Draft 5/1/73

CONF-730811-7

LASL AEC OFFICIAL

ATMOSPHERE-ENTRY BEHAVIOR
OF A
MODULAR, DISK SHAPED, ISOTOPE HEAT SOURCE

by

J. W. Vorreiter

J. J. Burns

W. C. Pitts

H. A. Stine

Proposed Paper for the
8th Intersociety Energy Conversion Engineering Conference
Philadelphia, Pennsylvania, August 13-17, 1973

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.

LASL AEC OFFICIAL

MASTER

84

ATMOSPHERE-ENTRY BEHAVIOR OF A MODULAR, DISK SHAPED, ISOTOPE HEAT SOURCE

by

J. W. Vorreiter, J. J. Burns, W. C. Pitts, and H. A. Stine

ABSTRACT

The authors have studied the entry and impact behavior of an isotope heat source for space nuclear power that disassembles into a number of modules that would enter the earth's atmosphere separately in the event of a flight abort. These modules are disk-shaped units each with its own reentry heat shield and protective impact container. In normal operation the disk modules are stacked together inside the generator but during a reentry abort they separate and fly as individual units of low ballistic coefficient. Flight tests at hypersonic speeds have confirmed that a stack of disks will separate and assume a flat-forward mode of flight. Free fall tests of single disks have demonstrated a nominal 100 ft/sec sea level impact velocity for a practical range of ballistic coefficients. Thermal and structural calculations indicate reentry survival under all possible reentry conditions for speeds up to and including earth ^{escape} speed.

INTRODUCTION

The protection requirements for space-destined nuclear materials has become more severe due to increasing environmental concerns, more severe possible launch failure environments, and the demonstrated likelihood of booster malfunction of approximately one infive.

Capsule damage resulting from ground impact after a reentry abort is one motivation for developing a modular heat source. The problem of capsule impact damage can be solved in two ways. First, existing designs such as SNAP27, and SNAP 19 can be improved by adding impact protection. The second solution, and the one employed here, is to substantially reduce the kinetic energy of ground impact. This is accomplished by dividing the heat source into modular disk-shaped units which when released from the generator because of aerodynamic forces ~~on the generator~~ fly as individual disks with a significantly lower mass to drag ratio than the assembled cylinder. The resulting lower mass to drag ratio not only lowers the free fall terminal velocity of individual disks as compared to other systems but also reduces the aerodynamic heating environment during reentry. In this paper, the authors show that such a modular disk heat source can be designed that is compatible with existing generator systems and that represents a viable solution to the capsule damage problem.

The concept of a disk-shaped heat source is not new. The disk shape was briefly considered by Brunner, et.al. (1) but rejected because of uncertain aerodynamic performance. However, subsonic aerodynamic characteristics of flat objects that included disks were investigated by Bustamante and Stone (2) among others and found to be systematic and predictable. Dix (3) discussed the disk as a possible solution to impact damage problems encountered on virtually all previous Radioisotope Thermoelectric Generator (RTG) programs. A tentative version of the design discussed herein was considered by La Porte and Osmeyer(4).

To consider the survivability of a disk module configuration the authors first

designed a modular disk shaped heat source that would be compatible with existing RTG generators. Temperature distributions of the assembled heat source were first calculated for normal operation. These temperature levels were checked against known material compatibility results to appraise the desired long-life integrity and operation. Then, this design was evaluated for reentry survival in the event of launch vehicle malfunction. Several possible high speed abort reentry conditions were postulated for the modular heat source and the thermal history of critical components was calculated those conditions. The flat-forward mode of flight was experimentally confirmed by ballistic range tests at hypersonic speeds. The subsonic portion of flight following a high speed abort reentry was also experimentally investigated. Finally, the post impact temperature of the modular heat source was calculated.

SYSTEM DESCRIPTION

Heat Source

An exploded view of an individual module is shown in Figure 1. The cost and fabrication of the fuel represents the major expense of most RTG systems. With that fact in mind, a fuel form in the shape of a right circular cylinder of dimensions 22.86mm in diameter and 8.66mm long was chosen because of the ease of fabrication. The exact dimensions of the fuel pellet were chosen after a dimension, thermal and weight minimization study which did not yield any clear optimum fuel form dimensions.

