trim conditions. The test results confirmed pretest predictions of dynamic instability at zero angle of attack below Mach 1.0. An even more significant observation is that the dynamic instability appears to move with the near-end-on stable trim points above Mach 1.0, which indicates that the aerodynamic phenomena producing the statically stable trim condition, is dynamically destabilizing.

5.11.5.3.1 Hypersonic Static Stability Test Results

Static force and moment data were obtained in the NASA-Ames 3.5 ft. hypersonic tunnel primarily to accurately determine the hypersonic stability characteristics between 0° and 40° angle of attack. Data were obtained at Mach number of 7.3 and at a Reynolds number of 1.2×10^6 , based on model diameter.

The results of the hypersonic tests are shown in Figures 5.11.5.3-1 through 5. Basic aerodynamic coefficients are presented as a function of angle of attack, and plotted symbols are representative of data points obtained with the six support installations for model attitude orientation.

The static stability characteristics are evaluated from the variation in pitching moment coefficient and aerodynamic center of pressure with angle of attack, as shown in Figures 5.11.5.3-1 and 2, respectively. The moment data clearly indicates an unstable moment

Figure 5.11.5.3-1. MHW-HSA Hypersonic Static Stability Characteristics

Figure 5.11.5.3-2. MHW-HSA Hypersonic Static Stability Characteristics

Figure 5.11.5.3-3. MHW-HSA Hypersonic Static Stability Characteristics

Figure 5.11.5.3-4. MHW-HSA Hypersonic Static Stability Characteristics

Figure 5.11.5.3-5. MHW-HSA Hypersonic Static Stability Characteristics

variation for the end-on attitude. The reversal in the moment curve at about 15° angle of attack reflects the bow shock attachment at the edge of the front face, creating a stabilizing trend which results in an aerodynamically stable trim point at about a 22° angle of attack. This stable trim condition is also observed in Figure 5.11.5.3-2 where the aerodynamic center of pressure moves aft of the center of gravity. It is also observed in Figure 5.11.5.3-2 that no more than a 1% static margin is obtained over a small range in angle of attack, which indicates a relatively weak stable trim condition. Once the cylinder gets above the unstable trim point at 30° it will continue to diverge to 90° where it should maintain a preferred stable broadside flight orientation. The actual HSA response (motions) to these aerodynamic characteristics will depend on the vehicle inertial characteristics and flight dynamic perturbations, as determined in trajectory simulation studies.

5.11.5.3.2 Transonic/Subsonic Static Stability Test Results

Static force and moment data were obtained in the NASA-Ames 6 x 6-ft. Supersonic tunnel to determine the transonic and subsonic static stability and drag. The objectives of this test program were to verify the aerodynamic predictions of end-on static stability at subsonic speeds and to obtain sufficient data for trajectory studies of subsonic flight behavior.

Some sample test results, selected for discussion purposes are shown in Figures 5.11.5.3-6 through 17. Variations in the aerodynamic coefficients with angle of attack are shown to have some highly irregular discontinuities in the subsonic and transonic flow regime. Most of the irregularities in the data are considered to be valid aerodynamic characteristics.

5.11.5.3.3 Subsonic Results

The variation in pitching moment coefficient with angle of attack is shown in Figure 5.11.5.3-6 at $M = 0.8$, as being a representative sample of the subsonic static stability characteristics. Similar results are shown at $M = .6, .7, .9$, and $.95$ in the test report (Ref. 34). The characteristic slope of the pitching moment curve indicates end-on static stability at zero angle of attack. This end-on stability, which begins at $M = 1.10$ and persists at subsonic speeds, is evident only in the angle of attack envelope of about $\pm 5^{\circ}$ about $\alpha = 0$. Although the normal forces are extremely small in this angle of attack range

(Figure 5.11.5.3-8), the center of pressure is very far aft as shown in Figure 5.11.5.3-7, primarily because of a negative normal force distribution acting at the nose. At $M = .90$ this effect is large enough to produce a negative normal force coefficient slope at $\alpha = 0^\circ$.

The discontinuity in the moment curve (Fig. 5.11.5.3-6) at $\alpha = 52^\circ$ is observed at Mach numbers of 0.6, 0.7, and 0.8, and is attributed to leeward flow separation. Although the separation is undoubtedly affected in some degree, it is a characteristic flow phenomena, which will tend to increase the destabilizing moment between 50° and 70° angle of attack.

