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P.A. O'Riordan
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**RTG SPACECRAFT MISSION ABORT STUDY
RTG STUDY - VOYAGER TASK C**

Prepared By:

D. Sergay,
D. Montgomery,
Advanced Interplanetary Project

H. Bermanis,
Advanced Nuclear Systems Operation

Approved By:

A. Kirpich

A. Kirpich
Cognizant Engineer
Voyager Task C, RTG Study
Voyager Spacecraft System Project

Prepared For

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1.0 INTRODUCTION

The utilization of Radioisotope Thermoelectric Generators (RTG's) for the Voyager Spacecraft is predicated on the assurance that at no time in the life of the generators will the general population be exposed to undue radiological hazards. The assumed approach in this study for providing this assurance is to consider that the RTG's must be designed to contain the isotopic fuel for all credible abort environments. The objectives of this study are to:

- a. Examine abort environments of the Voyager mission.
- b. Review appropriate RTG fuel containment concepts.
- c. Examine the effect of fuel containment concepts on the spacecraft design.

2.0 ABORT ENVIRONMENTS

The selection of RTG reentry protection methods is dependent on the relative probability of abort environments. Since RTG weight and its implication on spacecraft weight are dependent on such a selection, it is important to develop a reasonable insight into abort environment possibilities. It is not within the province of this study to conduct a rigorous safety analysis of RTG's applied to a Voyager spacecraft. Along with a specified set of safety criteria, such analysis requires extensive correlation of data from associated contractors and cognizant agencies. It is possible, however, through a cursory examination of abort environment possibilities to detect dominant characteristics and thereby establish probable RTG requirements relative to the abort environments.

For this purpose Figure 2-1 presents possible abort environments resulting from failures during mission phases prior to Earth escape. Qualitatively assessed, it is judged that the abort environment is potentially severe during prelaunch and the early launch phase due to the presence of large amounts of booster propellant. The severity decreases as this propellant is consumed, but increases sharply thereafter when critical altitudes and velocity are achieved. This is associated with the possibility of reentry heating. In all subsequent phases the situation is aggravated by ignition and/or rupture of the Voyager propulsion systems brought on by reentry heating. Information from Aerojet-General, designers of the solid propellant retropropulsion engine, indicates that auto-ignition would occur when the engine casing reached 600°F. Two possibilities could exist thereafter. The engine could operate in its normal propulsion manner, or overpressure could develop in local casing regions with rupture occurring at 650 psi. Propellant burning would then proceed through the rupture. No detonative explosion would be expected, this having been borne out in ground tests aimed at inducing such an explosion.

Reentry heating of the monopropellant midcourse engines on the other hand could be violent since the fuel is contained in spherical tanks at a pressure of about 3600 psi. A safety factor of 2.2 is used in the tank design but this would decrease with heating which, together with an increase in pressure, would result in violent energy release with tank rupture. Estimates also indicate this would occur around 700°F.

If the spacecraft reenter separately or attached to the S-IVB stage, rough calculations indicate that the shroud will only provide minor protection and will melt through rapidly. It seems certain that the propulsion equipment will eventually reach the temperatures that

Mission Phase	Possible Initial Environment	Subsequent Environments	Possible Causes of Abort
Prelaunch	Fireball	Seawater immersion, impact, burial, etc.	1. Saturn V failure 2. Voyager propulsion failure
Launch to Parking orbit	Fireball	Ballistic reentry, impact, etc	1. Intentional abort 2. L/V failure
	Ballistic reentry	Fireball (Voyager propulsion) due to reentry heating	1. Intentional abort 2. L/V failure
Parking orbit	Orbital decay & Earth reentry	Ignite Voyager propulsion by reentry heating	1. S-IVB restart failure
Transfer trajectory injection	Superorbital reentry	Ignite Voyager propulsion by reentry heating	S-IVB Guidance failure
	Decay from eccentric orbit & Earth reentry	Ignite Voyager propulsion by reentry heating	Deficiency in escape velocity

Figure 2-1. Possible Abort Environments

would result in rupture. The main question is when in the reentry phase this occurs. The shroud and other spacecraft equipment may sufficiently delay the impending rupture until after peak reentry heating. In this case, the RTG reentry protection will have served its principal function and the propulsion release of energy thereafter may not impair its fuel containment capability.

From the above considerations the probability of the propulsion rupture is considered to be very high with orbital reentry. It is therefore close in value to the probability of reentry itself. Taking one possible cause that would ultimately result in reentry, namely failure of the S-IVB stage to restart and inject into a transfer trajectory, it is evident that its failure probability is less than the reentry probability since other causes also contribute to the latter.