254

The fuel pellet is encapsulated in a .254mm wall thickness container of Iridium or ~~Iridium-Platinum alloy~~. Helium is vented from each fuel capsule through a porous plug. The encapsulation container represents the primary barrier for long-term post-impact survival. In the arrangement studied, sixteen encapsulated fuel pellets are deployed in an

Nb-Be Metal Alloy

igned a modular disk shaped heat source that would be compatible with existing RTG generators. Temperature distributions of the assembled heat source were first calculated for normal operation. These temperature levels were checked against known material compatibility results to appraise the desired long-life integrity and operation. Then, this design was evaluated for reentry survival in the event of launch vehicle malfunction. Several possible high speed abort reentry conditions were postulated for the modular heat source and the thermal history of critical components was calculated for these conditions. The flat-forward mode of flight was experimentally confirmed by ballistic range tests at hypersonic speeds. The subsonic portion of flight following a high speed abort reentry was also experimentally investigated. Finally, the post impact temperature of the modular heat source was calculated.

SYSTEM DESCRIPTION

Heat Source

An exploded view of an individual module is shown in Figure 1. The cost and fabrication of the fuel represents the major expense of most RTG systems. With that fact in mind, a fuel form in the shape of a right circular cylinder of dimensions 22.86mm in diameter and 8.66mm long was chosen because of the ease of fabrication. The exact dimensions of the fuel pellet were chosen after a dimension, thermal and weight minimization study which did not yield any clear optimum fuel form dimensions.

The fuel pellet is encapsulated in a ²⁵⁴24mm wall thickness container of ~~dium or Iridium-Platinum alloy~~. Helium is vented from each fuel capsule through a porous plug. The encapsulation container represents the primary barrier for long-term post-impact survival. In the arrangement studied, sixteen encapsulated fuel pellets are deployed in an

annular pattern and encased in a molybdenum alloy strength member. The wall thickness of the strength member is 2.54mm on the outer diameter, 1.25mm on the top and bottom of the fuel annulus and .254mm thick on the top and bottom of the center cavity. The center cavity skin is supported in the center by a porous plug that vents helium to the environment. Molybdenum a

functions as the primary strength member to minimize damage to the fuel pellets and their ~~iridium~~ cans in the event of ground impact, blast or fragment impact. TZM was the particular alloy studied for this application because of its high strength to weight ratio and high melting temperature. *However, it would require a compatibility-higher reentry heat*

The carbon-carbon composite shield surrounding the molybdenum strength member performs three functions. First, it serves as an ablative heat shield upon high speed atmospheric reentry. Second, it functions as a barrier to high speed fragments ^{that might be booster} generated in ~~a~~ fuel tank explosions. Third, it forms a secondary strength member for ground impact.

tern and encased in a molybdenum alloy strength member. The
ess of the strength member is 2.54mm on the outer diameter,
the top and bottom of the fuel annulus and .254mm thick on the
com of the center cavity. The center cavity skin is supported in
by a porous plug that vents helium to the environment. Molybdenum alloy
ns as the primary strength member to minimize damage to the
s and their ~~iridium~~ cans in the event of ground impact, blast or
pact. TZM was the particular alloy studied for this
because of its high strength to weight ratio and high melting

However, it would require a compatibility-high reentry heat
carbon-carbon composite shield surrounding the molybdenum strength *EMISSIVE PARTIAL*
forms three functions. First, it serves as an ablative heat shield
peed atmospheric reentry. Second, it functions as a barrier
ed fragments ^{that might be *booster*} generated in ~~ex~~ fuel tank explosions. Third, it
ondary strength member for ground impact.

A carbon-carbon composite material was chosen for this shield because of its superior strength and high thermal shock resistance as compared with bulk graphites. The shield measures 182.9mm outside diameter, 13.97mm thick on the outer edge, and 4.445mm thick on the flat sides. There is a 4.6mm radius on the outside corner. The shield is made in two halves and threaded together. To minimize stress concentrations caused by the threads, the shield was first structurally analyzed as a continuous body and then the threads were placed in an area of minimum stress.

During normal operation, a number of modular disk heat sources are stacked together to form a complete heat source. For the MHW generator, as an example, spacers would be required between disks to completely fill the generator cavity. End caps would also be necessary to match the existing generator hardware and to provide additional protection against blast fragment penetration. An assembled heat source unit that would produce 2400Wt is shown in Figure 2. The assembled heat source shown is dimensionally, thermally, and structurally compatible with the thermoelectric subsystems designed for the Multi-Hundred Watt (MHW) Generator.

Table 1 is a listing of item weights. The total heat source weight is 13 19.61 Kg (43.16 lb).

