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**AEROSPACE
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SAFETY**

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SANDIA LABORATORIES QUARTERLY REPORT
AEROSPACE NUCLEAR SAFETY PROGRAM
APRIL 1 THROUGH JUNE 30, 1970

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AEROSPACE NUCLEAR SAFETY DEPARTMENT 9510

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SC-PR-70-435

**SANDIA LABORATORIES QUARTERLY REPORT
AEROSPACE NUCLEAR SAFETY PROGRAM
APRIL 1 THROUGH JUNE 30, 1970**

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July 1970

ABSTRACT

This report describes research, development, support, and test activities in Sandia Laboratories Aerospace Nuclear Safety Program from April 1 through June 30, 1970.

Key words: SNAP-27, Pioneer, Brayton, DART

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SUMMARY

SNAP-27

Trajectories were computed for the GLFC reentering from the aborted Apollo 13 mission; the trajectories were used to supply AEC/SEPO with preliminary locations for stationing sampling aircraft.

PIONEER/VIKING

The average impact velocities of four different RTG heat source shapes were determined by helicopter free-fall drop tests at the Tonopah Test Range. Seventeen velocity and reentry angle pairs were selected as initial conditions for Pioneer heat source aerothermodynamic analyses. In the embrittlement HEAT facility test of the heat source, heat shield burnthrough occurred, but the fuel capsule appeared to have survived the test in excellent condition.

ISOTOPE BRAYTON HEAT SOURCE

Activities on the IBHS during May and June were mainly related to organization of the IBHS Task Force and information procurement. Terminal velocities were calculated for ablated and unablated configurations and for the heat source capsule alone. Free molecule, continuum, and transition hypersonic and the subsonic ballistic coefficients were calculated for the unablated IBHS reentering crosswise with the flat face forward. A two-dimensional thermal model of the proposed IBHS was constructed and used to predict the thermal response during a side-on-stable orbital decay reentry.

DART MINITHRUSTER

A series of simulated launch abort tests was begun to investigate the behavior of the DART capsule when it is subjected to postulated launch abort overpressures, fragments, fire, and impact. Terminal

velocities were calculated for five configurations of the minithruster drop test models; four each of the five configurations were drop tested at the Tonopah Test Range.

FLIGHT TEST STUDIES

Plans are proceeding for participating in the second Athena H research and development flight which is presently scheduled for February 1971; six heat sources will be flown in a piggyback payload.

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RADIOISOTOPE SYSTEMS SAFETY SUPPORT

SNAP-27

GLFC Lunar Return Thermal and Ablation Analyses (D. W. Larson)

An analysis was conducted to determine the thermal response of the Graphite Lunar Module Fuel Cask (GLFC) for a side-on-stable, 38,000 fps, -6.25 degree lunar return reentry. The analysis indicated that the beryllium secondary heat shield would melt over a 60-degree arc near the stagnation line. A refinement in the thermal model is required to correctly predict the peak temperature of the Haynes capsule clad; however, preliminary indications show that the clad will also melt over an arc of approximately 30 degrees near the stagnation line.

An ablation analysis of the GLFC was conducted for a high-speed lunar return atmospheric reentry (38,000 fps, -6.25 degree reentry angle). The results indicated a maximum surface temperature of 5400°F and a total graphite surface recession of 0.11 inch.

Apollo 13 GLFC Aerodynamic Analysis (H. R. Spahr)

On April 8, 1970, AEC/Houston requested that hypersonic ballistic coefficient data for the GLFC as a function of altitude be supplied for several GLFC attitudes. These data were needed April 9, 1970, so that NASA could calculate the GLFC impact point location for abort trajectory analyses prior to the Apollo 13 flight. The requested aerodynamic models were developed for crosswise, random tumbling, planar tumbling, and end-on orientations. These four models correspond, respectively, to the minimum ballistic coefficient, the expected ballistic coefficient, the maximum realistic ballistic coefficient and the absolute maximum ballistic coefficient.

The necessary information from these aerodynamic models was transmitted verbally to AEC/Houston. The complete aerodynamic models were transmitted to NASA/Houston as supplementary data for use in case of Apollo aborts.

Apollo 13 GLFC Abort Trajectory Analysis (H. R. Spahr)

At the request of AEC/SEPO on April 14, 1970, trajectories were computed for the GLFC reentering from the aborted Apollo 13 mission. The first trajectory computed was a -6.25 degree reentry angle, 38,000 fps "fast return" trajectory. Two sets of three "normal" reentry trajectories were computed. Each set of trajectories was based on a different set of input data (as mid-course corrections were made) and contained trajectories for the GLFC reentering crosswise, random tumbling, and planar tumbling.

The two sets of trajectories were used to supply AEC/SEPO personnel with the requested preliminary locations, relative to the NASA predicted impact point, where sampling aircraft should be positioned.

TRANSIT

Launch Abort Models (L. L. Keller)

Launch abort models have been developed only to the extent necessary to critique the test plans written by TRW/AI. The Fragmen-tation Test Plan was revised to include a high velocity (1190 fps) fragment representing the remains of the FW-4S Igniter Assembly. The Overpressure Test Plan was revised to call for a larger block of TNT (290 lb) so that the test items would be exposed to a more representative peak static overpressure.

Calorimeter Test Program

The Transit calorimeter test was initiated following the Pioneer Heat Source embrittlement test. The initial check run indicated that additional smoothing of the raw data was required in the data reduction program due to noise in the thermocouple output. The second check run resulted in a malfunction, and the 8-inch nozzle required some shop repair. This work was completed, and the test will be completed at the NASA/Houston plasma arc tunnel.

PIONEER/VIKING

RTG Heat Source Drop Tests (J. L. Entrekin)

Eighteen units representing four different reentry ablation profiles were drop tested at the Tonopah Test Range. The primary objective of the tests was to obtain terminal velocities and reduced trajectory data for the different shapes. A secondary objective was to obtain photographic coverage of the impact craters, measurement of the craters, and a brief general description of soil conditions at the various impact points. The different shapes were as follows. Four units, which represented an Earth orbital decay side-on stable reentry, were 6.50 inches long ablated 0.20 inch on one face of the hexagon with a 0.406-inch radius on all adjacent edges and corners and had a nominal weight of 10.328 pounds. Six units, which represented a multiple orbit side-on stable reentry, were 6.50 inches long ablated 0.40 inch on one face of the hexagon with a 0.625-inch radius on all adjacent edges and corners and had a nominal weight of 10.105 pounds. Six units, which represented a superorbital side-on spinning reentry, were 6.50 inches long ablated 0.050 inch on all faces of the hexagon with a 0.875-inch radius on all edges and corners and had a nominal weight of 10.015 pounds. Two units, which represented the unablated heat source, were 6.50 inches long and 3.50 inches across the faces of the hexagon with all edges and corners sharp and had a nominal weight of 10.461 pounds. All units were fabricated to duplicate the size, shape, weight, center of gravity, and moments of inertia of the actual heat source. All drops were made from a UH-IF helicopter flying at an altitude of 15,000 feet msl and traveling at a forward speed of approximately 40 mph.

The free-fall flight of all units was predominantly in a tumbling spinning mode. Average impact velocities were: (1) 300 fps for the Earth orbit decay side-on stable reentry shape, (2) 312 fps for the multiorbit side-on stable reentry shape, (3) 405 fps for the superorbital side-on spinning reentry shape, and (4) 268 fps for the unablated shape. The units were all impacted near or on the Yucca target which is firm, dry, sandy, soil impregnated with small rocks. Impact craters ranged from under 1/2 square foot and 5 inches deep to 1-1/2 square feet and 1 foot deep.

Aerodynamic Force Tests (H. R. Spahr)

Preliminary NASA/Ames hypersonic continuum drag data for ablated and unablated Pioneer heat sources were transmitted verbally to TRW. Terminal velocity data from drop tests of ablated and unablated Pioneer heat sources were also transmitted. These data were used by TRW in their short time scale analysis of the effects of several possible design modifications on the Transit and Pioneer heat sources.

Final data from the NASA/Ames hypersonic continuum flow aerodynamic force tests were received. The data on the three heat source configurations are being used to develop aerodynamic models for the heat source.