Figure 2-2 shows recent reliability estimates for the Saturn V launch vehicle. The S-IVB reliability is given as 0.975263 and is defined by $1 - Cr \times 10^{-6}$ where Cr is the flight criticality number whose value is 24,737 as shown. Cr is equivalent to λt where λ is failure rate and t is time. For a given value of λ , Cr is proportional to time. The critical phases with respect to reentry are associated with the "first coast" and "second burn" periods. Assuming that the "first coast" or parking orbit period for the Voyager mission does not exceed 5400 seconds, the value of Cr is proportionately adjusted for time as follows:

$$Cr' = \frac{13,108.9 - (10,792.9 - 5400)}{13,108.9} \times 24,737$$

$$= 14,530$$

The percentage of the total S-IVB operating time that it operates during the first coast and second burn phases is:

$$\frac{5400 + 335.7}{7716.0^*} = 74.2\%$$

$$* \text{ New flight operating time} = 172.3 + 5400 + 335.7 + 1808$$

Then the criticality during these phases is $0.742 \times 14,530 (=10,800)$ and the corresponding reliability is 0.9892. Thus there is a 1.08% chance of the S-IVB failure and, based on the previous conclusion, this is the corresponding likelihood of reentry along with an undefined blast environment resulting from ignition and/or rupture of the propulsion systems. Assuming that fuel containment must be assured to much higher probabilities ($>>0.9892$), it is reasonable that the RTG be designed to survive this combined reentry/propulsion rupture environment.

Data Extracted From:

SATURN V
 RELIABILITY ANALYSIS MODEL SUMMARY
 SA-501
 MSFC DWG 10M 30570
 December 2, 1966

Stage	Flight Operating Time (Seconds) ***	Percentage of Operating Time to Total Flight Time	Flight Criticality	Flight Reliability (R_L)	No. of Single Failure Points Flight & Prelaunch			
					CAT I	CAT II	Prelaunch Vehicle Loss	Total*
S-IC BURN	148.8	1.09	3,435	0.996565	94	73	39	115
S-II BURN	359.4	2.64	28,130	0.971870	61	74	6	121
S-IVB	13,108.9	96.27	24,737	0.975263	41	192	3	215
FIRST BURN	172.3	1.27						
FIRST COAST	10,792.9	76.26						
SECOND BURN	335.7	2.47						
SECOND COAST	1,808.0	13.27						
INSTRUMENT UNIT	13,618.1	100.00	12,285	0.987715	0	43	0	43
SATURN V	13,618.1	100.00	67,031**	0.932969	196	382	48	494

* Some components are critical in all three categories; i. e., CAT I, CAT II, and prelaunch vehicle loss. Therefore, the total quantity of single failure points is less than the sum of the quantities as reflected in the three categories.

** Adjusted criticality number to reflect product of reliabilities (R_L) of stages.

*** Flight operating times were obtained from memorandum I-V-E-619-66, "Nominal Burn Times on Saturn V Vehicles", July 7, 1966, L.B. Bell I-V-E, to J.W. Moody, I-V-Q.

Figure 2-2. Saturn V Reliability Summary

3.0 RTG REENTRY TECHNIQUES

There are two primary schemes for the safe reentry of radioactive fuel for space nuclear power devices. In one case, the fuel is ejected from its containment at a sufficiently high altitude so that the composition and form during eventual reentry would result in a concentration of fuel which is said to be not hazardous to the population. The other method aims at containing the fuel through reentry and impact to a smallest possible geographic area to minimize the possibility of interaction between the fuel and the general population. All of the reentry modes studied here fall under the second category because it is believed to be the more reliable method.

The two intact reentry schemes studied here are separable RTG's with uni-directional reentry heat protection and integral RTG's with omni-directional reentry heat protection.

3.1 SEPARABLE REENTRY

In this method the RTG's are separated from the spacecraft before the onset of reentry heating. They are designed to be aerodynamically oriented so that the subsequent reentry heat pulse is absorbed by a protection shield installed on one side of the RTG. The successful application of this technique is, in general, limited to abort situations in which structural integrity of the booster stage(s), spacecraft and shrouds still exists. This integrity implies that separation circuits, mechanisms and logic are intact to permit initiation and completion of the separation sequence. The following paragraphs present some of the consideration of this approach.