Hypersonic Flight Tests

Before the aerodynamic advantages of the disk module design must be realized, two events must occur after the separation of the diskpack from its container. First, the disks must separate from one another and second, each must assume a stable, flat-face-forward attitude. To determine whether or not both of these events will occur, the separation conditions were simulated in the Ames Hypervelocity Free Flight Aerodynamic Facility. In this facility models are fired from a light gas gun into a 120 foot-long chamber held at controlled pressure. Shadowgraph pictures are taken while in flight. The pressure in the test chamber may be set at any value between several Torr to several atmospheres.

A typical model used in this test is shown in Figure 3. This model consists of four aluminum discs separated by five lexan spacer discs. This assembly is held together in a lexan sabot which is split axially to make four segments. The sabot is designed to open by aerodynamic pressure as it leaves the gun muzzle. The model then flies down the range ahead of the sabot pieces.

A comparison is made in Table II between pertinent flight and test parameters. The flight values are for the altitude range between 35 and 70 km.

It was not practical to match the flight Mach number in this test, but the qualitative aerodynamic effects are expected to be the same for both flight and test Mach number ranges. The Reynolds number range was matched for flight altitudes above about 60km. The thickness to diameter ratio for the test discs was

0.167 and the ballistic coefficient, $(W/C_D A)$, could not be matched because of material limitations.

The dimensionless ratio of dynamic pressure to the ballistic coefficient, $q/(W/C_D A)$, was matched and varied to cover most of the expected flight range for this parameter. It is this parameter that determines the deceleration of a body in unpowered flight so that the time for separation and stabilization in the test should be comparable to that in actual flight.

The top and side views of a typical disc separation sequence are shown in Figures 4 and 5. The models are moving from right to left. This disc pack consisted of 6 brass discs with no lexan spacers. The first frame (upper left) shows an x-ray picture of the sabot separating from the disc pack about two feet from the gun muzzle. The sabot is free from contact on all sides and the disc pack is still intact. The second frame was taken about 8 milliseconds later and subsequent frames were taken at about 0.8 millisecond intervals. The four sabot pieces are seen to fall behind the disk pack. The parallel lines are reference lines and extraneous marks are flaws in the windows. A careful study shows that the disks quickly reorient themselves in a flat-forward attitude. During this process, they separate from one another and fly as if they had been released as single bodies. This same result was seen for all combinations of Reynolds numbers and $q/(W/C_D A)$ for this test. For the sequence shown in Figures 6 and 7 $Re = 2500$ and $q/(W/C_D A) = \frac{40}{20}$. One shot was also made with the axis of the disc pack initially aligned with the flight direction. The disc pack also separated for this mode of launch and no tendency was noted for discs to remain in contact.

Picture sequences of Figures 6 and 7 show the discs seek a stable flat face forward attitude at hypersonic speeds. To observe the terminal phase of the disc flight,

the deceleration was increased by increasing the dynamic pressure and by making the discs of aluminum to decrease the weight, hence $(W/C_p A)$. For this phase of the test $q/(W/C_p A)$ was of the order of 6000. By the time the discs reached the first picture station the discs were already well separated from each other and the sabot pieces were out of the picture.

The leading disc and the other discs, while still in view, are shown rocking about the stable flat-face forward attitude for the entire sequence, about 29 milliseconds. Figure is an enlargement of the 14th frame of the sequence shown in figure .

Edges stable gyro - ✓
 TI. Corwin ✓

FREE FLIGHT, TERMINAL VELOCITY TESTS

by Vorreiter and Tate (5)

Free fall tests were conducted ~~to~~^{would be expected} to determine whether or not the terminal velocity ~~to~~ to measure the lift to drag ratio, and to measure differences in flight between identical bodies that had different initial release conditions. The models were constructed of plexiglass ballasted to match the shape, size, mass, and mass moment of inertia in both the axial and radial direction of the heat source module under study.

The disk modules were released by hand from the rear ^{door} of a C130 aircraft flying at 4600m altitude and were tracked to a low altitude by radar. The models were launched either with low spin rates and random attitude, horizontally spin stabilized (flat down) or vertically spin stabilized (edge down). No photographic coverage was available. The test set-up is shown in Figure 9. The prevailing wind was measured vs altitude on the day of the test and subtracted from the observed motion to yield a "wind corrected" velocity vs altitude.