A Sandia representative attended part of the NASA/Langley low density aerodynamic force tests of the heat source on June 2, 1970. Preliminary data from the runs completed on this test were received.

Reentry Meeting (S. McAlees/H. R. Spahr/D. W. Larson)

Representatives from Sandia attended the Reentry Meeting of the Pioneer Launch Vehicles Systems Safety Studies Subgroup at Convair/San Diego on April 23, 1970. Isotopes, Inc., and Sandia presented summaries of the planned reentry analyses for Pioneer. The input data required from Convair for these reentry analyses were then defined.

Reentry Data (H. R. Spahr)

Detailed documentation defining the reentry data to be supplied by Convair for the Pioneer Data Submittal Plan was completed. The documentation was transmitted to AEC/SEPO for review and retransmittal to the chairman of the Pioneer Launch Vehicles Systems Safety Studies Subgroup. This completed the Sandia action item from the Reentry Meeting of April 23, 1970.

Convair Phase I Analysis (H. R. Spahr)

A Sandia representative attended the design review for the Convair Phase I launch vehicle analysis at NASA/Lewis on June 3, 1970. The Phase I launch vehicle data, received at this meeting, are currently being reviewed.

Aerodynamic and Trajectory Analyses (H. R. Spahr)

Seventeen velocity and reentry angle pairs were selected as initial trajectory conditions for Pioneer heat source aerothermodynamic analyses. These initial conditions include orbital decay, prompt reentries, the boundary between prompt and multiple revolution reentries, and both parabolic and hyperbolic multiple revolution reentries.

On April 8, 1970, AEC/SEPO requested that Sandia investigate the possibility of running the arc-jet test of the Pioneer heat source with the heat source in an end-on orientation. Aerodynamic models were developed for the Pioneer heat source end-on, without ablation and with an assumed hemispherically ablated end. These aerodynamic models were used to compute orbital decay trajectories to bound the reentry environment which would have to be simulated if the heat source were tested in the end-on orientation.

Preliminary initial trajectory conditions for three parabolic velocity multiple revolution Pioneer heat source reentries were transmitted verbally to Isotopes, Inc. These initial conditions were transmitted by Isotopes, Inc., to NASA/Ames for use as initial conditions for 6-degree-of-freedom trajectories.

Embrittlement Evaluation in HEAT Facility (D. E. Randall)

The heat source was assembled without the tantalum felt compliance pads or the TZM compliant cups. The orbital decay reentry test was conducted May 19. The fuel capsule was preheated to 1580°F over a period of 4 hours and held at this temperature for an additional 1-1/2 hours prior to the reentry pulse. The preheating was accomplished with the test chamber filled with N₂ at 650 mm Hg (Albuquerque ambient = 626 mm Hg) and helium continuously supplied to the heat shield interior. At the time the test chamber was evacuated prior to starting the reentry pulse, the fuel capsule temperatures climbed from 1650° to 1680°F.

The heating pulse along the nozzle axis provided an integrated heat load of 67 percent of the continuum orbital decay heat load and 80 percent of the orbital decay integrated square root of the model impact pressure. These conditions resulted in a heat shield surface

recession of 0.208 inch on the nozzle axis and 0.240 inch (heat shield burnthrough) at points approximately 2 inches either side of the nozzle axis. The fuel capsule visually appears to have survived the test in excellent condition; there is no evidence of clad melting or any other damage apparent. A helium leak test of the clad vent hole before and after the heat pulse provided positive evidence of hole venting. The capsule is being disassembled at LASL for evaluation of the T-111 strength member condition.

HEAT Facility Embrittlement Test Analysis (D. W. Larson)

A one-dimensional thermal model of the Pioneer heat source was constructed and used to predict the thermal response of the Pt-Rh clad in an orbital decay heat pulse in Sandia's HEAT facility. Results indicate that the Pt-Rh clad will melt if either no insulator or a Carbitex insulator is used to replace the pyrolytic graphite in the heat source.

A CMA run was made to determine the expected ablation during the HEAT facility embrittlement test. Results indicated an expected ablation of 0.127 inch. In the actual test, the maximum graphite recession was approximately 0.24 inch, causing total graphite removal at two locations on the heat source stagnation line. The discrepancy between predicted and measured ablation has not been resolved.

Reentry Thermal Analyses (D. W. Larson)

A two-dimensional thermal model of the Pioneer heat source was constructed and used to perform a scoping study of several heat source design variations. The model was used to determine the operational, reentry, and impact temperatures of the heat source. Temperatures were determined for the cases where the three-layer pyrolytic graphite insulator was replaced with (1) a beryllium sleeve, (2) a titanium sleeve, (3) a combination pyrolytic graphite and titanium sleeve, (4) a single-layer pyrolytic graphite sleeve, and (5) a double-layer pyrolytic graphite sleeve. The reentry temperatures were determined for an orbital decay, side-on-stable (flat forward) reentry. The results are shown in Table I.

TABLE I
Pioneer Temperatures From Two-Dimensional
Thermal Model for Side-on-Stable Orbital Decay

Design Condition (Air Gaps) Material	Operational Temperature (°F)	Peak Reentry Temperature (°F)	Impact Temperature (°F)
0.2-inch Beryllium Sleeve*			
Beryllium	1350	3749 (melted)	2121
Pt-20 Rh	1532	3371	2284
T-111	1556	3199	2326
Ta-10 W	1582	3078	2372
0.2-inch Titanium Sleeve			
Titanium	1336	4055 (melted)	2001
Pt-20 Rh	1652	3490 (melted)	2433
T-111	1686	3357	2481
Ta-10 W	1724	3271	2536
0.067-inch PG Sleeve and 0.133-inch Titanium			
Titanium	1580	3468 (melted)	2322
PT-20 Rh	1830	3100	2595
T-111	1860	3030	2642
Ta-10 W	1894	2989	2691
Two 0.067-inch PG Sleeves			
PT-20 Rh	1900	3134	2504
T-111	1927	3062	2563
Ta-10 W	1970	3016	2620
One 0.067-inch PG Sleeve			
PT-20 Rh	1677	3530 (melted)	2326
T-111	1712	3396	2407
Ta-10 W	1761	3310	2519
*Fuel loading for this case was low (450 Watts thermal); therefore operational and impact temperatures are low.			

ISOTOPE BRAYTON HEAT SOURCE

Isotope Brayton Heat Source Feasibility Study (IBSFS) (L. A. Hanchey)

The IBSFS is to provide the AEC with an in-depth risk assessment of the Isotope Brayton System as proposed for use on the Space Station. It is intended that safety problems be identified and recommendations relative to this application be included in the study. The IBSFS is

directed by SNS and managed by AEC through Sandia Laboratories using a Task Force type approach. The Task Force includes members from NASA (OART, LeRC, MSFC, MSC, ARC), LASL, NUS, SNS, and Sandia. The study started in May and is to be completed within 4 months, with a final report in draft form submitted to SNS by August 31.

Activities on the IBSFS during the first month of the study were mainly related to organization and information procurement. General definitions of the MSFC baseline mission Space Station design, launch vehicle, and Advanced Logistic System (ALS) were received. Information on two ALS vehicles being considered by MSC was also received. Sandia began work in the areas of defining accident environments and heat source response to these environments and is also actively working with the Radiological Working Group. General Electric started work in support of Sandia in the areas of prelaunch analyses, secondary explosive source definition and modeling, heat source thermal analyses, and reentry analyses.

Abort Models (L. L. Keller)

Preliminary definition of the launch vehicle (INT-21) was obtained from Marshall Space Flight Center. Estimated launch abort models are being prepared to support the feasibility study. The launch abort fireball model will be the same as used for the Saturn V.

Terminal Velocities (A. C. Bustamante)

Terminal velocities were calculated for ablated and unablated heat source configurations and also for the capsule alone. These velocities were calculated as a function of weight for 6000 feet and sea level (Figures 1 and 2). The data shown in these figures are based largely on experimental data, with proper corrections made for Reynolds Number effects. The ablated heat source configuration data shown in Figure 1 are based on an orbital decay reentry.

Orbital Decay Trajectory Initial Conditions (S. McAlees/ H. R. Spahr)

At the request of NASA/Ames, complete documentation was provided to NASA/Ames on computer code HRS004 which computes initial trajectory conditions for objects decaying from circular orbits.

