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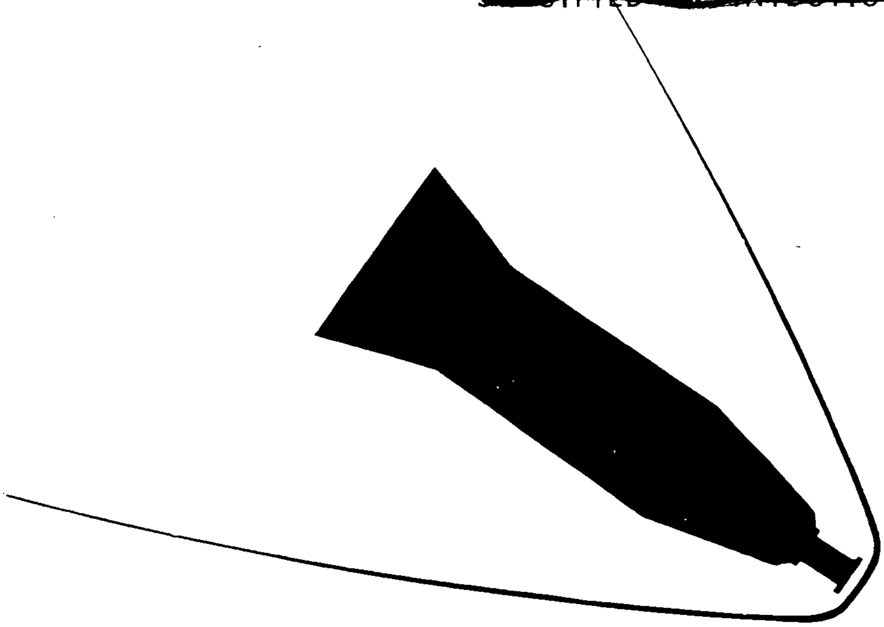

AEROSPACE NUCLEAR SAFETY

A STUDY OF POTENTIAL PRELIMINARY GUIDELINES AND TECHNIQUES FOR AEROSPACE NUCLEAR SAFETY AS APPLIED TO FUTURE SPACE MISSION REQUIREMENTS

Prepared by
Hittman Associates, Inc.

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A STUDY OF POTENTIAL PRELIMINARY GUIDELINES AND TECHNIQUES FOR AEROSPACE NUCLEAR SAFETY AS APPLIED TO FUTURE SPACE MISSION REQUIREMENTS*
(Title Unclassified)

Sandia Corporation
Aerospace Nuclear Safety Department
Albuquerque, New Mexico

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October 1965

ABSTRACT

This report evaluates the proposed United States space programs to 1980 relative to the feasibility and necessity of satisfying the auxiliary power requirements of each mission by nuclear means, and the missions which have an apparent nuclear requirement are selected. In a preliminary fashion the report determines the nature of the nuclear auxiliary power supply best able to satisfy mission requirements.

Implications of mission and power supply characteristics on aerospace nuclear safety for each mission are postulated. The report then evaluates the potential and limitations of techniques and solutions for the aerospace nuclear problems found in the mission and power supply analyses. (U)

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*This report was prepared for the Sandia Corporation Aerospace Nuclear Safety Department by Hittman Associates, Inc., Baltimore, Maryland, under Sandia Contract No. 48-2394. The content of this report is the work of Hittman Associates and does not necessarily reflect the opinions or recommendations of Sandia Corporation.

The information contained in this report is quite comprehensive and is believed to be of value to other organizations which have development or evaluation responsibilities in the SNAP programs. Therefore, we are distributing the report as submitted by Hittman Associates, Inc., to provide the specified distribution with this information on a timely basis.

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I. INTRODUCTION AND SUMMARY OF RESULTS

A. Introduction

As the scope and goals of the National Space Program begin to become clearer and more definitive, the role that nuclear auxiliary power will play in helping to reach these goals also solidifies.

Many missions in the vicinity of earth and the moon, to the planets in our solar systems, and beyond, cannot be seriously contemplated without the use of a dependable long lived power supply independent of solar or chemical energy sources.

As the need and application of nuclear power to these missions grows so do the attendent safety and systems integration problems. Present aerospace safety philosophies for both reactor and radioisotope auxiliary power supplies have been tied to earth orbiting systems in the low and medium power ranges of 10 - 500 watts. However, the full potential of nuclear power is realized in the long duration non-earth orbital missions and the higher power orbital missions.

In these future missions the velocity requirements, mission staging requirements and/or the quantity of radioisotope or reactor fission product inventory will surpass present usage and may constitute a hazard or an impediment to systems reliability or utility if present aerospace nuclear safety philosophies are employed exclusively.

It is the purpose of Task I of this study to:

(1) Evaluate the proposed National Space Program missions over the next 15 year period, relative to the feasibility and/or necessity of satisfying the auxiliary power requirements of the mission by nuclear means, select those missions which have a definite nuclear requirement, and in a preliminary fashion determine the nature of the nuclear auxiliary power supply best able to satisfy mission requirements.

(2) Postulate to the degree possible at this time the implications of the mission and power supply characteristics on aerospace nuclear safety for each of these missions.

(3) Evaluate the potential and limitations of techniques and solutions to the aerospace nuclear safety problems uncovered in the mission and power supply analysis.

The evaluation of potential aerospace nuclear safety techniques includes reentry burnup, intact reentry with recovery, space recovery, orbital storage and escape and the possibility of destruction in orbit. The analysis is primarily concerned with radioisotope nuclear auxiliary power supplies but

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the preliminary aerospace nuclear safety techniques of reactors are considered as well wherever applicable.

The results of this study have uncovered the following preliminary safety guidelines which are considered most pertinent to the application and limitations of the techniques discussed.

B. Summary of Preliminary Safety Guidelines and Results

1. Atmospheric Burnup

a. Complete Burnup

(1) Based on the concept of the NSIa (Nuclear Safety Index for air) which is defined as the ratio of maximum fallout concentration in the troposphere to the MPC for each isotope release for high altitude burnup, the most critical radioisotope relative to eventual fallout tropospheric concentrations is Pu-238. Based upon analyses carried out in this study, a limit of 800 Kw(t) of Pu-238 may be tolerated as injection into the mesosphere before the resulting surface tropospheric concentrations exceed 10 percent of the MPCa for this isotope as defined by NBS Handbook 69 data revised to 1964 values by Reference 10.

Because of the variations in analytical methods now used to predict fallout, a most pessimistic value for allowable burnup power was estimated for Pu-238 and is approximately 8 Kw(t). The optimistic values resulting in an allowable limit of 800 Kw(t) can be justified by the analysis presented herein. However, the pessimistic values were estimated based on the possible variations in present MPC's and fallout analyses assumptions evident from the literature. Table I of this summary presents the burnup performance of all the isotopes considered.

b. Based on the concept of the NSIs (Nuclear Safety Index for surface contamination) for a mesospheric release with minor restrictions, Pu-238 is still the most hazardous isotope. Only 20 Kw(t) can be tolerated in the most optimistic case before 10 percent of the MPCs, including a resuspension factor, are exceeded.

c. For all the nuclear missions and fuels considered in this analysis except Po-210, a mesosphere release yields more critical surface concentrations than the eventual trospospheric concentrations. Pu-238 is the fuel which consistently yields the highest NSI. Po-210 is the only fuel of the group studied which displays a higher fallout NSI for air than for surface concentration. However, for the missions studied, the application of Po-210 consistently yields NSI's five to seven orders of magnitude below the MPC. Hence, no significant hazard is envisioned with the use of Po-210 coupled to high altitude burnup if this burnup can be achieved.

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Table I

Range of Allowable Injection Power Levels of Radioisotopes of Interest for Atmospheric Burnup

I. Mesospheric Release (Typical Satellite Decay)

A. Ground Level Air Hazard — B. Surface Hazard (Minor Restrictions)

Isotope	A. Optimistic Power * Level (Kw(t))	A. Pessimistic Power* Level (Kw(t))	B. Optimistic Kw(t)	B. Pessimistic Kw(t)
Pu-238	800	8	20	5
Cm-244	3000	20	500	30
Sr-90	15,000	200	150	20
Pm-147	300,000	3,000	10^5	10^4
Po-210	500,000	5,000	8×10^6	8×10^5
Ce-144	1,500,000	20,000	10^7	10^6

II. Stratospheric Release (Typical Ascent Abort)

A. Ground Level Air Hazard — B. Surface Hazrd (Minor Restriction

Isotope	A. Optimistic Power Level (Kw(t))	A. Pessimistic Power Level (Kw(t))	B. Optimistic (Kw(t))	B. Pessimistic (Kw(t))
Pu-238	150	2	20	1
Sr-90	5,000	30	100	10
Po-210	30,000	150	10^5	$5 \times 10^2 - 10^4$
Pm-147	80,000	300	15,000	10^3

*The explanation of the basis of the optimistic and pessimistic Hazard Indicies are given in Section IV. A of the text.

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For the majority of missions where Pm-147 is considered applicable as a fuel and where high altitude burnup is desired NSI's of from four to seven orders of magnitude result. Therefore, it is felt that Pm-147 will be acceptable as a burnup fuel if the fuel form can reach the desired size range during burnup.

d. Stratospheric release of the fuel approximating ascent aborts with partial burnup yields a limit of 100 - 200 Kw(t) for an NSIa = 0.10, the most optimistic case for Pu-238 systems, which are still the most critical. This represents a real power range of from 10 - 40 Kw(e) depending on the power conversion system used and the amount of partial burnup experienced in the stratospheric release and this yields a most optimistic limit of 20 Kw(t) for an NSIs = 0.10 (minor restrictions). Pu-238 systems are still most critical even considering full release of high vapor pressure fuels such as Po-210 versus partial release of Pu-238.

e. Low altitude and ground releases are the most serious relative to earth surface contamination with Pu-238 again being the most critical isotope for contamination if vaporization is considered. However, this event has a low probability of occurrence with Pu-238 due to characteristics of its fuel forms and the available energy from ground aborts. Therefore, Po-210 must be considered the chief danger under these circumstances.

f. Not enough energy is available during satellite decay to destroy advanced reactor fuel elements such as UC, UO_2, or UN. High power plant operating temperatures are more critical than high power levels relative to achieving burnup because of the high temperature materials they require in fuel form and fuel element design.

g. In general, there are two main mechanisms for molten debris reduction during reentry, mechanical breakup due to aero pressures and aerodynamic heating of the resulting debris. Such thermodynamic assets as superheat upon release yielding flashing are added benefits which the fuel form designer should try to employ but are difficult to come by in all fuel forms of interest except Sr and Po-210 metals.

h. The isotope fuel forms most susceptible to chaotic spray as a primary destruction mecahnism are Sr metal, SrF_2 and SrSi.

i. The fuel forms most susceptible to total reentry vaporization are Pu and Sr metals and Po-210 as either a metal or a rare earth polonide.

j. In general, simple vaporization coupled with low vapor pressures, oxidation heating and possible chaotic spray make the possibility of reaching submicron sizes a real one for those materials which can demonstrate the necessary properties.

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k. If the designer were to define those properties of greatest importance which should be incorporated into a fuel form expected to burnup to acceptable sizes upon reentry into the earth's atmosphere, they would be:

(1) low melting point

(2) a molten release from the fuel capsule

(3) low heat of fusion

(4) low surface tension

(5) low heat of vaporization

(6) low vaporization temperature

(7) low emissivity

(8) high heat of reaction

(1) From the analysis carried out in this study it can be stated that those fuel forms, whether reactor or radioisotope, which have the capability of being coupled to high temperature, i. e., advanced power systems, will be by their very nature, unable to meet the burnup criteria. Metallic fuel forms seem to show the best burnup performance but their use in high temperature radioisotopic systems ($T \sim 1800^{\circ}F$) requires them either to operate in a molten state or as an oxide fuel form (solid state).

The molten state may require reflectory liners (i. e., tantalum) in the capsule to prevent internal corrosion under normal operation and these in turn may hamper the burnup of the capsule during reentry. If a capsule liner could be developed to hold liquid metal under high temperature generator operation and still be able to be breached during reentry to release the molten metal, the burnup aerospace safety philosophy might be able to be extended to advanced power systems. The breaching mechanisms of internal pressure and oxidation hold promise to achieve this, especially for the α- emitters, since the refractory liners needed would be low in strength after burnaway of the outer capsule wall and surface oxidation.

Relative to the radioisotope fuel forms of most interest, as defined by the mission analysis in Section II (Pu-238 and Po-210) high altitude burnup is an acceptable safety philosophy provided that the Nuclear Safety Index as defined in the text is not violated. Cm-244 would have to reenter intact. For reactors using uranium-zirconium hydride fuel elements, burnup seems marginal at best and final reduction of the fuel droplets must come from oxidation and explosion phenomena whose effects are as yet not completely

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defined. Based on the mode of release of the fuel elements, their size and the reentry processes which are expended upon the fuel form and resultant drops, reactors seem not to make as optimum use of the available destruction potential of the atmosphere as do radioisotope generators.

2. Partial Burnup with Disperal

a. With the advent of the higher performance UO$_2$ and UC reactor systems acceptable end of life destruction on reentry will become even more remote. Therefore, SNAP reactor fuel elements made of UO$_2$ micropsheres utilizing a dispersal reentry safety philosophy should be actively considered as a solution to the end of life reentry problem with high temperature systems. Possible sintering of the fuel particles is the greatest drawback.

b. Reentry from cis-lunar or planetary missions will inherently increase energy available to burnup the nuclear device. However, although burnup is assured, it may no longer be high altitude burnup. Because of the greater speeds of reentry and the steeper angles inherent in the trajectory for reentry in one earth revolution, burnup can occur at much lower altitudes. This is especially true if the nuclear device is buried within a reentering probe vehicle.

Therefore, eject mechanisms or placing the fuel source on the outside of the reentry vehicle to assure early fuel releases is a prerequisite for considering burnup as a nuclear safety philosophy in this case.

c. The concept of PuO$_2$ or other type of microsphere fuels and utilizing a fuel capsule which burns up at high altitudes and allows for adequate dispersal of inert fuel form is applicable as an aerospace nuclear safety technique to many missions of interest. Satellite decay cases are the best application for unmanned systems. The most critical hazard is skin dose on the earth's surface since ingestion, water contamination and inhalation hazards can be minimized by proper selection of fuel form.

It was found that powers of the order of 20 Kw(t) can be released as microspheres under the conditions assumed before the percent of background surface radioactivity reaches the natural level assuming a 500 square mile impact area. Hence, all radioisotope power systems up to at least 1 - 5 Kw(e) look safe with the micropshere capsule burnup and dispersal technique assuming no vaporization of the fuel and an insoluble fuel form. Effects on solubility may occur if surface melting and oxidation take place during reentry. In addition, re-solidification may fracture the microspheres into inhalable sizes. Experiments are recommend to establish validity of this potential hazard.

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For reentry at planetary return speeds, ~ 50,000 ft/sec, a significant portion of the microspheres may vaporize at a low altitude, thereby injecting inhalable sized particles into the troposphere. Hence, precautions must be taken to expose the fuel capsules to the air flow immediately upon reentry.

3. Fragmentary Burnup

Burnup reentry resulting in large inert fragments reaching the earth's surface was also evaluated. It was concluded that this approach negates the advantages of the microsphere approach because it severely reduces the benefits of the dispersal philosophy.

However, its application to real systems is highly dependent on the character of the fuel form. Pu-238O_2 is the most likely candidate for large component fragmentary burnup. The reduction in possible ingestion hazards by keeping the source in large pieces or close to it and the dispersal of fairly large pieces of insoluable relatively low radiation material would serve the purpose of localizing the hazard and not exposing as many people to the radioactive debris as with micropsheres. However, in all cases, if exposure to the reentry debris does occur, it will be more severe than with the microsphere approach. In the limit, complete intact reentry of the source is the logical extension of this technique.

It was concluded that large fragmentary burnup cannot be used in a random reentry case unless the probability of hitting a highly populated area (cities) is very low. A tradeoff between debris size and contamination must be made for each mission in order to finally access the application of this technique

4. Intact Reentry and Attendant Recovery

a. Uncontrolled Random Intact Reentry

(1) The controlling factor in application of uncontrolled random reentry is the magnitude and nature of the radioactive inventory being returned to earth. Because of this uncontrolled random reentry is limited to the lower power radioisotope systems. The increase in radiation shielding weights for population protection due to the use of Sr-90 for example could make uncontrolled random intact reentry unattractive even a low power levels.

(2) Pu-238, Cm-244, Po-210, and Pm-147 fuel forms seem to represent the best compromises of minimum weight random intact reentry system and minimum hazards to the general public if impact in an inhabited area results.

(3) In general, the total heat loads expected for a Martian or Venusian planetary entry would be less than those experienced in a normal earth satellite decay reentry. Hence, designing the heat source for random earth reentry represents a good design limit even for planetary operations.

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(4) The critical reentry cases uncovered by this study are earth return at super satellite speeds from cis-lunar and planetary missions.

(5) Uncontrolled random intact reentry of small systems can be accomplished by intact reentry of the fuel capsules, intact reentry of the total fuel block, or intact reentry of the complete generator.

(6) The minimum weight intact random reentry system being limited to small generator (less than 100 watt (e)) seems to resolve itself in the form of reentering the total generator. Two philosophies are possible to achieve this result. The first is to allow the generator structure to absorb the reentry heating and be destroyed at a low enough altitude so that the remaining reentry energy will not destroy the fuel capsules. The second is to protect the entire generator assembly with a thermal system which can reject reentry energy at a rate fast enough to protect the support structure of the generator and keep the fuel capsules from melting down during the reentry.

The first approach is straightforward in nature and does not require any change in present generator design other than replacing the fins and shell with a high heat capacity material such as beryllium. This will allow good heat transfer during normal operation and sufficient heat capacity to prevent capsule meltdown during reentry.

However, there are several basic flaws with this approach from a safety standpoint which make its use questionable. First, beryllium which seems to be the only material capable of playing the dual role of a good thermal conductor and good reentry heat sink is quite weak at the average surface temperatures encountered during reentry (2000°F) and advanced techniques such as composite structures or addition of ceramic fibers are required to increase high temperature strength. These are not state-of-the-art at present. Therefore, the question of whether the generator structure will stay together long enough to absorb the majority of the reentry heating as anticipated is not clear. This is not too critical in a ballistic abort due to the short heating times, but does become important as the aborts get closer to reentry angles reminiscent of the satellite decay case. If increase in strength of the beryllium has to be accompanied by lower temperatures it will reflect back to increased weight of the system under the same reentry conditions. Hence, it is not clear that a minimum weight system will result from this approach. Secondly, even if the reentry is successful and the fuel capsules are released at an altitude where they can survive the remaining aerodynamic heating, high impact velocities and the possibility of ground burial further complicate this technique. Based on past experience with low temperature ($T \sim 1500^{\circ}$F) PbTe thermoelectric systems, enough impact strength can be designed into the system to have reasonable assurance of survival. However, this is not the case with the higher temperature systems envisioned for future use, especially

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high temperature (Si-Ge) thermoelectric and thermionic systems. Some structure around the fuel capsule must remain to absorb impact energies and probably to avoid oxidation of the required refractory metals. Finally, in any case of free capsule impact, ground burial is a serious problem which could yield meltdown even of ceramic fuel forms and release of the fuel form if it occured. Drop tests conducted by Atomics International indicate burial in typical pasture soil of up to 18 inches can occur for cylindrical capsules of L/D of 4.

Therefore, since a conservative philosophy relative to eventual hazards from random intact reentry must be adapted, it would seem that the technique of maintaining the generator structure and using it both as a reentry body, impact energy absorber and preventative to ground burial, is the most reasonable alternate for intact reentry of the small radioisotope systems.

(7) For missions where both satellite decay or cis-lunar and planetary return reentries could occur special heat protection systems are required as a backup. The system could combine the advantages of a gaseous ablator and reradiator. Char ablators such as insul cork have demonstrated these properties.

(8) From the analysis presented herein, it can be seen that to reduce the total heat load a configuration must absorb, its ballistic coefficient should be maximized. This can be achieved practically by maximizing the drag and hence the profile area of the configuration. The implications of this fact might result in low power generator designs which use Be rear plate radiators which serve a dual purpose as drag brakes during reentry.

(9) Intact random reentry with delayed recovery achieved via a balloon technique as described in Section IV. B is applicable to low weight systems only due to inflation gas tankage weight limitations. It is an active system and its components must be developed to a high degree of reliability before it can be considered state-of-the-art. However, it should be studied further.

(10) It has been determined in Appendix H of this study that significant loss of impact strength of present fuel capsules due to evaporation could occur after high temperature vacuum operation in space for relatively short periods of time. Further study is recommended to uncover materials or fabrication methods to alleviate this problem.

b. Semi-Controlled Intact Random Reentry

(1) With the requirements for active reentry systems becoming more prominant to achieve safe intact reentry with large heat sources, the use of the lifting body as a reentry technique has been investigated.

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Semi-controlled random intact reentry was derived as a technique to use the inherent range and maneuver capability of a lifting body to assure disposal of the nuclear heat source in deep ocean areas after successfully completing random intact reentry. Its application was specifically designed for unmanned systems using large heat sources and not having the capability of fully controlled intact reentry. In addition, the system is attractive from the standpoint of not having to depend on ground station control in case of an abort.

(2) It was determined that from the standpoint of general range and lateral range the vehicle proposed has the performance necessary to land a nuclear heat source in water from any given reentry position.

(3) The limiting factors affecting feasibility of this technique all appear to be centered in development of flight controls and sensors capable of operating after long time exposure to space and the thermal and nuclear environment of the heat block. However, good short time reentry performance can be expected. These systems are not yet state-of-the-art but it is felt that they could be developed quickly. The control sequence for the vehicle is a simple one resolving itself into a go-no-go decision controlled by a large water body sensing device.

(4) Minimal shielding will be sufficient to protect typical electronic circuits from even a 10 percent degradation in performance considering a 20 Kw(t) Pu-238 or Po-210 source in a 300 n. mile orbit. The most severe radiation problem might come from space radiation rather than from the heat source.

(5) Therefore, semicontrolled intact reentry is seen to require a very complicated reentry vehicle whose reliability would be low compared to other possible techniques. However, active systems are the only way controlled intact reentry of large intact sources can be accomplished and it is felt that the components required can be developed and made reliable to the extent necessary to achieve semi-controlled reentry.

(6) The basic question which affects the applicability of semi-controlled intact reentry is not feasibility but application. That is, is the designer willing to use a reentry vehicle which has its main usefulness on short lived orbits (aborts yielding one orbital pass before random reentry) where ground control may not be functioning, or will he provide enough backup in the basic system so that ground control can never be lost. If the latter is the case, then controlled intact reentry making use of a ballistic vehicle can do all the jobs that semi-controlled intact reentry can do. At present the expense of providing a worldwide network of control stations along the launch orbital track of any nuclear system launch would be considerable. However, since this technique would be used with large heat sources and those in turn would be used with

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large heat sources and those in turn would be used with major missions (manned planetary or FM Broadcast, etc.) it is felt that the tracking net would be available anyway to assure basic mission success. Hence, semi-controlled intact reentry does not appear as attractive as controlled ballistic for the missions which would require its application.

(7) The lifting body concept, however, is still attractive for precise control of the final stages of controlled ballistic reentry and should be investigated further.

c. Controlled Intact Reentry and Recovery

(1) Controlled intact reentry is most suitable for large heat sources from both the hazards and economic standpoint. In addition, random intact reentry of large heat sources probably cannot be tolerated from a political standpoint because of international problems which could be raised by possible irradiation of an alien population or the associated delivery of large quantities of fissionable material to a potential enemy.

(2) Because of the nature of the end product of controlled intact reentry, that is, delivery of at least the nuclear heat source to a specific spot on the surface of the earth, only systems which are active in nature can be thought of for successfully meeting these goals. Passive systems which only have the capability to withstand reentry and impact can only satisfy the random intact reentry criteria. The active system must have the capability to leave the space environment at a particular point in its orbit or trajectory on command of a ground or space station, dissipate its inherent kinetic energy without compromising the integrity of the heat source during reentry or impact and signal their earth impact position to search parties if recovery is required.

(3) The complexity of achieving this goal is primarily a function of whether the vehicle system using the nuclear power supply is manned or unmanned. The heat source could be brought back with the crew in the mission module reentry vehicle or resupply vehicles thereby using the normally available mission reentry system or it could be jettisoned by the crew at a specific point in the orbit or trajectory of the manned vehicle to reenter ballistically within its own heat protection system.

(4) If the heat source returns with a manned reentry body the basic problems which arise are concerned not with the reentry but with integrating the heat source into the command and service modules during all the operational phases anticipated.

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(5) Location of large heat sources in the service module is suggested to take advantage of the inherent nuclear shielding of the crew by the mechanical systems and separation distances; and the reductions of thermal load to the ecological system of the command module.

(6) The heat source can be brought into the command module via thermally insulated boom before initiation of reentry. It can be stored in an insulated and shielded compartment during the reentry and cooled with residual cryogenics from the life support system. It can be dropped at the end of reentry($\sim$ 100,000) feet or carried on to impact with the reentry body if recovery is required.

(7) The implications of the technique probably will require that the heat source be radiatively coupled to the converter.

(8) It has been determined that radiation coupling of the heat source to the converter for all intact reentry designs is the technique which yields the highest probability of separation.

(9) For smaller heat sources which can be incorporated into the command module the area in the vicinity of the heat shield offers maximum volume and shielding potential to the crew. The techniques of cooling during reentry are the same as in the remote heat source case. Implications of this technique require the converter to be integrated directly with the heat source and coolant lines to the radiator to be explosively separated to free the command module from the service module upon reentry.

(10) For manned systems where the heat source is to be reentered without the benefit of a manned reentry vehicle the heat source must have its own reentry body which will protect the system from reentry heating and from meltdown by internal heat generation.

(11) The basic operational mode for the reentry body is separation from the manned system, spin stabilization in proper reentry attitude and firing of retros at a point in the trajectory where earth impact is assured within the landing range desired. This technique is applicable for both orbital and planetary return systems. However, space disposal is more attractive when planetary return is considered.

(12) The operational techniques for intact controlled reentry of the heat source for unmanned systems are the same as for manned systems except for the control of the events being strictly in the hands of the ground station. This, of course, affects the reliability of the operation since on board mechanical and electronic systems are the only bridge between activating the nuclear source reentry and the ground. Therefore, sufficient redundancy and nuclear shielding must be provided to assure a high probability of success.

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(13) It was determined that for orbital altitudes below 500 miles retro rocket weight is no larger than 10 percent of the reentry vehicle weight. However, the effects of mission profile on the required retro energy is quite severe in terms of impact at a given surface point after the command to fire is obeyed by the vehicle. No more than a ± 10° deviation in vehicle attitude can be allowed by assuming a ± 50 mile error is acceptable at impact.

(14) A gravity gradient stabilization system appears the most attractive for initial attitude control of unmanned intact reentry vehicles which are not attached to the end stage booster. Some active control will still be necessary to control rotation in a plane normal to the earth's radius.

(15) The design of reentry vehicles which are told when to disengage from the spacecraft is far simpler than semi-controlled random reentry. Since, under the circumstances, the vehicle contains no flight control system and must only survive a ballistic reentry. Therefore, the main problems are expected to be thermal and mechanical and not involved with complex electronic control systems. On board electronics will consist of a command receiver, action circuitry and surface location beacons and transponders.

(16) Any isotopic heat source designed for intact reentry must consider the unique heat transfer and heat storage problems associated with high internal heat generation rates during reentry. Specifically, based on present materials technology, the fuel capsules must be able to dump sufficient heat from their surfaces to limit the fuel-wall interface temperature to 1800°F or less in both normal and abnormal operating modes. During normal operation it is not difficult to select system operating parameters such that the temperature limit for materials compatibility is not exceeded. The problem of designing the heat source for intact reentry becomes difficult when consideration is given to the need to reject heat by some alternate mechanism in the case of failure of the primary heat rejection mode. The difficulty arises from the need for an alternate heat rejection path around a heat shield which is designed to reject the aerodynamic heat of reentry.

(17) Two techniques may accomplish this. The first is a standard reentry body which dissipates heat by jettisoning insulation during a coolant loss abort and has enough heat capacity to absorb the nuclear heat during reentry. The second is a device called a rotornet which has the capability to reduce the reentry heat protection weights considerably. However, no change in internal heat storage capacity weights is expected. It is this second technique, however, which holds promise of making intact controlled reentry from aborts at planetary return speeds feasible for nuclear systems.

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5. Recovery

a. Space Recovery. Fuel recovery vehicles for orbiting nuclear systems in space are beyond present technology, unless both systems are manned. Taking a nuclear heat source back to earth in a resupply vehicle is nearly state of the art and can be accomplished with minimal changes in the design of the reentry vehicle. However, if both systems were unmanned and the satellite randomly tumbling, securing the heat source to the rendezvousing vehicles would be almost impossible without men on board. Therefore, the most successful nuclear heat source recovery system is envisioned as a manned satellite interceptor type of vehicle.

Development of such a vehicle seems to be reasonable in the 1970-1980 time period. Special radiation protection for the crew will, of course, be necessary.

b. Ground Recovery. Analyses have shown that recovery of the nuclear heat source after controlled intact reentry within a predetermined impact area should pose no severe technical problems and can be carried out with present nuclear device handling techniques.

6. Orbital Storage and Escape

a. The use of orbital storage to minimize or eliminate potential nuclear safety problems due to reentry depends upon the success of the original launch, the choice of orbit, maintaining attitude control of the satellite, and choice of a propulsion system.

b. Three techniques are open to the designer all of which require propulsion:

(1) to maintain original orbit;

(2) to transfer to higher longer lived orbit;

(3) to escape.

c. In all cases of increasing orbital lifetime through the use of a thruster, attitude control is critical.

d. Small chemical propulsion systems are light in weight and best in reliability.

e. The best transfer technique is Hohmann with a single impulse.

f. Orbital storage appears to be an effective method in minimizing or eliminating potential nuclear safety hazards associated with future aerospace missions. It is, however, limited to larger systems where weight penalties induced by the propellant, tankage, guidance and other pertinent considerations do not appreciably affect system characteristics, such as specific power (watts/lb), size, or spacecraft-power system integration considerations. For missions characterized with orbital altitudes greater than 400 to 600 n. mi. (circular orbits),

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depending upon the radioisotope, the system can be abandoned or recovery of fuel can be achieved when techniques for this are applicable and available because the orbital lifetime will be sufficient to eliminate any reentry hazard via an orbit decay.

g. For missions below 400 n. mi. and above 100 n. mi. Po-210, because of its short half-life, may be acceptable if the systems ballistic coefficient is between 1 and 1000, respectively.

h. Escape from earth in all cases considered required an amount of fuel in excess of 3 - 4 times the payload weight. Attainment of a solar orbit requires 30 times the payload weight. Therefore, a prohibitive penalty in weight for this technique seems evident for any system considered.

7. Potential of Nuclear Systems for Destruction in Space

a. Isotope Systems

(1) Inherent Destructive Capability. Two basic forcing functions appear to offer some measure of inherent destruction capability for radioisotope systems. They are the stored thermal energy and possibly the pressure buildup for α-emitting radioisotopes which generate helium gas.

It appears from our analysis that both of these forcing functions do not have the capability to reduce the fuel form to micro-sized debris because the energy cannot be released quickly and efficiently transmitted to the fuel.

However, this may not be necessary for some fuels because of the potential of space evaporation as a useable destruct technique for the complete fuel mass. Hence, the use of the thermal and mechanical potential of the isotope fuel form could be one of breaching the fuel form containment rather than trying to destroy the fuel. After fuel capsule breaching evaporation in space vacuum can deplete the fuel form considerably before eventual reentry. Therefore deliberate cooling system shutdown is one technique available under these circumstances to breach the fuel capsule.

However, before this technique is employed, one must be assured that sufficient time exists in the space environment, before reentry, to evaporate the fuel to an acceptable degree.

It must be pointed out, however, that the meltdown of the capsule in space and subsequent evaporation of the capsule and fuel form could produce problems upon reentry of the residue into the earth's atmosphere. An alloy of the fuel form and the containment material could form, whose thermophysical properties are much different than the original constituents, and thereby inhibit burnup. Therefore, this technique must be applied with caution as a function of the fuel form and containment material used.

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(2) Fragmentation and Evaporation. It was determined for both Pu-238 and Sr metals that no gain is obtained by fragmenting the fuel form relative to the amount of mass loss which can be obtained by evaporation. The dependence of vapor pressure on temperature proved the most significant variable. When fragmentation occurred the equilibrium temperature of the fragments dropped as a function of the number of fragments, causing a significant decrease in the overall mass loss in space. Po-210 with its high vapor pressure will evaporate very quickly.

Hence, keeping the heat source intact with as large a ballistic coefficient as possible will tend to maximize the abort lifetime and provide maximum time for evaporation. For Po-210 fuel forms and Sr metal breakup of the capsule (via mechanical or thermal means) during a short life orbital abort is acceptable and represents a disposal technique with definite potential. For Pu-238 on the other hand, this technique is marginal and its usefulness is a function of orbital lifetime. If post-abort orbits of the length of a year or greater are assured then the evaporative technique is a valuable one.

In any case pure fragmentation in space even without significant evaporative mass loss is a valuable safety technique since it will assure excellent dispersal of the fuel form before reentry and eventual lower surface contamination. However, the problem here is in assuring the degree of fragmentation which will be significant. This may be easily done for large systems such as the NERVA but it is much more mechanically difficult and costly in terms of weight on the smaller isotope systems. Therefore, the pre-fragmented microsphere approach with simple fuel capsule meltdown on reentry, followed by microsphere dispersal, appears to be the most logical choice to augment space dispersal.

In general, it can be stated that for materials with high vapor pressures it is best not to break up the fuel form in space, but just breach the capsule and allow evaporation to deplete the inventory as much as possible before reentry. There appears to be a correlation between high vapor pressure and burnup reentry potential. Therefore, space evaporation coupled with reentry burnup for these materials is attractive. However, for low vapor pressure materials, an explosive system and/or a preformed fuel form (microspheres) coupled with capsule meltdown is attractive from the standpoint of reducing eventual hazards through maximizing reentry dispersal.

(3) Chemical Destruct. It may be possible especially in manned systems (such as a space station) to dispose of a spent radioisotope generator fuel charge by chemical techniques.

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The key characteristic affecting application of this technique is rate of reaction which in turn is controlled by the mass and geometry of the fuel capsule. In general, it is felt that times of the order of days and weeks, not hours, may be required to bring complete dissolution. Hence application to deep space orbital systems seems most reasonable.

(4) Orbital Destruction by External Means. It has been determined that using external means such as satellite or ground launched missiles to destroy the generator in space could not be effective unless a nuclear warhead were used. Even a direct hit with a chemical explosive system may not breach the fuel containment structure, hence only temperatures of the level to produce full vaporization would be effective.

In addition, based on the limitations of the present test ban treaty nuclear destruct in orbit probably cannot be thought of as a practical orbital destruct technique.

b. Reactor Systems

(1) Inherent Destructive Capability. Analyses have shown that complete destruction of SNAP or NERVA type cores by deliberate excursions is not obtainable. The mechanical forces in the core generated by the temperature of the excursion tend to expand and/or breakup the core prior to maximum power generation.

If complete destruction of a space power reactor is a specification for an operating reactor then certain design innovations can be attempted which may not affect normal operating characteristics. These are:

(a) Design the core to have a control element permanently installed in the core; this element, when withdrawn, could increase reactivity by several percent. A non-nuclear explosive or fuse could be used to eject the control element at time of destruct.

(b) Design the core to have a very short prompt neutron generation time. In addition, use fissile fuel such as Pu-239 which has a small delayed neutron fraction and possibly a positive Doppler coefficient.

(c) Design the core to have a very flat power distribution both radially and axially such that the entire core can achieve maximum temperature and the themal gradients that produce mechanical disassembly will be minimized.

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(2) One alternative to a complete destruct of the core is to use the nuclear excursion to trigger a chemical explosion. The core design would then have to contain materials that could have a violent chemical reaction at high temperature but be chemically inert at operating temperatures.

Coolant shutoff and subsequent core meltdown in space was also investigated. It was found that a decrease of 17 percent of the total fission product inventory could be expected for a typical power reactor. However, this still leaves a significant inventory which may reenter.

C. Applications of Safety Guidelines to Early Missions

Table II represents a preliminary estimate of early missions which may be flown with nuclear power in the 1965-70 time period and the attendant safety philosophy which is best suited based on the information available and generated in the study. For the low power orbital systems reentry burnup has the capacity to serve as the final disposal technique. The fuel forms are predominantly Pu-238 and Pm-147, however, Po-210 and Sr-90 may also be used.

The choice of burnup or dispersal of microspheres as the ultimate disposal technique for these smaller systems is dependent on the available suitable fuel form. The weight penalties which would be charged to these generators if intact reentry were used and the fact that judicious use of the burnup and microsphere techniques can keep atmospheric and surface contamination well below MPC limits was also considered. This is based on an average total isotope launch inventory of 300 Kw(t) in the 1965-1970 time period and a probability of launch success of 0.950. This yields a possible total abort inventory of 15 Kw(t) which may be added to the atmosphere. Pu-238 in its various forms was considered to represent approximately 50 percent of this inventory (i.e., 7 Kw(t)) of which 3.5 Kw(t) was considered to enter the atmosphere as high altitude burnup and 3.5 Kw(t) as microspheres. Pu-238 was considered the most significant isotope to base a safety criteria selection on because of its obvious high launch preference and its most serious final disposal hazards position relative to the other isotopes of interest.

The 3.5 Kw(t) of Pu-238 burnup yields a maximum fallout hazard considering a low altitude (stratospheric) release an order of magnitude below MPC. If a high altitude release is assumed, as would be the more probable event from ascent aborts or orbital decay, the hazard drops to between 3 and 4 orders of magnitude below MPC.

The 3.5 Kw(t) of Pu-238 microspheres represent a maximum hazard about equal to the natural background radioactivity assuming a contaminated area of 500 square miles. The most critical fuel form in this class of launches relative to selection of a safety philosophy is Po-210. Because of its volatile nature the Po-210 fuel form yields a greater hazard than all fuel forms considered if the capsule is breached at low altitude or in an on the pad abort.

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Table II

Preliminary Estimate of Early Missions Which May be Flown With Nuclear Power in the 1965-70 Time Period

	Mission	Approx. Time Period	Type of Mission	Approximate Power Level	Possible Fuel Source	Approx. No. of Missions in This Period	Most Suitable Disposal Technique
1.	Syn. Com.	65-70	orbital	25-50 w(e)	Pu-238		Burnup or microspheres
2.	Comsat	65-70	orbital	50-80 w(e)	Pu-238, Sr-90	23 (4/yr)	Burnup or microspheres
3.	Relay	65-70	orbital	50	Pu-238, Pm-147	3	Burnup or microspheres
4.	Transit	65-69	orbital	25	Pu-238	3 - 4	Burnup or microspheres
5.	Advanced Air Force	67-70	orbital	40 w(e)	Pu-238, Pm-147	6-7 (2/yr)	Burnup or microspheres
6.	Nimbus	65-70	orbital	50-250 w(e)	Pu-238	3 - 4	Orbital storage with microsphere backup
7.	OGO	65-70	orbital	300-500	Pm-147	1 - 2	Orbital storage burnup backup
8.	Explorer	65-69	planetary	15-50 w(e)	Pu-238, Pm-147		Orbital storage or escape, burnup backup or microspheres
9.	Mariner	65-70	planetary	150-250	Pu-238, Po-210, Pm-147		Orbital storage or escape, burnup backup or microspheres, intact reentry with Po-210
10.	Extended Pioneer	65-70	planetary	500 w(e)	Pu-238	1 - 2	Solar orbit, burnup backup
11.	Advanced Pioneer	65-70	planetary	1 Kw(e)	Pu-238	1 - 2	Solar orbit, intact reentry backup

Table II (Cont'd)

Mission	Approx. Time Period	Type of Mission	Approximate Power Level	Possible Fuel Source	Approx. No. of Missions in This Period	Most Suitable Disposal Technique
12. RAO	67-70	orbital	250-500 w(e)	Pu-238	1	Orbital storage, burnup backup
13. MORL	68-70	orbital	4 - 8 Kw(e)	Pu-238, Po-210	1 - 2	Intact reentry
14. AOSO	69-70	orbital	300-500 w(e)	Po-210, Pu-218, Pm-147	1 - 2	Orbital storage, burnup backup
15. Extended Apollo	70-71	cislunar	1.5 - 6 Kw(e)	Po-210, Pu-238	1 - 2	Intact reentry
16. FM Broadcast	70-71	orbital	1 Kw(e)	Pu-238, Po-210	1	Orbital storage, intact reentry backup

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However, Po-210's properties make it an ideal fuel for high altitude burnup and dispersion. Therefore, the key to application of burnup rests with generator and fuel capsule design. If the generator design is such that the integrity of the fuel capsule can be assured under all ground and ascent abort consequences without greatly compromising the burnup of the generator components and release of the fuel under reentry conditions, a high altitude burnup philosophy is justified. If this cannot be done then intact reentry, with all its weight penalties, must be accepted.

For those missions in the 1965-70 time period which are manned such as MORL, Apollo, and Extended Apollo, return of the heat source with the crew, controlled intact reentry of the heat source alone, or orbital storage are all practical and feasible techniques. The selection of one particular technique over any other is difficult at this stage of planning because the interaction of the safety technique chosen and the basic vehicle and mission profile cannot yet be fully defined. All systems could achieve orbital storage. All Apollo and MORL systems could reenter the heat source with the crew or a resupply vehicle, however, MORL looks most attractive for unmanned controlled intact reentry.

For those missions in the 1965-70 time period which are planetary, orbital storage and/or escape with a burnup backup has been selected as the preferred safety philosophy because of the inherent escape capability of the vehicle. These missions are long term ($\sim$ 1 year) for the most part and will be launched with present day boosters. Hence not much latitude will be available in the payload weights for intact reentry systems. Therefore, if flight aborts do occur the inherent capability for escape which is built into the vehicle should be used to place the heat source in a deep space non-earth collision trajectory or to increase the lifetime (i.e., altitude) of a short lived abort orbit. If the nature of the abort is such that earth reentry is assured then high altitude burnup or the microsphere technique should suffice safely for the quantity of fuel characteristic of these missions. Both the primary and backup philosophies should be chosen to minimize the weight and reliability penalties on the launch configuration.

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II. DEFINITION OF FUTURE MISSION TRENDS AND THEIR REQUIREMENTS FOR NUCLEAR POWER

A. Missions

In order to fully evaluate the potential and limits of application of th[redacted] various aerospace nuclear safety techniques analyzed in this report, the missions and nuclear power supplies to which they may apply must be defi[redacted]. By considering specific mission objectives, duration, operational character, vehicle configuration and other pertinent factors one can isolate in a preliminary fashion the most appropriate power system for the mission. For those missions where nuclear powered systems would be most desirable nuclear safety criteria are postulated and system requirements investigated to determine the degree of compliance with these criteria that can be expected.

The scope of this effort is quite involved and requires both tangible technical factors amenable to analysis and intangible factors based on independent judgments with respect to mission abort situations and since many space missions are not clearly defined and exist only in a conceptual stage, specific consideration of each mission was not possible. Therefore, the approach was to identify as many missions as possible and to delineate their characteristics as thoroughly as could be done, within the limitations of this study. The results are presented in Table III. The majority of the infomation to complete Table III was obtained from References 1, 2 and 3. For purposes of analysis and for the development of an aerospace nuclear safety approach for those missions utilizing nuclear devices, all missions were grouped into the four distinct categories listed below.

(1) Earth orbital.
(2) Lunar.
(3) Planetary.
(4) Solar and deep space

Within these categories the missions are chronologically listed in three separate time periods (1965-70, 1970-75, and 1975-80), and identified as being either a manned or an unmanned mission. This permits a logical evaluation of the missions in terms of priority the feasibility of meeting space environment, time span, trajectories, vehicle configurations and power requirements. The implications of the mission vehicle requirements upon the selection of a power system and the development of safety guidelines can then be determined.

It should be noted that our investigations into the type of missions which DOD will run in this time period revealed that surveillance, interception, communications and general R and D experimentation for space weapons systems were predominant. All these missions could be grouped into the earth orbital and lunar categories. Therefore, although primarily NASA missions are presented in Table III, they are typical of the DOD missions which are anticipated (in the categories mentioned above) and from the

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Table III

Planned Aerospace Missions

PART I. Earth Orbital Missions
Part I-A. Manned

Mission No.	Mission - Objectives	Time Period, Number Req'd	Space Environment	Manned or Unmanned	Mission Profile	Electrical Power Required	Mission Duration	Trajectories (Launch and Final)	Launch Vehicle		
1	Extended Apollo a. Remote sensing of earth's surface	Late 1960's, Early 1970's	Earth orbit 200 n mi., 50 deg	Manned	Earth mapping in visible and near-visible spectrum; artificial-gravity qualification; medical	1.5 - 6 Kw(e)	up to 1 year	AMR - parking orbit, non-escape lunar orbit, 19k to k n mi., synchronous launched due east	Saturn 1B, V, SIVB		
	b. Biomedical/behavioral and technology	Same as above	Earth orbit 200 n mi., 28.5 deg	Manned	Biomedical/behavioral; lunar-orbit survey system qualification	Same as above	30 days	Same as above	Same as above		
	c. Bioscience/physical science laboratory	Same as above	Earth orbit 200 n mi., 28.5 deg	Manned	Biomedical/behavioral-prolonged weightlessness effects; zero-gravity behavior of solids, liquids and gases	Same as above	45 days	Same as above	Same as above		
	d. Space operations/ bioscience laboratory	Same as above	Earth orbit 200 n mi., 28.5 deg	Manned	Rendezvous with (c); extravehicular cargo and personnel transfer, rescue operations, spacesuit evaluation; spaceflight effects on biological systems	Same as above	45 days	Same as above	Same as above		
	e. Remote sensing of earth's surface and atmosphere	Same as above	Earth orbit 200 n mi., 83 deg (retrograde)	Manned	Sun-synchronous; multispectral sensors; solar-cell array, erection and orientation	Same as above	45 days	Same as above	Same as above		
	f. Astronomy/bioscience laboratory	Same as above	Earth orbit 19,350 n mi.,0 deg	Manned	Synchronous orbit; X-ray, visual and IR spectral and photometric data; deployable satellites; extendable radio-astronomy antenna	Same as above	45 days	Same as above	Same as above		
	g. Artificial-gravity laboratory	Same as above	Earth orbit 200 n mi., 28.5 deg	Manned	Evaluate human performance in rotating environment; cable-connected, CSM and LEM-Lab are rotated	Same as above	45 days	Same as above	Same as above		
	h. Remote sensing of earth's surface and atmosphere; Echo observation	Same as above	Earth orbit 200 n mi., 83 deg (retrograde)	Manned	Sun-synchronous; multispectral sensors; solar-cell array, erection and orientation; CSM-inspection of Echo satellite	Same as above	45 days	Same as above	Same as above		

Table III (Cont'd)

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Part 1-A. Manned (Cont'd)

Mission No.	Mission - Objectives	Time Period, Number Req'd	Space Environment	Manned or Unmanned	Mission Profile	Electrical Power Required	Mission Duration	Trajectories (Launch and Final)	Launch Vehicle		
1 (contd)	i. Space physics and subsystem development	Late 1960's, Early 1970's	Earth orbit 19,350 n mi., 0 deg	Manned	Space environment near earth, e.g., aurora, magnetic fields, micrometeoroid fluxes; life support, communications, satellite launch	1.5 to 6 Kw(e)	45 days	AMR-parking orbit, non-escape lunar orbit, 19k to 23 k n mi., synchronous launched due east	Saturn 1B, V, SIVB		
	j. Astronomy/bioscience laboratory II	Same as above	Earth orbit 200 n mi., 28.5 deg	Manned	Astronomy, optical and radio; micrometeorites; gamma rays, biomedical	Same as above	45 days	Same as above	Same as above		
	k. Astronomy/logistics and bioscience laboratory	Same as above	Earth orbit 200 n mi., 28.5 deg	Manned	Rendezvous with (j); cargo and personnel transfer; biomedical effects of flight duration beyond 45 days; continue astronomy as on (j)	Same as above	45 days	Same as above	Same as above		
	l. Astronomy/logistics II	Same as above	Earth orbit 200 n mi., 28.5 deg	Manned	Rendezvous with (k); extend mission to total of 135 days; cabin atmosphere, propellant handling	Same as above	45 days	Same as above	Same as above		
	MORL	Same as above	Earth orbit	Manned	Conduct experiment, zero-gravity, geophysical, refueling, resupply	4 - 8 Kw(e)	1 year 90 day resupply	AMR	Same as above		
	Space Station	Early 1970's	Earth orbit (polar and equatorial)	Manned	Similar to MORL, except larger	7 - 10 Kw(e)	2-5 yrs 90 day resupply	AMR PMR	Saturn V		
	Gemini (Operational)	1964-1967	Earth orbit	Manned	Space effects upon man and preliminary space data for Apollo mission (rendezvous and docking)	1.5 - 2 Kw(e)	<14 days	AMR	Titan II		
	FM Broadcast Station	1970-71 2 1/yr	Earth orbit plus synchronous orbit	Unmanned	Communications	1 Kw(e)	5 years	AMR	Delta, Scout Saturn 1		
	TV Broadcast Station	1975-80	Earth orbit plus synchronous orbit	Unmanned	Communications	30 - 60 Kw(e)	5 years	AMR	Same as above		
7	LORL	Post '75	Earth orbit	Manned	Extend space missions, all types of experiments	15 - 40 Kw(e)	1-5 yrs Kw(e)	AMR	Saturn V		
8	Manned Orbiting Telescope	1979-80	Earth orbit (3000 n. mi)	Manned	To study noise sources from solar system and stars	1 - 1.5 Kw(e)	1 year	AMR	Atlas/Centaur or others		

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Table III (Cont'd)

Part 1-B. Unmanned

Mission No.	Mission - Objectives	Time Period, Number Req'd	Space Environment	Manned or Unmanned	Mission Profile	Electrical Power Required	Mission Duration	Trajectories (Launch and Final)	Launch Vehicle		
9	Syncom (operational)	1963 or later	Orbital (Synchronous)	Unmanned	High altitude communication satellite	25 - 50 w (e)	1-3 yrs	AMR	Thor-Delta		
10	Explorer (operational) a. Interplanetary b. Ionosphere c. Pegasus d. Direct Measurements e. Biosatellites (Bios) f. Lunar Anchored Interplanetary g. Upper Atmosphere Life Probe h. University Explorer (repeated in other space regimes)	1965-69	(orbital/lunar/interplanetary), deep earth orbital (150,000 n mi.) apogee earth orbital	Unmanned	a. Study various space environments, radiation and magnetic fluxes (wt. 135 lbs) b. Measure electron distribution and density; also study cosmic noise (wt. 110 lbs) c. Measure effects of meteoroid impact (wt. 3400 lbs) d. Study ion and electron density (wt. 157 lbs) e. Biological studies (wt. 200-250 lbs) f. Interplanetary magnetic fields (wt. 181 lbs) g. Determine the existance of living organisms in upper atmosphere h. Various objectives	15 - 50 w (e)	3-6 mo	a. AMR b. AMR-PMR c. AMR d. AMR e. AMR f. AMR g. --- h. ---	Thor-Delta Scout Atlas/Agena		
11	Telstar (operational)	1962 or later	Earth orbit	Unmanned	Low altitude communication satellite	60 w(e)	1 year	AMR	Delta		
12	Comsat	1965-70 23 4/yr	Earth orbit, also synchronous orbit	Unmanned	Medium altitude communication satellite (both military and commercial)	50 - 80 w(e)	5 years	PMR-AMR	Titan, Atlas/Agena		
13	Tiros (operational)	1960-70 58	Earth orbit	Unmanned	Exp't meteorological satellite	25 - 50 w(e)	3 mo	PMR-AMR	Thor-Delta		
14	Relay (operational)	1962-70 6 1/yr	Earth orbit (low)	Unmanned	Active-repeater-type communication satellite	50 w(e)	1 year	PMR-AMR	Same as above		
15	Transit (operational)	1964-69 6 2/yr	Polar orbit	Unmanned	Exp't navigational satellite	25 w(e)	2 years	PMR	Thor/Able/Star		
16	Advanced Air Force	1967-75 2/yr		Unmanned		40 w(e)	1 year	PMR	Thor/Able/Star or other similar type		

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Table III (Cont'd)

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Part I-B. Unmanned (Cont'd)

Mission No.	Mission - Objectives	Time Period Number Req'd	Space Environment	Manned or Unmanned	Mission Profile	Electrical Power Required	Mission Duration	Trajectories (Launch and Final)	Launch Vehicle		
17	Nimbus	1964-71 1971-80	Polar Earth Orbit Polar Earth Orbit	Unmanned Unmanned	Meteorological Satellite (Second Generation) Same as above	250 w(e)-total with 50w(e) nuclear	2 years 2 years	PMR PMR	Thor/Agena B Atlas/Agena		
18	Radioastronomy Observations	1967-80	Earth Orbit (3000 n. m.	Unmanned	To study noise sources from solar system and stars	0. 25-0. 50 Kw(e)	5 years	AMR	Thor- Delta or others		
19	Advanced Orbiting Solar Observatory	1969-80 22	Earth Orbital	Unmanned	Continued solar observatory studies solar phenomena (better resolution than earlier OSO)	0. 30-0. 50 Kw(e)	2 year	PMR			
20	Advanced Navy Mission	1970-72	- - - - - - - - - -	Unmanned	- - - - - - - - - - - -	15-50 w(e)	1 year	AMR-PMR	Thor/Able/ Star		
21	Operational Weather Satellite (Post-Tiros-System	Early 1970's	Earth Orbital	Unmanned	Combined Tiros, Nimbus, Synchronous Missions	* 0. 5 Kw(e)	5 years	AMR-PMR	Atlas/Agena or (similar type)		
22	Advanced Meterological Satellite	1970-74	Earth Orbital	Unmanned	Basic weather surveillance	50 - 100 w(e)	2 year	AMR-PMR	Atlas/Agena or similar type launch vehicle		
[illegible]	Orbiting Astronomical Observatory (OAO)	through 1974 5	Earth Orbital	Unmanned	Satellite Optical Observations, Mapping Mainly in uv Spectrum, Secondary X-ray Experiments (3600 lb Satellite)	0. 5-0. 75 Kw (e)	6 mo - 1 year	AMR-PMR	Atlas/Agena or similar launch vehicle		
24	Synchronous Meterological Satellite	1971-78	Synchronous Earth Orbit	Unmanned	Provide continuous monitoring of short-lived storms and cloud cover	0. 5 Kw(e)	1 year	AMR	Atlas/Agena others		
25	Orbiting Geophysical Observatory	through 78	Earth Orbital	Unmanned	Satellite equipped to measure geophysical data	0. 3 - 0. 5 Kw(e)	6 mo - 1 year	AMR-PMR	Atlas/Agena B or similar type		
26	Solar Geophysical Monitor	through 75	Earth Orbital	Unmanned	Solar activity measurements	0. 15-0. 20 Kw(e)	2 years				
27	Satellite Inspector	Mid 1970's	Earth Orbital	Unmanned	Maneuverable unit in space	1. 0 Kw(e)	5 years	AMR-PMR			
28	Advanced Orbiting Astronomical Observatory	1975-80	Earth Orbital	Unmanned	Continue the experiments of the OAO mission (No. 23)	0. 75 - 1 Kw(e)	2 years	AMR-PMR	Atlas/Centaur or similar type		

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Table III (Cont'd)

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Part II-Lunar

Mission No.	Mission - Objectives	Time Period Number Req'd	Space Environment	Manned or Unmanned	Mission Profile	Electrical Power Required	Mission Duration	Trajectories (Launch and Final)	Launch Vehicle		
29 (Same as No. 10)	Explorer a. Interplanetary b. Ionosphere c. Pegasus d. Direct Measurements e. Biosatellites (Bios) f. Lunar Anchored Interplanetary g. Upper Atmosphere Life Probe h. University Explorer (repeated in other space regimes)	1965-69	(orbital/lunar/interplanetary), deep earth orbital (150,000 n mi.) apogee earth orbital	Unmanned	a. Study various space environments, radiation and magnetic fluxes (wt. 135 lbs) b. Measure electron distribution and density; also study cosmic noise (wt. 110 lbs) c. Measure effects of meteoroid impact (wt. 3400 lbs) d. Study ion and electron density (wt. 157 lbs) e. Biological studies (wt. 200-250 lbs) f. Interplanetary magnetic fields (wt. 181 lbs) g. Determine the existance of living organisms in upper atmosphere h. others	15 - 50 w(e)	3-6 mo	a. AMR b. AMR-PMR c. AMR d. AMR e. AMR f. --- g. --- h. ---	Delta Scout Saturn 1		
30	Surveyor Orbiter and Lander	1966-67 6 3/yr	Lunar	Unmanned	Photograph lunar topography for Apollo-LEM probe; Lander - soft land on the moon to analyze surface	0.025 - 0.125 Kw	0.5 - 1 yr	AMR	Atlas/Agena, Atlas/Centaur		
31	Apollo - Land two astronauts and scientific equipment on the moon's surface	Late 1960's Early 1970's	Lunar	Manned	Land man on the moon and explore its surface	1.5 Kw(e)	1 mo	AMR	Saturn 1B, V, SIVB		
32	LEM	Early 1970's	Lunar	Manned	Land 250 lb & 10 ft^3, Return 80 lb & 2 ft^3	0.1 - 0.5 Kw(e)	7 days	Equatorial Launch	Saturn V		
33	Roving Vehicles	Early 1970's	Lunar	Manned	Lunar operations operate up to 2 years	3 - 15 Kw(e)	14 - 730 days	Equatorial Launch	Saturn V		
34	a. Lunar Exploration Shelter (28 days)	Thru 1980 6-9	Lunar	Manned	Lunar surface investigation	2 Kw(e)	28 days	AMR	Saturn V		
	b. Lunar Exploration Shelter (90 days)	2-4	Lunar	Manned	Lunar surface investigation	4 Kw(e)	90 days	AMR			
	c. Lunar Exploration Shelter (2 years)	4-6	Lunar	Manned	Lunar surface investigation	5 Kw(e)	2 years	AMR			

[redacted]

[redacted]

[redacted]

Table III (Cont'd)

Part III - Planetary

Mission No.	Mission - Objectives	Time Period, Number Req'd	Space Environment	Manned or Unmanned	Mission Profile	Electrical Power Required	Mission Duration	Trajectories (Launch and Final)	Launch Vehicle		
35	Scientific probes - to be deployed to planetary orbits or surfaces for data acquisition		Interplanetary	Unmanned	To determine gravitational fields, high particle fluxes, micrometeoroid fields	0.05-0.50 Kw(e)	2 years	AMR-PMR	Atlas/ Agena or similar type launch vehicle		
36 (Same as No. 10)	Explorer a. Interplanetary b. Ionosphere c. Pegasus d. Direct Measurements e. Biosatellites (Bios) f. Lunar Anchored Interplanetary g. Upper Atmosphere Life Probe h. University Explorer (repeated in other space regimes)	1965-69	(orbital/lunar/interplanetary), deep earth orbital (150,000 n mi.) apogee earth orbital	Unmanned	a. Study various space environments, radiation and magnetic fluxes (wt. 135 lbs) b. Measure electron distribution and density; also study cosmic noise (wt. 110 lbs) c. Measure effects of meteoroid impact (wt. 3400 lbs) d. Study ion and electron density (wt. 157 lbs) e. Biological studies (wt. 200-250 lbs) f. Interplanetary magnetic fields (wt. 181 lbs) g. Determine the existance of living organisms in upper atmosphere h. others	15 - 50 w(e)	3-6 mo	a. AMR b. AMR-PMR c. AMR d. AMR e. AMR f. --- g. --- h. ---	Delta Scout Saturn 1		
37	Asteroid-Jupiter-Outer Solar System Probes	1967 thru 1980's	Asteroid-Jupiter Environmental Study	Unmanned	Study asteroid belt between Mars and Jupiter, probe Jupiter environment and other planets - Mariner or Pioneer type crafts	50 w(e)	2 years*	AMR	Saturn class		
38	Jupiter Flyby	1972 - 1977	Interplanetary	Unmanned	TV pictures, radiation environment atmospheric study	1 - 1.2 Kw(e)	2 years*	AMR	Atlas/ Centaur or others		
39	Mercury Flyby	1973 - 1978	Interplanetary	Unmanned	TV Pictures, radiation environment, atmospheric study	0.4 Kw(e)	2 years*	AMR	Atlas/ Centaur or others		

Table III (Cont'd)

[redacted]

Part III - Planetary (Cont'd)

[redacted]

Mission No.	Mission - Objectives	Time Period, Number Req'd	Space Environment	Manned or Unmanned	Mission Profile	Electrical Power Required	Mission Duration	Trajectories (Launch and Final)	Launch Vehicle		
40	Mariner-Mars Flyby	Late 1960's through Early 1970's	Interplanetary	Unmanned	Return enroute data on particles and fields plus perform Mars experiment including "TV" (1000 lb)	0.15-0.25 Kw(e)	6 mo	AMR	Atlas/ Centaur		
41	San Marco	Late 1960's Early 1970's	Interplanetary	Unmanned	Measure solar pressures and other experiments	25 w(e)	1 yr*	------	Scout		
42	Interplanetary Module a) Training, checkout and flyby	1970's 1980's	Interplanetary	Manned	Same basic design as Apollo, conduct interplanetary experiments	10 - 20 Kw(e)	90-120 days 350-600 days	AMR	Saturn V		
	b) Lander type mission	1970's 1980's	Interplanetary	Manned	Surface and environment exploration	Same as above	400-1000 days	AMR	Saturn V		
43	Voyager (Mars or Venus Planet Study)	1971-1977	Interplanetary	Unmanned	Spacecraft 4000 to 8000 pounds	0.6 Kw(e)	2 years*	AMR	Atlas/ Centaur or others		
44	ABL (Automated Biological Lab) for detection and classification of Martian life	early 1970's	Mars	Unmanned	Martian life - Voyager type spacecraft - to be landed on Mars for detecting and classifying life	1 Kw(e) *	2 years*	AMR	Saturn class		
45	Venus/Mars Recon. Mission (Redefined)	Late 1980's	Interplanetary - 400 day flyby	Unmanned	Voyager type spacecraft	1 Kw(e)*	2 years*	AMR	Saturn class		
46	Mars Landing (Post Apollo)	Late 1980's	Martian environment	Unmanned	Voyager type spacecraft	1 Kw(e)*	2 years*	AMR	Saturn class		
47	Manned Mars Mission	Post 1980's	Interplanetary	Manned	Manned exploration of Mars environment - extend space studies	5 - 30 Kw(e)	1 - 2 years	AMR	Saturn 5		

Table III (Cont'd)

[redacted]

Part IV - Solar and Deep Space

Mission No.	Mission - Objectives	Time Period, Number Req'd	Space Environment	Manned or Unmanned	Mission Profile	Electrical Power Required	Mission Duration	Trajectories (Launch and Final)	Launch Vehicle		
[redacted]	Extended Pioneer	Late 1960's Early 1970's	Interplanetary	Unmanned	Monitor solar and interplanetary phenomena form orbits of 0. 4 - 0. 6 Au of Sun (130 - 140 lb)	0. 5 Kw(e)*	5 years*	AMR	Atlas/ Centaur		
[redacted]	Advanced Pioneer (solar probe)orbits 0. 02 - 0. 3 Au of Sun, also 0. 05 - 0. 1 Au	Late 1960's Early 1970's	Interplanetary	Unmanned	Orbit 0. 2 - 0. 3 Au from the sun - solar phenomena study (neutron fluxes, etc.) possible to 0. 05 - 0. 1 Au orbits	1 Kw(e)*	5 years *	AMR	Saturn class		
[redacted]	Out of Ecliptic Probe (25°-30° out of ecliptic orbit) study interplanetary particles cosmic phenomena	Early 1970's	Interplanetary Space	Unmanned	To be launched outside of the ecliptic phenomena (spacecraft will be Extended or Advanced Pioneer type)	0. 5 Kw(e)*	2 years *	AMR	Saturn V/ Centaur or Atlas/ Centaur		
[redacted]	Interstellar Probes	1967 thru 1980's	Extra Solar System	Unmanned	Investigate nearest stars, escape trajectory	0. 5 Kw(e)*	2 years *	AMR	Saturn class		
52	Comet Intercept	1970's	Interplanetary	Unmanned	Earth orbit (Mariner type craft)	0. 03 Kw(e)	0. 5 - 1 year	AMR PMR	Atlas/ Agena, Atlas/ Centaur		

* Estimated

[redacted]

[redacted]

standpoint of aerospace and nuclear safety will require the same techniques and countermeasures for launch approval.

B. Mission Induced Problems Affecting Aerospace Nuclear Safety

Having identified future aerospace missions involving different space regimes (earth orbital, lunar, planetary, and solar and deep space), a summary of the conditions that could be imposed upon a nuclear system if aborts should occur during any phase of the mission is presented below. The abort conditions strongly influence the selection and design of a power system in that safety must be assured under all abort conditions. Therefore, possible mission induced aerospace safety problems which must be considered in the different space regimes are:

1. Problems Common to All Space Regimes

a. Type of spacecraft - geometry, materials, structure, etc.

b. Location of the power system within the spacecraft.

c. System design to satisfy both reentry and launch pad abort requirements.

d. Radiobiological shielding requirements to meet permissible radiation design criteria for manned space exploration.

e. Radiation shielding of sensitive on board components.

f. Exposure of the system to meteorite and high vacuum effects.

g. Possible collision between spacecrafts during rendezvous and docking operations.

h. Tracking, guidance and control operations.

i. Spacecraft environmental control.

2. Earth Orbital

a. Orbital injection errors - decreasing orbital lifetime, higher reentry heat rate.

b. Post-injection aborts during space maneuvers which might cause premature reentry of the system and/or poor attitudes for the reentry.

c. Post-injection aborts yielding a breach in fuel containment via a coolant loss, etc., combined with early reentry for large fuel inventories designed for intact reentry.

[redacted]

3. Lunar

   a. Translunar injection error - reentry at super orbital velocities, direct intersect lunar trajectory, yielding hypervelocity impact.

   b. Translunar midcourse guidance malfunction.

   c. Lunar approach guidance malfunction (retro from translunar trajectory).

   d. Lunar descent (retro from lunar orbit to lunar surface).

   e. Lunar ascent and rendezvous malfunctions.

   f. Transearth injection, guidance, and subsequent reentry and touchdown (impact the earth's surface).

   g. Long term exposure of any nuclear lunar surface systems to meteorite effects, high vacuum and temperature cycles (± 250°F, day and night conditions). This point would have greatest importance only when the nuclear system were to be returned to earth.

4. Planetary, Solar and Deep Space

   a. Longer flight time (system component failure yielding planetary impact).

   b. More susceptible to environmental effects, i.e., exposure to both high vacuum and temperatures (on missions closer to the sun) and meteorites.

   c. Increased number of guidance maneuvers as well as increased communication and tracking difficulties.

   d. Planetary capture and subsequent entry into unknown types of atmospheres.

   e. Possible earth reentry at superorbital velocities several times that of earth orbital.

   f. Possible collision with interplanetary debris, such as the asteroids. This would not yield a safety problem to earth but is in the same class of accidents as planetary impact. Contamination of natural space environment before it can be studied.

In summary, the problems generated by the mission on aerospace safety all seem to reflect themselves in the reliability of basic spacecraft components and operational techniques. Two basic concerns become immediately evident:

(1) The interaction of the nuclear payload with the vehicle must never compromise the basic component reliability vehicle.

[redacted]

(2) If an abort occurs, the nuclear device must have the inherent capability to survive (or be destroyed by) the resulting environment induced by the mission profile, without creating the hazard to earth populations.

C. Missions and Applicable Power System Identification

In order to place the evaluation of basic aerospace safety philosophies and techniques, which is the major aim of this study, into proper perspective relative to the overall space program shown in Table III, those missions which may be flown with nuclear power plants must be identified and the character of the possible nuclear fuels and plants defined. Mission requirements are complex and vary in power requirements from 15 watts(e) for an Explorer-type mission (1965) to 60 kilowatts(e) for a Television Broadcast System (1975-1980) and in duration from 14 days for an Apollo mission to five years for a communications satellite. The characteristics of three primary energy sources - nuclear, solar, and chemical - and numerous methods for energy conversion were studied. Each of these methods or combinations of methods for producing electrical power in space may be shown to have specific advantages for use in certain space missions or experiments. For instance, a battery powered system for a Jupiter Flyby might be desirable based on the type of experiments to be performed, but would unquestionably be too heavy as the system has both long life and high power requirements. This is illustrative of the fact that there are many parameters which must be carefully considered in the selection or identification of a space power system for a particular mission. Because of the large number of variables associated with a given mission involving both tangible technical analysis and intangible value judgments, it is not possible to present any simplified formula for establishing the optimum auxiliary power system for a given mission. For example, the selection must depend upon the following types of considerations:

(1) Mission duration.
(2) Manned or unmanned.
(3) Mission trajectories.
(4) Mission environment (earth orbital, lunar, planetary, and solar and deep space), temperature, radiation, meteorites, etc.
(5) Type of experiments to be performed.
(6) Weight and volume limitations.
(7) Power requirements.
(8) Power cycling characteristic (peak to average ratio).
(9) Type of heat source.
(10) Reliability and maintainability requirements (converter)
(11) Nuclear safety.

[redacted]

In addition, certain characteristics such as the degree of system development at the planned mission initiation time and spacecraft power system integration must be considered.

In order to define those systems which can be flown non-nuclear, the characteristics of non-nuclear power plants are shown in Table IV, compiled from Reference 4 and updated. Those missions which are most amenable to nuclear power were then selected and their general characteristics are shown in Table V. The following system features were determined:

(1) Nuclear or non-nuclear energy source.
(2) Radioisotope or reactor system.
(3) Type of energy converters.
(4) Heat source characteristics.
    (a) Applicable radioisotopes.
    (b) Quantity required.

Recognizing that mission experiments, instrumentation, and schedules in the late 1960's and beyond are somewhat speculative, since each space mission if possible must be based upon the cumulative knowledge of space gained up to the time of mission preliminary design; a primary and a secondary type of energy source was selected. At present, the selection of the primary energy source as the method to be developed for the mission would provide the most flexibility and, in general, be consistent with mission requirements. If, after thorough consideration of mission and power system requirements, a clear decision could be made for a specific type system it was identified as either nuclear or non-nuclear.

For purposes of this study, radioisotope systems were considered feasible up to 10 Kw(e) and above this level reactor systems were considered more desirable. Of course, the exact power level at which reactors become more logical nuclear heat sources the isotopes may vary from mission to mission, and such levels have not been accurately established as yet.

Since nuclear safety is a direct function of the quantity of isotope launched, the maximum inventory for each mission was based upon a thermoelectric system at an efficiency of 5 - 7 percent which is a projected efficiency based on anticipated technology improvements. The inventory for dynamic conversion was also calculated where applicable using an efficiency of 15 - 20 percent. Although these efficiencies may be considered optimistic based on present SNAP-8 technology, it is anticipated that materials and design breakthroughs now in evidence (Reference 5) will justify these values in the 1970-80 time period. Also considered in the inventory requirements was a 30 day hold time between fueling and launch.

Based on mission and system requirements the following radioisotopes were considered: plutonium-238, promethium-147, polonium-210 and strontium-90. Plutonium, promethium and polonium were selected on the bases of their low characteristic radiation levels both neutron and gamma,

Table. IV

Characteristics of Non-Nuclear Auxiliary Power Systems*

[redacted]

*Data Compiled from Reference II-4 and Updated Where Applicable

Item	Specific Weight lb/Kwe	Approx. Specific Cost $ Kw-Hr	Geometric Limitations	Operating Temp. °F	Duration (t) Hours	Power Range of Best Application	Reliability	Remarks
Solar Cells p- on n- and n- on p-	$3 \times 10^2 - 4 \times 10^3$	$3 \times 10^2 - 1 \times 10^4$ unoriented $2 \times 10^1 - 2 \times 10^3$ oriented	100-250 ft^2/Kw	100-150	Years	0-2 Kw	Excellent when out of radiation fields and no temp. cycling. Very sensitive to over temperatures.	1. N - on - P - cells decrease radiation degradation up to 25% with only 1% loss in efficiency. 6-20 mil covers still necessary for severe radiation environments. 2. Most failures in operational systems due to electrical system storage and converter failures. 3. Costs may drop by a factor of 3 over next 5 years.
Solar Thermo-electric	100-150 collection heat sink T. E. generator concept. 200-300 flat plate solar panel concept including structure and erection neck.	*$9.3 \times 10^1 - 1.86 \times 10^2$	0.1 - 0.2 $\frac{ft^2}{kw}$ 300-800 $\frac{ft^2}{kw}$	900-1300	Years	Up to 10 Kw	Good in non-thermal cycling applications.	1. 4% conversion efficiency for PbTe-GeSi to compete with solar cells arrays. 2. Radiation insensitive. 3. LiH heat sink generator concept now under development at Westinghouse.
Solar Thermionic	$2 \times 10^2 - 7 \times 10^2$	$4 \times 10^1 - 9 \times 10^2$	Collector sizes limiting 50' max. dia. and errectable mirror	3000-10000	Potentially Years	Up to 10 Kw	No conclusive test data on solar systems.	1. System very sensitive to mirror attitude Limits of ± 3 min. of arc must be maintained. 20% power drop for 30 min. of arc aiming error. 2. Highest test times are solar converters to date 100-200 hrs., with 50-100 thermal cycles. 3. Highest systems eff. to date 2-5%. 4. Advantageous for operations in radiation or high temperature environment.
Solar Dynamic	$7 \times 10^1 - 3 \times 10^2$	$7 \times 10^1 - 4.5 \times 10^2$	Collector sizes limiting	1250 Hg 1750 Rb	Years	Up to 50 Kw	Same as Reactor Dynamic Systems except for mirror erection mechanisms.	1. Orientation requirements only ± 3/4° not severe compared to Mariner Venus probe. 2. Subject to same development problems as reactor Rankine Cycles. 3. Ideal as low cost space test bed for reactor dynamic systems. 4. Not as sensitive to mirror distortion as thermionic systems.
Batteries	$5 \times 10^4 - 5 \times 10^5$	$3 \times 10^3 - 5 \times 10^3$	0.003 Kw-hr/in^3	0 - 80	60 hrs Based on a 1500 lb, 2 Kw power payload req.	0 - 100 watts	9,000 recharging cycles at 75°F 100 cycles at -30°F 1800 cycles at 120°F	1. Future battery design objectives 0.20 Kw-hr/lb.
Fuel Cells	$8 \times 10^2 - 2 \times 10^4$	$6 \times 10^2 - 1.3 \times 10^3$	.015 Kw-hr/in^3 with tanks	140-180 Gemini H_2-O_2 Tungsten cell 400-500 Apollo H_2-O_2 KOH-cell	500 hrs present capability R & D for 2500 hrs now in progress to meet Apollo Extension Requirements	0.1 - 5 Kwe	Gemini cell stacks have run over 10^3 hrs. Single cells over 5000 hrs in lab many Apollo units in test. 15 re-starts and 400 hr runs have been achieved to meet mission specs.	1. Water recovery prime importance. 2. Regeneration of cell reactants possible in future increasing duration. 3. Resupply concepts being considered for space station use. 4. Fuel consumption decreases markedly on standby power. Fuel cell batteries may be pulsed to high powers for short terms.
Chemical Engines	$5 \times 10^3 - 5 \times 10^4$	$1.5 \times 10^1 - 2.5 \times 10^2$	0.07 Kw-hr/in^3	5700-6200	300 hrs	0.1-20 Kwe 100 Kwe on short-term emergency power	Good short-term applications (1500 hours). [redacted]	1. High fuel consumption. 2. Most attractive for peak power applications when planetary orbits are reached.

*Based on a 6 month mission with estimated costs from "Thermoelectric Power Systems" by N. F. Schuh, Astronautics and Aerospace Engineering, May, 1963.

[redacted]

[redacted]

PART I. Earth Orbital Missions

Part I-A. Manned

[redacted]

[redacted]

Table V

Mission and Probable Power System Identification

Mission No.	Mission	Power System (Nuclear/Non-Nuclear)		Power Level Kw(e)	Radioisotope or Reactor	Type of Converter	Heat Source Character			Projected Period for Mission
		Selection Primary Secondary	Discussion				Type of Isotope or Fuel Element	Initial Quantity Kw(t)*		
								T/E-T/I	Dyn	
1	Extended Apollo	1. Non-Nuclear 2. Nuclear	For early manned flights of short duration fuel cells are attractive (no orientation required or large external surfaces, and produce water as a by-product) but long life and reliability are problem areas. However, nuclear systems may be integrated into later missions where longer life and higher power systems are required studies have been completed by NAA and NASA which shows substantial gains by use of R.I.	1.5 to 6.0	Radioisotope	Thermoelectric (T/E) or Dynamic (Brayton, Rankine or Stirling)	Po^{210} (45d) Pu^{238} (1 yr) or Cm^{244}	124 *including 30 day hold time 86.4	42.8 30.2	late 60's early 70's
2	MORL	Nuclear	The one year mission will pose significant problems for non-nuclear power systems. Solar panels for the power level would provide considerable drag and a large area for meteorite impact. Nuclear systems definitely have the potential capabilities to satisfy mission requirements.	4.0 to 8.0	Radioisotope (possible reactor)	T/E or Dynamic	Pu^{238} (1 yr) Po^{210} (90 d resupply)	115.5 210	40.2 73.5	late 60's early 70's
3	Space Station	Nuclear	Considering mission duration and power level, a nuclear would be more attractive. Non-nuclear would not satisfy the long life requirements. Solar energy collection would provide a difficult problem, interfere with station operation. Batteries would be too heavy and fuel cells would require large resupply operations.	7.0 to 10.0	Radioisotope or reactor	T/E, Dynamic or T/I	Po^{210} (90 d resupply) Pu^{238} (5 yr)	262 150.1	91.6 52.8	eary 70's
4	Gemini	Non-Nuclear	Insufficient development and testing time for a nuclear system. Fuel cells are ideally suited because of the short mission duration. They have been selected and are current by being used on the Gemini craft.	---	----------	----------	----	-----	-----	1964-1967
5	FM Broadcast (unmanned)	Nuclear	Nuclear system are well suited to the long mission power requirements. Non-nuclear system would not have the reliability required. Solar powered system would require both orientation and over design to offset losses due to meteorite impact, shade portions of the orbit and long term degradation.	1.0	Radioisotope	T/E or T/I	Pu^{238} (5 yr) Po^{210} (90 d resupply)	15 26.2	-----	1970-1971

[redacted]

[redacted]

Table V (Cont'd)

Part I-A. Manned (Cont'd)

Mission and Probable Power System Identification

I is s on I ɔ.	Mission	ɔower System (Nuclear/Non-Nuclear) S lection ?rimary iecondary	Discussion	Pow-er Level Kw(e)	Radioisotope or Reactor	Type of Converter	Heat Source Character Type of Isotope or FuelElement		Initial Quantity Kw(t) T/E-T/I	Dyn.	Projected Period for Mission
5	:V Broa(:ast nmannec	ıclear	oth power level and mission duration ɛquire the use of a reactor system. The ɔwer requirements are too high for iso- ɪpe systems, even in module form (fuel ıventory).	30.0 to 60.0	Reactor	Dynamic or T/I	ZrHx UC, UO_2 etc.	T/I	at 13% ≃ 230	~150	1975[redacted]
	ORL	ıuclear	eactor system (same as above)	15 to 40	Reactor; possible large radioisotope system	T/E, T/I or Dynamic	Pu^{238} (1 yr) Pm^{147} (1 yr)	T/I T/I	115-308 160-430	----- -----	Post 1975
	·RB ELE	ıuclear or lon-Nucle:	oth energy sources appear feasible from mission standpoint (selection depends on reater definition of mission character), ıel cells might be desirable, if long term eliability can be achieved (the systems enerator water as a by-product desirable ɔr manned applications). Orientation ıay produce difficult for solar systems. 'he compactness, non-orientation and ːeedom from environment indicate that uclear systems offer considerable advantages. The planned orbit also provides ɔr in-orbit decay disposal.	1.0	Radioisotope	T/E, T/I or Dynamic	Pu^{238} (1 yr)	T/E T/I	14-20 7.7	5-6.5	1979-1980

ɴOTE: FM and TV large satellites are unmanned but are called out as missions 5 and 6.