The x, y, z velocity components are shown in Figure 10 along with the total velocity V_T . The sinusoidal variation in the x and y velocity components indicates a spiraling motion. All but one of the randomly launched models flew in this helical pattern. None of the models launched either horizontally spin stabilized or vertically spin stabilized exhibited this behavior. This helical flight is explained by Vorreiter and Tate (5) as being due to a gyroscopic precession of a rotating body which has a displaced center of gravity. The data were analyzed to yield lift coefficient and drag coefficient as a function of Reynolds number. They are shown in Figure 11. The change in Reynolds number correspond to changes in air density vs altitude.

The data were also reduced to yield the mean lift and drag coefficients for each model ~~X-47~~

LASL-AEC-OFFICIAL

LASL-AEC-OFFICIAL

IN ADDITION TO

~~also~~ λ a calculated sea-level terminal velocity. These results are shown in Table III.

Noticeable differences existed between models launched with different initial attitudes. The models launched vertically spin stabilized (edge down) demonstrated the lowest terminal velocity. The horizontally spin stabilized models had the highest terminal velocity and the randomly launched were between these two extremes. The reasons for these differences are not known.

[illegible]

REENTRY ANALYSIS

Since reentry experiments are prohibitively expensive considerable effort was expended in terms of both man-power and computer time to generate and exercise mathematical models which would yield believable predictions of the response of the modular disk heat source to various reentry environments. The first step in this part of the study was to define the reentry environment that could be experienced. Since any inadvertant reentry of isotope power sources is the ^{RESULT} of an accident or system malfunction, any assumed set of conditions is at best a guess. One study by Hogfors (7) yields the abort entry conditions that might be encountered by a spacecraft intended for synchronous orbit. These results, available as combinations of velocity and atmospheric flight path angle at 122 Km altitude, were used for this investigation and are shown in Figure 12. In general, only the highest velocity for any angle was used since this represents the case of maximum kinetic energy for that condition.

The trajectories were calculated using the Ames Laboratory Trajectory Program (ALTRAJ) which is a finite difference, point mass program that utilizes a 1962 standard atmosphere, and an oblate rotating earth. Drag coefficients of 2.0, 1.8 and 1.18 were used for the free molecule, hypersonic continuum, and subsonic flow regions respectively. A lift coefficient of 0.344 was used for the subsonic flow regime and since no autorotation was observed in any of our hypersonic free flight tests, a zero lift coefficient was used in the calculation for supersonic speeds.

The trajectories were calculated two different ways. First, for the majority of the calculations, it was assumed that the modular disk heat source was a single reentry body independent of the spacecraft and generator

at an initial altitude of 100,000 ft. Finally, to investigate the effects of spacecraft breakup on heat source survivability, the case of near vertical entry at 10,180 m/sec was calculated using the ballistic coefficient of an intact generator as the initial condition. In this calculation, the ballistic coefficient was instantly changed to that of the modular disk heat source at various ^{assumed breakup} altitudes. This sudden exposure resulted both in an increase in deceleration and a sudden onset of aerodynamic heating.

The trajectory calculations also yielded a ground dispersal pattern for the heat sources which would take the shape of a circle of about 10.2 km radius. The feasibility of recovery is not the subject of this study but it will be mentioned that the search of such an area for lost nuclear payloads as described by Bustamante(8) is fairly routine and has been conducted in the past.

Although over twenty detailed trajectories were calculated, only three will be discussed in detail. These reentry conditions are listed in Table IV. The entry at 10,180 m/sec and -88° is of interest because it is the trajectory that causes the greatest thermal stresses in the heat shield and has the largest decelerations. The entry at 10,881 m/sec and -25° was found to yield the highest internal temperatures of all those calculated. A large number of entries at speeds of 11,200 m/sec and various angles from -5° to -10° were investigated and all found to yield much lower internal temperatures than the -25°, 10,881 m/sec trajectory. Therefore, the -8°, 11,201 m/sec trajectory was selected as typical of these shallow angle trajectories.

Those entry conditions which result in more than one pass through the atmosphere were analyzed ^{on} a parametric basis and are discussed in the section on ablation response. The change in altitude, velocity and stagnation point heat transfer rate are shown for these three trajectories in Figures 13, 14, and 15.

To investigate the effects of spacecraft breakup on the performance of the heat shield, the trajectory with entry conditions of -88° and 10,180 m/sec was calculated with the generator breakup altitude treated as a parameter. The change in stagnation point heat transfer rate for three different breakup altitudes shown in Figure 16.