Figure 1. Isotope-Brayton heat source terminal velocity

Figure 2. Isotope-Brayton capsule (alone) terminal velocity

Aerodynamic and Trajectory Analyses (H. R. Spahr)

Free molecule and continuum hypersonic ballistic coefficients were calculated for the unablated IBHS reentering crosswise flat-on. The transition flow hypersonic ballistic coefficients were calculated using computer code HRS015. The subsonic ballistic coefficient was also calculated for this configuration.

The above aerodynamic data, with initial trajectory conditions from computer code HRS004, were used to compute an orbital decay trajectory. This trajectory supplies the input data for the ablation and thermal response analyses of this configuration and orientation.

The ascent abort trajectory analysis for the launch vehicle is being conducted by McDonnell Douglas. The abort trajectory data that Sandia needs from this analysis and the rationale being used to select "worst cases" were discussed with McDonnell Douglas personnel.

Reentry Thermal Analyses (D. W. Larson)

A two-dimensional thermal model of the proposed IBHS was constructed and used to predict the thermal response during a side-on-stable orbital decay reentry. The results indicate that, with vacuum gaps external to the capsule, the heat source will have a peak Pt-Rh clad temperature of 3383°F and a peak T-111 strength member temperature of 3215°F when a 0.2-inch Carbitex sleeve insulator is used. The analysis indicated a peak temperature of 2940° and 2818°F for the Pt-Rh clad and T-111 strength member, respectively, when a zirconium oxide foam-nickel thermal switch was used as an insulator. The thermal model is presently being used to predict the reentry and impact temperatures of the heat source for the above cases with air gaps replacing the vacuum gaps. The model is also being used to determine these temperatures with pyrolytic graphite insulators of varying thicknesses.

DART MINITHRUSTER

Launch Abort Tests (F. L. Baker)

A series of simulated launch abort tests was started to investigate the behavior of the DART capsule (Figure 3) when it is subjected to postulated launch abort overpressures, fragments, fire, and impact.

Two capsule designs were used in the tests. The two designs were identical with the exception of the fuel can which was TZM alloy in one and 50 percent by weight Mo-50 percent by weight Re alloy in the other. Units 1, 2, and 3 were TZM fuel cans; Units 4, 5, and 6 were molybdenum-rhenium fuel cans.

Figure 3. DART minithruster capsule

Overpressure tests at approximately 600 and approximately 1200 psi were performed. The Sandia 2-foot shock tube was used for the tests with two capsules per shot, i.e., one each of the two designs. The capsules were heated to 500°F which is the anticipated launch pad temperature.

In the 600-psi shots, the graphite heat shield was fragmented on both heat sources. A crack could be seen visually in the TZM strength member of Unit 1. Dye penetrant inspection revealed cracks in both ends of Unit 4. X-rays revealed no cracked fuel disks in either capsule.

In the 1200-psi shots, the graphite was fragmented on both units. The Unit 5 capsule survived relatively intact; however, dye penetrant inspection shows both ends of the capsule to be badly cracked. Unit 2 was broken (Figure 4), and the solid-solution cermet simulant (SSCS) disks were released with probably four of the six disks being broken. The TZM strength member was shattered, and only a few small pieces of it were recovered. The fragment test was performed with the bare capsule, Unit 4, which previously had been subjected to the 600-psi overpressure test. The fragment was a piece of 0.063-inch 2014-T6 aluminum.

which was oriented to strike the impact-capsule dome edge-on (Figure 5). The capsule was heated to 500°F, and the impact velocity was approximately 1500 fps. The TZM strength member was shattered (Figure 6). The 50 percent by weight Mo-50 percent by weight Re alloy fuel can remained intact and contained the SSCS fuel (Figure 7). X-ray examination showed all the fuel disks to be shattered. A postmortem examination will be made to determine the extent of the disk breakage.

Figure 4. Unit 2 after 1200-psi test

Figure 5. Fragment test setup

Figure 6. Recovered DART capsule pieces
from fragment test

Figure 7.
Postfragment test Mo-Re
fuel can

Terminal Velocities and Subsonic Ballistic Parameters
(A. C. Bustamante)

Terminal velocities were computed for five configurations of the LASL minithruster drop test models at sea level and three different altitudes--2600, 4000, and 6000 feet (Tonopah altitude). The five configurations analyzed are shown in Figure 8. In addition, the subsonic ballistic parameters for these configurations were computed and are given in Table II along with their terminal velocities. Preliminary analyses of Tonopah drop test data for the above configurations show good agreement with the predicted velocities. It appears from the drop test data that the motion of the cylindrical configurations tends to be between an end-on attitude and a tumbling mode. For the flat plates, the motion seems to be between a side-on ($\alpha = 90^\circ$) attitude and a tumbling (autorotating) type motion.

Figure 8. Minithruster configurations analyzed

Drop Tests (J. L. Entrekin)

Twenty units consisting of five different shapes supplied by LASL were drop tested at Tonopah Test Range for the DART minithruster programs. The objective of the tests was to obtain impact velocities for the units and documentary movies of the units during flight. The different configurations were: (1) 4 units 4.63 inches in diameter by 4.65 inches long with sharp corners and a nominal weight of 11.06 pounds; (2) 4 units 3.59 inches in diameter by 3.90 inches long with sharp corners and a nominal weight of 4.90 pounds; (3) 4 units 3.59 inches in diameter by 3.90 inches long with a 0.25-inch radius on the corners and a nominal weight of 4.87 pounds; (4) 4 units 2.75 inches square by 1.0 inch thick with sharp corners and edges and a nominal weight of 0.625 pound; and (5) 4 units 2.75 inches square by 1.0 inch thick with a 0.25 inch radius on the four corners and a nominal weight of 0.618 pound (Figure 8). All units were fabricated to duplicate the shape, size, weight, center of gravity, and moments of inertia of the actual units. All drops were made from a C-47 fixed-wing aircraft flying at 15,000 feet msl altitude and at approximately 100 mph forward speed.

During descent, the units assumed a predominantly tumbling mode. Unfortunately the magnetic tape recorder which was recording the trajectory data had a power failure after drop number 4 so no reduced data are available. Impact velocities were calculated from radar plots and Mitchell camera films. These may be considered reasonably accurate since the methods correlated well with each other and they have correlated well with reduced data in the past. Average velocities were: Units 1-4: 311 fps; Units 5-8: 260 fps; Units 9-12: 284 fps; Units 13-16: 109 fps; Units 17-20: 115 fps. Most of the drops were documented by motion picture films.

Thermal Analysis (D. W. Larson)

An analysis was conducted to determine the energy input required to maintain the thruster heaters at 500°F in a shock tube facility. The results indicate that approximately 800 watts would be required.

Reentry Heating Tests (D. E. Randall/J. E. Deveney)

On June 24, 1970, Los Alamos Scientific Laboratory supplied five DART minithruster heat sources to Sandia Laboratories for reentry heating tests. Two of the units contained fuel pucks of solid solution cermet simulant (SSCS). The remaining three units contained a molybdenum heat sink. Tests were performed on June 24, 25, and 29. All units were subjected to a simulated orbital decay heating pulse. All tests units performed very well during testing, and the maximum measured graphite recession was 0.183 inch. Unit 3 was tested end-on to the flow, and Unit 5 was preheated to 1550°F. Test results are summarized in Table III.

TABLE III

DART Minithruster Orbital Decay Reentry Tests

<u>Test Date</u>	<u>Unit Number</u>	<u>Capsule Material</u>	<u>Test Orientation</u>	<u>Material Loss Graphite Ablator (g)</u>
June 24	1	Sink*	Side-on-stable	71
June 24	2	Sink*	Side-on-stable	72
June 25	3	Sink*	End-on-stable	50
June 25	4	SSCS**	Side-on-stable	70
June 29	5†	SSCS**	Side-on-stable	75
* Molybdenum heat sink.				
** Solid solution cermet simulant.				
† Preheated to 1550°F.				

SAFETY TECHNOLOGY

ADVANCED FUELS SAFETY

Isotope Fuel Impact Facility (W. J. Dalby)

The AEC Operational Safety Division, Sandia Area Office, reviewed the proposed operation of the project and indicated that they feel that the radioisotope fuel and simulant impact program can be accomplished safely if undertaken in accordance with the procedures described.