Table V (Cont'd)

Mission and Probable Power System Identification

[redacted]

Part I-B. 'nmanned

Mis sior No.	Mission	Power Systein (Nuclear/Non-Nuclear) ielection Primary Secondary	Discussion	Pow- r I evel K v(e)	Radioisotope or Reactor	Type of Converter	Heat Source Character Type of Isotope or Fuel Element	Initial Quantity Kw(t) T/E-T/I	Dyn.	rojected eriod for lission
9	Syncom	Non-Nuclear	Solar Cells (Operational)	)25	ladioisotope	'/E	Pu^{238} (3 yr)	0. 335	-----	1963 or later
10	Explorer	Nuclear or Non-Nuclear	Non-nuclear energy sources have been used and probably will continue to use thi type. However, as small nuclear system becomes more adaptable to this type mission they may be used on later experiments particularly those involving lunar, planetary or deep space mission where the source would not interfere with mission objectives. Solar and chemical powered devices are most attractive fron both a power level and mission duration standpoint.	)15 :o)5	Radioisotope	T/E	Pu^{238} (6 mo) Pm^{147} (6 mo)	ι. 250-0. 30 0. 757	----- -----	1965-1969
11	Telstar	Non-Nuclear	Batteries (Nickel-Cadmium) and Solar Cells	)60	--------	T/E	Pu^{238} (1 yr)	0. 866	-----	1962 or later
							Pm^{147} (1 yr)	1. 14	-----	1965-1970
12	Comsat	Non-Nuclear and Nuclear	Solar cells are being used on this type satellite, but long term reliability is questionable (5 yrs). This application is, however, very suitable for nuclear power (freedom from environment and in-orbit decay would minimize post mission disposal - 6000 n. mi. or syn. (22, 500 n. mi.	. 05	ladioisotope	T/E	Pu^{238} (5 yr) Sr^{90} (5 yr)	1. 20 1. 30	----- -----	
13	Tiros	Non-Nuclear	Solar and chemical (solar cells are being used).	)25 to)5	--------	----	-----	-----	-----	1960-1970
14	Relay	Non-Nucleai or Nuclear	Solar cells and batteries	)5	Radioisotope	T/E	Pu^{238} (1 yr) Pm^{147} (1 yr)	0. 73 0. 95	----- -----	1962-1970
15	Transit	Nuclear	SNAP-9A (Pu^{238}) power unit; also solar cells and batteries (nickel-cadmium)	)25	--------	-----	-----	-----	-----	1964- 969
16	Adv. AF	Nuclear	The compactness and freedom space environment point to the use of nuclear power However, solar cells or batteries may be desirable based upon mission profile.	. 04	Radioisotope	T/E	Pu^{238} (1 yr)	0. 577	-----	1967- 975
							Pm^{147} (1 yr)	0. 761	-----	1964- 971
17	Nimbus	Non-Nucleai or Nuclear	Solar and batteries are currently being used, but nuclear power appears very satisfactory and is compatible with missi requirement. Planned activities call for the use of a nuclear system (SNAP-19) to tested with the Nimbus craft. Possibly th total power requirements (250 w(e)) may be satisfied with a nuclear system if associated safety problems can be resolved	. 05 o . 25	Radioisotop	T/E or T/I	Pu^{238} (2 yr)	0. 725-3. 66	-----	1971-1980

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[redacted]

[redacted]

Table V (Cont'd)

Mission and Probable Power System Identification

Part I-B. Unmanned (Cont'd)

Mission No.	Mission	Power System (Nuclear/Non-Nuclear)		Power Level Kw(e)	Radioisotope or Reactor	Type of Converter	Heat Source Character			Projected Period for Mission
		Selections Primary Secondary	Discussion				Type of Isotope or Fuel Element	Initial Quantity Kw(t) T/E-T/I	Initial Quantity Kw(t) Dyn	
18	RAO	Non-Nuclear	Solar system will probably be used because of integration, but long term reliability for this mission is definitely a problem (5 yr life) also orientation difficulties may pose restrictions. The long life and non-orientation characteristics of nuclear system make it desirable, if compatability can be achieved between the experiments and the nuclear source.	0.25 to 0.50	Radioisotope	T/E or T/I	Pu^{238} (5 yr)	2.5-7.5 T/E 1.9-3.9 T/I	-----	1967-1980
19	Adv. Orb. Solar Obs.	Non-Nuclear	Solar and chemical power is probably most desirable for early mission throughout the 60's. The inherent orientation toward the sun for solar experiment enhance the use of solar power systems. The long life is a limiting factor for non-nuclear systems, hence nuclear power for early 70's might be very desirable.	0.30 to 0.50	Radioisotope	T/E or T/I	Pu^{238} (1 yr) Pm^{147} (1 yr)	7.19 9.47	----- -----	1969-1980
20	Adv. Navy	Nucler or Non-Nuclear	Mission profile definition is required before a power system may be selected. The proposed mission duration and power level may be adquately resolved by either system Specific data required for selection includes: mission environment, type of experiments, sensitivity of equipment and power demand profile.	.015 to .05	Radioisotope	T/E	Pu^{238} (1 yr) Pm^{147} (1 yr)	0.723 0.95	----- -----	1970-1972
21	Operational Weather Satellite	Nuclear	The power level, mission duration and type of mission are ideally suited to nuclear power because of its compactness, non-orientation, long life and compatibility with mission objectives. If space nuclear power safety problems occur do no keep pace with the requirements for the systems solar system may be implemented, even though they are subject to both radiation and temperature degradation.	0.5	Radioisotope	T/E or T/I	Pu^{238} (5 yr) Sr^{90} (5 yr)	7.5 8.15	----- -----	Early 70's

[redacted]

[redacted]

[redacted]

Table V (Cont'd)

Part I-B. Unmanned (Cont'd)

Mission and Probable Power System Identification

[redacted]

Mission No.	Mission	Selections Primary Secondary	Discussion	ower evel w(e	Radioisotope or Reactor	Type of Converter	Heat Source Character: Type of Isotope or Fuel Element	Initial Quantity Kw(t): T/E-T/I	Dyn	Projected Period for Mission
22	Adv. Meterol. Sat.	Nuclear	(Same as above) The normal orientation of the spacecraft toward the sun for an experimental purpose makes it desirable for solar cells. However, the potential of nuclear systems to eliminate the large panel areas may be very desirable in that it may extend the field of observation (so that specific orientation may not be required).	.05 to 100	Radioisotope	T/E	Pu^{238} (2 yr)	0.725	-----	1970-1974
23	OAO	Non-Nuclear	Solar cells on first systems later units may consider nuclear power.	5 o 75	Radioisotope	T/E or T/I	Pu^{238} (1 yr) Pm^{147} (1 yr)	7.25-10.8 14.4	----- -----	through 1974
[redacted]	SMS	Nuclear	The power level and duration requirements for this mission are suited to a nuclear system, also the mission profile at prese would appear compatible with a nuclear source. Non-Nuclear power (solar) may be used but long life and orientation requir ments may pose significant problem areas	5	Radioisotope	T/E or T/I	Pu^{238} (1 yr) Pm^{147} (1 yr) Po^{210} (1 yr)	7.16 15.0 19.50	----- ----- -----	1971-1978
[redacted]	OGO	Non-Nuclear	Solar and chemical powered devices may be used to achieve quater mission compatability than could be obtained with a nuclear system. For the longer missions nuclear systems may be incorporated in the system.	3 o 5	Radioisotope	T/E or T/I	Pm^{147} (1 yr)	7.5	-----	through 1978
26	SGM	Non-Nuclear	Since orientation is required from an operational standpoint solar powered device may be effective integrated without additional wt. for orientation. Also the power level is well within the capabilities of solar devices, however, the 2 year mission life may be limiting and a nuclear source could become attractive.	15 o 20	Radioisotope	T/E or T/I	Pm^{238} (2 yr)	2.91	-----	through 1975
27	Sat. Inspec	Non-Nuclear	This selection is based upon mission requirements (pressure natural background) The high power level and long duration requirements may be more consistent with the potential of nuclear systems.	.0	Radioisotope	T/E or T/I	Pu^{238} (5 yr)	7.5	-----	mid 1970's
28	AOAO	Nuclear	In the time period desired, a nuclear system might be most attractive based upon power level and mission time.	0	Radioisotope	T/E or T/I	Pu^{238} (2 yr)	14.6	-----	1975-[redacted]

[redacted]

Part II. Lunar

Table V (Cont'd)

Mission and Probable Power System Identification

Mission	Mission	Power System (Nuclear/Non-Nuclear)		Power Level Kw(e)	Radioisotope or Reactor	Type of Converter	Heat Source Character			Projected Period for Mission
		Selections Primary Secondary	Discussion				Type of Isotope or Fuel Element	Initial Quantity Kw(t) T/E-T/I	Initial Quantity Kw(t) Dyn.	
29	Explorer		(Same as Mission No. 10)							1965-1969
30	Surv. Orb. and Lan.	Nuclear	The first flights will use non-nuclear devices but latter flights will probably use nuclear power because of their compatability with the lunar environment (less dependent upon varying so/or conditions), also, the higher specific power and compactness of design point toward the selection of a nuclear system.	.025	Radioisotope	T/E	Pu^{238} (1 yr)	7.81	-----	19[redacted]-1967
							Pm^{147} (1 yr)	2.38	-----	
31	Apollo	Non-Nuclear	Fuel cells have been selected for the mission, but their use will depend upon the establishment of reliability. If nuclear system developments proceeds rapidly, and associated safety problems can be effectively resolved, they may be used on later flights.	1.5 to 2	Radioisotope	T/E or Dynamic	Po^{210} (30 d)	38.6	13.3	late 60's
							Pu^{238} (30 d)	28.6	10.0	early 70's
							Pm^{147} (30 d)	30.1	10.5	
32	LEM	Non-Nuclear	Fuel cells have been selected for the mission, but their use will depend upon the establishment of reliability. If nuclear system developments proceeds rapidly, and associated safety problems can be effectively resolved, they may be used on later flights. Power level and duration are well suited fo non-nuclear devices.	0.10	Radioisotope	T/E	Po^{210} (7 d)	8.51	-----	early 70's
							Pu^{238} (7 d)	7.51	-----	
							Pm^{147} (7 d)	7.36	-----	
33	Roving Vehicles	Nuclear	The power level and duration variation (14 days to 730 days) suggest the use of a reactor system. However, for the low power requirements only radioisotope systems may be used (less than 10 Kw(e). Non-nuclear systems may be used for short duration missions.	3.0 to 15.0	Radioisotope Preferred Possibly Reactor	T/E or Dynamic	Po^{210} (14 d)	53.6	18.75	early 70's
							Pu^{238} (2 yr)	146	51.1	
34	Lunar Exploration	Nuclear	The use of nuclear powered systems is desirable for this type mission, both the duration and the power requirements are well within the capabilities of either an isotope or reactor systems. Fuel cells may be considered for the 28 and 90 day missions.	2(28d	Radioisotope (Possible Reactor)	T/E, Dynamic or T/I	Po^{210} (28 d)	38.6	-----	through 1980
				4(90d			Pm^{147} (90 d)	62.3	-----	
				5(2yr			Pu^{238} (2 yr)	73.9	-----	

Table V (Cont'd)

Part III. Planetary

Mission and Probable Power System Identification

[redacted]

Mission No.	Mission	Power System (Nuclear/Non-Nuclear)		Power Level Kw(e)	Radioisotope or Reactor	Type of Converter	Heat Source Character			Projected Period for Mission
		Selections Primary Secondary	Discussion				Type of Isotope or Fuel Element	Initial Quantity Kw(t) T/E-T/I	Dyn.	
35	Scientific Probe	Non-Nuclear	A selection cannot be made until the type of experiments to be performed have been defined. Solar cells or other solar devices will be limited based on the space regime to be explored. The mission duration, the reliability and the low specific power of chemical systems (batteries) limits their usefulness to a narrow region. Also, they are sensitive to large temperature fluctuations. If nuclear shielding does not impose stringent weight requirements to satisfy mission radiation fields the nuclear systems will be very attractive.	0.05 to o.50	Radioisotope	T/E	Pu^{238} (2 yr)	7.3	-----	
[redacted]	Explorer		(Same as Mission No. 10)							1965-1969
[redacted]	Aster- - Jupiter	Nuclear	The mission duration and type of space environment are well suited to the use of a nuclear device. Solar cells are not attractive because of the environment (meteorite and solar flux). The long mission duration is not compatable with chemical systems.	0.05	Radioisotope	T/E or T/I	Pu^{238} (2 yr)	T/E 0.73 T/I 0.385	-----	1967-1980
3[redacted]	Jupiter Flyby	Nuclear	The power level, duration and environment are well suited to the characteristics of nuclear power. Both solar and chemical systems would be severely limited in this environment for the planned mission duration time.	1.0 to 1.2	Radioisotope	T/E, T/I, or Dynamic	Pu^{238} (2 yr)	T/E 17.5 T/I 9.25	6.13	1972-1977
39	Mercury Flyby	Non-Nuclear	Solar powered devices are ideally suited to the high thermal flux environment, conversely the chemical systems are not as desirable in this region. Nuclear systems offer considerable flexibility in this region due to their greater freedom from the environment, relatively to other type systems. Their compactness is also very desirable.	0.4	Radioisotope	T/E or T/I	Pu^{238} (2 yr)	5.83	-----	1973-1978
40	Mariner - Mars Flyby	Non-Nuclear	Solar cells are currently being used. Again the power level, duration, and type of thermal environment are well suited to non-nuclear (solar) systems. For later flight, nuclear systems may be used to achieve a more compact design.	0.5 to 0.25	Radioisotope	T/E or T/I	Pu^{238} (6 mo) Pm^{147} (6 mo)	3.60 3.80	----- -----	late 60's through early 70's

[redacted]

[redacted]

[redacted]

Table V (Cont'd)

Part III. Planetary (Cont'd)

Mission and Probable Power System Identification

Mis ior No.	Mission	Power Syste (Nuclear/Non-Nuclear) Selections Primary Secondary	Discussion	Pow-er Leve	Radioisotope or Reactor	Type of Converter	Heat Source Char er ype of otope uel E	 r men	 Initial Quantity Kw(t) T/E-T/	 Dyn	Projected Period for Missi
1	an Marco	on-Nuclear r Nuclear	his type of power system must be selected ased upon the type of experiments to be erformed on a given mission. **Both** uclear and solar systems are capable of atisfying mission requirements.	.025	adioisotope	/E	^{238} yr) m^{147} yr)		0.36 0.475	----- -----	ate 60's early 70's
2	iterplane ry Modul	luclear	ased on power level and duration a reactor ystems is most attractive. The feasibility on-nuclear system in this region is ques- oned.	0.0 o 0.0	adioisotope	ynamic, T/I T/E	238 or m^{244}	/E	146-T/I 73 (2 yr)	51.1	970's to 1980's
3	oyager	Nuclear	he freedom from the space environment nd compactness of design for this type nission makes nuclear power the logical noice. Both solar and battery systems ould be severely limited on this type of nission because of temperature fluctua- ons and the planetary environments.	.6	adioisotope	/E or T/I	238 yr)	/E /I	8.6 4.6		971-1977
4	ABL	Non-Nuclear	olar or chemical systems would be more ompatible with mission requirements but ne long mission life may pose significant ifficulties with the chemical system and ossibly, to a lesser extent with the solar ystem. If permissible radiation levels an be met, a radioisotope system would rovide both a compact and high specific ower system.	..0	adioisotope	/E or T/I	238 yr)	/E /I	14.6 7.7	----	early 1970's
5	enus / ars econ	Nuclear	he freedom from environmental effects, s well as, suitable power and life require- nent makes the selection of a nuclear devic esirable. Non-Nuclear considerations are imilar to those identified for mission 44.	.0	adioisotope	/E or T/I	238 yr)	/E /I	14.6 7.7	----	late 1980's
6	ars anding	Nuclear	(Same as Mission No. 45)								late 1980's
7	anned ars ission	Nuclear	eactor power system is required based or oth power and duration. **The isotope** nventories would be prohibitive if the total ower were to be generated via isotope ystem design might be used to satisfy the ow power requirements.	.0) 0.0	Reactor, Radioisotope up to 10 Kw(e)	Dynamic or T/I T/E, T/I or Dynamic	238 yr) or m^{244}	/E /I	71.5 38.5	51.1	late 1980's

[redacted]

[redacted]

Part IV. Solar and Deep Space

Table V. (Cont'd)

Mission and Probable Power System Identification

Mission No.	Mission	Power System (Nuclear/Non-Nuclear)		Power Level Kw(e)	Radioisotope or Reactor	Type of Converter	Heat Source Character			Projected Period for Mission
		Selections Primary Secondary	Discussion				Type of Isotope or Fuel Element	Initial Quantity Kw(t) T/E-T/I	Initial Quantity Kw(t) Dyn.	
48	Ext. Pioneer	Nuclear or Non-Nuclear	Based on a 5 yr long mission duration and a power level of 0.5 Kw(e) a nuclear system is most desirable. The selection can only be made after thorough consideration of mission profile (experiments, radiation limits, space environment, mission duration and power demand profile).	0.5	Radioisotope	T/E or T/I	Pu^{238} (5 yr)	T/E 7.3 T/I 5.4	-----	late 1960's early 1970's
49	Adv. Pioneer	Nuclear or Non-Nuclear	(Same as Mission No. 48)							
50	Out of Ecliptic Probe	Nuclear	If compatibility can be achieved between the nuclear source and on board experiments the selection of a nuclear system would permit the design of a very compact system thus yielding a low cross-section for meteorite impact. Also, the freedom afforded by a nuclear device from the flight environment is very desirable for this type mission.	0.5	Radioisotope	T/E or T/I	Pu^{238} (2 yr)	T/E 7.3 T/I 5.4	-----	late 1960's early 1970's
[illegible]	Interstellar	Nuclear	(Same as Mission No. 50) In this case the mission environment and duration may dictate the type of power source. Temperature variations along the trajectory could place stringent limitations upon both solar and chemical powered system.	0.5	Radioisotope	T/E or T/I	Pu^{238} (2 yr)	T/E 7.3 T/I 5.4	-----	1967 through 1980's
5[illegible]	Comet Intercept	Nuclear	(Same as Mission No. 51)	0.03	Radioisotope	T/E or T/I	Pu^{238} (1 yr)	T/E 0.429 T/I 5.4	-----	1970's

[redacted]

their decayhalf-life, and their specific power (watts/gm). For missions where radiation level was not considered a limiting factor, strontium-90 was selected. This does not imply that other radioisotopes are not desirable for space useage when the mission profile, instrumentation, and isotope availability and cost have been analyzed for a particular mission. Also, technological advancements in the development of radioisotope fuel forms may provide additional flexibility in terms of nuclear safety for certain isotopes. This is especially true for Cm-244 which may be considered for some of the same missions as Pu-238. However, in all cases where shielding is required, lighter weight systems occur for the Pu-238. However, Reference 6 indicates that because of the availability problems with Pu-238, Cm-244 should still be considered to fill in the power requirements gap. Therefore, this isotope has also been considered in Section of this study.

A summary of the expected launch inventories based on Tables III and V is shown in Table VI. It can be seen that throughout the 1965-70 time period, Pu-238 and Pm-147 represent the majority of the inventory launched, in the 1970-75 time period, Pu-238 and Po-210 seem to show the highest use, and from 1975-80 Pu-238, Cm-244 and Pm-147 appear most often.

From the standpoint of aerospace safety therefore, the safety techniques which have most application to Pu-238 systems should have highest developmental potential with Po-210 and Pm-147 systems being considered for the intermediate and short mission times respectively.

The peak possible useage of radioisotopes appears in the 1970-75 time period which means that if the full potential of nuclear auxiliary power is to be realized in this time period decisions to institute development of aerospace nuclear safety techniques, which will both keep hazards to an acceptable level and not compromise the main mission objectives, must be made in the 1965-67 time period.

D. Power Plant Induced Problems Affecting Aerospace Nuclear Safety

Identification of nuclear power plant induced problems which affect system design and in turn the safety approach to be followed throughout the mission, resolve themselves into two basic areas: (1) the type of energy converter, and (2) the type of nuclear energy source (radioisotope or reactor).

1. Energy Conversion Techniques

a. Dynamic Systems. Rankine, Brayton and Stirling. Overheating and meltdown of the nuclear source resulting from loss of coolant flow or complete loss from the system - failure may be induced by meteorite puncture, seal leakage or failure (high speed shaft or high pressure pumps), component failure (pumps, valves, bearing, etc.), zero gravity conditions and corrosion caused by the working fluid. Other causes of overheating and meltdown include vacuum welding of moveable heat dump shutters and doors or changes in emissivity coating on radiating surfaces.

[redacted]

Table VI

A Preliminary Estimate of Future Nuclear Mission Launch Inventories (Radioisotope)

Estimated Power Which May be Launched, Kw(t) (Manned and Unmanned)

Year	1966	1967	1968	1969	1970	1971	1972	1973	1974	1975	1976	1977	1978	1979	1980
Pu-238	10	11.3	11.3	39.6	62.4	73.8	149.8	102.1	116.2	240	193.1	247.5	186.5	114	36.2
Po-210		11.9	20.4		33	65	125	122	125	91	31.8		31.8		
Pm-147		14.45	5.51	38.2	25.85	33.95	30.95	49.65	46.85	36.57	45.9	42.8	35.8		
Cm-244		Can be considered as backup for many Pu-238 missions in this time period											70	70	150
Sr-90	8	8	10	10		8.5	8.5	17							
Totals (including an indetermancy of ± 50%) Kw(t)	10-30	25-75	25-75	25-75	75-100	100-250	200-240	150-300	150-450	250-500	250-350	250-500	200-450	100-300	100-300

[redacted]

b. Thermionics. Several intrinsic characteristics complicate the design of a nuclear source to satisfy basic safety requirements, such as the high normal operating temperatures, the compactness of heat source design (auxiliary heat dump required for abort situation to prevent meltodown) and fuel containment difficulties under missile abort conditions because of limited materials which have good high temperature oxidation resistance, strength, and impact properties.

c. Thermoelectrics. This type converter imposes less stringent conditions upon the design of a heat source that will satisfy the basic nuclear safety requirements than the previous type converters. However, if the primary heat rejection system utilizes a circulating coolant then the problems associated with the system are in many areas similar to the dynamic conversion system.

2. Energy Source

a. Radioisotopes. The selection of a particular radioisotope and its fuel capsule for a given mission to minimize induced safety problems is a complex iterative process involving the following considerations:

Nuclear properties - Half-life, decay scheme (type and energy of emitted radiation), and impurity isotopes.

Chemical properties - Fuel form, material compatibilities, and impurity effects on stability.

Thermal and Physical Properties - Power density, thermal conductivity, heat capacity, heat of fusion, density, gas generation and retention, melting point, boiling point, specific activity, viscosity, and surface tension.

The fuel and the containment structure must be evaluated throughout all phases of the mission to insure compatibility and compliance to the selected safety design criteria. In addition, the radiological and radiobiological characteristics of the radioisotope must be considered under conditions of containment and/or planned release. Specific considerations must be given to the effects upon certain critical organs (i.e., Sr-90 bone seeker, organ burden, total body burden and maximum permissible concentration and surface contaminations. In general, the magnitude of induced hazard is a function of radioisotope inventory except for those isotopes which are fissionable.

b. Reactors. The primary induced safety problem is one of post-mission disposal (should it be orbital destruct, orbital storage, reentry burnup, etc.). The magnitude of induced hazard is correlatable to the total fission product inventory which is a function of power level and operating time. As in the case of radioisotopes, the fuel element matrix is of chief importance in establishing the aerothermodynamics and aerochemistry reentry characteristics of the fuel which dictates the feasibility of burnup and high altitude dispersal. To minimize

[redacted]

[redacted]

the magnitude of nuclear accidents and to reduce the probability of their occurrence numerous reactor flight safety criteria have been developed to provide a guideline for system evaluation. Currently plausible reactor design criteria includes:

(1) Reactor system shall be designed to prevent accidental criticality.

(2) Reactor shall not be started up until desirable orbit has been verified.

(3) Inherent shutdown mechanisms shall be incorporated such as moderator loss on cladding failure or coolant flow stoppage, and negative temperature coefficient.

(4) Termination of reactor operation upon mission completion.

(5) System and component design to satisfy reentry burnup and atmospheric dispersal when desirable based on mission characteristics of low orbital lifetime, operating time and power level.

(6) The reactor system shall be designed, experimentally verified, transported and integrated into the spacecraft without endangering the general public or handling personnel.

Only after thorough consideration of power plant induced safety problems, as well as mission induced problems, can an acceptable safety design approach be established that will permit the utilization of nuclear power in space. As in the design of a system from an operational standpoint, specific data on the particular mission and vehicle under consideration must be utilized to develop a system which will satisfy mission power requirements and integrate into the spacecraft. Accordingly, the degree to which a nuclear system complies to selected nuclear safety criteria can only be determined after a detailed analysis of the final spacecraft and its probable modes of failure throughout the mission. Since many of the studied aerospace missions have not been sufficiently detailed (type of spacecraft, location of components, launch trajectory and/or vehicle, other nominal trajectories, type of power system, etc.) to permit specific safety analysis, nuclear safety design criteria were developed that would permit general design and development of nuclear systems for the planned missions within acceptable safety guidelines. The objectives being to determine the limits of application of feasibility of the safety technique to a general class of missions rather than to their specific missions shown in Tables III and V. This was accomplished by considering both mission and power plant induced aerospace nuclear safety problems.

[redacted]

[redacted]

[redacted]

[redacted]

III. POSTULATION OF AEROSPACE NUCLEAR SAFETY PHILOSOPHIES AND THEIR APPLICATION TO FUTURE MISSION PROFILES

The definition of future nuclear missions undertaken in Section II of this study can now serve as a basis to which various Aerospace Safety Philosophies can be applied. This is shown in Table VII which considers ten aerospace safety techniques applicable to end of life disposal of the nuclear heat source as applied to the four classes of missions -- earth orbital, lunar, planetary, and solar and deep space.

These ten techniques can be categorized into the following general classes of ultimate disposal techniques:

(1) Intact Reentry
  (a) Intact controlled reentry with recovery
  (b) Intact controlled reentry (non-recovery)
  (c) Intact uncontrolled - random reentry
  (d) Intact uncontrolled reentry with recovery
(2) Reentry Burnup
  (a) Complete burnup with high altitude dispersion
  (b) Partial burnup - microspheres
(3) Orbital Storage
(4) Escape and/or Space Disposal
(5) Space Destruct
(6) Space Recovery

For purposes of consistency, these philosophies shall be defined as follows:

1. Intact Controlled Reentry With Recovery

The heat source can be returned intact to a predetermined site or sites on the surface of the earth and be recovered intact after impact without causing undue hazard to the earth's population during any phase of the mission profile.

2. Intact Controlled Reentry (Non-Recovery)

The heat source has the ability to reenter intact and in a semi-controlled fashion. That is, the heat source can be guided to large remote areas such as an ocean for ultimate disposal but does not have the capacity to land within a small predetermined recovery area.

[redacted]

Table VII. Aerospace Nuclear Safety Criteria and Their Potential Application to Space Regimes

Reentry Burnup with High Altitude Dispersion (5)
Intact Uncontrolled Reentry with Recovery (4)
Intact Uncontrolled Reentry (3)
Intact Controlled Reentry (Non-Recovery) (2)
Intact Controlled Reentry with Recovery (Manned and/or Unmanned Missions) (1)

*Primary Disposal Philosophy
**Secondary or Backup

(6) Reentry Burnup with Partial Dispersion (Microspheres)
(7) Orbital Storage*(Intact or Burnup Reentry Mode)**
(8) Escape Space Disposal* (Intact or Burnup Reentry Mode)**
(9) Space Destruct* (Nuclear and/or Non-Nuclear) (Intact or Burnup Reentry Mode)**
(10) Space Recovery* (Intact or Burnup Reentry Mode)**

Space Regime	Abort Condition	Consequences	(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)
Earth Orbital	a) Propulsion failure (fire and explosion)	Ballistic reentry or short orbit	▲	▲	⬡	▲	▲	▲	▲	●	▲	▲
	b) Explosion (propellant tank failure)	Fire, overpressure, reentry										
	c) Orbital injection errors (guidance failure)	Ballistic reentry or short orbit										
	d) Orbital transfer collisions	Power system damage, reentry altered										
	e) Meteorite and vacuum induced effects (component failure)	Loss of coolant, possible meltdown, and vacuum welding of components (system failure)										
	f) Spacecraft environmental control system failure	Overheating and/or meltdown										
	g) Spacecraft docking accidents	Power system damage, reentry altered										
	h) Premature startup or excursion (reactor systems)	High radiation and fission product inventory (short-lived orbit)										
Lunar	a) Translunar injection errors from an earth parking orbit	Super orbit reentry (earth), lunar impact, or heliocentric or selinocentric orbit)	▲	▲	⬡	⬡	▲	▲	▲	▲	▲	⬡
	b) Midcourse and lunar approach guidance and control malfunction	Super orbit reentry (earth), lunar impact, or heliocentric or selinocentric orbit)										
	c) Lunar orbit and retro	Lunar orbit or impact										
	d) Lunar landing malfunction (loss of coolant, radiator damage)	Possible overheat with release of fuel										
	e) Lunar ascent and rendezvous	Spacecraft collison, overheat and meltdown, degrade ablator or reentry scheme										
	f) Transearth injection (guidance, control, or communication)	Superorbital reentry (earth), heliocentric trajectory, earth orbit, and subsequent earth impact										
	g) Meteorite and vacuum induced effects (component failure	Superorbital reentry (earth), heliocentric trajectory, earth orbit, and subsequent earth impact										
	h) Propulsion system failure	Ballistic reentry or short orbit										

Symbol		Definition
▲	(Applicable)	This class of technique has the potential of satisfying to varying degrees the overall aerospace nuclear safety philosophy within the operational and design limitations of the mission and vehicles considered.
⬡	(Marginal)	This class of technique can be used in only very limited situations or is thought to be extremely sensitive to the quantity and type of radioisotope, reliability, point of use in a mission, or the development of new technology. In addition its use may induce complications in the design of the vehicle or power plant which could compromise the mission.
●	(Non-Applicable)	This class of technique would definitely compromise public safety, the mission, vehicle or power plant to such an extent as to make its use prohibitive.

Remarks

Space Regime	Criterion No.	Discussion
Earth Orbital	3	Depends on fuel quantity and type.
	8	Excessive weight penalty.
Lunar	3	Depends on fuel quantity and type. Cislunar return velocities may dictate heavy reentry vehicles.
	4	Limited to small systems
	10	Possible for aborts which do not perturb initial earth parking orbits or create satellite instabilities which cannot be damped.

Table VII (Cont'd)

(1) ntact Controlled Reentry with Recovery (Manned and/o Jnmanned Missions)

(2) ntact Controlled Reentry (Non-Recovery)

(3) ntact Uncontrolled Reentry

(4) ntact Uncontrolled Reentry with Recovery

(5) eentry Burnup with High Altitude Dispersion

(6) Reentry Burnup with Partial Dispersion (Microspheres)

(7) Orbital Storage*(Intact or Burnup Reentry Mode)**

(8) Escape Space Disposal*(Intact or Burnup Reentry Mode)**

(9) Space Destruct* (Nuclear and/or Non-Nuclear) (Intact or Burnup Reentry Mode)**

(10) Space Recovery* (Intact or Burnup Reentry Mode)**

pace Regime	Abort	Consequences	(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)
lanetary	a) Transplanetary trajector errors b) Guidance and communication failures c) Hostile environment (planetary) chemical, extreme temperature fluctuations, composite of atmospheres (higher gravity) d) Meteorite and high vacuum induced effects (component failure) e) Power system control (startup/shutdown), (power excursion) f) Rendezvous and docking g) Planetary landing h) Propulsion system (fire and explosion)	Superorbit reentry, heliocentric orbit or planet impact Superorbit reentry, heliocentric orbit or planet impact Breach of fuel containment via overheat, corrosion or high velocity impact Breach of fuel containment via overheat, corrosion or high velocity impact High radiation and fission product inventory Collision - degrade thermal control system (overheat or meltdown), alter system aerodynamic characteristics. Surface burial and meltdown. Surface burial and meltdown.	▲	▲	▲	●	▲	▲	○	▲	▲	○
olar and eep Space	a) Launch injection b) Guidance and communication failures c) Hostile environments of space (thermal and nuclear d) Meteorite and high vacuum induced effects (component failure) e) Propulsion system (fire and explosion)	Superorbital reentry (earth) and heliocentric trajectory Earth reentry or heliocentric trajectory Breach of fuel containment structure Breach of fuel containment structure Breach of fuel containment structure	○	●	▲	●	▲	▲	○	▲	▲	●

[redacted]

Space Space Regime	Criterion No.	Remarks Discussion
Planetary	4	Weight penalty too severe for even small planetary systems.
	7	O.K. for parking orbit aborts and earth return missions.
	10	Limited to near earth.
Solar and Deep Space	1	Not applicable to end of mission disposal, but applicable to ascent abort
	2	Excess reentry velocities at end of mission. O.K. for launch aborts and small systems.
	4	Not applicable to large systems or end of mission disposal.
	7	Use fuel only if escape capability fails.
	10	With escape capability, recovery not necessary, mission frequency low.

*End of life disposal

** Ascent abort safety technique

[redacted]

3. Intact Uncontrolled Reentry

The heat source has the capability to survive random reentry but has no control over its impact point on the earth's or a planetary surface.

4. Intact Uncontrolled Reentry With Recovery

The heat source has the capability to survive random reentry and is still uncontrolled, however, it is equipped with a balloon system which opens below 100,000 feet and suspends the heat source at reasonable aircraft flight altitudes until it can be identified and picked up by aircraft in flight. The technique is much the same as that already demonstrated by the Air Force in recovery of the Discoverer series of space capsules. The only difference is in the balloon system.

Since the system proposed is an active one, there is a finite probability of its failure. If this occurs, the system reverts back to the basic random intact reentry postulated in 3 above.

5. Reentry Burnup With Altitude Dispersion

The heat source is designed to be completely destroyed upon reentry into the earth's atmosphere into debris which will assure a high residence time and dispersion in the earth's atmosphere. The eventual fallout from such a release should be such as to not represent a hazard to the earth's population.

6. Reentry Burnup With Partial Dispersion (Microspheres)

The heat source is composed of microspheres of an inert biological character and impervious to any chemical or thermal changes during or after reentry. The containment of these microspheres is designed to be breached at a high altitude and the microspheres fall freely to the earth's surface. The basic safety concept is such that biological hazards are minimized due to:

(1) Inertness of fuel form.

(2) Microspheres are too large to be inhaled.

(3) Normal fallout dispersion is such that the final ground concentration will not be a hazard.

(4) Ingestion hazards are light and the event remote.

7. Orbital Storage

The heat source is left in orbit until complete decay is achieved. If aborts are such as to yield a short lived orbit, these techniques will be available to add the necessary velocity requirement to attain and keep the required orbit.

[redacted]

[redacted]

8. Escape and Space Disposal

In orbital missions, the heat source and attendent equipment can be put into an escape trajectory yielding deep space and/or solar disposal. In planetary missions, the final trajectory of the heat source can be controlled so as to avoid earth reentry and lunar or planetary impact.

9. Space Destruct

The heat source can be completely or partially reduced in activity and eventual reentry hazard by operations made to occur in space. These operations, such as vaporization and fracture of the heat source, can be triggered remotely or by an abort sequence on board the spacecraft.

10. Space Recovery

The heat source can be recovered from the original space vehicle while still in space and returned to earth via techniques described in item 1.

The preliminary judgments made in Table VII show that there are applicable disposal techniques for each operational space regime that our future mission profiles define. The questions of which to use, under what circumstances, and what their technical limitations and feasibility are remains to be answered.

In general, the determination of a particular reentry mode for either a reactor or radioisotopic system must be primarily based on the lift-off to orbital insertion and the probable reentry trajectories (including the post-mission trajectory), the magnitude and nature of the ratioactive material involved, and the accountability required. Each reentry mode has certain advantages and disadvantages with respect to satisfying both mission requirements and the safety philosophy. For example, if an evaluation of mission trajectory, system and fuel character reveal that burnup is more desirable, the power system would become much lighter and more desirable from an overall mission viewpoint. However, independent of the final reentry mode selected for the mission the radioisotopic fuel for RTG systems must be contained throughout all ground and launch operations in a manner so as not to create any undue radiological hazard. Similarly reactor systems must be rendered radiologically safe under the same conditions, especially under impact or environmental conditions which could yield an excursion. Also, the design (geometry) and materials used in RTG systems which involve large quantities of fissionable nuclei must satisfy non-critical array requirements under abort conditions.

The following technical discussion of these aerospace safety techniques attempts to put them into perspective relative to feasibility limitations and applications.

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IV. TECHNICAL EVALUATION OF THE TECHNIQUES POSTULATED TO SATISFY FUTURE AEROSPACE NUCLEAR SAFETY REQUIREMENTS

Present aerospace safety philosophies can be broken into two distinct operational categories, philosophies for reactor systems and philosophies for isotope systems.

For reactor systems, to date, most attention has been given to formulation of a flight safety philosophy for the SNAPSHOT experiment and to a degree can be expected to influence future safety philosophies for advanced systems. The important points in this safety plan are:

(a) No hazard to operating personnel or to general public from any assembly, pre-launch, launch or ascent aborts.

(b) No startup until a long life orbit has been achieved.

(c) Inherent disassembly of the reactor into a non-critical geometry in any pre-orbital aborts or end of life flight aborts, either by impact or reentry heating.

(d) Burnup of the reactor fuel elements into sub-micron sized particles above 100,000 feet during a random reentry after failure of all reactor shutdown mechanisms.

Therefore because of the key features of not starting the reactor until a long-lived orbit is achieved and special design characteristics which assure a non-critical geometry resulting from impact or during reentry, hazards to the general public can be minimized. In general, any aborts on the ground or during the ascent trajectory to orbit will not constitute a hazard to the general public because any released radioactivity will be completely contained in a particular area remote from any populated areas. This is achieved by virtue of the control of the downrange impact points through choice of initial trajectory.

Hence, the only questionable area arises from the magnitude of the hazards which would result from end of life random reentry after failure of all shutdown systems. Here both the impact points of the resulting debris and the concentration of radioactivity are unknown because of the random entry into the earth's atmosphere from a decaying orbit and the indefinite amount of burnup of the reactor system.

The present aerospace safety philosophy for most isotope systems flown or in the conceptual stage to date can be given as:

(a) Complete containment of the radioisotope fuel on all pre-launch, launch and pre-orbital aborts.

(b) Burnup of the fuel to submicron sizes at altitudes above 100,000 feet upon random reentry into the earth's atmospher[redacted]

[redacted]

[redacted]

(c) Intact reentry of the fuel form if no biological shielding is required, and the fuel form is inert enough to assure its ecological stability. This may take the form of intact reentry with containment or microspheres.

Depending on the ascent profiles, there could be some aborts late in the pre-orbital flight which would generate enough energy upon reentry of the radioisotope generator to only partially consume the fuel. However, because the impact points can be controlled by proper choice of the launch trajectory, remote ocean disposal of abort debris can be achieved to a high probability.

A most difficult paradox exists if this condition occurs. Radioisotope fuel forms with high vapor pressures are very dangerous in the partial burnup case. Once breach of the capsule occurs at low altitude ($h \approx 10^5$ ft), a partial breakup and vaporization of the fuel form may release all of the fuel in the final descent leg of the trajectory. Even though this is over water, a radioactive cloud could easily be formed and quickly migrate to a land area.

Therefore, one of the most favorable properties which makes a fuel form desirable from a high altitude burnup standpoint could produce a severe hazard in a low altitude ascent abort which yields partial burnup.

Several solutions to this problem exist. First, as mentioned above, the probability of partial burnup occurring is tied to the reliability of the launch vehicle, which is continually being increased as is evidenced by the success of the National Space Program. Second: if fuel forms are chosen that have a high vapor pressure only at temperatures which can be reached by sustained heating approximating that of satellite decay, the release under 100,000 feet from partial burnup conditions can be minimized.

Therefore, the only portion of the present isotope aerospace nuclear safety criteria which is in doubt is the probability of complete reduction of the fuel form to acceptable sizes above 100,000 feet via complete reentry burnup or the possibility of feasibly achieving intact reentry in both the normal and microsphere definitions.

Since the terminal stages of the flight are obviously the most critical from a safety standpoint two choices are open to the designer.

The first is to accept the eventual reentry and design the nuclear system to either be destroyed to the degree required by the aerospace safety burnup criteria, to be destroyed partially and still achieve acceptable ground concentrations, or to survive the reentry intact. The second is to prevent or delay the reentry from occurring and to use space as the eventual disposal sink for the radioactive material to as great a degree as possible.

In special cases a combination of both techniques such as space destruction with eventual reentry burnup may be required. The following discussion investigates the feasibility of these two choices and evaluates their implications on system design.

[redacted]

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A. Reentry Burnup and Dispersion

The history of using the inherent and kinetic energy of a reentering device as an aerospace safety technique to destroy the device and reduce eventual ground hazards dates back to SNAP 3 and 9A, the first operational isotope generators in space. The implications of the technique are very attractive from a generator design standpoint since they assure a minimum weight system. However, with the advent of higher temperature systems, the material and fuel forms required to maintain their integrity under normal operating conditions are no longer conducive to easy burnup during reentry.

In order to determine the limitations of the general burnup philosophy and their implications relative to the choice of this philosophy on future nuclear missions, two basic questions have to be answered:

1. What is the level of hazard which can be tolerated in terms of nuclear power plant inventory added to the earth's atmosphere?
2. What is the destruction potential of the fuel form of interest and does it satisfy present aerospace nuclear safety criteria?

1. Determination of Consequences of High Altitude Dispersion of Radioisotope Fuel

Because of the potential radiation hazard to man through resulting fallout, an analysis was undertaken to evaluate the safety of the present aerospace burnup criterion, assuming that high altitude burnup has occurred and one micron particles are the result. A convenient way of indicating relative hazard is through the use of a nuclear safety index (NSI). Several of the radioisotopes considered most often for space use were used in this analysis:

(1) Pu^{238}

(2) Sr^{90}

(3) Po^{210}

(4) Ce^{144}

(5) Pm^{147}

(6) Cm^{244}

In order to evaluate a nuclear safety index, certain hypothetical conditions must be assumed. The following assumptions were made for this study:

(1) Complete burnup in the mesosphere to 1 μ

(2) Complete circumferential dispersion in a 20-70 degree

[redacted]

[redacted]

latitude band in the hemisphere of injection.

(3) Mean residence times as follows (Reference 7):

(a) Mesosphere (above 160, 000 feet) - 5 years

(b) Stratosphere (40, 000 - 160, 000 feet) - 2 years

(c) Troposphere (below 40, 000 feet) - 21 days

(4) Beginning of life fuel inventory of 80 Kw(t). This represents an average of proposed fuel inventories relative to the single mission profiles discussed in Section II of this report. Since the effects of release inventory are linear, definition of only one release power level is necessary. The variations with power level can then be proportioned.

Using the above assumptions, maximum activity releases for each of the proposed isotopes can be determined. Based on the power densities of the various radioisotopes investigated the activities associated with a fuel inventory of 80 Kw(t) are listed in Table VIII.

Table VIII

Maximum Activity for 80 Kw(t) Release

Isotope	Activity Release (curies)
Pu-238	2.3×10^6
Sr-90	1.3×10^7
Po-210	4.6×10^6
Ce-144	1.7×10^7
Pm-147	2.9×10^8
Cm-244	2.4×10^6

Appendix A contains all the fallout and deposition equations necessary to calculate activity burdens realized in the troposphere and on the earth's surface. Equations (9) and (12) of Appendix A were solved and are shown graphically in Figures 1 and 2. Maximum activity burdens for 80 Kw(t) of each isotope under consideration and the corresponding times are shown in Table IX.

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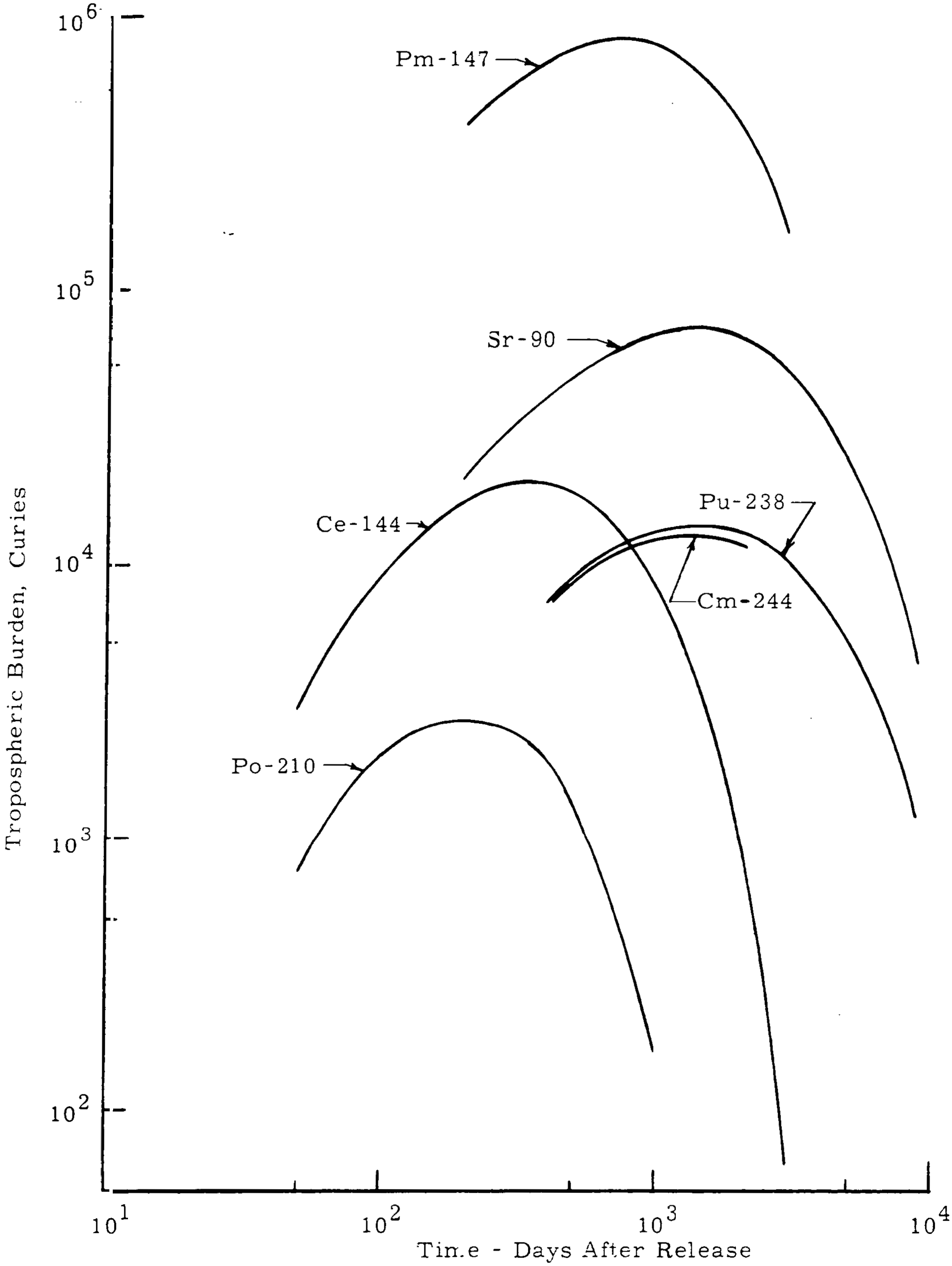

Figure 1. Tropospheric Burden vs Time from a Mesospheric 1 μ Release [redacted] Initial Loading of Isotopic Fuel.

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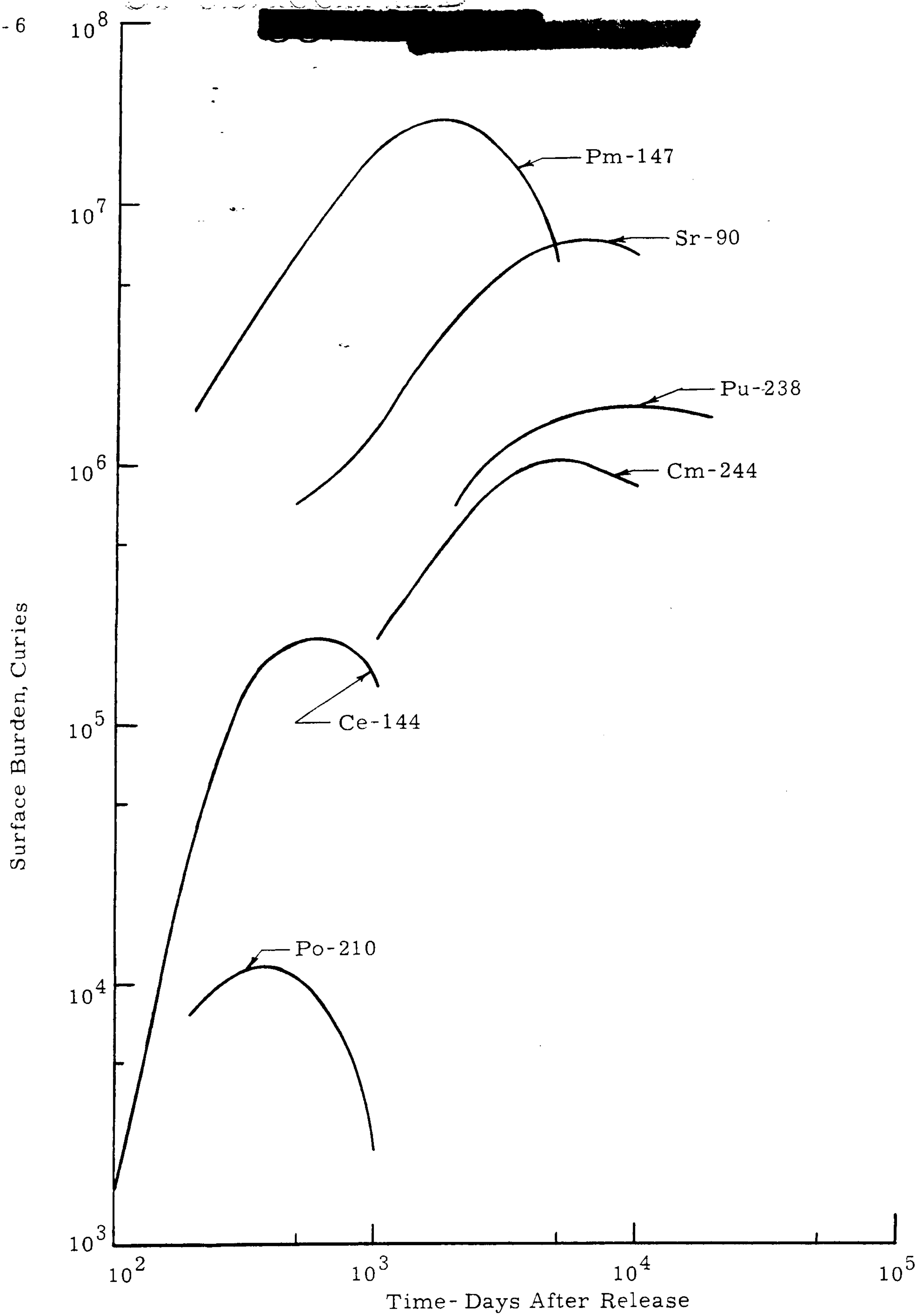

Figure 2. Earth's Surface vs Burden Time from a Mesospheric 1 μ Release of 80 kw(t). Initial Loading of Isotopic Fuel.

[redacted]

[redacted]

Table IX

Maximum Activity Burdens

Isotope	Troposphere		Surface	
	Burden (curies)	Time after release (days)	Burden (curies)	Time after release (days)
Pu-238	1.4×10^4	1600	1.7×10^6	9000
Sr-90	7.2×10^4	1400	7.2×10^6	6300
Po-210	2.7×10^4	200	1.2×10^4	375
Ce-144	2.1×10^4	350	2.1×10^5	600
Pm-147	8.3×10^5	750	2.1×10^7	1700
Cm-244	1.3×10^4	1300	1.05×10^6	5000

A "fallout ratio" can now be determined for each of the isotopes by comparing the values of Table IX with those of Table VIII. "Fallout ratios" are shown in Table X .

Table X

Fallout Ratios

$$\frac{\text{(Maximum Burden)}}{\text{(Maximum Release)}}$$

Isotope	Tropospheric Ratio	Surface Ratio
Pu-238	6.08×10^{-3}	7.39×10^{-1}
Sr-90	5.54×10^{-3}	5.54×10^{-1}
Po-210	5.86×10^{-4}	2.61×10^{-3}
Ce-144	1.18×10^{-3}	1.24×10^{-2}
Pm-147	2.86×10^{-3}	7.5×10^{-2}
Cm-244	5.4×10^{-3}	4.37×10^{-1}

These "fallout ratios" are convenient in that they directly relate maximum activity realized in the troposphere and on the earth's surface to the initial high altitide release of each isotope.

[redacted]

Recent evaluation of fallout patterns (Reference 8) shows that about 80 percent of the worldwide tropospheric and surface activity burdens lies fairly uniformly between 20° and 70° latitude in the hemisphere of injection. Corresponding earth area and tropospheric volume for this region are as follows (see Appendix B):

$$\text{Area}_{\text{Earth}} = 1.54 \times 10^{14} \text{ m}^2$$

$$\text{Volume}_{\text{Troposphere}} = 1.65 \times 10^{18} \text{ m}^3$$

Using the above information and that given in Tables VIII, IX, and X, contamination values for each quantity of fuel can be found (see Appendix B for sample calculations).

Nuclear safety indices are defined as follows:

$$(1) \quad \text{NSI}_a \text{ (air)} = \frac{\text{maximum tropospheric concentration}}{\text{MPC}_a \text{ (air)}}$$

$$(2) \quad \text{NSI}_s \text{ (surface)} = \frac{\text{maximum surface contamination}}{\text{MPC}_s \text{ (surface)}}$$

Graphs of NSI derived in this study are shown in Figures 3, 4 and 5. Figures 6, 6a, 6b, and 6c comprise a bar chart showing NSI values for each of the 52 proposed missions. The values of MPC's for air and surface were obtained from NBS Handbook 69, 10CFR, Part 20, revised to 1964 levels as shown in Reference 10. Ce-144 was not considered in Figure 6, due to its high radiation level which manifests itself in more shielding than the other isotopes of interest with no apparent performance advantage. Cm-244 was not considered due to its high neutron level requiring high shielding weights for those missions where background radiation is important. However, in many cases it is a good substitute for Pu-238 and could be used. Exact specification of where both might be used and why was felt dependent on mission and power design details beyond the scope of this study.

Maximum permissible concentrations based on a 168 hour week were used as standard for NSI_a calculations (Figure 3). Some question arises, however, as to the basis of maximum permissible surface contamination values.

Since there has never been any extensive research into standardizing permissible surface contaminations, other than those studies based on dairy farming, or nuclear excursions, maximum permissible values used in this report (Table XI from Reference 10) require some explanation.

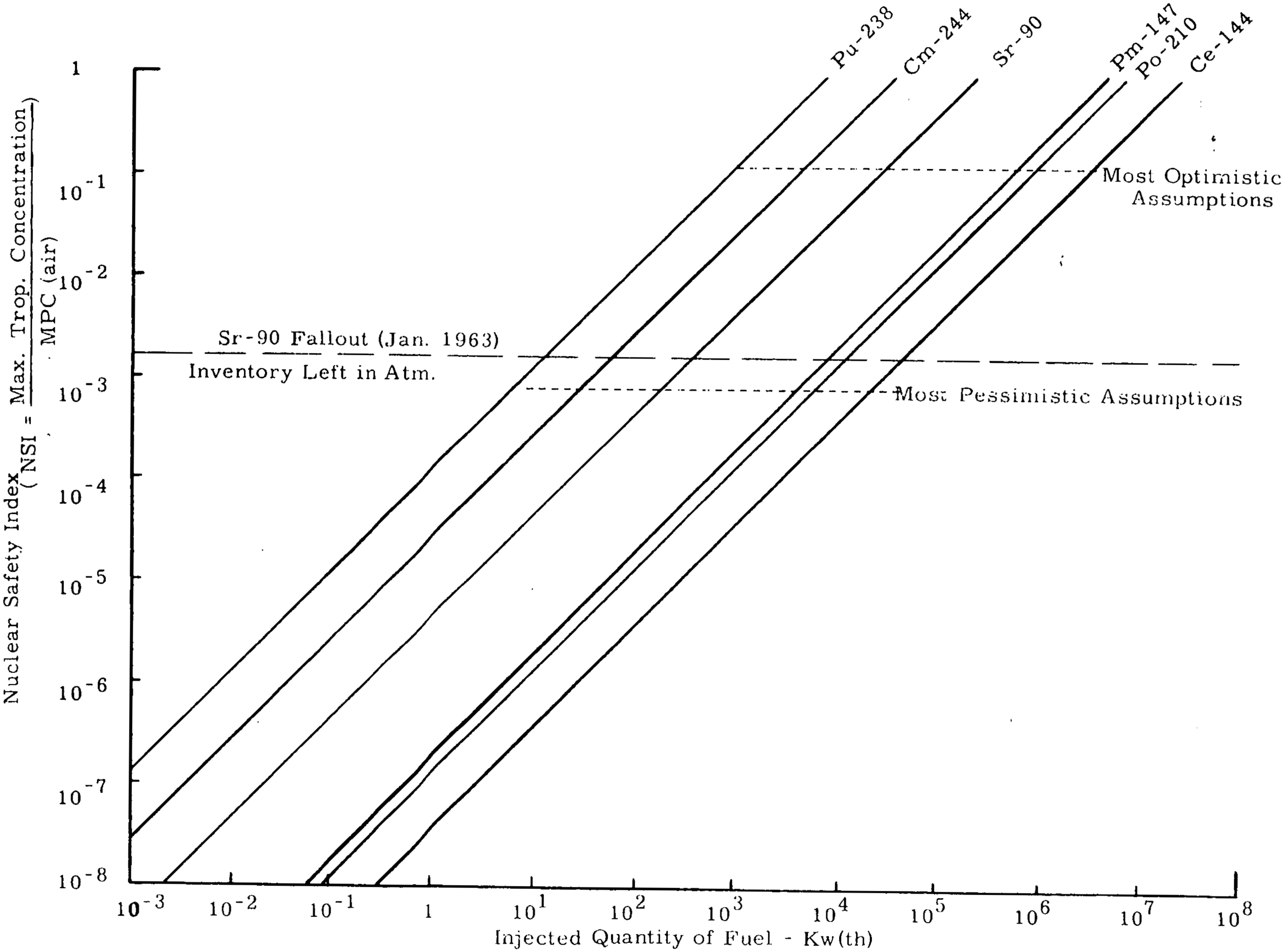

Figure 3. Nuclear Safety Index for Tropospheric Air Contamination Between 20° and 70° Latitude for a Mesospheric 1 μ Release.

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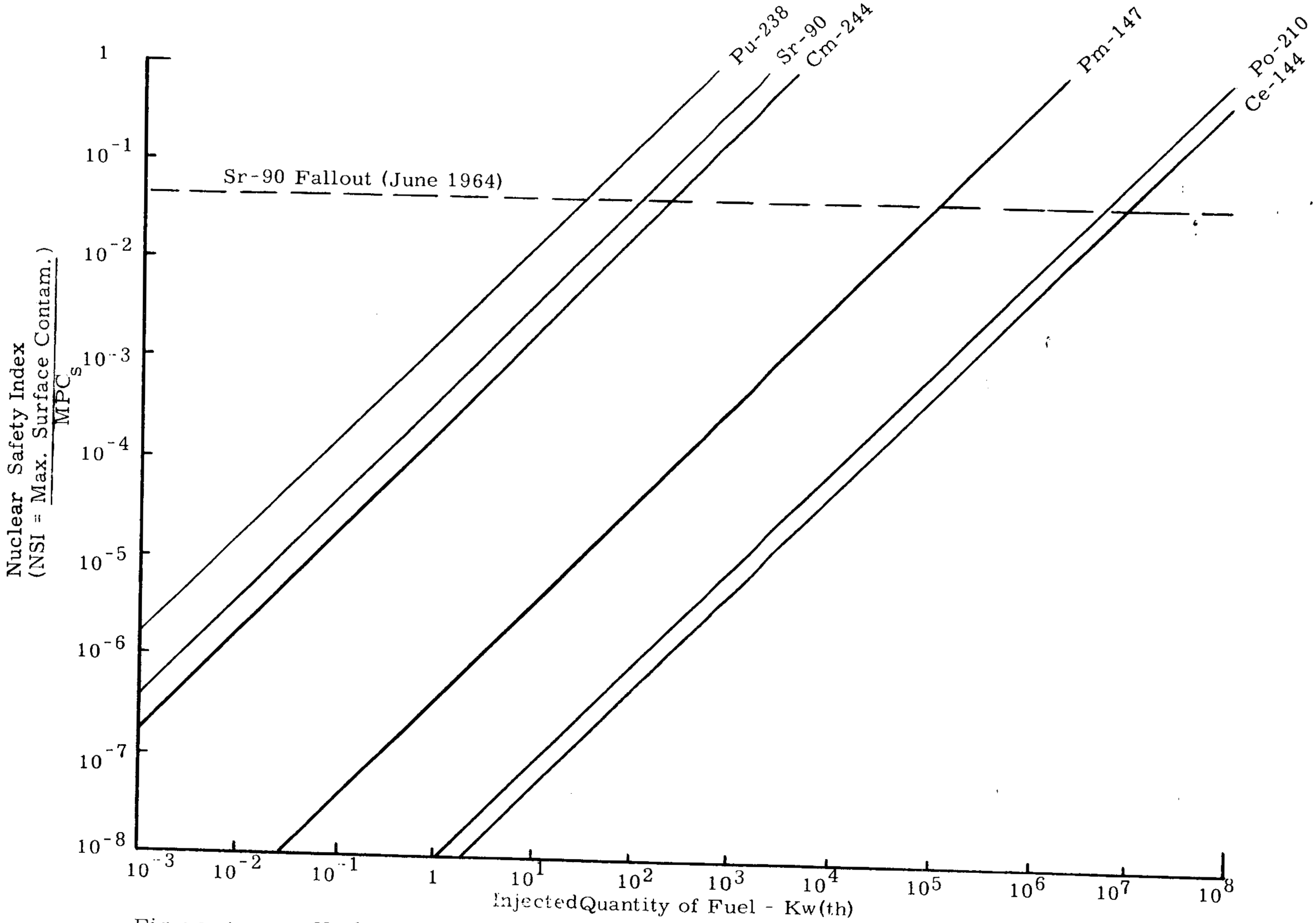

Figure 4. Nuclear Safety Index for Earth Surface Contamination (Minor Restriction) for a Mesospheric 1 μ Release.

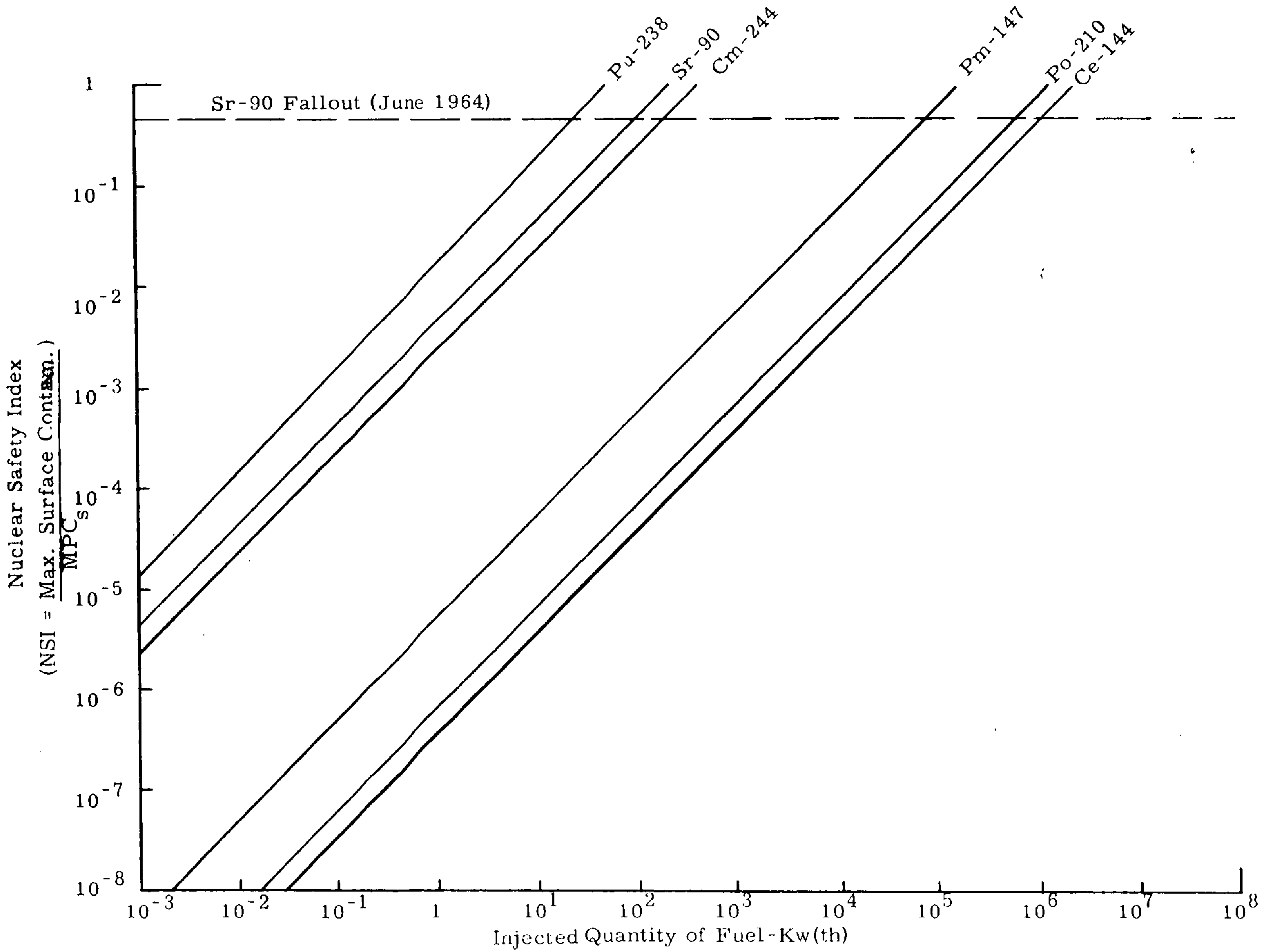

Figure 5. Nuclear Safety Index for Earth Surface Contamination Between 20° and 70° Latitude (No Restrictions) for a Mesospheric 1μ Release

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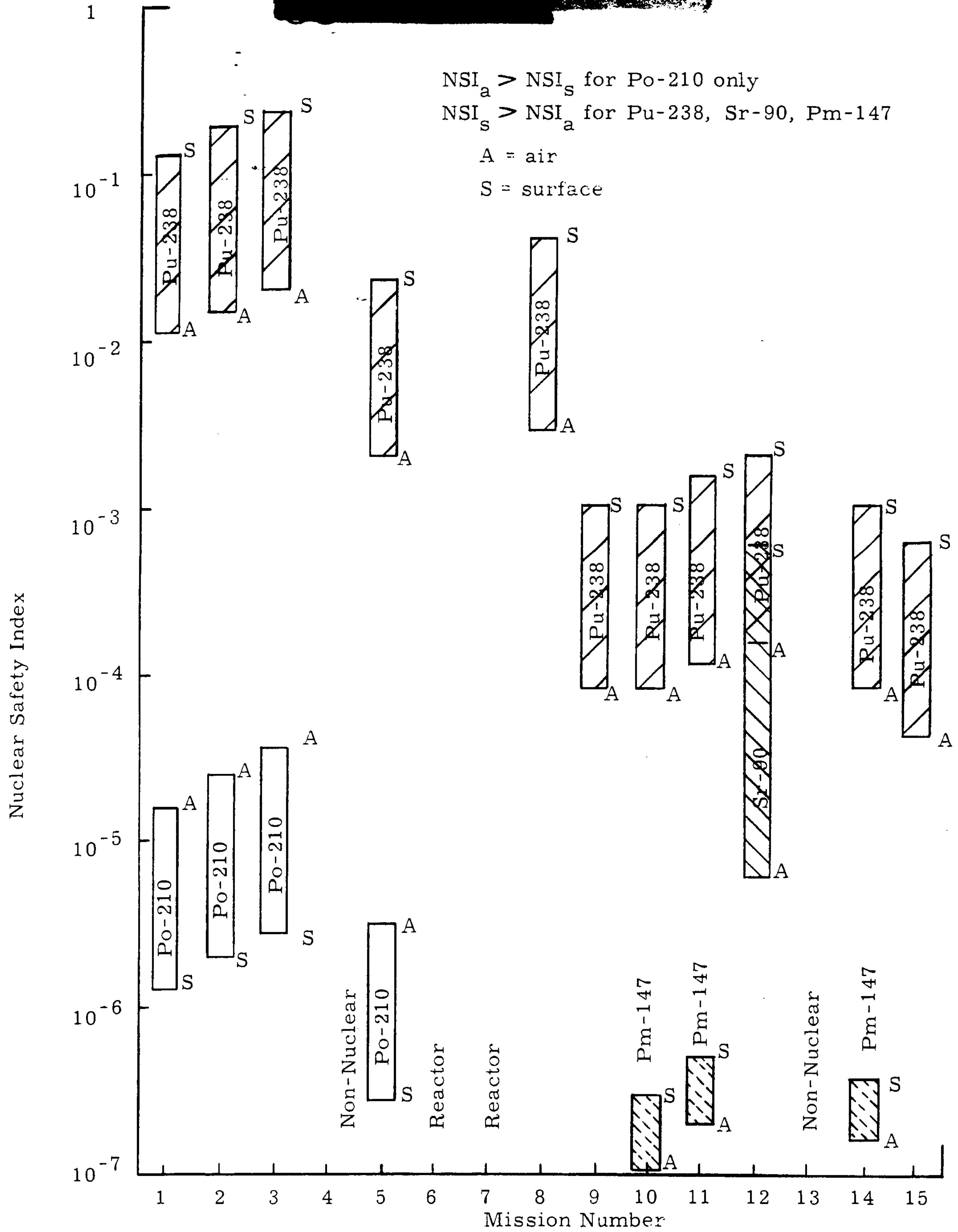

Figure 6. Nuclear Safety Indices for Both Air and Surface Contamination Assuming Burnup in the Mesosphere. (Based on actual power level of each mission and the analysis of Appendix A & B)

[redacted]

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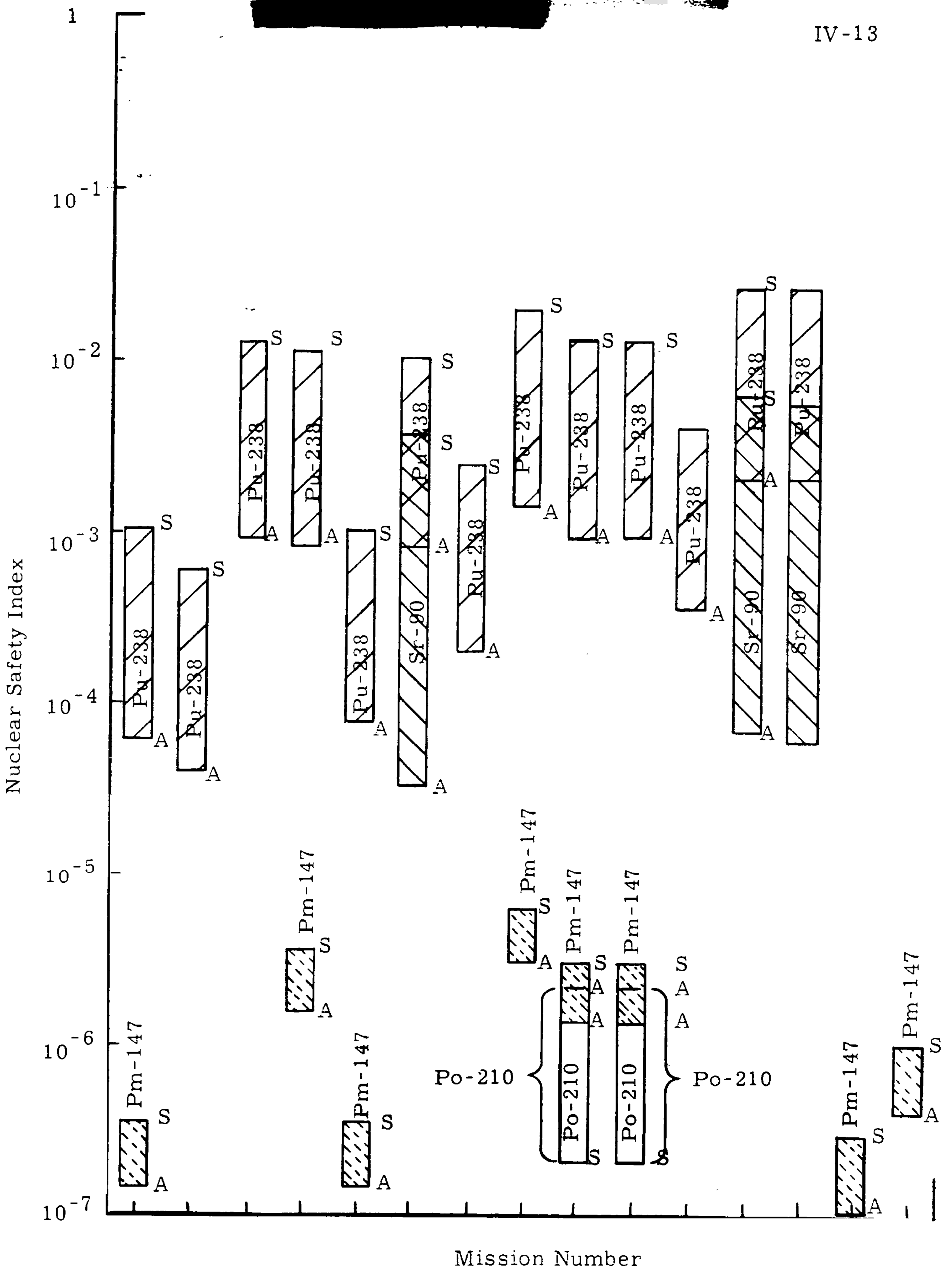

Figure 6a. (Cont'd)

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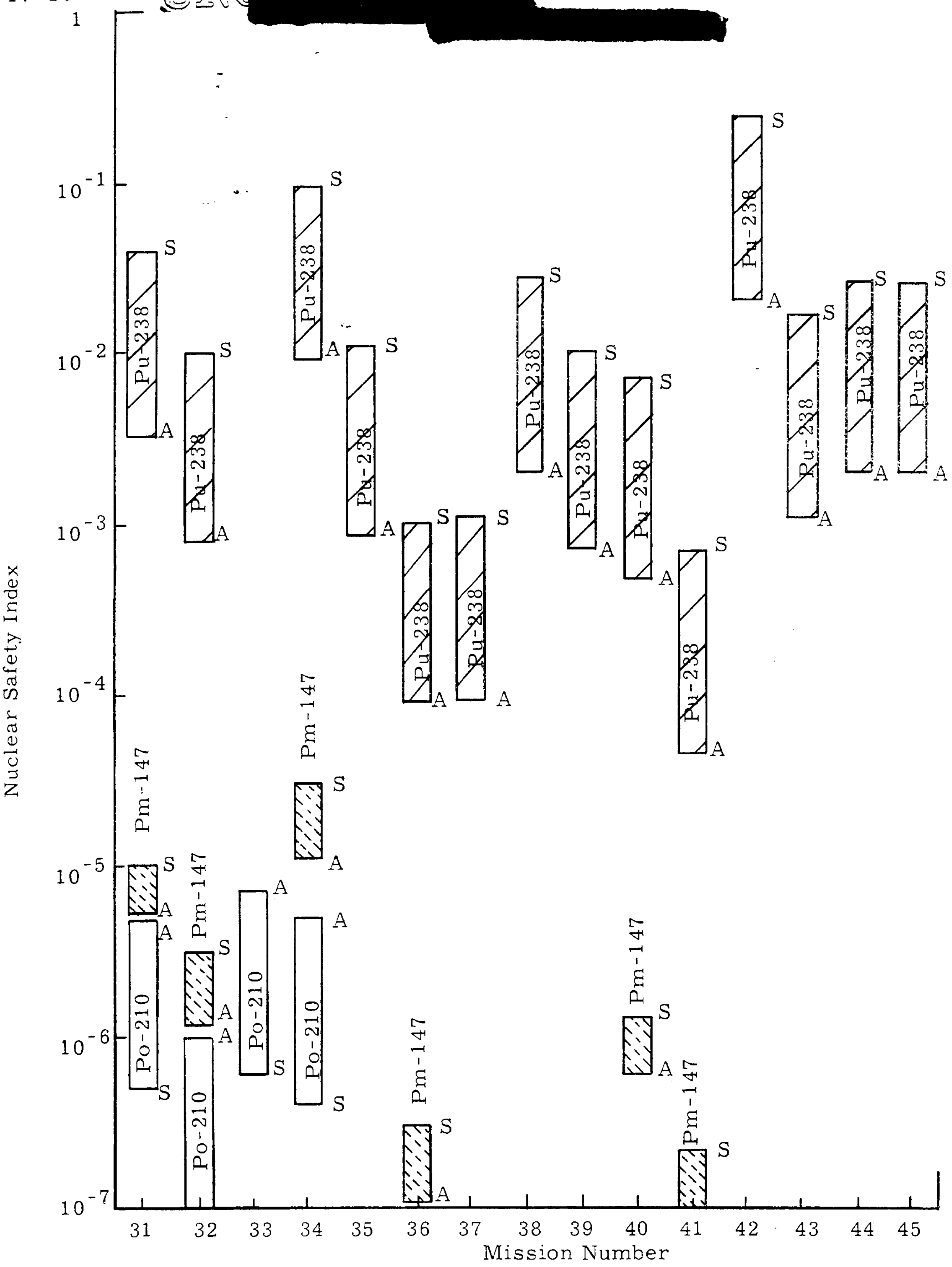

Figure 6b. (Cont'd)

[redacted]

[redacted]

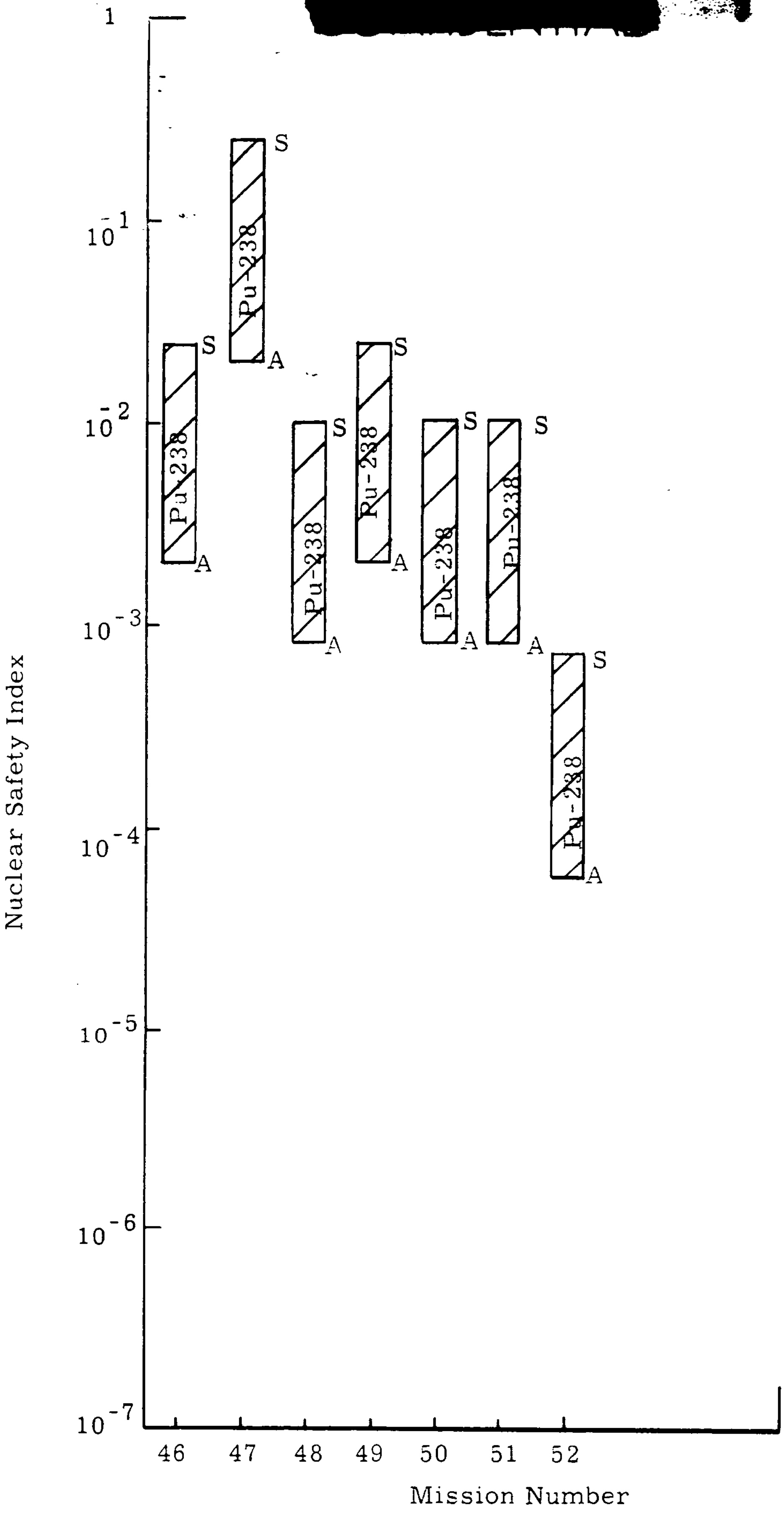

Figure 6c. (Cont'd)

[redacted]

Table XI

Maximum Permissible Surface Contamination Ranges Used in This Study (Reference 10)

Isotope	168-hr MPC Minimum[a] (c/cc)	$t_{1/2}$ (yr)	No Restrictions (curie/m^2)	Range III Minor Restrictions: No evacuation Minor decontamination some crops destroyed (curie/m^2)	Range II Moderate Restrictions Short-term evacuation Moderate decontamina. No crops for ≥ 1 yr (curie/m^2)	Range I Severe Restrictions Long-term evacuation Total loss of value No crops for ≥ 5 yr (curie/m^2)
Co-60	3×10^{-9}	5.2	$<3 \times 10^{-5}$	3×10^{-5} to 3.4×10^{-4}	3.4×10^{-4} to 6×10^{-3}	$>6 \times 10^{-3}$
Sr-90[a]	10^{-10}	27.7	$<1 \times 10^{-7}$	1×10^{-7} to 1×10^{-6}	1×10^{-6} to 1.1×10^{-5}	$>1.1 \times 10^{-5}$
Sb-124	7×10^{-9}	0.167	$<7 \times 10^{-5}$	7×10^{-5} to 4.6×10^{-2}	$>4.6 \times 10^{-2}$	-
I-131[a]	3×10^{-9}	0.022	$<6.5 \times 10^{-7}$	$>6.5 \times 10^{-7}$	-	-
Cs-137[a]	5×10^{-9}	26.6	$<5 \times 10^{-6}$	5×10^{-6} to 5×10^{-5}	5×10^{-5} to 5×10^{-4}	$>5 \times 10^{-4}$
Ce-144	2×10^{-9}	0.78	$<2 \times 10^{-5}$	2×10^{-5} to 1.4×10^{-3}	1.4×10^{-3} to 1.2	>1.2
Pm-147	2×10^{-8}	2.6	$<2 \times 10^{-4}$	2×10^{-4} to 2.7×10^{-3}	2.7×10^{-3} to 9.6×10^{-2}	$>9.6 \times 10^{-2}$
Tm-170	1×10^{-8}	0.354	$<1 \times 10^{-4}$	1×10^{-4} to 7×10^{-2}	$>7 \times 10^{-2}$	-
Ir-192	9×10^{-9}	0.203	$<9 \times 10^{-5}$	9×10^{-5} to 2.7×10^{-2}	$>2.7 \times 10^{-2}$	-
Po-210	7×10^{-11}	0.378	$<7 \times 10^{-7}$	7×10^{-7} to 4.5×10^{-5}	4.5×10^{-5} to 3.6	>3.6
U-233[b]	1.4×10^{-11}	7.7×10^{4}	$<1.4 \times 10^{-7}$	1.4×10^{-7} x 4×10^{-6}	1.4×10^{-6} to 1.4×10^{-5}	$>1.4 \times 10^{-5}$
U-235	4×10^{-11}	7×10^{8}	$<4 \times 10^{-7}$	4×10^{-7} to 4×10^{-6}	4×10^{-6} to 4×10^{-5}	$>4 \times 10^{-5}$
Pu-238	7×10^{-13}	86.4	$<7 \times 10^{-9}$	7×10^{-8} to 7×10^{-8}	7×10^{-8} to 7×10^{-7}	$>7 \times 10^{-7}$
Pu-239	6×10^{-13}	2.4×10^{4}	$<6 \times 10^{-9}$	6×10^{-9} to 6×10^{-8}	6×10^{-8} to 6×10^{-7}	$>6 \times 10^{-7}$
Cm-242	4×10^{-11}	0.444	$<4 \times 10^{-7}$	4×10^{-7} to 1.9×10^{-5}	1.9×10^{-5} to 4.5×10^{-1}	$>4.5 \times 10^{-1}$
Cm-244	3×10^{-12}	17.9	$<3 \times 10^{-8}$	3×10^{-8} to 3×10^{-7}	3×10^{-7} to 4×10^{-6}	$>4 \times 10^{-6}$

[a] Contamination levels for these isotopes based on dairy farming. Limits for all other isotopes based on resuspension and inhalation.

[b] Based on U-233 containing 550 ppm U-232. Specific activity = 2×10^{-2} curie/g U-233 + U-232.

[redacted]

Following the Windscale accident of October 10, 1957, studies were made on I-131, Sr-90 and Cs-137 content in milk as a function of the contamination of the surrounding area (Reference 10). Based on dietary recommendations by the English Medical Research Council, maximum permissible contamination levels were made. A significant amount of research has been done in standardizing the food uptake cycle, particularly with respect to strontium-90.

The contamination level basis for non-dairy isotopes is handled differently. In the case of these isotopes, resuspension factors have been used in combination with the maximum permissible concentrations in air, MPC_a. Using Table XI, it can be seen that the lower contamination levels for Range III differ with the MPC_a values by a factor of 10^{-4}. This factor of 10^{-4} is the resuspension factor and is related to the amount of activity increase in the air due to wind pickup of surface particles. Values for Ranges I and II are determined by the pattern set by Sr-90 and Cs-137.

Experiments have indicated a resuspension factor of 4×10^{-5} units per cubic meter in air per 1 unit per square meter of surface activity for a dusty operation in a confined space. Other experiments have indicated resuspension factors in the range of 2×10^{-3} to 4×10^{-5} (Reference 11). Oak Ridge National Laboratories have conducted studies that indicate values ranging from 1.3×10^{-5} to 1.7×10^{-7} (Reference 12). Based on wind pickup of particles on various types of ground, resuspension factors of 7×10^{-2} (1.5 μ particles) to 8×10^{-5} (14 μ particles) can be predicted in a 5 m/sec wind.