THERMAL RESPONSE

The distribution of heat transfer rate around a disk is shown in Figure 17 as measured by Harvin and Sinclair(9). Since with the exception of the region around the fuel pellet, the heat source is radially symmetric, only the small portion of the unit shown on Figure 18 need be thermally analyzed.

This geometry shown on Figure 18 was modeled with the Chrysler Improved Numerical Differencing Analyzer (CINDA) transient heat transfer program. This model contained 136 thermal elements or nodes.

Considerable detail and attention was given to the effects of various gaps and voids which are shown on Figure 18. Along with the steady operating temperature, when the external air pressure was in excess of 0.009 atmospheres, the gaps were assumed to be filled with air at 1.0 atm pressure and evacuated at lower pressures. The gaps dimensions were adjusted continuously depending on component thermal expansion coefficients and component temperature. When the deceleration exceeded 0.005g the gaps on the forward side were reduced to zero. *AND THE AFT GAPS WERE DOUBLED,* The cold wall heat input calculated in the trajectory analysis was continuously adjusted depending on the enthalpy difference between the hot graphite surface and the surrounding gas-cap. Corrections for ablative blowing were estimated, found to be very small for a body with this ballistic coefficient and thus were not used. The logic used in this solution technique is shown on Figure 19.

To investigate the effects of spacecraft breakup on the performance of the heat shield, the trajectory with entry conditions of -88° and 10,180 m/sec was calculated with the generator breakup altitude treated as a parameter. The change in stagnation point heat transfer rate for three different breakup altitudes is shown in Figure 16.

THERMAL RESPONSE

The distribution of heat transfer rate around a disk is shown in Figure 17 Marvin and Sinclair(9). Since with the exception of the region around the pellet, the heat source is radially symmetric, only the small portion of unit shown on Figure 18 need be thermally analyzed.

This geometry shown on Figure 18 was modeled with the Chrysler Improved Spherical Differencing Analyser (CISDA) transient heat transfer program. This model contained 136 thermal elements or nodes.

Considerable detail and attention was given to the effects of various gaps and conditions shown on Figure 18. Along with the steady operating temperatures, when the external air pressure was in excess of 0.009 atmospheres, the gaps were assumed to be filled with air at 1.0 atm pressure and vented at lower pressures. The gaps dimensions were adjusted continuously depending on component thermal expansion coefficients and component temperatures.

When the deceleration exceeded 0.005g the gaps on the forward side were reduced to zero. AND THE AFT GAPS WERE DOUBLED.

The cold wall heat input calculated in the trajectory analysis was continuously adjusted depending on the enthalpy difference between the hot graphite surface and the surrounding gas-cap. Corrections for ablative mass loss were estimated, found to be very small for a body with this ballistic coefficient and thus were not used. The logic used in this solution technique is shown on Figure 19.

Shown on Figures _____, 2.1, and 2.2 are the maximum temperatures existing in the shield, fuel ^{LINER} and strength member as a function of time for the three reference trajectories.

At no time does any particular component exceed its melting temperature. Another unusual feature of this design is observable on these three figures is that of the very rapid cool down of the external shield. This cool down is so rapid that for all cases except the entry at -88° and 10,180 m/sec, the graphite shield temperature dropped below the graphite oxidation limit of about 1350°K before the heat source falls to sufficiently low altitudes to encounter significant quantities of oxygen.

The mechanism of convective cooling down during free-fall following a high speed reentry was calculated by a Reynolds number correlation suggested by Kreith (II) for irregular objects. These approximate calculations indicate a shield temperature and strength member temperature of about 30°K and 60°K above the ambient air temperatures respectively.

An analysis of the temperature levels to be expected after ground impact was conducted. The modular heat source was assumed to be stripped of its graphite heat shield and lying on coastal plain soil. The calculations were done with a transient analysis program and the results yielded a post impact temperature of about 670°K which is thought to be allowable for long term exposure of ~~Ti and Iridium~~ ^{Noble Metals to Air} or Iridium-Platinum alloy.

STRUCTURAL ANALYSIS

14-00000-00000

A stress analysis was made ~~for the forward half~~ of the heat shield. This analysis was done with the NASTRAN (NASA Structural Analysis) finite element computer program. Thermal, pressure and inertia loads were included.