The corresponding center of pressure, normal force, and axial force data at $M = 0.8$ are shown in Figures 5.11.5.3-7, 8 and 9, respectively. The center of pressure data in Figure 5.11.5.3-7 dramatically shows the stability at low angles of attack, which in effect reflects the negative normal force distribution at the nose. The resultant normal force variation with angle of attack is shown in Figure 5.11.5.3-8 which also demonstrates the effect of the front face to the point that the total normal force is nearly zero between 0 and 5 degrees angle of attack. With the substantial moment variation and nearly zero normal force, the center of pressure goes to infinity near $\alpha = 0$ as expected. The effect of leeside flow separation at 42° is also observed in the center of pressure data, indicating a shift forward as expected with separation toward the aft end on the cylinder.

5.11.5.3.4 Transonic Results

A sampling of transonic data at $M = 1.2$ is shown in Figures 5.11.5.3-10 through 5.11.5.3-13. These results exhibit significantly different aerodynamic characteristics than the subsonic data, and again the data indicate irregular variations with angle of attack. A complete reversal in the end-on static stability occurs between Mach 1.10 and 1.20. The effect of the formation of a bow shock at $M = 1.2$ changes the pressure distributions at the nose, eliminating the negative normal force distribution phenomena observed in subsonic flow. This effect tends both to increase the total normal force (as shown in Figure 5.11.5.3-12) and move the center of pressure well-forward of the c.g. location on the body (Figure 5.11.5.3-11) at small angles of attack. The result is a highly unstable pitching moment variation between -10° and $+10^\circ$ angle of attack.

At about $\alpha = 10^\circ$, a highly irregular discontinuity is observed in the moment data, which shows an almost instantaneous reversal in stability. The data obtained in Mach numbers 1.5 and 1.8 also reflect this discontinuity but to a lesser degree as the Mach number increases. The interpretation is that this discontinuity in the pitching moment curve is a valid data measurement and is attributed to the bow shock-edge-attachment, which has an instantaneous effect on the pressure distributions over the entire body. The result is an increase in the normal force and a significant movement aft in center of pressure, with a statically stable trim point at $\alpha = 10^\circ$.

Figure 5.11.5.3-10 also indicates a slight discontinuity in the moment data at about $\alpha = 25^\circ$. Measured support interference effects demonstrate that this irregularity is caused by support interference, which is also evident at $M = 1.10$ and disappears at $M = 1.50$.

The center of pressure data at $M = 1.2$ is shown in Figure 5.11.5.3-11. The characteristics of the low angle data, with center of pressures located at about 20% of length, are in direct contrast to the subsonic data shown in Figure 5.11.5.3-7. Also, the sudden movement aft in center of pressure at $\alpha = 10^\circ$ reflects the change in the pressure distributions with bow shock attachment on the leading edge of the cylinder.

Variations in normal force coefficient and axial force coefficient with angle of attack at $M = 1.2$ are shown in Figures 5.11.5.3-12, and 13, respectively. The discontinuity at about $\alpha = 10^\circ$ is also indicated in the normal data in Figure 5.11.5.3-12. In addition to the increase in normal force at the discontinuity, the normal force coefficient derivative (C_N) also increases substantially from about .036 per deg. to about .065 per degree. Scatter in both the normal force and axial force data at the higher angles are indicative of support interference effects, which are apparent from the data overlap of the different support systems.

5.11.5.3.5 Supersonic Results

Data obtained at $M = 1.80$ are shown in Figures 5.11.5.3-14, 15 and 16, and are considered representative of the supersonic aerodynamic characteristics. The moment data shown in

Figure 5.11.5.3-14 continue to reflect the end-on static instability and the discontinuity at $\alpha = 10^\circ$, to a much lesser degree. However, the stable trim point has moved to about $\alpha = 18^\circ$. The pitching moment characteristics at $M = 1.80$ are beginning to approach similarity with the hypersonic data shown in Figure 5.11.5.3-1 and, accounting for differences in plotting scales, the moment coefficient variations (stability and trim points) and magnitudes are comparable. The variations in normal force and axial force coefficients with angle of attack at $M = 1.80$ are indicative of the more stabilized conditions of low supersonic flow in which support interference effects and flow irregularities are significantly reduced.