For the purpose of this study, a resuspension factor of 10^{-4} (curies/m^3 / curies/m^2) has been selected as representing the mean value. It is assumed that Range III (Table XI) represents the action that would be taken at a contamination level that would result in the air being above MPC_a. As was mentioned previously, the lower limit chosen for Range III is $10^4 \times MPC_a^{168}$, curies/m^2, for all isotopes other than Sr-90, I-131, and Cs-137. In choosing ranges for longer restriction, radioactive decay must be considered in addition to normal availability decrease due to washout or absorption in the soil. The following equations were used to calculate the lower limits for Range II and Range I, and are consistent with the ratios used for Sr-90, I-131, and Cs-137:

$$\frac{\text{Range II}}{\text{Range III}} = 10 \times 2^{(1/t_{1/2})}$$

$$\frac{\text{Range I}}{\text{Range III}} = 100 \times 2^{(5/t_{1/2})}$$

[redacted]

Referring back to Figures 3, 4 and 5, it can be seen that NSI values based on surface contamination are more critical than those based on air contamination. This is particularly true of the long half-life isotopes such as Pu-238 and Sr-90 which exhibit significant surface activity buildup. This seems to indicate that more extensive research into the standardization of maximum permissible surface contaminations is advisable. Studies should be carried beyond the experimental levels reached to date and should strive for the development of empirical relationships which may well prove to be at least as significant as MPC_a values.

The limits on maximum inventory that can be tolerated in the atmosphere from high altitude and stratospheric burnup as shown in this analysis was based upon the MPC data and calculational procedures listed in Appendices A and B, and Reference 10. However, at the time of publication of this study, new calculational techniques for fallout predictions are being developed and worked with (see AI-65-144, 9/20/65). These techniques tend to increase the atmospheric activity predicted by the analysis used herein. The validity of these new methods has not been corroborated at this writing and AI-65-144 gives no details as to the calculative techniques used. However, preliminary comparisons of the data presented in this report with new techniques indicates a possible range of application of 1 to 2 orders of magnitude lower than acceptable power limits shown in Figures 3 - 5. Therefore, conclusions on atmospheric burnup limits are given as a range of values, felt by the authors to represent safe releases within present state of uncertainty of the analytical techniques.

2. Low Altitude Releases

The occurrence of partial burnup, releasing radioactive material in a low atmosphere region, is a definite possibility in any launch abort of a nuclear system that is designed for complete burnup upon reentry. The degree of hazard created in a partial burnup situation is extremely difficult to analyze because of its random nature, the interaction of the launch vehicle and the power system upon release and subsequent burnup, the radionuclide and fuel matrix being released, and the location of the release. The approach in analyzing the degree of hazard associated with a low altitude injection of radioactive material was to determine the atmospheric concentration and surface contamination resulting from releases in the stratosphere. The resulting contamination levels were then compared with maximum permissible values to formulate both air and surface Nuclear Safety Indices, NSI's, as a function of radioactive material injected into the stratosphere. The results are presented in Figures 7 through 9. Details concerning the method of analysis are presented in Appendix B.

It can be seen that stratospheric injection of the fuel assuming 1 micron size and good circumferential dispersion due to the jet stream, still yields a limit of several hundred Kw(t) ($\text{NSI}_a \approx 0.10$) as a maximum for Pu-238 systems which are most critical. Assuming a range of uncertainty of two orders of magnitude for the NSI_a based on calculational procedures used,

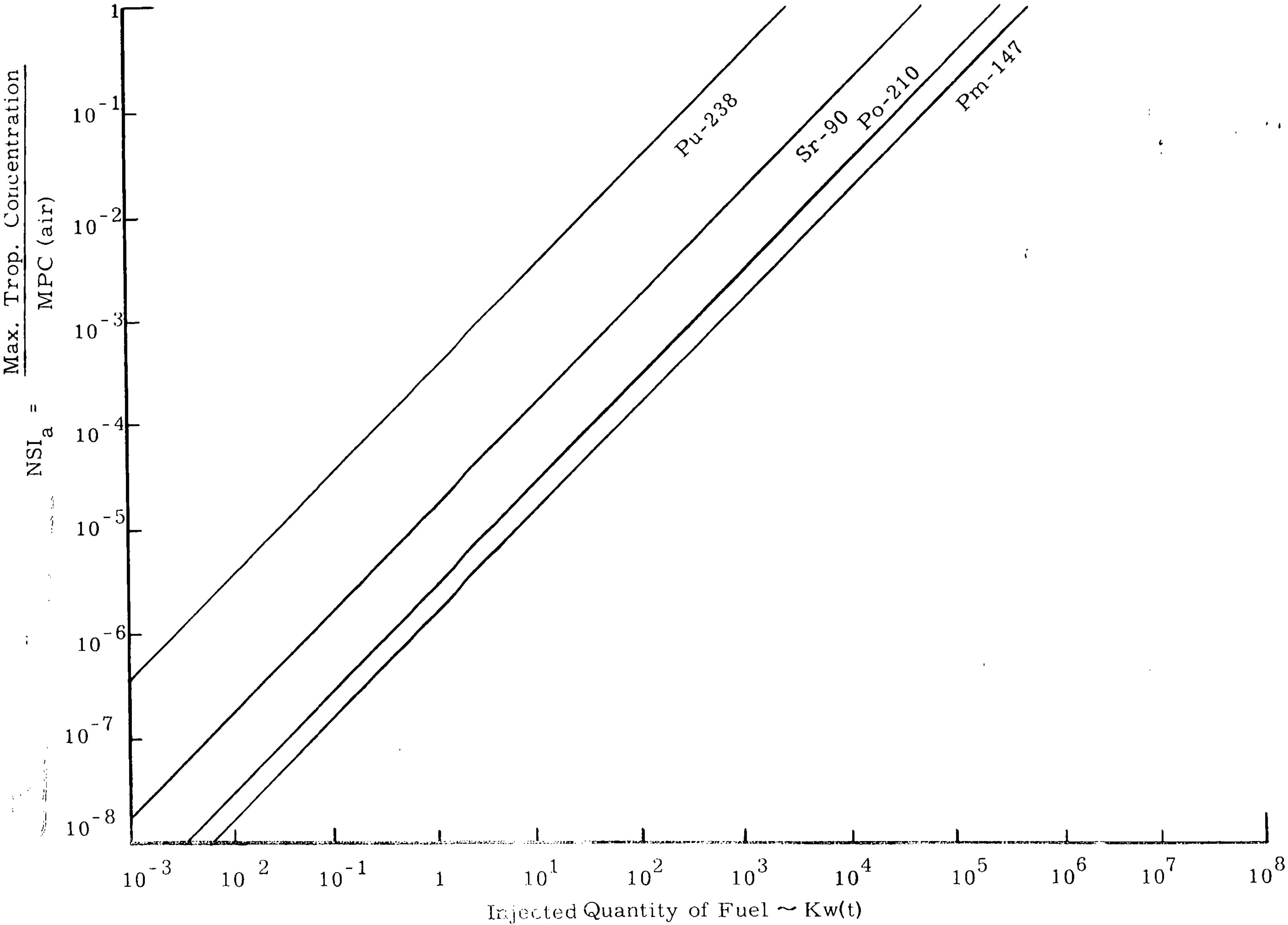

Figure 7. Nuclear Safety Index for Tropospheric Air Contamination Between 20° and 70° Latitude for a Stratospheric 1μ Release

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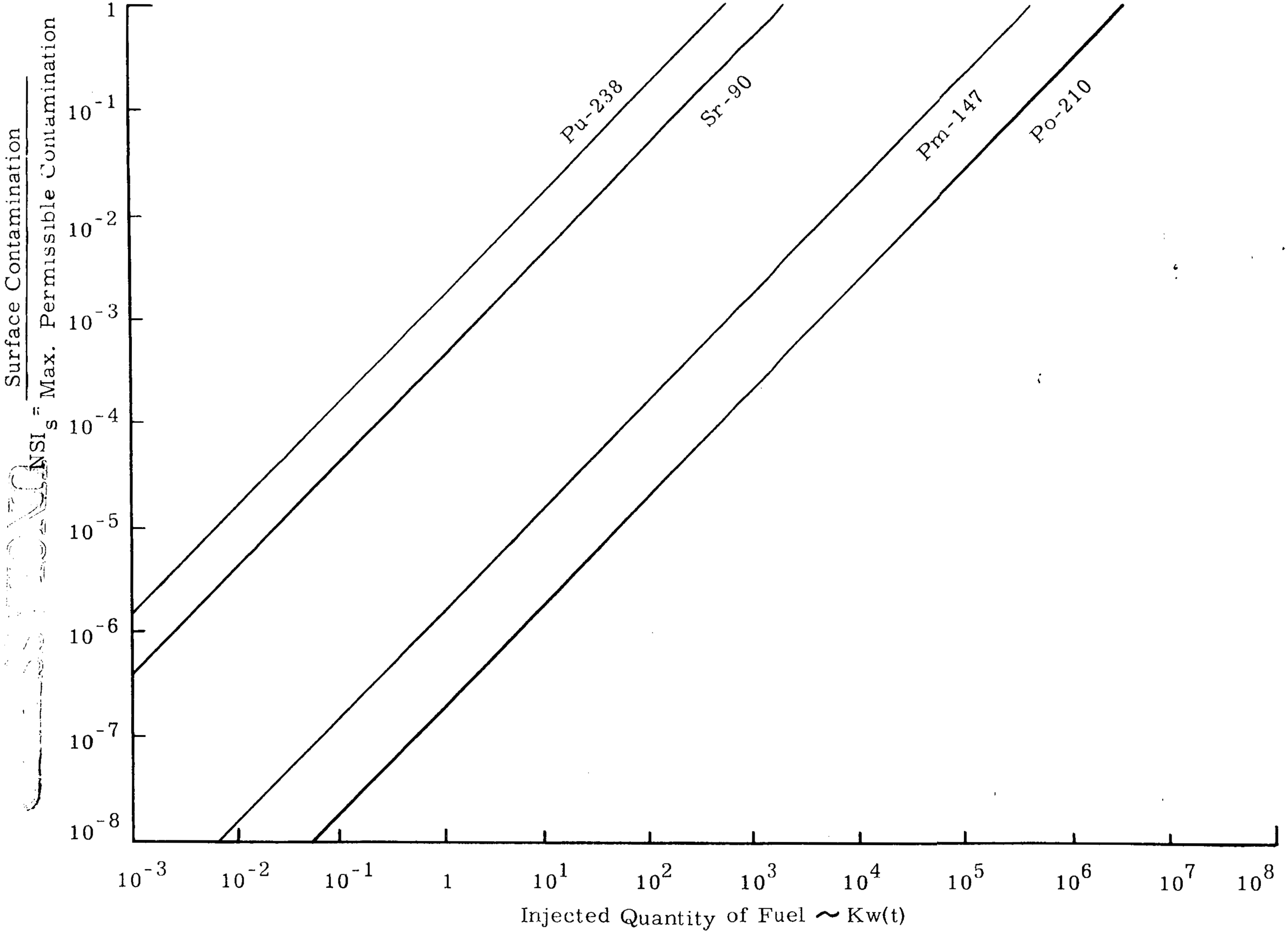

Figure 8. Nuclear Safety Index for Surface Contamination (Minor Restrictions) Between 20 and 70° Latitude for a Stratospheric 1μ Release

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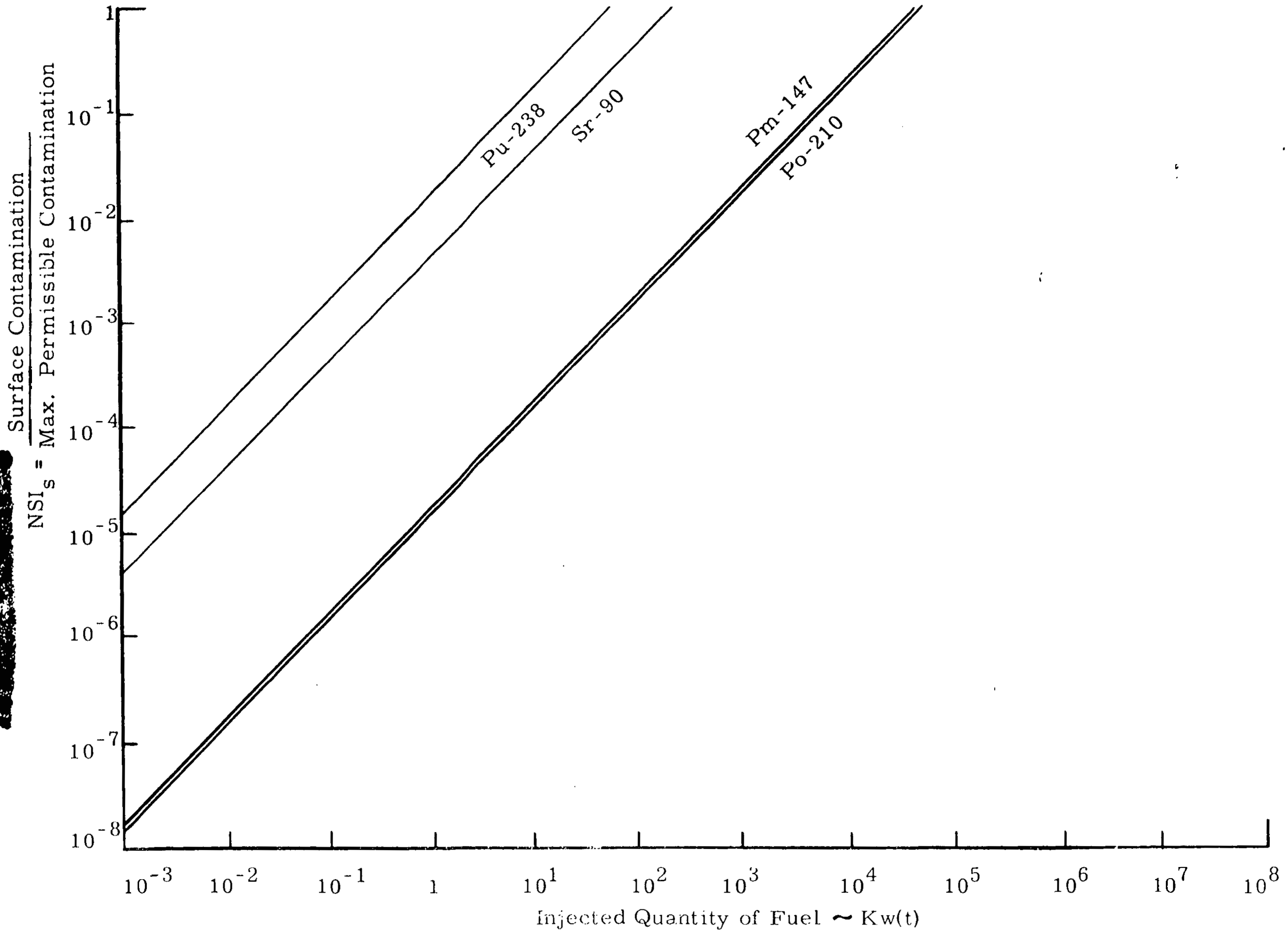

Figure 9. Nuclear Safety Index for Surface Contamination (No Restrictions) Between 20 and 70° Latitude for a Stratospheric 1 μ Release

[redacted]

this gives a minimum limit of approximately 2 Kw(t). The maximum values represent a power range of from 15-50 Kw(e) depending on conversion system used for the most optimistic case. Surface contamination is still the most serious considerstion for a stratospheric release and this yields a limit of 4-16 Kw(e) for an $NSI_s = 0.10$ (no restrictions). Pu-238 systems are still most critical. This is close to the range of power requirements of Extended Apollo, MORL, and manned Mars missions.

As would be expected, ground and ascent releases are the most serious. The data in Figures 10 and 11 were coupled parametrically relative to contaminated area. This can be related back to altitude of release during the early ascent phase if an abort which breaches the capsule should occur. Since the character of the released fuel form is unknown (i. e., metal, microspheres, etc.) a parametric contamination plot was presented (Figures 10 and 11) to allow assessment of the ground hazard under a series of release accidents.

a. Ascent Trajectory Considerations. The significance of the low altitude releases discussed above has to be evaluated in the light of choosing the original launch trajectory, because only through proper choice of launch trajectory can the hazards from an ascent accident be minimized.

For anticipated ascent trajectories from AMR or PMR, an early ascent abort occurring at relatively low velocities and altitudes will result in the system landing intact in the Atlantic or Pacific Ocean because aerodynamic heating will be practically non-existent. However, once the spacecraft attains orbital altitude and velocity systems designed for reentry burnup will be aerodynamically consumed upon reentry. Between the intact and the burnup area there is a partial burnup zone, a region where reentry can produce a breached fuel containment structure and the typical attendant fallout patterns shown in Figures 10 and 11. This condition is only acceptable if it occurs over a deep ocean area away from land masses. Also intact reentry systems pose similar problems in that the source may impact down range unshielded (AMR-Africa or PMR-Antarctica) creating a radiological hazard or undesirable political situation.

These conditions may be minimized through the selection of an ascent trajectory which achieves orbital insertion early in the launch phase so as to provide minimum ground range for ballistic reentry should aborts occur close to orbital insertion. Ascent profiles from AMR and PMR were investigated to determine the influence of the trajectory on nuclear safety.

The manned Apollo mission was selected as being representative of a typical AMR mission. Figure 12 illustrates the trajectory profile from lift-off through orbital insertion. The launch is such that the vehicle rises almost vertically until the planned orbital altitude is obtained. It then tilts over and accelerates until such a time that the desired orbital velocity is achieved. This type launch is characterized by a relatively short ground range which permits aborts to occur late in the launch phase and still achieve deep ocean burial or,

[redacted]

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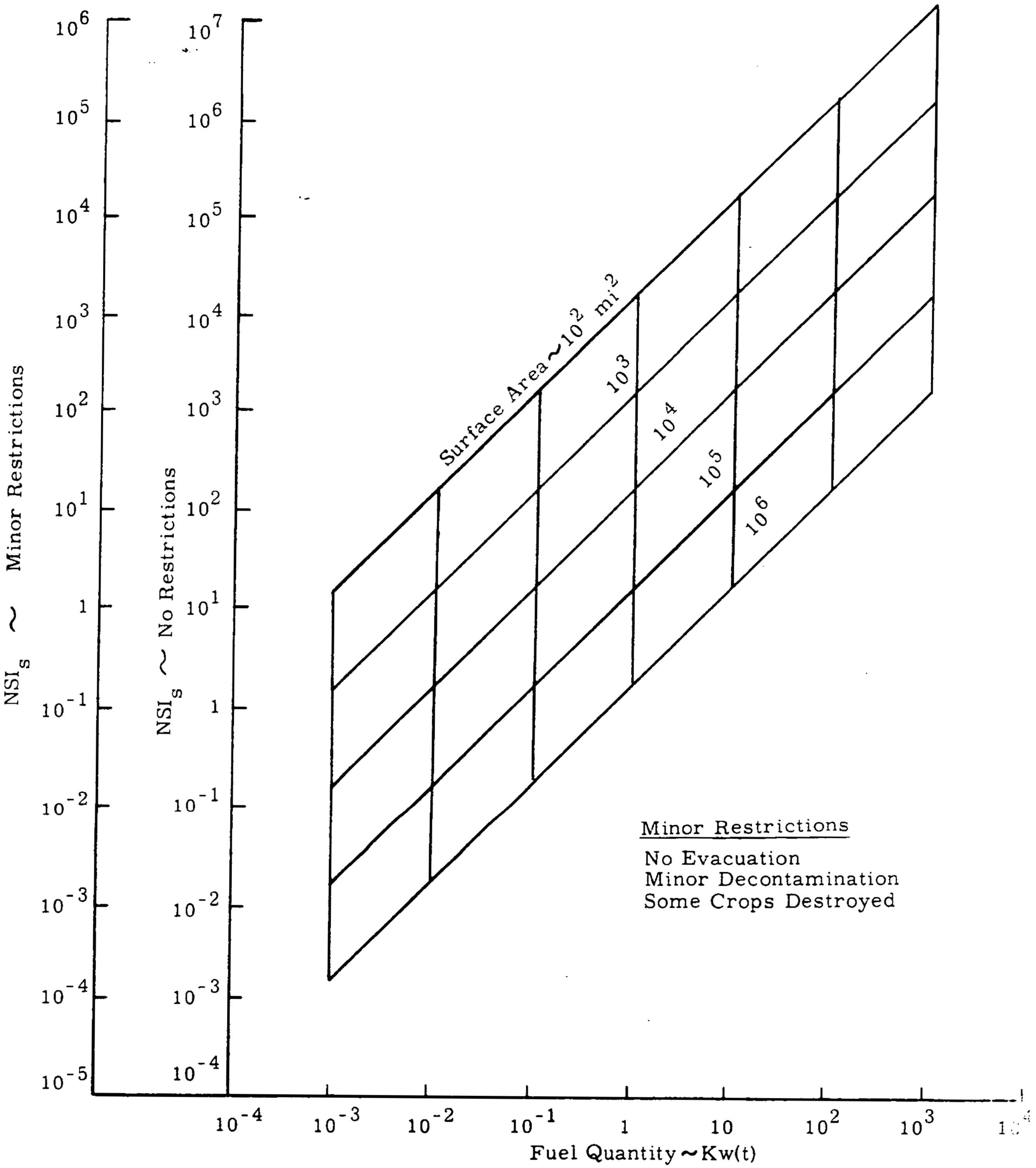


Figure 10. Nuclear Safety Index For Earth Surface Contamination Pu-238 Ground Release

[redacted]

[redacted]

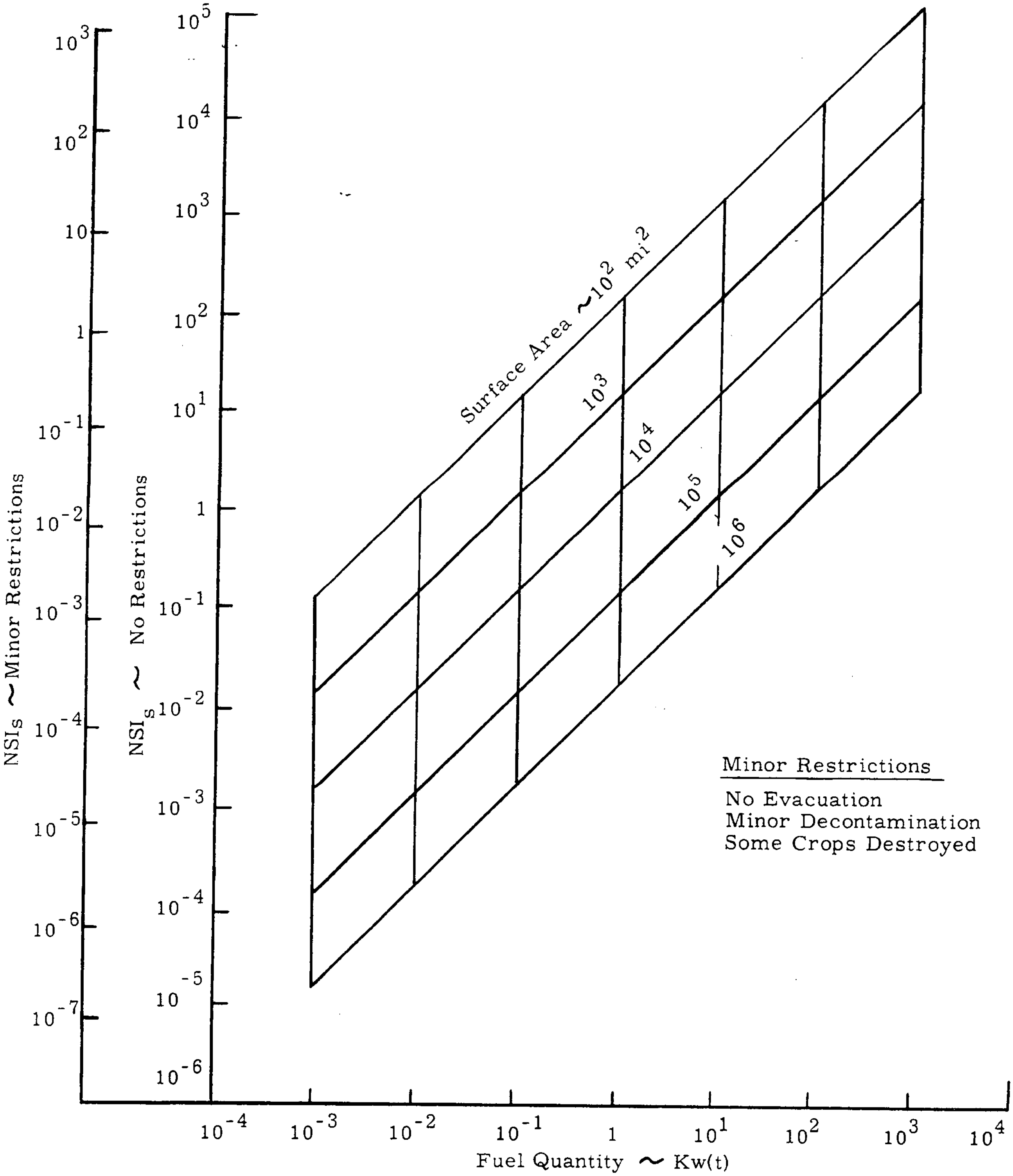


Figure 11. Nuclear Safety Index for Earth Surface Contamination Po-210 Ground Release

[redacted]

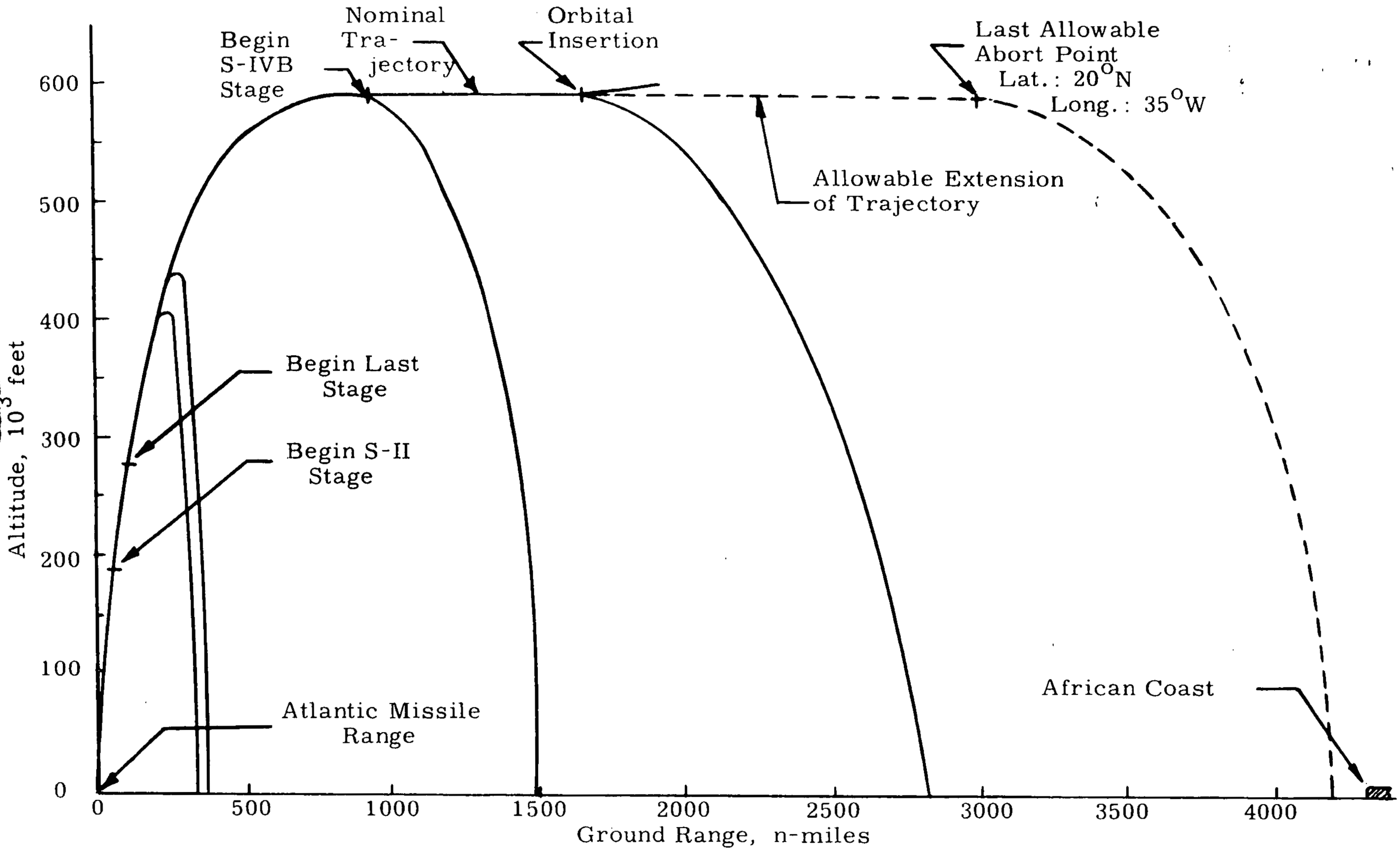


Figure 12. Typical Profile of Atlantic Missile Range Launch and Abort Characteristics*

*Based on Apollo/Saturn Mission - Reference "Apollo Mission Planning Task Force", Volume 1, Mission Description

[redacted]

for burnup systems it confines the partial burnup region to deep ocean. In an abort situation, as shown, the last possible abort point for a deep ocean impact is displaced toward the African Coast over 1000 nautical miles from the planned orbital injection point, assuming a ballistic type of entry. Furthermore, if the ground range can be reduced even more for other types of missions the probability of deep ocean burial under an abort situtation increases.

Figure 13 illustrates an ascent trajectory from PMR for a medium altitude communication satellite. It can be seen that, unlike the Apollo launch, this trajectory does not allow a displacement of the orbital insertion point towards the land mass. This is due to the relatively short ground range for this launch which does not allow aborts to occur late in the launch and increases the partial burnup region over deep ocean. The fact that an abort point, that may cause land impact, exists in the trajectory before orbital insertion is exactly what should be avoided. Figure 13 therefore shows an undesirable trajectory while Figure 12 shows a desirable one for nuclear safety considerations.

In general, it may be concluded that the desired characteristics for a safe launch trajectory are:

(1) Launch over ocean areas.

(2) Choice of launch course to maximize open water.

(3) Vertical ascent to desired altitude to decrease trajectory ground range.

(4) High energy upper stages to yield orbital insertion early in the launch and consequent increase in water impact probability if an ascent abort occurs.

Hence, it is clear that nuclear safety problems emanating from the ascent trajectory, independent of the design criteria (intact or burnup), may be minimized through trajectory selection.

b. Other Considerations. One last consideration in evaluating the safety of the aerospace burnup criterion is the effect of particle size. It was mentioned earlier that particles of 1 micron size and smaller exhibit mean residence times in the upper atmosphere of the order of seven years. In the case of short half-life isotopes, this is sufficient time to render the activity safely below permissible levels. However, if the particles were larger, residence times would be less and the above statement may prove to be erroneous.

When considering the displacement of fallout from the point of initial release, size is also a very important factor. Figure 14 shows the effect of particle size on the ratio of vertical to horizontal displacement. Earlier, it was discussed how the 1 micron sized particles exhibited complete dispersion resulting in significant lowering of activity levels at the earth's surface. Larger particles, of the order of several hundred microns, will not undergo the same complete dispersion as the indexed 1 micron particles and subsequently could

[redacted]

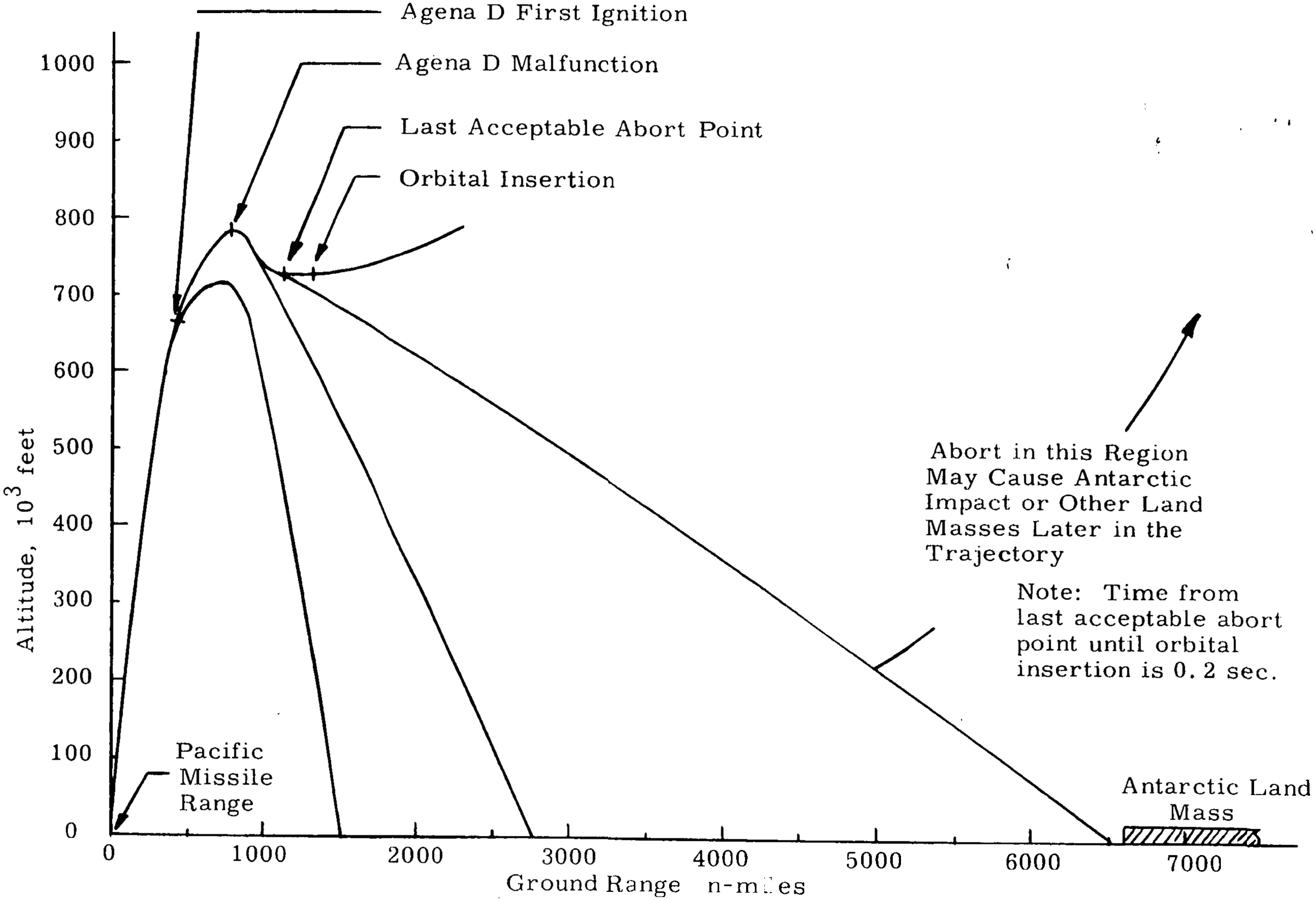
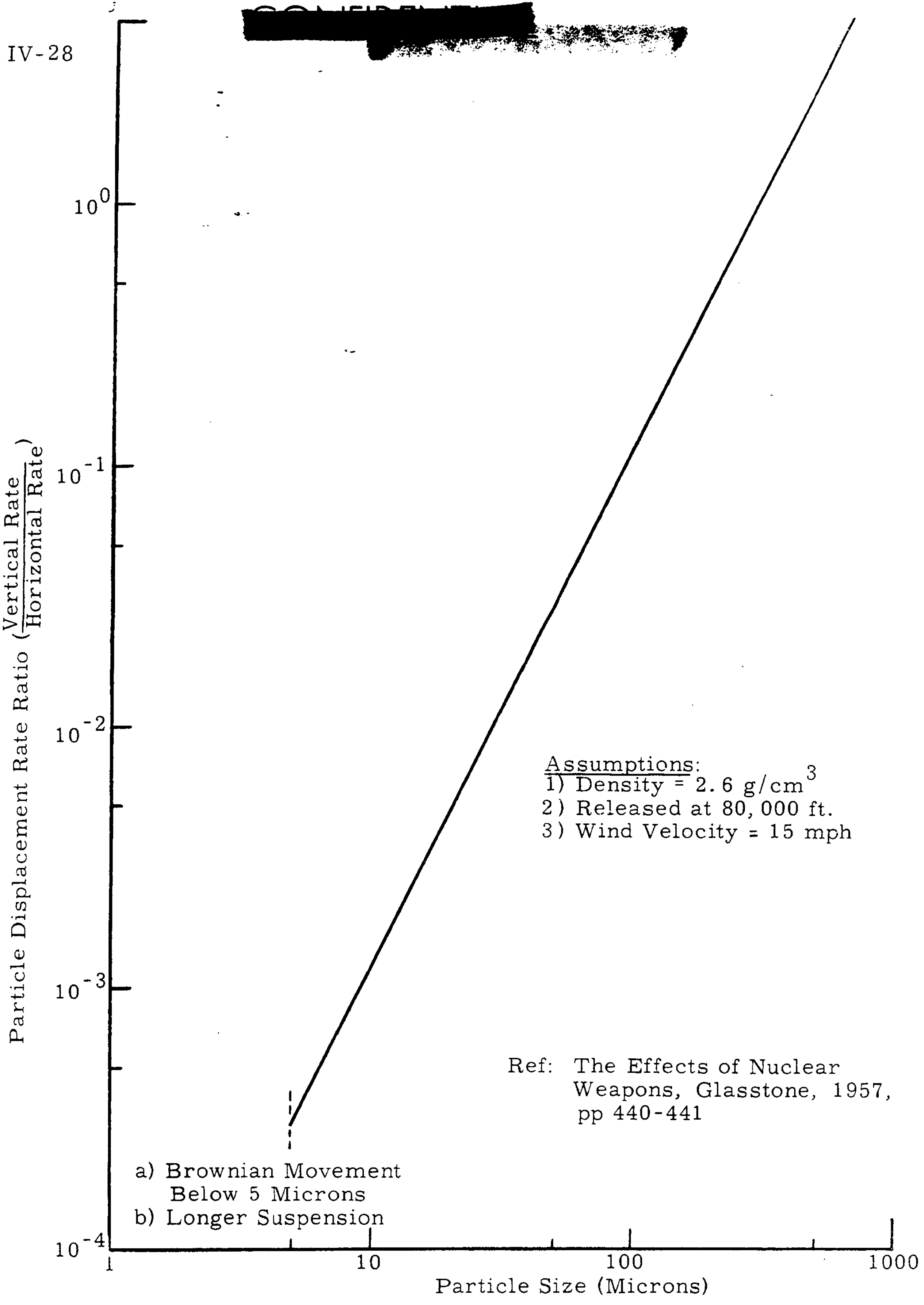


Figure 13. Typical Profile of Pacific Missile Range Launch and Abort Characteristics*

* Based on SNAP-17A - Reference "SNAP-17A Safety Phase I Final Summary Report", MND-3307-33-1, August 1964 (CRD)

[redacted]



Assumptions:
1) Density = 2.6 g/cm^3
2) Released at 80,000 ft.
3) Wind Velocity = 15 mph

Ref: The Effects of Nuclear Weapons, Glasstone, 1957, pp 440-441

a) Brownian Movement Below 5 Microns
b) Longer Suspension

Figure 14. Fallout Displacement Rate, Vertical-to-Horizontal Ratio for Various Particle Sizes.

[redacted]

deposit the same amount of activity over a smaller area, thereby increasing concentration. This is exactly the case in the proposed safety philosophy for SNAP-25. This microsphere release technique is discussed in detail in Section IV-A-4. Although ground concentrations will always rise with this technique, the hazard may still be acceptable as a function of radioisotope, fuel form, and quantity. Figure 15 shows that, within the most optimistic and pessimistic limits as defined by present fallout analyses and MPC assumptions, the consequences of burnup are acceptable for Po-210 and Pm-147 and marginal, depending on fuel form character, for Pu-238 and Sr-90. All of the missions considered in Figure 15 are for the maximum inventory expected to be flown. If the optimistic Nuclear Safety Indices are assumed, which are backed up by the calculations in this report, then no serious problems from high altitude burnup should occur.

3. The Potential of Satisfying the Aerospace Nuclear Safety Burnup Criterion for the Random Reentry of Reactor and Isotope SNAP Systems

It has been established that if reentering reactor or isotope systems could be reduced to particles of one micron or less above 100,000 feet, the resulting fallout would constitute an acceptable hazard, relative to the MPC, in all but the most powerful systems contemplated for the 1980 time period and beyond.

The problem remains, however, to investigate the possibility of actually achieving the desired particle sizes during typical reentries of reactor and isotopic SNAP systems. This, rather than the prediction of general system and component meltdown altitudes, is the most critical problem affecting the satisfaction of the general aerospace safety burnup criterion.

a. Satellite Decay. Fairly good estimates of gross reentry heating effects on the macroscopic systems components can be generated by present analytical techniques (References 14, 15 and 16). However, very little analysis and data exists on what the spectrum of particle diameters and their character is when they are released from the melted fuel form; what effects release into the main body wake have, and what heat transfer and combustion processes take place to reduce them to the critical one micron size necessary to satisfy the aerospace safety burnup criterion.

Figure 16 represents a typical reactor system reentry energy and temperature profile. The sequence of events on the reentry input energy curve are those predicted for the reentering SNAP 2/10A reactor (Reference 17). Superimposed on the energy profile are the temperature profiles for spinning and tumbling fuel elements after their release from the core. In addition, the melt temperatures of various fuel element materials which are being considered for present and future reactor systems are shown, as well as the effects of varying fuel element diameter and emissivity.

[redacted]

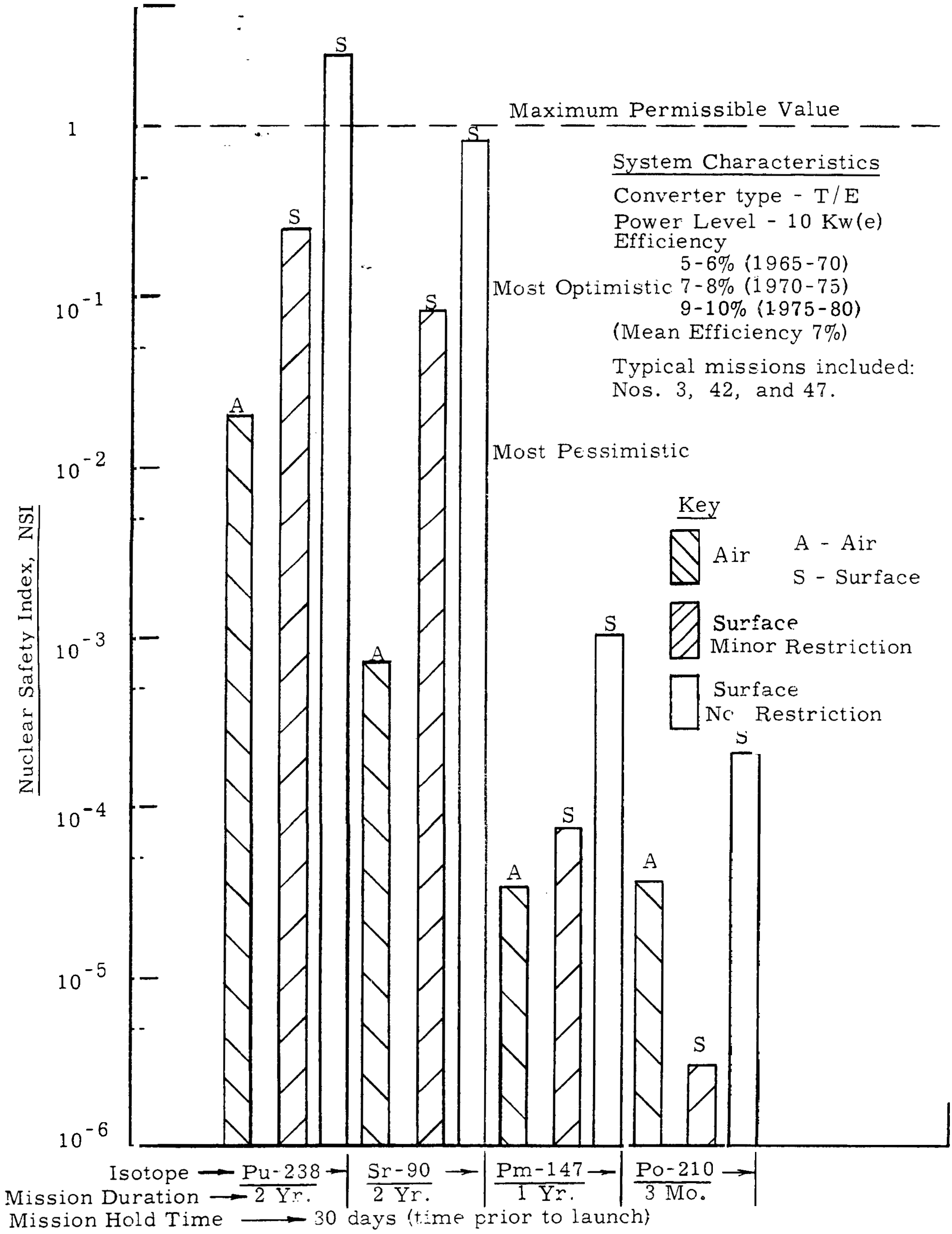
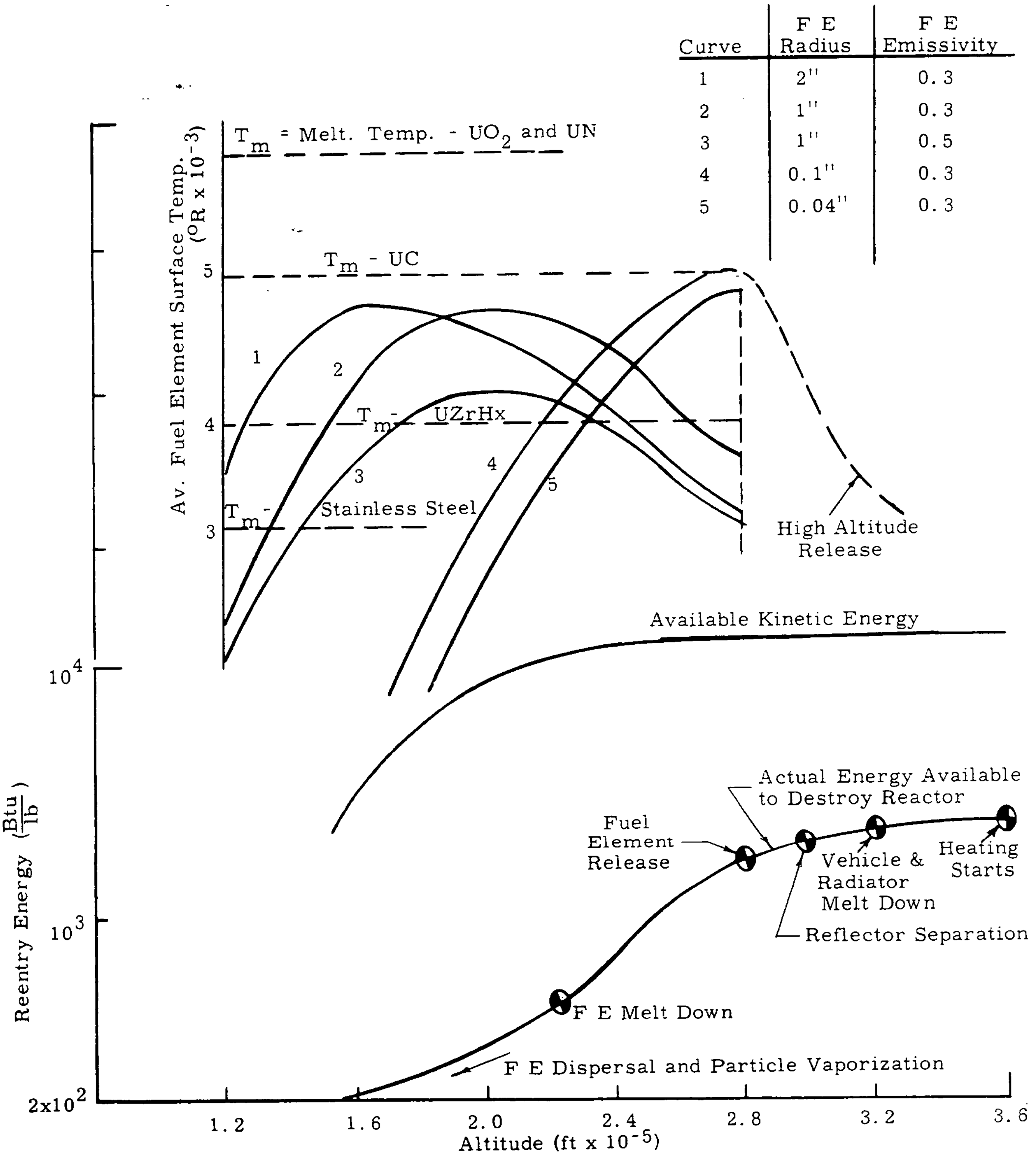


Figure 15. Nuclear Saftey Index for Large Radioisotopic Systems - 10 Kw(e) for a Mesospheric 1μ Release

[redacted]

[redacted]



Curve	F E Radius	F E Emissivity
1	2''	0.3
2	1''	0.3
3	1''	0.5
4	0.1''	0.3
5	0.04''	0.3

Figure 16. Typical Energy and Temp. Profiles of SNAP Reactors and Fuel Elements During Random Satellite Reentry (Reference 17).

[redacted]

[redacted]

It is seen that, for typical fuel elements with radii of approximately one inch, the average surface temperatures after release will exceed the meltdown temperatures of $UZrH_x$. The temperature peaks at approximately 200,000 feet and will stay above the melt temperature of the ZrH_x fuel element for approximately 100 seconds, representing the fall from 220,000 to 180,000 feet and a surface emissivity of $\epsilon = 0.5$. This indicates good prospects for meltdown of the ZrH_x fuel element.

However, if the reactor were designed with UC, UO_2 or UN fuel elements, indicative of the advanced high temperature SNAP reactors, meltdown temperatures could not be reached. Even varying the size of the fuel elements down to thin wire rod bundles does not seem to yield enough heat flux for meltdown. Therefore, special thermal countermeasures, which are integral with the fuel element are necessary to to add sufficient energy for meltdown and vaporization. Simple mechanical destruct devices (explosives) are not enough.

The same type of analysis was completed for typical SNAP isotope systems during random reentry from a decaying satellite orbit. Average total heat flux and capsule surface temperature versus altitude are shown in Figure 17. These curves were compiled for Pu-238 and SrF_2 fueled 25 watt(electric)generators representing present and possible future fuel forms of interest. In the most conservative case of a tumbling and spinning fuel capsule (minimum heat flux during reentry), both fuel forms will lose their containment and be released close to the molten state above 230,000 feet. Therefore, the best possible conditions for fuel element breakup seem within reasonable attainment with isotope systems. However, this may not guarantee attainment of the size levels of interest as is shown below.

After determining the probability of meltdown of typical reactor and isotope fuel forms the next step is to evaluate the final particle sizes to which these debris could be reduced by the remaining reentry energy.

It was shown that typical meltdown conditions for both SNAP reactor and isotope systems would occur at about 200,000 feet and at velocities of approximately 20,000 to 23,000 ft/sec based on random reentry from a decayed satellite orbit. These conditions can be then assumed as those which various sized particles of reactor fuel element material or isotope fuel form first see upon release from the melted main fuel form.

A detailed analysis of the potential of various radioisotope fuel forms to be reduced to submicron size has been completed by Hittman Associates and is reported in Reference 18. The discussion that immediately follows draws heavily upon that report.

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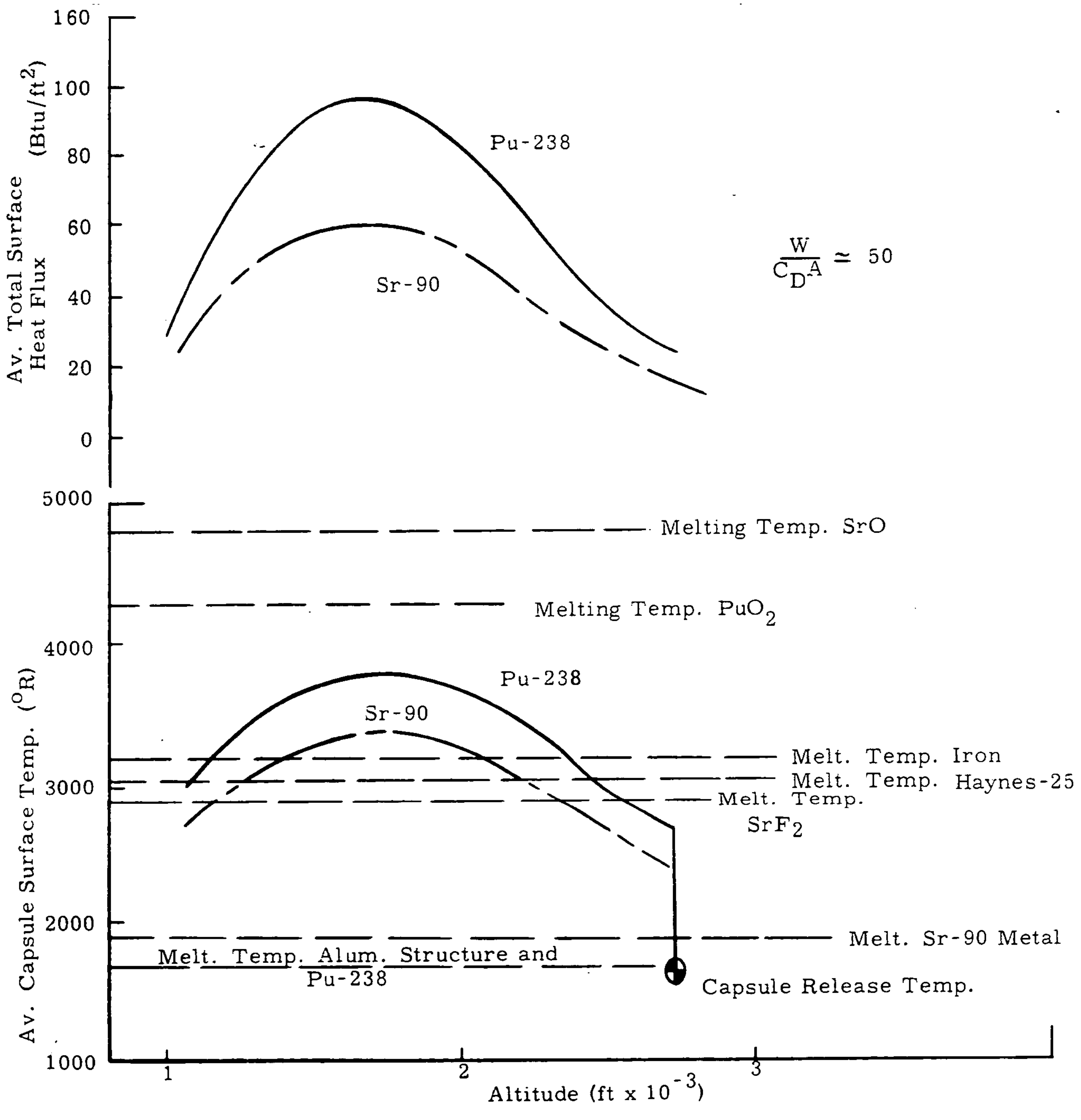

Figure 17. Typical Energy and Temp. Profiles of Sr-90 and Pu-238 Fuel Capsules During a Typical Random Reentry of a Radioisotope Generator and Satellite System.

[redacted]

[redacted]

The reentry destruction potential of any fuel form material is a function of many complex phenomena acting simultaneously after release of the fuel form from the capsule. As explained in Reference 18, these are aerodynamic instabilities, thermodynamic instabilities, boiling instabilities, surface oxidation and basic surface vaporization due to aerodynamic heating.

In general, there are two main mechanisms for particle reduction, mechanical breakup due to the aerodynamic pressure and aerodynamic heating of the resulting debris. Such thermodynamic assets as superheat upon release yielding flashing are added benefits which the fuel form designer should try to employ but are difficult to come by in all the fuel forms of interest except Sr and Po.

Figure 18 shows two different characters of breakup which could be expected from typical molten fuel form debris upon release from the capsule. The chaotic breakup has the potential of reducing the melt to a fine spray, whose average drop size is 1 - 10μ immediately upon release, leaving maximum time for the aerodynamic heating to complete the reduction to 1μ or below. The possibility of inducing this breakup is a function of material and is shown in Figure 19 for eight materials of interest representing reactor and isotope fuel forms. Figure 20 shows the potential for flashing or thermodynamic instability of these materials and Figure 21 shows the burnup potential of the critical sized drops of these materials when exposed to aerodynamic heating and oxidation.

Table XII attempts to correlate the full potential of these materials to be destroyed by all the phenomena of interest.

It can be seen that simple vaporization based on aerodynamic heating is not enough to reduce any of the fuel forms, released critical size drops, to submicron levels. However, Pu^{238}, Po^{210} and Sr metallic fuel forms show the most promise.

This simple vaporization coupled with low vapor pressures, oxidation heating and possibly chaotic spray upon release (explained in References 11 and 12 and shown in Table XII) makes the possibility of reaching submicron sizes a real one. Hence, the present aerospace safety burnup criteria of 1μ above 100,000 ft has a distinct possibility of being achieved by the fuel forms of greatest interest, relative to satisfying mission requirements discussed in Sections II, III and IV of this report.

If the designer were to define those properties of greatest importance which should be incorporated into a fuel form expected to burnup to acceptable sizes upon reentry into the Earth's atmosphere, they would be:

[redacted]

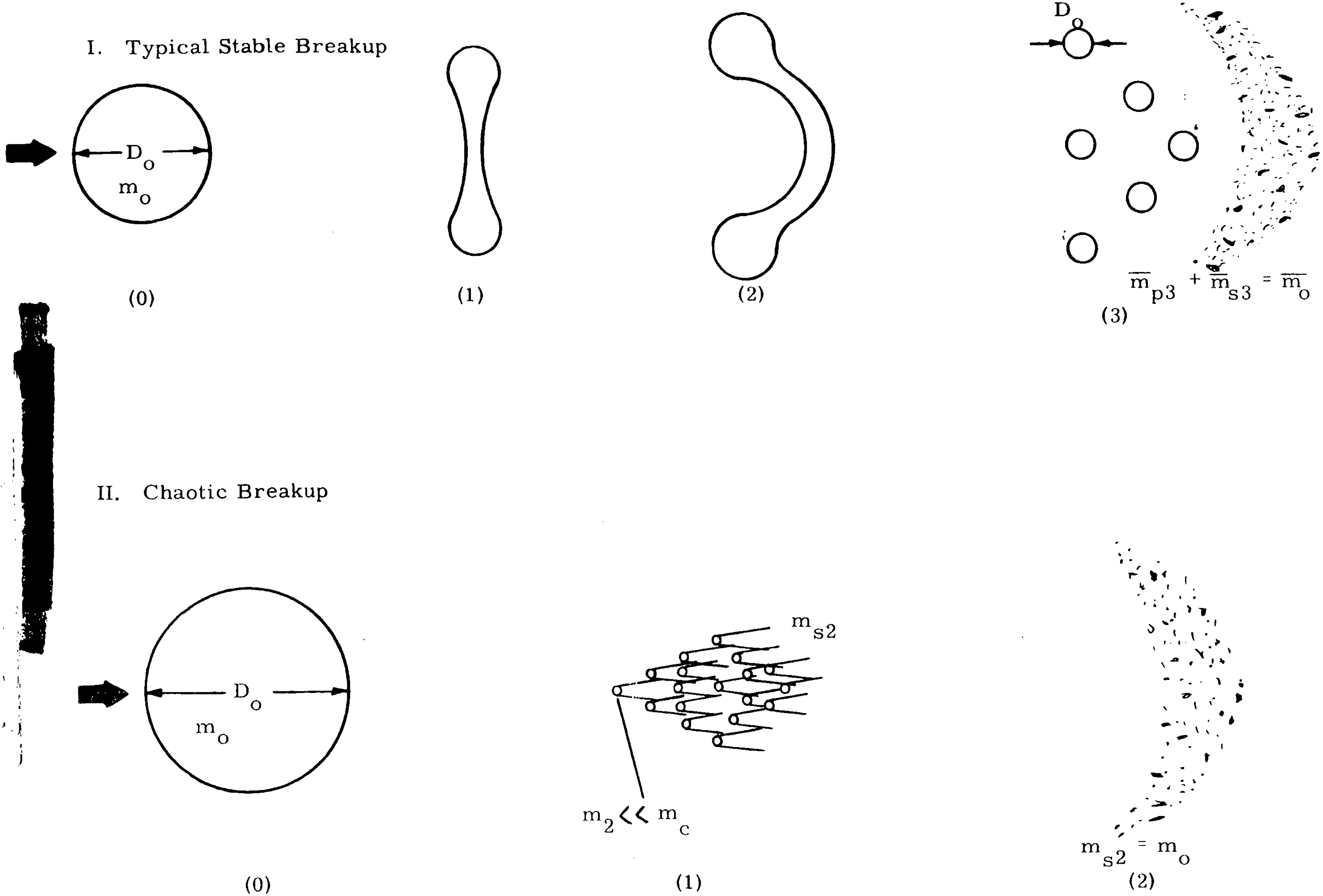

Figure 18. Characteristic Large Drop Breakup

[redacted]

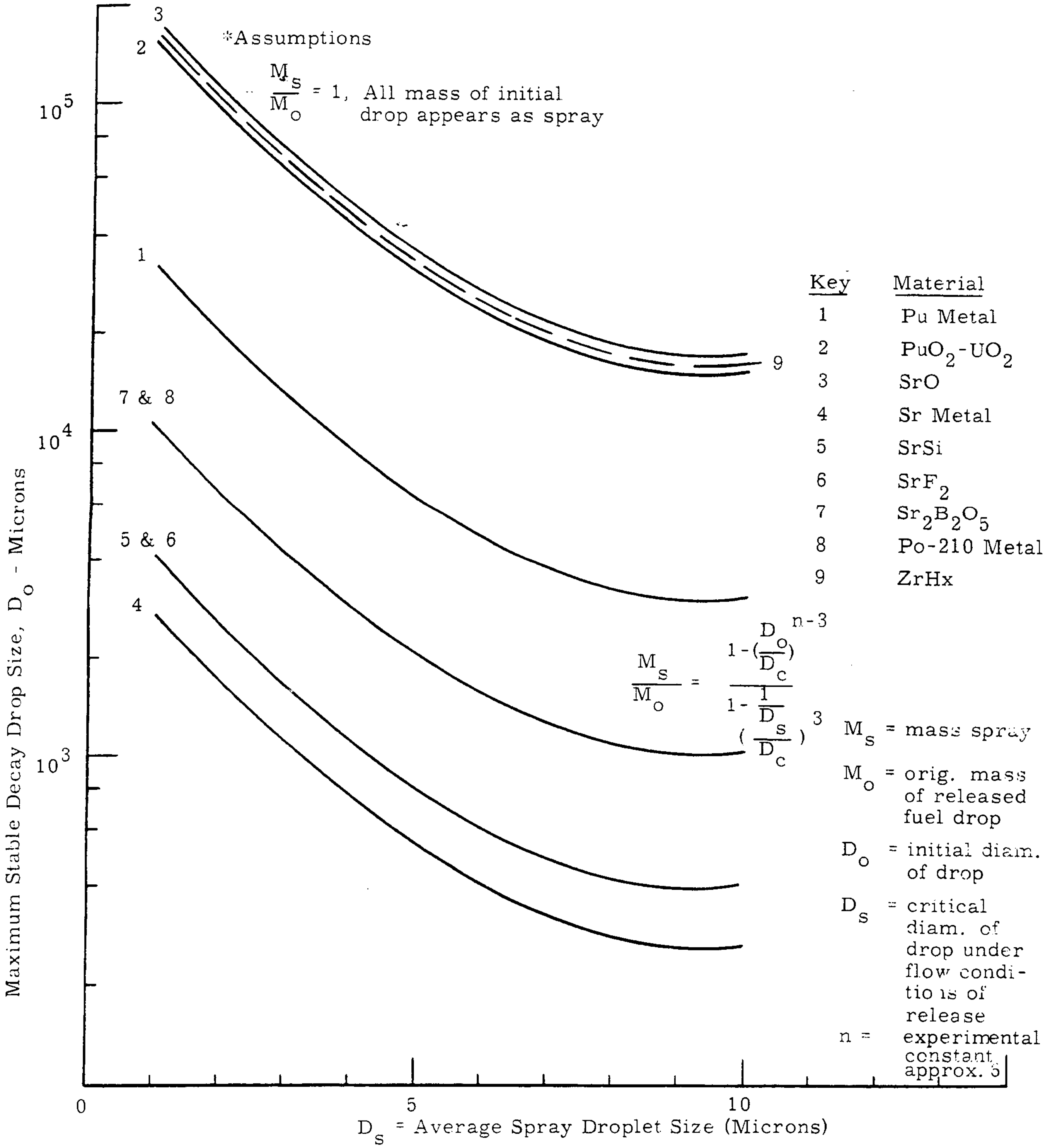

Figure 19. Maximum Initial Drop Size Above Which Large Spraying Instabilities will Occur at 200,000 Feet and 23,000 Ft/sec for Fuel Forms of Interest.

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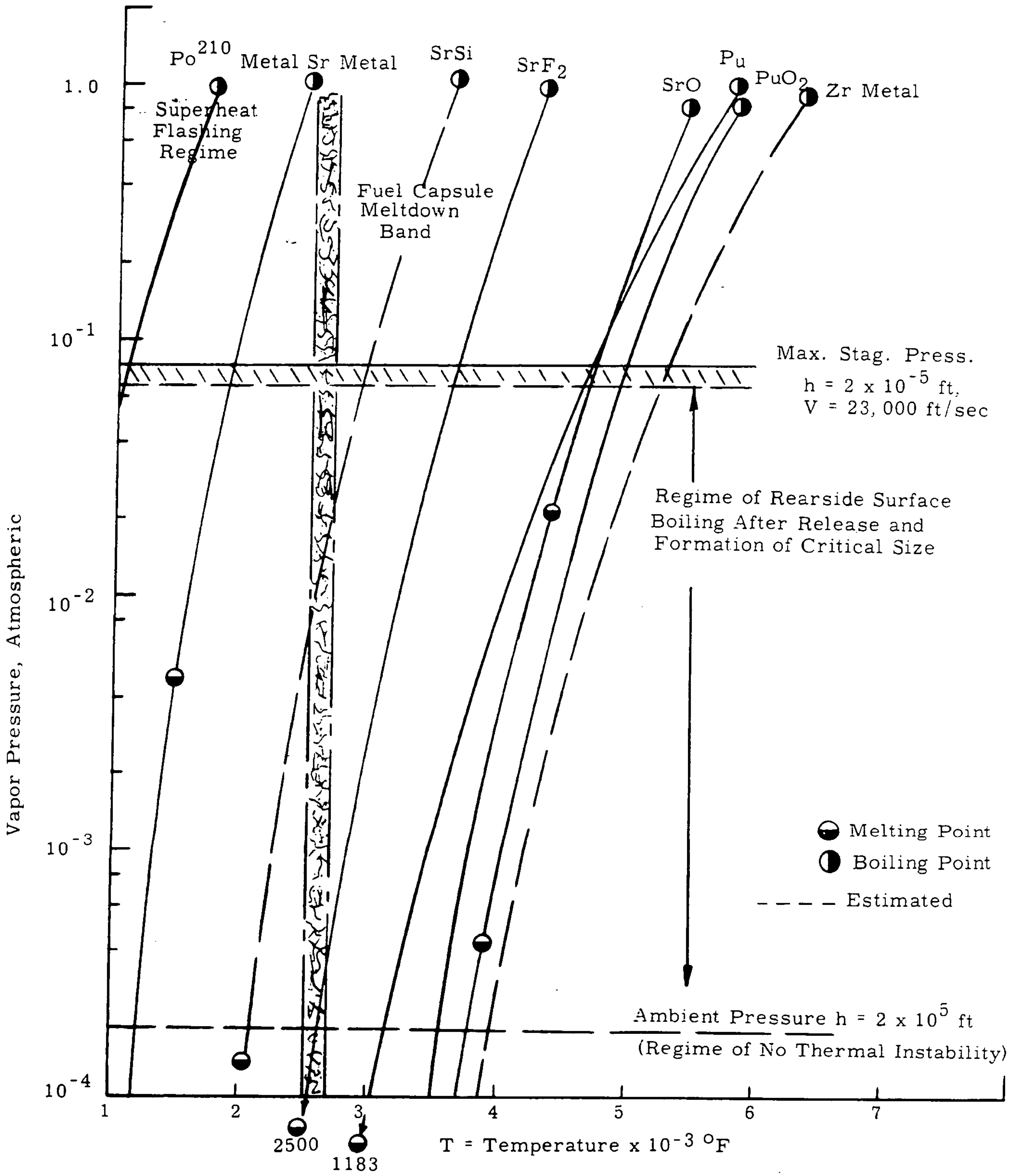
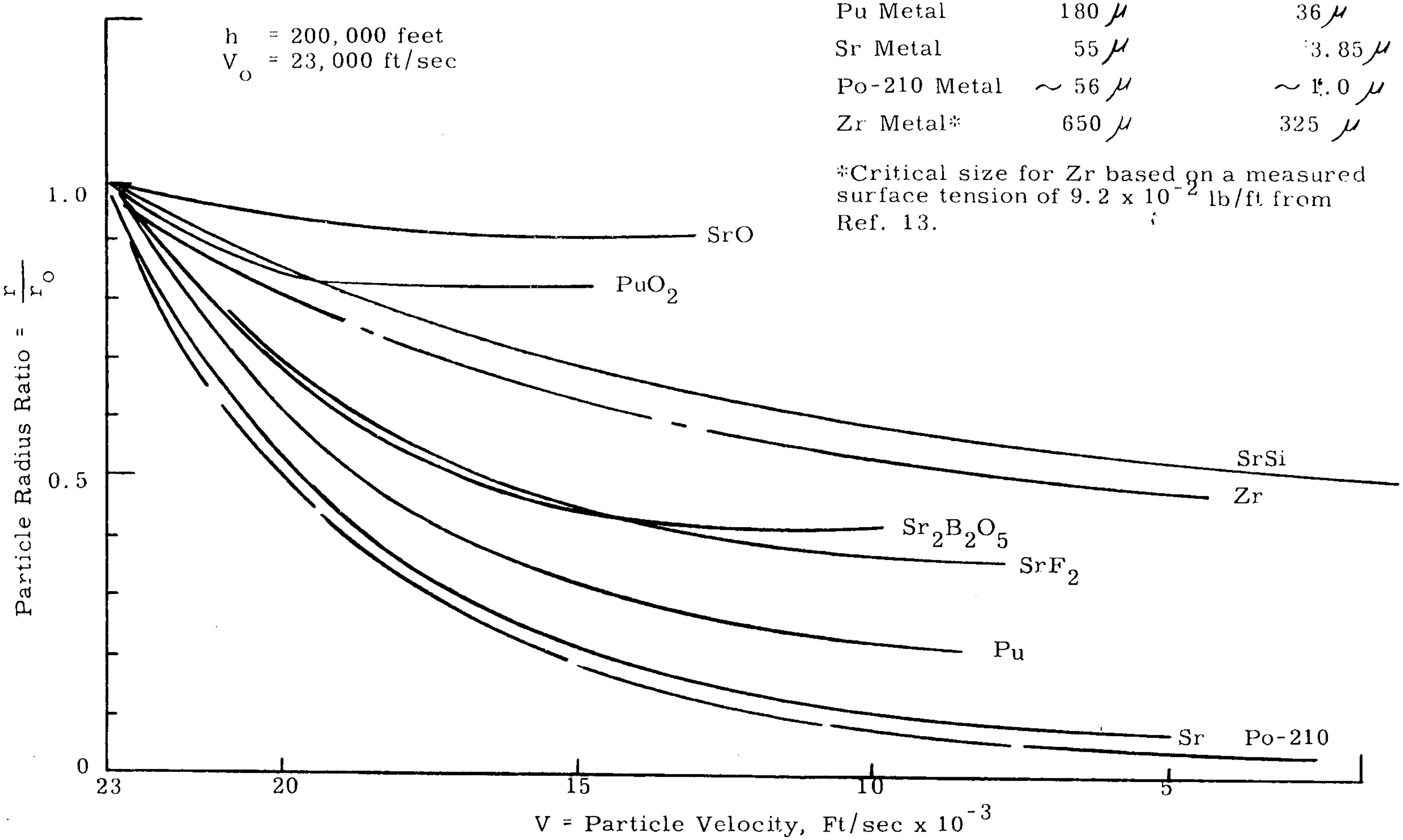


Figure 20. Potential for Flashing and Thermodynamic Breakup of Typical Fuel Forms at the Instant of Release.

[redacted]



Material	Av. Critical Size	Final Size
SrO	430 μ	390 μ
PuO_2	400 μ	330 μ
SrSi	67 μ	34 μ
$Sr_2B_2O_5$	107 μ	45 μ
SrF_2	65 μ	24 μ
Pu Metal	180 μ	36 μ
Sr Metal	55 μ	3.85 μ
Po-210 Metal	~ 56 μ	~ 1.0 μ
Zr Metal*	650 μ	325 μ

*Critical size for Zr based on a measured surface tension of 9.2 x 10^{-2} lb/ft from Ref. 13.

Figure 21. Comparison of the Vaporization Potential of Various Fuel Forms During Reentry Deceleration.

Material	Melting Temp °F Ratio T_m fuel / T_m capsule	Vaporization Temp. T_v °F	Surface Tension lb/ft	Heat of Vaporization H_v Btu/lb	Heat of Reaction H_R Btu/lb O_2	Chaotic Spray Dispersal Drop Size Limit	Critical Size Drop (Microns)	Thermodynamic Instabilities at Release	Approximate Significance of Boiling Instabilities During Flight	Oxidation Reactions H_R = Btu/lb	[redacted] Character of Oxide Expected	Final Vaporization Sizes (Microns)	Susceptible to Explosion Phenomena	[redacted] Remarks
Pu Metal	T_m = 1183 R = 0.455 molten release	5852	3.60×10^{-2}	605	2.7×10^4	Above 0.120 inch	177 - 200	None	Very Small	yes H_R = 2.7×10^4	Broken and swept off surface, some absorption	18 - 36 for = 0.1 - 0.5 respectively	Expected	Final size assuming all oxide stays with particle = 40
[redacted]	T_m = 3862 R = 1.55 solid release	5852	0.41×10^{-2}	597	....	Above 0.60 inch	380 - 425	None	None	No	....	330	No	No burnup expected in real release case
[redacted]	T_m = 1420 R = 0.545 molten release	2442	1.13×10^{-2}	710	1.6×10^4	Above 0.013 inch	52-57	Superheat at Release Yields Flashing	Great	yes H_R = 1.6×10^4	Broken and swept off surface, some absorption	3 - 4 small effect of oxide.	Expected	Final size assuming all oxide stays with particle = 3.5 Sr is affected by all destruction phenomena
[redacted]	T_m = 4400 R = 2.76 solid release	5432	0.9×10^{-2}	2180	....	Above 0.87	410 - 450	None	None	No	....	390	No	No burnup expected in real release case
[redacted]	T_m = 2480 R = 0.95 - 1.0 plastic release	4316	1.45×10^{-2} @ 2500°F	1010	....	Above 0.016	64 - 70	None	Some Significane if it occurs	No	....	24	No	Questionable state of release. A' lation of slug expected. Non-opti mum release results. Particles susceptible to spray dispersal
[redacted]	T_m = 2102 R = 0.820 molten release	5432	2.35×10^{-2} @ 2100°F	730	....	Above 0.039 inch	102 - 112	None	Very Small	No	....	45	No	
[redacted]	T_m = 2000 R = 0.77 plastic release	3630	1.5×10^{-2} @ 2000°F estimated based on SrF_2	3000	8.5×10^4	Above 0.0160 inch	64 - 70	None	Some Significane if it occurs	yes H_R = 8.5×10^4	Most likely to from protective shield - expected to be swept off in vapor	34	Unknown but expected	High heat of fusion and vaporiza-tion. Non-optimum release. Drops susceptible to spray dispersal
Po-210 as GdPo	T_m = 3000 R = 1.2	1750 Po-210 Separates from GdPo at meltdown	2.5×10^{-2}	650	..........*	Above 56 μ	55 - 60	Violent Flashing at GdPo melt-down	....	....	PoO_2 is a gas	Smallest of all probably < 1 μ	No	Po-210 metal has a low melting point, also GdPo is expected to exhibit good reentry burnup characteristics
Zr	T_m = 3370 Reactor use	6470	0.25×10^{-2}	780	1.45×10^4 *PoO_2 is a Gas at Reentry Temperatures	Above 400 μ	650	None	....	yes, H_R = 1.45×10^4	High absorption characteristics	$\frac{R}{R_o}$ = .55	Expected and known to occur	Poor vaporization performance due to high vapor temperature yielding high re-radiation losses. Oxidation and explosive phenomena are significant and could yield optimum debris at high altitudes

[redacted]

Table XII

General Comparison of Fuel Form Reentry Destruction Potential at h=200,000 Feet and V_o = 23,000 ft/sec

[redacted]

1) Low melting point
2) A molten release from the fuel capsule
3) Low heat of fusion
4) Low surface tension
5) Low heat of vaporization
6) Low vaporization temperature
7) Low emissivity
8) High heat of reaction with air

From this brief discussion and analysis it becomes clear that those fuel forms, whether reactor or radioisotope, which have the capability of being coupled to high temperature, i. e., advanced power systems, will be, by their very nature, unable to meet the burnup criteria. Metals seem to show the best burnup performance but their use in high temperature radioisotopic systems (T >1800°F) requires them either in a molten state or as an oxide fuel form (solid state).

The molten state may require refractory liners (i. e., tantalum) in the capsule to prevent internal corrosion under normal operation and in turn may hamper the burnup of the capsule during reentry. If a capsule liner could be developed to hold liquid metal under high temperature generator operation and still be able to be breached during reentry to release the molten metal, the burnup aerospace safety philosophy might be able to be extended to advanced power systems. The breaching mechanisms of internal pressure and oxidation hold promise to achieve this, especially for the α emitters, since the refractory liners needed would be low in strength after burnaway of the outer capsule wall.

Relative to the radioisotope fuel forms of most interest, as defined by the mission analysis in Section II (Pu^{238} and Po^{210}), high altitude burnup is an acceptable safety philosophy, provided that the Nuclear Safety Index as outlined previously is not violated. For reactors using Zr fuel elements, burnup to acceptable sizes seems marginal at best, and final reduction of the fuel droplets must come from oxidation and explosion phenomena which are as yet not completely defined. Based on the mode of release of the fuel elements and the reentry processes which are expended upon the fuel form and resultant drops, reactors seem not to make as optimum use of the available destruction potential of the atmosphere as radioisotope generators do.

With the advent of the higher performance UO_2 and UC reactor systems, acceptable destruction on reentry will become even more remote.

[redacted]

b. Cis-Lunar and Planetary Return. The conclusions discussed above are based on reentry conditions with potential energies of the order of 13,500 Btu/lb, representing typical satellite decay velocities. However, missions which will be flown in cis-lunar and planetary space would have reentry velocities of 36,700 ft/sec to 50,000 ft/sec, respectively compared to 25,000 ft/sec for orbital decay.

In these cases, substantially greater peak heating is available, although the reentry angles are steeper and hence total heating time shorter. The reentry angles are determined by the skip limits of the reentering body. For velocities typical of lunar and planetary return, the vehicle will not reenter in one pass if the reentry angle is less than 7-10°, respectively. The vehicle will skipout, degrade in velocity and eventually reenter. The reentry after skipout will be typically a satellite one in one or two more passes if the initial angle was close to the 7 - 10° limit.

It is expected under the conditions postulated above that reentry systems will be severely and, in most cases, completely destroyed. The radiation heating from the ionized shock layer during earth reentry at these speeds is very significant. The selection of burnup, therefore, as an eventual destruction mode will probably be enhanced by the higher reentry velocities.

However, the one danger that high velocity, steep angle reentries hold for the general safety of the burnup concept is that if the nuclear system is buried in a reentering satellite or probe vehicle, it will not separate until very low altitudes are attained ($\approx$75,000 ft). Here, although complete burnup at speeds of $\approx$30,000 ft/sec is assumed, the burnup is low enough to present a significant fallout problem. Therefore, eject mechanisms or placing the fuel source in the high velocity flow as early as possible during the reentry is still an important consideration to assure high altitude burnup.

4. Partial and Fragmentary Burnup - (Microspheres)

The constant attempt to increase radioisotope generator efficiency has produced another safety technique which relies on partial burnup to operate. The technique was developed to make use of high temperature fuel forms with inherently poor burnup potential, but with the capability to utilize higher efficiency conversion techniques in the generator.

The technique consists of fabricating the fuel form as microspheres and releasing these upon capsule burnup high enough in the atmosphere so that the ensuing fallout and dispersal will produce an acceptable ground hazard. The key point in ensuring the mechanics of the dispersal work is non-sintering of the microspheres during normal operation. Obviously the application of this safety philosophy to the missions described in Section II of this report depends on the type and quantity of isotope released, the character of the ensuing dispersal pattern and the fact that no chemical changes affecting the biological inertness of the fuel form occur during the reentry process.

[redacted]

To assess the limitations of this partial burnup technique, a typical $Pu^{238}O_2$ microsphere particle size distribution was assumed as shown in Figure 22, compiled from Reference 20. The acceptance of intact reentry of the microspheres from a safety point of view must imply minimal hazards. This is the case with the inert Pu^{238} fuel forms. As shown in Figure 23, the dissolution of $Pu^{238}O_2$ microspheres in seawater would lead to complete dissolution of a typical 150μ particle in about 6.83 x 10^5 days. It is felt, therefore, that the criteria of insolubility is reasonable. The question of chemical changes of the PuO_2 microspheres during reentry via surface melting and further oxidation fractioning of the microspheres due to re-solidification can best be answered experimentally.

In addition, burnup from typical satellite reentry will not ablate much from a 150μ PuO_2 microsphere. Since this size is below the critical size, only direct vaporization will remove material and not surface shear forces. Therefore, these sizes are reasonable approximations of the final surface impact debris. The majority of the fuel form will be above sizes where significant inhalation can take place. Hence, the only hazards are through direct ingestion and skin doses.

Ingestion hazards from microspheres will not affect large portions of the population in any given area unless the water supply is contaminated. This probability is low, especially in light of the filtering fresh water undergoes before usage. Therefore, based on the claimed ecological inertness of this fuel form, external skin dose from surface contamination seems the most severe hazard from this form of aerospace safety technique.

A parametric analysis was undertaken to try and determine the maximum allowable PuO_2 microsphere inventories which can be tolerated relative to surface dose rates. Appendix C was derived to estimate the surface dose rates from PuO_2 microsphere releases from approximately 200,000 ft., representing a capsule breach during a typical satellite reentry.

Figure 24 shows that powers of the order of 20 Kwt can be released instantaneously as microspheres under conditions assumed before the percent of background radioactivity reaches the natural level. Hence, radioisotope power systems in the 1-10Kw(e) range look safe with the microsphere technique, assuming no vaporization and an insoluble fuel form.