Fuels Impact (W. J. Dalby)

The first full scale SSCS disk was impacted, and the capsule was returned to LASL for particle size analysis. The disk was approximately 2.15 inches in diameter by 0.21 inch thick and weighed approximately 118 grams. The test conditions were: $850 \pm 10^\circ\text{F}$, 141 fps, and 45 degrees disk attitude.

Eleven additional disks are presently available at Sandia for impact testing. However, testing has been suspended because the disk deformed the impact surface of the capsule and some of the material from the disk appeared to be embedded in the capsule. An order was placed for heat treatment of the capsules to obtain a harder impact surface. This is expected to relieve the problem for the scheduled impact tests at 270° and 850°F . Investigation is continuing to determine a suitable capsule material for higher temperature impacts.

Compounds in the Th-Mo-O System (C. J. M. Northrup/R. P. Wemple)

It has been known for several years that thorium and molybdenum oxides react to form a complex system of compounds. Such a reaction may occur when an oxygen atmosphere comes in contact with either a solid solution cermet or a thorium diffusion barrier on molybdenum at temperatures that oxidize the molybdenum. Little is known about the physical properties of the resultant ThMo_2O_8 , and a series of experiments was performed to produce this compound and study it. Several grams of the compound were made by heating a $\text{ThO}_2 + \text{MoO}_3$ mixture at

900°C for 4 hours. The resultant ThMo_2O_8 was a friable, flaky powder such as is common to refractory metal oxides. This compound has been formed at 600°C and has a reported phase transition at approximately 900°C. $\text{PuO}_2 + \text{MoO}_3$ is expected to undergo similar reaction at slightly lower temperature. No detectable vapor species containing thorium were found in the 4-hour heat treatment at 900°C which suggests that the cause of small particulate after such reactions is more apt to be erosion than emission of radioactive vapor species.

Alpha Irradiation of Molybdenum (C. J. M. Northrup/
R. P. Wemple/G. J. Thomas/C. Hill)

In $^{238}\text{PuO}_2$ radioisotopic fuel capsules, molybdenum frequently may be found in contact with the alpha emitting isotopes. This can occur when molybdenum is used for a cermet matrix, fuel coating, fuel liner, etc. The alpha irradiation of the metal produces a number of significant changes, and these effects are being studied by helium implantation studies of molybdenum.

A cold discharge source operating at a potential of 10 kV was used to implant He ions into prethinned molybdenum foils at room temperature. A high density of helium atoms, amounting to approximately one atom percent, was obtained in the specimens, with an average penetration range of about 700 Å. Electron microscope observation after irradiation revealed the expected very dense radiation damage and also some curling of the foil edges due to internal stresses. The foils were then examined after successively higher temperature isochronal anneals. It was observed that voids, presumably containing significant amounts of the helium present, formed in specimens annealed at 600°C or higher. Since the formation of these bubbles would require large numbers of vacancies, this result suggests that vacancies become mobile in the Stage IV temperature region ($\sim 0.3 T_m$).

An example of an annealed sample is shown in Figure 9. This micrograph was obtained after the sample was annealed at 900°C for 10 minutes. The upper grain is in diffraction contrast, and, in addition to the bubbles, resolvable dislocation loops and smaller black spot defects can be seen. Two commonly observed features of the bubbles can be seen in the example. Firstly, the bubbles have a polyhedral shape, as reported previously by R. S. Nelson, et al,¹ for α -irradiated

Figure 9. Sample after annealing at 900°C for 10 minutes (250,000X)

Cu, Al, and Mo samples. Secondly, strings of bubbles, which have an elongated shape, are often observed along the grain boundaries.¹ Although the instrument resolution makes the bubble outline uncertain in the lower-temperature annealed samples, the polyhedral shape is fairly well defined after annealing above approximately 800°C, and it therefore appears that sufficient numbers of vacancies were absorbed for an equilibrium configuration to be attained.

Measurements of the average size and densities of both the bubbles and the loop defects were made for various annealing temperatures (Table IV). It is seen that the average bubble size increases slightly with a corresponding decrease in bubble density as the temperature is raised. The total void volume, however, remains essentially the same. A similar behavior is observed with the dislocation loops, except that the total loop length (proportional to the number of defects in the loops) decreased by an order of magnitude between the 800° and 1100°C annealing temperatures. Preliminary tilting experiments suggest that the loops are interstitial in character.

TABLE IV
Average Size and Density
of Bubbles and Loop Defects

T _{anneal} (°C)	Bubbles		Loops	
	Density	Av. Dia (Å)	Density	Av. Dia (Å)
640	5 x 10 ¹⁷	≤20	(CM ⁻³)	--
790	3 x 10 ¹⁷	20-25	~10 ¹⁷	450
900	1 x 10 ¹⁷	~40	~10 ¹⁶	800
1100	5 x 10 ¹⁶	~50	2-3 x 10 ¹⁵	~1800

Release of helium from the shallow implantation specimens is detectable at temperatures as low as 150°C. As the temperature is increased, the nature of the gas release changes from a continuous emission to a "burst" type release characteristic of bubbles. The gas release continues at temperatures as high as 1500°C for periods as long as 15 to 20 minutes, verifying that once the bubbles are formed they are relatively stable.

The above behavior is consistent with the following explanation. Bubbles begin forming after anneals around 600°C due to the onset of vacancy motion. The bubbles, once formed, are quite stable and effectively trap these vacancies at all annealing temperatures. It is presumed that any changes in the bubble size and density occur by coalescence.² At high temperatures, the interstitial clusters which have survived vacancy recombination begin to dissociate, causing the larger loops to grow at the expense of smaller ones. Similar breakup of defect clusters found after neutron irradiation has been reported.³ At higher temperatures (~1000°C) dislocations are found to be pinned at the bubbles.

References

1. Nelson, R. S., Mazey, D. J., and Barnes, R. S., Phil. Mag., 11, 91 (1965).
2. Barnes, R. S., and Mazey, D. J., Proc. Roy. Soc., A, 275, 47 (1963).
3. Brimhall, J. L., Mastel, B., and Bierlein, T. K., Acta Met., 16, 781 (1968).

TRACKING, SEARCH, DETECTION, AND RECOVERY

Airborne Infrared Detection Tests (J. L. Entrekin)

A second series of infrared detection tests was performed at the Edgewood Bombing Range in April. Primarily, Sandia's Aerospace Nuclear Safety Department provided support to the U. S. Forest Service for these tests. The Forest Service provided the aircraft and the infrared equipment, and Sandia provided the target area and the targets. Both day and night flights were made. Generally, the daytime results were poor and the nighttime results were good.

Targets for the tests were laid out in an east-west line with approximately 500 feet between targets. From east to west, the targets were: 1 square foot of burning charcoal at 1200°F; 3- x 12-inch heater

assembly at 1500°F; 3- x 12-inch heater assembly at 1000°F; 3- x 6 inch heater assembly at 1000°F; 3- x 12-inch heater assembly at 500°F; 1000-watt light bulb; 1 square foot of burning charcoal at 1200°F; and 2 square feet of burning charcoal at 1200°F. Background temperature was 63°F.

In daylight, only the burning charcoal could be detected. Night results were better and are given in Table V. Results from these tests indicate that we can reliably detect 0.25 square foot, 1500°F targets 15,000 feet above terrain and 0.25 square foot, 1000°F targets at 10,000 feet. If detection of the 0.25 square foot, 500°F and 0.125 square foot, 1000°F targets is needed, the aircraft could not be flown over 4000 feet above terrain. Daytime detection of these small targets does not appear to be feasible. For targets of 2 square feet at 1000°F or greater, daytime detection may be feasible.

FLIGHT TEST STUDIES

Athena H Piggyback Flights (A. C. Bustamante)

Plans are proceeding for participating in the second Athena H research and development flight which is presently scheduled for February 1971. Originally, it appeared that Sandia might participate in both Athena research and development flights which had been scheduled for October 8, 1970, and December 10, 1970. The piggyback payload will consist of six heat sources (Figure 10): an instrument unit, a Transit heat source, a Pioneer heat source, a DART minithruster, and two Pioneer heaters. With the exception of simulated fuel, each of the above systems will be identical to the live systems. The test objectives of the piggyback flight are to obtain interim information for the primary test in 1972 on recovery capability and instrumentation survivability.