5.11.5.3.6 Supersonic Drag Results

Subsonic drag data were obtained on the HSA configuration at reduced Reynolds numbers with an $\alpha = 90^\circ$ orientation in an effort to measure boundary layer transition effects on drag. The additional drag data were obtained at Reynolds numbers of 0.60, 0.45, and 0.3×10^6 based on cylinder diameter and at Mach numbers of .8, .7, and .6. The latter condition of each represented the tunnel lower limits. The drag data indicated transition was not achieved, presumably because the Mach number was not low enough to obtain critical flow conditions and because the tunnel turbulence level was too high, which in effect reduces the critical Reynolds number.

The variation in the "broadside" drag coefficient with Mach number in the subsonic-transonic regime is shown in Figure 5.11.5.3-17. The drag coefficient is reduced here in terms of projected area ($L \times d$) for comparison purposes with other test data and with two-dimensional flow empirical predictions. The drag data obtained on the HSA configuration was considerably lower than predicted and did not exhibit the classical transonic drag rise that is predicted by two-dimensional empirical techniques.

The effects of Reynolds number on "broadside" drag at subsonic speeds is shown in Figure 5.11.5.3-18. The subsonic drag on right circular cylinders, perpendicular to the flow, is highly dependent on both the Mach number and Reynolds number. As the Mach number decreases, the drag coefficient at supercritical Reynolds numbers will continue to decrease to a minimum prior to transition. This minimum drag level will be dependent on surface

roughness, on fineness ratio, and on the turbulence factor of the flow (in the case of wind tunnel data). When both the critical Mach number and critical Reynolds number conditions are achieved the boundary layer separation conditions change, causing a significant increase in drag, as shown in Figure 5.11.5.3-18. The MHW-Ames data at Mach numbers .8, .7, and .6 indicates that subcritical conditions were not achieved. Superimposed on the data in Figure 5.11.5.3-18 are the ideal 2-dimensional flow data and the low speed supercritical (turbulent separation) and subcritical (laminar separation) levels in drag coefficient. Providing the local boundary layer conditions are obtained, transition could occur at higher Mach numbers and therefore at higher drag coefficients than indicated by 2-dimensional flow predictions. Transition Reynolds number will vary with flow turbulence, roughness, wall temperature, and possibly fineness ratio (especially with low L/D's, where end-effects alter the base flow). The subcritical (laminar) level ($C_D = 1.2$) appears sacred from available literature, however most of this information is based on 2-dimensional data and this laminar drag level could be affected by fineness ratio as observed in supercritical flow.

5.11.5.3.7 Transonic/Subsonic Dynamic Stability

Free oscillation dynamic damping data were obtained in the AEDC 4-ft Transonic Tunnel to evaluate the transonic and subsonic dynamic stability characteristics. The objectives of these tests were to verify the predictions of dynamic instability and to obtain some quantitative damping measurements as a function of angle of attack and Mach number, specifically for trajectory simulation studies of flight motion behavior.

The dynamic damping data are shown in Figure 5.11.5.3-19 to be highly erratic as a function of both angle of attack and Mach number, which reflects the difficulty in both obtaining and analyzing the test results. Data were obtained at different Reynolds numbers in an effort to obtain meaningful data traces, and the results presented are the best interpretation of the data at this time. Time-consuming curve-fitting of the data traces are required to complete the data reduction, and therefore these results are considered "preliminary".

The data to demonstrate consistent trends with angle of attack and Mach number that can be used effectively in 6-degree-of-freedom trajectory studies. Variations with angle of attack

are shown in Figure 5.11.5.3-19 and suggest a close correlation with the static stability data. The correlation is the extent of dynamic instability, which coincides exactly with the statically stable conditions over a relatively small range in angle of attack. At Mach numbers less than $M = 1.0$ the HSA is dynamically unstable at small angles of attack, becomes stable between $\alpha = 5^\circ$ and $\alpha = 8^\circ$ and appears to go unstable again about $\alpha = 20^\circ$. At transonic Mach numbers about $M = 1.0$, the body is dynamically stable at small angles of attack, but goes unstable at higher angles of attack, which appears consistent with the variations in statically stable trim conditions between 10° and 22° with increasing Mach number.