If the reentry velocities are increased to the 50,000 ft/sec range, then a significant portion, if not all of the microspheres, may vaporize. This is quite severe since this vaporization would occur at a relatively low altitude compared to satellite decay fuel capsule burnup and consequent microsphere release. For example, on return from a typical manned Mars mission, should the astronauts fail to survive beyond the first trans-earth correction, the earth reentry velocity will be about 50,000 ft/sec. The penetration of the atmosphere prior to spacecraft disintegration has been estimated to be to an altitude of less than 150,000 ft. At this time, the PuO_2 fuel block would be

[redacted]

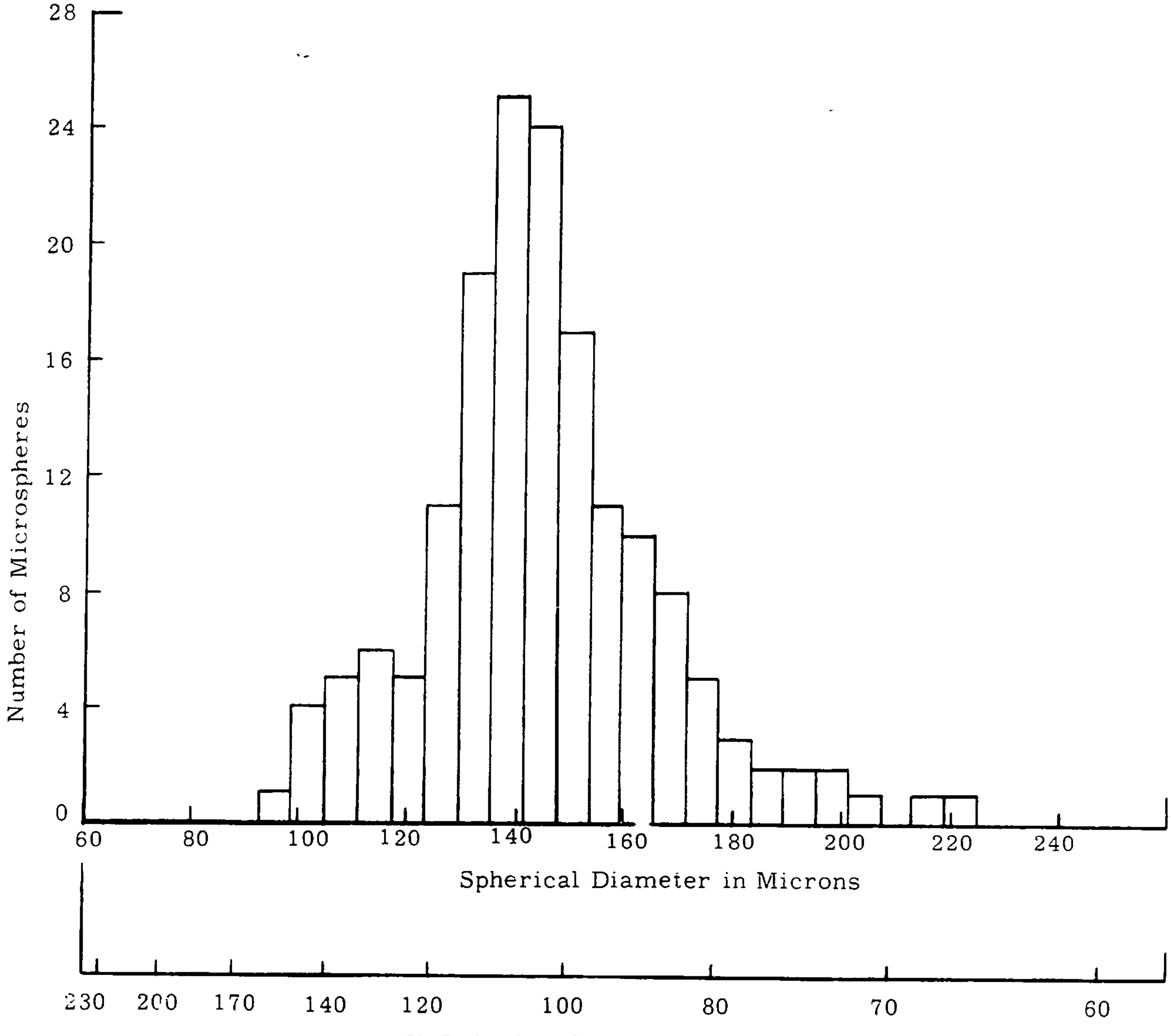
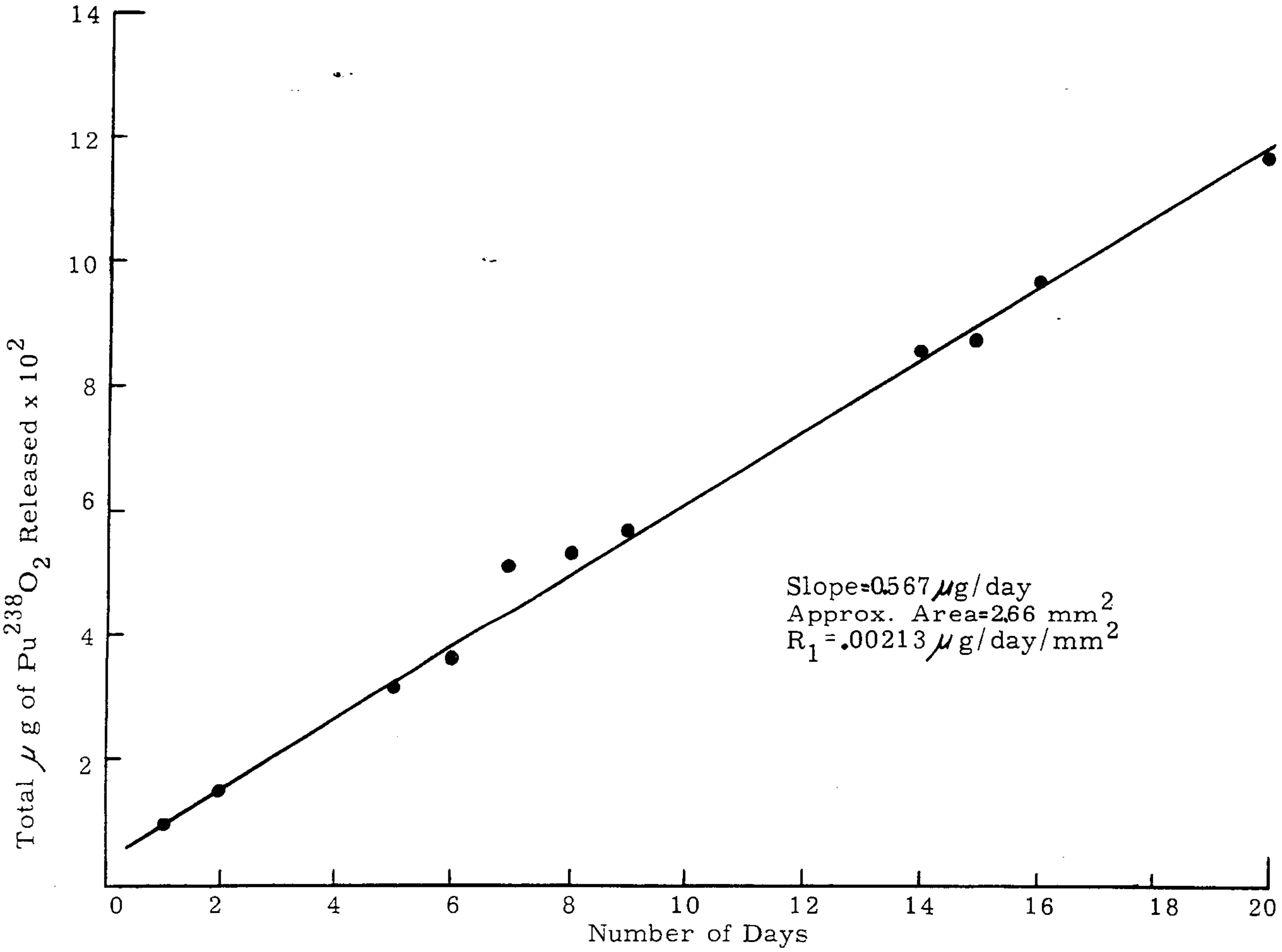


Figure 22. The Spherical Particle Size Distribution of 163 $Pu^{238}O_2$ Microspheres Made from Crushed Sintered Oxide of 100 to 200 Mesh.



Derivation of Dissolution Time for a 150 μ $Pu^{238}O_2$ Microsphere

Given:

$$R_t = 2.13 \times 10^{-3} \frac{\mu g}{day} / mm^2 = \frac{dm/dt}{4\pi r^2}$$

$$\rho = 9.7 \text{ gm/cc}$$

$$m = \rho \frac{4}{3} \pi r^3$$

$$\frac{dm}{dt} = \rho 4 \pi r^2 \frac{dr}{dt} \longrightarrow \text{hence} \quad \frac{dm/dt}{4\pi r^2} = \frac{\rho\, dr}{dt} = R_t$$

$$\int_0^t dt = -\frac{\rho}{R_t} \int_{r_0}^0 dr$$

$$t = \frac{\rho r_0}{R_t} = 6.83 \times 10^5 \text{ days}$$

Figure 23. Dissolution of $Pu^{238}O_2$ Microspheres in Sea Water.

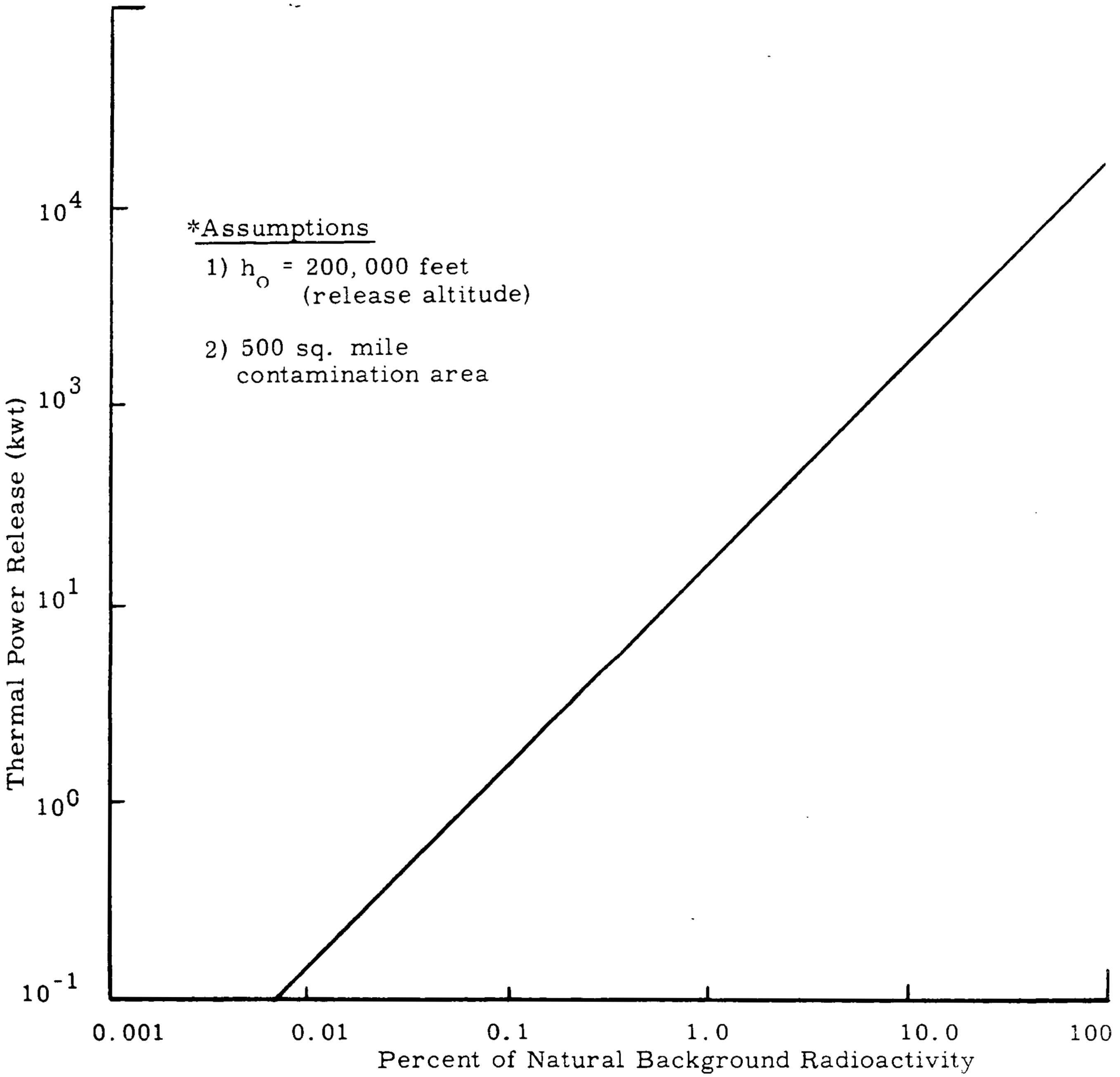


Figure 24. Surface Dose Rates as a Function of Thermal Power Released for PuO_2 Microspheres.

[redacted]

traveling approximately 30,000 ft/sec upon release from the destroyed reentry vehicle. Therefore, it is conceivable that the PuO_2 microspheres could be greatly reduced in size by vaporization in this accident and that dispersal could occur in the lower stratosphere or upper troposphere, thereby, creating a possible inhalation hazard.

Several techniques have been proposed to solve the low altitude burnup problem (Reference 21) of PuO_2 microspheres upon aborted planetary return. They are the source extension module, graphite coated microspheres and the fuel capsule intact reentry.

The source extension module essentially keeps the source exposed to the air flow at high altitudes until a successful reentry is assured then retracts the fuel assembly into the reentry vehicle heat shield for the majority of the heating regime. The advantage here is that the source is kept outside of the reentry vehicle so if an abort does occur during reentry, high altitude dispersal will result and the reentry vehicle will not shield the heat source until the lower altitudes.

The second concept is one of putting a prolytic graphite coating on the microspheres to prevent burnup. Analysis (Reference 21) have shown that this technique will not prevent burnup of the microspheres at 75,000 - 100,000 ft with a velocity of 30,000 ft/sec due to the oxidation rates of the graphite under these conditions. In addition, the lower thermal efficiency of this concept may ultimately manifest itself by increasing the fuel inventory significantly in any single design concept using this fuel.

Finally, intact reentry of the PuO_2 microsphere fuel in its original containment is defeating the purpose of the microsphere approach. All the problems of random impact of a large inventory of fuel and the attendent burial overheating and possible vaporization problems appear. In addition, the added weight and complexity of a reentry body to survive 50,000 ft/sec may be prohibitive. Certainly the reentry technology existing today would be hard pressed to develop this reentry vehicle. It is not state-of-the-art.

5. Large Component Fragmentary Burnup

The same general safety philosophy as presented by the microsphere approach may be extended to larger debris. The basic satellite and generator structure fractures and releases several fuel blocks. The limitations of this fragmentary burnup approach extend from microspheres on the low end to complete intact reentry of the source at the high end. The hazards involved in any approach in between would be a function again of the character and quantity of the radioisotope being reentered.

[redacted]

[redacted]

If the isotope were one of the fuel forms of Sr-90, then significant shielding would have to be provided especially for reentry survival of large portions of the fuel block. If the isotope were one of the Pu-238 fuel forms then there would be a gain by partial fragmentation of the source. The reduction in ingestion hazards by keeping the source in one piece or close to it and the dispersal of fairly large pieces of insoluble relatively low radiation material would serve the purpose of localizing the hazard and not exposing as many people to the radioactive debris as with microspheres. The limit of this philosophy is, of course, completely intact reentry of the source.

In general fragmentary release of the fuel block may

a) Prevent shielding of large heat sources which could reenter intact.

b) Minimize contaminated area and population exposure.

c) Enhance recovery and/or cleanup of the fuel form.

However, in all cases, if exposure to the reentry debris does occur it will be more severe than with the microsphere approach. A trade-off between low level irradiation of a large mass of people vs. higher irradiation of a smaller group is the philosophy of this concept. Therefore, its applicability can only be assessed in the framework of the importance and characteristics of a real mission and power plant. In general, large component fragmentary breakup is not recommended based on the increased difficulty in locating the source and the increase in contamination area.

6. Results and Conclusions: Atmosphere Burnup

a. Based on the concept of the NSI_a (Nuclear Safety Index - Air) for high altitude burnup, the most critical radioisotope for burnup is Pu-238. In the most optimistic case powers of the order of several hundred kwt can be tolerated with high altitude injection. In the most pessimistic appraisal, approximately 10 kwt may cause a hazard. Estimates of maximum injection powers for other radioisotopes are given in Table I in the summary of this report.

b. Based on the concept of the NSI_s (Nuclear Safety Index for Surface Contamination) and a mesospheric release with minor restrictions, Pu-238 is still the most critical isotope from a hazards standpoint. Only 20 kilowatts internal can be tolerated in the most optimistic case. Estimates of the maximum injection powers for other radioisotopes yielding surface contamination are given in Table I in the summary of this report.

c. For all the nuclear missions and fuels considered in this analysis except Po-210 a mesosphere release yields more critical surface concentrations than the eventual tropospheric concentrations. Pu-238 is the fuel which consistently yields the highest NSI. Po-210 is the

[redacted]

only fuel of the group studied which displays a higher fallout NSI for tropospheric air than for surface concentration. However, for the missions studied, the application of Po-210 consistently yields NSI's five to seven orders of magnitude below the MPC. Hence, no significant hazard is envisioned with the use of Po-210 coupled to high altitude burnup if this burnup can be achieved.

For the majority of missions where Pm-147 is considered applicable as a fuel and where high altitude burnup is desired NSI's of from four to seven orders of magnitude below critical result. Therefore, it is felt that Pm-147 will be acceptable as a burnup fuel if the fuel form can reach the desired size range during burnup.

d. Stratospheric release of the fuel approximating ascent aborts with partial burnup yields an optimistic limit for Pu-238 (100,200 Kw(t)) for an $NSI_s = .10$. However, most pesimistic case results show limits of the order of 8 Kw(t). This represents a real power range of from 10-40 Kw(e) depending on the conversion system used and the amount of partial burnup experienced in the ascent abort. Surface contamination is still most critical for a stratospheric release and this yields a most optimistic limit of approximately 20 Kw(t) for an $NSI_s = .10$ (minor restrictions). Pu-238 again being the most critical, even considering full release of high vapor pressure fuels such as Po-210 vs partial release of Pu-238.

e. Low altitude and ground releases are the most serious relative to earth surface contamination with Pu-238 again being the most critical isotope for contamination if vaporization is considered. However, this event has a low probability of occurrence with Pu-238 due to its fuel forms and the available energy from ground aborts. Therefore, Po-210 must be considered the chief danger under these circumstances.

f. Not enough energy is available during satellite decay to destroy advanced reactor fuel elements such as UC, UO_2, or UN. High power plant operating temperatures are more critical than high power levels relative to achieving burnup because of the high temperature materials they require infuel form and fuel element design.

g. In general there are two main mechanisms for molten debris reduction during reentry; mechanical breakup due to aero pressures and aerodynamic heating of the resulting debris. Such thermodynamic assets as superheat upon release yielding flashing are added benefits which the fuel form designer should try to employ but are difficult to come by in all fuel forms of interest except Sr and Po-210 metals.

h. The isotope fuel forms most susceptible to chaotic spray as a primary destruction mechanism are Sr metal, SrF_2 and SrSi.

[redacted]

i. The fuel forms most susceptible to total reentry vaporization are Pu and Sr metals and Po-210.

j. In general simple vaporization coupled with low vapor pressures, oxidation heating and possible chaotic spray make the possibility of reaching submicron sizes a real one for those materials which can demonstrate the necessary properties.

k. If the designer were to define those properties of greatest importance which should be incorporated into a fuel form expected to burnup to acceptable sizes upon reentry into the earth's atmosphere, they would be:

(1) Low melting point
(2) A molten release from the fuel capsule
(3) Low heat of fusion
(4) Low surface tension
(5) Low heat of vaporization
(6) Low vaporization temperature
(7) Low emissivity
(8) High heat of reaction

l. From the analysis carried out in this study, it can be stated that those fuel forms, whether reactor or radioisotope, which have the capability of being coupled to high temperature, i.e., advanced power systems, will be by their very nature, unable to meet the burnup criteria. Metals seem to show the best burnup performance but their use in high temperature radioisotopic systems ($T > 1800^{\circ}F$) requires them either to operate in a molten state or as an oxide fuel form (solid state).

The molten state may require refractory liners (i.e., tantalum) in the capsule to prevent internal corrosion under normal operation and in turn may hamper the burnup of the capsule during reentry. If a capsule liner could be developed to hold liquid metal under high temperature generator operation and still be able to be breached during reentry to release the molten metal, the burnup aerospace safety philosophy might be able to be extended to advanced power systems. The breaching mechanisms of internal pressure and oxidation hold promise to achieve this, especially for the α emitters, since the refractory liners needed would be low in strength after burnaway of the outer capsule wall and surface oxidation.

Relative to the radioisotope fuel forms of most interest, as defined by the mission analysis in Section II (Pu-238 and Po-210) high altitude burnup is an acceptable safety philosophy provided that the Nuclear Safety Index as defined in the text is not violated. Cu-244 would have to reenter in tact. For reactors using Zr fuel elements, burnup seems marginal at best and final reduction of the fuel droplets must come from oxidation and explosion phenomena whose effects are as yet not completely defined. Based on the mode of release of the fuel elements, their size and the reentry processes which are expended upon the fuel form and resultant drops, reactors seem not to make as optimum use of the available destruction potential of the atmosphere as radioisotope generators can.

[redacted]

[redacted]

m. With the advent of the higher performance UO_2 and UC reactor systems acceptable destruction on reentry will become even more remote. Therefore SNAP reactor fuel elements made of UO_2 microspheres should be actively considered as a solution to the end of life reentry problem with high temperature systems. If sintering can be avoided during operation the technique should work.

n. Reentry from cislunar or planetary missions will inherently increase the energy available to burnup the nuclear device. However, although burnup is assured, it may no longer be high altitude burnup. Because of the greater speeds of reentry and the steeper angles inherent in the trajectory for reentry in one earth revolution, burnup can occur at much lower altitudes. This is especially true if the nuclear device is buried in a reentering probe vehicle.

Therefore, eject mechanisms or placing the fuel source on the outside of the reentry vehicle to assure early fuel release is a prerequisite for considering burnup as a nuclear safety philosophy in this case.

o. The concept of PuO_2 microspheres is applicable as an aerospace nuclear safety technique to many missions of interest. Satellite decay cases are the best application for unmanned systems. The most critical hazard is skin dose on the earth's surface since ingestion, water contamination and inhalation hazards can be minimized.

It was found that powers of the order of 20 Kw(t) can be released as microspheres under the conditions assumed before the percent of background surface radioactivity reaches the natural level assuming a 500 square mile impact area. Hence all radioisotope power systems up to at least 1-5 Kw(e) look safe with the microsphere technique assuming no vaporization and an insoluble fuel form.

Effects on solubility may occur if surface melting and oxidation take place during reentry. In addition, refreezing may crack microspheres into inhalable sizes. Experiments are recommended to define this.

For reentry at planetary return speeds of 50,000 ft/sec, a significant portion if not all of the microspheres may vaporize and at a low altitude ($\sim 10^5$ ft). Hence, precautions must be taken to expose the fuel capsules to the flow immediately upon reentry.

[redacted]

[redacted]

B. Intact Reentry and Attendent Recovery

As was seen in Section IV-A, the increase in nuclear APU (Auxilliary Power Unit) operating temperature rather than fuel inventory threatens to negate reentry burnup as a workable aerospace safety philosophy. Fuel form materials with the required high temperature properties for normal operation, in high thermal efficiency nuclear systems, may well be able to withstand all reentries up to 50,000 ft/sec. In addition, only intact reentry offers the potential of minimizing the hazards attendent with the flight of nuclear systems. The ultimate capability of being able to return a nuclear fuel package intact to a preselected point on the earth's surface and recover or dispose of it at will represents a solution which could assure the application of nuclear power in space.

It is the purpose of this discussion to define the various categories of intact reentry, investigate their applicability to various nuclear missions, and evaluate in a preliminary fashion their technical feasibility and operational problems.

For the sake of simplicity, the discussion has been presented in three (3) general categories:

(1) Uncontrolled Random Reentry
(2) Semi-Controlled Reentry
(3) Fully Controlled Reentry

1. Uncontrolled Random Intact Reentry

Definition: The reentry of a system into the earth's atmosphere in an uncontrolled random manner without any active means to predetermine its impact points on the earth's surface or modify its trajectory during the reentry.

The implications of uncontrolled random reentry are such that the hazards from this technique must be acceptable relative to the general definition of aerospace nuclear safety. Therefore, the controlling factor in application of this technique is the magnitude and nature of the radioactive inventory being returned to earth. Because of this,uncontrolled random reentry (as a safety technique) is limited to the lower power radioisotope systems (∿100-500 W(e)). The increase in radiation shielding weights for general population protection due to use of Sr-90,for example Table XIII, could make uncontrolled random intact reentry unattractive even at these lower power levels. Hence, Pu-238, Po-210 and Pm-147 fuel forms seem to represent the best compromise of a minimum weight random intact reentry system and minimum hazards to the general public if impact in an inhabited area results. Table XIII was compiled for shielded capsules of various radioisotopes from a 10 watt(e) and a 100 watt(e) generator. The shielding given provides a dose rate of 1 rem/hr at 1 foot from the reentered fuel capsule which represents personnel protection emergency dose rates.

[redacted]

Table XIII

Low Power Systems

Isotope	Dose Rate rem/hr at 1 ft	Shield Material	Total Shield Weight Lbs.	
			10 watt(e)	100 watt(e)
Sr-90	1	Uranium	88	459
Cs-137	1	Uranium	189	758
Pm-147	1	Uranium	---	36.4
Pu-238	1	Lithium Hydride and Uranium	---	9.4
Cm-244	1	Uranium	---	18.7
Po-210	1	Lithium Hydride and Uranium	---	10.3

The basic question of feasibility and limits of application of this technique revolve around the following:

1. The character of the reentry.
2. The choice of reentry configuration.
3. Thermal protection problems of intact reentry systems.
4. General design considerations of a typical intact random random reentry vehicle.
5. Intact random reentry with delayed recovery.

a. The Character of the Reentry. Based on the conceivable missions that these heat sources would be called upon to fly, the following reentries could exist:

1. Ballistic reentry into earth atmosphere from ascent aborts.
2. Decaying satellite reentry from earth orbit.
3. Earth reentry from lunar mission aborts or planetary return at super satellite speeds.
4. Reentry into non-earth atmospheres

[redacted]

Figure 25 shows the comparison of maximum heating rate parameter for earth satellite, lunar return, Earth, Mars and Venus ballistic, and satellite and lifting body reentries. Since the escape velocity from Mars and Venus is less than Earth escape velocities, then reentry into the earth's atmosphere upon return from these planets, will not exceed maximum heat fluxes at earth escape velocities. For manned missions, however, minimum transit time must be a consideration, hence velocities of the order of 50,000 ft/sec could occur in an abort from a typical manned Mars return. Therefore, the whole concept of intact reentry under these conditions is marginal. The analyses presented herein was, therefore, compiled for satellite and probe vehicles whose reentry velocities would not exceed earth-escape.

The character of the reentry not only depends on peak heat fluxes but on the total heating and impact. Figure 26 shows estimates of the total heating for earth ballistic, satellite and lunar return reentries to Earth. Total heating of Mars and Venus reentries depend on more detailed investigation of their atmospheres. However, it has been experimentally estimated that the Martian surface densities are approximately 1-2 percent of earth densities (References 22, 23, and 24). The Martian atmosphere scale height is similar to earth's according to Mariner data, hence the rate at which atmospheric pressure increases with decreasing height is also similar although its magnitude at the surface is only 1-2 percent of Earth values. Because of this, higher drag bodies or more retro will be required to give acceptable impact velocities at the Martian surface.

In general, the total heat loads expected for a Martian or Venusian planetary entry would be less than those experienced in a normal earth satellite decay reentry. Consequently, it is expected that designing the heat source reentry system for earth reentry represents a good design limit for planetary operations. The critical cases are obviously earth return at super satellite speeds from cis-lunar and planetary missions. Of the two, cis-lunar return is the most probable since planetary flyby trajectories can be designed to have a solar termination or if the mission is manned and must return, deep space disposal of the nuclear source is feasible. It is felt that the technology to handle cis-lunar return below the skipout limit is feasible and state of the art.

b. Choice of Reentry Configuration. Uncontrolled random intact reentry of the smaller radioisotope systems under present discussion can theoretically be accomplished in several ways as shown in Figure 27.

1. Intact reentry of the fuel capsules as individual entities.

2. Intact reentry of the total fuel block.

3. Intact reentry of the complete generator.

[redacted]

[redacted]

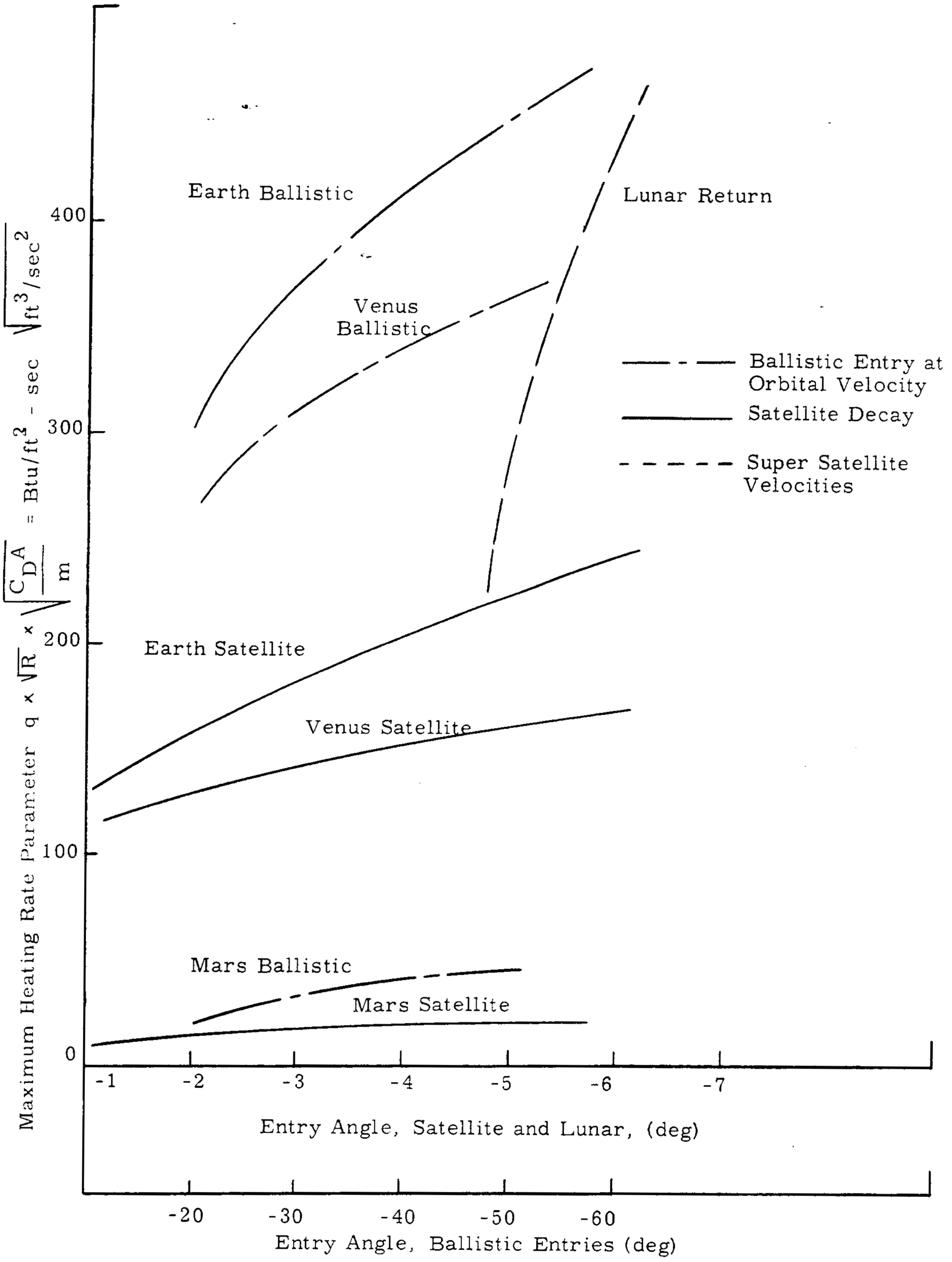


Figure 25. Comparison of Maximum Heating Rates for Various Reentries

[redacted]

[redacted]

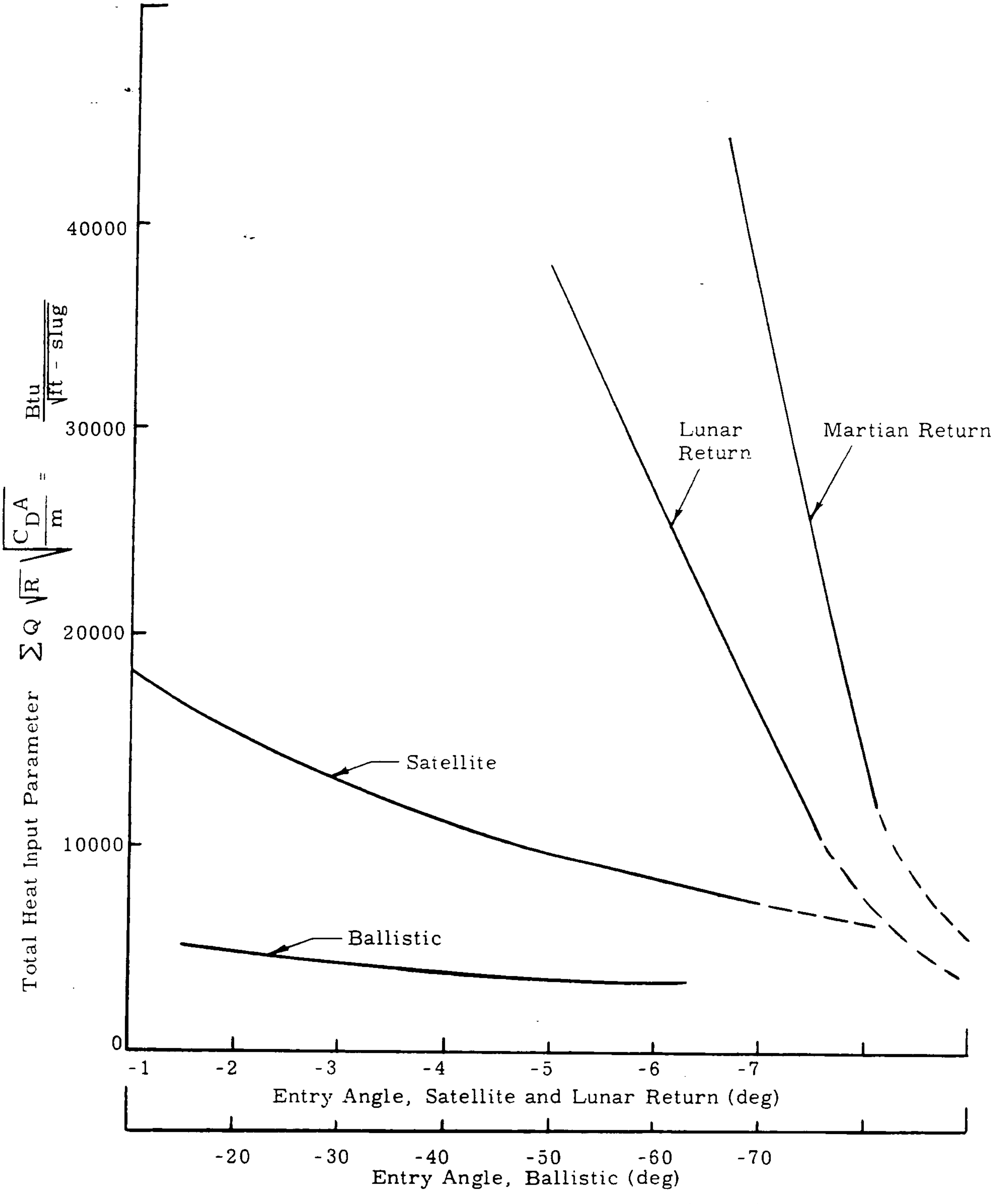


Figure 26. Approximation of the Total Heating Parameter for Various Earth Reentries

[redacted]

[redacted]

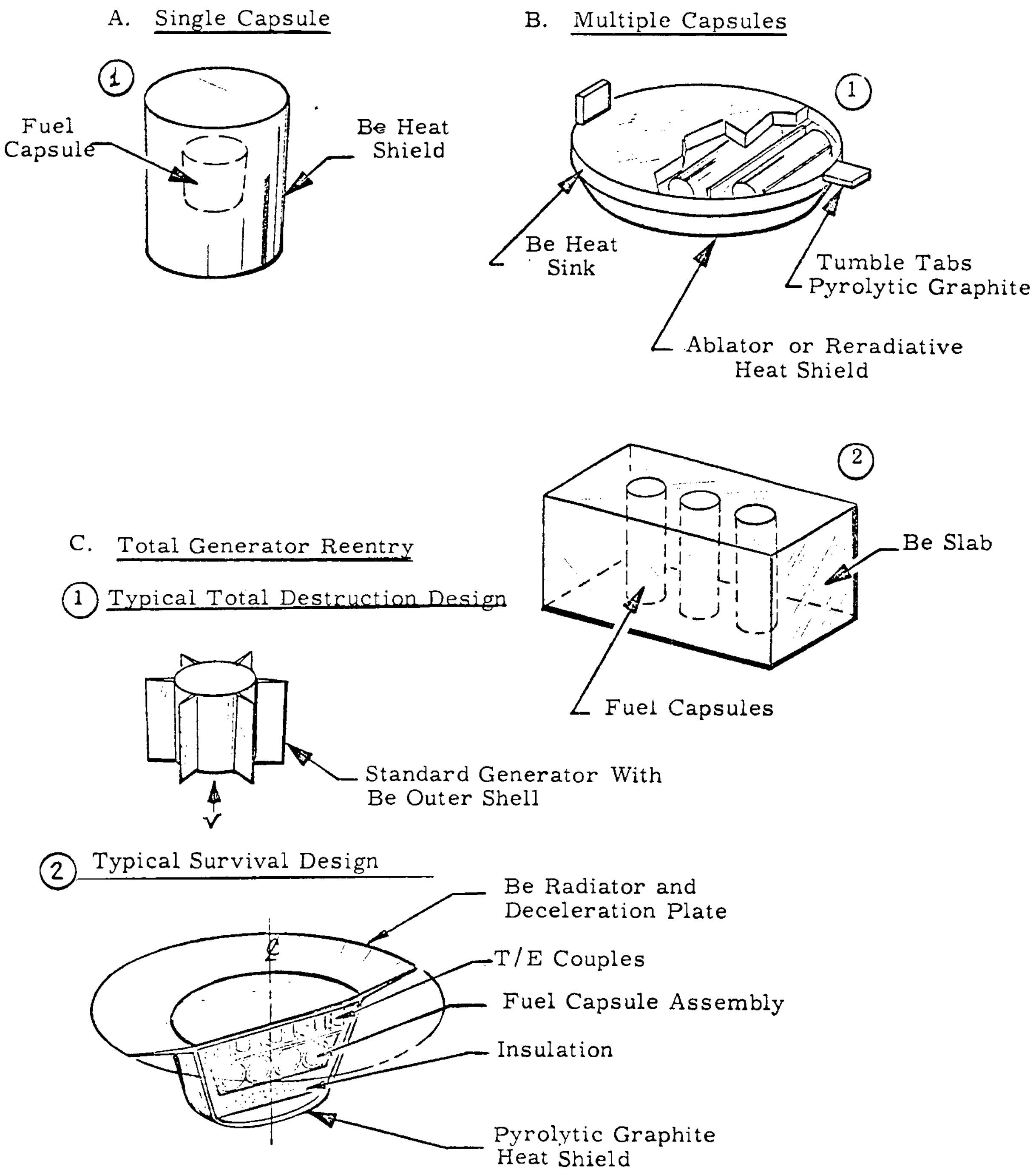

A. Single Capsule
①
Fuel Capsule
Be Heat Shield
B. Multiple Capsules
①
Be Heat Sink
Tumble Tabs Pyrolytic Graphite
Ablator or Reradiative Heat Shield
②
Be Slab
Fuel Capsules
C. Total Generator Reentry
① Typical Total Destruction Design
Standard Generator With Be Outer Shell
V
② Typical Survival Design
Be Radiator and Deceleration Plate
℄
T/E Couples
Fuel Capsule Assembly
Insulation
Pyrolytic Graphite Heat Shield

Figure 27. Intact Reentry Techniques of Fuel Block Assemblies for Small Radioisotopic Generators

[redacted]

[redacted]

In addition to the three choices mentioned above, the designer has open to him the selection of an active versus a passive reentry system. Active systems would make use of such devices as lifting bodies, parachutes or balloons, drag brakes, and devices which vary ballistic coefficient. Passive systems would depend only on their inherent thermal and geometric design to achieve intact reentry.

Since it has been shown that random intact reentry would be limited to the smaller heat sources or to reentry of discrete modules of a large heat source, it would seem most reasonable to adapt the passive system to achieve the goals outlined. Only when the heat sources get large enough to represent such a significant hazard upon intact reentry, that control is necessary, will active systems be justified. The inherent increase in weight and decrease in reliability attendant with active systems also penalize the smaller systems excessively relative to larger (kw) power supplies.

1) Reentry of the Individual Fuel Capsule. Maximum design flexibility would be obtained if it were possible to reenter individual fuel capsules safely regardless of the design of the subsystem or vehicle to which they were attached. A typical fuel capsule was chosen as representative of several generator configurations. It is a right circular cylinder 1.162 inch in diameter by 5.35 inches long, weighing 2.5 pounds and having a ballistic coefficient of 0.70 for random tumbling reentry. The capsule is assumed to reenter the earth's atmosphere in the larger heat source shell and to be released at 300,000 feet altitude and 25,000 ft/sec velocity which are typical dynamic parameters for this type of reentry. The heat source shell is assumed to be burned away at this height. The reentry angle at capsule release is $\phi_E = 1^c$.

The basic trajectory from this point on can be expressed by:

$$V = V_E \, e^{\frac{-B \rho_o e^{-\beta h}}{\beta \sin \phi_E}} \qquad (1)$$

The heat flux to a random tubling cylinder can be approximated by:

$$(q_{av}) \hat{=} \frac{2340}{\sqrt{R_N}} \left(\frac{\rho}{\rho_o} \right)^{1/2} \left(\frac{V}{V_E} \right)^{3.15} \qquad (2)$$

cold wall

[redacted]

where:

V = reentry velocity

V_E = circular velocity

B = ballistic parameter

ρ_o = sea level density

ρ = altitude density

R_N = characteristic radius

ϕ_E = reentry angle

β = 1962 atmosphere constant

Three typical heat shields can be applied to the capsule which represent heat sink, re-radiative and ablation types of cooling systems. Typical materials for these shields are beryllium, pyrolytic graphite, and phenolic resins, respectively.

However, the capsule heat shield for reentry must not severely compromise the normal heat flux from the capsule. If this is the case, lower overall thermal efficiency will result and the probability of capsule meltdown or over-temperature failures during normal operational transients is increased.

Since both pyrolytic graphite and the phenolic resins are inherently low thermal conductivity materials, beryllium looks most attractive from overall operational view.

The required thickness of beryllium must be iterated upon for survival of the fuel capsule from Equation (3):

$$t^3 + \left(\frac{4r_o + 1}{2}\right) t^2 + r_o (r_o + 1)\, t - \frac{\sum Q_{aero}}{C_p (T_m - T_o)} + \frac{H_f}{X} = 0 \quad \text{(Ref. 4)} \qquad (3)$$

Where:

t = shield thickness

r_o = capsule radius

$\sum Q$ = total aero heating

C_p = heat capacity

T_m = melt. temperature

T_o = original system temperature

H_f = heat of fusion

$\bar{X}$ = constant representing that portion of the heat of fusion allowed to enter shield

Assuming the equilibrium temperature of the shield is 1800°F for structural purposes and that it is allowed to absorb half its heat of fusion during reentry, its heat absorption capability is expressed by:

$$C_p (T_m - T_o) + \frac{H_f}{2} = 710 \text{ Btu/lb} \qquad (4)$$

The iterative solution yields a total capsule weight to survive satellite reentry of 15.8 pounds. In reality, the weight would be about 30 percent less if re-radiation were taken into account. In view of the bare capsule weight of 2.5 pounds, it appears that protection of individual fuel capsules through a heat sink technique is quite prohibitive from a weight standpoint, especially if the heat source contains several capsules. The penalty that small generators would pay is clearly evident.

Other special techniques for protecting the capsule during reentry were investigated, especially those using pyrolytic graphite. This material was of interest because it may be possible to design a heat dump system which could take heat from the capsule radially and transform it 90° to re-radiate on a converter. This property would also allow minimal heat input into the capsule during reentry. However, the problem of nuclear heat absorption during the reentry and heat rejection after ground impact on the ends of the radiation lamina, still remain a serious meltdown threat. In addition, even if these could be solved, the weight of this system based on preliminary calculations is greater than an equivalent beryllium system, due to the restrictions on the minimum thickness of pyrolytic graphite for a 90° bend.

In view of the weight penalty of both a beryllium heat sink shield and an undirectional graphite shield interest reverts to use of one of the alternate types of reentry configurations.

2) <u>Reentry of Total Fuel Block</u>. The same reasoning which was applied to reentry of a single fuel capsule can be extended to reentry of several capsules as a cluster. The choice of any heat shield for the cluster other than a heat sink type would interfere with optimum heat transfer during normal operations for the cluster as well as a single capsule. This, however, forces the designer to choose a material like beryllium and the weight penalties are much the same as for single capsule reentry.

There are some modifications in geometry, however, which can reduce total heat input during total fuel block reentry. These are shown in Figure 27. Based on heat source and converter integration, the maximum amount of surface area for heat transfer is desirable, hence a slab geometry with the fuel capsules buried in the slab represents a typical approach. Depending on the generator, the ballistic parameter $\frac{W}{C_D A}$ may be made small enough to ensure stabilization at a high altitude and hence conventional heat sinks and ablation systems can be used. If this is the case, minimum slab thickness can be obtained. Stablilization in an undersirable or metastable configuration can be avoided by simple tumble tabs made of pyrolytic graphite arranged around the periphery of the heat block slab. The tabs would yield a pitching moment in all configurations except nose forward.

3) <u>Reentry of the Total Generator.</u> Based on the arguments presented above, the minimum weight intact random reentry system seems to resolve itself in the form of reentering the total generator. Two philosophies are evident to achieve this result. The first is to allow the generator structure to absorb the reentry heating and be destroyed at a low enough altitude so that the remaining reentry energy will not destroy the fuel capsules. The second is to protect the entire generator assembly with a thermal system which can reject reentry energy at a rate fast enough to protect the support structure of the generator and keep the fuel capsules from melting down during the reentry.

The first approach is straight forward in nature and does not require any change in present generator design other than replacing the radiator fins and shell with a high heat capacity material such as beryllium. This will allow good heat transfer during normal operation and sufficient heat capacity to prevent capsule meltdown during reentry (Reference 26).

However, there are several basic flaws with this approach from a safety standpoint which make its use questionable. First, beryllium which is the only material capable of playing the dual role of a good thermal conductor and good heat sink is quite weak at the average surface temperatures encountered during reentry ($\sim 2000^{\circ}F$), and an increase in present material design strength is required. This could take the form of composite structures or the addition of ceramic fibers to the basic metal to increase its high temperature strength. At present no such modifications to basic beryllium designs are in production. Therefore, the questions of whether the generator structure will stay together long enough to absorb the majority of the reentry heating as anticipated is not clear. This is not too critical in a ballistic abort due to the short heating times, but does become important as the aborts get closer to reentry angles reminiscent of the satellite decay case. If increase in strength of the beryllium has to be accomplished by lower temperatures, it will reflect back to increased weight in the system under the same reentry conditions. Hence, it

[redacted]

is not clear that a minimum weight system will result from this approach. Secondly, even if the reentry is successful and the fuel capsules are released at an altitude where they can survive the remaining aerodynamic heating, high impact velocities and the possibility of ground burial further complicates this technique. Based on past experience with low temperature (~1500°F) thermoelectric systems, enough impact strength can be designed into the system to have reasonable assurance of survival. However, this is not the case with the higher temperature systems envisioned for future use, especially thermionic systems. Some structure around the fuel capsule must remain to absorb impact energies. Finally, in any case of free capsule impact, ground burial is a serious problem which could yield meltdown and release of the fuel form if it occurred. Drop tests conducted by Atomics International (Reference 27) indicate burial in typical pasture soil of up to 18 inches can occur with the fuel capsules of L/D=4. In addition, degradation of the impact strength of the fuel capsules after abort residence in space under high temperatures and vacuum conditions, is a definite possibility. Appendix G shows that if this factor is credible, survival of the base capsule at impact may be questionable.

The use of a high melting temperature fuel form, however, may reduce this problem to acceptable limits. Presently, intact reentry coupled to PuO_2 fuel forms, via SNAP-27, is being considered. The most serious problem to be resolved in this case is the definition of meltdown of the oxide upon burial. If this has a possibility of occurring, then the insolubility of the fuel form is threatened and its usefulness in question.

Therefore, since a conservative philosophy relative to eventual hazards from intact reentry must be adapted, it would seem that the techniques of maintaining the generator structure and using it both as a reentry body, impact energy absorber and preventative to ground burial, is the most reasonable alternate for intact reentry of the smaller radioisotope systems.

c. Thermal Protection Problems of Intact Reentry Systems. An integrated approach must be used in designing the thermal protection system for any intact reentry system. This approach must be able to resolve three basic problems:

1. Prevention of meltdown due to loss of coolant or heat rejection capability in space.
2. Meltdown during reentry due to inability of the system to absorb nuclear heat.
3. Protection of the fuel capsules from the reentry heating proper.

[redacted]

[redacted]

1) Loss of Heat Rejection Capability in Space. If the system cannot dump its heat to space, meltdown of the fuel capsules will surely occur and intact reentry of this system would create a serious hazard since integrity of the reentry structure upon impact cannot be assured. For the small systems which fall into the category of possible random intact reentry, active coolant loops are not anticipated, hence, the problem of heat rejection in space is a simple one depending on basic radiation to the converter and consequent dumping to space from the backside of the converter. Therefore, unless the radiator is covered, destroyed, or the view factor modified by aborts which would deform the heat rejection system, small systems should not suffer from this problem.

In larger systems, this would be a problem and is discussed in a later section of this report.

2) Nuclear Heat Absorption During Reentry. In view of the high weight penalties for individual capsule reentry, it is evident that the more attractive solutions will involve reentry of integrated structures. If a reentry design is to consist of a multiplicity of capsules within a reentry shield, which must function at least in part as a thermal insulator, it is of interest to investigate the ability of the fuel capsules to survive during the period of reentry without any cooling. This is computed by calculating the rise in temperature of a capsule during a reentry of 500 seconds duration, typical of the time from 400, 000 to 100, 000 feet during satellite reentry. Assuming all the decay heat goes to raise the capsule temperature above a chosen maximum operating temperature of 1800°F. The results are as follows for a 100 W(e) thermoelectric system :

No. of Fuel Capsules	Fuel Form	Aprox. Max. Equilibrium Temperature	
4	$Pu^{238}O_2$	$T_{max} \simeq 2200^{\circ}F$	No extra heat capacity needed
4	$Cm^{244}O_2$	$T_{max} \gtrsim 3000^{\circ}F$	Small increase in support structure weight will suffice
1	$Po^{210}Gd$	$T_{max} \gtrsim 5800^{\circ}F$	Large heat capacity required

[redacted]

It may thus be noted that a typical Haynes 25 capsule may have sufficient heat capacity to survive when fueled with Pu-238 and $Cm-244O_2$ but not with Po-210 Gd suggesting that one useful design modification would be to increase the heat capacity of a Po-210 Gd generator, if intact reentry is required.

This is simply done by increasing the capsule weight, and/or structural mass. Therefore, for small generators overheating from decay during reentry does not seem a problem.

3) Reentry Heat Protection Systems. To minimize the weight of the reentry heat protection systems required for intact random reentry a study was undertaken to define the relationship between the reentry parameters controlling heat protection system selection. These parameters are maximum stagnation heat flux; total heat input per unit frontal area, ballistic coefficient, reentry angle and lift/drag (L/D) ratio of the configurations. All these parameters can be presented as shown in Figure 28 which relates maximum heating rate to total load as a function of ballistic coefficient during the reentry. The figure shows present operating regimes for both ablative and radiative heat protection techniques. The line for the sublimation of graphite at $T = 7000^\circ R$ is superimposed on the figure to indicate an absolute materials limit on heat flux for the use of radiation systems. The area of operation of the reentry vehicles for the heat sources of interest in this study is also shown as the shaded zone. From Figure 28 it can be seen that radiation systems are most suitable for vehicles with low heating rates, i. e., heating rates low enough so that the reradiation temperature can permit use of current materials. For normal satellite and lifting body reentry times this yields a limiting surface temperature of $\lesssim 4500^\circ R$. Back side temperatures of the heat protection systems can thus usually be kept below $1800^\circ F$ and hence state-of-the-art structural materials will still function. Ablation systems are not limited by heating rate but rather by total heat load, because of the direct relation between that and the weight requirement, they are usually thought of for use with relatively rapid deceleration systems in the lower atmosphere (ballistic nose cones).

Therefore, for the range of ballistic coefficient ($1 < W/C_DA < 100$) shown on Figure 28 as representative of the class of small nuclear device reentry bodies of interest, a reradiation system such as pyrolytic graphite seems most attractive for satellite decay. However, for cis-lunar return, an ablation system is definitely required. Since aborts yielding both types of reentries could occur on a lunar mission, heat protection systems with the advantages of an ablator and reradiator could be used. Such systems as char ablators, i. e., insul cork would have these properties.

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Modified from Data in Reference 25

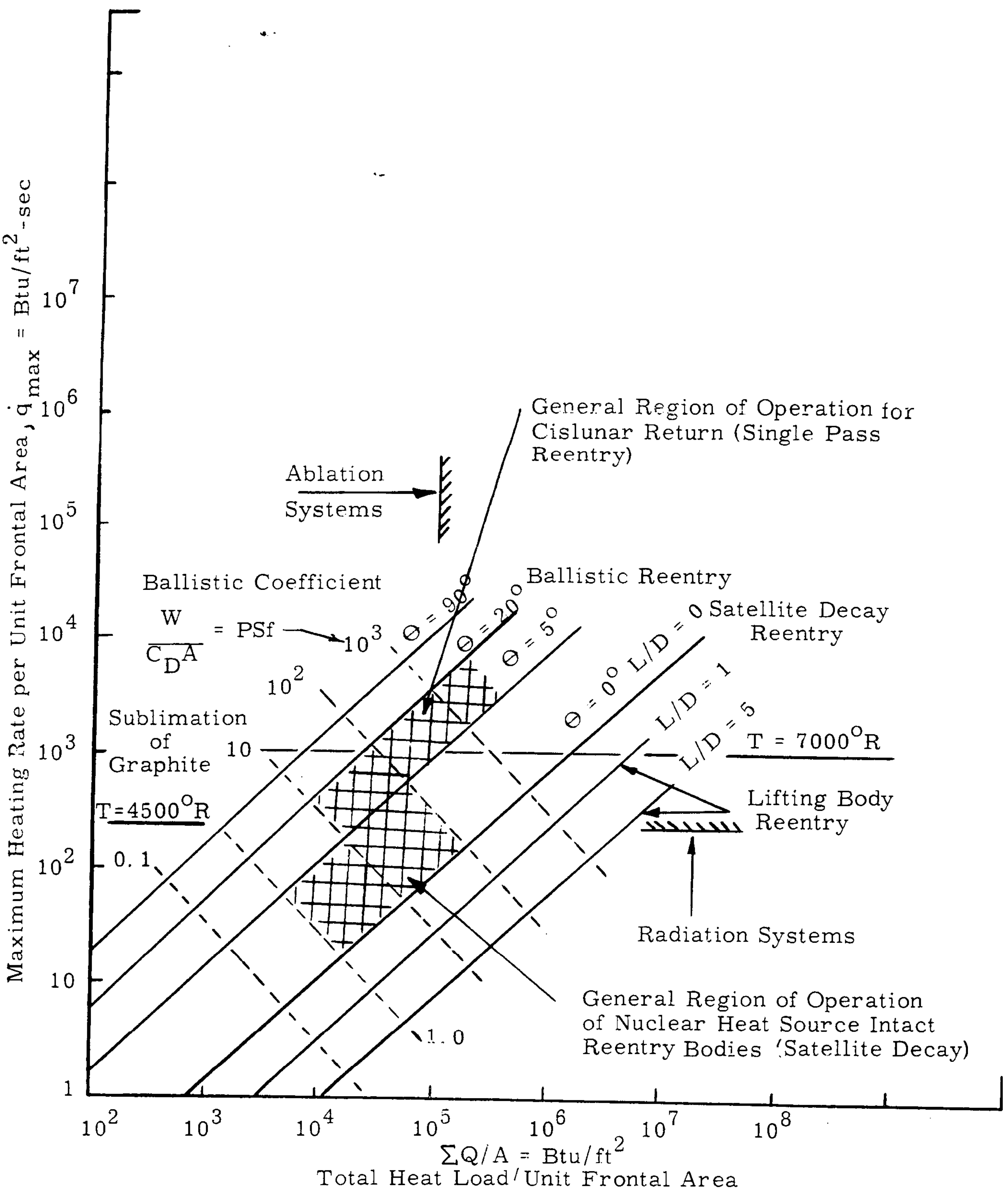

Figure 28. Maximum Heating Rate and Total Heat Load During Entry Into Earth's Atmosphere

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Figure 28 also allows some basic conclusions on the concept of using a lifting body as a reentry vehicle. From the thermal standpoint the total heat load for the same ballistic coefficient vehicle is increased due to the longer reentry times and very close control over the reentry trajectory may be required in order not to exceed peak heat flux levels. However, reradiation systems are definitely indicated and pyrolytic graphite should be capable of handling the thermal loads fairly easily for ballistic coefficients up to 100 and L/D's above 0.5 Figure 29 shows the approximate ablation cooling system weight requirements for atmospheric entry at orbital velocity.

d. General Design Considerations of a Typical Reentry Vehicle.

1) The General Configuration. Figure 30 shows the effect of increasing the ballistic coefficient on the total reentry heating input parameter $\sum Q \sqrt{R_N}$, during satellite reentry. The implications clearly show that to reduce the total heat load a configuration must absorb, its ballistic coefficient should be maximized. This can be achieved practically by maximizing the drag and hence the profile area of the configuration. With the drag profile maximized, the reentry vehicle will undergo maximum decelerations at high altitudes, thereby reducing peak heating through an early velocity reduction and yielding lower velocity profile in the higher density lower atmosphere.

Three general passive configurations shown in Figure 31 immediately appear feasible. They are typical of the present Mercury - Gemini-Apollo type reentry vehicles, present ICBM warheads, and planetary lander reentry vehicle design, respectively. The Mercury-Gemini type configurations are inherently unstable at high altitudes because of the close proximity of the center of pressure and center of gravity of the vehicle. They would definitely need an active attitude control system to prevent tumbling or they would have to be spun stabilized prior to the reentry and this may be an unacceptable design contraint for small heat source intact reentry bodies from the standpoint of added weight, volume and an inherent decrease in reliability.

Therefore, the general ICBM and/or planetary lander type vehicles are most attractive for the smaller reentry systems because of this inherent drag stability. A reference configuration, Figure 31(d) is shown for purposes of discussion in this study.

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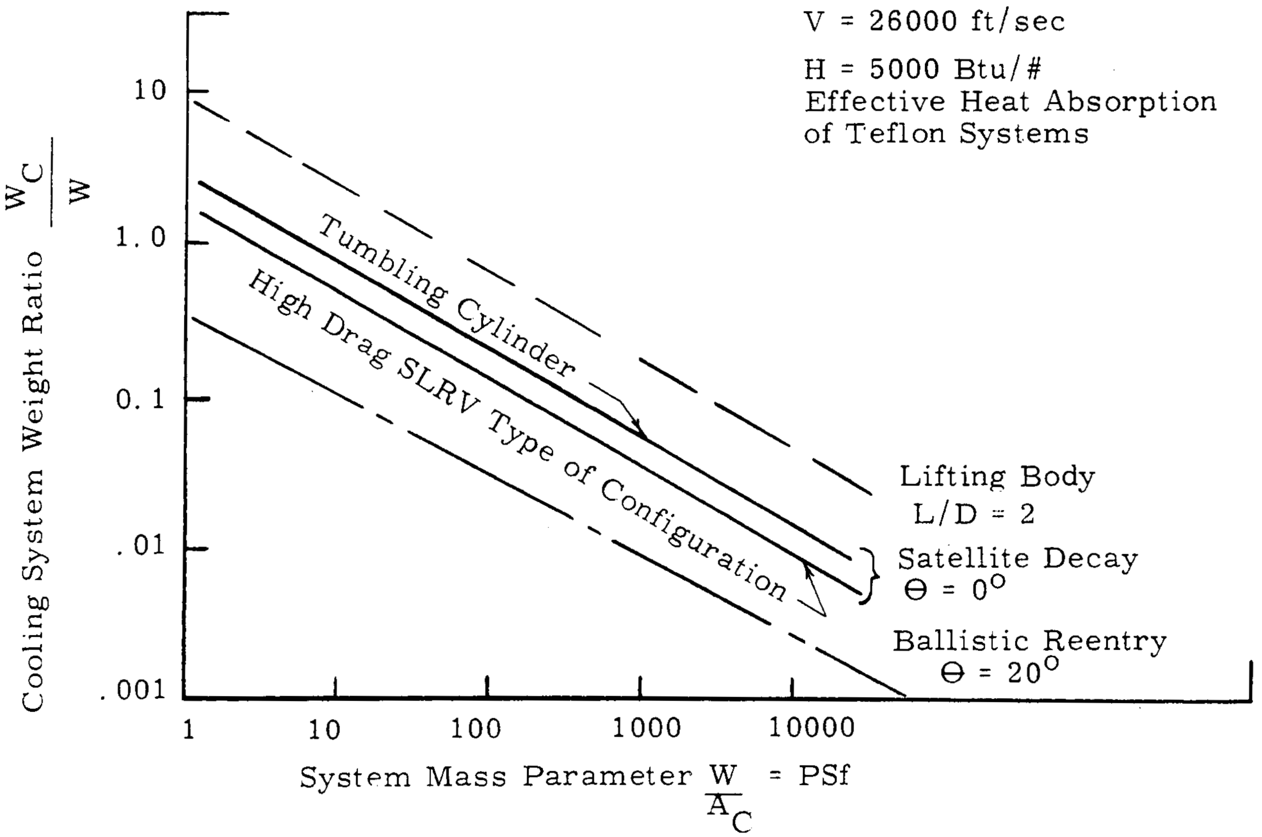

Figure 29. Ablation Cooling System Weight Requirements for Atmospheric Entry at Orbital Velocity

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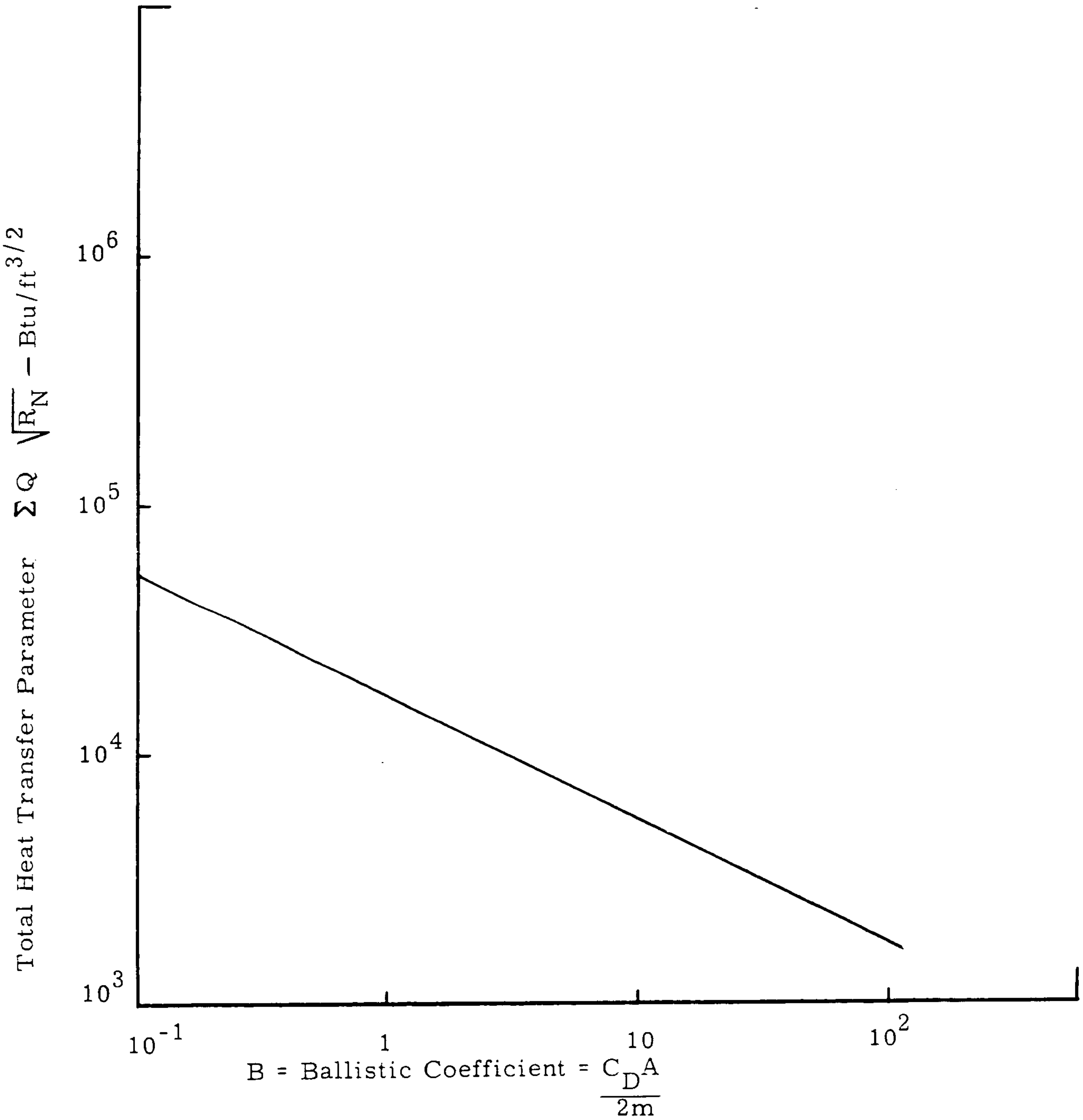

Figure 30. Effect of Ballistic Coefficient on Integrated Heating Duri [redacted]

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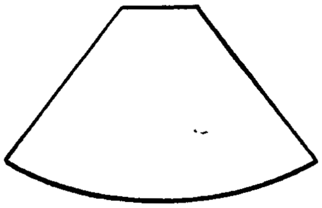
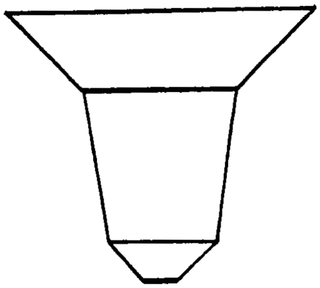
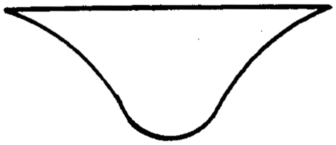


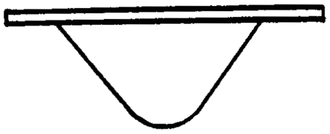
(a) Mercury - Gemini



(b) ICBM Nose Cone



(c) Planetary Landers



(d) Reference Configuration

Figure 31. Typical Reentry Configurations Which Minimize Total Heat Input During Satellite Reentry

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2) The Expected Trajectory. Before the heating and dynamics of any reentry conditions can be defined, conditions of velocity, flight path angle and altitude must be known. From these and the ballistic parameters of the configuration a full trajectory chronology can be derived. Such a derivation is presented in Figure 32 for the reference configuration of Figure 31 with a ballistic coefficient of B = 2.5 during satellite reentry. The data was compiled from Reference 39 and is typical of a 30 watt electric radioisotope generator.

3) Stability of the Reference Configuration. Because of the random nature of the expected aborts which would lead to satellite reentry of the heat source, a stable mode of reentry (nose first) cannot be considered to be the most logical. Indeed, if any heat source reenters, it most probably will be with the structure it services as a power supply. Upon separation from this structure, due to aerodynamic heating or deceleration forces, a component of rotational momentum is bound to be transferred to the reentry body. Hence, tumbling must be considered as the initial reentry mode.

In order to minimize the reentry heat shield weights and design the complete thermal protection system, the altitude at which tumbling can be damped and the total heat input into the unprotected rear end of the vehicle must be computed. If this heat load is too severe for the thermal capacity of the unshielded rear structure of the reentry body to absorb, then redesign of the basic aerodynamic configuration and addition of protection systems must be initiated. In real heat source designs, the rear side of the drag plate is an ideal radiation surface for dumping the normal thermal load in space. Hence, insulation will most likely not be provided on this surface. Figure 33 derived from analyses in Reference 29, and applied to the reference configuration, shows the effects of initial tumbling rate on damping altitude. For a ballistic coefficient of B = 2.5 the average damping potential indicates stability above 300,000 feet for initial tumbling rates of 1 rps or below. These results are compatible with observed ICBM decoy warhead observations.

Meltdown of the plate during tumbling flight could expose the fuel capsules to the reentry environment. Therefore, the average heat flux on the back of the drag plate in the reference configuration was computed and is shown in Figure 34. It is seen that damping of this configuration can be made to occur at altitudes above where peak heating takes place and hence the drag plate can be made to survive this phase of the reentry without a large weight penalty.

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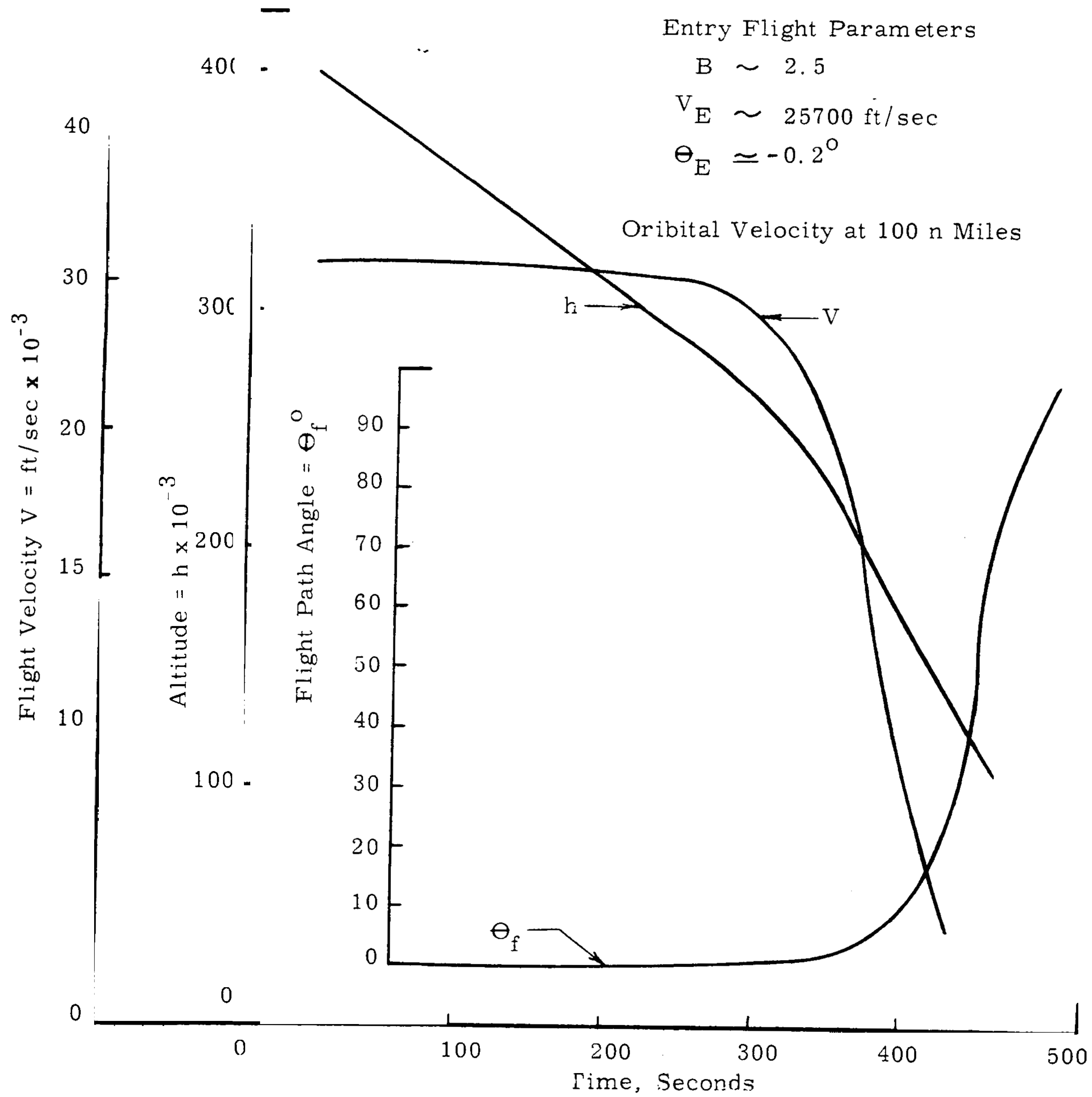

Figure 32. Typical Trajectory for the Reference Configuration Satellite Decay Reentry

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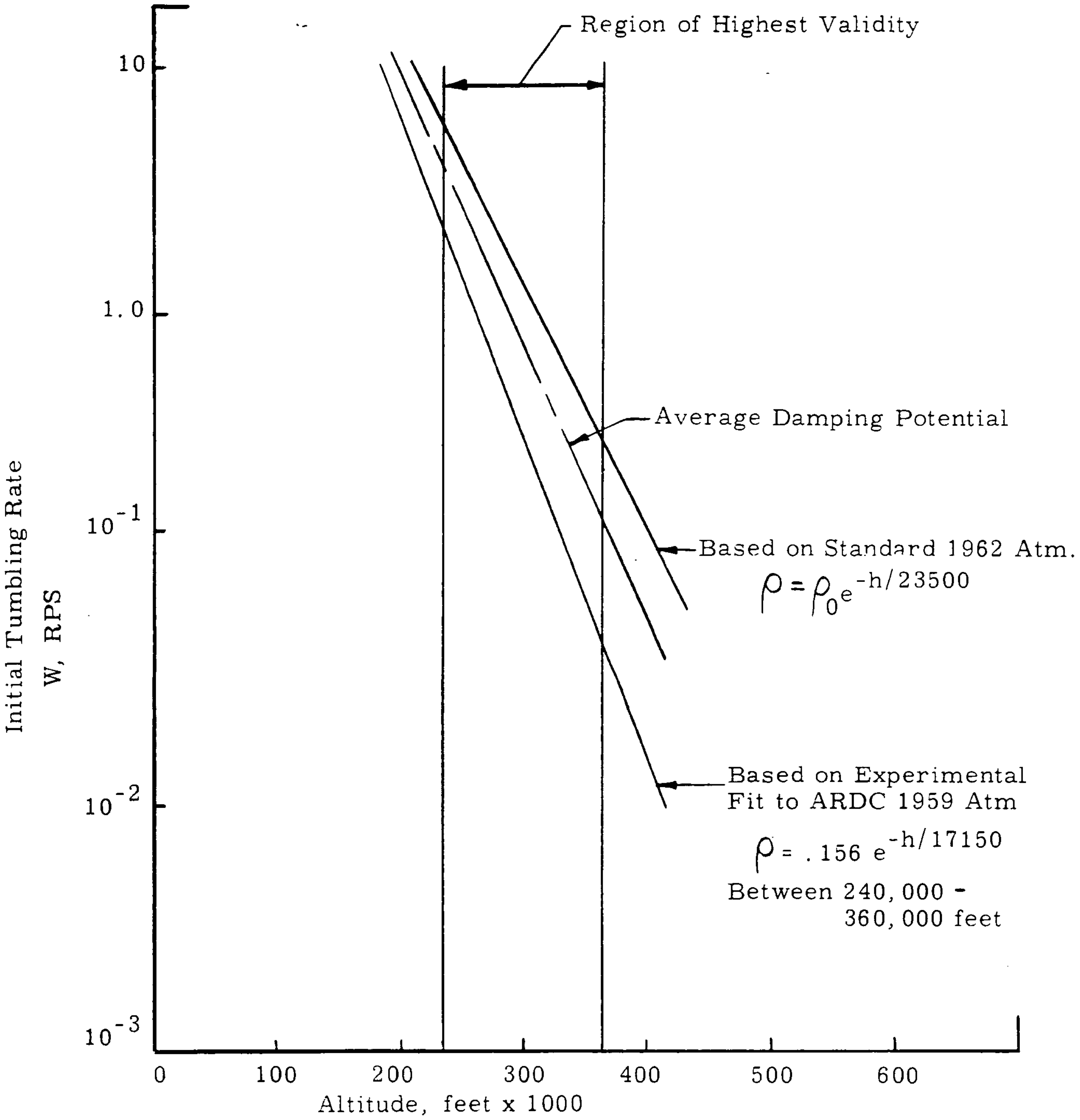

Figure 33. Effects of Initial Tumbling Rate on Damping Altitude of Reference Configuration

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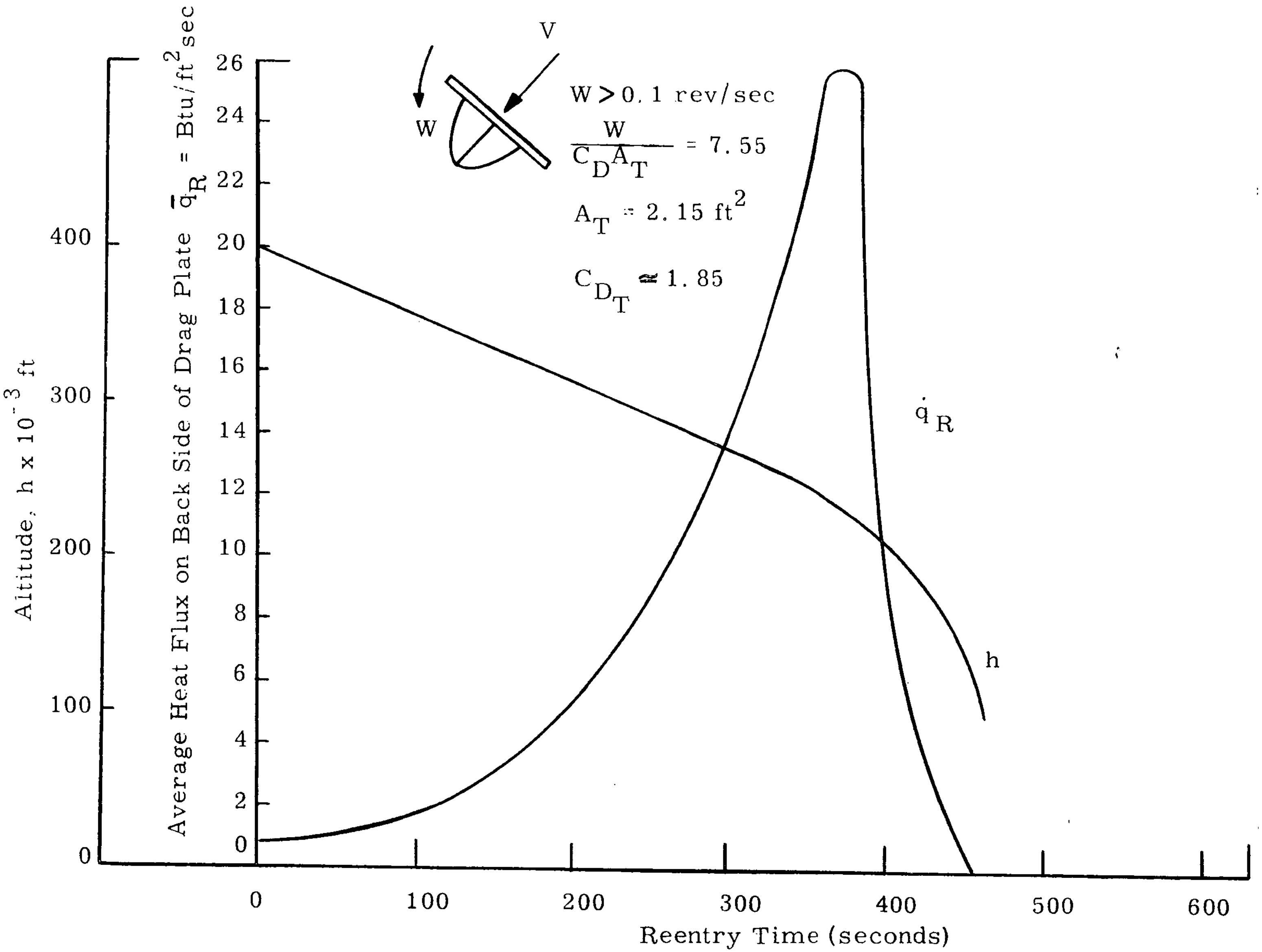

Figure 34. Typical Heat Flux on Back of Drag Plate of a Tumbling Reference Body During Satellite Reentry

4) Typical Heating Rates in Stable Flight and Choice of Reentry Heat Shield Materials. After stabilization, surface heat flux profiles on the reentry configurations of interest can now be defined and used as inputs to the design of the required heat protection system. Figure 35 shows the expected heat fluxes for the reference configuration considered versus reentry flight time. The peak cold wall heat flux at the stagnation point is approximately 130 Btu/ft^2-sec and the integrated heating is of the order of 20,000 Btu/ft^2. The W/C_DA is equal to 2.5 putting this reference configuration squarely in the satellite intact reentry vehicle operation band predicted in Figure 30.

Since reradiation has been determined to be the desired heat dump mechanism during the heat source reentry, high emissivities and low thermal conductivities are the most desirable characteristics of the heat shield. Pyrolytic graphite with an effective emissivity of unity and a thermal conductivity normal to the surface of between 0.2 and 0.7 Btu/hr Ft - ^oF is the most logical choice.

Other materials such as gaseous or char ablations also show promise for heat protection systems. However, materials which are impregnated with a high vapor pressure filler which will vaporize upon exposure to reentry heating and release a gas must have coatings for long time and high temperature operation in space. Since the heat shield could be called upon to function after a considerable residence time in space much of the impregnated vapor shield could be already lost before reentry.

In general, foamed metal shields impregnated with high vapor pressure fillers such as metal chlorides are still slightly heavier than pyrolytic graphite shields for satellite decay reentries. For the critical case of cis-lunar return at angles of 7-8^o this technique is better than typical teflon ablators and graphite but is still heavier than systems using char ablators like insul-cork.

5) Oxidation Performance of Pyrolytic Graphite. The only other reentry consideration which might affect the choice of pyrolytic graphite as a heat shield material is the susceptibility of the graphite to oxidation in the reentry environment and the addition to the total heat load of that amount of heat released by the graphite while oxidizing.