The inertia loads due to aerodynamic deceleration of the fuel and strength member were distributed uniformly over the inner portion of the shield. The inertia load of the rear half of the heat shield was uniformly distributed over the outer lip. The body-inertia force of the front half of the heat shield was accounted for in the NASTRAN program as were the thermal load forces. The pressure load was obtained from the calculated stagnation pressure and the pressure-distribution profile given by Inouye⁽¹⁰⁾.

A portion of the finite-element model used in the analysis is shown in Figure 23 along with the manner in which the loads were applied. The center portion (not shown) of the model was made up entirely of rectangular elements of the same size and shape seen at the left edge of this figure.

The input data for the stress analysis was obtained from the thermal analysis and the reentry analysis previously described.

An inter-

polation was made of the temperatures from the thermal analysis to obtain temperatures at each corner of the finite elements in Figure _____. The deceleration and stagnation pressure were calculated for the same time of flight as were the temperatures using the reentry analysis program.

14-00000-00000

Stress in the heat shield was calculated for five release altitudes of the FAD's from their package: 36.5, 48.7, 57.9, 76.1 and 122 Km. For each of these five cases stresses were calculated for thermal loads only and also for combined thermal, pressure and inertia loads. The material properties used in these calculations are shown in Table V.

Lines of constant stress are plotted in Figures 24 and 25 for conditions when the release altitude is 36.5 Km the most severe of the cases examined. The lines of constant stress continue from the left of these figures to the center of the heat shield as essentially parallel lines. The corresponding plots for other release altitudes are qualitatively the same as those shown for the 36.5 Km release altitude. It is generally true that for all conditions circumferential stress is larger than the radial stress and axial stress is negligibly small. Adding pressure and inertia loads increases the magnitude of the maximum stress, particularly the circumferential stress. For the unconstrained shield these stresses are well within the ultimate stress limits of 90×10^6 to 120×10^6 N/m² in tension and 70×10^6 to 165×10^6 N/m² in compression (depending on temperature).

A Summary of the maximum (circumferential) stress for the unconstrained heat shield is shown in Figure 26 as a function of the release altitude.

ABLATION ANALYSIS

The expected ablation of the graphite heat shield was investigated for many trajectories including both single and multiple pass ~~entires~~ ^{re-entries}. The calculations used the ablation equations of Metzger (12) in the reaction limited regime, Scala (13) in the diffusion limited regime, and Scala (14) in the sublimation regime. In only two cases was the ablation found to be significant. For the entry at -88° and 10,181 m/sec the ablation during the subsonic and hypersonic portion of the flight amounted to 0.85mm and 0.25mm respectively for a total of 1.1mm.

The range of entry angles which results in multiple passes through earth's atmosphere was investigated and the ablation vs entry angle shown on Figure 27. Under the worst multiple pass entry condition, only 2.5mm of graphite would be ablated away from the original 4.45mm material thickness.

CONCLUSIONS

The authors have evaluated the design of a disk shaped isotope heat source under atmospheric entry conditions. By launching stacks of disks at hypersonic speeds, we discovered that the disks quickly separate and fly as individual units in a flat forward attitude. When released from an aircraft at high altitude, the disks were observed by radar to fly in a predominately spiral flight pattern with a sea level equivalent terminal velocity of 29.1m/sec.

Considerable effort was expended to predict the thermal and structural response of the modular disk heat source to a variety of potential earth entry environments. It was found that no component of the heat source overheated during reentry and the stresses induced by temperature gradients, pressures, and dynamic loads were far below the expected failure stresses of the heat shield material. The amount of ablation that would occur during a variety of reentry situations was calculated and found to be well below the selected ablator thickness.

The authors feel that the modular disk heat source represents a practical solution to the more severe reentry environments that can be expected ~~in~~ ^{for} future space missions and that the low sea level terminal velocity observed by these units represents a solution to the ground impact damage problem that has plagued almost all space nuclear heat source programs to date.