Variations in dynamic damping with Mach number are shown in Figure 5.11.5.3-20 for $\alpha = 0^\circ$ and for $\alpha = 4^\circ$ and 10° . Variations in Reynolds number are also included in the $\alpha = 0^\circ$ data, which is more indicative of the ability to obtain consistent measurements than a meaningful Reynold number effect. The data obtained at $\alpha = 0^\circ$ are consistent with pretest predictions and indicates dynamic instability below Mach 1.0. The variations in the data with Mach number at $\alpha = 4^\circ$ and 10° demonstrate how erratic the dynamic stability characteristics are in the transonic flow regime. The most consistent trend in both Figure 5.11.5.3-19 and Figure 5.11.5.3-20 is that the dynamic instability appears to follow the statically stable trim point in terms of both Mach number and angle of attack. Although the phenomena is not predictable, it is not unusual either because of the interdependencies of the static and dynamic stability derivatives.

Figure 5.11.5.3-6. MHW-HSA Subsonic Static Stability Characteristics

Figure 5. 11. 5. 3-7. MHW-HSA Subsonic Static Stability Characteristics

Figure 5.11.5.3-8. MHW-HSA Subsonic Static Stability Characteristics

Figure 5.11.5.3-9 . MHW-HSA Subsonic Static Stability Characteristics

Figure 5.11.5.3-10. MHW-HSA Transonic Static Stability Characteristics

Figure 5.11.5.3-11. MHW-HSA Transonic Static Stability Characteristics

Figure 5.11.5.3-12. MHW-HSA Transonic Static Stability Characteristics

Figure 5.11.5.3-13. MHW-HSA Transonic Static Stability Characteristics

Figure 5.11.5.3-14. MHW-HSA Supersonic Static Stability Characteristics

Figure 5.11.5.3-15. MHW-HSA Supersonic Static Stability Characteristics

Figure 5.11.5.3-16. MHW-HSA Supersonic Static Stability Characteristics

Figure 5.11.5.3-17. MHW-HSA Subsonic-Transonic-Drag Characteristics

Figure 5.11.5.3-18. MHW-HSA Subsonic Drag Variation with Reynolds Number

Figure 5.11.5.3-19. MHW-HSA Transonic/Subsonic Dynamic Stability

Figure 5.11.5.3-20. MHW-HSA Transonic/Subsonic Dynamic Stability

REFERENCES

1. Knight, D., "MHW Heat Source Operational Temperature", G. E. PIR ISAO 4618, 9/6/73.
2. Knight, D., "Operational Temperatures for Coated Iridium PICS and Clad", G. E. PIR ISAO 4661, 10/2/73.
3. Quinn, J. E., "Transient Response of MHW HSA when Removed from Mound Chill Blocks", G. E. PIR ISAO 4319, 1/4/73.
4. Quinn, J. E., "HSA Temperature Profile Prediction for MRC Thermal Gradient Test", G. E. PIR ISAO 4521, 6/11/73.
5. Knight, D., "Analysis of MRC Δ T Test with Walk Through HSA", G. E. PIR ISAO 4728, 11/6/73.
6. Quinn, J. E., "MHW-HSA Temperatures for Assembly, Test and Normal Operation", G. E. PIR ISAO 4428, 3/28/73.
7. Quinn, J. E., "MHW Heat Source Assembly Temperature Profiles and Transients - Launch to Orbital Operation", G. E. PIR ISAO 4544, 7/5/73.
8. Quinn, J. E., "Outer Clad - Emissivity Sleeve Temperature Penalties for Various Clad/ Sleeve Temperature Levels", G. E. PIR ISAO 4446, 4/10/73.
9. Knight, D., "Pad Abort Thermal Analysis for the Updated Safety Analysis Report (USAR)", G. E. PIR ISAO 4971, 5/2/74.
10. Multi-Hundred Watt Radioisotope Thermoelectric Generator Program Preliminary Safety Analysis Report, Vol. II, GESP-7075, July 1972.
11. Pigeon, J. G., "Post Impact Burial Analysis of FSA and PISA", G. E. PIR ISAO 4747, 11/19/73.
12. Knight, D., "Minimum Gamma Aerothermal Analyses with Iridium Coatings/ with Increased Heating/with a Bare Aeroshell, G. E. PIR ISAO 4784, 12/10/73.
13. Tantino, D. C., "Thermal Analysis for the Reference Design Steep Angle Trajectories", G. E. PIR ISAO 4617, 9/5/73.
14. Perini, L. L., "Review of Graphite Ablation Theory and Experimental Data", ANSP-M-1, December 1971, Page 29.
15. Quinn, J. E., G. E. ISAO, 20B12, Personal Communication.