In an earlier study of the subject completed in Reference 29, it was determined that the contribution to the total heat flux due to the oxidation of the pyrolytic graphite shield at its stagnation point is only 11 Btu/ft^2 - sec as a maximum in a reentry of the reference configuration. This is less than 10 percent of the

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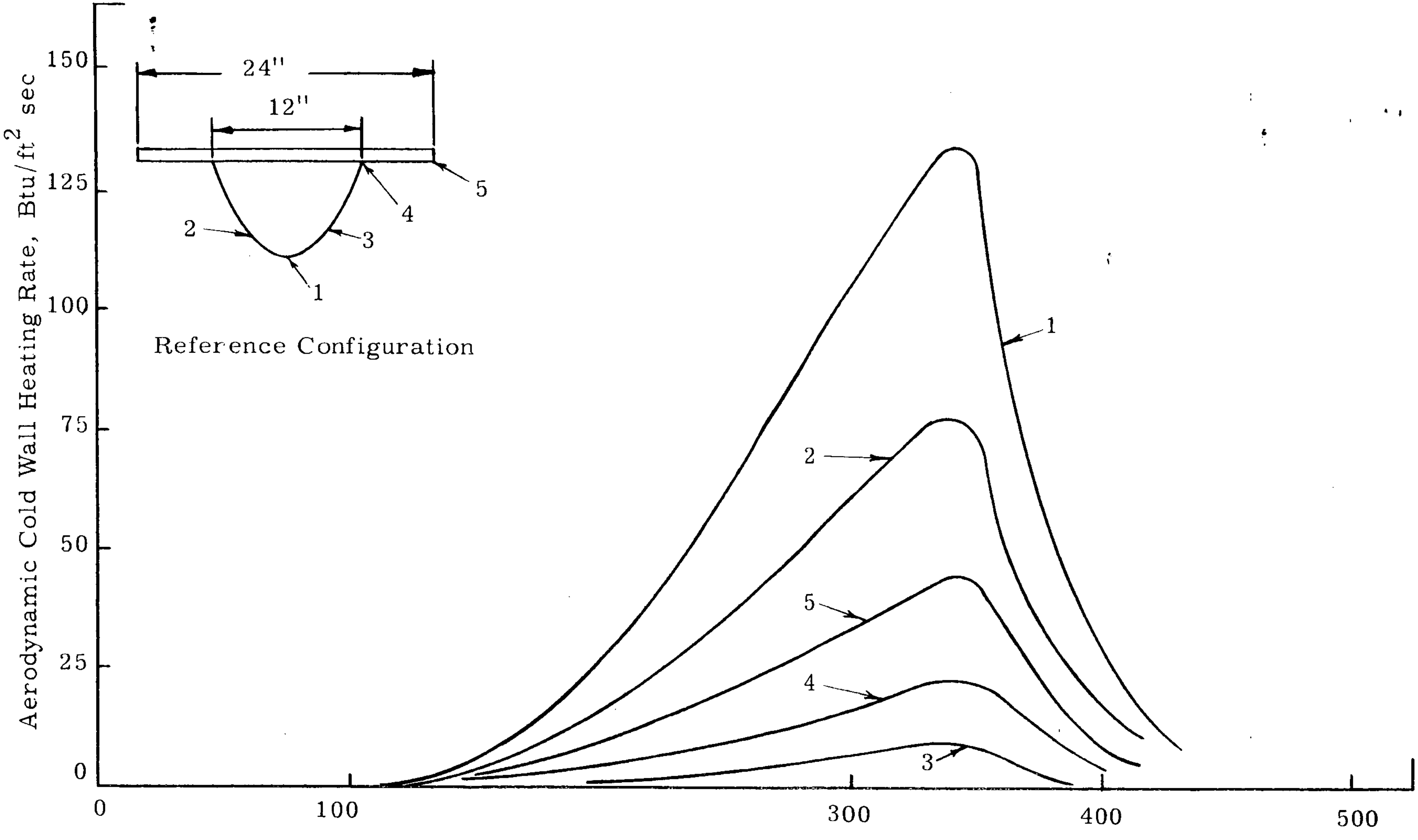

Figure 35. Surface Heat Flux Profiles on Reference Reentry Body During Satellite Decay Reentry

maximum aerodynamic heat flux. In the same study, it was determined that the mass loss rate of the pyrolytic graphite shield due to oxidation under reentry conditions of the reference configuration 2 x 10^{-3} pounds/ft^2-sec as a maximum. Integration over the complete reentry down to 100,000 ft yields an average loss of the order of only 0.01 inch from the stagnation region.

Therefore, it can be concluded that the pyrolytic graphite shields are not oxidation limited. However burning of the shield has been observed after reentry heating is completed in the terminal stages of the flight but effects on the integrity of the shell are minimal and impact or burial performance is not affected.

6) Required Thickness of the Pyrolytic Graphite Shields. Using the conservative assumptions of Reference 39 the thickness can be determined by reducing the problem to one of transient heat conduction with a variable temperature on one surface and a perfectly insulated inner surface. This problem can be solved analytically by use of Duhamel's theorem (Reference 30). The basic procedure is to take a solution with a constant surface temperature and perform the partial differentiations and integrations necessary to yield the solution with a variable surface temperature. The details of the technique and the expression relating the temperature within the body as a function of time and location can be found in Appendix B of Reference 29.

The solution for the satellite reentry of the reference configuration yield stagnation point thicknesses of about 1/2 inch which are conservative since the surface temperatures were based on emissivities of about 0.7. The approximate weight of the total shield including nose, forebody, and forward portion of the drag plate on the reference design amounted to between 15 and 20 percent of the total system weight.

For larger systems and emissivities it is expected that this would drop to 5 to 10 percent of total system weight for heat source reentry bodies of the order of 1000 pounds.

7) Dynamic Loading of the Drag Plate. During the various types of reentries which the heat source reentry body will be called upon to survive considerable aerodynamic loads can be expected on the drag plate. Since this plate is the main source of stability of the system at high altitudes, it is important that it remain intact.

Figure 36 shows the magnitude of the dynamic loads which may be expected on the reference design for typical ballistic and satellite decay aborts.

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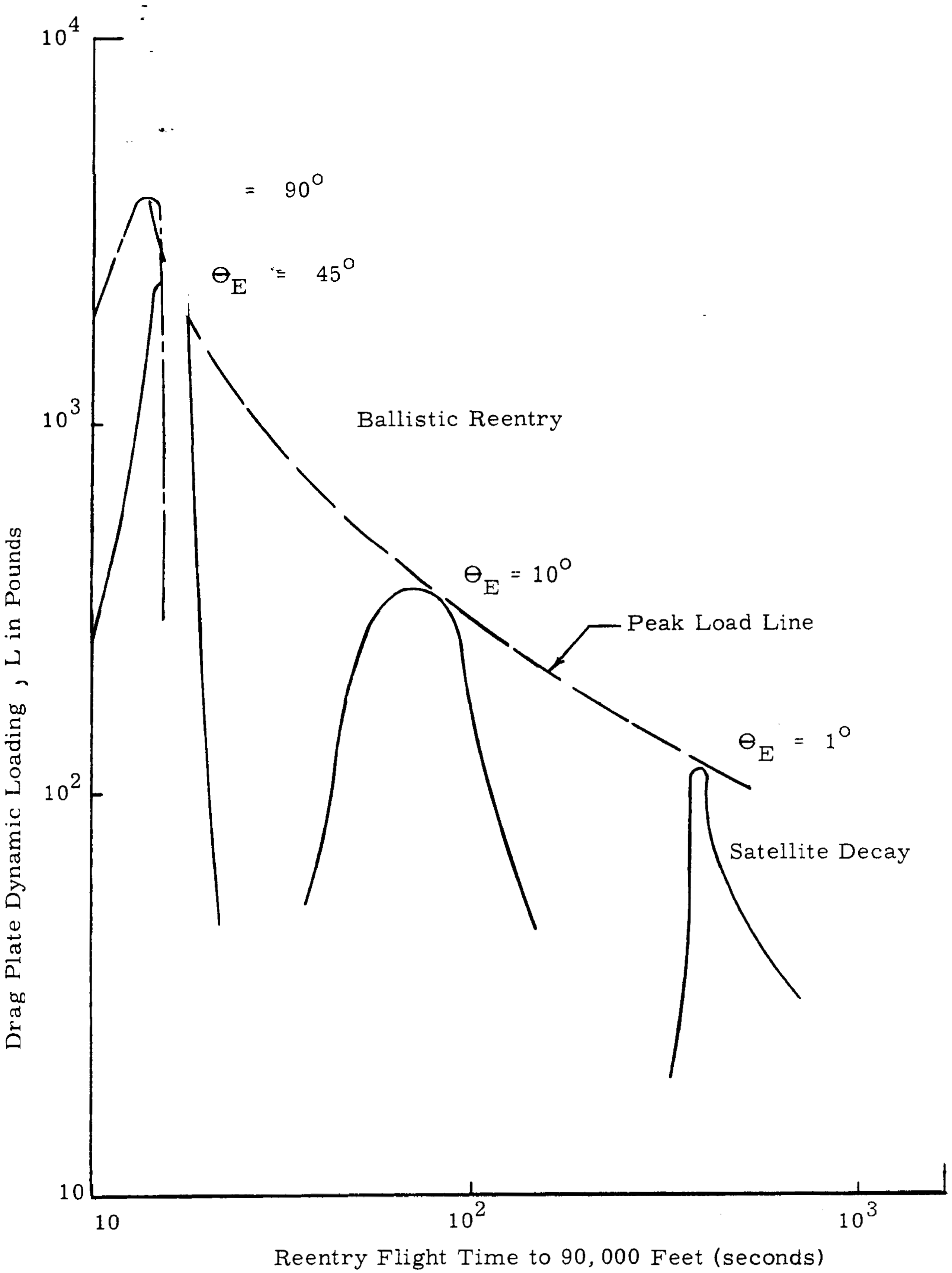

Figure 36. Magnitude of Total Dynamic Loading on the Reference Configuration During Ballistic and Satellite Decay Reentry.

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Clearly structural problems may exist during ballistic reentry and must be solved in design.

8) Structural Design Considerations for Reentry Bodies. A definite problem with entry vehicles of the type in question is the rapid growth of the structural weight if the ballistic parameter must be reduced due to design considerations other than the reentry. Analyses have recently been made (Reference 3) to determine new vehicle shapes which minimize structural weight growth while retaining the desired high aerodynamic drag and stability coefficients.

A conventional blunt nose conical shell under the combined action of aerodynamic and inertial loadings during deceleration develops circumferential compressive stresses. Structurally, more efficient shapes such as illustrated below, in Figure 37 by the dashed line, have been derived which are subject to tensile rather than compressive stresses in the side walls under entry loadings. Compressive forces are carried in a ring at the aft end, a condition highly favorable to reducing the weight of the overall design. The resulting shell is basically a tension member having less surface area than the original conical shell and a significantly higher drag coefficient. However, volume does suffer somewhat from this approach.

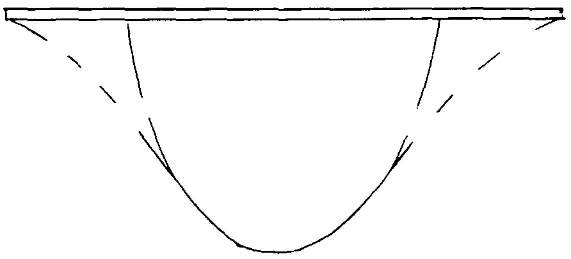

Figure 37. Application of Tension Shell Technique to Reference Configuration

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e. Intact Random Reentry With Delayed Recovery. An interesting technique with application to those small systems which would require large amounts of nuclear shielding (i. e., Sr-90 and Cs-137) for safe random intact reentry or to recover light weight experimental nuclear systems after significant operation in space has been investigated in the course of the present study and is shown schematically in Figure 38.

The basis of the technique as described in Section III is to deploy a balloon system after the reentry body is no longer exposed to an appreciable heat flux. The balloon acts to decelerate and eventually float the intact reentry system at an altitude where ground hazards are minimal and where aircraft recovery can be employed. The balloon is aluminized and acts as a radar beacon. The aircraft acquires the balloon on radar, makes the pickup via the same techniques used in the Discoverer series of experiments, and reels the system into a shielded compartment returning it to a facility capable of handling the system.

This is not a passive technique of the type described in earlier sections and hence questions of reliability of the deployment system arise immediately.

The question of active versus passive systems begins to arise, whenever semi-controlled, controlled or recoverable systems are required. As will be seen in this study, the only way to achieve the degree of control necessary to make large radioisotope intact reentry systems safe or to use isotopes requiring heavy shielding is through the use of an active system.

Therefore, attempts at evaluation of active systems are considered strongly in the overall Aerospace Nuclear Safety analyses presented in this study.

The feasibility of such a delayed recovery system is quickly defined by fixing the weight of the floating system. For any particular equilibrium altitude, the temperature, pressure, and density corresponding to that altitude coupled with the floating system weight allow the balloon volume, and number of moles of suspension gas to be calculated. Figure 39 shows the balloon volume, gas weight, and pressure vessel size and weight as a function of floating system weight. The pressure vessel radius was limited to one foot to keep the volume consistent with launch vehicle capabilities for small systems. Titanium was chosen as the pressure vessel material based on its light weight at temperatures of the order of 600°F which was chosen as maximum design temperature limit for material temperatures of the reentry body during atmosphere penetration. Preliminary analysis of this concept has shown that the pressure vessel weight is the controlling factor in the design. Typically for a 15 pound final floating system weight at least 30 pounds of jettisonable tankage is required.

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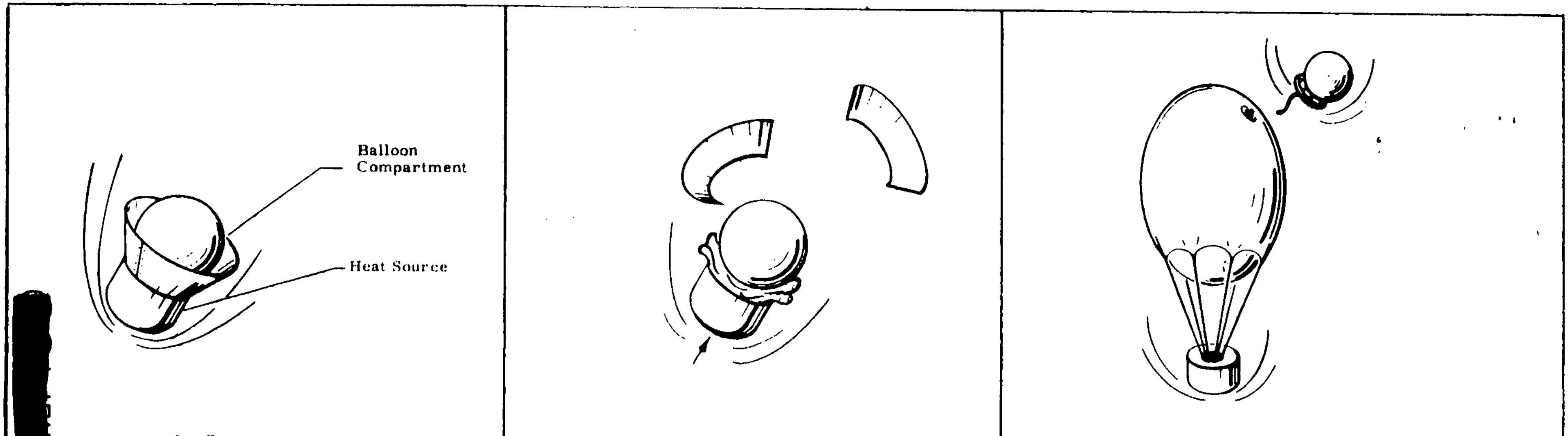
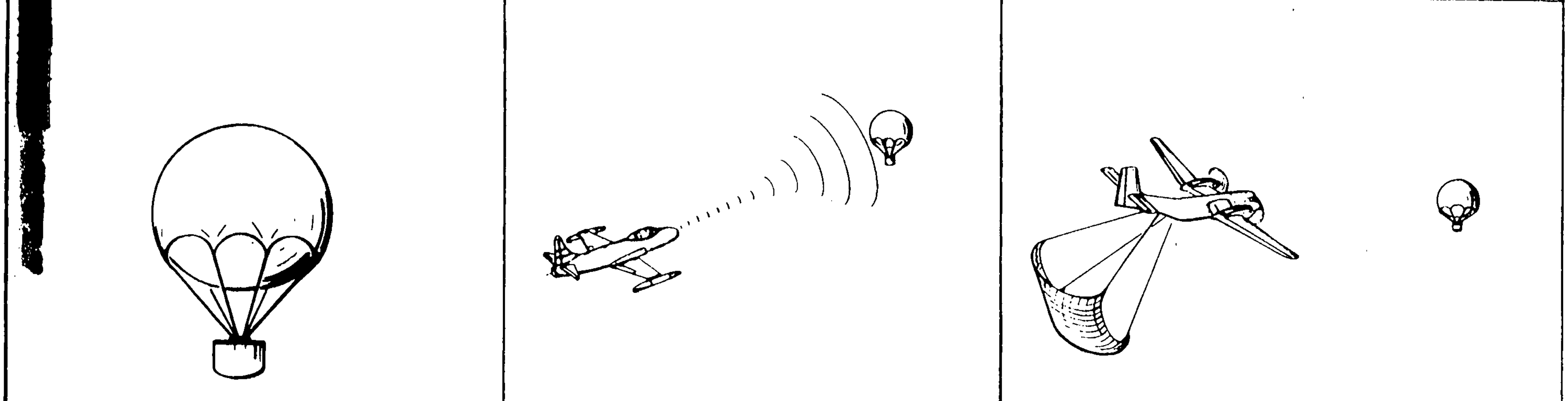


1. Reentry

2. Ejection of Balloon Compartment Shroud Below 100,000 Feet

3. Balloon Deployment and Ejection of He Tankage - 50,000 Feet



4. Suspension of Fuel Block at Approximately 15,000 Feet

5. Aircraft Acquisition

6. Aircraft Capture

Figure 38 Intact Random Reentry With Delayed Recovery Sequence of Events

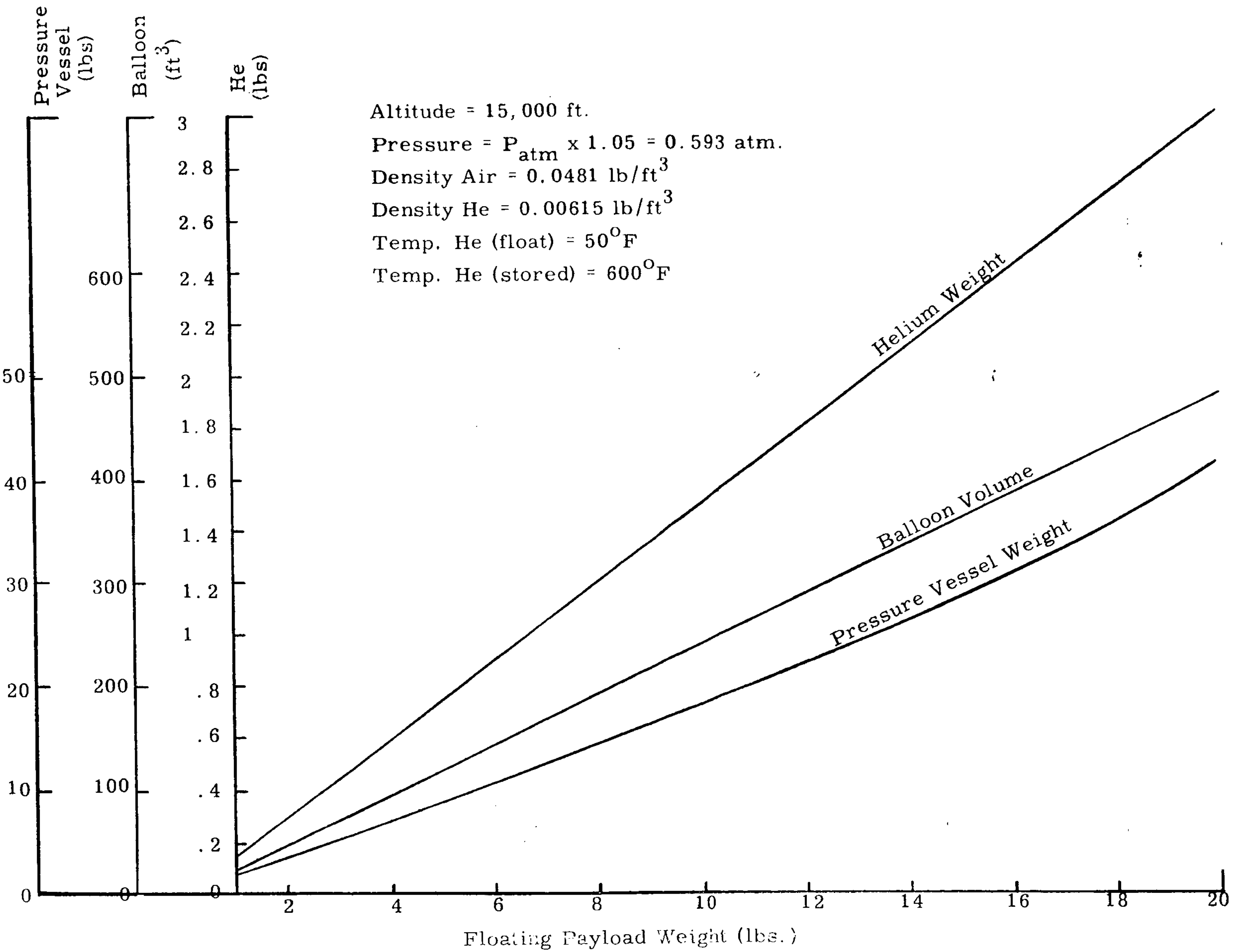


Figure 39. Component Weights vs Floating Payload Weights for Intact Random Reentry With Delayed Recovery

Therefore, the application of this technique would not be too applicable to systems which were launch weight limited. In addition, reentry of the heavier configuration would require more thermal protection.

Hence, this technique is obviously limited to small systems due to the penalties described above. In addition, such problems as the effects of storms and mountains on the eventual safety of the system plus its normal lifetime at 15,000 - 20,000 feet further complicate the concept. However, it does represent a developable system, well within the state-of-the-art, and if it can be made reliable, represents a valuable technique to recover small nuclear or non-nuclear systems which have operated in space for long periods of time

f. General Conclusions Uncontrolled Random Intact Reentry.

1) The controlling factor in application of uncontrolled random reentry is the magnitude and nature of the radioactive inventory being returned to earth. Because of this uncontrolled random reentry is limited to the lower power radioisotope systems. The increase in radiation shielding weights for general population protection due to use of Sr-90 for example could make uncontrolled random intact reentry unattractive even at these power levels.

2) Pu-238, Po-210 and Pm-147 fuel forms seem to represent the best compromises of minimum weight random intact reentry system and minimum hazards to the general public if impact in an inhabited area results.

3) In general, the total heat loads expected for a Martian or Venusian planetary entry would be less than those experienced in a normal earth satellite decay reentry. Hence. designing the heat source for earth reentry represents a good design limit even for planetary operations.

4) The critical reentry cases uncovered by this study are earth return at super satellite speeds from cis-lunar and planetary missions.

5) Uncontrolled random intact reentry of small systems can be accomplished by intact reentry of the fuel capsules, intact reentry of the total fuel block, or intact reentry of the complete generator.

6) The minimum weight intact random reentry system seems to resolve itself in the form of reentering the total generator. Two philosophies are evident to achieve this result. The first is to allow the generator structure to absorb the reentry heating and be destroyed at a low enough altitude so that the remaining reentry energy will not destroy the fuel capsules. The second is to protect the entire generator assembly with a thermal system which can reject reentry energy at a rate fast enough to protect the support structure of the generator and keep the fuel capsules [redacted] the reentry.

[redacted]

The first approach is straightforward in nature and does not require any change in present generator design other than replacing the fins and shell with a high heat capacity material such as beryllium. This will allow good heat transfer during normal operation and sufficient heat capacity to prevent capsule meltdown during reentry.

However, there are several basic flaws with this approach from a safety standpoint which make its use questionable. First, beryllium which is the only material capable of playing the dual role of a good thermal conductor and good heat sink is quite weak at the average surface temperatures encountered during reentry ($\sim$ 2000°F). Advanced techniques such as composite structures or addition of ceramic fibers are required to increase high temperature strength. These are not state-of-the-art at present. Therefore, the question of whether the generator structure will stay together long enough to absorb the majority of the reentry heating as anticipated is not clear. This is not too critical in a ballistic abort due to the short heating times, but does become important as the aborts get closer to reentry angles reminiscent of the satellite decay case. An increase in strength of the beryllium has to be accompanied by lower temperatures which reflect back to increased weight of the system under the same reentry conditions. Hence, it is not clear that a minimum weight system will result from this approach. Secondly, even if the reentry is successful and the fuel capsules are released at an altitude where they can survive the remaining aerodynamic heating, high impact velocities and the possibility of ground burial further complicate this technique. Based on past experience with low temperature ($\sim$ 1500°F) thermoelectric systems, enough impact strength can be designed into the system to have reasonable assurance of survival. However, this is not the case with the higher temperature systems envisioned for future use, especially thermionic systems. Some structure around the fuel capsule must remain to absorb impact energies. Finally, in any case of free capsule impact, ground burial is a serious problem which could yield meltdown and release of the fuel form if it occurred. Drop tests conducted by Atomics International indicate burial in typical pasture soil of up to 18 inches can occur.

Therefore, since a conservative philosophy relative to eventual hazards from intact reentry must be adapted, it would seem that the technique of maintaining the generator structure and using it both as a reentry body, impact energy absorber and preventative to ground burial, is the most reasonable alternate for intact reentry of the small radioisotope systems.

[redacted]

[redacted]

7) For missions where both satellite decay or cis-lunar and planetary return reentries could occur special heat protection systems are required as a backup. These systems should combine the advantages of an ablator and reradiator. Char ablators such as insul cork have these properties.

8) From the analysis presented herein, it can be seen that to reduce the heat load a configuration must absorb, its ballistic coefficent should be maximized. This can be achieved practically by maximizing the drag and hence the profile area of the configuration. The implications of this fact result in generator designs which use Be rear plate radiators which serve a dual purpose as drag brakes during reentry.

9) Intact random reentry with delayed recovery is a technique applicable to small systems only due to tankage weight limitations. It is an active system and its components must be developed to a high degree of reliability before it can be considered state-of-the-art.

10) It has been determined in Appendix G of this study that significant loss of impact strength of present fuel capsules could occur after high temperature vacuum operation in space for relatively short periods of time, approximating aborts that would yield random intact reentry. Further study is recommended to uncover materials or fabrication methods to alleviate this problem.

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2. Semi-Controlled Intact Reentry

With the requirements for active reentry systems becoming more prominent to achieve safe intact reentry with large heat sources the use of the lifting body as a reentry technique has been investigated.

The lifting body has higher payload capabilities and range than simple ballistic reentry systems and also has the capability to maneuver which allows a great leeway in controlling final impact points of the heat source.

Semi-controlled reentry was derived as a technique to use the inherent range and maneuver capability of a lifting body to assure disposal of the heat source in deep ocean areas after successfully completing random intact reentry. Depending on the size of the heat source several of these lifting body modules may be used. In addition, it has the capability to be reentered under full control from either ground or space stations.

The technique was proposed as a method of satisfying remote disposal of large radioisotope heat sources from aborts of unmanned systems yielding random reentry. Although the complexity of this type of intact reentry device is much greater than the passive systems discussed previously and hence its reliability is decreased; the potential represented by these systems is such that their feasibility should be investigated and the problem areas defined.

It is immediately evident that the range of applicability of this concept is most attractive in shallow angle satellite decay reentries ($\phi_E \simeq 0^{o}$). During ascent aborts which yield ballistic reentries a lifting body would tend to skip and increase its impact range over a typical ballistic body. Therefore the basic safety backup of choosing a launch trajectory so that low ascent aborts will yield water impact could be negated. A short study of this point was made assuming typical launches from AMR and PMR. It was found that for ballistic reentries of a lifting body with $L/D \geq 0.5$ $V_E \simeq 16,000$ ft/sec and $\gamma_E = 45^{o}$ several trajectory perturbations may occur in the vertical plane and impact range is increased but splash points within the downrange limits still occur far enough from land not to negate the launch trajectory azimuth choices.

Figure 40 shows a preliminary schematic of the system required. The heat block would be incorporated directly into the lifting body reentry vehicle which in turn would be fitted into a special ejection heat exchanger to clear the R.V. if over temperature conditions occur (i.e., during reentry or coolant loss).

The inherent design of the reentry vehicle is such that attitude control will be required to assure the proper reentry control. The most convenient way this can be accomplished is through a frangible drag plate designed to break above 350,000 feet before much of the available K. E. is dissipated. Once controlled reentry begins, pressurized helium is released

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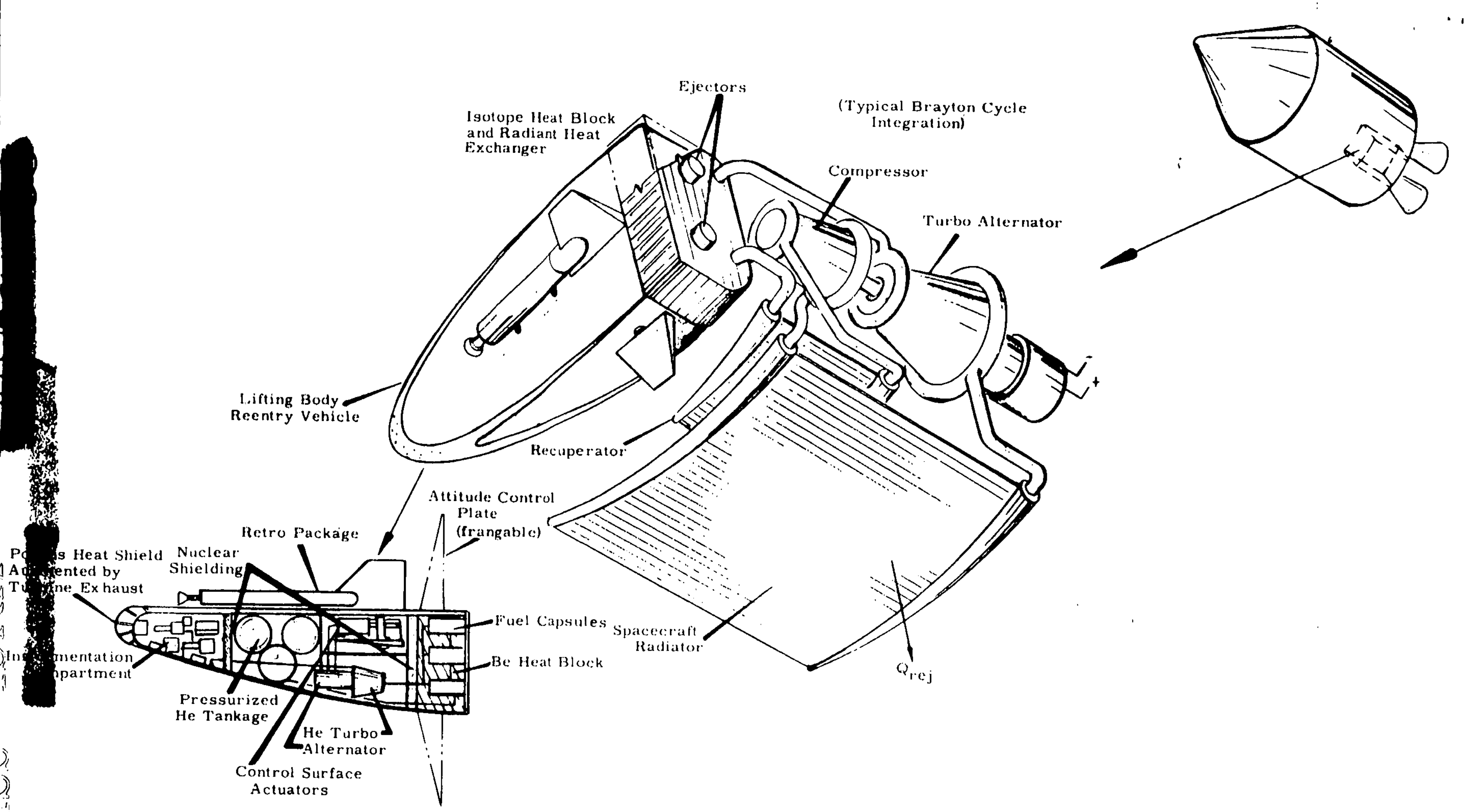
Ejectors
Isotope Heat Block and Radiant Heat Exchanger
(Typical Brayton Cycle Integration)
Compressor
Turbo Alternator
Lifting Body Reentry Vehicle
Recuperator
Attitude Control Plate (frangable)
Retro Package
Po[illegible]s Heat Shield A[illegible]ented by Tu[illegible]ne Exhaust
Nuclear Shielding
In[illegible]mentation [illegible]partment
Fuel Capsules
Be Heat Block
Spacecraft Radiator
Q_{rej}
Pressurized He Tankage
He Turbo Alternator
Control Surface Actuators

Figure 40 Typical Preliminary Schematic of a Lifting Body Reentry System and Integration with Spacecraft Electrical System.

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through the fuel block which serves to cool the heat source and acts as a pre-heater for the same helium which is used as a working fluid in a small turbo alternator. The turbo alternator supplies power to the instrument package and control system. The turbine exhaust is dumped through the heat shield to provide film cooling during the reentry. Five hundred watts for 2000 seconds can be supplied with a coolant supply of only three pounds of helium. However, other working fluids such as water may be more attractive since they can be stored under lower pressures. For helium, tankage weights are below 10 pounds and volumes are of the order of 500 - 600 cubic inches maximum. A possibility of using the working fluid exhaust for attitude control during the reentry also exists. However a cold gas system may be required for pre-heating attitude control before and during the reentry.

The instrumentation must be able to tell the difference between land and water area on the earth's surface (in the presence of clouds) and guide the system into a water impact in the operational sequence is shown in Figure 41.

The feasibility of the concept rests on several points:

(1) Range and maneuver capability.

(2) Character of the control systems and sensors.

(3) Reliability of components in a high temperature and nuclear environment.

(a) Range and Maneuver Capability. The fundamental purpose of this investigation is to examine the possibility of using aerodynamic lift to effect the intact recovery of a satellite or space vehicle from an orbit about the earth. The main effort is directed towards obtaining relatively large landing areas on the surface of the earth while remaining within the structural and heating limitations consistent with the current state of the art in these areas.

The primary considerations involved in the non-destructive return to earth of a space vehicle are:

(1) Maximum deceleration.

(2) Maximum aerodynamic heating rate.

(3) Total amount of heat absorbed.

(4) Radiative capability.

During reentry, the influence of any one of these three factors on the design of a particular vehicle is generally not independent of the others, and hence all of them must be considered simultaneously. For instance, a vehicle may have a low maximum heating rate and absorb a relatively small amount of heat, but if the vehicle is not able to transfer this heat to the atmosphere, the temperature of the structure may exceed the limit imposed by the g-loading, and result in a structural failure. Conversely, a high heating rate may cause failure or burnup, even with a high radiative capability.

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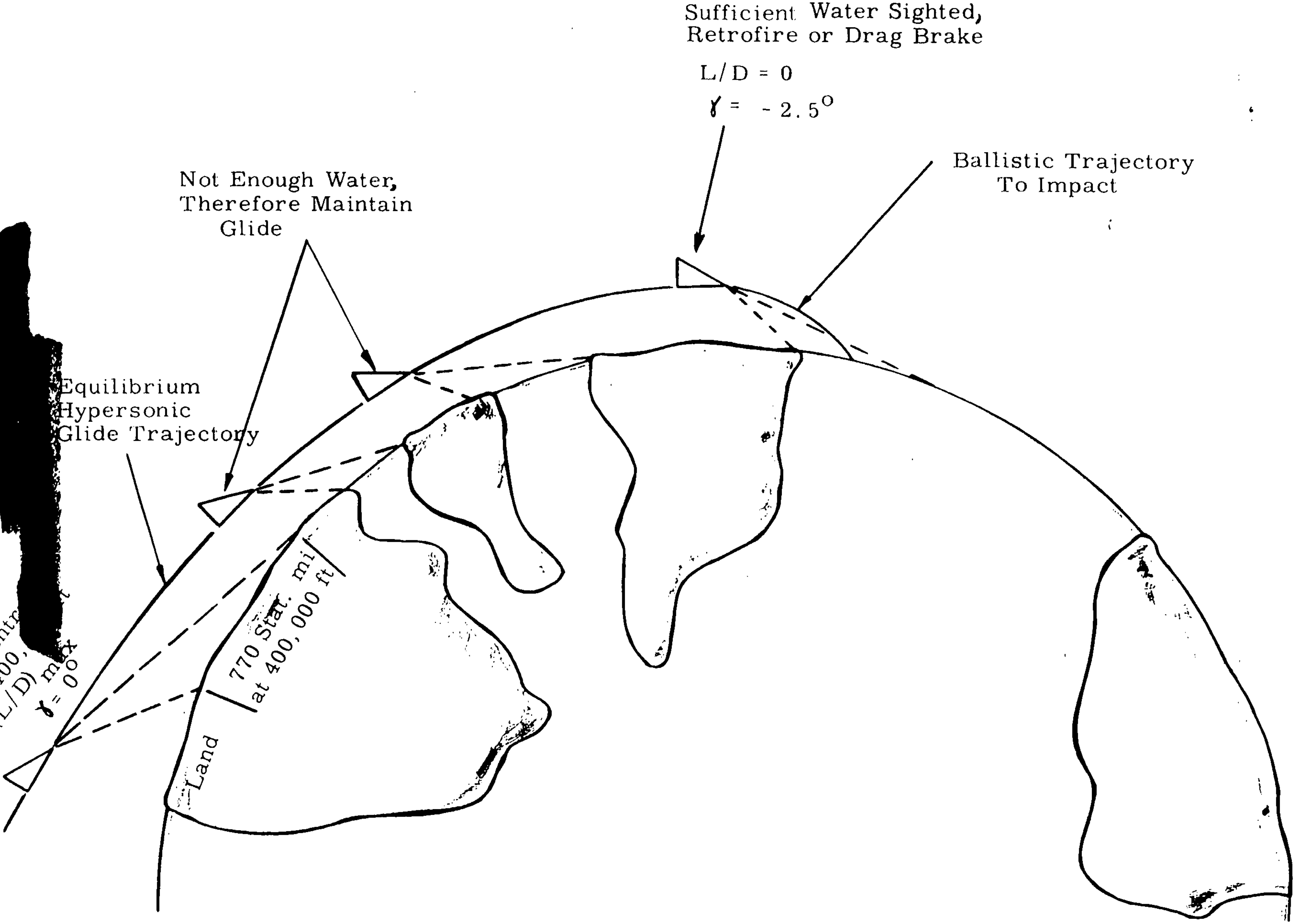

Figure 41. Typical Reentry Operational Sequence for Water Landing

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The more recent investigations conducted by NASA and private organizations have been directed towards the feasibility of employing aerodynamic lift to reduce the severe heating rates and decelerations encountered during atmospheric entry. The results indicate that lifting trajectories show a marked improvement in maximum heating rate and deceleration over ballistic (non-lifting) entries, although the total amount of heat absorbed is higher. The increase in absorbed heat results from the longer reentry times associated with lifting bodies.

Many studies have shown that for lifting vehicles, deceleration is essentially no problem, and Chapman's results (Reference 32) are shown in Figure 42.

Lifting trajectories may be classified into three types:

(1) Skip trajectories, characterized by high L/D's and small flight path angles.

(2) Direct entry trajectories, which usually have negative lift and negative flight path angles.

(3) Equilibrium glide, which is the boundary between skip and direct entry.

In the equilibrium glide trajectory, the weight of the vehicle is balanced by the lift and centrifugal force at all points along the path. Thus, as aerodynamic drag slows the vehicle down, it must drop to a lower altitude where the higher density can maintain the required lift force. Since the magnitude of the vertical velocity is small compared to the horizontal velocity, small flight path angles are required to maintain equilibrium. When the flight path angle is decreased to a negative value greater than that required for equilibrium, the vehicle penetrates the atmosphere fast enough for the lift force to increase faster than the centrifugal force decreases, resulting in either oscillations or a skip out of the atmosphere.

The sensitivity of the trajectory to the flight path angle has been given extensive consideration by Chapman and it is seen that for L/D's of about 1.0, flight path angles on the order of -1° to -2° are sufficient to induce oscillations and skips. For decaying orbits, however, the flight path angle will be very close to zero and Chapman shows that in this case, the angle does not decrease significantly until the altitude is so low that there is no danger of skipping out of the atmosphere.

The equilibrium glide trajectory has been chosen for this investigation because aside from the fact that it is particularly well-suited for satellite decay orbits, it also represents the optimum trajectory for range on a single pass entry.

Appendix D discusses the calculational techniques used to determine range, maneuver time and heating rates during equilibrium glide reentries.

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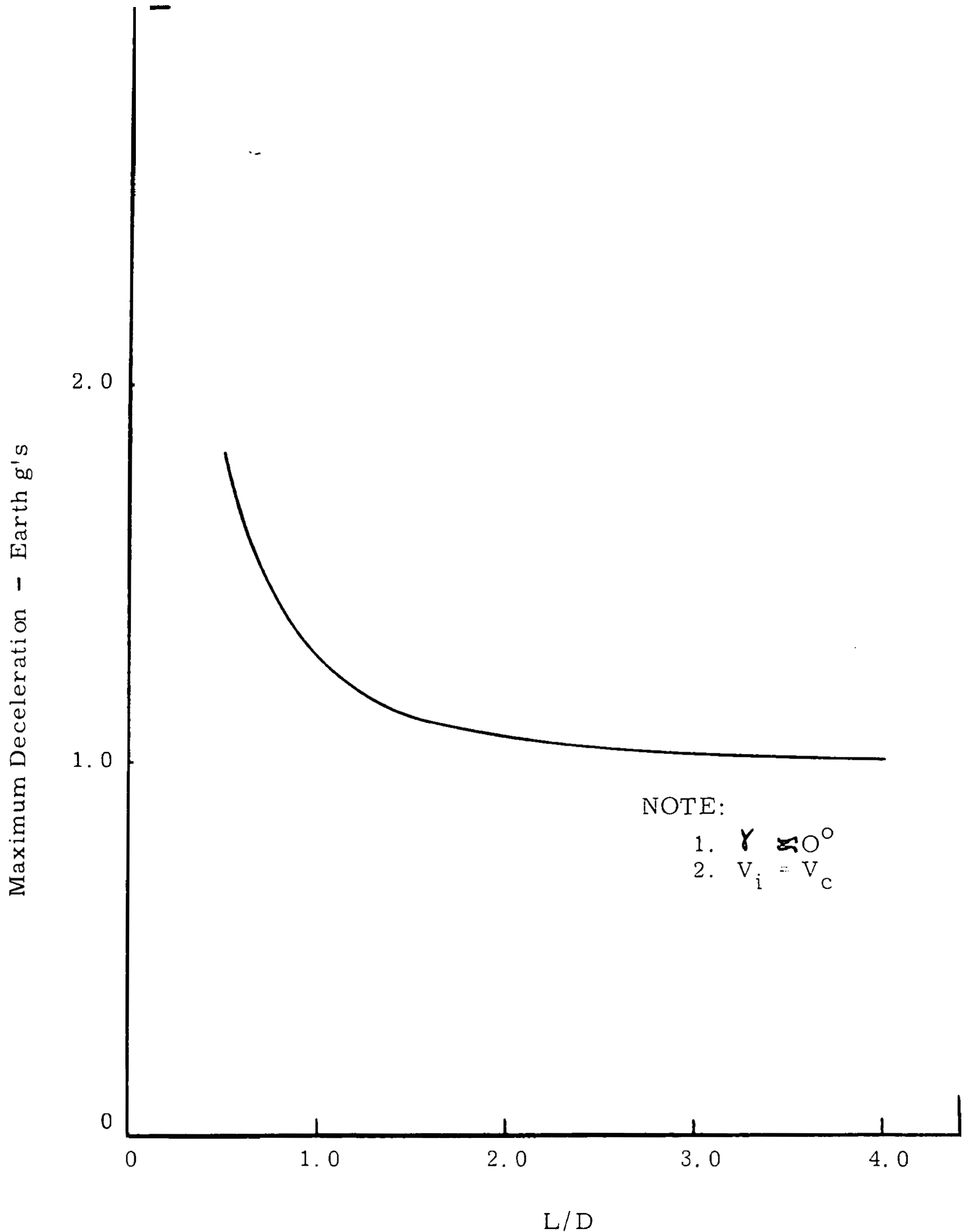

Figure 42. Effect of L/D On Maximum Deceleration

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b. Discussion of Results

(1) Optimum Roll Angle. Figure 43 shows the effect of L/D on optimum roll angle for maximum lateral range. The reason for the decrease of ϕ, roll angle, with an increase in L/D is readily observable in Equation (D-14) in Appendix D. The term $\left[-L/D \sin \phi \ln (x)\right]$ is the instantaneous heading angle, and its value must remain below $\pi/2$ along the significant portion of the trajectory or else the vehicle will fly in a spiral path and subsequently decrease both lateral and longitudinal range.

Actually, the heading angle increases beyond $\pi/2$ in the lower portions of the trajectory (approximately 300° for L/D = 3), but this occurs at such a low altitude that the loss in range is small. It should be pointed out that an attempt to remedy this situation by reducing the roll angle would result in a loss in lateral range at the higher altitudes that would be greater than that gained at the end of the trajectory.

(2) Range. The effect of L/D on both lateral and longitudinal range is illustrated in Figures 44 - 47. From these figures, it is evident that appreciable maneuverability can only be gained with L/D's on the order of 1.0 or above. Although Reference 35 indicates that a hypersonic L/D of 2.0 or above is possible with a blunted half-cone, the fineness ratios required for low drag would present some packaging difficulties. Also, since ailerons or control fins seem to be the most feasible type of roll-control system, a winged vehicle would probably be the best arrangement.

The data presented in Reference 36 shows that L/D's up to about 5.0 can be obtained with delta-wing-half cone combinations. This type of vehicle is apparently very attractive because of its high L/D's and large amount of storage space available for a control system power supply and payload. Also, ailerons could be easily incorporated for pitch and roll control. The effect of volume requirements on the hypersonic aerodynamic characteristics of this type of vehicle is shown in Figure 48, taken from Reference 36. Although increasing the volume is detrimental to the performance of the vehicle, it appears that for the L/D's required here (2.0 - 3.0), a relatively large volume can be realized. In addition, this configuration could be easily adapted into the overall heat source design of a large radioisotope power system.

The interesting features of Figures 44 - 47 are that ballistic coefficient has no effect on lateral range and that the longitudinal range is not decreased appreciably when $\phi = \phi_{opt}$.

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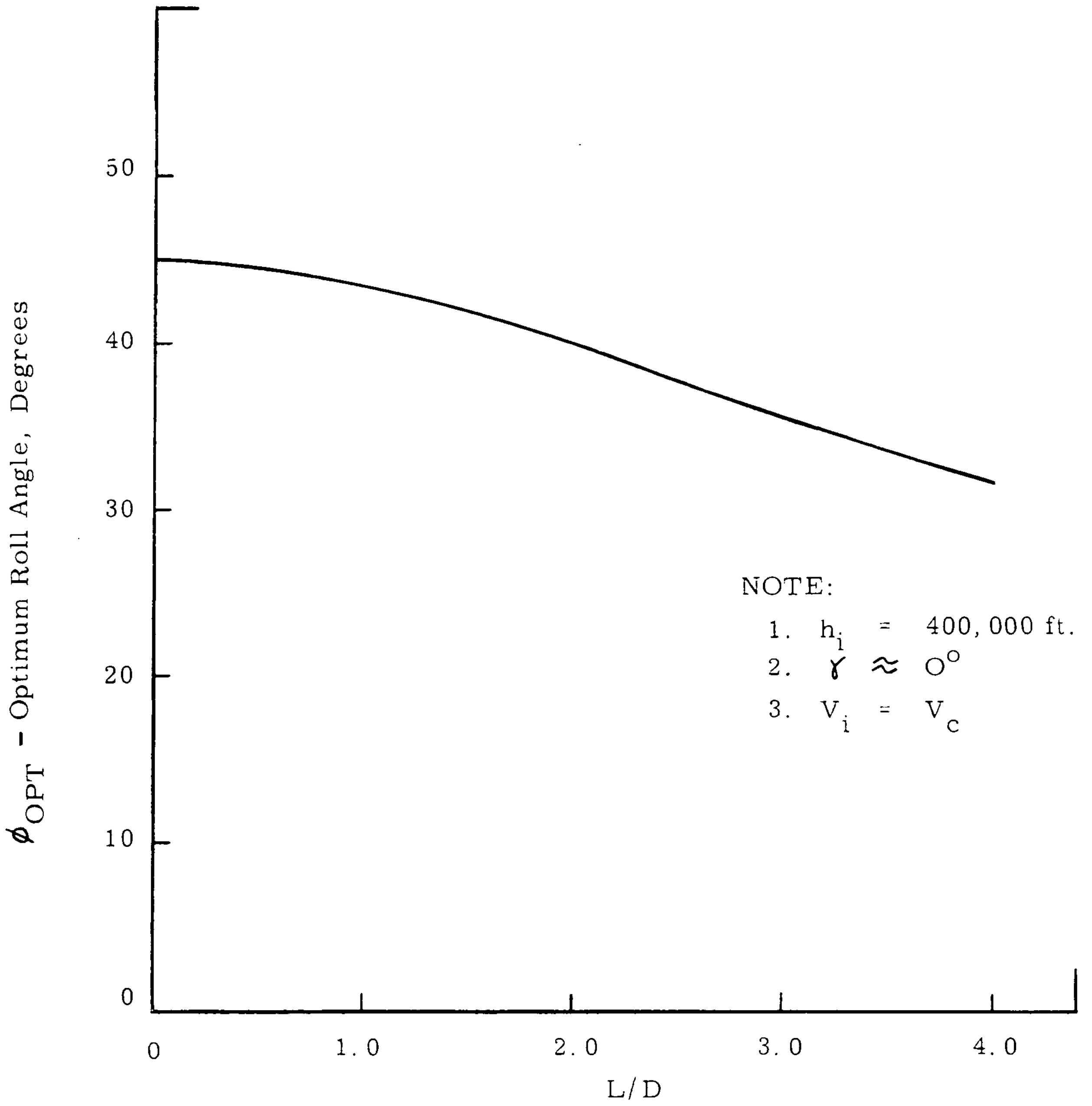

Figure 43. Optimum Roll Angle for Maximum Lateral Range

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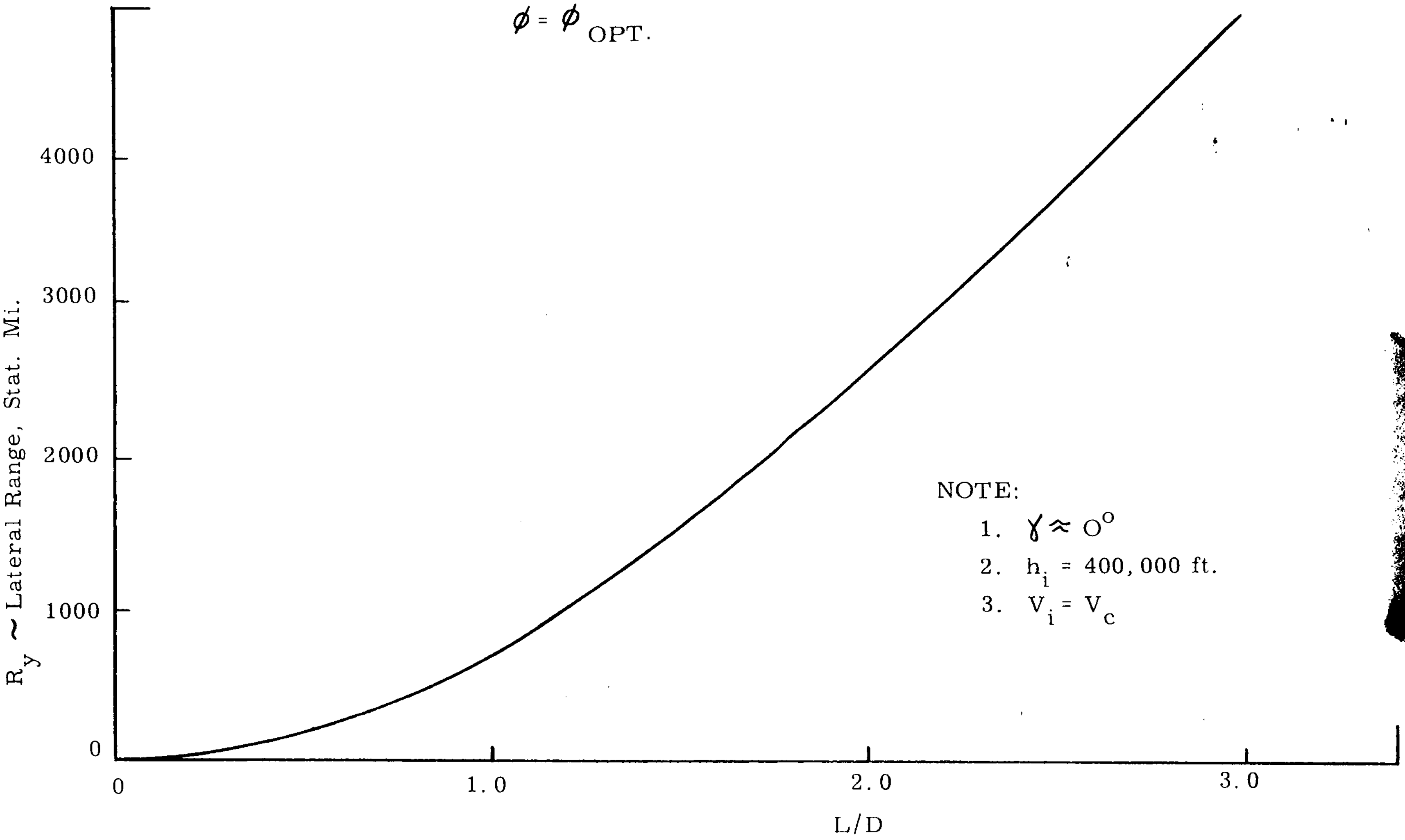
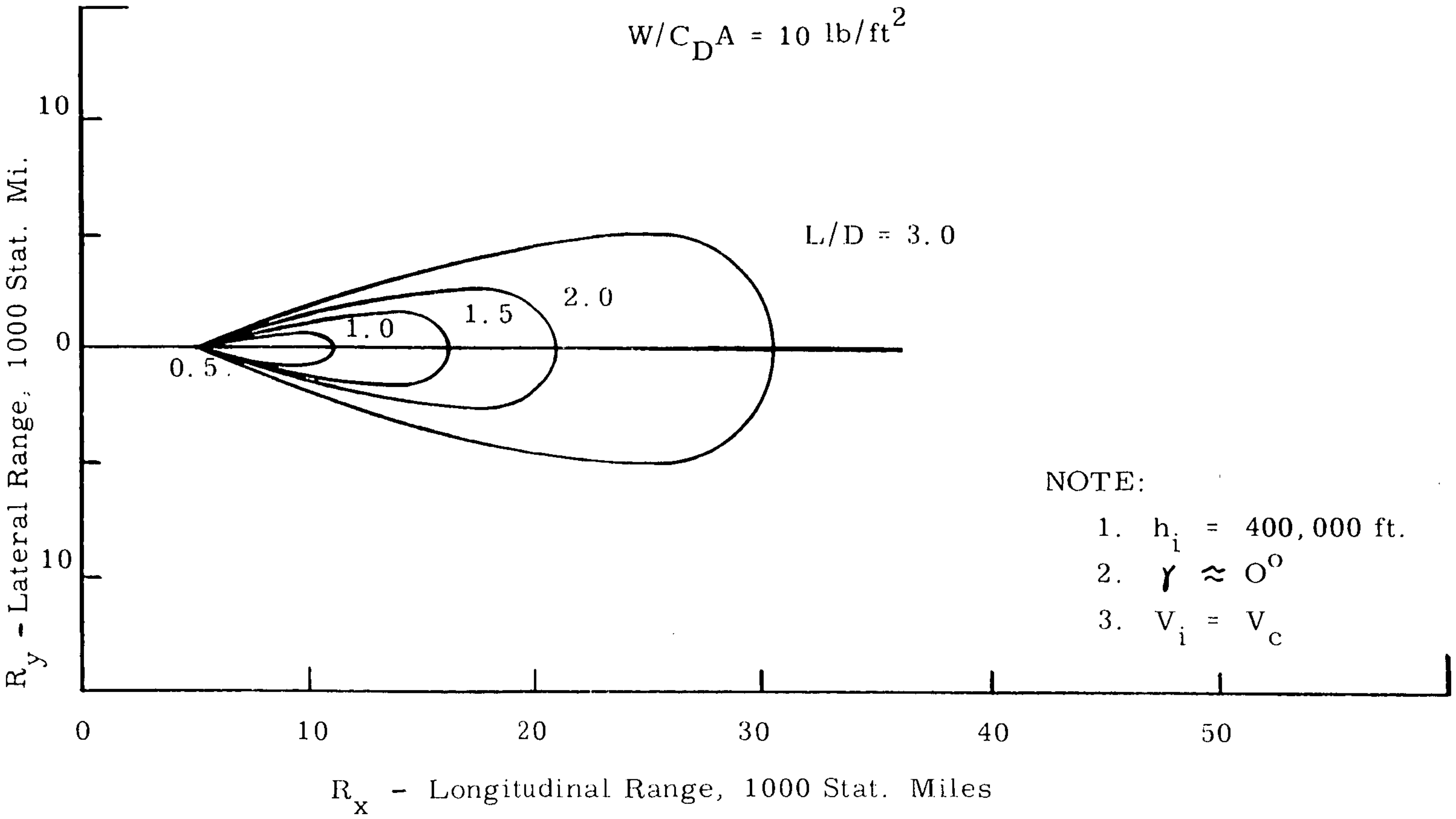


NOTE:

1. $\gamma \approx 0^o$
2. h_i = 400,000 ft.
3. $V_i = V_c$

Figure 44. Effect of L/D On Lateral Range



NOTE:
1. h_i = 400,000 ft.
2. $\gamma \approx 0^o$
3. V_i = V_c

Figure 45. Effect of L/D On Range

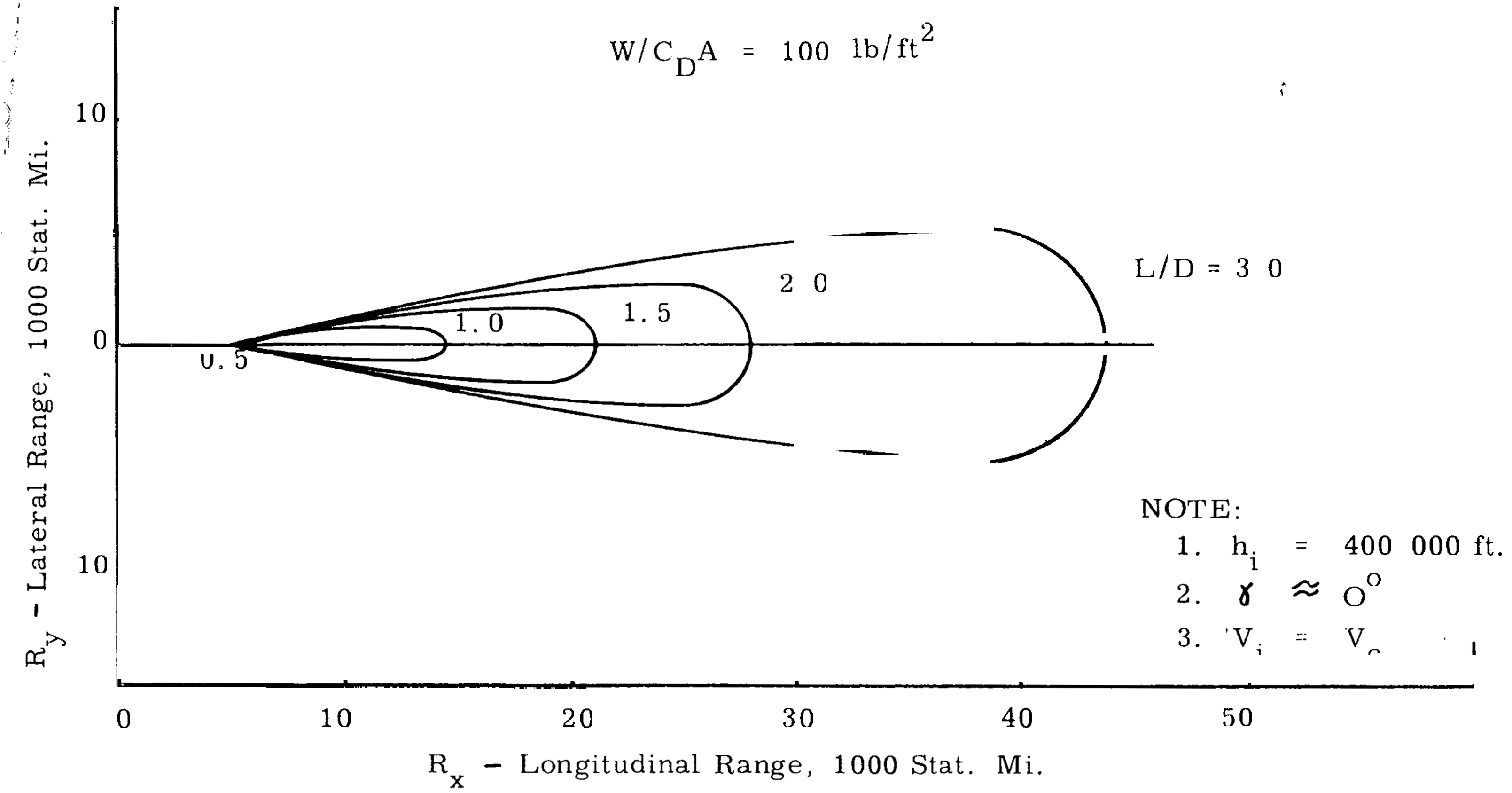
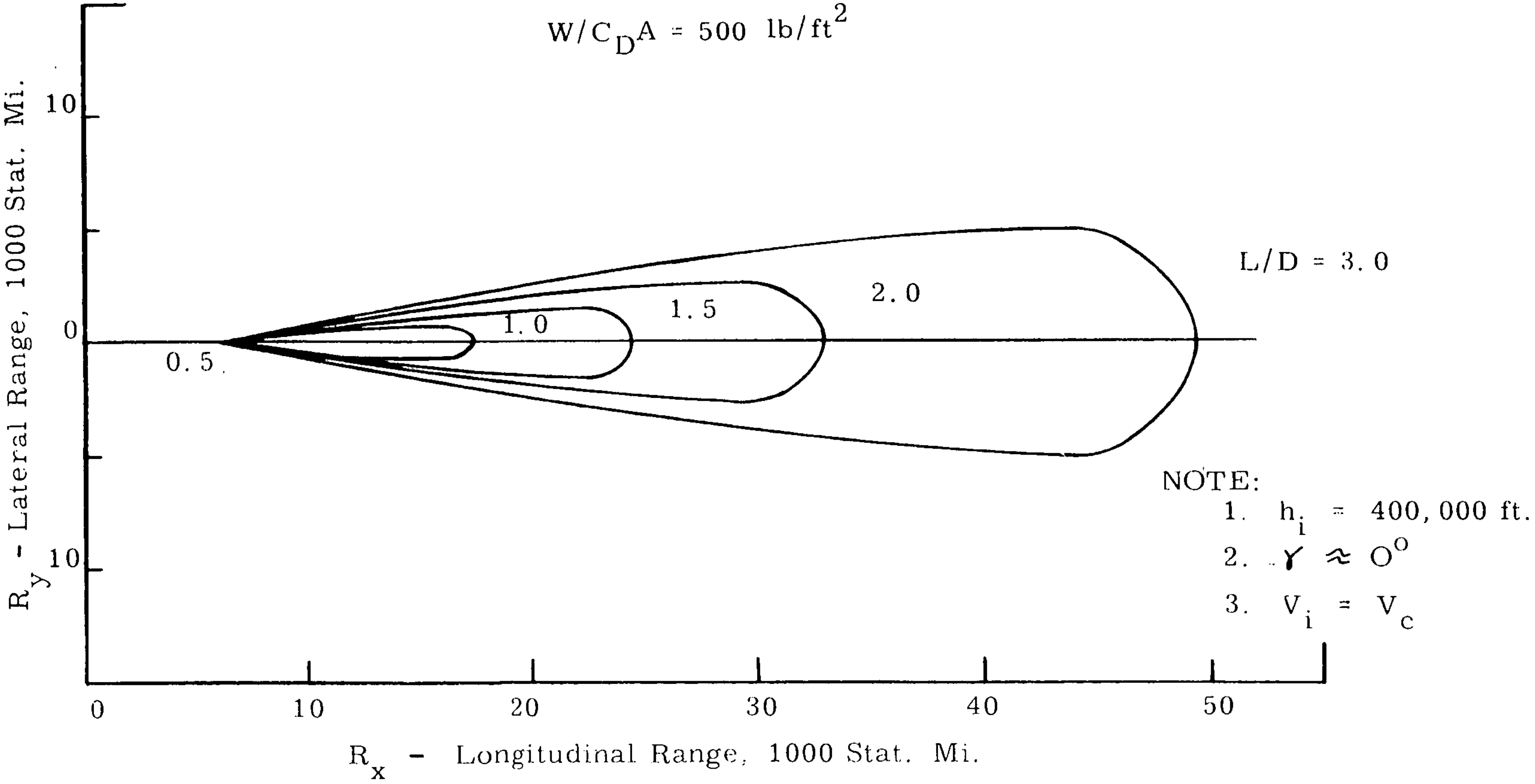


Figure 46. Effect of L/D On Range



NOTE:
1. h_i = 400,000 ft.
2. $\gamma \approx 0^o$
3. $V_i = V_c$

Figure 47. Effect of L/D On Range

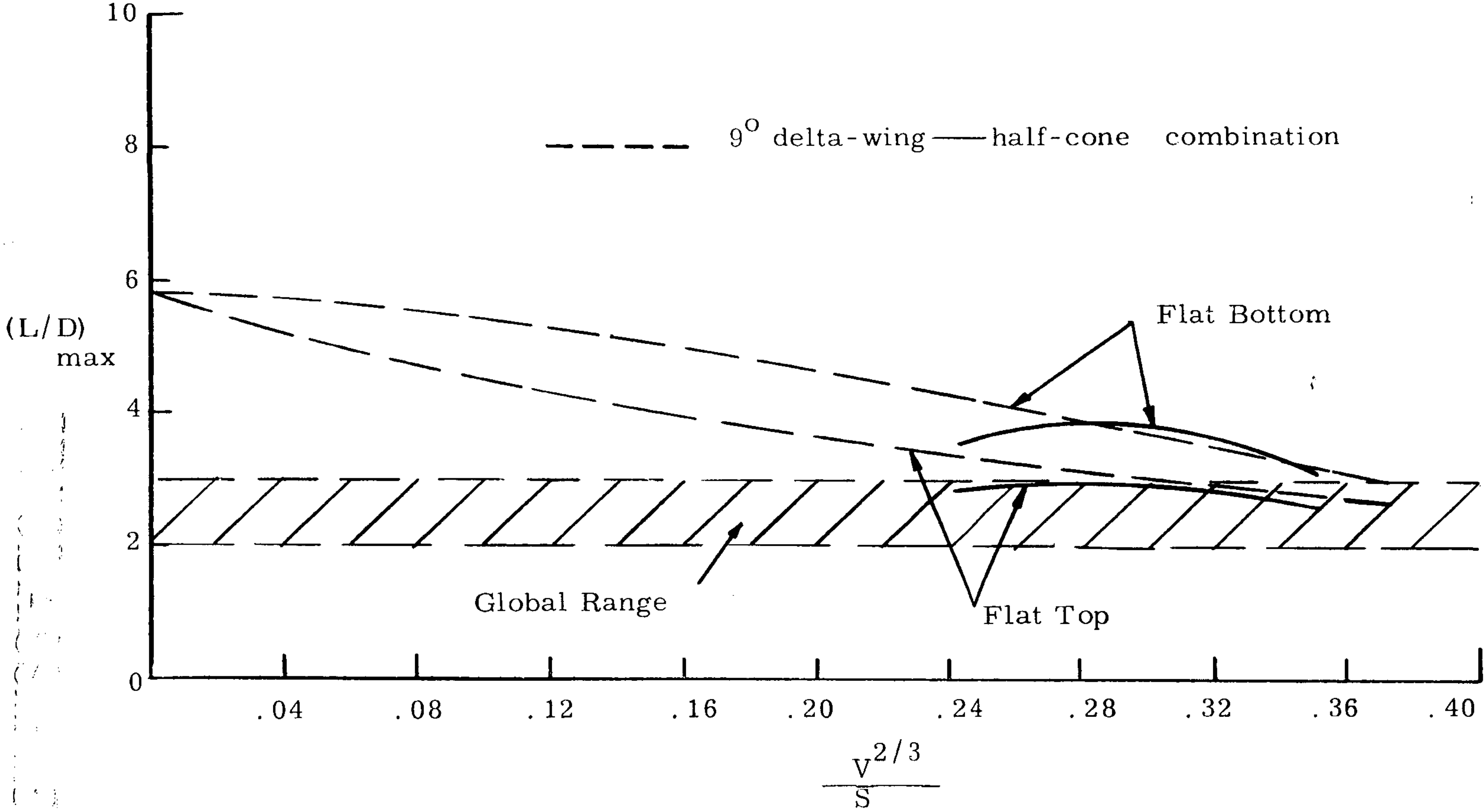


Figure 48. Maxium Lift-Drag Ratios of Isolated Half-Cone Bodies. M = 20; R = 3.5x10^{6}

This results because the momentum of the vehicle carries it a great distance before the kinetic energy is reduced enough to cause a descent to an altitude where the side force becomes significant. Hence, the vehicle has only a relatively short longitudinal range-to-go capability over the portion of the trajectory where maneuvering is possible. This is evident in the tear-drop shape of the footprint curves. Figure 48 shows the effects of L/D_{max} and configuration on range.

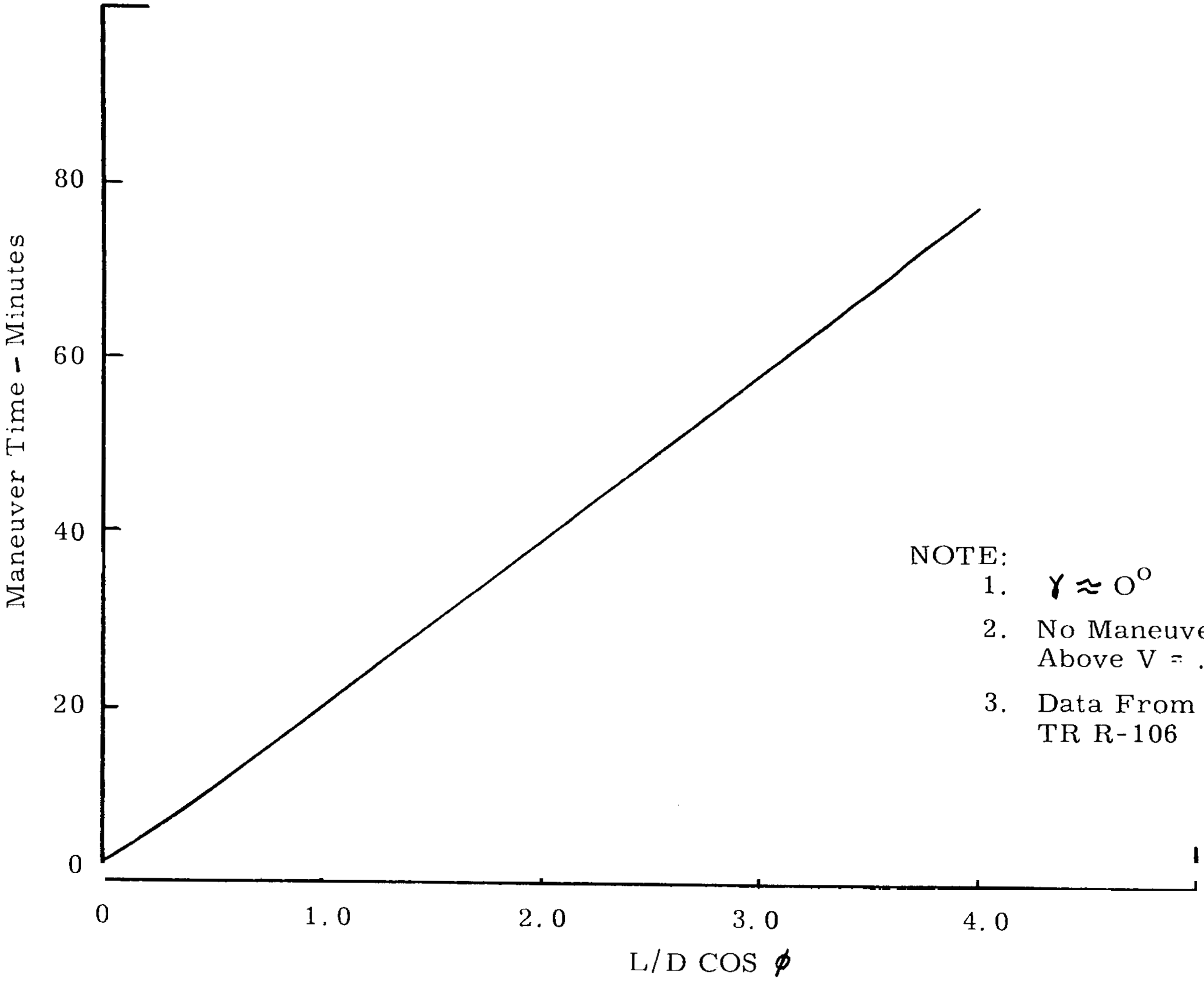
(3) Maneuver Time. Figure 49 gives an estimate of the order of magnitude of the available maneuver time, the implications of which are that if the control system is required to be active along the entire maneuverable part of the trajectory, power supply weights may become prohibitive. In other words, if the vehicle is to have the capability to change its course at any point along the trajectory, the L/D of the vehicle may be limited to keep the weight within reasonable limits.

An answer to this problem might be to have an active control system to turn the vehicle to the pitch and roll angles required for a desirable L/D before reentry and remain passive thereafter.

(4) Aerodynamic Heating. Maximum stagnation point heating rate and total heat absorbed during reentry are shown in Figures 50 and 51 as a function of L/D and ballistic coefficient. Here, it is seen that the advantage of using lift to decrease the heating rate is offset by the increase in the amount of heat absorbed by the vehicle. This effect is due to the fact that $\dot{q}$ is proportional to $\frac{1}{\sqrt{L/D}}$ while Q is proportional to $\sqrt{L/D}$.

Figure 52 , taken from Reference 37, gives a comparison of the weights of ablation heat protection systems for lifting and ballistic entry into the earth's atmosphere from a circular orbit. These results indicate that because of the large amounts of ablation material needed for surface protection, some other type of system is almost certainly required for lifting vehicles.

Recent developments in heat protection systems have shown that a combination of radiation and ablation type cooling system offers an effective means of removing large quantities of heat. One of the more promising types of radiation-ablation materials is pyrolytic graphite, which has been investigated in Reference 38. The data (presented in Reference 38) for a typical lifting reentry vehicle ($L/D = 0.5$, $W/C_DA = 500$, $V_i = 26{,}000$ fps) show that only about 0.5 inch of pyrolytic graphite is required to keep the wall temperature in the neighborhood of 1200°R after 30 minutes of exposure to reentry conditions, and that less than 0.2 inch of material is ablated away during the reentry.



NOTE:

1. $\gamma \approx 0^{\circ}$
2. No Maneuverability Above $V = .9\ V_c$
3. Data From NASA TR R-106

Figure 49. Effect of L/D and Roll Angle on Maneuver Time

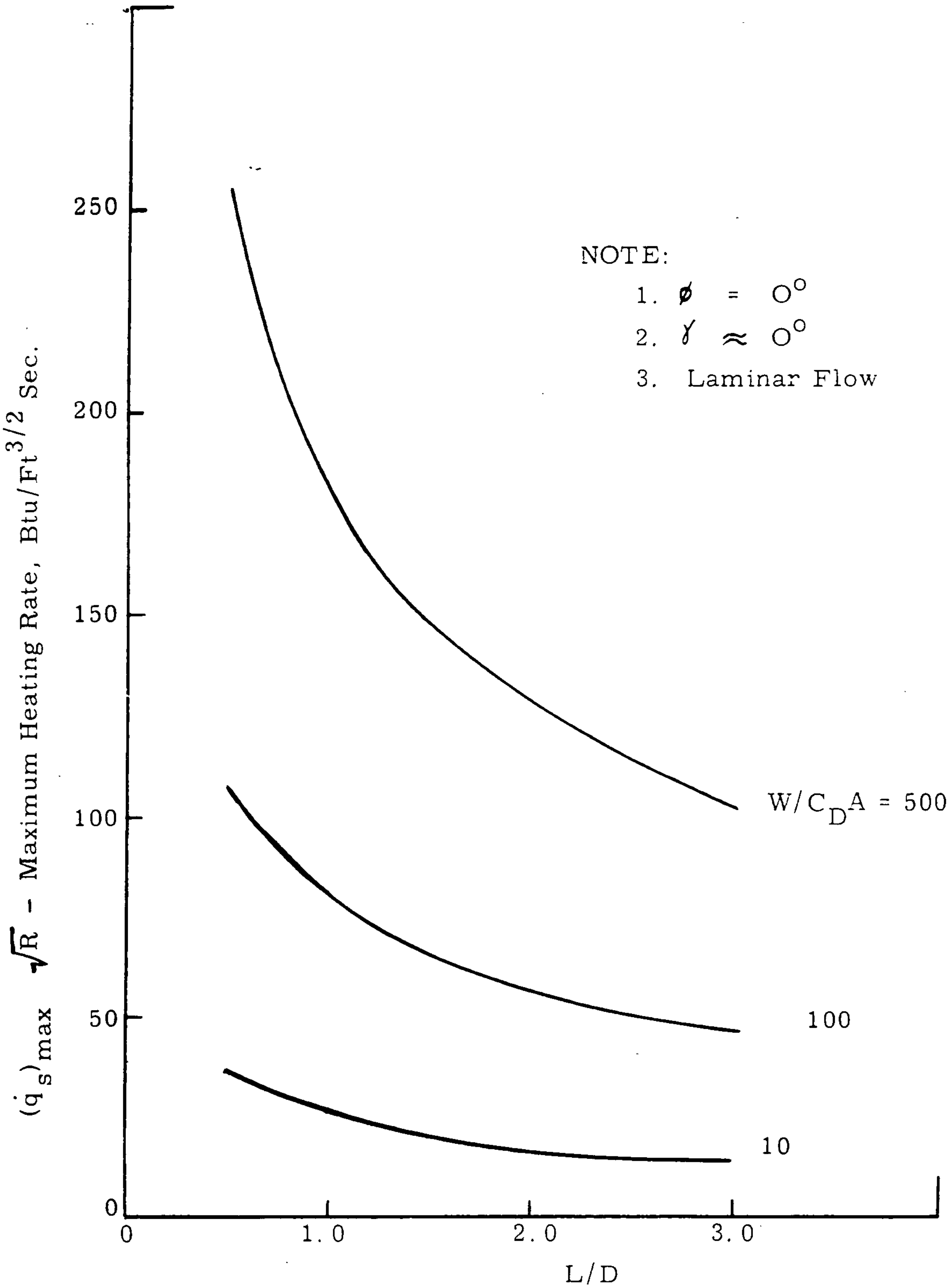


Figure 50. Effect of L/D and Ballistic Coefficient on Maximum Stagnation Point Heating Rate

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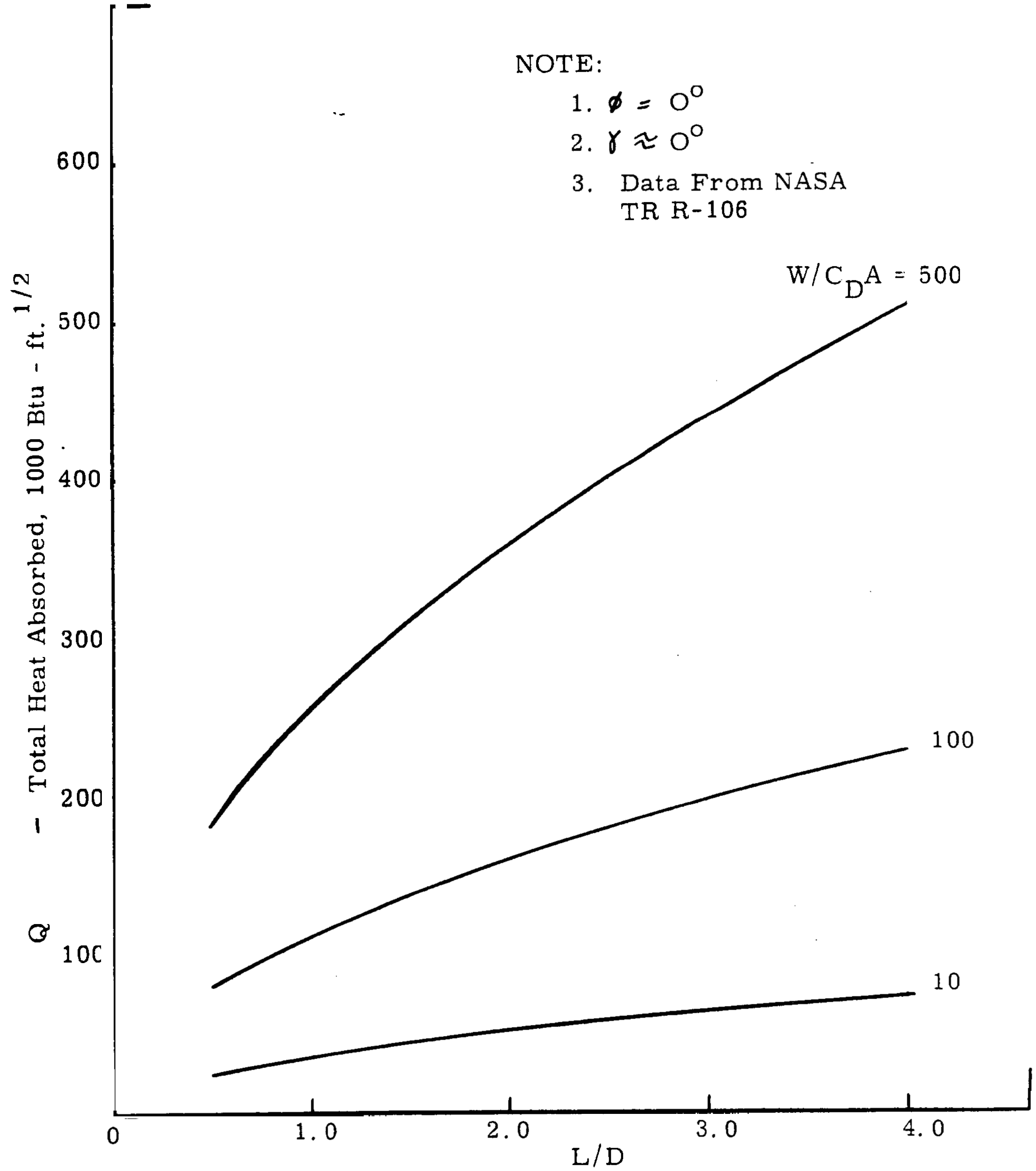

Figure 51. Effect of L/D and Ballistic Coefficient on Total Heat Absorbed

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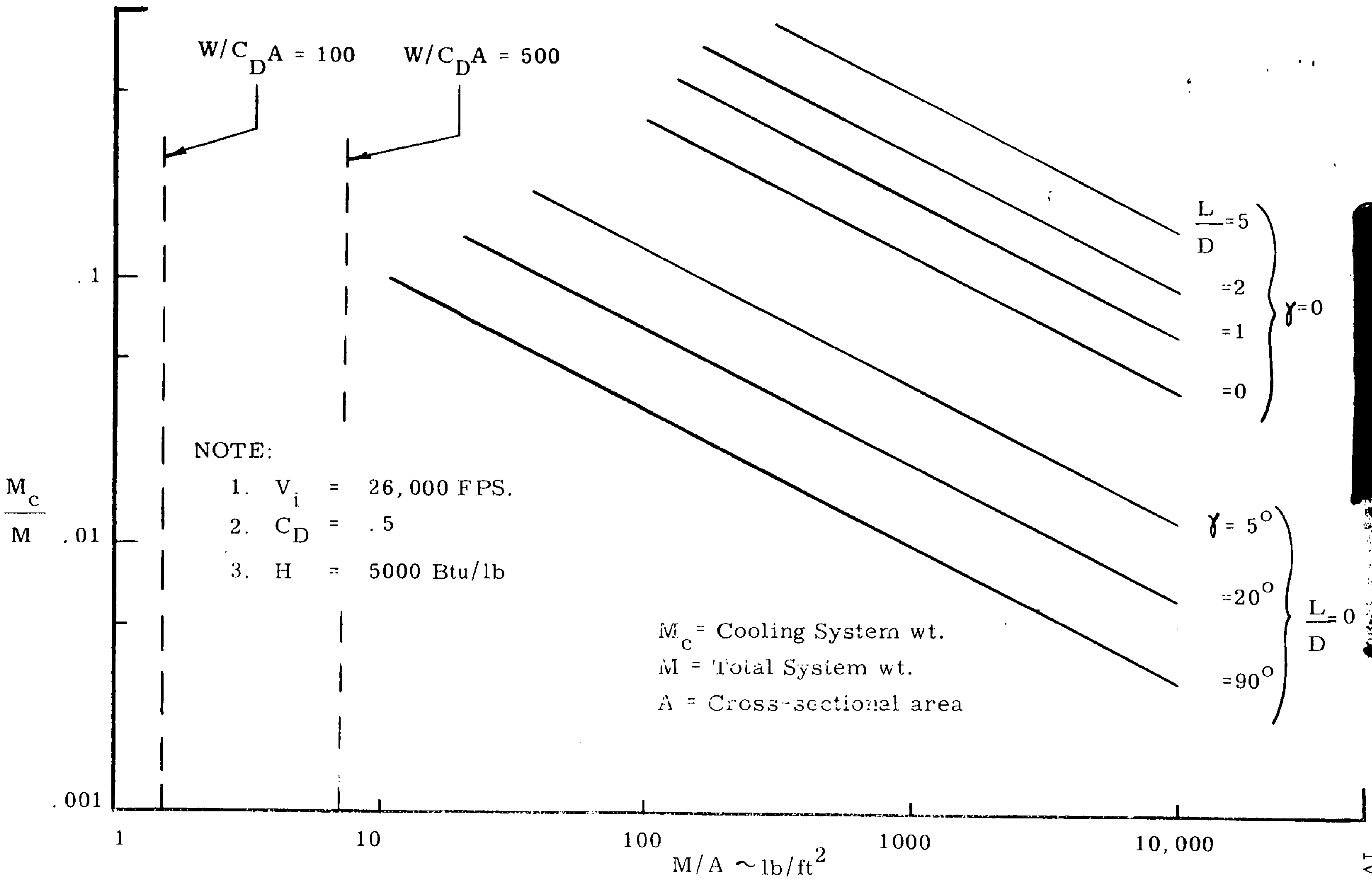

Figure 52. Comparison of Heat Protection System Weights for Satellite and Ballistic Reentry

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Pyrolytic graphite heat protection systems are also desirable from a weight standpoint, as is illustrated by the vehicle described in Reference 39. In this particular ballistic entry vehicle ($W/C_DA \approx 9$, satellite decay orbit) the heat protection system weight is only about 13 percent of the total weight, which is far better than the weights predicted for ablation systems.