References

1. Brunner, M. J., Dohner, C. V., and Lawit, R. L.: Re-Entry of Radioactive Power Sources, Journal of Spacecraft and Rockets, Vol. 5, No. 4, April 1968.
2. Bustamante, A. C., and Stone, G. W., Jr.: The Dynamic Characteristics of Autorotating Configurations in Subsonic and Hypersonic Flows, Sandia Laboratories, SC-DC-68-2395, November 1968.
3. Dix, G. P.: Advances in the Safety of Space Nuclear Power Systems, The Second International Symposium on Power from Radioisotopes, Madrid Spain, May 1972.
4. LaPorte, A. H., and Osmeier, W. E.: Growth Potential of the Snap 19 High Performance Generator Through Technology Application, The International Conference on Nuclear Solutions to World Energy Problems, American Nuclear Society, November 1972, Washington D. C.
5. Vorreiter, J. W. and Tate, D. L.: Observations of Disk-Shaped Bodies in Free Flight at Terminal Velocity, NASA TM X 62,262 May 1973
6. Willmarth, W. W., Hawk, N. E. and Harvey, R. L.: Steady and Unsteady Motions and Wakes of Freely Falling Disks, The Physics of Fluids, Feb. 1964, Vol. 7, No. 2, pp. 117-121.
7. Hogfors, H. E.: Velocity-Flight Path Envelopes for Different Failure Modes in a Titan IIIC Synchronous Mission, The Aerospace Corporation Report No. TOR-0172-(2701-C1)-5, March 6, 1973.
8. Bustamante, A. C., Randall, D. E. and McAlees, J.: Hove Sineu-1 Postflight Report, Sandia Laboratories Report #SC-RR-72 0225

LASL AEC OFFICIAL

LASL AEC OFFICIAL

References (Cont)

9. Marvin, J. G., and Sinclair, A. R.: Convective Heating in Regions of Large Favorable Pressure Gradient, AIAA Journal, Vol. 5 No. 11, November 1967.
11. Kreith, F., Principles of Heat Transfer, International Textbook Company, 1958, page 379.
12. Metzger, J. W., Engel, M. J. and Diaconis, N. S.: The Oxidation and Sublimation of Graphite in Simulated Re-Entry Environments, AIAA Thermophysics Specialists Conference, Monterey Calif., Sept 13, 1965, Paper # 65-643.
13. Scala, S. M., and Gilbert, L. M.: Aerothermochemical Behavior of Graphite at Elevated Temperatures, G. E. Report #R63SD89, November 1963
14. Scala, S. M. and Gilbert, L. M.: Sublimation of Graphite at Hypersonic Speeds, AIAA Journal, Vol. 3, No. 9, pp1635-1644, Sept. 1965
10. Inouye, M., Marvin, J. G., and Sinclair, A. R., : Comparison of Experimental and Theoretical Shock Shapes and Pressure Distributions on Flat Faced Cylinders at Mach 10.5, NASA TN D-4397, Feb. 1968

TABLE 1

MODULAR HEAT SOURCE ASSEMBLY WEIGHT

Item	Unit Mass, Kg	Number required	Item Mass	
			Kg	(lbm)
Fuel cylinder	0.037	160	5.95	(13.10)
Encapsulation Shell	0.008	160	1.32	(2.90)
Strength member	0.341	10	3.41	(7.50)
Heat shield	0.572	10	5.72	(12.60)
Spacers	0.176	9	1.59	(3.50)
End cap	0.810	2	1.62	(3.56)
TOTAL			19.6	43.16

TABLE II -- COMPARISON BETWEEN FLIGHT AND TEST PARAMETERS

	<u>FLIGHT</u>	<u>TEST</u>
Mach Number	17-34	5-6
Reynolds Number	11,000-400,000	2500-25,000
Thickness/Diameter	0.114	0.157
Ballistic Coef. (M/P_{DA}), (k_2/k)	35.2	12.7
<u>Dynamic Pressure</u>		
Ballistic Coef.	20-500	40-400

TABLE III
Results, Free Fall Terminal Velocity Tests

MEASUREMENTS	MODELS					
	A1	A2	A3	A4	A5	A6
Mass (Kg)	1.640	1.636	1.645	1.645	1.636	1.636
Initial Conditions	Random Low Spin	Random Low Spin	Horizontal Spin Stabilized	Horizontal Spin Stabilized	Vertical Spin Stabilized	Vertical Spin Stabilized
Equivalent Sea- level Velocity m/Sec	29.134	23.892	31.360	30.189	26.70	28.35
Mean Lift Coefficient	0.305	0.309	0.176	0.263	0.408	0.321
Mean Drag Coef.	1.135	1.153	1.002	1.065	1.332	1.196
C_L/C_D	.268	.264	.175	.246	.306	.268

TABLE IV

<u>Inertial Velocity</u> <u>in/sec</u>	<u>Inertial Flight Path</u> <u>Angle (Degree)</u>
11,202	-8
10,881	-25
10,180	-88