TABLE V

Data From Night Flight on April 20, 1970, Sandia Edgewood Test Site
Test Site Elevation was 7000 Feet MSL

Slate Number	Altitude (ft)*	Heading	Charcoal Bucket-2 ft ² at 1200°F	Charcoal Bucket-1 ft ² at 1200°F	1000 W Light Bulb	Heater 3 x 12 in. 500°F	Heater 3 x 6 in. 1000°F	Heater 3 x 12 in. 1000°F	Heater 3 x 12 in. 1500°F	Charcoal Bucket-1 ft ² at 1200°F
15	4,000	West	X	X	0	X	X	X	X	X**
16	4,000	East	X	X	0	0	X	X	X	X
17	4,000	West	X	X**	0	X	X	X	X	X**
18	Clouds, no data.									
19	8,000	East	X	X	0	0	0	X	X	X
20	8,000	East	X	X	0	X	0	X	X	X
21	15,000	East	X	X	0	0	0	0	X	X
22	15,000	West	X	X	0	0	0	0	X	X
23	15,000	East	X	X	0	0	0	0	X	X
24	15,000	West	X	X	0	0	0	0	X	X
25	Clouds, no data.									
26	15,000	West	X	X	0	0	0	0	X	X
27	Clouds, no data.									
28	10,000	West	X	0†	0	0	0	0	X	X
29††	10,000	East	X	0†	0	0	0	X	X	X
30	10,000	West	X	0†	0	0	0	X	X	X

*Feet above terrain.

**Target appears on imagery, but no TDM trips.

†Ground crew reported bucket was cooling.

††Light cloud skiff drifted in over area during this pass.

NOTE: X = detected
0 = not detected

Figure 10. Athena H piggyback payload

The instrumented unit shown in Figure 11 will be used to obtain thermal and motion data on a right circular cylinder configuration. The unit will be instrumented with a 3-axis rate gyro to obtain angular rates about the pitch, yaw, and roll axes. In addition, 11 thermocouples will be placed at various locations on the unit in order to obtain its temperature distribution. These locations are shown in Figure 12. Some of the important characteristics of this instrumented heat source are described in Table VI. The heating rate pulses for two of the Athena piggyback flights are shown in Figure 13. At the present, it appears that Sandia's flight trajectory will consist of a reentry velocity of 22,500 fps and a reentry angle of -20 degrees. The heating environment from this trajectory is severe enough to generate good temperature data. The POCO surface temperature on the unit should reach a peak of approximately 5300°F (Figure 14). (This includes a safety factor of 1.3 times the heating rate.) A CINDA thermal analysis was conducted on the instrumented unit in order to determine the amount of insulation required to protect the instrumentation from the high

temperature environment. A fibrous carbon felt thickness of 0.651 inch was required in order to keep the instrumentation below a temperature of 175°F (Figure 15). The total cost for the instrument unit is shown in Table VII.

Figure 11. Instrument package for motion and thermal data

Figure 12. Thermocouple locations

TABLE VI
Instrumented Heat Source Description

Total Weight	35.2 lb
Ballistic Parameter (Hypersonic Continuum)	47
Terminal Velocity at 6000 Feet Altitude	332 ft/sec
Maximum Deceleration Load	90 g
Reference Trajectory:	
Velocity = 22,500 ft/sec	
$\gamma = -20$ deg	
Instrumentation:	
A. Three Axis Rate Gyro:	Pitch ± 3600 deg/sec
	Yaw ± 3600 deg/sec
	Roll ± 1800 deg/sec
B. VCO	
C. Commutator	
D. Tape Recorder	
E. 11 Thermocouples	

Figure 13. Heating rate vs time for two piggyback Athena trajectories

Figure 14. Instrumented heat source temperature vs time

Figure 15. Athena piggyback flight

TABLE VII
Cost for Instrumented Unit

Tape Recorder	\$ 2,100
VCO	1,500
Commutator	3,000
Three Axis Gyro	5,000
Thermocouples	3,000
Heat Shield	5,000
Mechanical Assembly	1,000
Miscellaneous	500
Total Hardware	\$21,100
Manpower	45,900
Total	\$67,000

Flight Test Briefings (A. C. Bustamante)

A briefing on the proposed FY72 Athena H flight test of an isotopic heat source was given at the Pentagon by Sandia to representatives of the Air Force, AEC, DDRE (Space Technology), and NASA.

A briefing was given by Sandia on June 10 to a group of TRW's Space Systems staff on the proposed FY72 Athena H and Athena H research and development piggyback flights.

Thermal Gradient Study (A. C. Bustamante)

A temperature study to determine the relative difference between the temperature gradients resulting from a heat source with internal heat generation and a heat source without internal heat was completed. The study was conducted for a Transit heat source, using environmental conditions for Transit abort No. 5. The results of this study will be used for a thermal stress analysis.

Athena H Performance Capabilities (A. C. Bustamante)

An Athena flight test interface meeting between SAMSO, Aerospace, Atlantic Research, and Sandia was held on April 9, 1960, at Aerospace Corporation, San Bernadino, California. The two items for discussion

at the meeting were (1) standard Athena H performance capabilities relative to flight test requirements, and (2) feasibility of conducting piggyback flights on Athena H or standard Athena flights.

REENTRY AND HEAT CONDUCTION STUDIES

Heat Conduction in Solids (R. E. D. Stewart)

There are a number of possible approaches for obtaining numerical solutions of heat conduction problems. Two of the more common ones are the "finite element" and the "finite difference" approaches. In the finite element approach, the body is subdivided into a meshwork of finite elements, and an energy balance is written for a typical element. In the finite difference approach, one starts with an appropriate form of the differential heat conduction equation and replaces each term in the equation with an appropriate finite difference representation. Both approaches are completely equivalent, as they must be if they are to yield identical results. However, strict attention must be paid to detail in order to avoid certain potential error sources associated with each approach.

One potential source of error is the apparently rather common but erroneous assumption that, for the case of a homogeneous, isotropic material with temperature dependent properties and no internal heat generation, the heat conduction equation can be written as

$$\nabla^2 T = \frac{1}{\alpha(T)} \frac{\partial T}{\partial t}.$$

Under the above conditions, the correct form of the heat conduction equation is

$$\nabla^2 T + \frac{1}{k(T)} \frac{dk}{dT} (\nabla T \cdot \nabla T) = \frac{1}{\alpha(T)} \frac{\partial T}{\partial t}.$$

Other common errors involve evaluating the thermal conductivity at the interface between elements when it should be evaluated at the center of the element, and vice versa.

If one takes the finite difference approach, all properties must be evaluated at the center of the element since the differential heat

conduction equation applies at a point. In the finite element approach, however, the thermal conductivity for conduction heat transfer between adjacent finite elements must be evaluated at the mean temperature of these two elements, i.e., at the temperature of the interface between adjacent elements. Various combinations of the errors described above can effectively result in the term

$$\frac{1}{k(T)} \frac{dk}{dT} (\nabla T \cdot \nabla T)$$

being either neglected or included twice. This term stems from a spatial dependence of the thermal conductivity which is induced by its temperature dependence in the presence of any thermal gradient. Its magnitude varies directly as the product of the temperature gradient squared and the slope of the thermal conductivity versus temperature curve, and inversely as the thermal conductivity. The significance of this term is illustrated in Figures 16, 17, and 18 which show the front face temperature versus time of a homogeneous isotropic slab irradiated on the front face by a constant and uniform heat flux and insulated on the rear face. Curves are shown for each of two heating rates, 350 and 700 Btu/ft²-sec, for each of three materials, zirconium dioxide, thorium dioxide, and graphite. These heating rates were chosen as being representative of reentry stagnation heating rates, and these materials were chosen because of their applications in isotopic space power supplies which must be designed to survive the reentry environment. These three materials were also chosen for illustration purposes because they cover a wide range of both thermal conductivity and sensitivity of thermal conductivity to temperature change. The thermal conductivities of these materials as a function of temperature are shown in Figures 19, 20, and 21. In Figures 16, 17, and 18, the solid curve represents the true temperature as a function of time, the dashed curve the erroneous results obtained by twice including the term

$$\frac{1}{k(T)} \frac{dk}{dT} (\nabla T \cdot \nabla T),$$

and the dot-dash curve the erroneous result of neglecting this term. These results show that in situations involving severe thermal gradients, such as those generated during hypersonic reentry, improperly accounting for this term can cause significant, and in some cases catastrophic errors.