3.1.1 SYSTEM IMPLEMENTATION FOR SEPARABLE RTG'S

Any active means of RTG separation will depend on the following functions:

- a. Abort sensing (determine what has happened)
- b. Information assimilation and interpretation
- c. Decision making
- d. Perform separation
- e. Provide power for separation

Each of these functions is discussed below:

Abort Sensing

The first approach to sensing an abort of the Voyager mission, which requires separation of the RTG's, was to consider the use of logic sensors on each spacecraft. The criterion for the abort sensor was maximum reliability for initiating the jettison of the RTG's. This implies a minimum of in-line events to accomplish this function. The maximum reliability must be attained with minimum weight penalty, minimum demands on the spacecraft interface and system design, and minimum complexity. It is furthermore desirable to use developed techniques if possible in order to minimize development cost.

It soon became evident that the abort system would be complex. It would require multiple sensors such as accelerometers, gyros, memory logic, relays, etc. to develop the proper signal for the separation pyrotechnics and would be duplicated for each spacecraft. A simpler approach is to provide a counter and time register such that if the spacecraft had not separated from the shroud in a given time after lift off, the RTG separation sequence would be initiated. Another approach is to make use of the Emergency Detection System (EDS) which is presently part of the Saturn V launch vehicle equipment. This system is designed to automatically abort a manned mission since the crew may not have sufficient time to respond to emergencies. Since the EDS is designed both to initiate action as well as to display information, it can be used to provide telemetry data for ground control of the arming and initiation of the jettison sequence as well as to initiate abort actions. It would constitute no additional weight penalty for the abort sensing requirements imposed by the RTG's. Furthermore, once separation from the S-IVB has occurred, the planetary vehicles would be isolated from the EDS and thus could not receive false abort commands.

Of various possible methods of abort sensing, the use of the EDS appears most practical and provides a foundation on which detailed abort capabilities could be developed.

Information Assimilation and Interpretation

This is covered in "Abort Sensing"

Decision Making

This is covered in "Abort Sensing"

Perform Separation

Several alternative spacecraft arrangements were examined from the point of view of this function. With minor modifications, the spacecraft design described in Document No. VOY-C1-TR16 served as the model for this examination:

- a. Case 1 - The spacecraft arrangement is shown in Figure 3-1. The serial events required to separate the RTG's are:
 - (1) jettison of the over-the-nose shroud
 - (2) separation of Planetary Vehicle No. 1
 - (3) separation, or jettison, of each of the RTG's in the bus while the planetary vehicle is in free flight away from the S-IVB stage and the second Planetary Vehicle
 - (4) separation of the interstage assembly which joins the lower and upper shrouds and
 - (5) jettison of Planetary Vehicle No. 2 followed by
 - (6) jettison of each of the eight RTG's from the second Planetary Vehicle.

In conjunction with the use of the Emergency Detection System (EDS) identified earlier, an additional time delay function is required on board the spacecraft to prevent the RTG separation until the spacecraft has cleared the shroud.

Figure 3-1. RTG Ejection Sequence, Capsules Above Bus

Figure 3-2 is a functional flow block diagram of a possible control and sequencing system for the RTG separation. The signal to arm separation circuits can only come from the Instrument Unit. A backup path is provided for earth command to the command receiver on the S-IVB and thence by hardwire connection into the spacecraft to arm the separation circuits and to initiate the action of a special control and sequencer which controls the events of the RTG jettison sequence. This special C&S unit is completely isolated from the mission equipment on board the spacecraft, and, upon separation from the shroud assemblies and the S-IVB final stage, no signal can enter the sequencer to initiate jettisoning of the RTG's. The parallel path to the arming and fusing components is a double check on this disabling of the jettison controlling and sequencing unit. This affords a high probability that no spurious signal will cause the RTG's to be jettisoned from the spacecraft when the planetary vehicles are on a satisfactory Mars trajectory.

- b. Case 2 - In Figure 3-3 the two Planetary Vehicles are turned upside down to reduce the number of serial events in the RTG ejection sequence. These are:
 - (1) over-the-nose separation of the shroud
 - (2) direct jettisoning of the RTG's from the upper Planetary Vehicle

Figure 3-2. System Functional Flow Diagram

- (3) the separation of the No. 1 Planetary Vehicle and the combined upper and lower shroud as a unit, exposing the RTG's on the Planetary Vehicle No. 2 which are then
- (4) free to move away from the Planetary Vehicle No. 2

Besides the reduction in sequence steps, the time delay function identified for Case 1 is avoided since the EDS can directly provide the properly timed separation signal to the RTG's.