Based on the preceding remarks, the pyrolytic graphite protection system seems to be the best for the type of reentry vehicle discussed in this study.

b. Character of Flight Controls and Sensors

(1) Typical Operational Profile. For intact reentries, the most desirable type of landing area is water, since in most cases no special systems are required to guarantee survival of the impact. It has been demonstrated that the proper choice of L/D, ballistic coefficient, and roll angle for a given vehicle can result in relatively large landing areas on the surface of the earth. It must be pointed out, however, that this increase in the performance and maneuverability over ballistic vehicles, does not increase the probability that the vehicle will land in water. For completely random reentries, a ballistic vehicle has the same probability of hitting a given spot on the earth as a lifting vehicle no matter how great its range capability may be, if no active control of the lifting vehicle is available. Hence, the probability of landing in water for both vehicles is simply the ratio of the water area to the total area of the earth.

The main problem encountered in the attempt to use aerodynamic lift to insure a water landing is that the vehicle cannot "see" its impact point until it is too late to do anything about it. At 400,000 feet the visibility (earth range to the horizon) is only about 770 statute miles and decreases to about 390 miles at 100,000 feet. Therefore, the vehicle does not know where it will land until the range-to-go is somewhere between 770 and 390 miles, at which time most of the maneuvering range has already been used up. It appears then, that although a large maneuver capability can be incorporated into an unmanned reentry vehicle, something else must be done to make this potential useable. Unfortunately, this means that the complexity of the system must be increased.

It appears that one solution to the problem is to provide the vehicle with enough lift such that it can cruise at high alititudes until it can see a good impact area. Obviously, this means that no more than a global range is required (Figures 44-47 show this is possible), since it is not unreasonable to expect that in one circuit around the earth a large enough body of water will be encountered.

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In order to insure absolutely that the vehicle will land in water, it is necessary that once a landing area is spotted, the vehicle is capable of hitting this area regardless of the altitude. This immediately implies that a retro system is required. This sytem could be a simple drag brake or more likely a propulsion retro. Also, the vehicle must be able to withstand the heating and deceleration encountered after retrofire.

A workable system would operate in the following manner:

(1) The vehicle begins reentry at 400,000 feet and circular velocity in the (L/D) max configuration, and cruises in an equilibrium glide until there is nothing but water in sight.

(2) Immediately after the vehicle is surrounded by water (in its visibility cone), the vehicle assumes a zero-lift configuration, fires the retro system (or lowers its drag brakes), and follows a ballistic trajectory to impact. It could also combine a 90° roll with the retro and spiral into this impact area of its visibility cone.

Whether or not this type of system is feasible depends on the ability of the vehicle to land the payload(if nothing else) in the desired impact area. The most difficult condition occurs when the vehicle spots water immediately. If this is the case, the vehicle is at 400,000 feet, traveling at circular velocity, and must land somewhere within a 770 mile radius, since there may be land over the horizon. The tabulations in Reference 32 indicate that this is possible at zero lift with a -2.5° change in flight path angle, and that no more than a 10g deceleration occurs. The amount of heat absorbed during such a reentry will be less than that experienced for an equilibrium glide, hence, there is no problem in this area. Although the heating rate will be higher than the gliding heating rate, the heat protection system mentioned previously should be capable of coping with it. If not, however, and some portion of the vehicle is burned away, the payload will still be insulated by the structure of the vehicle and will survive.

The most probable case is that the vehicle will have to do some amount of gliding before coming to a suitable landing point. Although the vehicle's visibility is now something like 500-600 miles and it must therefore impact into a smaller area, Chapman's data shows that this is more easily done than the extreme case above.

[redacted]

Still better confidence in the vehicle's ability to hit the desired impact can be gained by rolling the vehicle 90° to obtain zero lift instead of trimming to zero lift. This would give the vehicle a side force, resulting in a spiral trajectory that would fall short of the horizon seen at retrofire.

An example of the type of logic diagram required for this vehicle is shown in Figure 53.

(2) Required Sensory and Control Systems. The implications of Figure 53 are that the following systems would have to be incorporated into the vehicle:

(a) A water-sensing system. This could probably be an infrared emissivity sensor or a microwave device. Also, this system would require a control to insure that it does not operate above 400,000 feet (i.e., before reentry).

(b) An attitude control system to provide a desired L/D (either (L/D) max or L/D = 0). No negative lift capability is required since the resulting trajectory would cause unreasonable decelerations and retrofire would be required anyway, hence there is no advantage to be gained.

(c) A retro system to change the flight path angle.

(d) A logic system to control the other systems in the vehicle.

(e) A control to turn the logic system off once a decision has been made and action has been taken.

As was mentioned earlier, incorporation of a water-only landing capability into the vehicle is done at the expense of increasing its complexity and weight. Because of the types of systems which are involved, however, it is reasonable to expect that their overall effects on the vehicle will not be prohibitive. Infrared sensors and "black box" logic systems, for instance, are known to be reliable and should cause no large weight problem. The types of retrorocket systems in use today are very light and should comprise only about 10 percent of the weight of the vehicle. The basic question of operation after long time storage in space is still a serious unknown.

One advantage that this vehicle would have is that the guidance system would be relatively simple. For reentry vehicles which are either attempting to hit a specific spot on the earth or fly a nominal trajectory, a rather complex guidance system is required. The systems usually employed have to measure a number of variables (altitude, velocity, range-to-go, etc.); use

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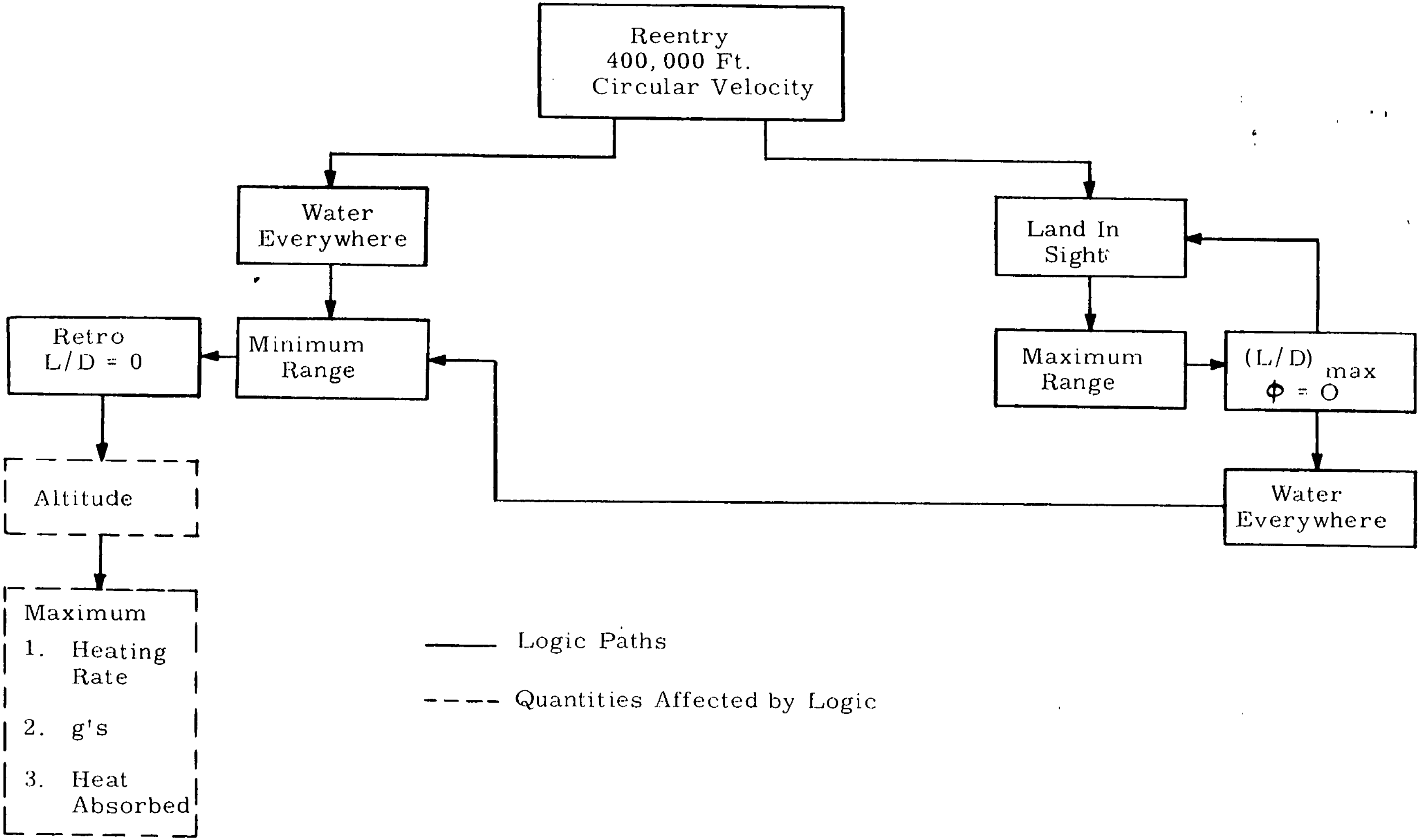

Figure 53. Typical Control System Logic Loop Semi-Controlled Lifting Body Reentry

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these to compute the trajectory and corrections to it; activate a control system; and check the new trajectory. Even if the calculations are done on the ground rather than using an on board computer, the telemetry system required to transmit and receive data is quite complex.

The control scheme proposed above, however, needs much less in the way of complex electrical or mechanical equipment. There is only one "variable" to measure - namely whether or not there is water below. Also, there are no calculations to be made, the control system does not have to be capable of performing a number of different maneuvers, and it only has to perform once. Essentially all that is needed is a reentry initiated off-on switch controlled by the water sensing device.

Although this system seems to represent a simplification, there are some inherent problems which must be overcome. One of these is the water sensor. Photographic means are limited by clouds and even light haze while infrared sensors, which are capable of penetrating fog and light clouds, are not able to see through storms. Low frequency microwave systems, however, are "all-weather", but they are more complicated since both a transmitter and receiver are required. There also may be trouble with reflections from choppy water (especially in cloudy storm areas) which might look like land to the receiver. It appears, then, that there are two choices available for the type of water sensor desired. One may select either the limited visibility type (photographic or infrared) and take the chance that all possible landing areas are not covered by clouds, or else the more complex total visibility type (microwave) which is not dependent on the weather.

There is also more than one way to go with the attitude control system, which determines L/D. If fins or trim tabs are employed the vehicle can be either rolled or pitched from the (L/D) max configuration to the zero-lift configuration. The rolling maneuver is the more attractive of the two since it helps in keeping the vehicle from overshooting the landing area and presents less of a stability problem. The main problem associated with an aerodynamic altitude control system is that it is needed most when it is least effective - at high altitudes. Also, from the overshoot standpoint, the most desirable maneuver is the quickest, and again the aerodynamic type of system has its drawbacks at high altitudes. Gas jet thrusters or inertia wheels, both of which are presently being used on ballistic missiles may be the answer to the problem. The same can be said for the deceleration system which is needed to change the vehicle's flight path angle. Rough calculations indicate that a drag brake would have to have a huge projected area for effectiveness at the 300,000 to 400,000 foot altitudes. Hence, the retro rocket system seems to be the most logical choice to deliver the braking impulse required in a reasonable time and without huge structural loads and deployment problems.

[redacted]

[redacted]

In summary, this vehicle would have all the problems of the design of a large manned lifting body reentry system at lower reliability levels.

Although the concept is sound the development of the system is felt not to be within present state of the art for the application to intact reentry of nuclear heat sources, however, they could be developed with a reasonable lead time (~3 years).

An analysis of the sensitivity of electronic components to the nuclear environment on board this type of vehicle was completed and is presented in Appendix E. It shows that there are no severe shielding problems for electronic components using transistors in a typical 2.5 Kw(e) Po-210 or Pu-238 fueled vehicle.

d. General Conclusions - Semi-Controlled Intact Random Reentry

(1) With the requirements for active reentry systems becoming more prominent to achieve safe intact reentry with large heat sources the use of the lifting body as a reentry technique has been investigated.

Semi-controlled random intact reentry was derived as a technique to use the inherent range and maneuver capability of a lifting body to assure disposal of the nuclear heat source in deep ocean areas after successfully completing random intact reentry. Its application was specifically designed for unmanned systems using large heat sources and not having the capability of fully controlled intact reentry. In addition, the system is attractive from the standpoint of not having to depend on ground station control in case of an abort.

(2) It was determined that from the standpoint of general range and lateral range the vehicle proposed has the performance necessary to land a nuclear heat source in water from any given reentry position on the surface of the earth.

(3) The limiting factors affecting feasibility of this technique all appear to be centered in development of flight controls and sensors capable of operating after long time exposure to space and the thermal and nuclear environment of the heat block. However, good short time reentry performance can be expected. These systems are not yet state of the art but it is felt that they could be developed quickly. The control sequence for the vehicle is a simple one resolving itself into a "go-no go" decision controlled by a water sensing device.

(4) Minimal shielding will be sufficient to protect typical electronic circuits from even a 10 percent degradation in performance considering a 20 Kw(t) Pu-238 or Po-210 source. The most severe radiation problem would come from space radiation rather than from the heat source.

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[redacted]

(5) Therefore, semi-controlled intact reentry is seen to require a very complicated reentry vehicle whose reliability would be low compared to ballistic reentry. However, active systems are the only way random intact reentry of large heat sources can be accomplished and it is felt that the components required can be developed and made reliable to the extent necessary to achieve semi-controlled reentry.

(6) The basic question which affects the applicability of semi-controlled intact reentry is not feasibility but application. That is, is the designer willing to use a vehicle which has its main usefulness on short lived orbits (aborts yielding one pass before reentry) where ground control may not be functioning, or will he provide enough backup in the basic system so that ground control can never be lost. If the latter is the case then controlled intact reentry making use of a ballistic vehicle can do all the jobs that semi-controlled intact reentry can do. At present the expense of proving a worldwide network of control stations along the launch orbital track of any nuclear system launch would be considerable. Hence, it is felt that there is a legitimate need for semi-controlled intact reentry. Further study on the logistics of intact reentry techniques for large heat sources is required before a final judgment is made, however.

[redacted]

3. Controlled Intact Reentry and Recovery

a. Operational Profile. The ultimate capability in disposing of nuclear power supplies used in space is to return the fuel loading to earth in a manner which does not compromise the containment structure of the fuel and allows complete control over the impact points of the reentering system. If this can be accomplished then disposal of the fuel in a remote area (i.e., oceans or deserts) or recovery of the fuel by selecting a particular landing area falls within the realm of possibility for the nuclear system designer.

As was shown in Section IV-B-1, the impetus for designing special reentry systems to achieve this is not pressing for the low power radioisotope systems. However, when power levels of the order of kilowatts are required, then the impetus is large from both the hazards and economic standpoint. In addition, random intact reentry cannot be tolerated from a political standpoint because of the international problems which could be raised by irresponsible irradiation of an alien population or the associated delivery of large quantities of fissionable material to a potential enemy.

Because of the nature of the end product of controlled intact reentry, that is, delivery of at least the nuclear heat source to a specific spot on the surface of the earth, only systems which are active in nature can be thought of for successfully meeting these goals. Passive systems which only have the capability to withstand reentry and impact can only satisfy random intact reentry criteria. The active system must have the capability to leave the space environment at a particular point in its orbit or trajectory on command of a ground or space station, dissipate its inherent kinetic energy without compromising the integrity of the heat source during reentry or impact and signal their earth impact position to search parties if recovery is desired.

The complexity of achieving this goal is primarily a function of whether the vehicle system using the nuclear power supply is manned or unmanned. The nature of the basic mission (i.e., orbital, lunar, planetary, etc.) also must be considered, but those manifest themselves in second order effects such as the change in weight of the heat shield or the amount of retrothrust required and do not have large effects on the character of active control systems and operational event sequences in the vehicle. It is the latter which are primary in achieving the desired goals.

b. The Techniques of Intact Controlled Reentry

(1) Manned Systems. Achieving controlled intact reentry of a nuclear heat source associated with a manned system allows some latitude in design and selection of the operational mode of the reentry vehicle. For example, the heat source could be brought back with the crew in the mission module reentry vehicle or resupply vehicles thereby using the normally available reentry system or it could be jettisoned by the crew at a specific

[redacted]

point in the orbit or trajectory of the manned vehicle and reenter ballistically within its own heat protection system. Accurate knowledge of the altitude, geometry, and inclination of the space vehicles orbit would allow selection of the proper retroenergies to allow precise control over the impact points to at least the same tolerances as present Mercury and Gemini flights.

If the heat source returns with a manned reentry body, the basic problems which arise are concerned not with the reentry but with integrating the heat source into the command and service modules during all the operational phases anticipated. Figure 54 shows some typical arrangements of the heat source and vehicles in various operational modes.

For large heat sources coupled to dynamic converters or for 2 - 3 Kw(e) thermoelectric systems as shown in Figure 54 separation from the command module under normal operating conditions will be required. In addition to the reduction in nuclear shielding weight obtained through separation and the inherent self-shielding of the service module, separation is anticipated because the life support systems in the command module might not tolerate the extra heat load imposed on them by a 30 - 50 Kw(t) thermal source in close proximity to the crew even with good insulation.

If this restriction is imposed and the heat source is outside of the reentry vehicle, then some means to reenter it, using the basic vehicle as a shield, must be devised. A preliminary solution is shown in Figure 54. The heat source is radiatively coupled to the service module heat exchanger hence it can be easily disengaged from the thermal loop. It is attached to a simple insulated manipulator boom. Upon separation of the service module in the reentry sequence, the heat source is rotated approximately 225 - 240° to the surface of the command module and there placed in a protective container flush with the vehicle skin. The boom can be designed to be alternately operated by the crew particularly for emergency use during a power failure preceding the reentry. The container has an ablative outer cover to protect the heat source from wake temperatures encountered during reentry and will contain a heat sink, such as beryllium, to absorb the nuclear heat on reentry (nuclear shielding for the crew may also be necessary depending on the isotope and exposure time). In addition, since the protective container is only activated on reentry, the residual cryogenics from the life support system may be bled through the heat sink during terminal stages of the flight. Below 100,000 feet the heat source can be separated from the vehicle or impact with it if recovery of the fuel is desired. It must be noted that the heat source protective container should be forward of the main parachute compartment and cannot interfere in any way with the operation of this subsystem. Figures 55 and 56 show the operational sequences envisioned for these systems for return from each space operational mode.

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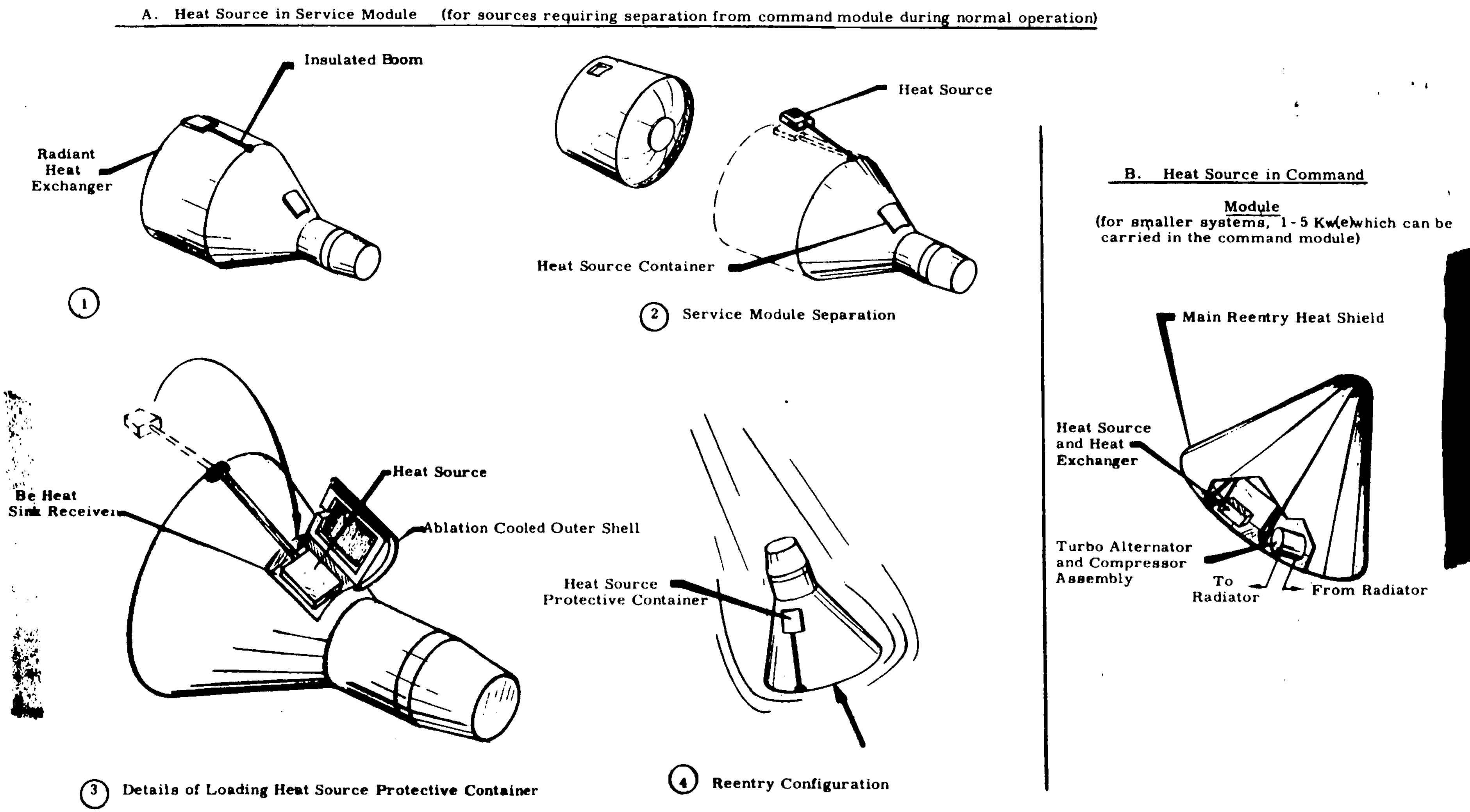
A. Heat Source in Service Module (for sources requiring separation from command module during normal operation)

①

② Service Module Separation

③ Details of Loading Heat Source Protective Container

④ Reentry Configuration

B. Heat Source in Command Module (for smaller systems, 1 - 5 Kw(e) which can be carried in the command module)

Figure 54 Typical Heat Source Integration with Space Vehicles in Various Operational Modes for Earth Reentry

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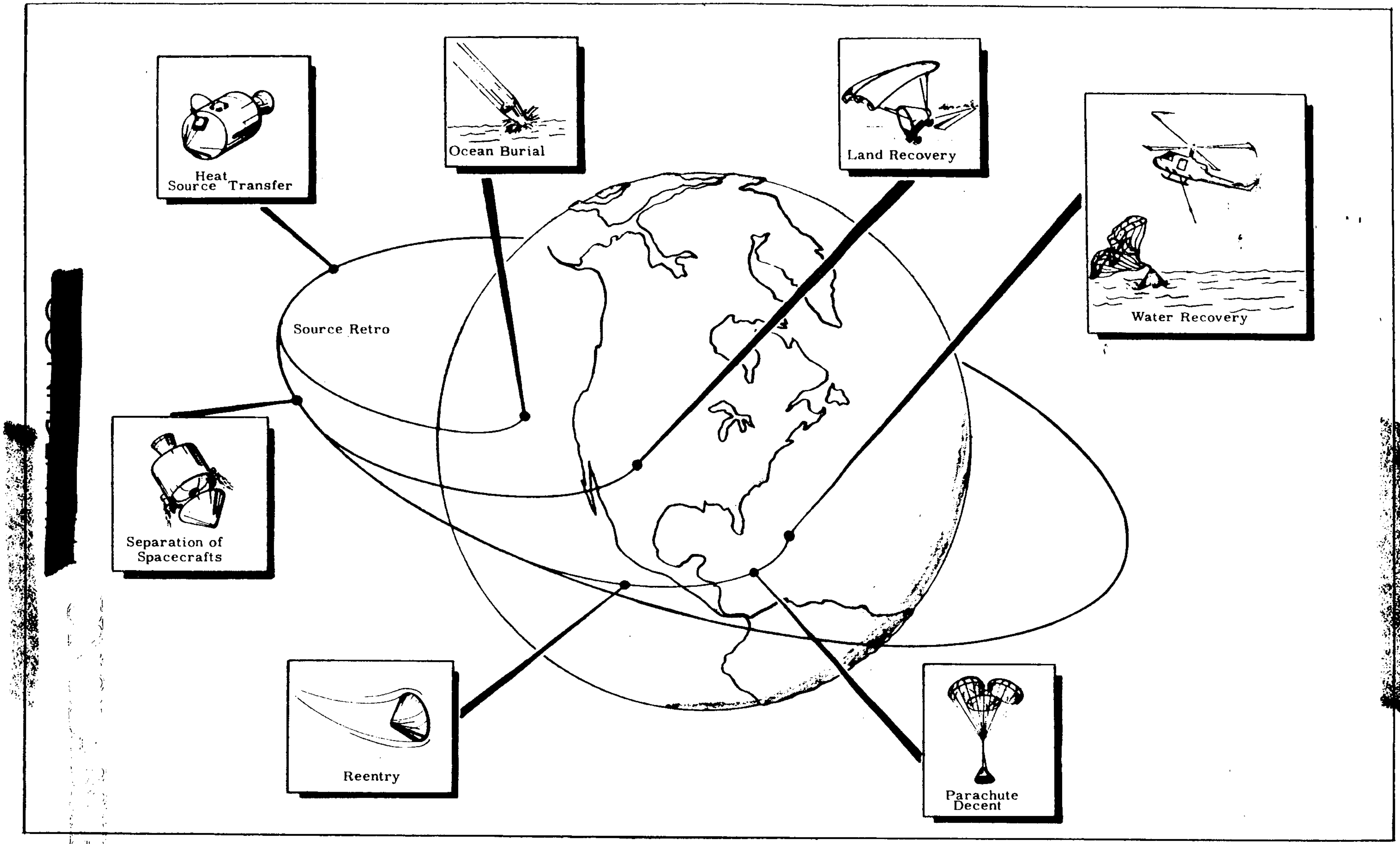

Figure 55 Earth Orbital: Intact Controlled Reentry With Recovery

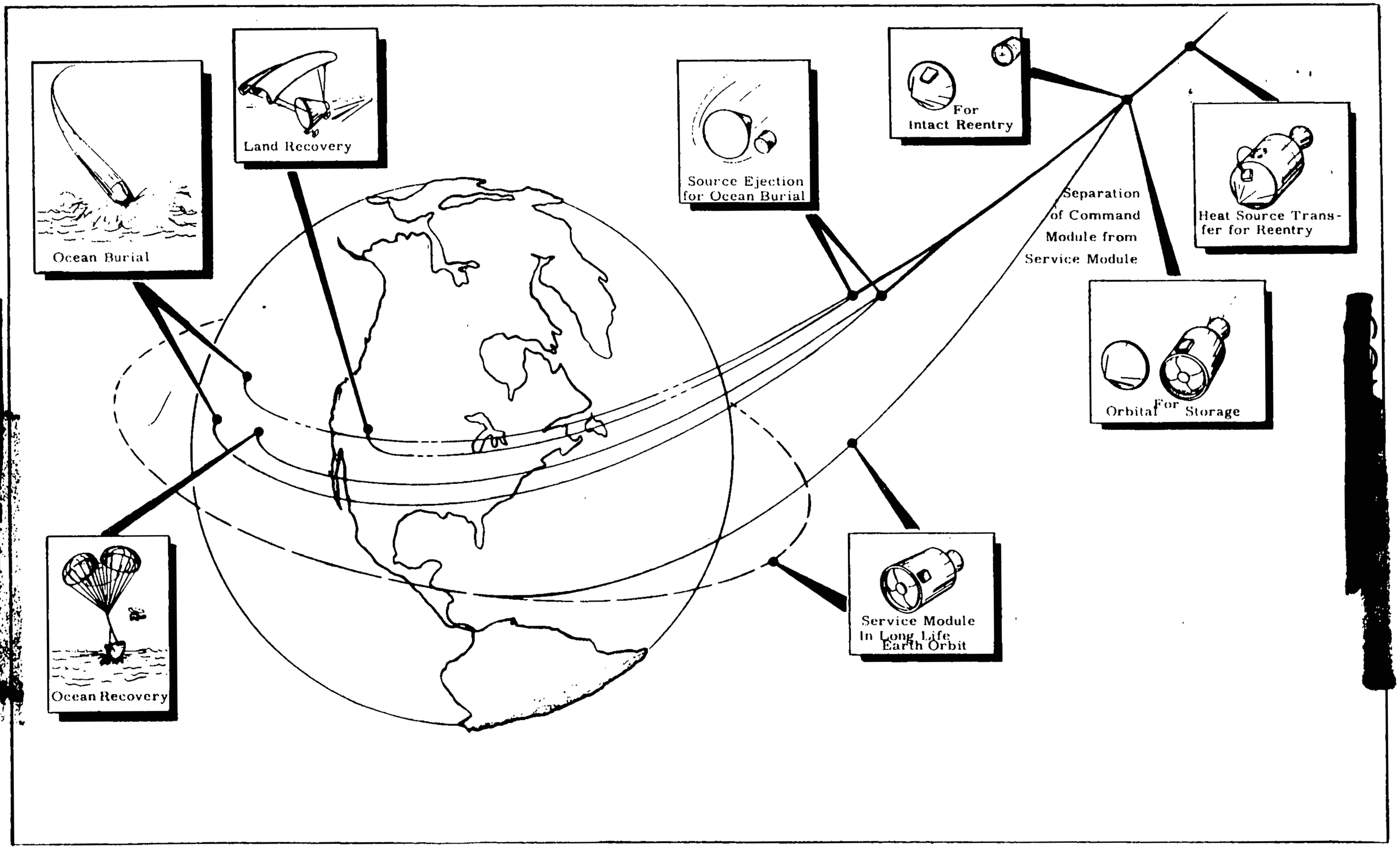
Ocean Burial
Land Recovery
Source Ejection for Ocean Burial
For Intact Reentry
Heat Source Transfer for Reentry
Separation of Command Module from Service Module
Orbital For Storage
Service Module In Long Life Earth Orbit
Ocean Recovery

Figure 56 Intact Reentry With Recovery or Non-Recovery

[redacted]

Figure 54 also shows a solution for smaller systems where the heat source is integrable directly into the command module. Since the heat source is already in the command module, it would make most sense to integrate the converter package directly as well to cut down on thermal losses and pressure drops. Turbine exhaust would be dumped to the radiators located in the service module.

Both the turbine exhaust and compressor and pump inlet lines would be blown off by explosive fittings at service module separation. The location of the heat source behind the heat shield as shown is forced on the designer based on maximum volume considerations. However, the best location for the heat source from a thermal standpoint during reentry is at the apex of the command module. But interference with the parachutes, excursion module egress hatch or flight instrumentation may make this position untenable. Therefore, the station right behind the heat shield is the only one suitable based on the preliminary inspection.

The exact implications of this choice on the heat source design is not clear without doing a detailed analysis, however, this station is one of the hottest on the body during reentry due to the aspiration of the stagnation air from the front of the main heat shield by the corner expansion. Therefore, active cooling as well as a heat sink may be required and could be accomplished by "bleeding" the residual cryogenics into the heat source heat sink.

(a) Reentry Independent of the Manned Vehicle. In manned systems where the heat source is to be reentered without the benefit of the manned reentry vehicle special techniques must be developed. As in the case of random intact reentry the heat source must have its own reentry body which will both protect the system from reentry heating and from meltdown by internal heat generation.

Figure 57 shows the operational sequences required to return a large heat source from a manned space station or laboratory such as MORL. It is envisioned that the heat source would be launched in position on the service module of the space laboratory and operate in this position through its life cycle. When reentry is desired the space station would separate from the module. This is accomplished by simply firing the explosive attachment bolts holding the reentry vehicle to the service module. No connections between an isotope heat source and the service module are envisioned because a radiant heat exchanger can be used. Those bolts may have to be shielded in the presence of the nuclear source to provide reliable separation after long time exposure, but this is not considered

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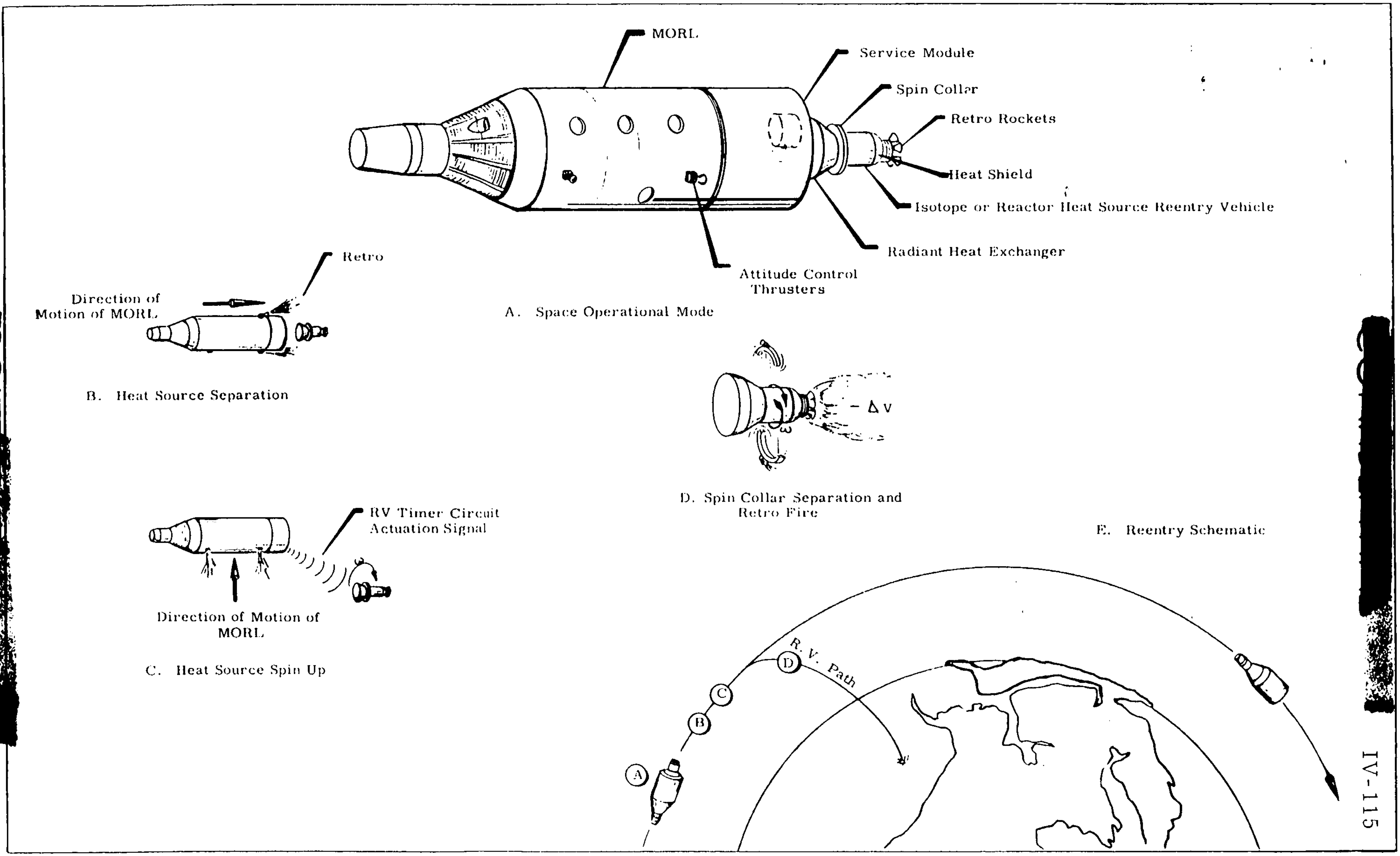
MORL
Service Module
Spin Collar
Retro Rockets
Heat Shield
Isotope or Reactor Heat Source Reentry Vehicle
Radiant Heat Exchanger
Attitude Control Thrusters
Retro
Direction of Motion of MORL
RV Timer Circuit Actuation Signal
Direction of Motion of MORL
- Δ v
R. V. Path
A
B
C
D

A. Space Operational Mode

B. Heat Source Separation

C. Heat Source Spin Up

D. Spin Collar Separation and Retro Fire

E. Reentry Schematic

Figure 57 Operational Sequences and Reentry Vehicle Configuration Required To Return A Large Heat Source From MORL

[redacted]

a serious problem. As soon as separation has occurred the MORL would change its orbital plane slightly (i.e., several hundred feet) relative to the reentry vehicle in order to clear itself before the retros fire. Stability of the reentry vehicle during this phase of the operation is critical therefore as soon as the explosive bolts fire the reentry vehicle would be spin stabilized by a collar around the vehicle to which small solid propellant rockets are attached and fire normal to the axis of the reentry vehicle and in opposite directions from each other.

When the MORL is clear of the reentry vehicle a signal is sent to the timer circuit aboard the reentry vehicle which will activate the retros at the right point in time to yield eventual impact in a predetermined area on the earth's surface. It must be stressed at this point that the firing time should not be activated by the explosive bolt firing although this is attractive, since it relieves the requirement for electronics aboard the reentry vehicle and hence gives greater system reliability. The MORL must have sufficient control over the reentry vehicle's operational sequencing so that if anything goes wrong and the MORL cannot change orbital position a collision between the reentry vehicle and MORL can be avoided. Reentry can always be instituted on the next orbital pass since the relative position of the MORL and the reentry vehicle will not change and the reentry vehicle will remain spin stabilized thereby avoiding tumbling.

Therefore, at the proper moment selected by the MORL crew and the ground stations which can track both the MORL and the reentry vehicle, the retro package is fired and reentry of the heat source is instituted on a precise ballistic trajectory which will yield any desired landing point on the orbital track. The retro package is allowed to burn off during reentry keeping the required mechanical functions to a minimum. This should not induce serious perturbations to the vehicle during the reentry if the package is designed to ablate uniformly or is held on by fusable bands.

(2) Unmanned Systems. The operational techniques for intact controlled reentry of the heat source reentry system for unmanned systems are the same as for manned systems except for the control of the events being strictly in the hands of a ground station. This, of course, affects the reliability of the operation since on board mechanical and electronic systems are the only bridge between activating the nuclear source reentry and the ground. Therefore, it is expected that sufficient redundancy and shielding will be required to assure a high probability of success.

[redacted]

[redacted]

Several important considerations have to be evaluated before the feasibility of the unmanned system can be established. They are:

(a) The weight of the retro rocket system.

(b) Attitude control of the unmanned vehicle.

(c) The effects of timing errors in retro rocket firing on impact position.

(3) Retro Rockets - Operational Limitations. For any initial circular orbit, there is a minimum velocity increment requirement opposite to the direction of the vehicle to insure earth impact within one orbit. As the altitude of the initial orbit increases, the velocity increment requirement increases (within our range of interest) as shown in Figure 58.

These velocity requirements can be represented by the ratio of propellant weight to vehicle weight. For typical chemical rocket specific impulses of 200 - 300 seconds, the ratio can be shown as a function of initial altitude(Figure 59). This figure shows that for a 100 pound payload at an altitude of 100 n. mi., it would take about 2 - 3 pounds of propellant.

With an unmanned mission, the retro rockets must be started and aimed by either ground control or a very complicated guidance system. Assuming that the rockets could be started, the direction of thrust is the main problem. If the vehicle is tumbling or the thrustors are approximately 180° out of phase, the rockets will increase orbital altitude rather than cause impact.

A study of the effect of angle deviation in the retro rockets (shown in Figure 60) shows that for altitudes below 500 miles retro rocket weight is no larger than 10 percent of the reentry vehicle weight. However, the effect of mission on the retros is quite severe in terms of impact at a given surface point after the command to fire is obeyed by the vehicle. No more than a ± 10° deviation in the attitude can be allowed assuming a ± 50 mile error is acceptable at impact. Therefore, the attitude control of an unmanned vehicle from which a nuclear heat source is to be reentered intact is the critical factor.

Since the use of the reentry technique in this case implies that the space vehicle may outlive its usefulness long before the nuclear heat source has, active attitude control, by gas jets, of these vehicles is probably out of the question. The only attitude control systems which are acceptable therefore are gyroscopic or gravity gradient. Since the gyroscopic systems require electric power to maintain themselves and this could easily fail; as a matter of fact it could be the prime reason to require return of the heat source and converter for investigation; the gravity gradient system is the most attractive for fulfilling the attitude control reliability necessary to achieve unmanned controlled intact reentry.

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[redacted]

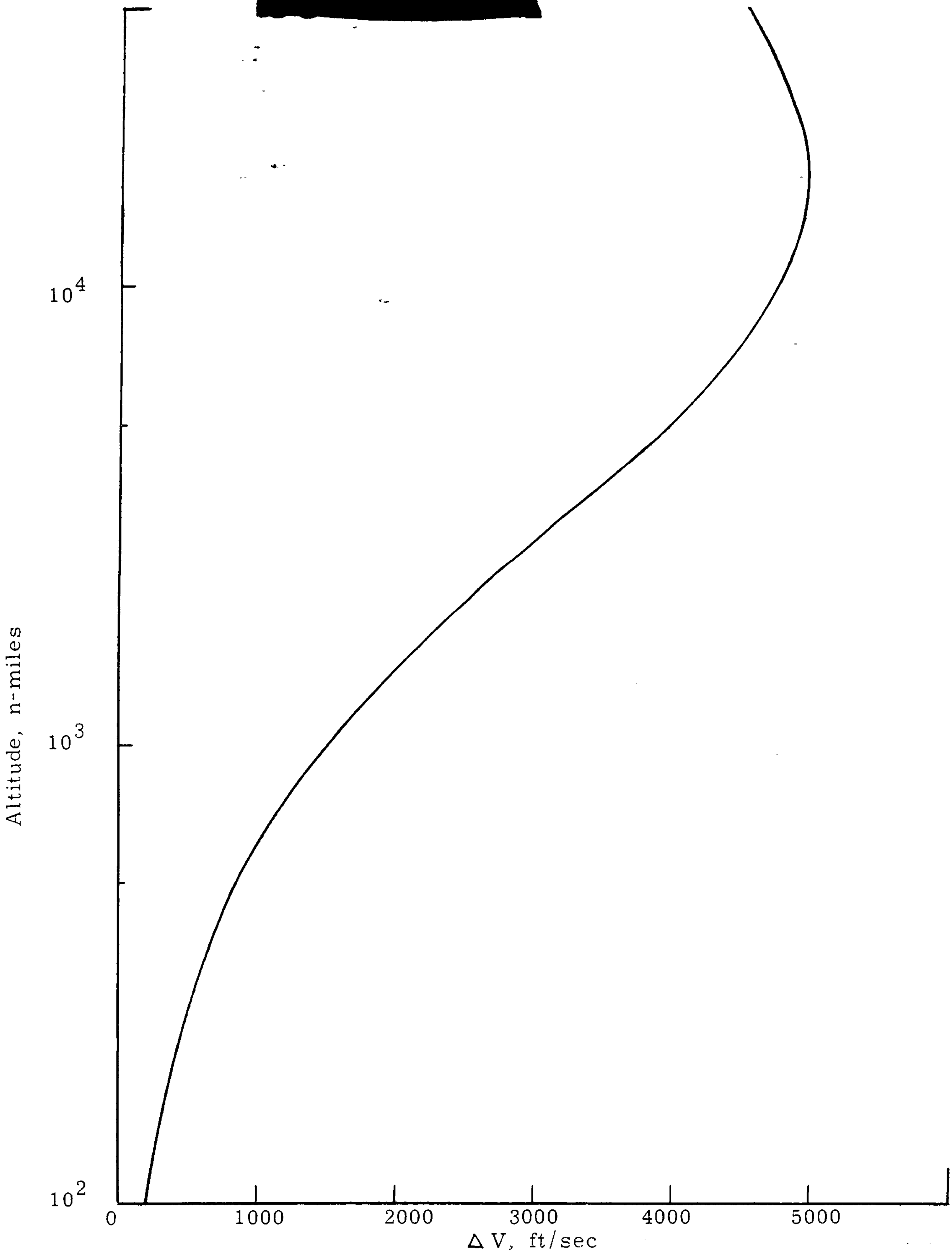

Figure 58. Retro-Rocket "ΔV" Requirements as a Function of Altitude for Earth Impact Within One Orbit.

[redacted]

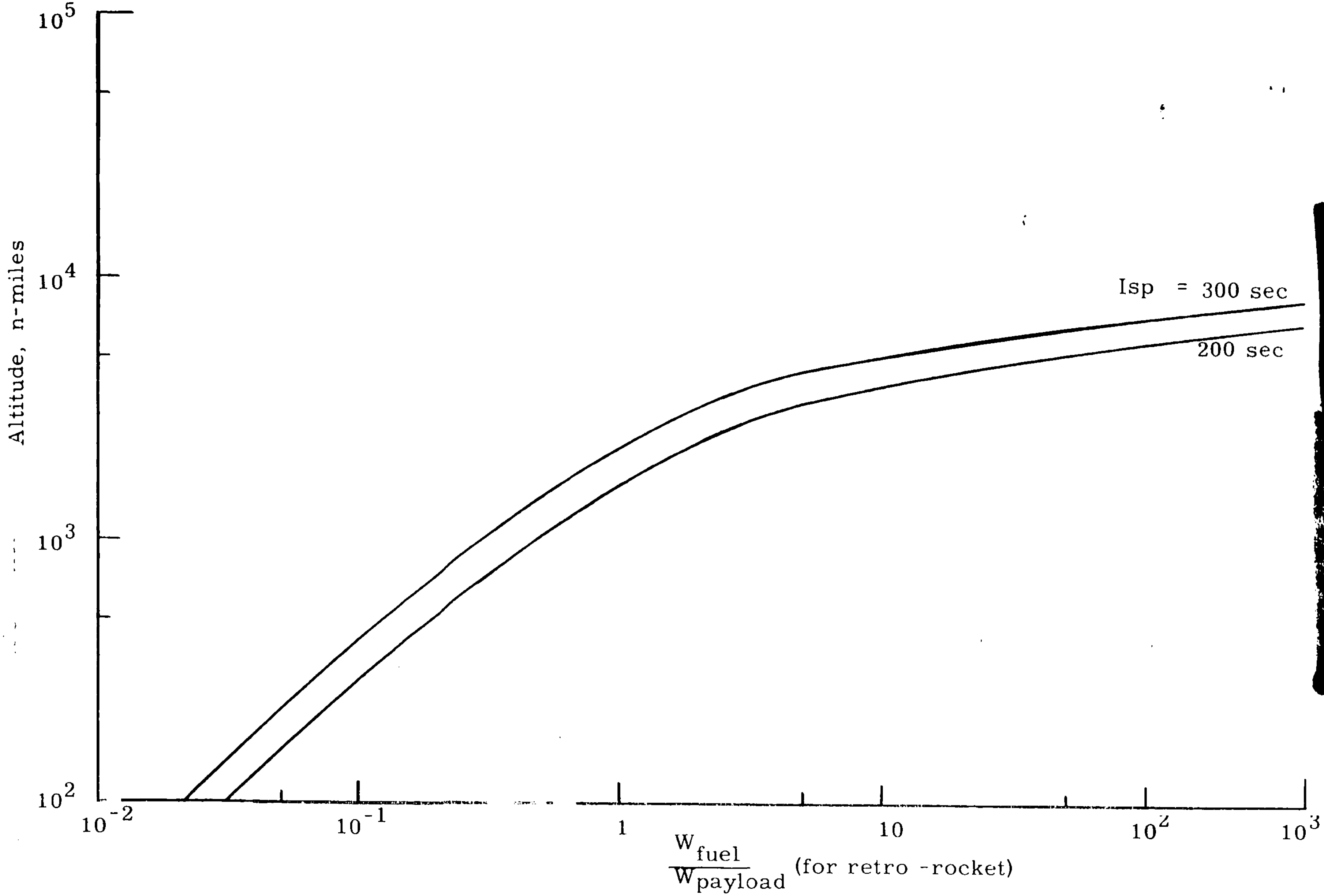

Figure 59. Ratio of Retro-rocket Fuel to Payload Weight vs Altitude for Earth Impact Within One Orbit.

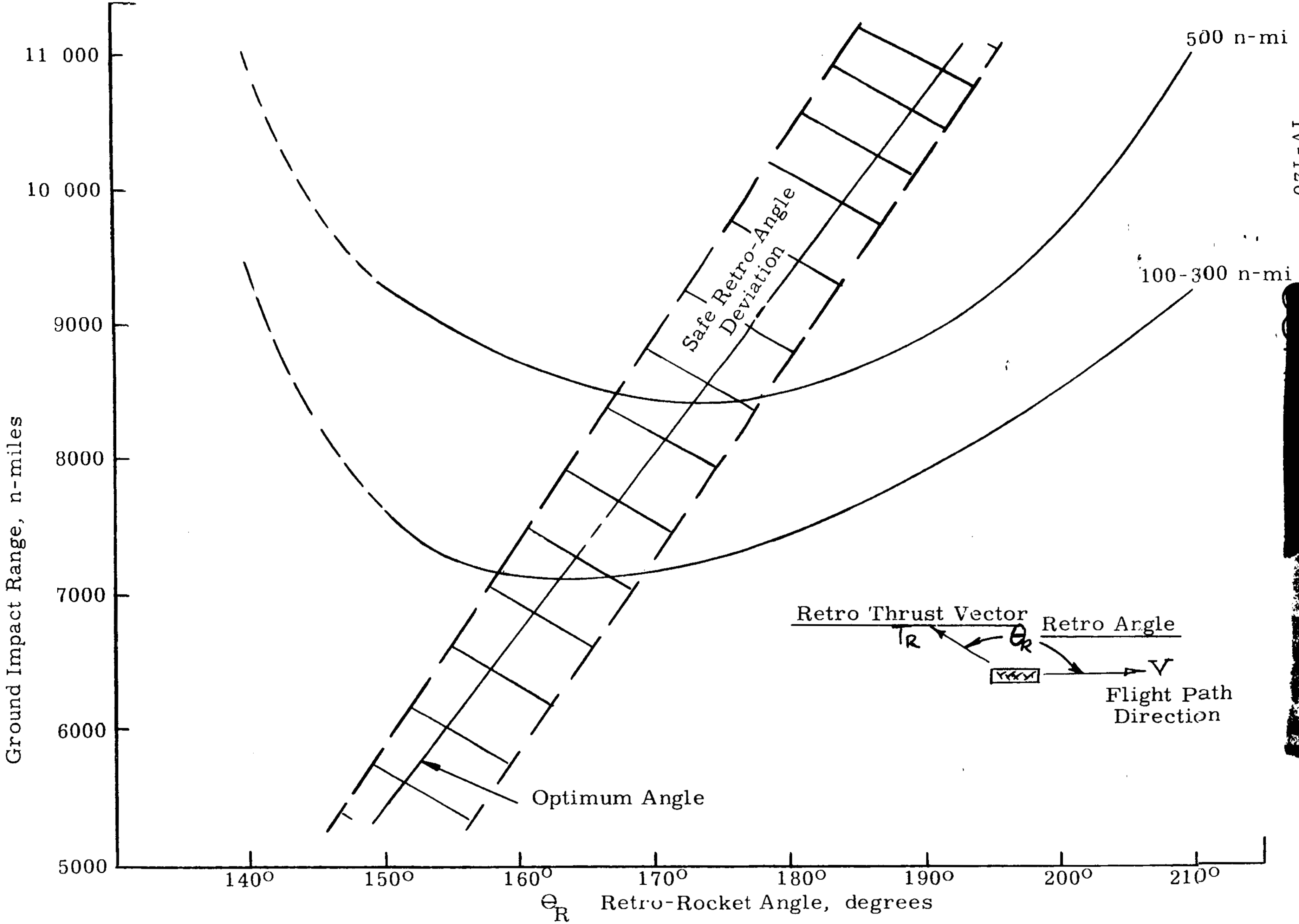

Figure 60. Ground Range vs Retro Angle for Various Initial Orbits.

The techniques of achieving controlled intact reentry in the unmanned case are roughly the same as described in Section IV-B-3-c-(1) after one has achieved a reliable and stable platform as described above to which commands can be given and obeyed remotely.

c. The Design of Intact Controlled Reentry Vehicles. The design of reentry vehicles which are told when to disengage from the spacecraft is far simpler than semicontrolled random reentry. Here the vehicle contains no flight control system and must only survive a ballistic reentry. Therefore, the main problems are thermal and mechanical and not involved with complex electronic control systems.

This section presents a number of potential solutions to the many problems that have been described in the preceding sections. They should be considered as indicating promising lines of attack rather than established solutions.

Any isotopic heat source designed for the intact reentry mode must consider the unique heat transfer and heat storage problems associated with a material whose internal heat generation rate cannot be altered by any way other than time. Specifically, the fuel capsules must be able to reject sufficient heat from their surfaces to limit the structure interface temperatures to 1800°F or less in both normal and abnormal operating modes. This materials compatibility problem varies from isotope to isotope and cladding to cladding. However, 1800° appears, at this time, to be an upper limit to the allowable interface temperature, based on using a Rene' 41 support structure.

During normal operation it is not difficult to select system operating parameters such that the temperature limit for compatibility is not exceeded. The problem of designing a heat source for intact reentry becomes difficult when consideration is given to the need to reject heat by some alternate mechanism, such as the case of failure of the primary heat rejection mode, i.e., working fluid loss, pump loss, or turbine seizure. The difficulty arises from the need for an alternate heat rejection path around a heat shield which is designed to reject the aerodynamic heat of reentry. The designs described in the following paragraphs offer several approaches to the solution of these problems.

(1) A Solution for Low Power Density Fuel. Figure 61 presents a design concept for a heat source employing $Pu^{238}O_2$. The drawing details are limited to the containment of the fuel capsules within a reentry body. Additional components such as the heat transfer system are omitted. The principal of the design is that the heat is extracted internally during normal operation but is radiated from the outer cylindrical surface during all abort situations. It is necessary that the body stabilize at a high altitude during reentry so that reentry heating will be confined largely to the nose cone and the flare from which it will be reradiated. This can be accomplished by spinning in space but a backup technique is necessary and results in the high drag design as shown in Figure 61.

[redacted]

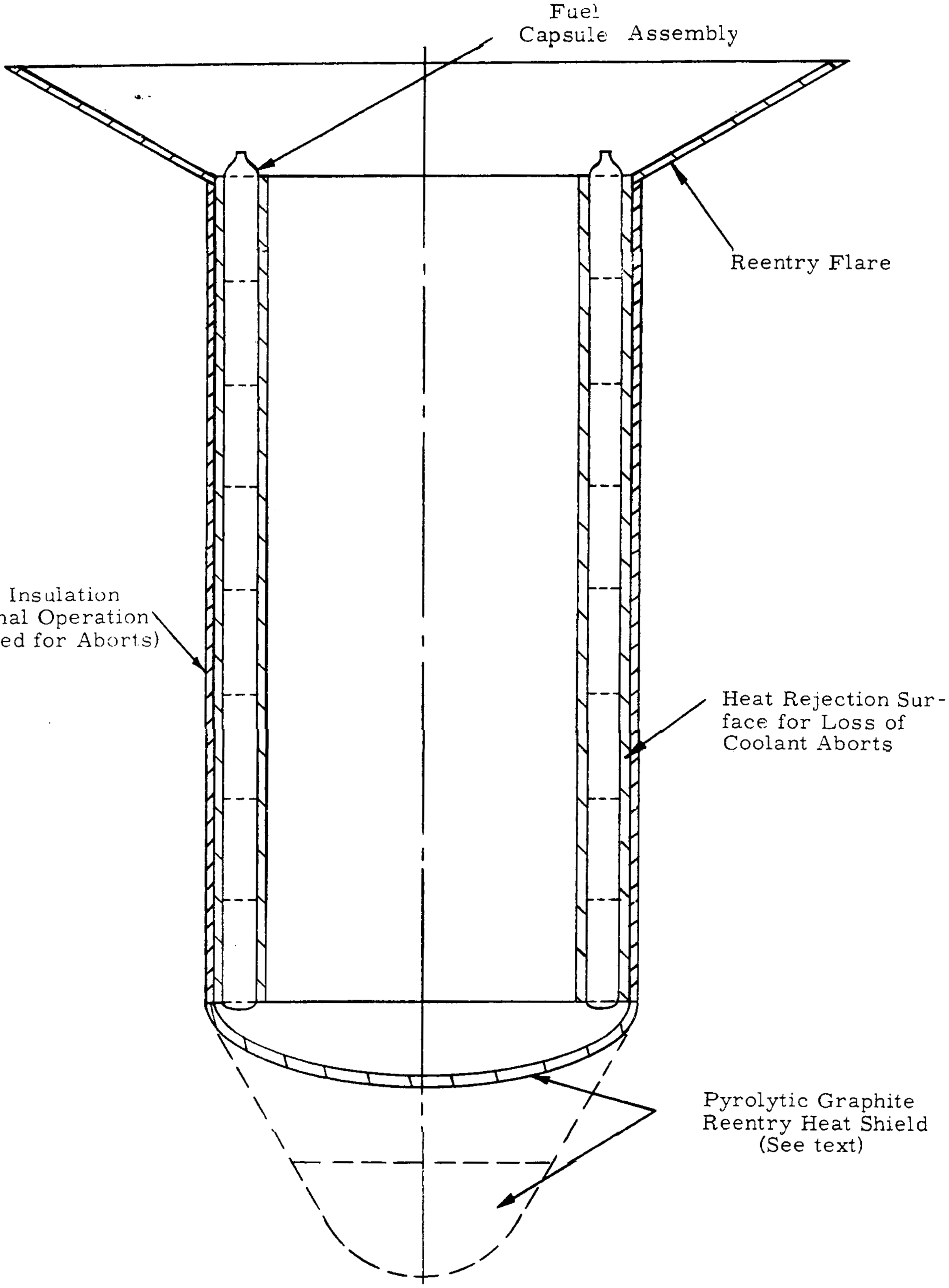
Fuel Capsule Assembly
Reentry Flare
Thermal Insulation for Normal Operation (Jettisoned for Aborts)
Heat Rejection Surface for Loss of Coolant Aborts
Pyrolytic Graphite Reentry Heat Shield (See text)

Figure 61. Design Concept for a Low Power Density Heat Source.

[redacted]

This concept has been sized for a fuel loading of 100 Kw(t) utilizing Pu-238 and encapsulated in containers typical of present day designs. With a cylindrical section 2 feet in diameter and 4 feet long, it is found to have slightly greater surface area than that needed to reject the heat in abort situations provided the thermal insulation can be jettisoned. Alternate means by which this can be accomplished are described later in this section. The cylindrical section consists of a thick walled beryllium tube bored out to contain the fuel capsules and the heat extraction system for normal operation. The nose cone is pyrolytic graphite and the flare is beryllium faced with pyrolytic graphite.

Any attempt to increase the power density of the reentry body will run into problems caused by the need for a second inner layer of capsules. These capsules would have to transfer their heat past the outer layer of capsules during aborts. This would lead to a higher temperature for the inner capsule layer than the outer capsules. Such an arrangement would require that the heat rejection surface temperature be lowered to avoid a materials compatibility problem in the inner capsules, thus limiting its ability to dump the full heat load.

The body shape shown in Figure 61 is a compromise between the ideal high drag configurations described in Section IV-B-1 and the requirement for providing sufficient uninsulated heat rejection area. Design of its aerodynamic characteristics will be very important to insure the backup that the tumbling could be damped at a high altitude in order to limit aerodynamic heating of the heat rejection surface and the back of the body. In general, stability will be improved by reducing L/D (length to diameter ratio), increasing flare area, and moving the center of gravity toward the nose. It is apparent that many of these requirements are in conflict and that achievement of an optimum design will be difficult.

The weight estimates for this configuration show that a 43 pound pyrolytic graphite heat shield is required at the nose to meet the necessary beryllium structural strength requirements at the attachment points of the nose cone. The pyrolytic graphite shield is also extended to the basic flare and weights of the order of 10 - 20 pounds are estimated. Approximately 350 pounds of beryllium are needed for the block itself. This block serves multiple functions. It must absorb that heat which is being generated by the isotopes if the body is undergoing tumbling reentry and it must absorb whatever heat input comes into the beryllium during the same period. These two constraints determine the amount of beryllium which is necessary in the barrel part of the assembly. Preliminary calculations indicate that the total heat source weight will be approximately 1280 pounds including an assumed weight of 40 pounds for the heat exchanger.

[redacted]

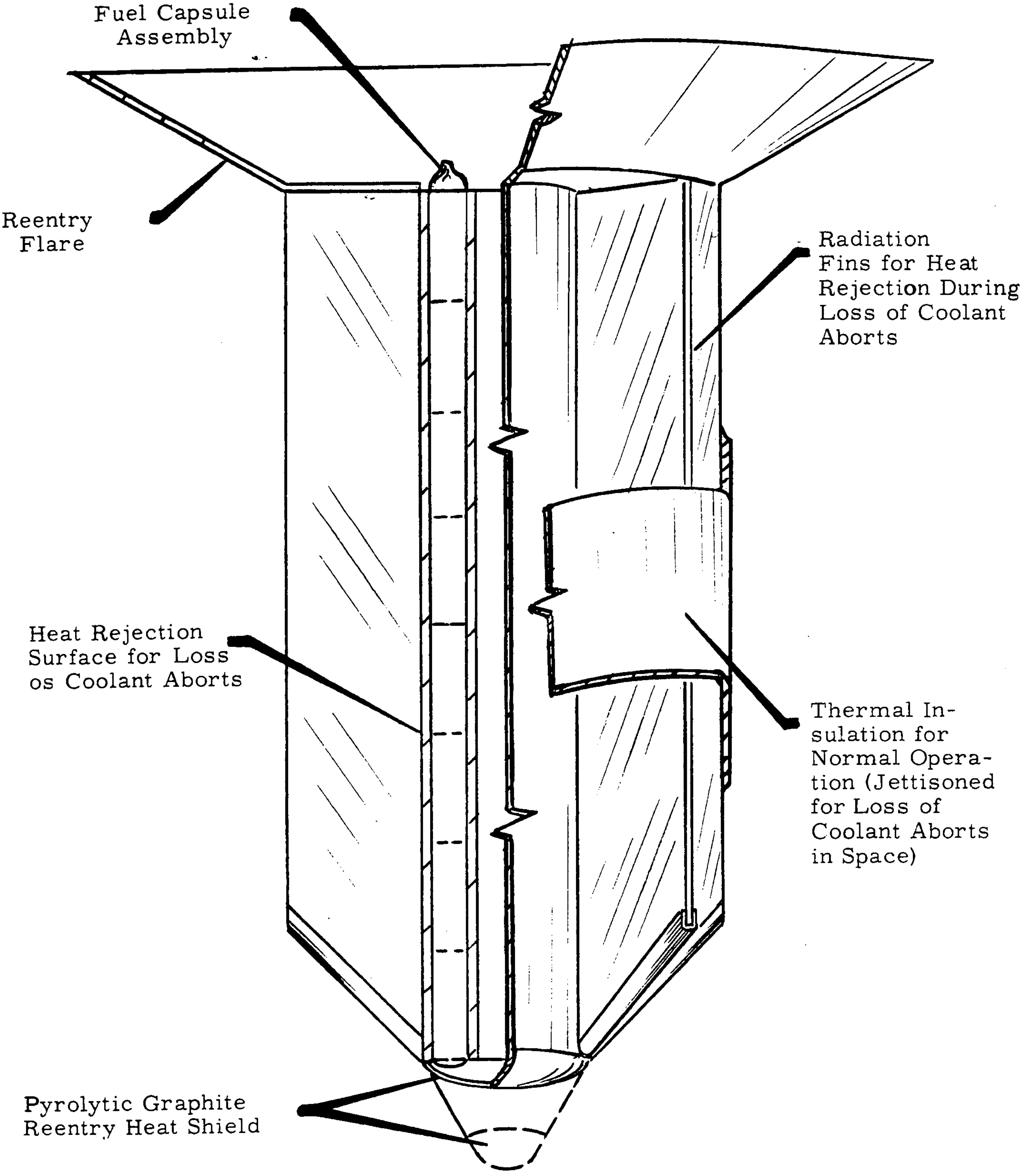
Fuel Capsule Assembly
Reentry Flare
Radiation Fins for Heat Rejection During Loss of Coolant Aborts
Heat Rejection Surface for Loss os Coolant Aborts
Thermal In-sulation for Normal Opera-tion (Jettisoned for Loss of Coolant Aborts in Space)
Pyrolytic Graphite Reentry Heat Shield

Figure 62. Design Concept for A High Power Density Heat Source

[redacted]

(2) A Solution for High Power Density Fuel. Figure 62 presents a design concept for a heat source employing Po-210 fuel. The design is similar to the previous one except that in this case it is limited by the heat rejection surface area and it is necessary to add fins to the basic cylindrical surface to provide sufficient area to protect the fuel capsules during loss of normal cooling. This configuration has been sized for 100 Kw(t) at time of loading and contains 21 capsules with a 25 percent void volume. Calculations indicate that the weight of this heat source would be approximately 475 pounds.

The same problems of aerodynamic design are present with the additional problem of predicting the behavior of the fins during reentry. It is felt that the fins will receive the least amount of heat if protected individually on their leading edges as indicated. However, the presence of fins obviously presents an additional path by which aerodynamic heat can be fed back into the fuel capsules. This problem would have to be analyzed in detail. It is also apparent that this design presents more opportunity to improve stability by decreasing L/D (length to diameter ratio), than the previous one but the interaction of fin heating presents a limit to increasing the frontal area. Also, the possibility of partial burnoff of the fins and its effect on stability and attitude must be considered.

A major point that would have to be established early in a design program would be whether or not there is a limit to the size of a single heat source module of this type. Many factors point to the possibility that the optimum solution may be to limit the size of a single source to some value lower than 100 Kw(t), achieving full power by use of several modules. Factors which tend to limit size are ground handling, i.e., shipping and loading operations, nuclear criticality problems (if Pu-238 and/or Cm-244 were used instead of Po-210) difficulties in handling internal heat during no cooling periods, and problems in integration with the space vehicle, i.e., it may be much easier to fit several modules than one large source into a vehicle configuration. Modularization would also provide greater flexibility and, in some applications, reliability. The compensating advantages of a single source are lighter weight and greater simplicity of attachment and heat transfer to the power conversion system.

(3) The Rotornet Decelerator. An alternate reentry vehicle design of great promise in its application to intact controlled reentry nuclear systems is the rotornet shown in Figures 63 - 65. Basically, it is an active system requiring spin stability to be effective, but it holds the promise of significantly reducing the weight of the heat source thermal protection system during earth reentry. This is particularly important if the reentry is at planetary return speeds. The systems discussed previously would be exorbitantly heavy at

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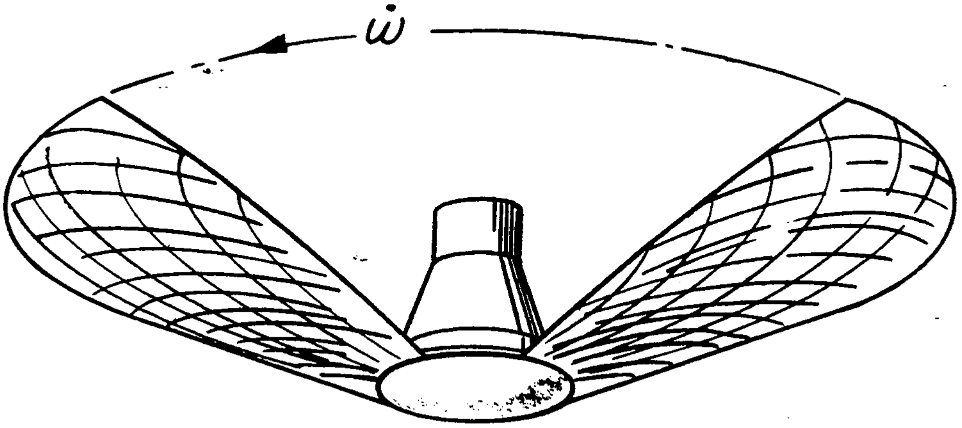
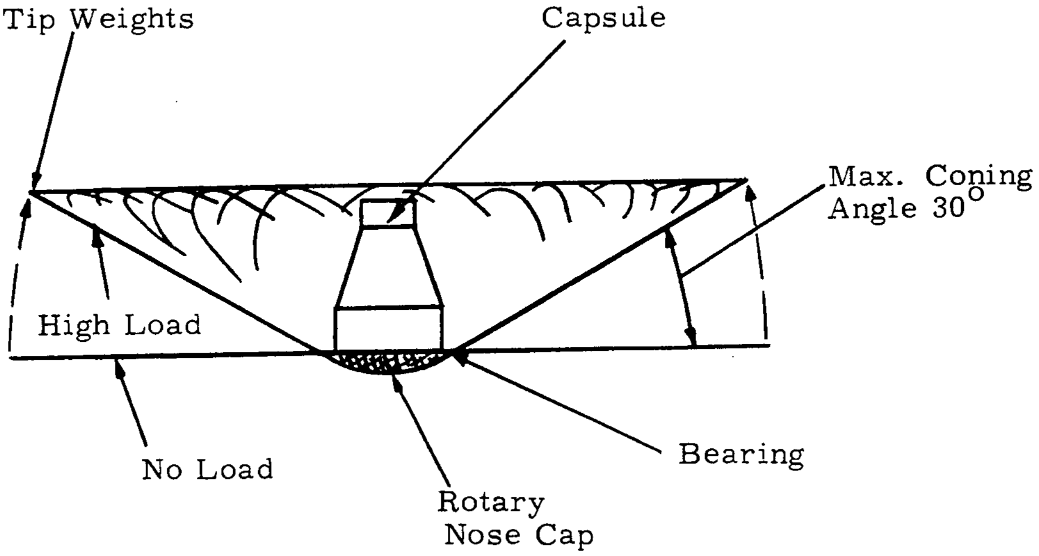


Rotornet Decelerator



General Arrangement

Figure 63. The Rotornet Decelerator General Arrangement

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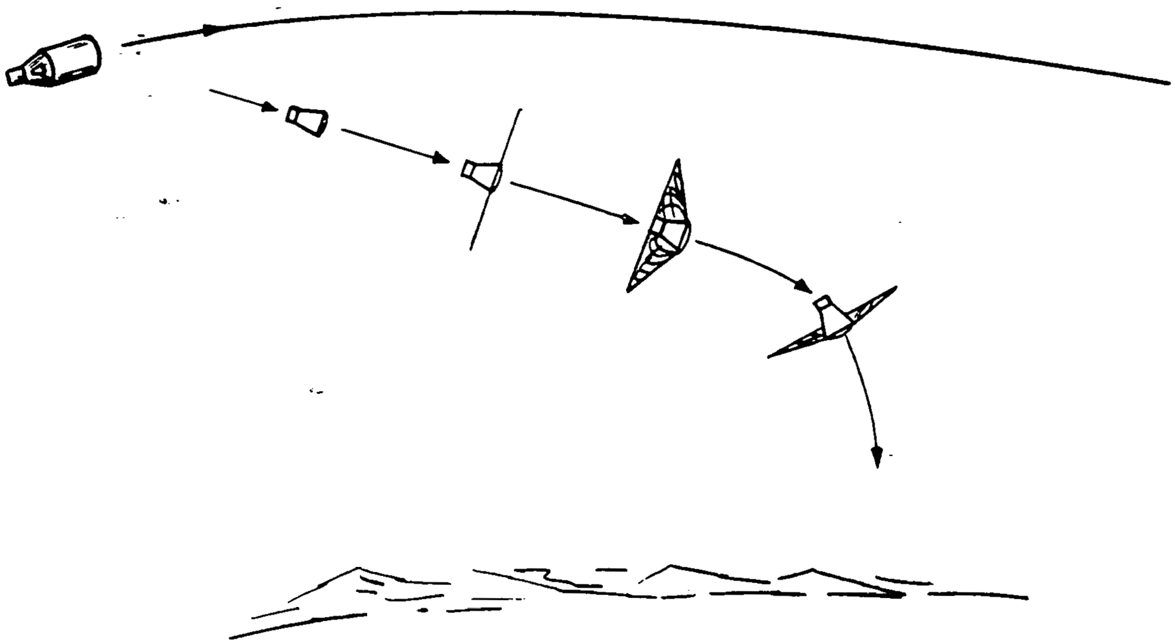
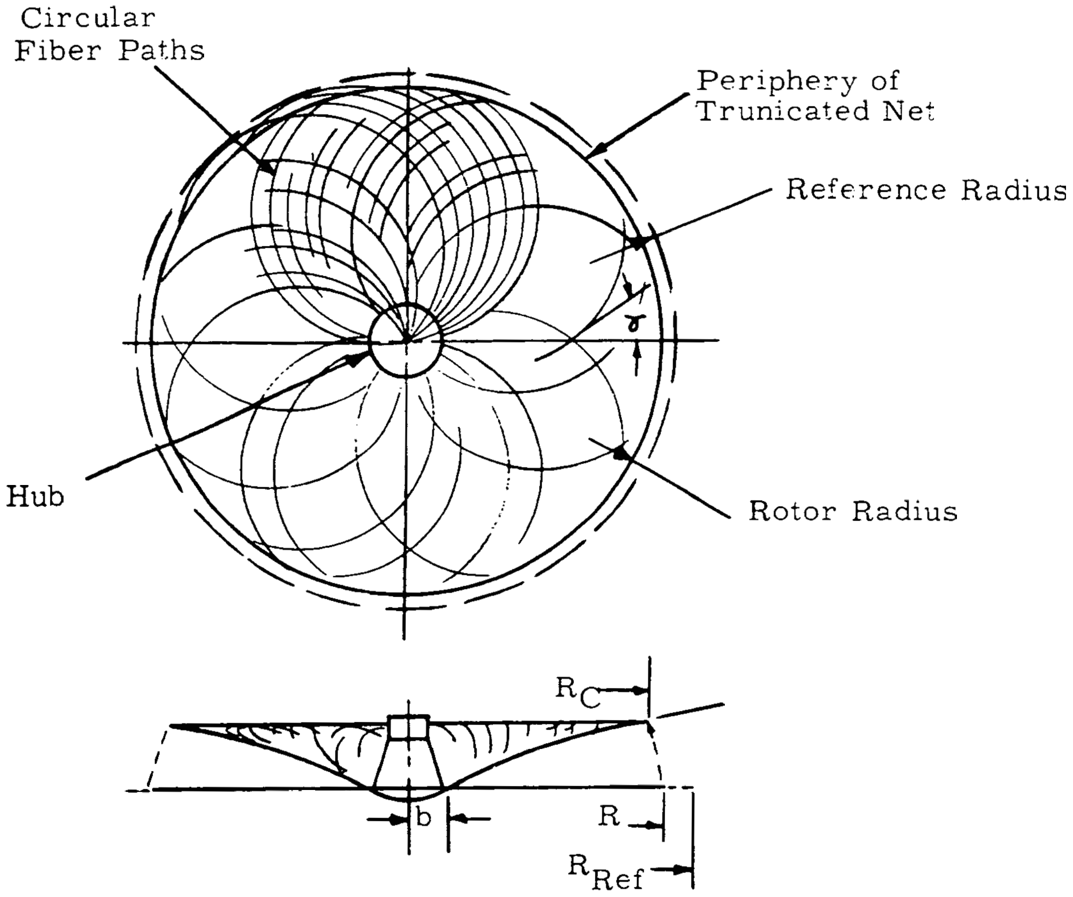


Typical Operational Sequence Earth Orbital Systems



Adaptation of Isotensoid Disk to Rotornet

Figure 64. Orbital Return Operational Sequence and Typical Structure of Rotornet.

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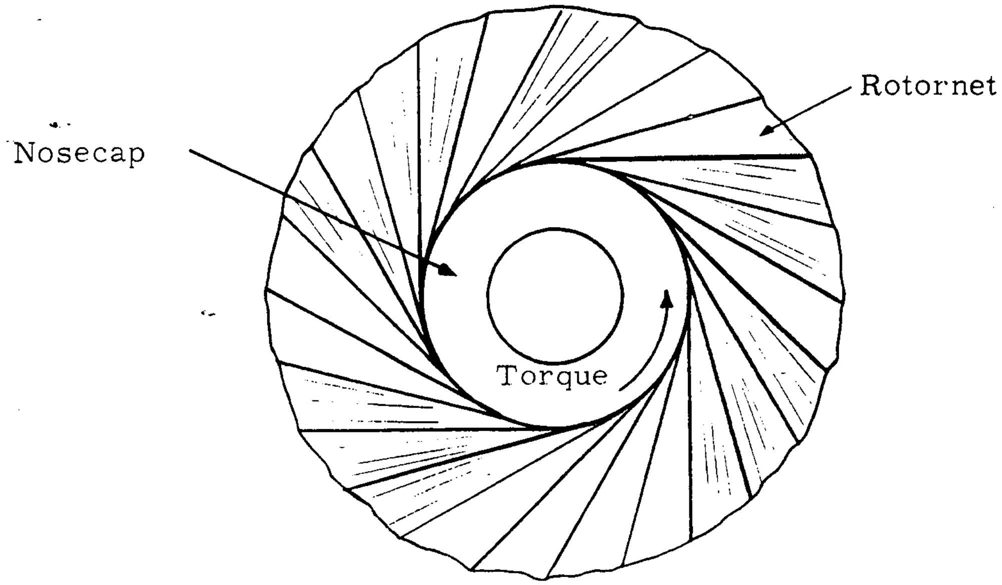

Typical Deployment of Rotornet

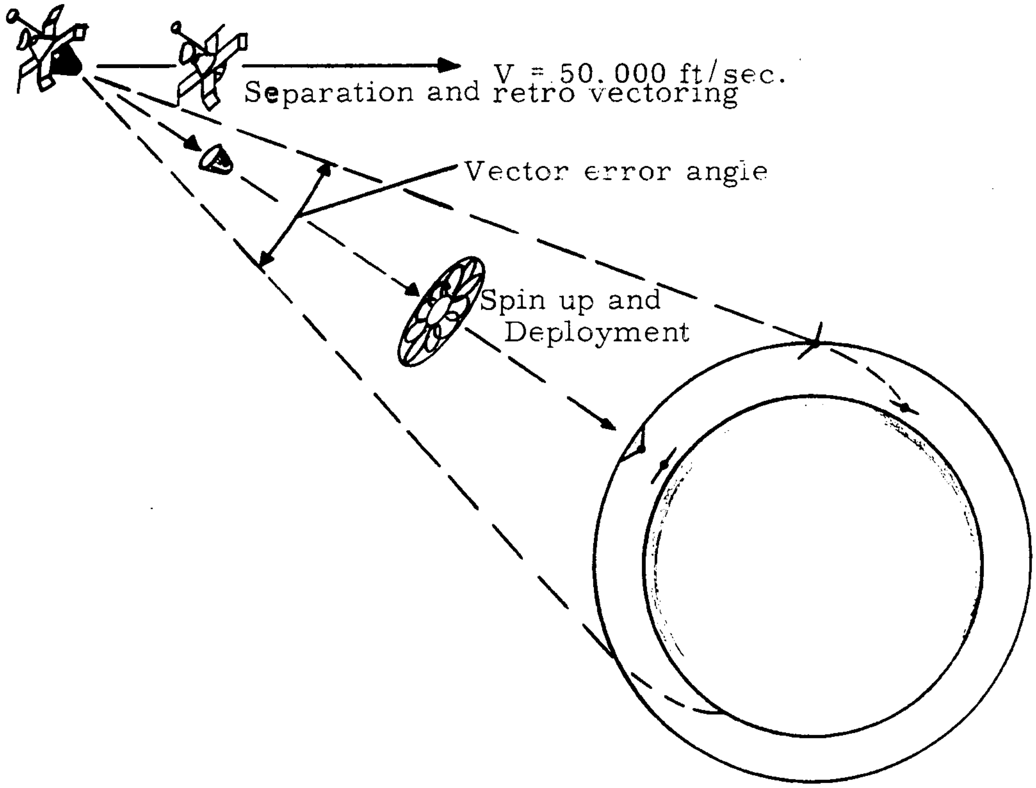

Figure 65. Earth Flyby Reentry from Planetary Mission and Typical Deployment.