Figure 16. Front face temperature of slab versus time for thorium dioxide

Figure 17. Front face temperature of slab versus time for graphite

Figure 18. Front face temperature of slab versus time for zirconium dioxide

Figure 19. Thermal conductivity versus temperature for thorium dioxide

Figure 20. Thermal conductivity versus temperature for graphite

Figure 21. Thermal conductivity versus temperature for zirconium dioxide

Thermal Shock Similarity Analysis (R. E. D. Stewart)

The similarity parameters necessary to model the thermal stresses occurring in a homogeneous, isotropic right circular cylinder with constant material properties subjected to axially symmetric heating, either over one end or over the cylindrical surface, were determined for two cases: (1) when radiation heat transfer from the cylinder surfaces is important, and (2) when radiation heat transfer from the cylinder surfaces can be neglected. For case (1), the following set of nondimensional similarity parameters is obtained:*

$$\left\{ \begin{array}{l} \nu, \frac{L}{D}, \bar{r} \equiv \frac{r}{R}, \bar{z} \equiv \frac{z}{L}, \bar{t} \equiv \frac{kt}{\rho CL^2}, \bar{q} \equiv \frac{qL}{kT_0}, \\ \bar{q}''' \equiv \frac{q'''L^2}{kT_0}, \bar{\epsilon} \equiv \frac{\epsilon \beta T_0^3 L}{k}, \bar{T} \equiv \frac{T - T_0}{T_0}, \bar{\sigma}_{ij} \equiv \frac{\sigma_{ij}(1 - \nu)}{\alpha E T_0} \end{array} \right\}$$

Making ν , L/D , $\bar{q}$, $\bar{q}'''$, and $\bar{\epsilon}$ equal for a model and prototype guarantees that $\bar{T}$ and $\bar{\sigma}$ will be identical functions of $\bar{r}$, $\bar{z}$, and $\bar{t}$ for the model and prototype.

The fact that ν appears in the above group of similarity parameters means that the model material must be chosen so that Poisson's ratio for the model and prototype materials are the same. However, a numerical investigation of thermal stresses in a cylinder with $1/8 \leq L/D \leq 8$ and $0 \leq \nu \leq 0.49$ has shown that for practical purposes similarity is obtained without requiring that ν be the same for model and prototype. Therefore for engineering purposes at least for $1/8 \leq L/D \leq 8$, ν may be deleted from the list of similarity parameters. Equating each of the remaining similarity parameters for a model and prototype yields the following scaling relations:**

$$\frac{L_m}{L_p} = \frac{D_m}{D_p} \quad (1)$$

---

*Symbols are defined at the end of this section.

**The subscripts m and p denote model and prototype, respectively.

$$\frac{q_m}{q_p} = \left(\frac{\epsilon_p}{\epsilon_m}\right)^{1/3} \left(\frac{k_m}{k_p}\right)^{4/3} \left(\frac{L_p}{L_m}\right)^{4/3} \quad (2)$$

$$\frac{T_{o_m}}{T_{o_p}} = \left(\frac{\epsilon_p}{\epsilon_m}\right)^{1/3} \left(\frac{k_m}{k_p}\right)^{1/3} \left(\frac{L_p}{L_m}\right)^{1/3} \quad (3)$$

$$\frac{q'''_m}{q'''_p} = \left(\frac{\epsilon_p}{\epsilon_m}\right)^{1/3} \left(\frac{k_m}{k_p}\right)^{4/3} \left(\frac{L_p}{L_m}\right)^{7/3} \quad (4)$$

$$\frac{t_m}{t_p} = \left(\frac{\rho_m}{\rho_p}\right) \left(\frac{C_m}{C_p}\right) \left(\frac{k_p}{k_m}\right) \left(\frac{L_m}{L_p}\right)^2 \quad (5)$$

$$\frac{T_m}{T_p} = \left(\frac{\epsilon_p}{\epsilon_m}\right)^{1/3} \left(\frac{k_m}{k_p}\right)^{1/3} \left(\frac{L_p}{L_m}\right)^{1/3} \quad (6)$$

$$\frac{\sigma_{ij_m}}{\sigma_{ij_p}} = \left(\frac{\alpha_m}{\alpha_p}\right) \left(\frac{E_m}{E_p}\right) \left(\frac{\epsilon_p}{\epsilon_m}\right)^{1/3} \left(\frac{k_m}{k_p}\right)^{1/3} \left(\frac{L_p}{L_m}\right)^{1/3} \quad (7)$$

To ensure similarity of the temperature and stress distributions in a model and prototype, the geometry, applied heat flux, initial temperature, and internal heat generation for the model must be scaled to satisfy relations (1), (2), (3), and (4) above. Equations (5), (6), and (7) give the ratio of the dimensional times, temperatures, and stresses existing at the same nondimensional times for model and prototype. The ratio of the dimensional temperatures and stresses occurring at the same dimensional times for model and prototype can be obtained by setting $t_m/t_p = 1$ in Eq. (5) and combining the result with Eqs. (6) and (7). This gives:

$$\frac{T_m}{T_p} = \left(\frac{\rho_m}{\rho_p}\right)^{1/3} \left(\frac{C_m}{C_p}\right)^{1/3} \left(\frac{\epsilon_p}{\epsilon_m}\right)^{1/3} \left(\frac{L_m}{L_p}\right)^{1/3} \quad (8)$$

$$\frac{\sigma_{ijm}}{\sigma_{ijp}} = \left(\frac{\alpha_m}{\alpha_p} \right) \left(\frac{E_m}{E_p} \right) \left(\frac{\rho_m}{\rho_p} \right)^{1/3} \left(\frac{C_m}{C_p} \right)^{1/3} \left(\frac{\epsilon_p}{\epsilon_m} \right)^{1/3} \left(\frac{L_m}{L_p} \right)^{1/3} \quad (9)$$

The ratio of the dimensional times at which the ratio $(\sigma_{\max}/\sigma_{\text{rupt}})$ is the same for the model and prototype can be found by multiplying Eq. (9) by $(\sigma_{\text{rupt}p}/\sigma_{\text{rupt}m})$, setting the left hand side of the resulting equation equal to unity, and combining the relation obtained with Eq. (6). The final result is:

$$\frac{t_m}{t_p} = \left(\frac{\alpha_m}{\alpha_p} \right)^6 \left(\frac{E_m}{E_p} \right)^6 \left(\frac{\epsilon_p}{\epsilon_m} \right)^2 \left(\frac{k_m}{k_p} \right) \left(\frac{\rho_m}{\rho_p} \right) \left(\frac{C_m}{C_p} \right) \left(\frac{\sigma_{\text{rupt}p}}{\sigma_{\text{rupt}m}} \right)^6 \quad (10)$$

Thus, whether or not a prototype will fail due to thermal shock when subjected to a heating pulse $q_p(r,t)$ over a period of time t_p can be determined experimentally by subjecting a model, whose geometry, initial temperature, and internal heat generation have been scaled according to Eqs. (1), (3), and (4) above, to a heating pulse $q_m(r,t)$ determined by Eq. (2) for a period of time t_m determined by Eq. (10).