REFERENCES (CONTINUED)

16. Ungvarsky, D., "Stresses in Aeroshell for Side-On Reentry", G. E. PIR ISAO 4481, 5/8/73.
17. Zucker, A., "Aeroshell Thermal/Stress Analysis", G. E. PIR ISAO 4546, 7/5/73.
18. Knight, D., "Reference Design Thermal Stress", G. E. PIR ISAO 4777, 12/5/73.
19. Knight, D., "Thermal Stress Results - HSA Release from a Reentering RTG", G. E. PIR ISAO 4783, 12/10/73.
20. Knight, D., "FSA Reentry after Release at Peak Heating in a Steep Angle Reentry", G. E. PIR ISAO 4739, 11/15/73.
21. Tantino, D. C., "Heating on a Spinning Sphere in Hypersonic Flow", G. E. PIR ISAO 4494, 5/16/73.
22. Knight, D. and Knechtel, E., "Graphite Oxidation at Low Temperature in Subsonic Air", AIAA Paper No. 73-735.
23. Tantino, D. C., "HSA Thermal Response with Converter, Clad and Girdle Effects", G. E. PIR ISAO 4648, 9/24/73.
24. Tantino, D. C., "MHW RTG Breakup Study", G. E. PIR ISAO 4502, 5/21/73.
25. Hobbs, R. and Corbett, S., "MHW Generator Aerodynamic Analyses", G. E. PIR ISAO 4328, 1/11/73.
26. Orbital Reentry Breakup Analysis Report P74-1 Payload System, TRW D00604, November 9, 1973.
27. Tantino, D. C., "MHW RTG Breakup Study for the End-On Orientation", G. E. PIR ISAO 4551, 7/9/73.
28. Tantino, D. C., "Thermal Performance of the MHW RTG for TRW's Cases I, IV, V, VI and X, G. E. PIR ISAO 4820, 1/10/74.
29. Pigeon, J., "MHW RTG Thermal Performance for TRW Simulated Reentry Case VII", G. E. PIR ISAO 4778, 12/6/73.
30. Tantino, D. C., "MHW RTG Thermal Performance for TRW Simulated Reentry Cases II and IV", G. E. PIR ISAO 4731, 11/8/73.

REFERENCES (CONTINUED)

31. Knight, D. , "HSA Thermal Stress Calculations for Safety Break Up Cases, G. E. PIR ISAO 4904, 2/28/74.
32. Hobbs, R. and Corbett, S. , "MHW Generator Aerodynamic Analyses", G. E. PIR ISAO 4328A, 5/29/73.
33. Hobbs, R. B. , "HSA-Hypersonic 6-DOF Trajectory Simulation", G. E. PIR ISAO 4655, 10/1/73.
34. Hobbs, R. B. , "MHW-HSA Aerodynamic Performance Test Report", 9151-001, August 1973.
35. Bell, J. M. (TRW System's Group), "Preliminary Subsystem Design Analysis Report/ Payload System Breakup Analysis", CDRL A 027, July 27, 1973.
36. Knight, D. D. , "Aerodynamic Heating on the MHW Heat Source/Test Results", G. E. PIR ISAO 4432, 2 April 1973, pgs. 19 and 29.
37. Bell, J. M. (TRW System's Group), "Preliminary Subsystem Design Analysis Report/ Payload System Breakup Analysis", CDRL A 027, July 26, 1973.
38. Final Report, Safety Test S-2 Liquid Propellant Fire Thermochemical Effects, GEMS 413, July 1973.
39. Metallurgical Society Conference Series "Refractory Metals and Alloys", Vol. II.
40. Quinn, J. E. , "MHW-PICS Reentry Temperature Profile During Reentry".

BLANK

GENERAL ELECTRIC

Space Division /

Headquarters: Valley Forge, Pennsylvania ☐ Daytona Beach, Fla. ☐ Cape Kennedy, Fla.
☐ Evendale, Ohio ☐ Huntsville, Ala. ☐ Bay St. Louis, Miss. ☐ Houston, Texas
☐ Sunnyvale, Calif. ☐ Roslyn, Va. ☐ Beltsville, Md.