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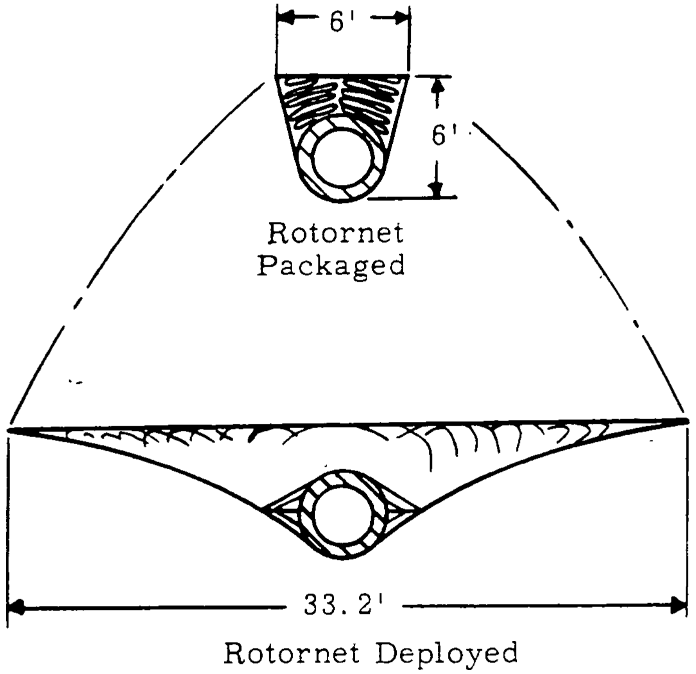
[redacted]



Payload	290 lbs
Decelerator	10 lbs
Rotornet	50 lbs
Total	350 lbs

Rotornet Design for Entry on Mars at 90° and 25,000 ft/sec.

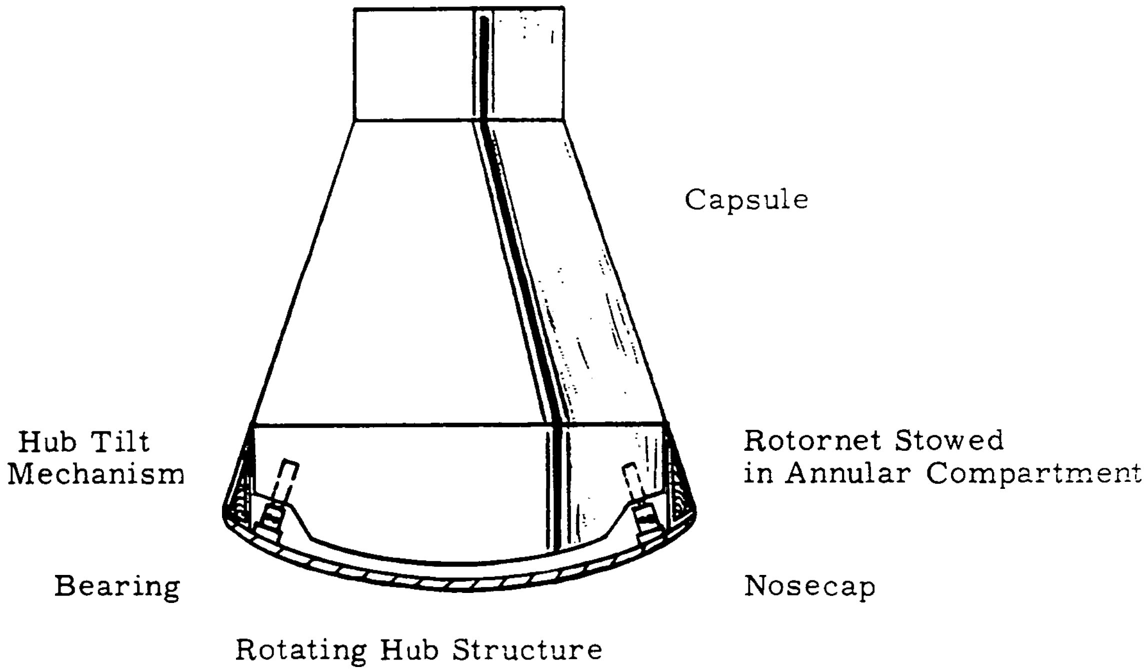


Figure 66. Typical of Earth Entry Vehicle Application of Rotornet.

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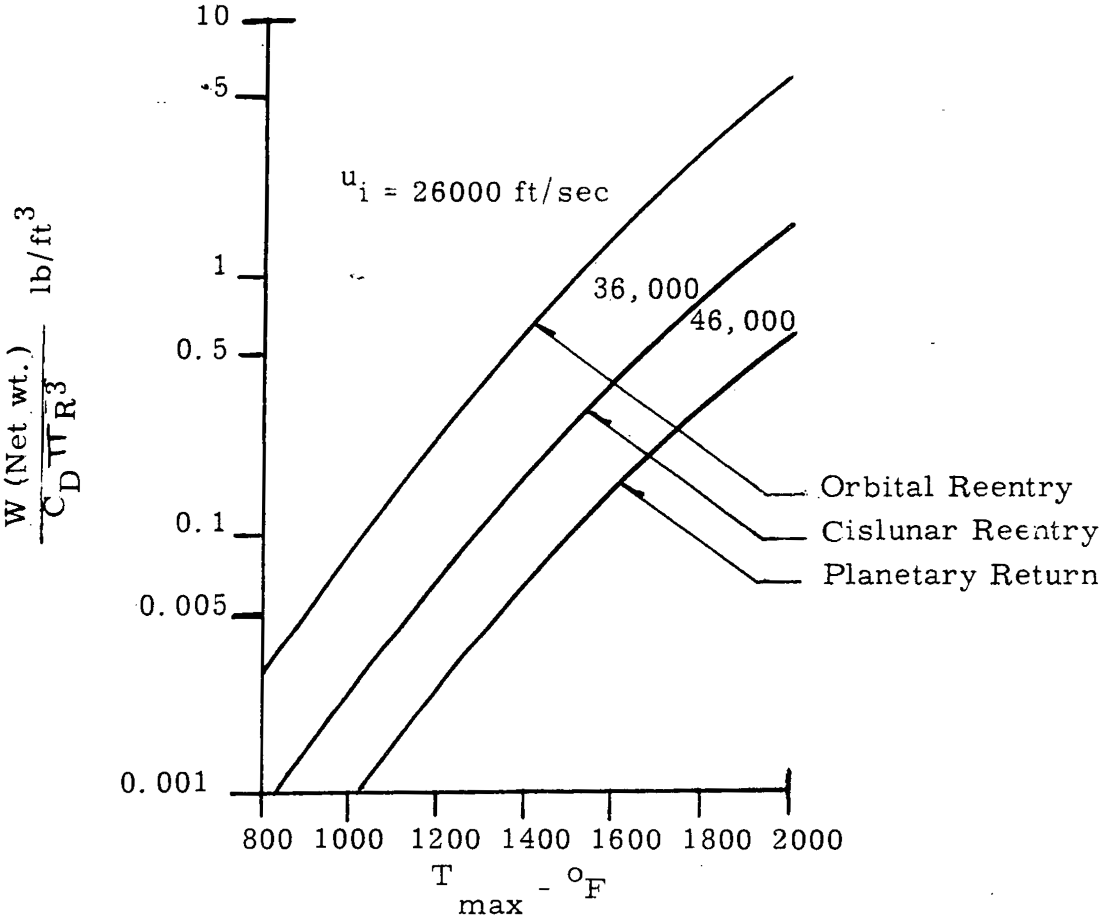
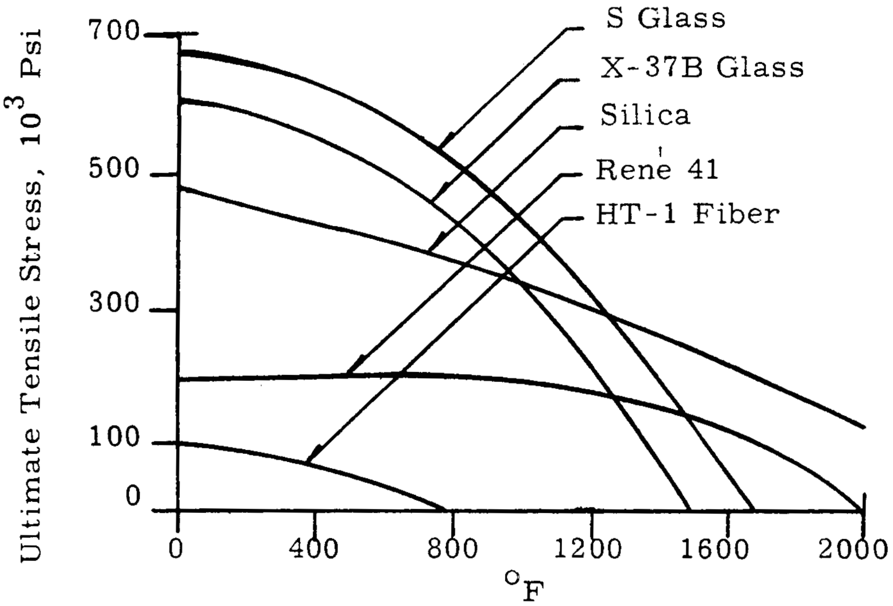


Maximum Temperature vs. Load Density



Strength vs. Temperature for Structural Fibers

Figure 67. Rotornet Load Density and Strength vs. Temperature

[redacted]

u_i 26,000
36,000
36,000
36,000
46,000
46,000
N_N 10
10
15
10
20
20
t 1600°F
1600°F
1600°F
1200°F
1600°F
1200°F
100,000

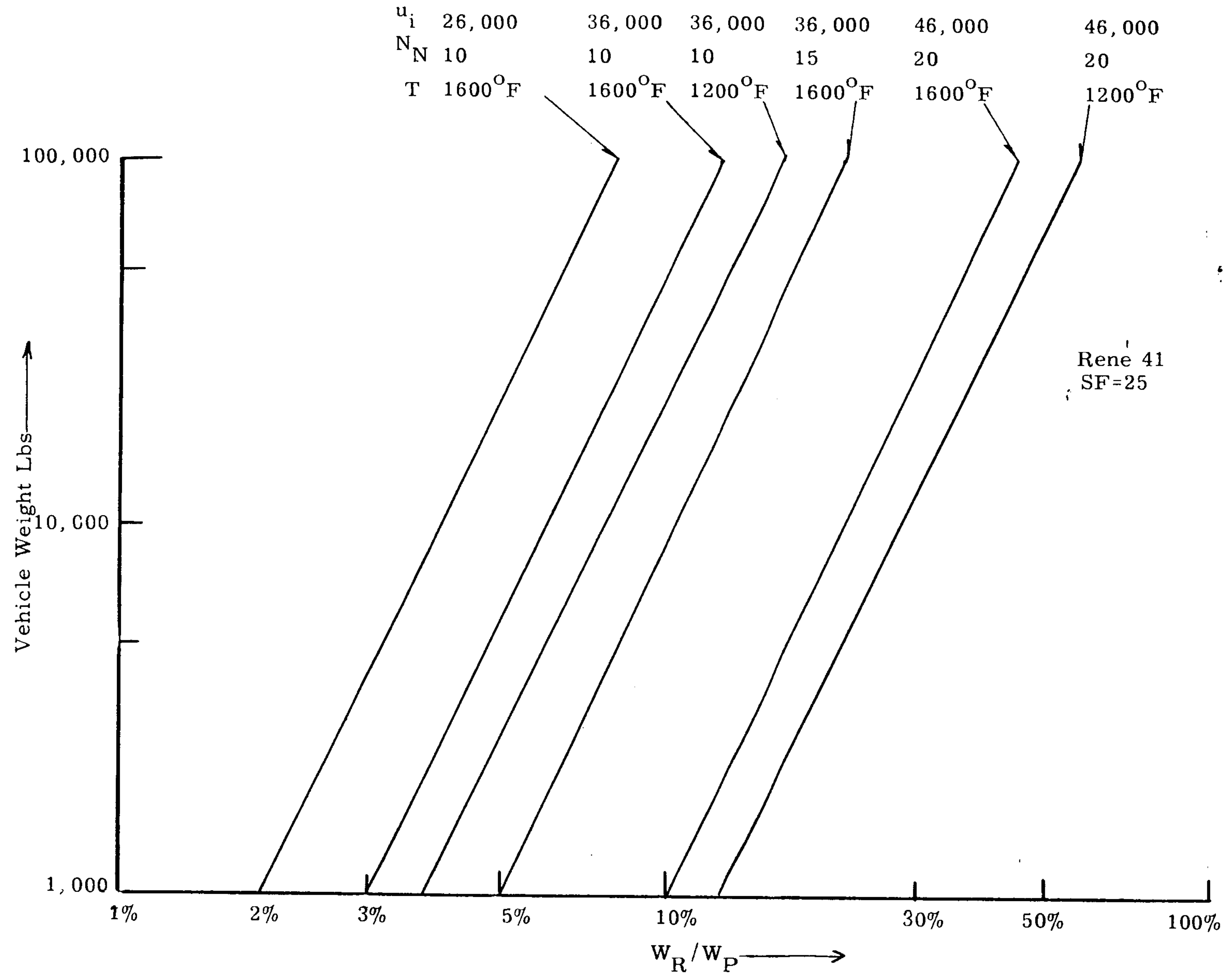


Figure 69. Rotornet Weight Fraction for Ballistic Entry (Rene 41 With Safety Factor of 2.5)

[redacted]

50,000 ft/sec which is typical of an emergency return from Mars. The rotornet principle and analysis was developed in Reference 43, NASA-CR-247.

The possibility of using low-wing-loading decelerators for entry from orbit into planetary atmospheres has been of continuing interest since the beginning of serious work in reentry mechanics. The heating rate for a given size vehicle decreases with decreases in W/C_DA; thus, if W/C_DA can be made compatible with the ability of structural materials to reject heat by radiation, practically no limit on reentry velocities and angles would exist. A cursory examination of the problem has shown that, for typical payload masses, the decelerator surface area is no greater than that required in the way of a parachute for terminal descent. Application to the reentry of large heat sources is felt to be a simple design problem with high probability of success.

The principal advantage to be gained from the low-wing-loading approach is that for a wide range of mission parameters it offers a potentially large decrease in the decelerator mass fraction (decel. mass/payload mass) over use of systems with conventionally high W/C_DA. Attendant to this feature, however, are a large number of important secondary advantages, such as operation at low temperatures, improved communication throughout the descent and potential for gliding and maneuvering without large increase in structural weight.

Figure 66, taken from Reference 43, shows two typical packaging concepts which are both applicable to planetary and earth reentry vehicles capable of nuclear heat source returns.

Figure 67 presents the maximum temperature versus load density curves for typical orbital, cislunar and planetary returns; and strength versus temperature for various structural fibers of interest.

Figures 68 and 69, from the same reference, shows the radius and rotornet weights for silicon fiber and Rene' 41 rotornet weaves with safety factors of 4 and 2.5, respectively.

d. Space and Ground Recovery

(1) Space Recovery. Controlled intact reentry has been evaluated and is applicable to recovery of the heat source. However, other techniques of recovery of the heat source in space should be investigated. Recovery can be accomplished in two ways:

(a) Reentering the heat source as a unit from an orbiting vehicle.

[redacted]

[redacted]

(b) Using a scavenging vehicle to pick up the heat source from the space vehicle and return it to earth.

The implications of the first technique have been discussed under controlled intact reentry. The second technique, however, has some interesting implications on vehicle design if an attempt were made to put it into practice.

A manned or unmanned scavange vehicle with the ability to pick up and return nuclear power supplies to earth must have the following characteristics:

(a) The ability to intercept and rendezvous with orbiting vehicles.

(b) The ability to capture the entire vehicle and adjust properly so that the nuclear heat source can be removed.

(c) The ability to store various sized heat sources and return them to earth.

The intercept and rendezvous capability implies a vehicle which has a large propellant capacity especially if more than one pick up is to be made. In addition, the vehicle may have to be manned to assure the reliability necessary to rendezvous with and disassemble a nuclear heat source from its mother vehicle in space. The task is quite difficult and to do it remotely, even with TV monitors, etc., is very questionable at this time. The closest practical system which is being planned and has at least some of the capability required is the satellite inspector. However, the system required for heat source removal is at least two orders more complicated than the satellite inspector, because it has to close exactly with the target, stop the target motion and control its relative alignment and remove the heat source.

One of the most difficult problems to overcome is that if the heat source to be captured is on board a spent satellite, that is, a satellite whose attitude control system has been depleted due to long space operation or an abort, then the capture vehicle must be able to stabilize the satellite before it can close with it and rendezvous. This, of course, is of the greatest difficulty and may be impossible without astronauts actually leaving the capture vehicle and physically altering the dynamics of the satellite with such things as gravity gradient collars, etc. All this being done in the presence of a nuclear source which may only have a shadow shield is very dangerous.

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[redacted]

Based on this short discussion it can be seen that scavenging nuclear sources from manned vehicles where good control can be maintained is reasonable, and depends only on development of present rendezvous techniques. However, trying to capture unmanned sources is not state of the art and significant technology has to be developed to make it possible.

Even if such a system were developed, it would undoubtedly be more complex than adding a controlled intact reentry system directly to the device. Accordingly, any practical application of scavenging would have to be a system capable of picking up a number of systems on a single mission to show any advantage over controlled intact reentry. Furthermore manned systems for this purpose show the highest level of success.

Table XIV shows the combination of systems which can presently be envisioned as feasible for space scavenging.

Table XIV

Feasibility of Space Scavenging

	Capture Vehicle	
Satellite	Manned	Unmanned
Manned	✓	✓
Unmanned	✓ (depending on satellite abort mode)	X (most difficult)

(2) Cost of Recovery Systems. Manned systems have the inherent capability of recovery and only modifications rather than complete redesign are required to produce a reentry vehicle which can bring back both men and heat source. The modifications to the basic command module as shown in Section IV B-3-c would be relatively minor and the R and D costs would be the major consideration. Very rough estimates indicate $\$1 - \$2 \times 10^6$ for the complete design and systems modification of an existing reentry vehicle thru the hardware stage.

However, unmanned systems have to start from scratch and these costs depend on what type of operations the reentry and scavenge vehicles are called on to perform. Definition of this point is important because it has a direct input into the decision to recover and reprocess the heat source fuel as discussed in Reference 44. If costs of the recovery system are prohibitive, recovery may not be financially attractive. Of course, recovery from a safety standpoint is not affected by the reprocessing argument. Hence, intact reentry of large heat sources for safety reasons will automatically profit the reprocessing argument.

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[redacted]

Since the basic design of scavenge systems are a spin-off from the advanced programs now under consideration (satellite interceptor, etc.) much of the basic technology may be chargeable to NASA and DOD programs making the AEC's application less costly than starting from scratch.

(3) Ground Recovery. Different techniques have been presented and discussed for achieving controlled and semi-controlled intact reentry. The feasibility of this terminal phase activity is, naturally, totally dependent upon the ability to return the manned or unmanned reentry vehicle to a designated land area. The actual recovery of the nuclear fuel from the reentry vehicle, the handling operations (land and water) and, the subsequent transporting of the source back to a fuel reprocessing facility are not controlling considerations in evaluating the feasibility of this approach for post-mission disposal. The problems are similar to those associated with the handling and transporting of any large radioactive source. This does not necessarily imply that the problems are easy or should not be thoroughly examined. Since the source can be adequately shielded to prevent excessive radiation doses to personnel directly involved in handling operations, the major task then for a given mission becomes the definition of recovery procedures, the type of equipment required, shielding limitations, and pertinent data related to the establishment of backup procedures. Other parameters to be considered should include the perturbation induced by different geographic locations and the effects of varying fuel inventory upon the method of recovery.

In any event, detailed recovery procedures and techniques for implementing the procedures must be developed to support not only the primary post mission disposal but any recovery operation that is required as a result of an aborted mission. Recovery from aborted situations can impose additional complexities due to the fact that landing may occur in unplanned areas or in undesirable landing modes, such as source ejection from the manned vehicle and direct impact. Hence, the recovery operation must be designed to meet a wide variety of conditions, it can take many forms depending upon the type and quantity of radionuclide involved, the type power converter, the vehicle design and the mode and type of surface impacted. It can be seen from these considerations that recovery teams must be equipped to handle all sizes of wreakage and varying degrees of structural complexity as well as thermal, chemical and nuclear effects.

In formulating controlled recovery plans consideration must be given to all modes of conceivable impact situations under which recovery must be achieved. It is impossible to maintain a recovery force sufficient to initiate immediate action, however, it would appear that the most logical approach would be to maintain the level of recovery support consistent with the

[redacted]

probability of the abort situation occurring. Since both land and water impact is possible, the recovery task should establish a balance between them to achieve the most flexible recovery task force in support of both abort and regular post-mission recovery.

e. General Conclusions

(1) Controlled Intact Reentry and Recovery

(a) Controlled intact reentry is tied to large heat sources from both the hazards and economic standpoint. In addition, random intact reentry of large heat sources probably cannot be tolerated from a political standpoint because of international problems which could be raised by irresponsible irradiation of an alien population or the associated delivery of large quantities of fissionable material to a potential enemy.

(b) Because of the nature of the end product of controlled intact reentry, that is, delivery of at least the nuclear heat source to a specific spot on the surface of the earth, only systems which are active in nature can be thought of for successfully meeting these goals. Passive systems which only have the capability to withstand reentry and impact can only satisfy the random intact reentry criteria. The active system must have the capability to leave the space environment at a particular point in its orbit or trajectory on command of a ground or space station, dissipate its inherent kinetic energy without compromising the integrity of the heat source during reentry or impact and signal their earth impact position to search parties if recovery is required.

(c) The complexity of achieving this goal is primarily a function of whether the vehicle system using the nuclear power supply is manned or unmanned. The heat source could be brought back with the crew in the mission module reentry vehicle or resupply vehicles thereby using the normally available reentry system or it could be jettisoned by the crew at a specific point in the orbit or trajectory of the manned vehicle or reenter ballistically within its own heat protection system.

(d) If the heat source returns with a manned reentry body the basic problems which arise are concerned not with the reentry but with integrating the heat source into the command and service modules during all the operational phases anticipated.

[redacted]

(e) Location of large heat sources in the service module is suggested to take advantage of the inherent nuclear shielding of the crew by the mechanical systems and separation distance and the reductions of thermal load to the ecological system of the command module.

(f) The heat source can be brought into the command module via a thermally insulated boom before initiation of reentry. It can be stored in an insulated and shielded compartment during the reentry and cooled with residual cryogenics from the life support system. It can be dropped at the end of reentry ($\sim$100,000 feet) or carried on to impact with the reentry body if recovery is required. The implications of the technique require that the heat source be radiatively coupled to the converter.

(g) For smaller heat sources which can be incorporated into the command module the area in the vicinity of the heat shield offers maximum volume and shielding potential to the crew. The techniques of cooling during reentry are the same as in the remote heat source case. Implications of this technique require the converter to be integrated directly with the heat source and coolant lines to be explosively separated to free the command module from the service module upon reentry.

(h) For manned systems where the heat source is to be reentered without the benefit of a manned reentry vehicle the heat source must have its own reentry body which will protect the system from reentry heating and from meltdown by internal heat generation.

(i) The basic operational mode for the reentry body is separation from the manned system, spin stabilize in proper reentry attitude, fire retros at the point in trajectory where earth impact is assured within the landing range desired. This technique is applicable for both orbital and planetary return systems. However, space disposal is more attractive when planetary return is considered.

(j) The operational techniques for intact controlled reentry of the heat source for unmanned systems are the same as for manned systems except for the control of the events being strictly in the hands of the ground station. This, of course, affects the reliability of the operation since on board mechanical and electronic systems are the only bridge between activating the nuclear source reentry and the ground. Therefore, sufficient redundancy and shielding must be provided to assure a high probability of success.

[redacted]

(k) It was determined that for altitudes below 500 miles retro rocket weight is no larger than 10 percent of the reentry vehicle weight. However, the effects of mission on the retros is quite severe in terms of impact at a given surface point after the command to fire is obeyed by the vehicle. No more than a ± 10° deviation in attitude can be allowed assuming a ± 50 mile error is acceptable at impact.

(l) A gravity gradient stabilization system appears the most attractive for initial attitude control of unmanned satellites which will launch intact reentry vehicles. Some active control will still be necessary to control rotation in a plane normal to the earth's radius.

(m) The design of reentry vehicles which are told when to disengage from the spacecraft is far simpler than semi-controlled random reentry. Hence, the vehicle contains no flight control system and must only survive a ballistic reentry. Therefore, the main problems are expected to be thermal and mechanical and not involved with complex electronic control systems. On board electronics will consist of a command receiver, action circuitry and surface location beacons and transponder.

(n) An isotopic heat source designed for intact reentry must consider the unique heat transfer and heat storage problems associated with high internal heat generation rates during reentry. Specifically, the fuel capsules must be able to dump sufficient heat from their surfaces to limit the fuel-wall interface temperature to 1800°F or less in both normal and abnormal operating modes. During normal operation it is not difficult to select system operating parameters such that the temperature limit for compatibility is not exceeded. The problem of designing the heat source for intact reentry becomes difficult when consideration is given to the need to reject heat by some alternate mechanism in the case of failure of the primary heat rejection mode. The difficulty arises from the need for an alternate heat rejection path around a heat shield which is designed to reject the aerodynamic heat of reentry.

(o) Two techniques can accomplish this. The first is standard reentry body which dissipates heat through jettisoning insulation during a coolant abort and has enough heat capacity to absorb the nuclear heat during reentry. The second is a device called a rotornet which has the capability to reduce the reentry heating to any desired value thereby reducing the reentry heat protection weights considerably. However, no change in internal heat storage capacity weights is expected. It is this second technique, however, which holds promise of making intact controlled reentry from aborts at planetary return speeds feasible for nuclear systems.

[redacted]

f. General Conclusions

(1) Recovery

(a) Space Recovery. Scavenging vehicles for nuclear systems in space are beyond present technology, unless both systems are manned. Taking a nuclear heat source back to earth in a resupply vehicle is nearly state of the art and can be accomplished with minimal changes in the design of the reentry vehicle. However, if both systems were unmanned and the satellite randomly tumbling, securing the heat source to the rendezvousing vehicles would be almost impossible without men on board. Therefore, the most successful nuclear heat source scavenger is envisioned as a manned satellite interceptor type of vehicle.

Development of such a vehicle seems to be reasonable in the 1970-1980 time period. Special radiation protection for the crew will, of course, be necessary.

Costs for a program to design and modify existing manned reentry vehicles to return nuclear heat sources is approximately \$1-\$2 x 10^6. Reentry of the fuel will be based on the safety argument mainly, however, this will auutomatically benefit reprocessing.

(b) Ground Recovery. Analyses have shown that recovery of the nuclear heat source after controlled intact reentry within a predetermined impact area should pose no severe technical problems and can be carried out with present nuclear device handling techniques.

C. Orbital Storage and Escape

Post-mission disposal of a space nuclear system (radioisotope or reactor) can be achieved by abandonment of the nuclear fuel in orbit, transferring the source to a higher orbit for greater lifetime, or by propelling it into an earth escape trajectory. The primary advantages associated with these approaches is the elimination of large radioactive release in the troposphere or biosphere, or impacting nuclear material in populated areas.

1. Orbital Storage

The effectiveness of orbital storage is dependent upon the orbital lifetime and the radioisotope (half-life and type of emitted radiation) being stored. In the following discussion, the applicability of orbital storage will be assessed. As a basic guideline, a nominal decay of ten half-lives (this reduces the original inventory by a factor of 1024) will be used to establish the isotope fuels and to assess the usefulness of other orbits for disposal. These guidelines are considered reasonable since a backup safety philosophy

[redacted]

of burnup will undoubtedly limit the launch inventory so that any orbital storage time will reduce the hazard further. The decay half-life of useable power generating isotopes varies from a fraction of a year (Po-210 - 138.5 days) to many years (Pu-238 - 89.6 years), therefore, the orbital lifetime requirements range from approximately 3 to 900 years depending on the half-life of radioisotope being used.

The orbital lifetime of a vehicle or fuel core is extremely sensitive to its altitude, ballistic coefficient and type of orbit (eccentricity). The orbital lifetime as a function of these variables is presented in Figures 70 and 71. The required orbital altitude for storage to permit ten half-life decay of Po-210, Pm-147, Cm-244, Sr-90, and Pu-238 are shown for ballistic coefficients from 1 to 10,000 lb/ft^2. The ballistic coefficient of different type fuel blocks may vary from approximately 5 to 100 lb/ft^2 because of the dependence upon shielding and basic design considerations for different type energy converters. It may be possible to design the fuel containment structure with a ballistic coefficient in excess of 100 lb/ft^2. This would be desirable since as the ballistic coefficient is increased the required altitude for ten half-life decay is decreased, as shown in Figures 70 and 71. For example, an altitude of 333 nautical miles (n. mi.) is needed to assure at least ten half-life decay prior to earth reentry for Po-210 within a configuration having a ballistic coefficient of 10 lb/ft^2. If an elliptical orbit is the final orbit achieved during a transfer, its lifetime is significantly less than the circular orbit as illustrated in Figure 71. For example, a 300 n. mi. perigee and a 700 n. mi. apogee with a ballistic coefficient of 1 lb/ft^2 is characterized with an orbital lifetime of approximately two years, whereas for a circular orbit of 700 n. mi. the lifetime is approximately 400 years. However, for systems with higher ballistic coefficients in the range of 100 to 10,000 lb/ft^2 the orbital lifetime for a 400 n. mi. orbit ranges from 160 years to greater than 10,000 years. Hence, the use of orbital storage to minimize or eliminate potential nuclear safety problems through isotope decay in space is very attractive.

2. Orbital Transfer

There are several different techniques for achieving orbital transfer. The following techniques are discussed in this section: a Hohmann Transfer, a radial thrust transfer and a spiral-out trajectory via tangential thrusting.

By definition, the Hohmann Transfer is the least energy orbital transfer technique. The use of this approach naturally minimizes any weight or volume requirements associated with orbital transfer, as disposal technique. A schematic illustrating the transfer is presented in Figure 72. The nuclear source is injected from a circular or elliptical orbit into an elliptical trajectory which is characterized with an apogee which when circularized at the apogee will yield an orbit that possesses the necessary lifetime to permit fuel decay. If the radioactive source is not circularized, as a result of system malfunction, the source will remain in the elliptical orbit with a lifetime much less than the lifetime of the circular orbit. The change in

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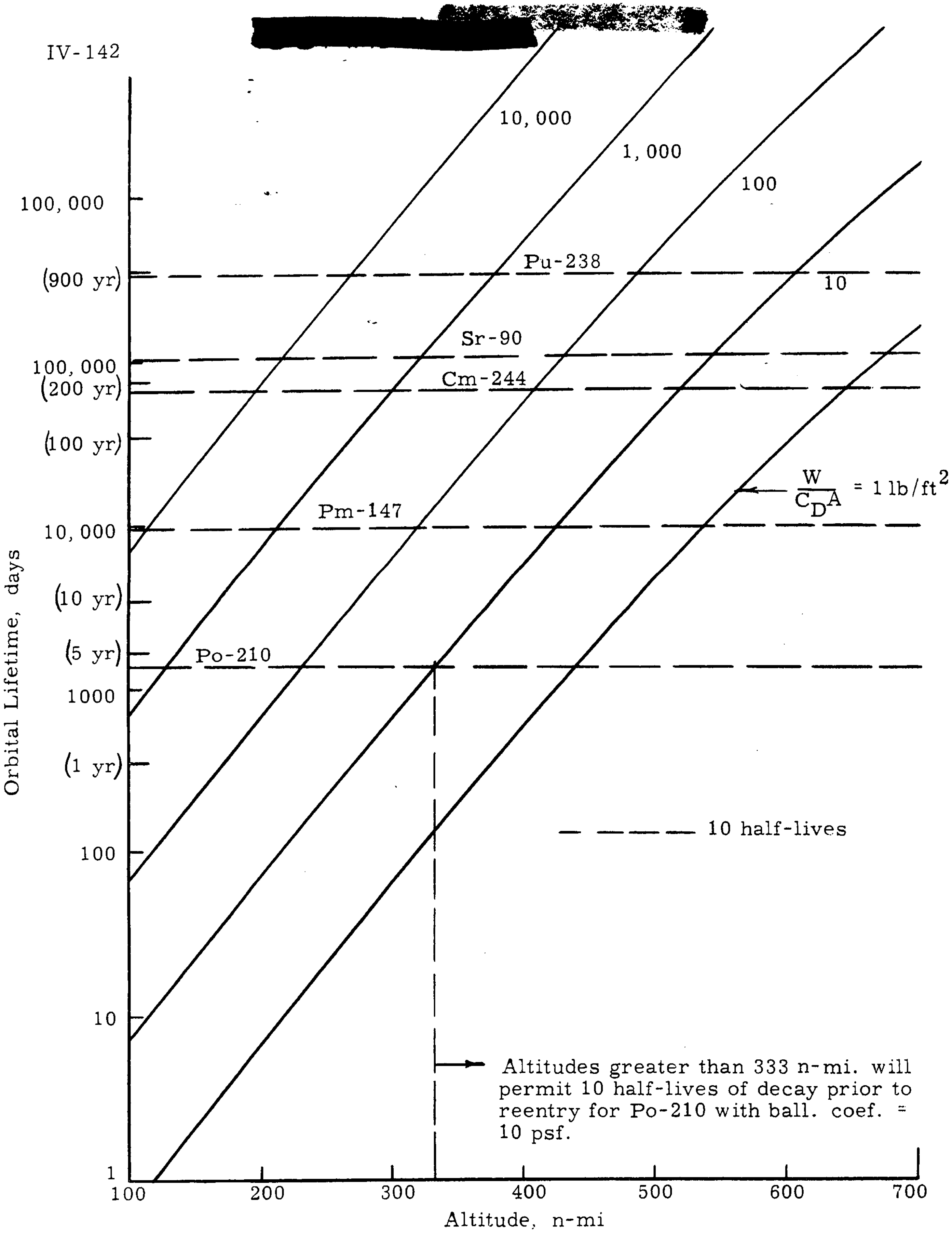

Altitudes greater than 333 n-mi. will permit 10 half-lives of decay prior to reentry for Po-210 with ball. coef. = 10 psf.

Figure 70. Orbital Lifetime for Circular Orbits

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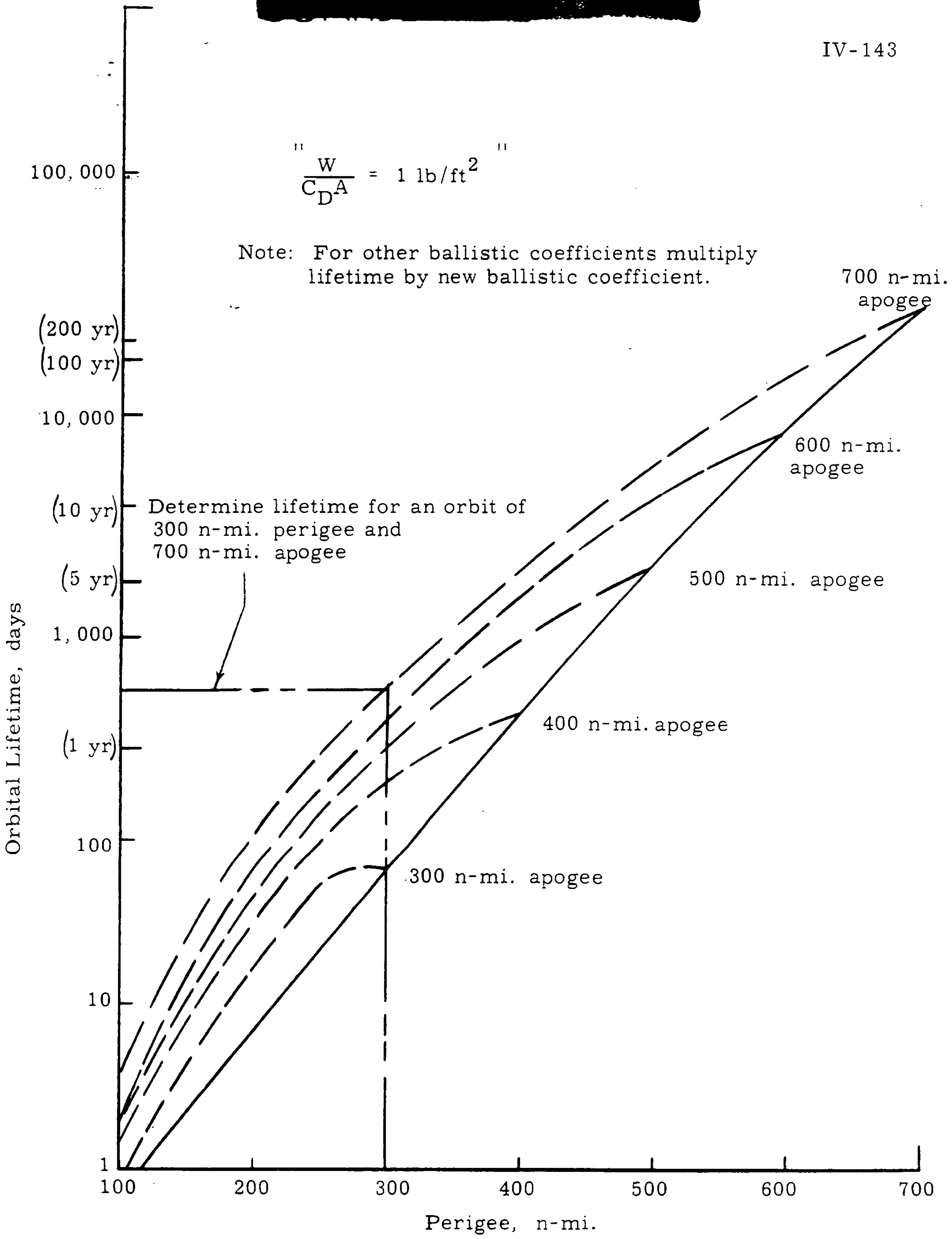

Figure 71. Orbital Lifetime for Elliptical Orbits

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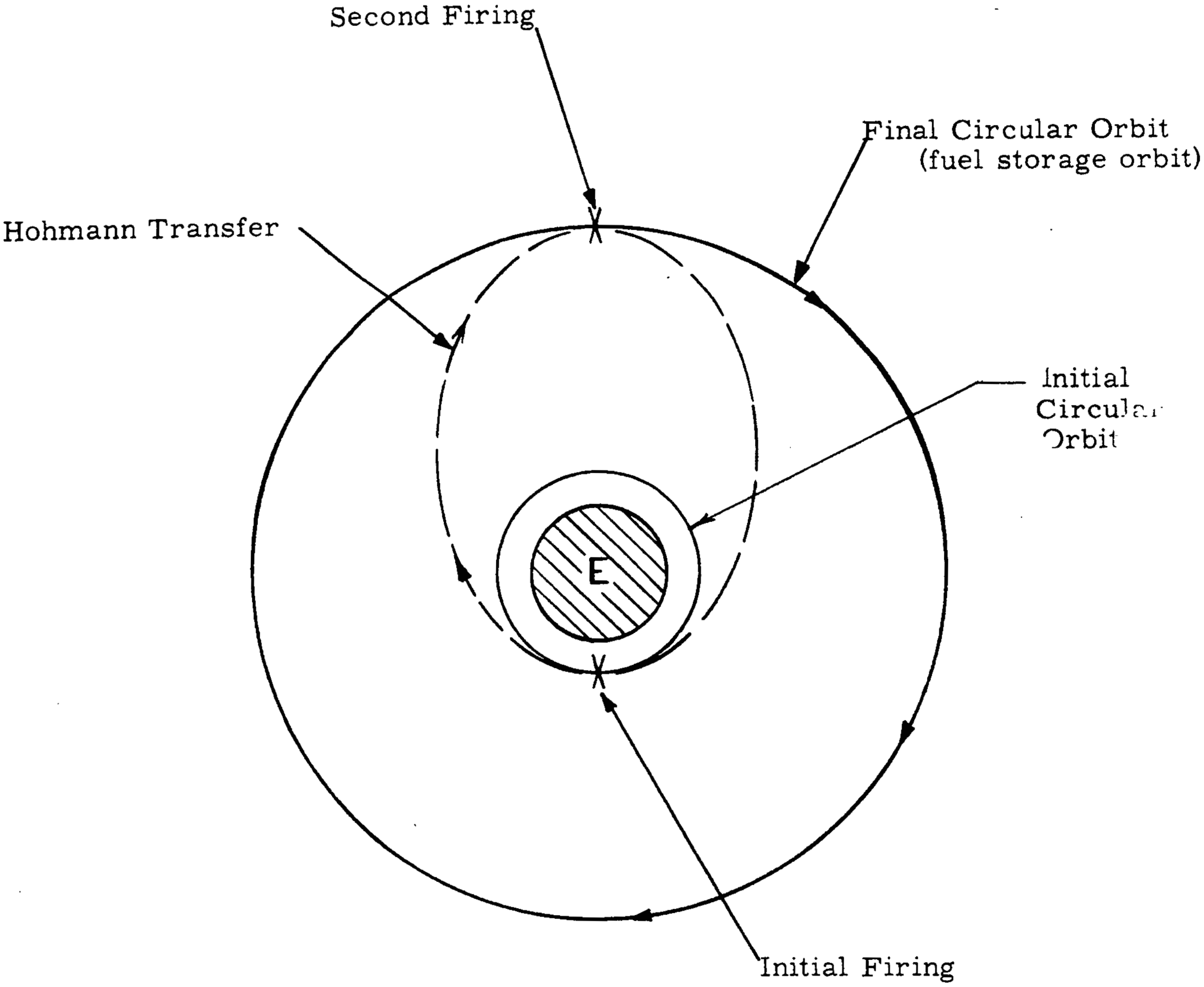

Figure 72. Hohmann Transfer Orbital Storage

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orbital lifetime may be determined for any system utilizing the data presented in Figures 70 and 71. By knowing the orbital lifetime, the quantity of radioactive material which will reenter the earth's atmosphere from the undesirable orbit can be determined.

The radial thrust transfer, illustrated in Figure 73, offers no significant advantages relative to the Hohmann transfer technique. It involves the same number of rocket firings and requires more propulsive power to accomplish than the same transfer via a Hohmann transfer approach. However, the positioning of the rocket for the first firing is probably less critical in the radial thrust technique than the positioning of the rocket for a pure tangetial firing from the initial orbit. Also, the required time for completing the radial transfer is much less than for the completion of a Hohmann transfer. These considerations become very important in increasing the reliability and usefulness in terms of a lighter system.

The spiral-out trajectory, a condition which yields a selenocentric (earth-moon orbit) or heliocentric (earth-sun orbit) orbit can be implemented by either a continuous or periodic thrustor. This approach to disposal again requires more propulsion power than a Hohmann transfer, but does not involve a major rocket firing to circularize the orbit after the apogee of the new orbit has been achieved. A schematic of the technique is shown in Figure 73. The elimination of the second firing and the associated guidance and control equipment is the primary advantage of this approach. However, the lack of available long life (continuous or periodic) thrustors, limits the usefulness of this technique. The feasibility of orbital maintenance is also remote for unmanned systems. Small electrical propulsion devices have been developed, but their applicability to orbital maintenance and a spiral-out trajectory have not been demonstrated. Potentially, this type of thrustor may be very attractive for both applications, but system duration and reliability are severely limiting factors at the present state of development. Present systems are heavy and require high power ($\sim$ 1/2 Kw(e)) for thrusts of 0.01 - 0.1 pounds.

It is evident that a Hohmann transfer offers the optimum mode of orbital storage for decreasing the fuel inventory prior to reentry. Due to the characteristics of this method, however, a significant amount of guidance and control are needed to assure correct orientation of the rocket while firing. For this reason, a gravity gradient system (GGS) was considered as a simple, reliable and tested (used on satellites now in space) means for achieving the degree of control required.

The system consists of a long bar attached to the system with a small weight at the other end, as shown in Figure 74. It orients the structure such that the bottom is always aimed toward the center of the earth, hence, the GGS provides two degrees of control (pitch and roll). However, yaw control must be provided through some other guidance system to prevent misalignment of the rocket during the first and second burns. To decrease the effect of drag on the system, the gravity gradient bar and the small end weight may be ejected after the second firing to increase the orbital lifetime of the structure containing the radioactive material.

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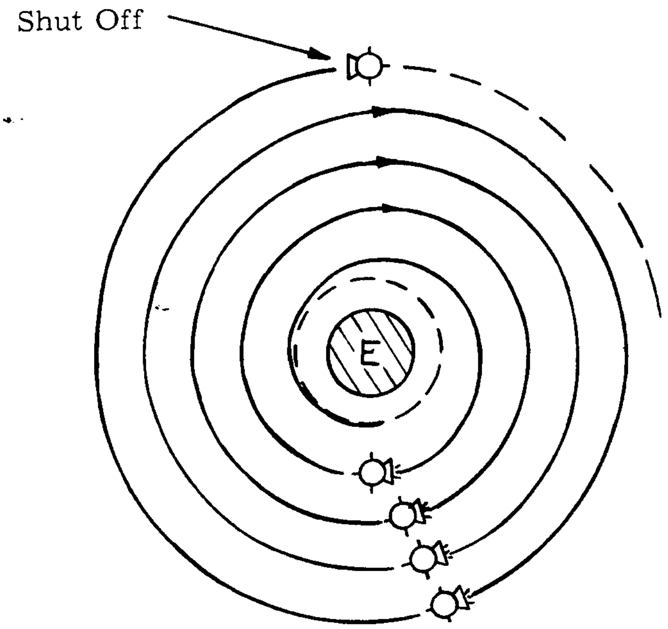
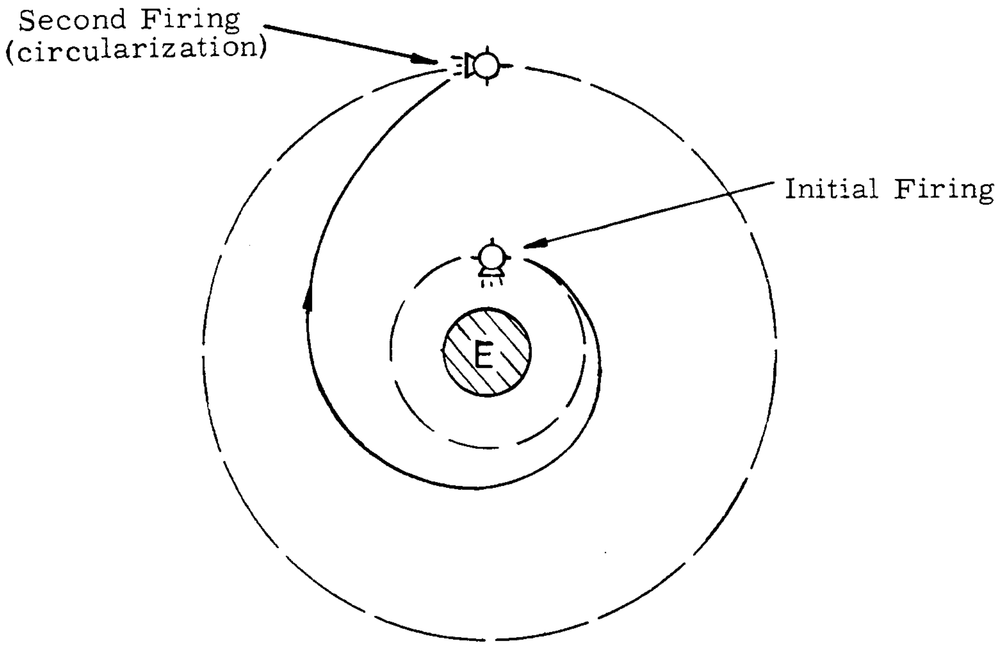


Spiral



Radial Transfer

Figure 73. Orbital Transfer Methods: Spiral and Radial

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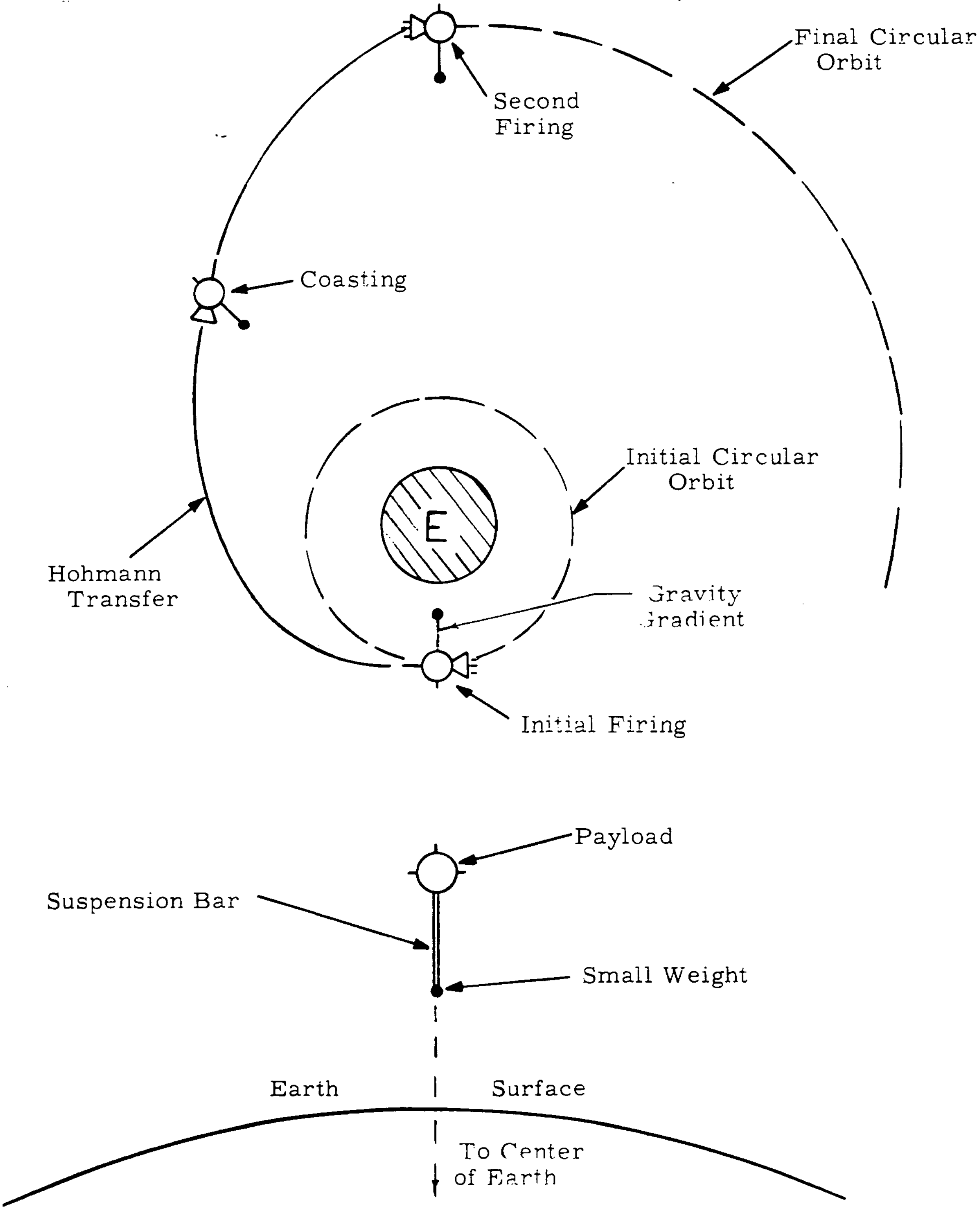

Figure 74. Gravity Gradient System for Altitude Control During Orbital Transfer

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A comparison of the propulsive and guidance requirements for the different orbital transfer techniques and their relative feasibility is presented in Table XV.

Table XV

Evaluation of Orbital Transfer Techniques

Modes of Orbital Transfer	Power Requirements	Guidance Requirements	Feasibility
Hohmann Transfer	Least	Medium*	Good
Radial Transfer	Medium	Low-Medium*	Fair
Spiral Transfer	Medium	Medium*	Good
Maintaining Original Orbit			
1. Continuous Thrust	Low	Low*	Poor
2. Periodic Thrust	High	Medium*	Fair
Earth Escape	High	Low-Medium*	Poor

* Using gravity gradient system

3. Required Propulsion Systems

The three types of propulsion systems that will be considered for orbital storage are chemical, nuclear, and electrical systems. The transfer from one orbit to another requires a specific velocity increase that is dependent upon the mode of transfer and the altitude change. The velocity increase is independent of the propulsion system, however, the weight requirements to achieve a given velocity increment varies with the type of propulsion system. The velocity increments for transfer between orbits is illustrated in Figure 75. Both the circular velocity (V_C) and the apogee velocity (V_A) as a function of altitude are presented. ΔV_{II} and ΔV_{III} values correspond to the velocity increments required to leave the original orbit and circularize at the desired altitude,respectively. From these velocity requirements, the fuel and propulsion system weights necessary to complete a given orbital transfer may be determined.

Total fuel requirements for a particular type propulsion system depends upon the specific impulse of the system, as shown by Equation 1, the higher the specific impulse the lower the fuel inventory. The fuel to payload weight ratio for a fixed velocity increment is obtained from the following equation:

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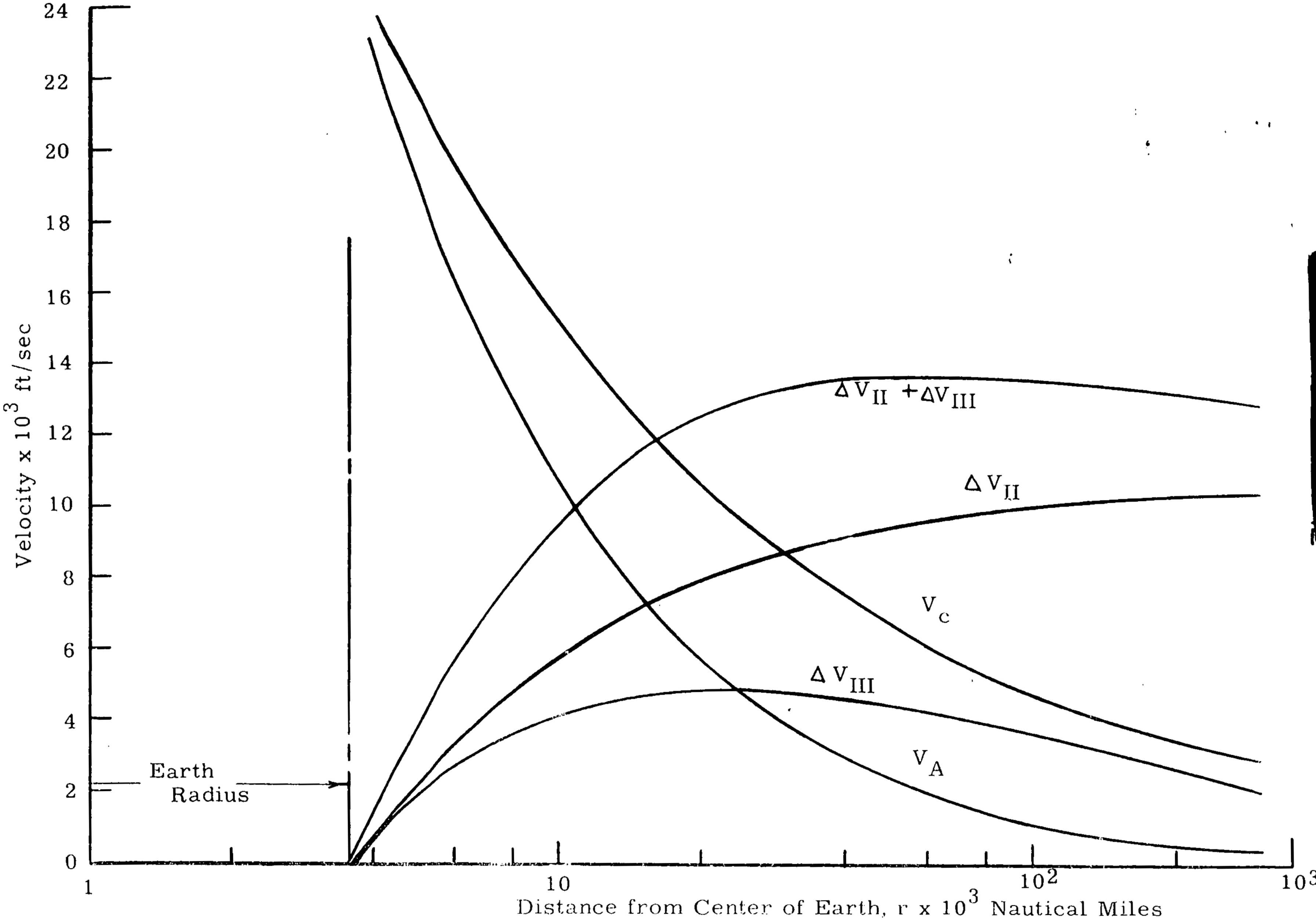

Figure 75. Variation of Circular Velocity, Apogee Velocity and the Increments ΔV_{II} and ΔV_{III} as a Function of Satellite Orbital Distance.

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$$\frac{W_{fuel} + W_{payload}}{W_{payload}} = \exp\left(\frac{\Delta V}{g\, I_{sp}}\right) \qquad (1)$$

where: V = velocity increments
I_{sp} = specific impulse
g = gravity

The specific impulse of the different type of propulsion systems considered are:

Propulsion System	Specific Impulse (Sec.)
Chemical	200 - 450
Nuclear Isotope	700 - 900
Reactor	700 - 1,100
Electrical	800 - 10,000

Since the fuel weight requirements decrease with increasing specific impulse, it is apparent that the electrical system uses the least amount of fuel. The values of specific impulse do not represent the complete system since the total propulsion system weight is not included. For example, the electrical system requires a heavy power generating system to provide the large amount of electrical power. The nuclear system requires containment vessels, controls, coolant and shielding which increase its total weight appreciably. Whereas, the chemical system has only minor weight penalties associated with tankage, approximately 10 percent of the fuel weight (Reference 50).

The ratio of propulsion system weight to payload weight for the three different systems considering a Hohmann Transfer from an 80 n. mi. original orbit to altitudes up to 700 n. mi. is presented in Figure 76. A comparison of the performance of each propulsion system for this type transfer may be made.

The decrease in propellant weight required for isotopic, reactor and electrical systems relative to the chemical system is clearly shown in the figure. As previously mentioned, however, the total propellant system weight is much greater than the fuel weight for the three higher specific impulse systems.

For the calculation, the following typical values of 300, 800, and 5000 seconds for specific impulses were used for the chemical, nuclear and electrical systems, respectively.

The method of spiraling out to a new orbit lends itself to the low thrust (continuous or periodic) electrical propulsion system. This system may offer considerable advantages in the future; however, the present electrical propulsion systems require so much electrical power that they are heavier than the conventional systems (Reference 46).

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Hence, the chemical propulsion system is the most appropriate method. The components of the system are the most tested of all the systems analyzed and therefore the most reliable to date. The weight of the system is not unreasonable. It can be seen from Figure 76 that the weight ratio to go to a 400 n. mi. orbit from an 80 n. mi. orbit is 0.19. This means that for a 100 pound payload the weight of the propellant would be about 19 pounds.

4. Earth Escape

The last method for achieving post-mission disposal to be analyzed will be earth escape. The modes of escape correspond to those mentioned for orbital transfer within the earth's gravitational field. And as before, spiralling out and radial thrust require more energy than a Hohmann Transfer. Figure 77 shows the propellant system weights required for various escape missions as a function of propulsion system. The method of escape eliminates any problem connected with earth reentry, but the energy requirements are too severe to make this technique more attractive than station keeping or slight change in orbit.

An evaluation of the possible abort modes in an orbital transfer for a specific mission has been completed. The MORL mission was selected as a typical earth orbital mission which could utilize orbital storage for safe disposal of on board radioactive fuel. The original circular orbit was considered to be 200 n. mi. and the ballistic coefficient of the container was assumed to be 10 lb/ft^2. A low value was used in order to determine the shortest orbital lifetime because the design will probably be characterized with a higher ballistic coefficient which will increase the orbital lifetime or decrease the required storage orbital altitude. The isotope considered for this analysis was Pu-238; other radioisotopes such as Po-210 and Pm-147 could be used at much lower altitudes due to their shorter half-life. The analysis considered the total sequence of events from mission orbit departure through the establishment of the desired circular storage orbit. The transfer sequences, possible abort modes and the effect upon orbital storage are represented in Table XVI and Figure 78.

If both stages of the rocket fire properly, the fuel block will be circularized at 600 n. mi. above the earth. This would allow 10 half-lives of Pu-238 decay before reentry. It should be noted that the worst case is a firing of a stage in the opposite direction; if the first stage fires 180° out of phase, the vehicle will impact the earth on its first orbit. From an altitude of 200 n. mi., a "Δ V" decrease of 400 ft/sec will cause impact within the next orbit (Reference 53). The "Δ V" necessary to increase to a 600 n. mi. orbit is approximately 500 ft/sec (see Figure 75). Hence, if this rocket fired as a retro, it would cause reentry and earth impact within the first orbit.

If the first stage fired properly but the second stage fired in the opposite direction, the resulting orbit would be elliptical with a low perigee. The low perigee and highly elliptical orbit would cause the fuel block to reenter before the inventory could be significantly reduced through decay.

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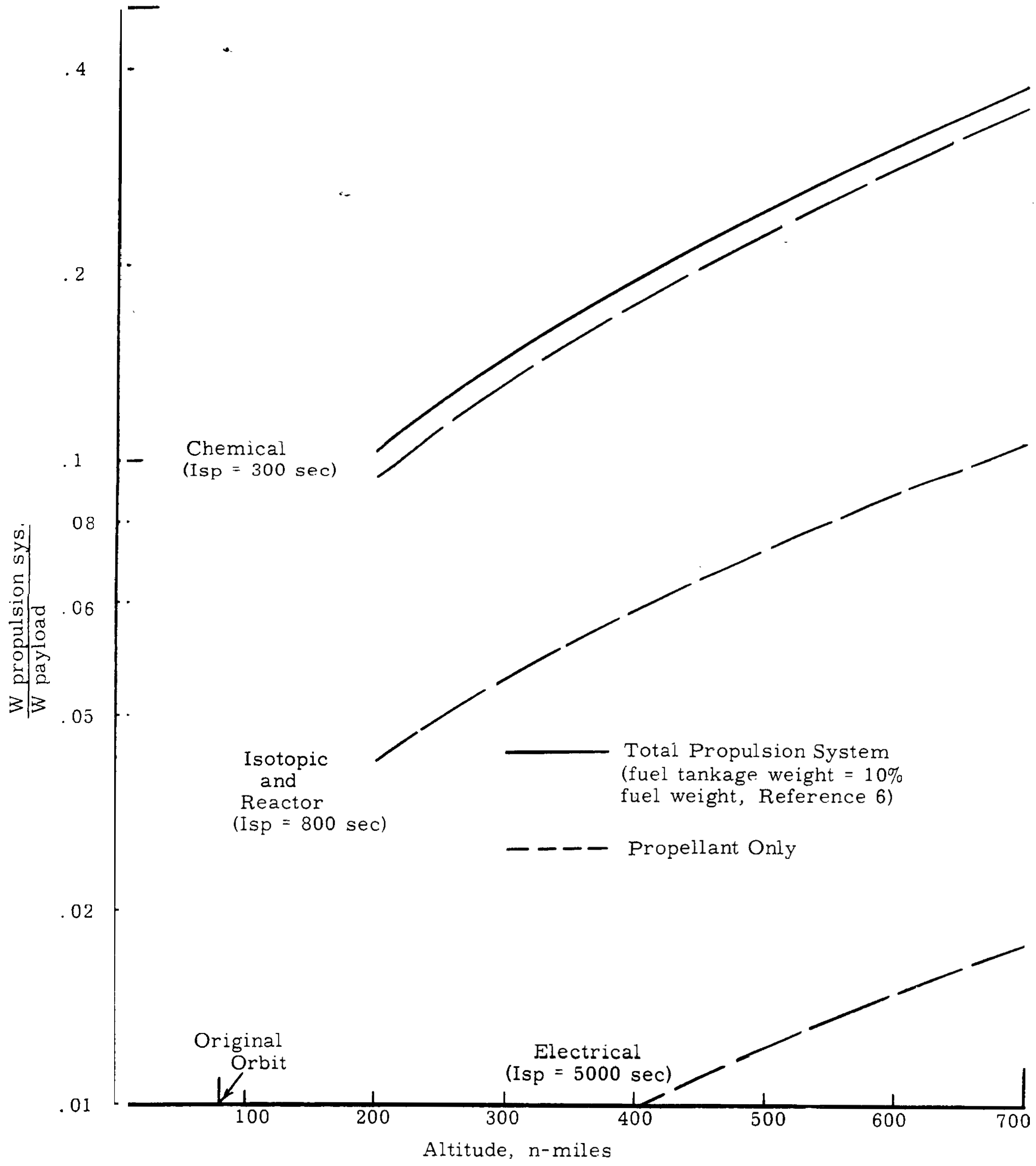

Figure 76. Ratio of Propulsion System Weights to Payload Weight for Hohmann Transfer from 80 n-mile Original Orbit.

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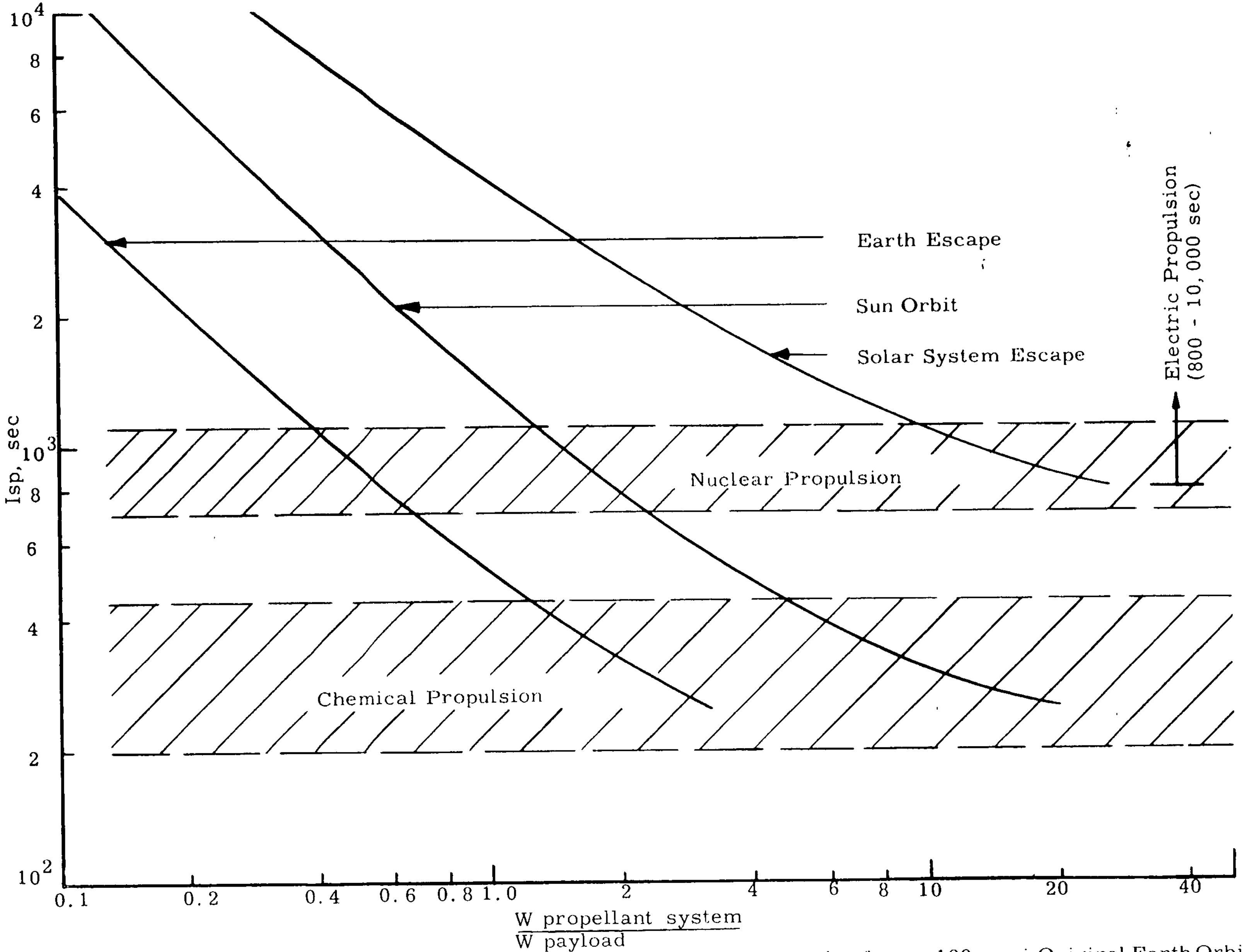

Figure 77. Ratio of Propellant System Weight to Payload Weight Versus Specific Impulse from a 100 n-mi Original Earth Orbit.

Table XVI

Evaluation of Orbital Transfer Missions for MORL

(Isotope: Pu-238 at 200 n-miles original orbit)

(Ballistic Coefficient = 10 psf)

Case *	Final Orbit (n. mi.) Perigee	Final Orbit (n. mi.) Apogee	Orbital Lifetime
1. First stage fires properly, Second stage fires properly	600	600	900 years
2. First stage fires properly, Second stage fires partially	200-600	600	2 years - 900 years
3. First stage fires properly, Second stage fails to fire	200	600	2 years
4. First stage fires properly, Second stage fires in opposite direction	50	600	1 hour
5. First stage fires properly, second stage fires partially in opposite direction or other than 180° angle	50-200	600	2 years
6. First stage fires partially, second stage fails to fire	200	200-600	70 days - 2 years
7. First stage fails to fire, Second stage fails to fire	200	200	70 days
8. First stage fires in opposite direction	0	200	30 minutes

* The abort conditions are shown on Figure 78.

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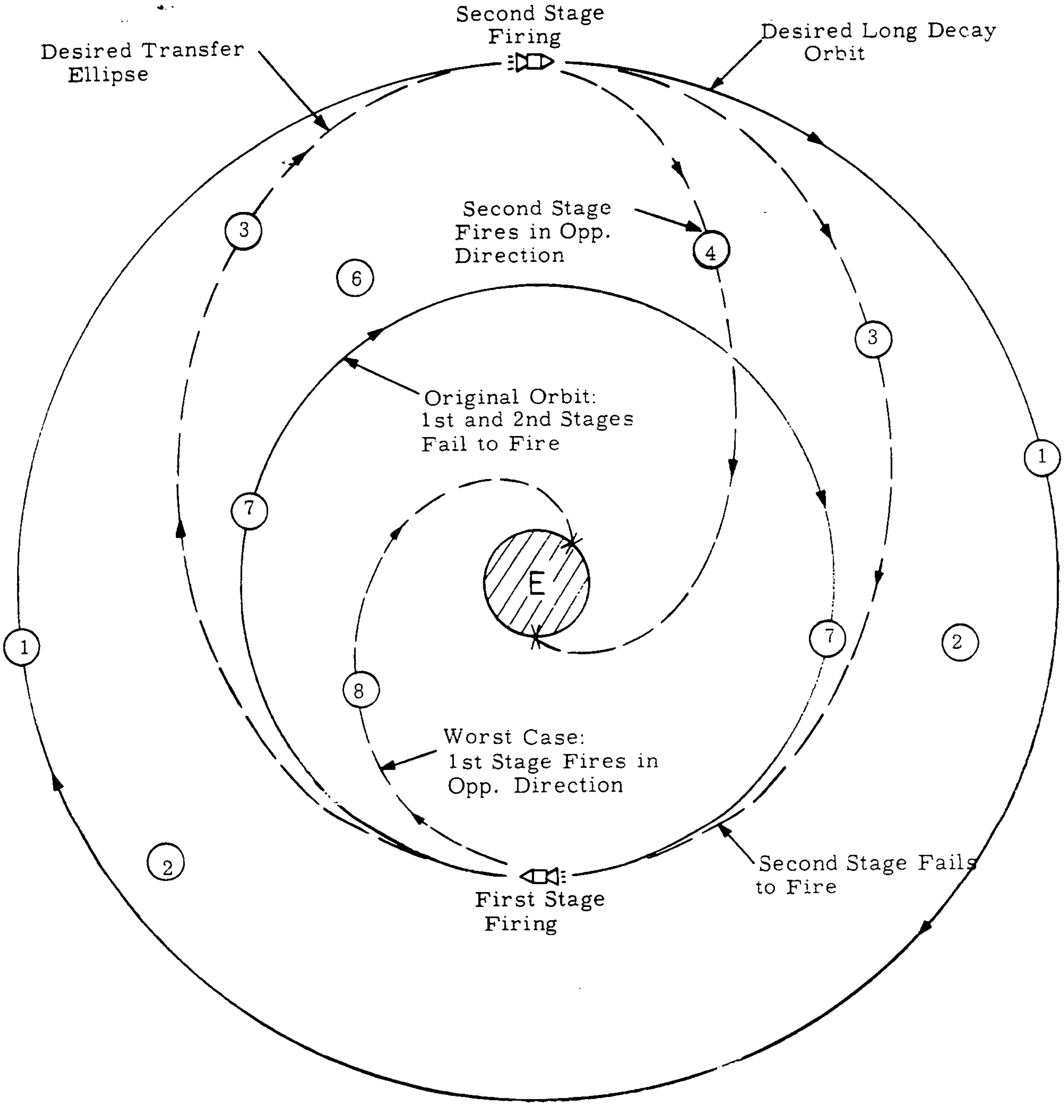
Second Stage Firing
Desired Transfer Ellipse
Desired Long Decay Orbit
Second Stage Fires in Opp. Direction
Original Orbit: 1st and 2nd Stages Fail to Fire
Worst Case: 1st Stage Fires in Opp. Direction
Second Stage Fails to Fire
First Stage Firing
E
1
2
3
4
6
7
8

Figure 78. Typical Orbital Transfer Sequences and Possible Abort Modes.

[redacted]

[redacted]

If the first stage fired properly but the second stage only partially burned or deviated in angle, the fuel block would be placed into an elliptical orbit between the desired circular orbit and the resulting ellipse if the second stage had failed to fire. Also, the partial firing of the first stage would result in an ellipse between the original circular orbit and desired transfer ellipse and being characterized with an orbital lifetime close to the original circular orbit.

It may be concluded that malfunction of either stage rocket would significantly reduce the orbital lifetime and effectiveness of this approach for post-mission disposal.

5. Mission Applicability

Of the planned space missions identified in Table II several are applicable to orbital storage. The applicability is controlled by the nature of the orbit (low geocentric orbit) and the quantity of radioactive fuel. In general, if the orbit is greater than 400 to 600 n. mi., depending on the type of isotope involved, orbital storage is not required because higher orbits will permit sufficient orbital lifetime for 10 half-life decay of the fuel. Also, interplanetary missions returning to earth may inject the fuel source into a heliocentric or geocentric orbit for subsequent decay. However, these missions do not need propulsion to achieve orbital storage altitudes.

To determine the requirements of orbital storage for particular missions, a Hohmann Transfer, with a chemical system, was considered in all cases and an abort into an 80 n. mi. orbit was assumed. To provide a consistent basis for the generation of data corresponding to the requirements of the different missions, typical electrical power to weight conversion factors were assumed for both thermoelectric $\sim$0.5 lb/watt(e) and dynamic systems $\sim$0.216/watt(e).

Missions that were considered applicable to orbital storage based on these considerations are listed in Table XVII.

In summary, orbital storage would appear to be an effective method in minimizing or eliminating potential nuclear safety hazards associated with future aerospace missions. It is, however, limited to larger systems where weight penalties induced by the propellant, tankage, guidance and other pertinent considerations do not appreciably affect system characteristics, such as specific power (watts/lb), size or spacecraft-power system integration considerations. For missions characterized with orbital altitudes greater than 400 to 600 n. mi. (circular orbits), depending upon the radioisotope, the system can be abandoned because the orbital lifetime will be sufficient to eliminate any reentry hazard via an orbit decay.

For missions below 400 n. mi. and above 100 n. mi. Po-210 because of its short half-life may be acceptable if the systems ballistic coefficient is between 1 and 1,000, respectively.

[redacted]

Table XVII

Missions Applicable to Orbital Storage With Associated Altitude and Weight Requirements

Mission No.	Mission	Power Level Kw(e)	Conversion Technique	Power Generating System Weight (lbs)	Isotope	Altitude Nec. for 10 Half-life Decay (2) (ni. mi.)	Orbital Storage System Weight Propellant & Tankage (lbs)
1	Extended Apollo (manned)	6.0	Dynamic	1200	Po-210	330	180
			Dynamic	1200	Pu-238	600	360
			T/E	3000	Po-210	330	450
			T/E	3000	Pu-238	600	900
2	MORL (manned)	8.0	Dynamic	1600	Po-210	330	240
			Dynamic	1600	Pu-238	600	480
			T/E	4000	Po-210	330	600
			T/E	4000	Pu-238	600	1200
3	Space Station (manned)	10.0	Dynamic	2000	Po-210	330	300
			Dynamic	2000	Pu-238	600	600
			T/E	5000	Po-210	330	750
			T/E	5000	Pu-238	600	1500
17	Nimbus	0.25	T/E	125	Pu-238	600	38
19	Adv. Orb. Solar Obs.	0.50	T/E	250	Pm-147	430	50
				250	Pu-238	600	75
21	Operational Weather Satellite	0.50	T/E	250	Sr-90	550	
				250	Pu-238	600	75
23	OAO	0.75	T/E	375	Pu-238	600	110
25	OGO	0.50	T/E	250	Pm-147	430	50
26	SGM	0.20	T/E	100	Po-210	330	15
27	Satellite Inspector	1.0	T/E	500	Pu-238	600	150
28	AOAO	1.0	Dynamic	200	Pu-238	600	60

(1) Ballistic Coefficient = 10 lb/ft^2. (2) From original orbit = 80 n. mi. (therefore, if fuel block is above 80 n. mi. originally, orbital storage system weight will be less.)

[redacted]

Escape from earth, in all cases considered, required an amount of fuel in excess of 3 - 4 times the payload weight. Attainment of a solar orbit requires 30 times the payload weight. Therefore, a prohibitive penalty in weight for this technique seems evident.

6. General Conclusions

(a) The use of orbital storage to minimize or eliminate potential nuclear safety problems due to reentry depends upon the success of the original launch, the choice of orbit, maintaining attitude control of the satellite, and choice of a propulsion system.

(b) Three techniques are open to the designer all of which require propulsion to:

(1) Maintain original orbit

(2) Transfer to higher longer lived orbit.

(3) Escape.

(c) In all cases of increasing orbital lifetime through the use of a thruster attitude control is critical.

(d) Small chemical propulsion systems are light in weight and best in reliability.

(e) The best transfer technique is Hohmann with a single impulse.

(f) Orbital storage appears to be an effective method in minimizing or eliminating potential nuclear safety hazards associated with future aerospace missions in eleven out of fifty-two missions analyzed. It is, however, limited to larger systems where weight penalties induced by the propellant, tankage, guidance and other pertinent considerations do not appreciably affect system characteristics, such as specific power (watts/lb), size or spacecraft-power system integration considerations. For missions characterized with orbital altitudes greater than 400 to 600 n. mi. (circular orbits), depending upon the radioisotope, the system can be abandoned because the orbital lifetime will be sufficient to eliminate any reentry hazard via an orbit decay.

(g) For missions below 400 n. mi. and above 100 n. mi. Po-210, because of its short half-life, may be acceptable if the systems ballistic coefficient is between 1 and 1,000, respectively.

(h) Escape from earth in all cases considered required an amount of fuel in excess of 3 - 4 times the payload weight. Attaninment of a solar orbit requires 30 times the payload weight. Therefore, a prohibitive penalty in weight for this technique seems evident for any system considered.

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D. The Potential of Nuclear Systems for Destruction in Orbit

An alternate safety philosophy often quoted by nuclear power systems designers, which may have the potential to alleviate or certainly reduce the eventual reentry hazards associated with the use of these systems, is destruction of the systems in space.

Two questions immediately arise when this philosophy is considered; first, what is the acceptable level of destruction required to satisfy safety consideration ? ; and second what techniques can be realistically employed to achieve this level of destruction?

1. Acceptable Levels of Destruction

To define the character and extent of the destruction necessary to significantly lower the eventual reentry hazards from nuclear system space debris one must first postulate the possible destruction modes and these consequences.

The destruction modes possible in space could range from breaking up any system into discrete fragments to complete vaporization. In some cases chemical dissolution of the fuel capsule may be a possible and practical technique.

If one considers the advantages to space destruction from a safety viewpoint, several basic facts stand out above all others, namely, the eventual objectives are to:

(1) Increase the orbital lifetime of the nuclear device and hence the time available for decay before reentry.

(2) Increase the exposed surface area of the nuclear debris to maximize loss of fission products or radioisotope fuel to space via evaporation before eventual reentry.

(3) Increase the final dispersal pattern upon the earth's surface via initial space diffusion.

In the limit, of course, complete vaporization is most desirable. However, as will be discussed later in this section, it is not likely to be achieved by utilizing the internal energy of the nuclear source.

If one consideres fracture of the device into smaller elements several points of interest appear.

The only way this can be achieved is through explosive or chemical means. Therefore, it can be expected that the ΔV induced to the resulting debris will both degrade and accelerate some of the particles. However, in all cases the orbital lifetime of any given device tends to be reduced by fracturing due to the decrease in effective ballistic coefficient. This occurs because the loss of mass predominates over loss of cross sectional area.

[redacted]

[redacted]

Therefore, a tradeoff between the decrease in orbital lifetime versus the increase in surface area exposed to space must be made on the basis of the evaporation potential of each fuel form of interest before a judgment can be made on the degree to which a given device can be fractured to enhance aerospace safety.

Another point of interest must be made relative to the fact that since evaporation mass loss is a function of both temperature and surface area, once fracture of device is induced the average temperature of the debris is less than the equilibrium temperature of the whole, because of the increase in effective reradiation area per unit of power generating mass, thereby further reducing the expected effects of evaporation.

The relative effects of the latter points can be approximated as shown in Figure 79 based on the analyses of Appendix F-2. Figure 79 shows that fracture of a given fuel block does indeed increase the effective exposed surface area much faster than the equilibrium temperature drops. However, the real mass loss is a function of the vapor pressure of the material and this must be factored into the analysis via the discussion in Appendix F-2.

Three fuel forms of interest, Sr metal Po-210 metal, and Pu-238 metal, were selected for analysis. Of the three it was found that the Po-210 had such a high vapor pressure that no destruction other than breakup of the fuel capsule was necessary to achieve complete vaporization in very short times.

Figures 80 to 83 show that for both Pu-238 and Sr metals no gain is obtained by fragmenting the fuel form relative to the amount of mass loss which can be obtained by evaporation. If evaporation were to be used as a disposal technique for Sr metal just breakup of the fuel capsule from overheating would be enough to vaporize 90 percent of the charge before reentry heating and atmospheric effects were effective. In addition, due to the high vapor pressure of the Sr metal, the times for this mass loss to occur are quite short, approximately 1/2 hour, which indicates that even on ascent aborts yielding short lived orbits this disposal technique would be a valuable one.

Pu-238 metal forms, on the other hand, show more sluggish (evaporation) behavior as would be expected due to its lower vapor pressure. Again, no advantage is seen in fracturing the fuel form into smaller particles relative to mass loss by evaporation. Although the surface area is increased as was shown in Figure 79, the vapor pressure is very sensitive to temperature and small changes in temperature mean large changes in vapor pressure. Hence, a 100 $^{\circ}$F temperature drop due to fragmentation of the source could lower the vapor pressure by a factor of at least 2 - 3. Therefore, temperature, not surface area, is the controlling factor in mass loss by evaporation in space.