If a cylinder is subjected simultaneously to axially symmetric heating over more than one of its surfaces, an additional similarity parameter must be added for each additional surface which is heated. If, for example, the two ends are subjected to axially symmetric heating pulses of $q_1(r,t)$ and $q_2(r,t)$, respectively, and the cylindrical surface to an axially symmetric heating pulse $q_c(z,t)$, then the similarity parameter qL/kT_0 must be replaced by three similarity parameters: q_1L/kT_0 , q_2L/kT_0 , and q_cK/kT_0 ; and the heat flux to each of the model surfaces must be scaled according to Eq. (2), i.e.,

$$\frac{q_{1m}}{q_{1p}} = \frac{q_{2m}}{q_{2p}} = \frac{q_{cm}}{q_{cp}} = \left(\frac{\epsilon_p}{\epsilon_m} \right)^{1/3} \left(\frac{k_m}{k_p} \right)^{4/3} \left(\frac{L_p}{L_m} \right)^{4/3} \quad (11)$$

For the case where radiation from the surfaces of the cylinder can be neglected, the similarity parameter $\bar{\epsilon} \equiv \epsilon \beta T_O^3 L / k$ vanishes. In this case, the scaling relations become:

$$\frac{L_m}{L_p} = \frac{D_m}{D_p} \quad (12)$$

$$\frac{q_m}{q_p} = \left(\frac{k_m}{k_p} \right) \left(\frac{L_p}{L_m} \right) \left(\frac{T_{O_m}}{T_{O_p}} \right) \quad (13)$$

$$\frac{q'''_m}{q'''_p} = \left(\frac{k_m}{k_p} \right) \left(\frac{L_p}{L_m} \right)^2 \left(\frac{T_{O_m}}{T_{O_p}} \right) \quad (14)$$

$$\frac{t_m}{t_p} = \left(\frac{k_p}{k_m} \right) \left(\frac{\rho_m}{\rho_p} \right) \left(\frac{C_m}{C_p} \right) \left(\frac{L_m}{L_p} \right)^2 \quad (15)$$

$$\frac{T_m}{T_p} = \frac{T_{O_m}}{T_{O_p}} \quad (16)$$

$$\frac{\sigma_{ijm}}{\sigma_{ijp}} = \left(\frac{\alpha_m}{\alpha_p} \right) \left(\frac{E_m}{E_p} \right) \left(\frac{T_{O_m}}{T_{O_p}} \right) \quad (17)$$

For this case, the equations corresponding to Eqs. (8), (9), and (10) are, respectively:

$$\frac{T_m}{T_p} = \frac{T_{O_m}}{T_{O_p}} \quad (18)$$

$$\frac{\sigma_{ijm}}{\sigma_{ijp}} = \left(\frac{\alpha_m}{\alpha_p} \right) \left(\frac{E_m}{E_p} \right) \left(\frac{T_{O_m}}{T_{O_p}} \right) \quad (19)$$

$$\frac{t_m}{t_p} = \left(\frac{\rho_m}{\rho_p}\right)\left(\frac{C_m}{C_p}\right)\left(\frac{\alpha_p}{\alpha_m}\right)\left(\frac{E_p}{E_m}\right)\left(\frac{k_p}{k_m}\right)\left(\frac{L_m}{L_p}\right)^2\left(\frac{T_{om}}{T_{op}}\right). \quad (20)$$

In most situations of practical interest, the internal heat generation will be either negligible or identically zero. For these situations, the similarity parameter $q'''L^2/kT_0$ vanishes. As a result, if radiation from the cylinder surfaces is also negligible, the parameter qL/kT_0 can be conveniently combined with the nondimensional temperature and stresses to further decrease the number of similarity parameters required to form a complete set. For this case, the following set of similarity parameters is obtained.

$$\left\{ \nu, \frac{L}{D}, \bar{r} \equiv \frac{r}{R}, \bar{z} \equiv \frac{z}{L}, \bar{t} \equiv \frac{kt}{\rho CL^2}, \right. \\ \left. \frac{\bar{T}}{\bar{q}} \equiv \frac{k(T - T_0)}{qL}, \frac{\bar{\sigma}_{ij}}{\bar{q}} \equiv \frac{\sigma_{ij}k(1 - \nu)}{\alpha E qL} \right\}$$

As mentioned previously, for engineering purposes ν may be deleted from this set. Thus, in this case, simply making L/D equal for a model and prototype guarantees that $\bar{T}/\bar{q}$ and $\bar{\sigma}_{ij}/\bar{q}$ will be identical functions of $\bar{r}$, $\bar{z}$, and $\bar{t}$ for both the model and prototype.

An existing thermal stress computer code was modified to provide, at the users option, plots of the nondimensional stresses and temperatures divided by the nondimensional heat flux as a function of nondimensional time or position; the remaining nondimensional similarity variables are treated as parameters. An error analysis was made to determine the maximum errors which can occur in

$$\frac{\bar{T}}{\bar{q}}, \frac{\bar{\sigma}}{\bar{q}}, \frac{\bar{\tau}}{\bar{q}}, \left(\frac{\sigma}{\sigma_{rupt}}\right)_{\max}, \text{ and } \left(\frac{\tau}{\tau_{rupt}}\right)_{\max}$$

from given inaccuracies in basic material properties and from experimental errors in the measurement of applied heat flux and test time for the model. The results of this error analysis are given below.

$$\frac{\bar{T}}{\bar{q}} = \left(\frac{\bar{T}}{\bar{q}}\right)_{\text{nominal}} \pm \left\{ \bar{t} \frac{\partial \left(\frac{\bar{T}}{\bar{q}}\right)}{\partial \bar{t}} \left[\frac{\Delta k}{k} - \frac{\Delta \rho}{\rho} - \frac{\Delta C}{C} + \frac{\Delta t}{t} \right] \right\}. \quad (21)$$

$\bar{\sigma}/\bar{q}$ and $\bar{\tau}/\bar{q}$ are given by equations identical to Eq. (21) except that $\bar{T}$ is replaced by $\bar{\sigma}$ and $\bar{\tau}$, respectively.

$$\begin{aligned} \left(\frac{\sigma}{\sigma_{\text{rupt}}}\right)_{\text{max}} &= \left(\frac{\sigma_{\text{max}}}{\sigma_{\text{rupt}}}\right)_{\text{max nominal}} \left\{ 1 \pm \left[\frac{\Delta k}{k} \left(\frac{\bar{t}}{\left(\frac{\bar{\sigma}}{\bar{q}}\right)}\right) \frac{\partial \left(\frac{\bar{\sigma}}{\bar{q}}\right)}{\partial \bar{t}} - \right. \right. \\ &\quad \left. \left. - \bar{t} \left(\frac{qL}{k\sigma_{\text{rupt}}}\right) \left(\frac{d\sigma_{\text{rupt}}}{dT}\right) + \left(\frac{qL}{k\sigma_{\text{rupt}}}\right) \left(\frac{d\sigma_{\text{rupt}}}{dT}\right) \left(\frac{\bar{T}}{\bar{q}}\right) - 1 \right] + \right. \\ &\quad \left. + \left(\frac{\Delta \rho}{\rho} + \frac{\Delta C}{C} - \frac{\Delta t}{t}\right) \left(\bar{t} \left(\frac{qL}{k\sigma_{\text{rupt}}}\right) \left(\frac{d\sigma_{\text{rupt}}}{dT}\right) \frac{\partial \left(\frac{\bar{T}}{\bar{q}}\right)}{\partial \bar{t}} - \right. \right. \\ &\quad \left. \left. - \frac{\bar{t}}{\left(\frac{\bar{\sigma}}{\bar{q}}\right)} \frac{\partial \left(\frac{\bar{\sigma}}{\bar{q}}\right)}{\partial \bar{t}} \right) + \frac{\Delta q}{q} \left(1 - \left(\frac{qL}{k\sigma_{\text{rupt}}}\right) \left(\frac{d\sigma_{\text{rupt}}}{dT}\right) \left(\frac{\bar{T}}{\bar{q}}\right) \right) + \right. \\ &\quad \left. \left. + \frac{\Delta \nu}{\nu} \left(\frac{\nu}{1-\nu}\right) + \left(\frac{\Delta \alpha}{\alpha} + \frac{\Delta E}{E} - \frac{\Delta \sigma_{\text{rupt}}}{\sigma_{\text{rupt}}}\right) \right] \right\}. \quad (22) \end{aligned}$$

$(\tau/\tau_{\text{rupt}})_{\text{max}}$ is given by an equation identical to Eq. (22) except that σ and $\bar{\sigma}$ are replaced by τ and $\bar{\tau}$, respectively. An option is provided in the thermal stress computer code which provides plots of the nominal values of

$$\frac{\bar{T}_{\text{max}}}{\bar{q}}, \frac{\bar{\sigma}_{\text{max}}}{\bar{q}}, \frac{\tau_{\text{max}}}{\bar{q}}, \left(\frac{\sigma}{\sigma_{\text{rupt}}}\right)_{\text{max}}, \text{ and } \left(\frac{\tau}{\tau_{\text{rupt}}}\right)_{\text{max}}$$

along with the maximum error band for these quantities which can result from given errors in basic material properties and from experimental measurement of the applied heat flux and test time for the model.