Table V

Thermophysical Properties of Carbon-Carbon
Composite Heat Shield

Property	<u>293°K</u>	<u>1273°K</u>	<u>2200°K</u>
Elastic Modulus (10^8 Newtons/m ²)	29		20
Thermal Coefficient of Expansion (10^{-6} m/m/°K)	-.9	1.8	3.4
Poisson's Ratio	0.15		0.15
Ultimate Tensile Strength (10^6 Newtons/m ²)	90		120
Ultimate Compressive Strength (10^6 Newtons/m ²)	90		165

Fig 1

FIGURE II-1-PAD CONFIGURATION

FIGURE 2
2400 WATT MODULAR HEAT SOURCE ASSEMBLY

Figure 3.- Typical model and sabot assembly.

Figure 4- Top view of disk separation sequence.

Fl.

Figure 1.- Side view of disk separation sequence.

LASL-AEC-OFFICIAL

LASL-AEC-OFFICIAL

Figure 6 - Top view of model stabilization sequence.

Figure 7.- Side view of model stabilization sequence.

has been the source of the left
side of the model

Figure 6 - Enlarged view of stabilized model.

Figure 9

Free Fall, Terminal Velocity Test Configuration

Figure 10
Velocity vs. Altitude for Free Falling
Modular Disk Heat Source

Figure 11
Lift and Drag Coefficient
vs.
Reynolds Number

Figure 11
Lift and Drag Coefficient
vs.
Reynolds Number

INERTIA VELOCITY m/sec

VORREITER
NASA-AMES

FIGURE 12

FLIGHT PATH ENVELOPE AT 122 Km ALTITUDE

Figure 13
 Relative Velocity, Altitude and Stagnation Heat Transfer
 V
 h
 q
 Time

Inertial Velocity = 10,881 m/sec
 Flight Path Angle = -25°

FIGURE 14
 RELATIVE VELOCITY, ALTITUDE, AND
 REFERENCE HEAT TRANSFER RATE
 VS
 TIME

FIGURE 15
RELATIVE VELOCITY, ALTITUDE, AND
REFERENCE HEAT TRANSFER RATE
VS
TIME

FIGURE 16

ALTITUDE ~~122~~ REFERENCE HEATING VS SEPARATION ALTITUDE

(Trajectory: $10,182 \text{ km/sec}$, $\gamma = 88 \text{ deg}$)

FIGURE 17

HEAT TRANSFER DISTRIBUTION ON A DISC

FIGURE 18
 THERMAL MODEL 240 WATT
 With Normal Operating Temperatures

FIGURE 20
 240 WATT REENTRY MODEL
 MAX COMPONENT TEMPERATURES VS TIME
 88° 10,170 m/sec TRAJECTORY

FIGURE 21

240 WATT ~~FA~~ REENTRY MODEL
 MAX COMPONENT TEMPERATURES VS TIME
 TRAJECTORY

- 25° X 10,281 M/sec

FIGURE 22

240 WATT REENTRY MODEL MAX COMPONENT TEMPERATURES
 VS TIME $8^\circ \times 11,186 \text{ m/sec}$ TRAJECTORY

Note: Final figure will have complete Model A Grid

FIGURE 23
FINITE ELEMENT PATTERN USED IN NASTRAN FOR STRESS
ANALYSIS OF
HEAT SHIELD

AXIS OF SYMMETRY

Note: Units of stress
are 10^6 Newtons/m²

FIGURE 24

LINES OF CONSTANT RADIAL STRESS FOR THERMAL, PRESSURE,
AND INERTIA-LOADS; ENTRY ANGLE $= -88^\circ$, VELOCITY = 10,180 m/sec
RELEASE ALTITUDE = 36.5 km

AXIS OF SYMMETRY

Note: Units of stress
are 10^6 Newtons/m²

FIGURE 25

LINES OF CONSTANT CIRCUMFERENTIAL STRESS FOR THERMAL,
PRESSURE AND INERTIA LOADS; ENTRY ANGLE = -88° , VELOCITY = 10,181 m/sec
RELEASE ALTITUDE = 36.5 km

BLANK PAGE

DEPENDENCE OF MAXIMUM CIRCUMFERENTIAL STRESS ON RELEASE ALTITUDE

LASL AEC-OFFICIAL

FIGURE 26

ALTITUDE, KM

2

END — DATE FILMED

FIGURE 27
MODULAR HEAT SOURCE ABLATION
(MULTI - PASS REENTRIES)