Figure 3-3. RTG Ejection Sequence, Inverted PV's

- c. Case 3 - In Figure 3-4 the RTG's are mounted outboard of the shroud enclosure. They are joined to the spacecraft by supports which pass through the shroud separation joint. All RTG's are jettisoned simultaneously with no shroud or spacecraft separation required. The major problems of this approach are the unknown aerodynamic effects of outboard RTG's, and mounting complexities associated with the shroud separation joints

For all the cases cited above, a means must be provided for physically separating the RTG's from the spacecraft. A variety of pyrotechnic devices could be used in conjunction with release mechanisms. Figure 3-5 illustrates a conceptual design for the release mechanism which uses redundant Mild Detonating Fuses (MDF) to sever a mounting ring which holds the RTG in place. Four springs subsequently eject the RTG clear of the spacecraft. The MDF's can be used at the elevated temperatures in the region of the RTG without detonating prematurely. A signal to detonate the MDF's would be supplied from an electrical harness routed to all of the RTG's.

Figure 3-4. RTG Ejection Sequence, Outboard RTG's

Provide Power for Separation

Power for RTG separation could be supplied by the RTG's themselves or by peaking batteries. This would depend on the nature of the separation circuit load demand. For Case 1 discussed earlier an onboard spacecraft supply would be required, whereas Cases 2 & 3 would permit the use of booster stage power sources.

3. 1. 2 WEIGHT

The RTG shown in Figure 3-5 is representative of the separable approach and produces 75 watts. The estimated weight breakdown is as follows:

Fuel, generator, heat rejection system	38 lbs
Protection Shield	11 lbs
Separation Mechanism	5 lbs
Mounting Hardware	5 lbs
Total	59 lbs

The greatest weight uncertainty is associated with the protection shield. Its weight was based on assuming that the proper reentry attitude is achieved before the onset of major reentry heating. By keeping the center of mass well ahead of the center of pressure in the reentry direction this

Figure 3-5. Separable RTG

was a reasonable assumption. More rigorous analysis and test of this stability question could result in increased weight for the protection shield.

3.2 INTEGRAL REENTRY

In this method the RTG's are appropriately cladded with reentry protection materials around its vulnerable portions. Such cladding would permit reentry in any attitude and therefore the need to separate the RTG's from the spacecraft is avoided, at least as such separation is required to permit the proper reentry attitude.

As indicated earlier, non-separated RTG's could encounter a severe environment associated with propulsion system rupture. Thus, in a way, avoidance of one problem, namely separation, results in the appearance of another, the rupture environment. The SNAP-27 RTG fuel cask must operate under very similar circumstances in connection with its mounting on the Lunar Excursion Module. Though differences in design and environment exist with respect to the Voyager application, the data gathered on the fuel cask performance will be of direct benefit for Voyager RTG designs.

Total avoidance of the need for separation is of course, a vastly simplifying feature and the ideal situation would be to have the reentry protection designed to survive the propulsion rupture environment as well.

3.2.1 SYSTEM IMPLEMENTATION FOR INTEGRAL RTG's

Assuming that integral reentry RTG's can survive the probable environments associated with and subsequent to abort no active means for implementing this capability appear necessary. The RTG's would be directly bolted to the spacecraft without any provisions for RTG separation in case of abort. An RTG has recently been designed with integral reentry capability. Its principal features are shown in Figure 3-6. The design is based on the use of basic SNAP-27 elements, namely the fuel capsule, thermopile and heat rejection fins. A slotted shell of graphitic material surrounds the hub of the RTG and serves as the reentry protection in combination with the beryllium substructure which acts as a heatsink.

3.2.2 WEIGHT

Analysis of a 75 watt RTG for use on Voyager with integral reentry protection has resulted in the following weight breakdown:

Fuel, generator, heat rejection system	38 lbs
Reentry protection cladding	20 lbs
Mounting hardware	<u>5 lbs</u>
Total	63 lbs

As with the separable RTG the greatest weight uncertainty is associated with the protection system, resulting principally from insufficient definition of the reentry environment after abort.

Figure 3-6. Integral RTG

4.0 CONCLUSIONS

The general conclusions of this study are as follows:

- a. Separable reentry RTG's are applicable to a limited class of abort situations. These are identified as those in which sufficient structural and functional integrity is maintained to permit RTG separation.
- b. The use of separable reentry RTG's would involve extensive design complexity in the spacecraft, shroud, and booster systems.
- c. Integral reentry RTG's are applicable to a broad class of abort situations.
- d. Weight estimates of integral and separable reentry RTG's are comparable. Uncertainty exists for both in the reentry protection weight. The separable RTG weight is uncertain because of aerodynamic considerations. The integral RTG weight is uncertain because of unknown environmental conditions during reentry.