Some typical computer plots for a solid cylinder of zirconium dioxide heated from one end are shown in Figures 22 through 28. Figures 22, 23, and 24 show the effect of radiation from the cylinder on $\bar{T}_{\max}/\bar{q}$, $\bar{\sigma}_{\max}/\bar{q}$, and $\bar{\tau}_{\max}/\bar{q}$ for various values of emissivity and heating rate. Figures 25 and 26 demonstrate that, for engineering purposes, similarity is obtained without requiring that Poisson's ratio be the same for both model and prototype. (For true similarity, all curves in Figures 25 and 26 would coalesce into a single curve.) Although Figures 25 and 26 are for $\epsilon = 0$ (i.e., no radiation) and $L/D = 1.5$, the same approximate coalescence also occurs for other combinations of ϵ and L/D , at least within the range $1/8 \leq L/D \leq 8$. Figures 27 and 28 show plots of

$$\left(\frac{\sigma}{\sigma_{\text{rupt}}} \right)_{\max} \quad \text{and} \quad \left(\frac{\tau}{\tau_{\text{rupt}}} \right)_{\max},$$

respectively, as a function of $\bar{t}$. In Figures 27 and 28, the central curve represents the nominal value of the independent variable, while the upper and lower curves are the maximum and minimum values, respectively, which can result from an assumed ± 5 percent inaccuracy in each of the basic material properties as well as the applied heat flux and test time. The values for the upper and lower curves are obtained from the error analysis results by choosing independently each of the 5 percent inaccuracies either plus or minus, as required, so as to result in the maximum total error in the independent variable. Note that the curves in Figure 27 are not perfectly smooth curves. This is caused by the fact that the position of the maximum tensile stress in the cylinder is constantly changing with time, together with the fact that the computer utilizes finite intervals of time and position to calculate $\bar{\sigma}$ and $\partial(\bar{\sigma}/\bar{q})/\partial\bar{t}$. These curves would become smoother for smaller intervals of time and position. This behavior does not occur in Figure 28 because the position in the cylinder at which the maximum shear stress occurs does not change with time.

An important conclusion to be drawn from Figures 27 and 28 is that a ± 5 percent inaccuracy in each of the basic material properties for zirconium dioxide, together with a ± 5 percent error in the experimentally measured applied heat flux and test time, results in a total maximum cumulative error in the independent variable of approximately ± 30 to 35 percent. The total cumulative error obviously varies somewhat with time and will also be different for different materials. In addition, it will be a function of the applied heat flux. The point to be made, however, is that relatively small errors in the individual material properties can result in a rather sizable cumulative error in the all important ratios

$$\left(\frac{\sigma}{\sigma_{\text{rupt}}}\right)_{\text{max}} \quad \text{and} \quad \left(\frac{\tau}{\tau_{\text{rupt}}}\right)_{\text{max}} .$$

This illustrates the importance of accurate material property data.

Symbols

- C - specific heat
- D - cylinder diameter
- E - Young's modulus
- k - thermal conductivity
- L - cylinder length
- q - applied heat flux per unit area and time
- q''' - internal heat generation per unit volume and time
- r - radial coordinate measured from cylinder axis
- R - radius of cylinder
- t - time
- T - temperature
- T_0 - initial temperature of cylinder
- z - coordinate measured along cylinder axis
- α - coefficient of thermal expansion
- β - Stefan-Boltzmann constant
- ϵ - emissivity
- ρ - density
- ν - Poisson's ratio
- σ_{ij} - components of stress tensor
- σ - indicates a tensile stress when no subscripts are used
- τ - shear stress

Figure 22. Maximum temperature/heat flux vs time for various values of emissivity and heating rate

Figure 23. Maximum tensile stress/heat flux vs time for various values of emissivity and heating rate

Figure 24. Maximum shear stress/heat flux vs time for various values of emissivity and heating rate

Figure 25. Maximum tensile stress/heat flux for various values of Poisson's ratio

Figure 26. Maximum shear stress/heat flux for various values of Poisson's ratio

Figure 27. Maximum value of ratio of tensile stress to tensile strength versus time

Figure 28. Maximum value of ratio of shear stress to shear strength versus time

ABORT MODEL

Scaling Laws for Fuel Capsules Subjected to Blast, Impact, and Thermal Loading (T. A. Duffey)

Scaling laws were recently developed for fuel capsules subjected to various loadings. The laws show that it is possible to test models of fuel capsules that were geometrically reduced in size. In addition, "material scaling," in which substitute materials are used to construct the model fuel capsules, was found to be possible in certain instances.

It is felt that an experimental verification and demonstration of the scaling laws would be desirable. Toward this end, a series of experiments is currently being designed to verify the scaling laws. Pure geometrical scaling will be attempted, followed by a material scaling effort. The configuration and loading chosen are intended to resemble fuel capsule earth impact.

Dynamic Material Properties from the Ring Test (T. A. Duffey)

Dynamic material properties are necessary for accurate predictions of structural response to suddenly applied loads, such as the earth impact of fuel capsules. Further, they are essential for meaningful selection of the "best" material (from a structural standpoint) once a given design is established.

For current fuel capsule candidate materials, these dynamic properties do not appear to be available even at room temperature in the intermediate strain rate range of 10 to 1000 per second. While there are several different methods of obtaining the required properties, such as the intermediate strain rate tensile testing machine and the Hopkinson bar, there is some question as to the validity of data obtained by these techniques.

One experimental technique that appears promising is the uniformly expanding ring technique being developed by Denver Research Institute. As reported last quarter, they had experienced difficulty with required double differentiation of the data. Since that time, discussions with them indicate that they have overcome this difficulty by observing that displacement-time records of the expanding ring take on a parabolic form to a high degree of accuracy. The resulting data

reduction is greatly simplified. As a consequence, there appears to be little justification for further simplification by making certain material assumptions, as proposed in the previous Quarterly Report.

At present, the expanding ring technique is limited to room temperature material properties. For fuel capsules, dynamic material properties at high temperatures are required as well.

Presently, efforts are underway to: (1) place the experimentally observed parabolic displacement-time behavior on a firm theoretical foundation, and (2) extend the technique to high temperatures.

COMPUTER CODES

RADFAC (H. R. Spahr)

Computer code RADFAC computes thermal and solar script F radiation interchange factors among surfaces having any combination of the following properties:

1. Diffuse emittance and reflectance
2. Diffuse emittance and specular reflectance
3. Diffuse emittance and components of diffuse and specular reflectance
4. Directional emittance and bi-directional reflectance

The reports documenting this code were received from NASA/Houston and were reviewed. A memo was written which briefly describes the code and its capabilities, and provides recommendations on priorities for the steps in obtaining this code and making it operational on Sandia computers.

COSMIC Computer Codes (H. R. Spahr)

The new catalog of computer codes available through the Computer Software and Management Information Center (COSMIC) were reviewed for codes useful to the Space Power Systems Directorate. The new catalog lists the following computer codes supplied by the Aerospace Nuclear Department:

1. HRS004 - computes initial trajectory conditions for objects decaying from circular orbits
2. HRS015 - computes hypersonic ballistic coefficients for blunt objects in the transition flow regime.

These codes were supplied at the request of AEC/ALO.

HRS017 (H. R. Spahr)

Computer code HRS017 computes the trajectory and reentry environment for nuclear systems decaying from highly elliptical orbits. SC-M-70-157, which describes briefly this code and its availability, was published.

Documentation on computer code HRS017 was supplied to Lockheed Missiles and Space Company, at their request. Lockheed is considering using this code in some of the aerothermodynamic analyses of the nuclear reusable vehicle under a NASA/Marshall contract.

The computer tape and documentation package for the two versions of this code for four computers were submitted to COSMIC, as requested by AEC/ALO. Information on this code was submitted for inclusion in the Sandia Laboratories Aerothermodynamic Program Library.

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