Figure 83 indicates that times of the order of 10^7 seconds would be required to reduce the mass of the index Pu system by 90 percent.

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[redacted]

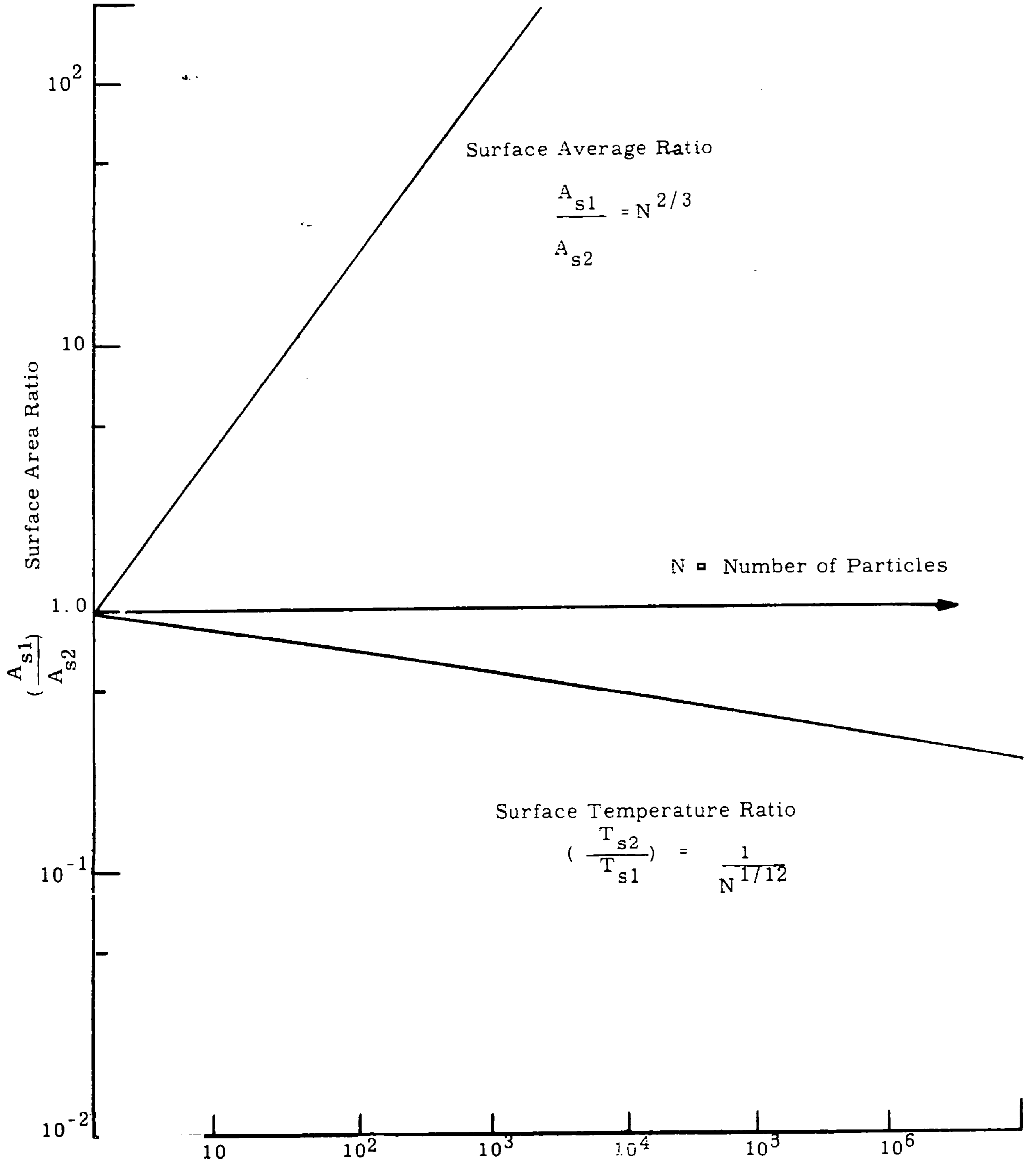

Figure 79. The Variation in Surface Area and Debris Surface Temperature as a Function of Number of Debris After Fracture

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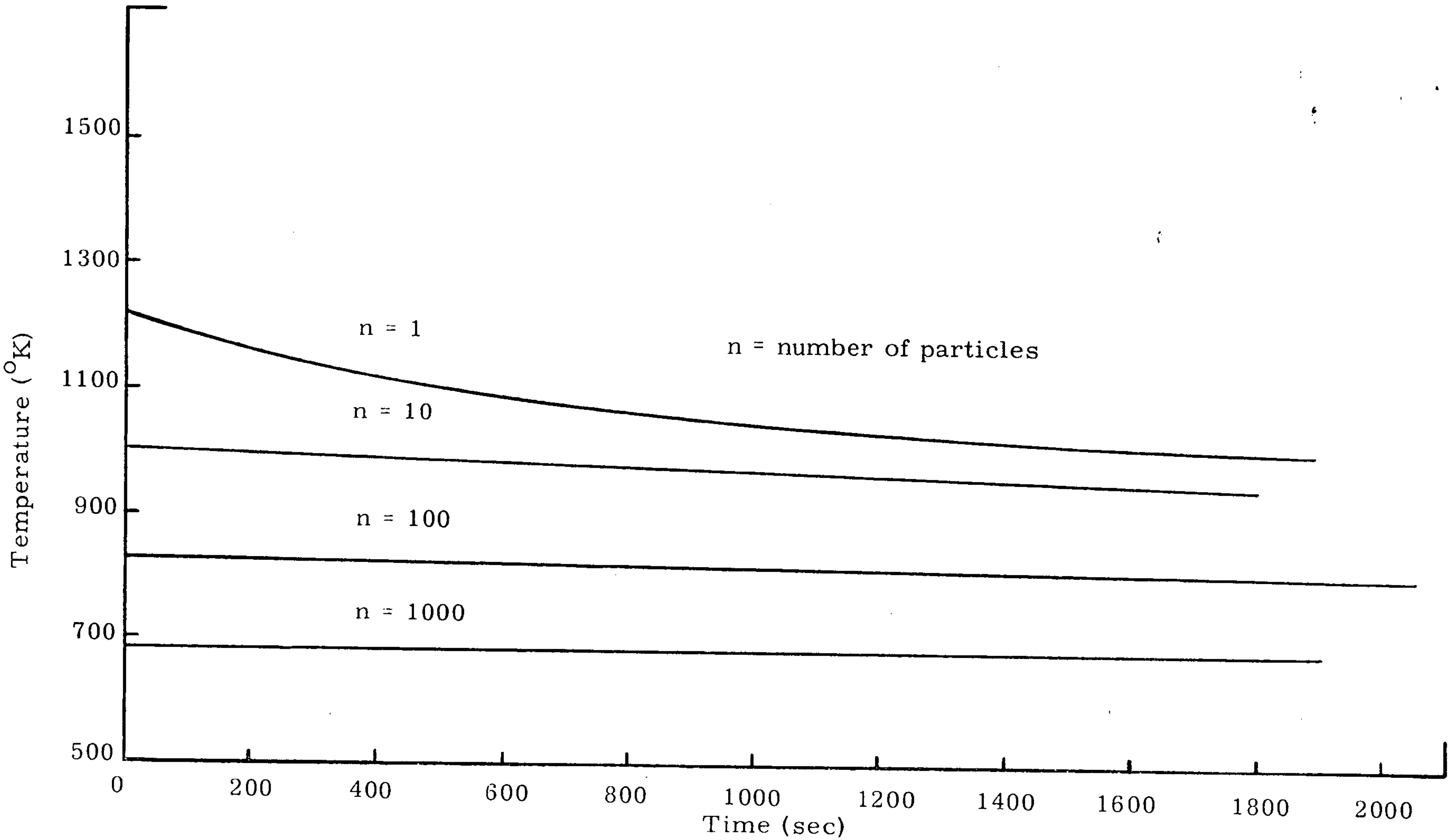

Figure 80. Temperature Profiles for Sr-90 Metal Debris In Space

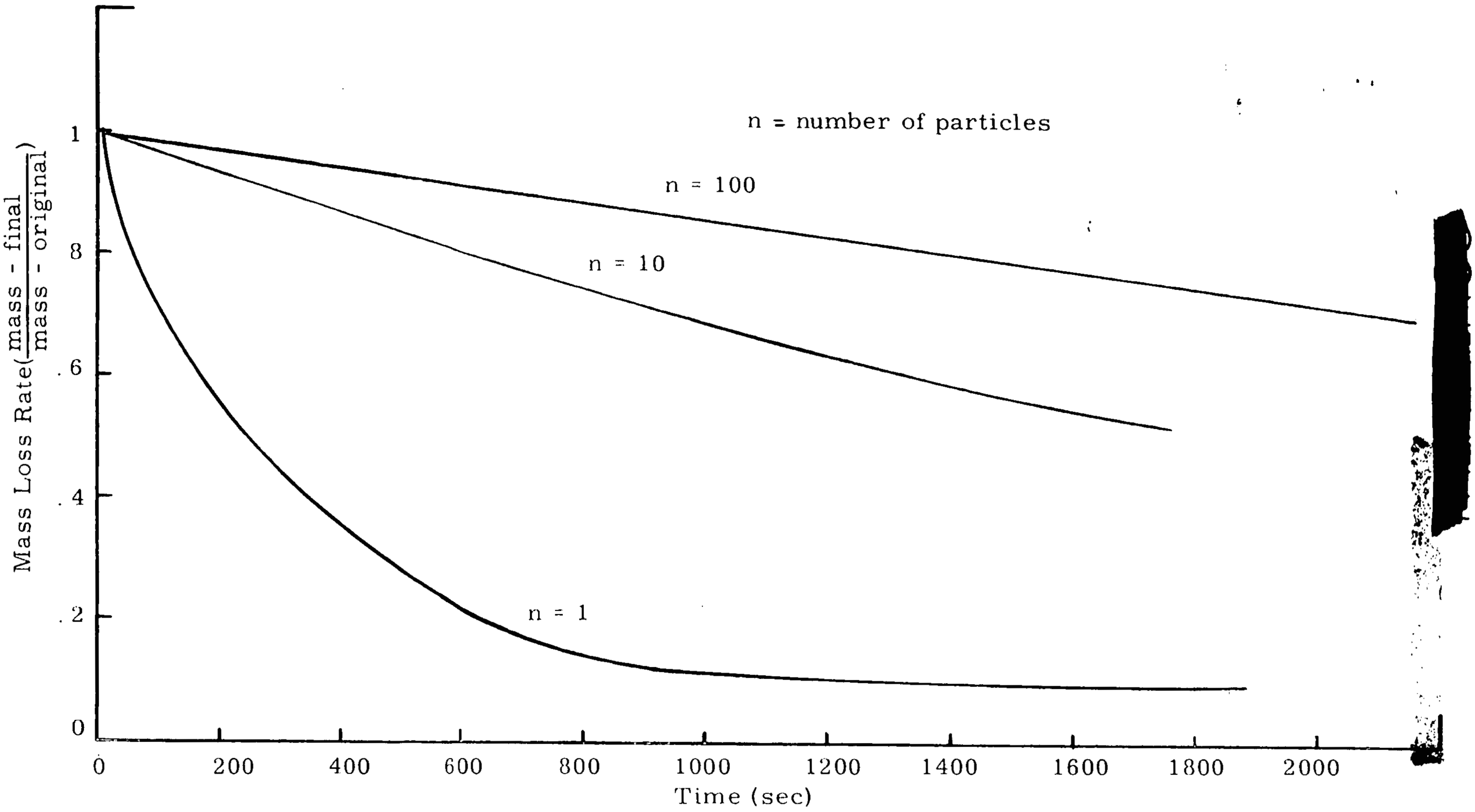

Figure 81. Mass Loss Profiles for Sr-90 Evaporation In Space

[redacted]

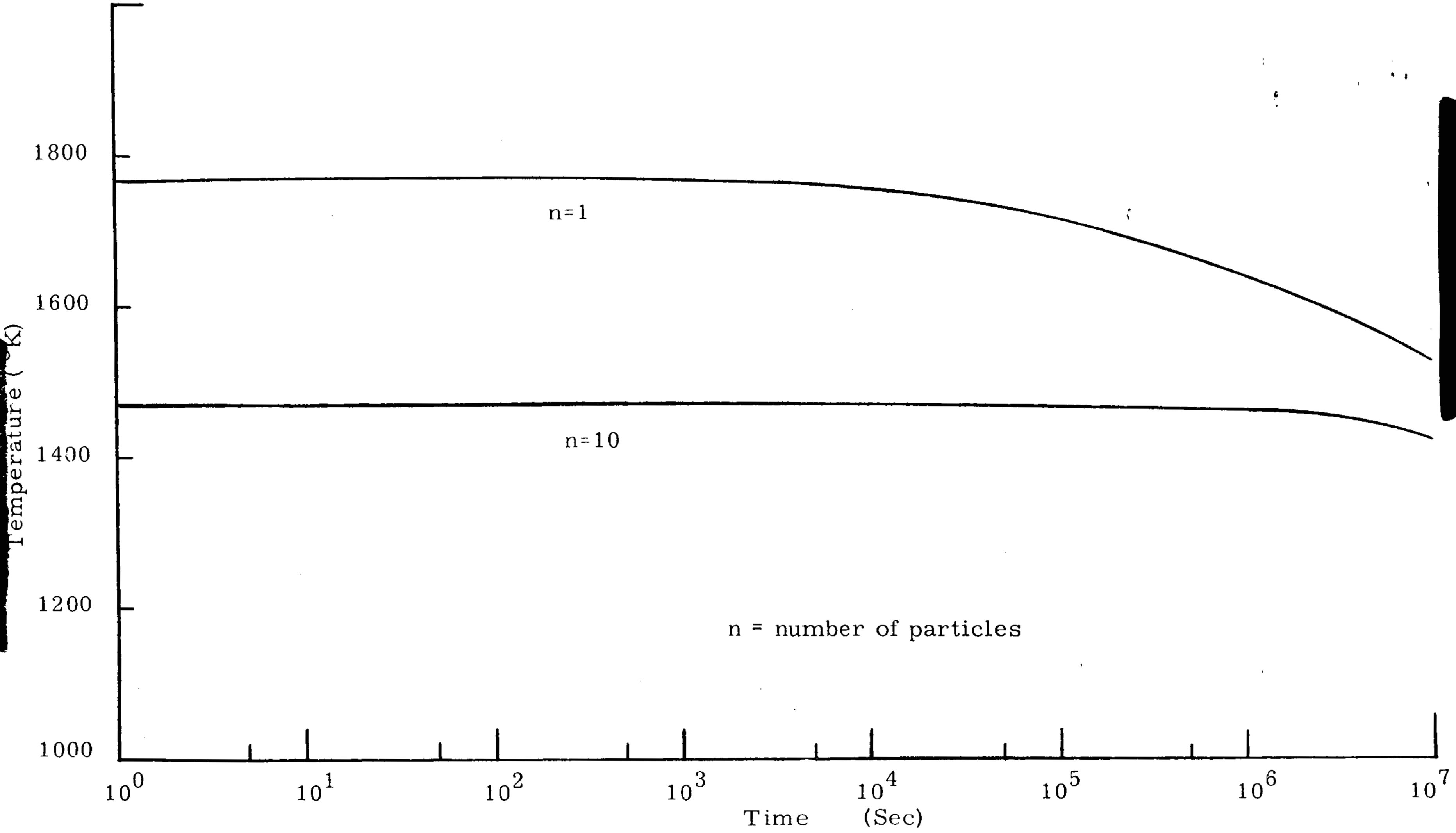

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Figure 82. Pu-238 Evaporation Performance as a Function of Size and Time

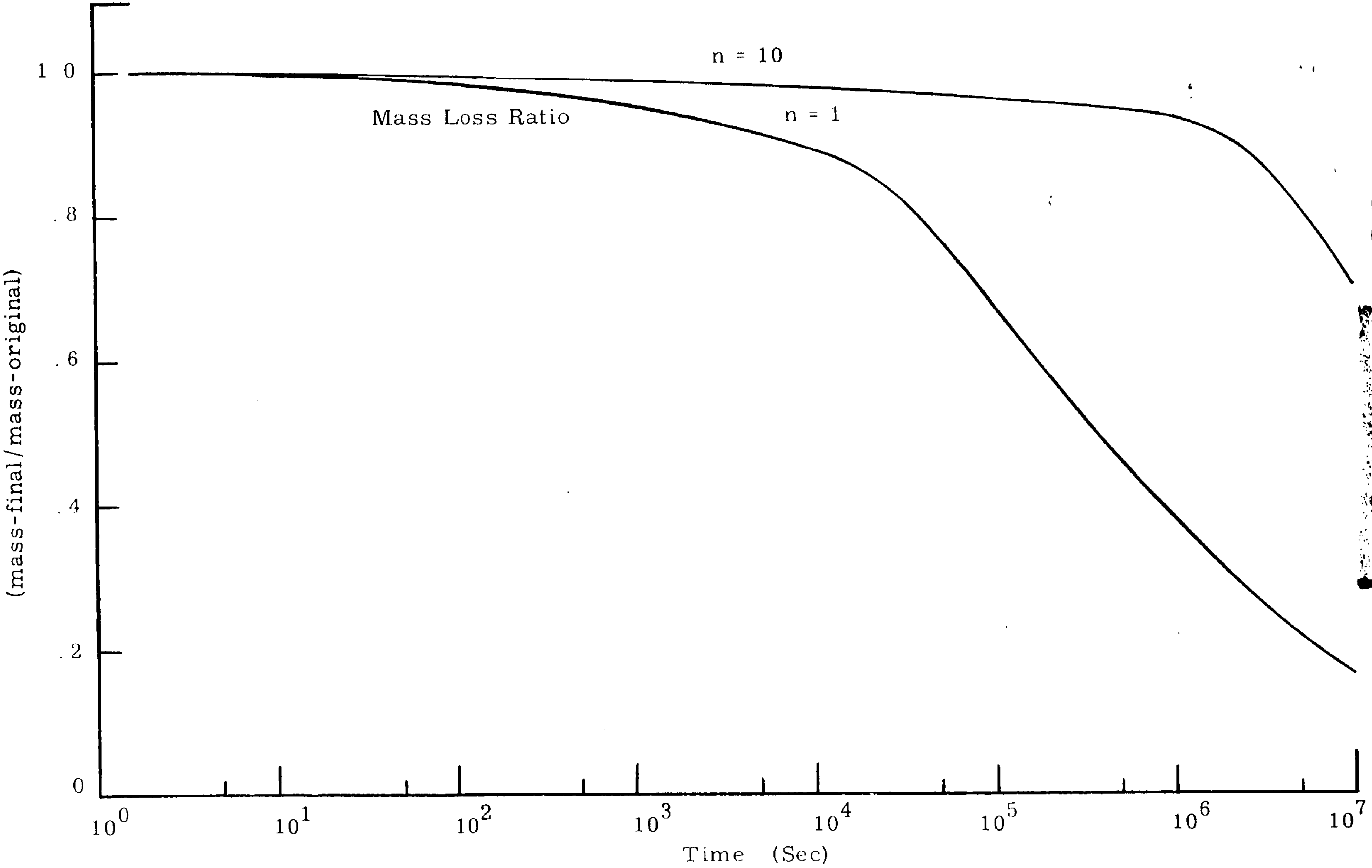

Figure 83. Pu-238 Evaporation Performance as a Function of Size and Time

[redacted]

In general, therefore, keeping the source intact with as large a ballistic coefficient as possible will tend to maximize the abort orbital lifetime and provide maximum time for evaporation. For Po-210 and Sr breakup and/or meltdown of the capsule (via mechanical or thermal means) during a short life orbital abort is acceptable and represents a disposal technique with definite potential. For Pu-238 on the other hand, this technique is marginal and is a function of orbital lifetime. If abort orbits of the length of a year or greater are assured, then the technique is a valuable one.

Further study on the mechanics of how a typical fuel capsule will melt under these conditions is advised. Derivation of design data, to assure proper exposure of the fuel for optimum evaporation to take place is a necessity before qualifying the technique.

The choice of fracture of the fuel form. based on the dispersion one may obtain in orbit before reentry, is still a valid approach, however, to minimize ground hazard. However, two points must be considered.

First, to what degree can fracture be obtained? Second, to what degree can dispersion in space be obtained before reentry.

A qualitative analysis has been carried out on the effects of chemical explosive on fracture of reactor cores in the NERVA program and the techniques would be roughly similar for large isotope systems. However, the results would be different since isotope systems would tend to have many small heat sources rather than one large one. The encapsulation would have to be impact and explosion qualified due to launch safety requirements and this would tend to increase the amount of explosive required to achieve the same fracture as in NERVA cores.

In addition, the work with NERVA has shown that maximum fracture efficiency results from explosive systems of the shaped charge type which can produce shock fronts in the core of 20.000 - 30,000 ft/sec. Therefore, optimum fracture would not give large Δ V's to the fragments. In fact, the NERVA pressure vessel is not greatly distorted or fractured by the internal blasts. Therefore, even though the fuel core is well fractured, it may still be contained.

Since large Δ V's are not expected, the cloud of fragments, if one is produced after the explosion, would tend to disperse slowly at best. However, they would disperse and yield an eventual advantage on reentry of increased surface area, longer exposure to the flow, and larger ground dispersion area.

In general, the explosion technique needs to be investigated in great detail and on a particular system before a final conclusion as to its effectiveness can be made.

[redacted]

[redacted]

2. Chemical Destruct

It may be possible especially in manned systems (such as a space station) to passively dispose of a spent radioisotope generator fuel charge by chemical techniques. Table XVIII shows some possible chemical solvents which could be used to dissolve various fuel forms of interest.

The technique is a simple one. It requires providing a reaction container into which the spent fuel core can be placed. The container is then sealed, pressurized, and the solvent is injected. A pressure valve prevents rupture of the container due to gaseous reaction products. After the reaction is complete, the container is ejected from the orbiting vehicle and vents itself to space boiling off the dissolved fuel core.

The key characteristic affecting application of this technique is rate of reaction which in turn is controlled by the mass and geometry of the fuel capsule. In general, it is felt that times of the order of days and weeks, not hours, would be required to bring about complete dissolution. Hence, application to deep space or orbital systems seems most reasonable.

3. Inherent Destruction Potential of Nuclear Systems in Space

a. Isotope Systems. The motivation for disassembly of a radioisotope fueled generator in orbit is the desire to render its nuclear debris less hazardous to the general public. Isotopes vary greatly in their radiation spectrum and biological activity, therefore, the hazards which they present are widely divergent in nature. What might be an acceptable method of disposal for one isotope would be completely unacceptable for another. For example, $Pu^{238}O_2$ is an α emitter with some spontaneous fission and an appreciable neutron contribution from the α-n reaction with O^{17} and O^{18}. The external radiation dose from a $Pu^{238}O_2$ source of one thousand thermal watts dispersed over 500 square miles represents an addition to the background radiation of approximately 5 percent.

$Pu^{238}O_2$ has been shown to be biologically inert, i.e., not subject to body concentration mechanisms. However, inhalation and retention of $Pu^{238}O_2$ in the lungs is very hazards, a single 5 micron particle being body burden.

Strontium-90 is a γ-emitter and has a considerable hazard from Bremsstrahlung emission. A dispersion of Sr-90 under the same conditions as outlined in the preceding paragraph would raise the background radiation by a factor of three. In addition, Sr-90 is a bone seeker, i.e., it tends to be concentrated by biological action. In its present space fuel form Sr-90 is highly soluble and could enter the ecological cycle quickly.

[redacted]

[redacted]

Table XVIII

Evaluation of Possible Chemical Solvents for Radioisotope Fuel Forms

Radioisotope	Fuel Form	Chemical Solvent
Sr-90	SrO	Hot water
	$SrTiO_3$	Fused Na_2O
	metal	perchloric acid
Cs-137	CsCl	Hot water (alcohol)
	CsF	Cold water
	Cs Polyglass	HF acid
Pm-147	Pm_2O_3	HCl
Po-210	metal	HNO_3
	GdPo	
Pu-238	metal	HNO_3
	PuO_2	HNO_3, H_2SO_4
Cm-242	Cm_2O_3	HCl and/or HNO_3
Cm-244	Cm_2O_3	HCl and/or HNO_3

Capsule Material	Chemical Solvent	Rate @ Boiling
Haynes-25	HCl	1000 mils/yr
Rene' 41	HCl	------------
TZM	H_2SO_4	------------
Tantalum	HF acid	1000 mils/yr

[redacted]

[redacted]

From the foregoing it may be concluded that different definitions of destruction may be used for different isotopes. A mechanism which allowed for breaching the fuel capsules and releasing large fuel particles which fall out rapidly over a relatively small area but cannot be inhaled, would appear desirable in the case of $Pu^{238}O_2$. However, such a mechanism would be completely unacceptable in the case of Sr-90.

A destruction method which produced very small particles dispersed over a large area might be technically permissible for Sr-90, but may not be acceptable for $Pu^{238}O_2$ because of this materials large inhalation hazard.

The only acceptable method of orbital destruction which would apply to all isotopes appears to be vaporization with the subsequent retention of the isotope vapor above the biosphere until appreciable radiological decay and/or dispersion had occurred. There is no inherent internal mechanism in isotope systems which would provide the energy required for vaporization while still in orbit. Such a mechanism does theoretically exist for nuclear reactor systems and this will be discussed in a later section.

(1) Definition of Destruction. For the purposes of this discussion, destruction in orbit shall be defined as the rending of the fuel into a particle size or state such that on its ultimate reentry into the biosphere it will not constitute a hazard to the general public. As shown in Section IV-D-1, this could encompass the whole spectrum of debris sizes to ultimate vaporization depending on the nature of the orbit and change in orbital lifetime of the debris after formation.

(2) Inherent Destructive Capability. Two basic forcing functions appear to offer some measure of inherent destruction capability for radioisotope systems. They are the stored thermal energy and possibly the pressure buildup for α-emitting radioisotopes which generate helium gas.

It is immediately apparent that both of these forcing functions do not have the capability to reduce the fuel form to microsized debris because the energy cannot be release quickly and efficiently transmitted to the fuel.

However, this may not be necessary for some fuels as shown in Section IV-D-1 because of the potential of space evaporation as a useable destruction technique for the complete fuel mass. Hence the use of the thermal and mechanical potential of the isotope fuel form should be one of breaking the fuel form containment rather than trying to destroy the fuel, so that normal evaporation in vacuum can deplete the fuel form as much as possible before eventual reentry.

[redacted]

[redacted]

Figure 84 shows the typical pressure potential versus time for a low void volume PuO_2 fueled radioisotope T/E generator. Since most aborts would lead to fairly quick reentries ($<$ 1 year), the pressure buildup in this case would not be available to break the capsule before reentry. Therefore, deliberate coolant loss would be the only inherent technique available under these circumstances to break the fuel capsule.

However, before this technique is employed, one must be assured that sufficient time exists in the space environment before reentry to evaporate the fuel to an acceptable degree.

It must be pointed out, however, that the meltdown of the capsule in space and subsequent evaporation of the capsule and fuel form could produce problems upon reentry of the residue into the earth's atmosphere. An alloy of the fuel form and the containment material could form, whose thermophysical properties are much different than the original constituents, and inhibit burnup. Therefore, this technique must be applied with caution as a function of the fuel form and containment material used.

In general, it can be stated that for materials with high vapor pressures it is best not to break up the fuel form in space, but just break the capsule and allow evaporation to deplete the inventory as much as possible before reentry. There appears to be a correlation between high vapor pressure and burnup reentry potential as shown in Section IV-A-3. Therefore, space evaporation coupled with reentry burnup for these materials is attractive. However, for low vapor pressure materials, an explosive system and/or a preformed fuel form (microspheres) coupled with capsule meltdown is attractive from the standpoint of reducing eventual hazards through maximizing reentry dispersal.

A change in generator design which would allow the fuel capsules to breach and simultaneously open one side of the generator to space might provide a means of allowing the fuel to sublime. This would be attractive for high vapor pressure fuels in long lived orbits (depending on the fuel form, see Section IV-D-1). This mechanism is not fully responsive to the destruction definition, however, since it may require a long lived orbit to utilize its full potential and since with decreasing temperature the rate of sublimation decreases markedly. These effects may result in incomplete vaporization of the fuel.

[redacted]

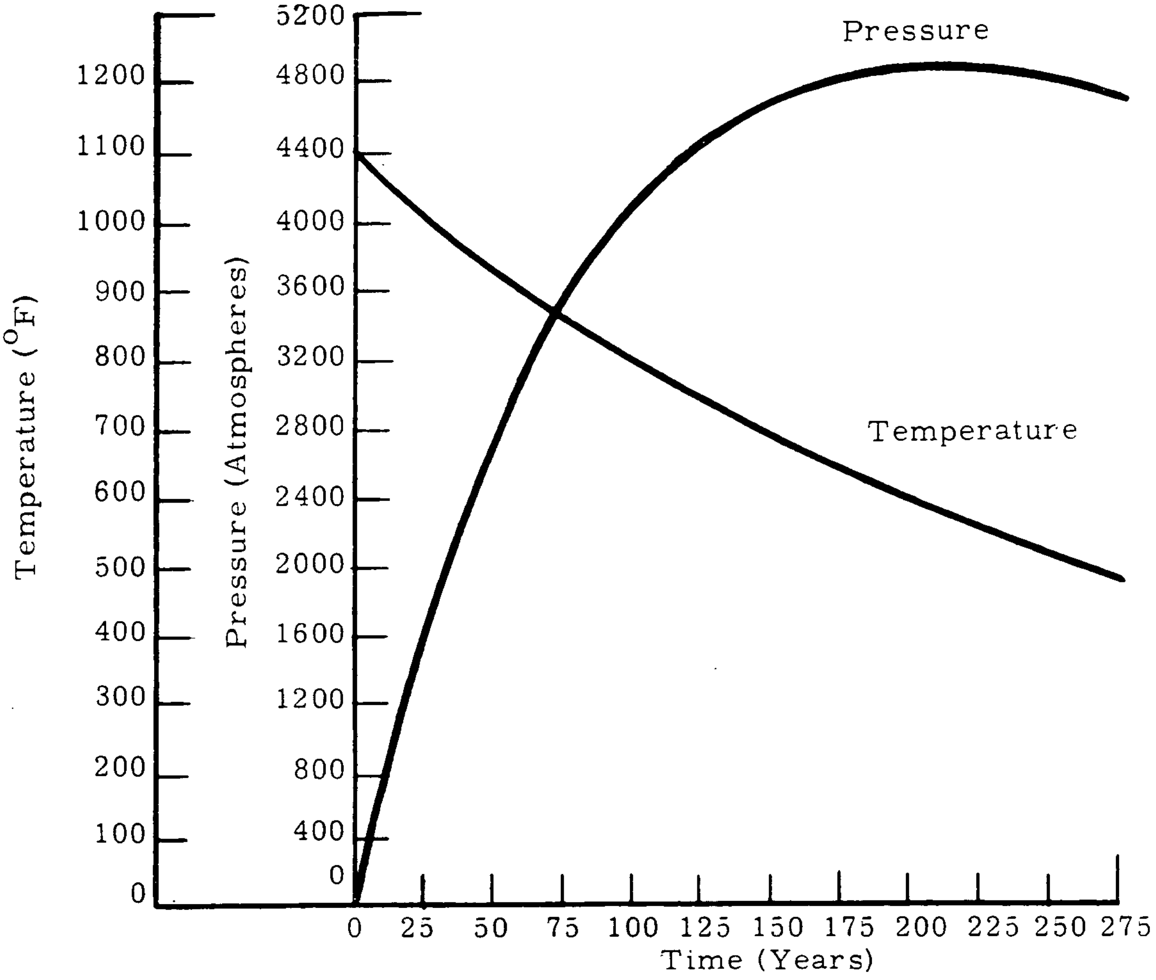

Figure 84. Variation in Capsule Pressure and Temperature with Time for a Typical Low Void Volume PuO_2 Fueled Radioisotope Thermoelectric Generator

[redacted]

Therefore, no generator design modification appears capable of offering a solution to complete destruction in orbit although changes may be made which could enhance sublimation and the probability of destruction on reentering the earth's atmosphere.

(3) Orbital Destruction by External Means. The destruction of the fuel by external means would require the addition of approximately 1218 Btu to vaporize a pound of PuO_2 from its normal operating temperature. Strontium fluoride would require the addition of approximately 2032 Btu per pound.

Normal chemical explosives do not function in a manner which would lead to fuel vaporization. This is because the rate at which energy can be released from a chemical reaction is controlled by chemical equilibrium. This leads to limits on effective flame front temperatures which in turn limit the availability of energy released from the explosive. Further; chemical explosives would tend to break the fuel capsules and fuel into shrapnel and expel them from the reaction field before they can reach equilibrium with the flame front temperature.

It is felt that conventional explosives do not release their energy at a rate or availability sufficient to allow them to be useful in destroying radioisotope fuel capsules within the previously defined definition of destruction.

The use of satellite or ground launched intercept missiles for orbital destruction would require nuclear warheads since conventional explosives will not provide the thermal effects for destruction. Nuclear warheads would probably be necessary in any event to provide for some guidance system inaccuracies. Destruction by external nuclear means would require detonation of warheads at orbital altitudes. This probably unacceptable at any reasonable future date because of the limited test ban treaty and because of the effects on satellite communications that the injection of additional electrons into the Van Allen belts would have. This latter effect was graphically demonstrated by the results of the Johnston Island "Starfish" shot.

b. Reactor Systems. The normal design of a nuclear power reactor incorporates many safety precautions which limit the total energy that can be released to the core in a short time. Thus, the possibility for a very large energy release capable of vaporizing an entire core can only be achieved with great ingenuity and effort. For large power stations, the chemical reaction, or gas explosion, that may be triggered due to the nuclear excursion warrant much more concern because, depending on the core materials, the chemical energy release can be orders of magnitude greater than the direct nuclear heat generation.

[redacted]

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Nuclear reactors for space application, because they do not pose a direct danger to populated areas during normal operation, may be permitted emergency operational characteristics which could include a self-destruct capability. A study of reactor self-destruction depends on the particular reactor type; it is necessary to specify the core size, configurations, and control system to attain accurate quantitative results. However, certain general conclusions are possible without such a detailed analysis.

(1) Safety Considerations Involved in the Design of a Nuclear Reactor.

There are certain basic ground rules which apply to the design of a nuclear reactor that have to be followed in order for the reactor to be licensed by the Atomic Energy Commission regulatory group. Some of these ground rules are discussed below together with a discussion of how they affect the self-destruct potential of the reactor.

(2) Reactor Shutdown With One Control Element Inoperable.

The basic control philosophy used in reactor design is that the control system shuts the reactor down at the most reactive time in life with the control element that controls the most reactivity in its most reactive position. As examples: for a reactor using poison control rods, this condition corresponds to the strongest control rod stuck out of the reactor; for a reactor using fuel control rods this condition corresponds to the strongest control rod stuck in the reactor; for a reactor using control drums, the condition corresponds to the strongest drum stuck in a position in which moderator or fuel is adjacent to the stationary fuel and the neutron poison, or neutron leakage path, is minimized.

If a preliminary design does not meet the one stuck rod condition, the number of control elements can be increased to decrease the worth of individual element or auxiliary control such as burnable poison may be used to decrease the excess reactivity. In any event, the price of meeting the one stuck element condition is to reduce the total excess reactivity that the reactor can gain by the movement of the entire control system from the least to the most reactive condition. Since the magnitude of the nuclear energy generated during an excursion is an exponential function of the magnitude of the reactivity input, this design limitation is very severe for possible self-destruction.

(3) Temperature Coefficient. The temperature coefficient depends on the reactor design and is a complicated function of reactor operation. Temperature changes in fuel, moderator, coolant and structure produce separate reactivity changes which are spatially and time dependent. It is inherent in reactor design that the temperature coefficient be negative and that under normal operating conditions, the reactor output be stable and any change in temperature does not produce an autocatalytic excursion.

[redacted]

[redacted]

The effect of the negative temperature coefficient is to negate the effect of a reactivity insertion; any increase in power and temperature causes a negative reactivity change which compensates for initial reactivity insertion and will terminate the excursion. Thus the magnitude and time dependence of the negative temperature coefficient is very significant in space reactors in order to determine the ability to generator sufficient heat to destroy a reactor.

(4) Rate of Reactivity Insertion. For normal power operation, it is desirable to start up a reactor slowly. Reactivity is inserted at a slow rate by slow movement of individual control elements to make certain that operating power and temperature limits are not exceeded. On the other hand, for a nuclear rocket propulsion reactor, the success of the mission is dependent on very rapid rate of reactivity insertion to achieve specified thrust over a specified time interval.

For self-destruction, it is necessary to insert the reactivity very rapidly. Propulsion reactors may have this capability while space power supplies would require significant design changes to achieve the capability.

(5) Non-Nuclear Disassembling of Space Reactors. Space reactors are designed on the basis of minimum weight, and high power density, therefore the fissionable fuel inventory is larger than thermal reactors. A launch abort or reactor reentry with subsequent immersion in water could produce a self-sustaining nuclear reaction. Therefore, the mechanical design must consider mechanisms to break the reactor into pieces such that each piece will be subcritical in water. The method of obtaining subcriticality is to use non-nuclear explosives to destroy the core or to design core support structure such that the core can be disassembled under specified conditions.

The use of non-nuclear explosive only provides sufficient energy to break the core into large pieces and cannot be used for complete core destruction such as required to vaporize the entire core.

(6) Characteristics of Supercritical Excursions in a Nuclear Reactor.

If the amount of reactivity that is inserted into a reactor is greater than the reactivity associated with the delayed neutron fraction, simplifications can be made in the transient equations and relatively simple expressions can be derived to determine the temperature, power and total energy generation of the core as a function of time.

[redacted]

[redacted]

The equations that govern the time behavior of a reactor described by time independent lumped parameters and neglecting delayed neutrons effects are based on the Fuch-Nordheim model (Reference 54). Scolettar has extended the model to include variable heat capacity within the core (Reference 55).

These equations are:

$$\frac{dP}{dt} = \frac{\delta k_p - dT}{l_p} \quad P \tag{1}$$

$$C \frac{dT}{dt} = P \tag{2}$$

$$C = C_o + \alpha T \tag{3}$$

δk_p = amount of reactivity in section
P(T) = reactor power at time t
T = core temperature
l_p = prompt neutron generation time
α = temperature coefficient of reactivity
C = heat capacity varying linearly with temperature

Division of Equation (1) by (2) and integrating yields the following expression for the power, P;

$$P = P_o + \frac{1}{l_p} \left[C_o \delta k_p T + \frac{1}{2} (\gamma \delta k_p - \alpha C_o) T^2 - \frac{1}{3} (\alpha \delta T^3) \right] \tag{4}$$

The final temperature T_{max} is:

$$T_{max} = \frac{3}{4} \frac{\delta k_p}{\alpha} \left(\frac{\alpha C_o}{\gamma \delta k_p} - 1 \right) \left[\sqrt{1 + \frac{\frac{16}{3} \frac{\alpha C_o}{\gamma \delta k_p}}{\left(\frac{\alpha C_o}{\gamma \delta k_p} - 1 \right)^2}} - 1 \right] \tag{5}$$

The latent energy release as a function of core temperature is:

$$E = \int_o^t P \, dt = \int_o^T C \, dT = C_o T + \frac{1}{2} \gamma T^2 \tag{6}$$

[redacted]

These solutions represent upper limits of the total energy generation. The assumption of instantaneous reactivity insertion and the neglect of the temperature coefficient all tend to yield a higher energy output than if these effects were described more accurately. Furthermore, mechanical expansion, or chemical reaction within the core prior to maximum power generation will terminate nuclear excursions at an earlier time.

Calculations by Moss and Wilson (Reference 56) in which the mechanical expansion of a SNAP reactor core was considered, show that fuel disintegration occurs over a small region at the center of the core. This disintegration produces sufficient pressure to expand the core and terminate the excursion prior to maximum power generation. The result of this analysis which was carried out for a 2% k excess reactivity is shown in Figure 85. Only 46 Mw seconds of nuclear energy has been generated as a result of the nuclear excursion.

(7) KIWI Transient Nuclear Test. The KIWI Transient Nuclear Test, or KIWI-TNT, was designed to investigate the behavior of a KIWI reactor under conditions of rapid insertion of abnormal excess reactivity (Reference 57). However, this test was not carried out with a standard core which was designed to meet safety considerations previously discussed. The shutdown margin of the reactor was decreased by removal of internal reactor poisons and the drum actuators were modified to increase their speed by a factor of approximately 100. These two modifications result in a reactivity insertion of $300 per second with a total super-prompt excess reactivity of $7.50.

The preliminary results of this test indicate that complete vaporization of the core did not occur. In fact, only about 5 percent of the core was vaporized.

(8) Possibility of Achieving Complete Core Destruction. The results of the KIWI-TNT test indicate that even under the best of conditions it may not be possible to completely vaporize the core solely by a nuclear excursion. Since KIWI is a graphite moderated core with a relatively long prompt generation time these results may not necessarily apply to all reactor types. Nevertheless, the results appear to indicate that short of designing a reactor that can function as a bomb at the time of destruction it may not be possible to vaporize a reactor core solely by a nuclear excursion.

If complete destruction of a space power reactor is a specification for an operating reactor then certain design innovations can be attempted which may not affect normal operating characteristics. These are:

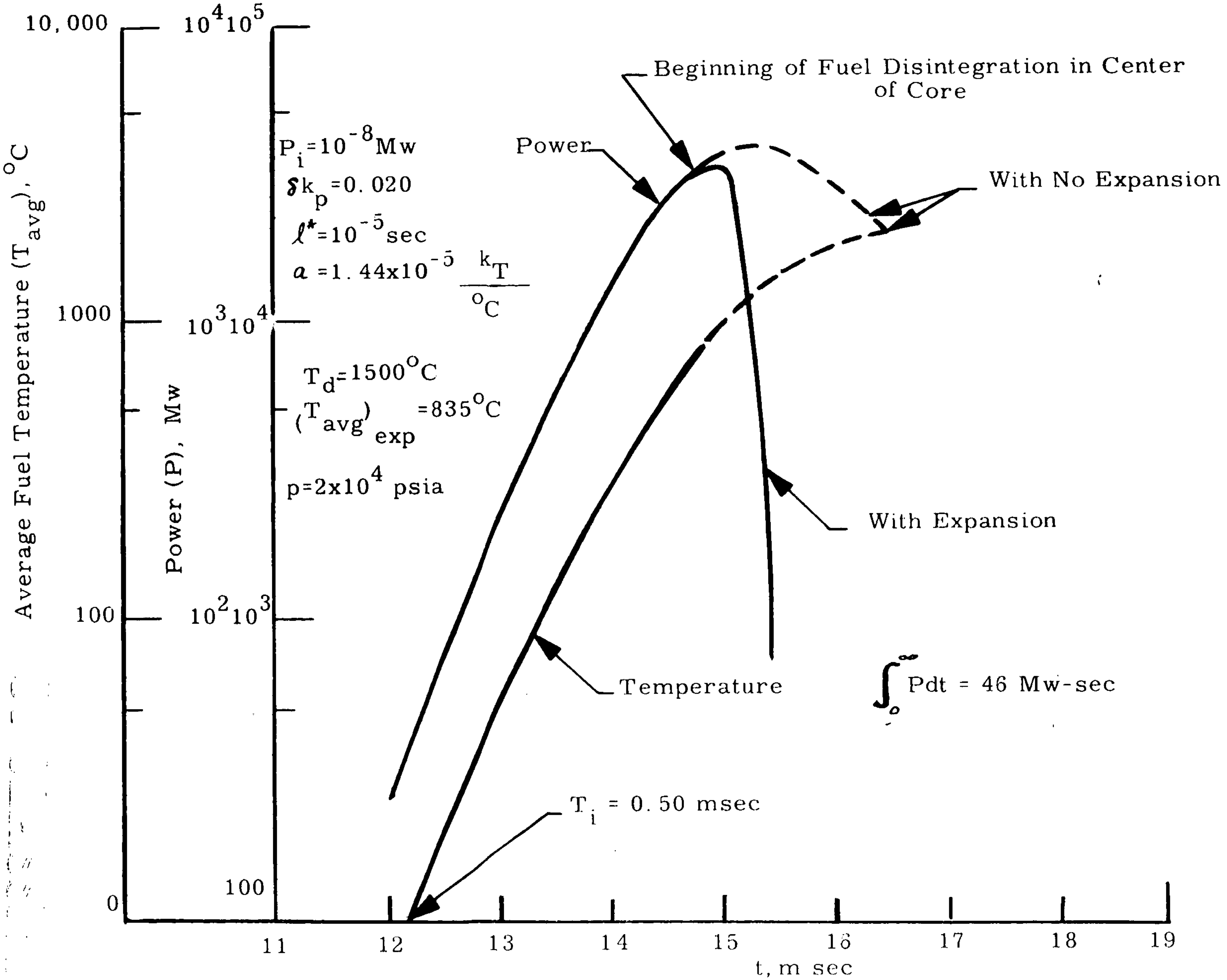

Figure 85 Temperature and Power Versus Time After Start of Transient.

[redacted]

(a) Design the core to have a control element permanently installed in the core; the element withdrawn increases reactivity by several percent. A non-nuclear explosive or fuse ejects the control element at time of destruct giving shortest possible reactivity insertion.

(b) Design the core to have a very short prompt generation time. In addition, use fissile fuel such as Pu-239 which has small delayed neutron fraction and possibly a positive Doppler coefficient.

(c) Design the core to have a very flat power distribution both radially and axially such that the entire core can achieve maximum temperature and the thermal gradients that produce mechanical disassembly will be minimized.

One alternative to a complete destruct of the core is to use the nuclear excursion to trigger a chemical explosion. The core design would then have to contain materials that could have a violent chemical reaction at high temperatures but be chemically inert at operating temperatures.

(9) Necessity for Destruction in Orbit. The major danger caused by the reentry of a reactor that has operated at high power is that it will represent a radiation hazard. However, if the gaseous fission products were to be eliminated then the hazard is lessened since the solid fission products can be more easily located and contained.

A possible operational procedure for the power reactor at the end of its space mission is to permit core meltdown by shutting off coolant flow. All the gaseous fission products, 50 percent of the halogens, and 1 percent of the solid fission products would be released in orbit which represents a decrease of 17 percent of the total fission product activity and the core would resolidify. This reduction would lessen the potential danger of the reactor core. However, a tradeoff between the reentry disassembly of the core as postulated in SNAP-10A designs versus pre-reentry meltdown relative to final hazards to the population, has to be made for a final decision.

4. Results and Conclusions

a. Isotope Systems

(1) Inherent Destructive Capability. Two basic forcing functions appear to offer some measure of inherent destruction capability for radioisotope systems. They are the stored thermal energy and possibly the pressure buildup for α- emitting radioisotopes which generate helium gas.

[redacted]

It was apparent from this analysis that both of these forcing functions do not have the capability to reduce the fuel form to microsized debris because the energy cannot be released quickly and efficiently transmitted to the fuel.

However, this may not be necessary for some fuels because of the potential of space evaporation as a useable destruction technique for the complete fuel mass. Hence the use of the thermal and mechanical potential of the isotope fuel form should be one of breaching the fuel form containment rather than trying to destroy the fuel, so that normal evaporation in vacuum can deplete the fuel form as much as possible before eventual reentry. Therefore, deliberate coolant loss would be the only inherent technique available under these circumstances to breach the fuel capsule.

However, before this technique is employed, one must be assured that sufficient time exists in the space environment before reentry to evaporate the fuel to an acceptable degree.

It must be pointed out however, that the meltdown of the capsule in space and subsequent evaporation of the capsule and fuel form could produce problems upon reentry of the residue into the earth's atmosphere. An alloy of the fuel form and the containment material could form, whose thermophysical properties are much different than the original constituents, and inhibit burnup. Therefore, this technique must be applied with caution as a function of the fuel form and containment material used.

(2) Fragmentation and Evaporation. It was determined for both Pu-238 and Sr metals no gain is obtained by fragmenting the fuel form relative to the amount of mass loss which can be obtained by evaporation. The dependence of vapor pressure on temperature proved the most significant variable. When fragmentation occurred the equilibrium temperature of the fragments dropped as a function of the number of fragments, causing a significant decrease in the overall mass loss in space. Po-210 with its high vapor pressure will evaporate very quickly.

Hence, keeping the source intact with as large a ballistic coefficient as possible will tend to maximize the abort lifetime and provide maximum time for evaporation. For Po-210 and Sr metal breakup of the capsule (via mechanical or thermal means) during a short life orbital abort is acceptable and represents a disposal technique with definite potential. For Pu-238 on the other hand, this technique is marginal and is a function of orbital lifetime. If abort orbits of the length of a year or greater are assured then the evaporative technique is a valuable one.

[redacted]

In any case pure fragmentation in space even without significant evaporative mass loss is a valuable safety technique since it will assure excellent dispersal of the fuel form before reentry and eventual lower surface contamination. However, the problem here is in assuring the degree of fragmentation which will be significant. This may be easily done for large systems such as the NERVA but is much more mechanically difficult and costly in terms of weight on the smaller isotope systems. Therefore, the pre-fragmented microsphere approach with just simple fuel capsule meltdown appears to be the most logical choice to augment space dispersal.

In general, it can be stated that for materials with high vapor pressures it is best not to break up the fuel form in space, but just breach the capsule and allow evaporation to deplete the inventory as much as possible before reentry. There appears to be a correlation between high vapor pressure and burnup reentry potential. Therefore, space evaporation coupled with reentry burnup for these materials is attractive. However, for low vapor pressure materials, an explosive system and/or a preformed fuel form (microspheres) coupled with capsule meltdown is attractive from the standpoint of reducing eventual hazards through maximizing reentry dispersal.

(3) Chemical Destruct. It may be possible especially in manned systems (such as a space station) to passively dispose of a spent radioisotope generator fuel charge by chemical techniques.

The key characteristic affecting application of this technique is the rate of reaction which in turn is controlled by the mass and geometry of the fuel capsule. In general, it is felt that times of the order of days and weeks, not hours, would be required to bring about complete dissolution. Hence application to deep space or orbital systems seems most reasonable.

(4) Orbital Destruction by External Means. It has been determined that using external means such as satellite or ground launched missiles to destroy the generator in space could not be effective unless a nuclear warhead were used. Even a direct hit with a chemical explosive system may not breach the fuel containment structure, hence only temperatures of the level to produce full vaporization would be effective.

Based on the limitations of the present test ban treaty nuclear destruct in orbit probably can not be though of as a practical orbital destruct technique.

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b. Reactor Systems - Inherent Destructive Capability.

Analyses have shown that complete destruction of SNAP or NERVA type cores by deliberate excursions is not obtainable. The mechanical forces in the core generated by the temperature of the excursion tend to expand and/or break up the core prior to maximum power generation.

If complete destruction of a space power reactor is a specification for an operating reactor then certain design innovations can be attempted which may not affect normal operating characteristics. These are:

(1) Design the core to have a control element permanently installed in the core; the element withdrawn increases reactivity by several percent. A non-nuclear explosive or fuse ejects the control element at the time of destruct giving shortest possible reactivity insertion.

(2) Design the core to have a very short prompt generation time. In addition,use fissile fuel such as Pu-239 which have small delayed neutron fractions and possibly a positive Doppler coefficient.

(3) Design the core to have a very flat power distribution both radially and axially such that the entire core can achieve maximum temperature and the thermal gradients that produce mechanical disassembly will be minimized.

One alternative to a complete destruct of the core is to use the nuclear excursion to trigger a chemical explosion. The core design would then have to contain materials that could have a violent chemical reaction at high temperature but be chemically inert at operating temperatures.

Coolant shutoff and subsequent core meltdown in space was also investigated. It was found that a decrease of 17 percent of the total fission product inventory could be expected for a typical power reactor. However, this still leaves a significant inventory which may reenter.

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V. APPLICATION OF GENERATED SAFETY GUIDELINES TO THE FUTURE MISSION PROFILES IN THE 1965-1980 TIME PERIOD

From the analyses of planned aerospace missions, power systems, and aerospace safety criteria, in which the problems and requirements of each were defined, the most appropriate power system and supporting aerospace safety criteria were estimated for each space mission, as illustrated in Table XIX.

The selections in Table XIX represent best estimates at this point in time as to the primary and backup safety philosophies which could be used to satisfy the requirements of the missions as postulated.

The results of this table show that intact reentry is tied very closely with manned systems and large fuel inventories. Reentry burnup or microspheres are the predominant backup safety philosophies for most of the missions analyzed and definitely are feasible for future mission work.

Orbital storage is surprisingly attractive as a primary disposal technique because it is tied to many missions which have the capability even after an abort to achieve orbital storage.

Deep space disposal is only attractive for those missions which have inherent escape potential or for manned interplanetary missions.

[redacted]

Table XIX -Space Missions, Associated Power Systems, and **Estimated Aerospace Nuclear Safety Criteria**

Mission		Power System Selected			Aerospace Nuclear Safety Criteria		
No. Manned or Unmanned	Name	Power Level Kw(e)	First Choice	Second Choice	Primary	Backup	Time Periods
1 Manned	Extended Apollo	1.5 to 6.0	Non-Nuclear	Nuclear	Intact controlled reentry with recovery or orbital storage for long lived orbits, e.g., sychronous altitudes	Ocean burial of the source may be accomplished during the mission via retro from orbit or ejection from the manned craft during the descent phase. Shielding may be incorporated into fuel block to prevent direct radiation hazard.	1965
2 Manned	MORL	4.0 to 8.0	Nuclear	Non-Nuclear	Intact controlled reentry with recovery	Same as mission No. 1. (to augment the fuel containment criterion in the terrestrial environment the microsphere fuel form should be selected)	
3 Manned	Space Station	7.0 to 10.0	Nuclear	Non-Nuclear	Intact controlled reentry with recovery	Same as mission No. 2	
4 Manned	Gemini	-----	Non-Nuclear	------	(Not Applicable)	(Not Applicable)	
5 Unmanned	FM Broadcast	1.0	Nuclear	Non-Nuclear	Orbital storage - primary with either intact or burnup system backup	Same as mission No. 2	
6 Unmanned	TV Broadcast	30.0 to 60.0	Nuclear	-----	Orbital storage primary with either intact or burnup system backup	Same as mission No. 2	
7 Manned	LORL	15.0 to 40.0	Nuclear	-----	Intact controlled reentry	Space disposal; explosive destruct to subcritical fragments prior to ocean burial, ocean burial via retro from orbit, or ejection during the descent phase of the trajectory yielding ocean burial	
8 Manned	ORB TELE	1.0 to 1.5	Nuclear	Non-Nuclear	Orbital storage - either intact or burnup system design criteria may be used or intact controlled reentry with recovery	Same as mission No. 2	
9 Unmanned	Syncom	0.025 to 0.050	Non-Nuclear	-----	(Not Applicable)	(Not Applicable)	
10 Unmanned	Explorer	0.015 to 0.05	Non-Nuclear	Nuclear	Orbital storage - may be used (type of mission yields in space of fuel before reentry) either intact or burnup system design criteria	a) Microsphere fuel form backup to the fuel containment design approach. b) Ascent trajectory shaping to provide maximum surface traverse over deep ocean c) Atmospheric suspension via baloon technique d) Shield the source to prevent radiation hazards resulting from intact random reentry	
11 Unmanned	Telstar	0.060	Non-Nuclear	Nuclear	Reentry burnup with high altitude dispersion or partial dispersion (microspheres)	a) Ascent trajectory shaping to provide maximum surface traverse over deep ocean b) Atmospheric suspension via balloon technique c) Shield the source to prevent radiation hazards resulting from intact random reentry	Early 70's

Space Missions, Associated Power Systems, and Estimated Aerospace Nuclear Safety Criteria

Table XIX (Cont'd)

[redacted]

Mission		Power System Selected			Aerospace Nuclear Safety Criteria		
No. Manned or Unmanned	Name	Power Level Kw(e)	First Choice	Second Choice	Primary	Backup	Time Periods
12 Unmanned	Comsat	0.050 to 0.080	Non-Nuclear	Nuclear	Reentry burnup with high altitude dispersion or partial dispersion microsphere (the planned mission orbit will yield in space decay)	Same as mission No. 11	1965
13 Unmanned	Tiros	0.025 to 0.050	Non-Nuclear	Nuclear	Reentry burnup with high altitude dispersion or partial dispersion (microspheres)	Same as mission No. 11	
14 Unmanned	Relay	0.050	Non-Nuclear	Nuclear	Same as Mission No. 13	Same as mission No. 11	
15 [redacted]anned	Transit	0.025	Nuclear	Non-Nuclear	Same as Mission No. 13	Same as mission No. 11	[redacted]
16 [redacted]anned	Advanced AF	0.040	Nuclear	Non-Nuclear	Same as Mission No. 13	Same as mission No. 11	[redacted]
17 [redacted]anned	Nimbus	0.050 to 0.250	Non-Nuclear	Nuclear	Orbital storage - either intact or burnup design criteria may be used as a backup	Same as mission No. 11	[redacted]
18 [redacted]anned	RAO	0.250 to 0.500	Non-Nuclear	Nuclear	Orbital storage - (type of mission yields in space decay of fuel before reentry) either intact on burnup system design criteria may be used	Same as mission No. 11	[redacted]
19 Unmanned	Adv. Orb. Solar Obs.	0.30 to 0.50	Non-Nuclear	Nuclear	Same as mission No. 10	Same as mission No. 10	[redacted]
20 Unmanned	Advanced Navy	0.015 to 0.050	Nuclear	Non-Nuclear	Reentry burnup with high altitude dispersion and/or partial burnup (microspheres)	Same as mission No. 10	[redacted]
21 Unmanned	Operational Weather Satellite	0.50	Nuclear	Non-Nuclear	Same as mission No. 10	Same as mission No. 10	
22 Unmanned	Adv. Meteorological Satellite	0.050 to 0.100	Nuclear	Non-Nuclear	Same as Mission No. 10	Same as mission No. 10	Early 70's

Space Missions, Associated Power Systems, and Aerospace Estimated Nuclear Safety Criteria
Table XIX (Cont'd)

Mission		Power System Selected			Aerospace Nuclear Safety Criteria		
No. Manned or Un-manned	Name	Power Level Kw(e)	First Choice	Second Choice	Primary	Backup	Time Periods
23 Unmanned	OAO	0.500 to 0.750	Non-Nuclear	Nuclear	Same as mission No. 10	Same as mission No. 10	1970
24 Unmanned	SMS	0.50	Nuclear	Non-Nuclear	Same as mission No. 10	Same as mission No. 10	
25 Unmanned	OGO	0.30 to 0.50	Non-Nuclear	Nuclear	Same as mission No. 10	Same as mission No. 10	
26 Unmanned	SGM	0.150 to 0.200	Non-Nuclear	Nuclear	Same as mission No. 10	Same as mission No. 10	
27 Unmanned	Satellite Inspector	1.0	Non-Nuclear	Nuclear	Same as mission No. 10	Same as mission No. 10	
28 Unmanned	AOAO	1.0	Nuclear	Non-Nuclear	Same as mission No. 10	Same as mission No. 10	
29 (same as mission No. 10) Unmanned	Explorer	0.015 to 0.050	Non-Nuclear	Nuclear	Same as mission No. 10	Same as mission No. 10	
30 Unmanned	Surv. Orb. and Lan.	0.025 to 0.125	Nuclear	Non-Nuclear	Intact reentry (controlled by shielding lunar impact consideration)	a) Ascent trajectory selection to minimize any possible land impact b) Shielded source minimizes radiation hazards	
31 Manned	Apollo	1.5 to 2.0	Non-Nuclear	Nuclear	Intact controlled reentry with recovery	Same as Mission No. 2	
32 Manned	LEM	0.10 to 0.50	Non-Nuclear	----	(Not Applicable)	(Not Applicable)	
33 Manned	Roving Vehicles	3.0 to 15.0	Nuclear	Non-Nuclear	Intact controlled reentry (translunar phase) system will be left on the lunar surface - disposal via decay	Same as Mission No. 2 (If a reactor system startup shall not occur until the system is on the lunar surface, except for orbital checkout, core fragmentation and subsequent ocean burial must be achieved.	1975

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Space Missions, Associated Power Systems, and Estimated Aerospace Nuclear Safety Criteria
Table XIX (Cont'd)

Mission No. Manned or Unmanned	Mission Name	Power System Selected Power Level Kw(e)	Power System Selected First Choice	Power System Selected Second Choice	Aerospace Nuclear Safety Criteria Primary	Aerospace Nuclear Safety Criteria Backup	Time Periods
34 Manned	Lunar Exploration	2.0 to 5.0	Nuclear	Non-Nuclear	Intact controlled reentry (Translunar phase) system will be left on the lunar surface - disposal via decay	Same as mission No. 2 (if a reactor system startup shall not occur until the system is on the lunar surface, except for orbital checkout, core fragmentation and subsequent ocean burial must be achieved.	1970
35 Unmanned	Scientific Probe	0.050 to 0.500	Non-Nuclear	Nuclear	Reentry burnup with high altitude release partial release (microspheres)	Same as mission No. 10	
36 (same as mission No. 10) [illegible]manned	Explorer	0.015 to 0.050	Non-Nuclear	Nuclear	Same as mission No. 10	Same as mission No. 10	
37 [illegible]manned	Asteroid Jupiter	0.050	Nuclear	Non-Nuclear	Space disposal (type of mission yields in space decay of fuel before reentry) the system may be designed for intact or burnup depending upon spacecraft requirements	Same as mission No. 10	
38 [illegible]manned	Jupiter Flyby	1.0 to 1.2	Nuclear	Non-Nuclear	Space disposal, intact reentry design based on fuel inventory; however, a burnup design may be necessary due to high reentry velocities (Jupiter escape velocity 200,000 ft/sec)	Same as mission No. 10	
39 [illegible]manned	Mercury Flyby	0.40	Non-Nuclear	Nuclear	Same as mission No. 37	Same as mission No. 10	1[illegible]
40 [illegible]manned	Mariner Mars Flyby	0.150 to 0.250	Non-Nuclear	Nuclear	Same as mission No. 37	Same as mission No. 10	
41 Unmanned	San Marco	0.025	Nuclear	Non-Nuclear	Space disposal (either an intact or burnup design criteria may be considered based upon planned experiments & spacecraft requirements) reentry burnup with high altitude dispersion would give highest specific power	Same as mission No. 10	
42 Manned	Interplanetary Module	10.0 to 20.0	Nuclear	----	Intact controlled reentry with recovery	Same as mission No. 2	
43 Unmanned	Voyager	0.60	Nuclear	Non-Nuclear	Same as mission No. 37	Same as mission No. 10	1975

Space Missions, Associated Power Systems, and Estimated Aerospace Nuclear Safety Criteria

Table XIX (Cont'd)

Mission		Power System Selected			Aerospace Nuclear Safety Criteria		
No. Manned or Unmanned	Name	Power Level Kw(e)	First Choice	Second Choice	Primary	Backup	Time Periods
44 Unmanned	ABL	1.0	Non-Nuclear	Nuclear	Same as mission No. 37	Same as mission No. 10	1975
45 Unmanned	Venus/ Mars Recon.	1.0	Nuclear	Non-Nuclear	Same as mission No. 37	Same as mission No. 10	
46 Unmanned	Mars Landing	1.0	Nuclear	Non-Nuclear	Intact reentry design criteria; the source will undergo decay on the Martian surfact	Same as mission No. 10	
47 Manned	Manned Mars Mission	5.0 to 30.0	Nuclear	----	Intact controlled reentry with recovery	Same as mission No. 2	
48 Unmanned	Ext. Pioneer	0.50	Nuclear	Non-Nuclear	Space disposal, reentry burnup would give the lightest system; however, inventory requirements or experiments and/or on board equipment may dictate intact reentry	Same as mission No. 10	
49 Unmanned	Advanced Pioneer	0.50	Nuclear	Non-Nuclear	Same as mission No. 48	Same as mission No. 10	
50 Unmanned	Out of Ecliptic Probe	0.50	Nuclear	Non-Nuclear	Same as mission No. 48	Same as mission No. 10	
51 Unmanned	Interstellar	0.50	Nuclear	Non-Nuclear	Same as mission No. 48	Same as mission No. 10	
52 Unmanned	Comet Intercept	0.030	Nuclear	Non-Nuclear	Space disposal (the system may be designed for either intact or burnup depending upon the type of measurements and spacecraft limitations)	Same as mission No. 10	1980

Reference Table VII

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VI. REFERENCES

1. NASA TMX-SN 14, "Selection of Radioisotopes for Space Power Supplies", July, 1965.

2. Space/Aeronautics, No. 1, Vol 43, Jan. 1965, "A Review of the National Aerospace Program in Perspective".

3. Aviation Week and Space Technology, Vol. 82, No. 11, "32nd Inventory of Aerospace Power", Mar. 15, 1965.

4. SNAP Study Report, Task 16, Non-nuclear Power Supplies, Sept. 1963.

5. HA-TR-139, "State of the Art of Thermoelectricity", 1964.

6. HIT-181, "Task II - A Study of the Economics and Consequences of Recovery of Radioisotope Fuel Forms from Space Systems", August 1965.

7. Stebbins, A. K., "HASP Second Special Report on High Altitude Sampling Program", DASA 539 B, August, 1961.

8. Eisenbud, M., "Environmental Radioactivity", McGraw-Hill, 1963.

9. "Maximum Permissible Body Burdens and Maximum Permissible Concentrations of Radionuclides in Air and in Water for Occupational Exposure", (NCRP), NBS Handbook 69, 1959.

10. Guthrie, C. E. and Nichols, J. P., "Theoretical Possibilities and Consequences of Major Accidents in U-233 and Pu-239 Fuel Fabrication and Radioisotope Processing Plants", ORNL-3441, April, 1964.

11. A. F. Bacher, "Health and Safety Laboratory Symposium on Occupational Health Experience and Practices in the Uranium Industry", New York City, October 15-17, 1958, HASL-58, p. 155, September, 1959.

12. F. L. Cullen, et. al., "Chemical Technology Division Annual Progress Report for Period Engind May 31, 1961", ORNL-3153, p. 57.

13. Glasstone, "The Effects of Nuclear Weapons", June 1957, pp. 440-441.

14. "SNAP-9A Radioisotope-Fueled Generator Final Safety Analysis for Transit Mission", MND-P-2275-1, August, 1963 (CRD).

15. "Evaluation of the Radiological Hazards due to Reentry of SNAP APU's" NAA-SR-5482, F. D. Anderson, January 1961 (SRD).

16. "Reentry of Nuclear Rockets into the Earth's Atmosphere", R. I. Weiner AEC R&D Report, TID-7653, August, 1962 (CRD).

17. "The Effects of Various Aerospace Safety Philosophies on the Design of Advanced Nuclear Electric Space Power Plants", by R. I. Weiner,

[redacted]

[redacted]

REFERENCES (CONT'D)

AIAA Paper delivered at the Third Biennial Aerospace Power System Conference, Philadelphia, Pa., September 2, 1964, (SRD).

18. "An Investigation of the Reentry Destruction Potential of Various Radioisotope Fuel Forms", R. I. Weiner, Hittman Associates, Inc., Technical Report HIT-160, April 21, 1965, (CRD).

19. Quarterly Report - CAL-I08100-1, "Dispersal of Reactor Core Materials by Reentry Processes", J. L. Beal, et al, May 1963.

20. RFP for a 25-Watt(e) Power Supply, SNAP 25, USAEC, NYOO, February 2, 1965.

21. AI-65-6, Vol. II - Radioisotope Dynamic Electrical Power for the Manned Mars/Venus Mission.

22. "Flight Regimes of Mars and Venus", P. P. Wagner, Rand Corp., 1962.

23. "Model of the Venus Atmosphere", L. D. Kaplan, Jet Propulsion Lab. Rpt - CIT-1963.

24. "Maneuver Data May Limit Voyager Payload", Av. Week 8/2/65.

25. HA-139P, Proposal for Preliminary Design and Safety Study of Large Radioisotope Heat Source Subsystem for Space Power Supply, 1/65.

26. 64SD5033, 40 Watt Pu-238 Power Supply Engineering Study Final Report, Phase 1, 1/15/65.

27. AI-65-144 "Large Radioisotope Heat Source Study, Mid-Term Report to NYOO of USAEC, 8/20/65.

28. "Handbook of Astronautical Engineering", Koelle, McGraw-Hill, 1963.

29. "A Radioisotope T/E Power Supply for the Surveyor Lunar Roving Vehicle", RCA Report AEDR-2496, October 1964.

30. "Conduction Heat Transfer", P. J. Schneider, Addison Wesley, 1955.

31. "Structures Technology, 1964", R. A. Anderson, Astronautics and Aeronautics, December 1964.

32. Chapman, Dean R., "Tables of Z Functions for Atmosphere Entry Analysis", NASA TR R-106, 1961.

[redacted]

[redacted]

REFERENCES (CONT'D)

33. Chapman, Dean R., "An Approximate Analytical Method for Studying Entry Into Planetary Atmospheres", NASA TR R-11, 1959.

34. Hankey, Wilbur L., Jr., "Optimization of Lifting Reentry Vehicles", ASK-TDR-62-1102, U. S. Air Force, March 1963.

35. Eggers, Alfred J., Jr., "The Possibility of a Safe Landing", Space Technology, edited by Howard S. Siefert, Wiley and Sons, N. Y., 1959.

36. Johnston, Patrick J., Snyder, Curtis K., and Witcofshi, Robert D., "Maximum Lift-Drag Ratios of Delta-Wing-Half-Cone Combinations at a Mach Number of 20 in Helium", NASA TN D-2762, May 1965.

37. Gazley, Carl, Jr., "Atmospheric Entry", Handbook of Astronautical Engineering, edited by Heinz Hermann Koelle, McGraw-Hill, 1961.

38. Nolan, E. J and Scala, S M., "The Aerothermodynamic Behavior of Pyrolytic Graphite During Sustained Hypersonic Flight", General Electric Space Sciences Laboratory Report No. R61SD051, March 1961.

39. HIT-168, "Engineering Analysis of a Proposed Thermoelectric Generator for the Apollo Lunar Instrument Package".

40. E. P. Blizard, et al., Proceedings of the Symposium on the Protection Against Radiation Hazards in Space, TID-7652 (1962)

41. Radiation Effects on Nimbus B, General Electric Missile and Space Vehicle Department Report No. 4185-016.

42. E. D. Arnold, Handbook of Shielding Requirements and Radiation Characteristics of Isotopic Power Sources for Terrestrial, Marine, and Space Applications, ORNL-3576 (1964).

43. NACA CR-247, "The Rotornet Decelerator", 1965.

44. HIT-181, "A Study of the Economics and Consequences of Recovery of Radioisotope Fuel Forms From Space Systems", 8/65.

45. Space Technology, 1959.

46. "Space/Aeronautics", Tech. Handbook, 1964/65, pp. 39-53.

47. "Astronautics and Aerospace Engineering", August 1963, pp. 88-105.

48. "Astronautics and Aeronautics", January 1965, pp. 30-51.

49. Some Requirements of Electric and Chemical Thrust Systems for Space Station Drag Cancellation, NASA-TN-D-1835.

[redacted]

[redacted]

REFERENCES (CONT'D)

50. Hesse and Mumferd, Jet Propulsion, p. 529.

51. Unmanned ElectricPropulsion Mission Capabilities of the SNAP 50/ Spur Power Plant, PWAC-447, p. 19.

52. Earth Orbital Satellite Lifetime, NASA-TND-1995.

53. Koelle, Handbook of Astronautical Engineering, 1961.

54. K. Fuchs, "Efficiency for Very Slow Assembly", LA-596 (1946).

55. R. Scolettar, "The Fuchs-Nordheim Model with Variable Heat Capacity", Nuc. Sci. and Eng. 16, 459 (1963).

56. L. Moss and T. Wilson, "SNAP Reactor Transient Tests, Proceedings Aerospace Nuclear Safety", SC-DC-3553, p. 65 (1963).

57. L. D P. King, "Description of KIWI-TNT Excursion," Trans. Am. Nuc. Soc. 8, No. 1, 126 (1965).

[redacted]

[redacted]

APPENDIX A

MESOSPHERIC FUEL RELEASE FALLOUT ANALYSIS*

1. Derivation and Solution of Fallout and Deposition Equations

The basic activity removal relationship is:

$$\frac{dA\ (t)}{dt} = -\ kA\ (t) \qquad \text{(A-1)}$$

where: $\frac{dA\ (t)}{dt}$ = activity depletion rate

$A\ (t)$ = regional activity burden at time t

k = proportionality constant = removal rate

$1/k$ = mean residence time

$0.693/k$ = half residence time

The solution to Equation (A-1) is:

$$A\ (t) = A\ (O)\ \exp\ (-kt) \qquad \text{(A-2)}$$

The activity equations for the three regions and the earth's surface are as follows:

a. Mesosphere

rate of change of activity in mesosphere = rate of loss of activity to stratosphere + rate of loss of activity due to radioactive decay

$$\frac{dA_1(t)}{dt} = -k_1 A_1(t) - \lambda A_1(t) \qquad \text{(A-3)}$$

where: λ = radioactive decay constant

The solution of Equation (A-3) is:

$$A_1(t) = A(O)\ \exp\ -(k_1 + \lambda)\ t \qquad \text{(A-4)}$$

b. Stratosphere

rate of change of activity in stratosphere = rate of gain of activity from mesosphere - rate of loss of activity to troposphere - rate of loss of activity due to radioactive decay

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* Typical of Reference 1 techniques.

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$$\frac{dA_2(t)}{dt} = k_1A_1(t) - k_2A_2(t) - \lambda A_2(t) \qquad \text{(A-5)}$$

The solution of Equation (A-t) is:

$$A_2(t) = \frac{k_1A(O)}{k_2 - k_1} \left(\exp\left[-(k_1 + \lambda)t\right] - \exp\left[-(k_2 + \lambda)t\right]\right) \qquad \text{(A-6)}$$

c. Troposphere

rate of change of activity in troposphere = rate of gain of activity from stratosphere - rate of loss of activity to the earth's surface - rate of loss of activity due to radioactive decay

$$\frac{dA_3(t)}{dt} = k_2A_2(t) - k_3A_3(t) - \lambda A_3(t) \qquad \text{(A-7)}$$

$$= \frac{k_1k_2A(O)}{k_2 - k_1} \left(\exp\left[-(k_1 + \lambda)t\right] - \exp\left[-(k_2 + \lambda)t\right]\right) - (k_3 + \lambda)A_3(t)$$

let $\left[k_1k_2A(O)/k_2 - k_1\right] = C_1$

Then: $A_3(t) = \frac{C_1}{k_3 - k_1} \exp\left[-(k_1 + \lambda)t\right] - \frac{C_1}{k_3 - k_2} \exp\left[-(k_2 + \lambda)t\right] + I_1 \exp\left[-k_3 + \lambda)t\right]$ (A-8)

let $\frac{C_1}{k_3 - k_1} = B_1$; $\frac{C_1}{k_3 - k_2} = B_2$; $I_1 = B_3$

Then: $A_3(t) = B_1 \exp\left[-(k_1 + \lambda)t\right] - B_2 \exp\left[-(k_2 + \lambda)t\right] + B_3 \exp\left[-(k_3 + \lambda)t\right]$ (A-9)

d. Earth's Surface

rate of change of activity deposited on earth's surface = rate of gain of activity from troposphere - rate of loss of activity due to radioactive decay

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$$\frac{dA_4(t)}{dt} = k_3A_3(t) - \lambda A_4(t) \qquad \text{(A-10)}$$

$$= k_3B_1 \exp\left[-(k_1 + \lambda)t\right] - k_3B_2 \exp\left[-(k_2 + \lambda)t\right]$$

$$+ k_3B_3 \exp\left[-(k_3 + \lambda)t\right] - \lambda A_4(t)$$

The solution of Equation (A-10) is:

$$A_4(t) = -\frac{k_3B_1}{k_1} \exp\left[-(k_1 + \lambda)t\right] + \frac{k_3B_2}{k_2} \exp\left[-(k_2 + \lambda)t\right]$$

$$- B_3 \exp\left[-(k_2 + \lambda)t\right] + I_2 \exp(-\lambda t) \qquad \text{(A-11)}$$

where: I_2 = integration constant

$$= \frac{k_3B_1}{k_1} - \frac{k_3B_2}{k_2} + B_3$$

let: $\frac{k_3B_1}{k_1} = D_1$; $\frac{k_3B_2}{k_2} = D_2$; $B_3 = D_3$; $I_2 = D_4$

then: $A_4(t) = -D_1 \exp\left[-(k_1 + \lambda)t\right] + D_2 \exp\left[-(k_2 + \lambda)t\right]$

$$- D_3 \exp\left[-(k_3 + \lambda)t\right] + D_4 \exp(-\lambda t) \qquad \text{(A-12)}$$

2. Calculation of Tropospheric Volume and Earth Surface Area Between 20° and 70° Latitude

Area of curved surface of spherical segment is

$A = 2\pi rh$

where r = earth radius = 4×10^3 miles

h = height between latitudes

[redacted]

a. Earth Surface Area

$$A = 2\pi rh$$

$$r \doteq (4 \times 10^3 \text{ mi}) \frac{(1.6 \times 10^3 \text{ m})}{\text{mi}} = 6.4 \times 10^6 \text{ m}$$

$$h = r \sin 70^\circ - r \sin 20^\circ$$

$$A = 2\pi r^2 (\sin 70^\circ - \sin 20^\circ)$$

$$= (6.28)(0.41 \times 10^{14} \text{ m}^2)(0.940 - 0.342)$$

$$= (6.28)(0.41 \times 10^{14} \text{ m}^2)(0.598)$$

$A \doteq 1.54 \times 10^{14}$ m^2 (Earth surface area betwen 20° and 70° latitude)

b. Tropospheric Volume

Assuming the thickness of the troposphere to be 10,700 meters, the volume of the troposphere between 20° and 70° is

$$V = (A)(1.07 \times 10^4) \text{ m}^3 = (1.54 \times 10^{14} \text{ m}^2)(1.07 \times 10^4) \text{ m}^3$$

$V = 1.65 \times 10^{18}$ m^3 (Volume of troposphere between 20° and 70° latitude)

3. Sample Calculation of Maximum Tropospheric and Surface Contamination

a. Tropospheric Contamination, C_T

$$C_T \doteq \frac{(\text{total release})(\text{fallout constant-trop})(\%\text{ between } 20^\circ \text{ and } 70^\circ\text{lat})}{(\text{tropospheric volume between } 20^\circ \text{ and } 70^\circ \text{ lat})}$$

e.g., 1 Kw(th) of Pu-238

Total Release = 10^3 watts(th) x 30 curies/watt(th)

= 3×10^4 curies

$$C_T = \frac{(3 \times 10^4 \text{ curies})(6.08 \times 10^{-3})(0.80)}{(1.65 \times 10^{18} \text{ m}^3)} = 8.83 \times 10^{-17} \text{ curies/m}^3$$

= 8.83×10^{-7} uc/cc (tropospheric concentration)

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b. Surface Contamination, C_S

$$C_S = \frac{(\text{total release})(\text{fallout constant-surface})(\%\text{ between }20^\circ\text{ and }70^\circ\text{ lat})}{(\text{earth surface area between }20^\circ\text{ and }70^\circ\text{ lat})}$$

e.g., 1 Kw(th) of Pu-238

$$C_S = \frac{(3 \times 10^4 \text{ curies})(7.39 \times 10^{-1})(0.80)}{(1.54 \times 10^{14} \text{ m}^2)} = 11.45 \times 10^{-11} \text{ curies/m}^2$$

$$= 1.145 \times 10^{-10} \text{ curies/m}^2 \text{ (surface contamination)}$$

Reference 1. "SNAP-9A Final Safety Report", MND-P-2775-2, T. J. Dobry, March 1963, SRD.

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APPENDIX B

STRATOSPHERIC FUEL RELEASE FALLOUT ANALYSIS

1. Derivation and Solution of Fallout and Deposition Equations

The degree of hazard associated with a low altitude (stratosphere) injection of radioactive material is obviously greater than that associated with a mesospheric release. Calculations similar to those shown in the first part of Appendix A were conducted with the following assumptions:

(1) One micron sized particles
(2) Complete worldwide dispersion
(3) Mean residence time of two years in the stratosphere (40, 000 - 160, 000 feet)

It should be pointed out that the assumption of worldwide dispersion is not necessarily correct because it is conceivable that an activity release in the lower region of the stratosphere (around the 40, 000 feet limit) may result in a partial dispersion over a limited surface area.

The necessary equations and their solutions are as follows:

a. Stratosphere (1)

rate of change of activity in the stratosphere = rate of activity loss to troposphere + rate of activity loss due to decay

$$\frac{dA\ (t)}{dt} = -k_1 A_1 (t) - \lambda A_1 (t) \qquad \text{(B-1)}$$

where:

$\frac{dA_1(t)}{dt}$ = depletion rate of activity in stratosphere
k_1 = proportionality constant
$1/k_1$ = mean residence time for stratosphere
$A_1(t)$ = activity burden at time t
λ = decay constant of material

Solution of Equation (B-1) is of the form:

$$A_1(t) = A(O) \exp \left[- (k_1 + \lambda) t \right] \qquad \text{(B-2)}$$

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b. Troposphere (2)

rate of activity change in the troposphere = rate of activity gain from the stratosphere + rate of activity loss to the earth's surface + rate of activity loss due to decay

$$\frac{dA_2(t)}{dt} = k_1 A_1(t) - k_2 A_2(t) - \lambda A_a(t) \qquad \text{(B-3)}$$

Solution of Equation (B-3) is of the form:

$$A_2(t) = \frac{k_2 A(O)}{k_2 - k_1} \left(\exp\left[-(k_1 + \lambda)t\right] - \exp\left[-(k_2 + \lambda)t\right]\right. \qquad \text{(B-4)}$$

c. Earth's Surface (3)

rate of activity change on the earth's surface = rate of activity gain from the troposphere + rate of activity loss due to decay

$$\frac{dA_3(t)}{dt} = k_2 A_2(t) - \lambda A_3(t) \qquad \text{(B-5)}$$

Solution of Equation (B-5) is of the form:

$$A_3(t) = \frac{A(O)}{k_2 - k_1} \left\{ k_1 \exp\left[-(k_2 + \lambda)t\right] - k_2 \exp\left[-(k_1 + \lambda)t\right] + (k_2 - k_1)\exp(-\lambda t) \right\} \qquad \text{(B-6)}$$

Using the same values for earth surface area and tropospheric volume between 20° and 70° latitude as given in Appendix A, Nuclear Safety Indices can be found for stratospheric releases. The procedure for carrying out these calculations is comparable to that use for a mesospheric release.

As expected, results shown in the text indicate a substantially higher degree of hazard associated with stratospheric release than with a mesospheric release.

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APPENDIX C

DISPERSAL AND DOSE RATE OF REENTERED MICROSPHERES

1. Fallout Patterns of Dispersed Microspheres

On the basis of information on particle size given in Reference 1 proposal, a typical 75 w(e) generator fuel loading will consist of about 245 million PuO_2 microspheres ranging in diameter from about 100 to 220 microns. Upon release at some specified altitude the particles will first be decelerated and then will descend through the atmosphere at rates which are dependent on the particle density, size, and relative vertical movement of the portions of the atmosphere through which they are falling. The particles will be displaced horizontally by prevailing winds as they fall. The smallest particles, which take the longest time to reach the ground, will be displaced a maximum distance from the initial point of deceleration: the largest particles, which fall at the maximum rate, are deposited a minimum distance from the starting point. Thus, the particles will be distributed on the ground over a considerable distance in the prevailing wind direction solely by virtue of their variation in diameter.

Dispersion normal to the prevailing wind direction will result from atmospheric turbulence and convection, but the extent of dispersion from this source is difficult to predict. However, observed fallout patterns of radioactivity following an atomic bomb detonation provide a guide to the extent of lateral dispersion (Reference 2).

In estimating the fallout pattern of PuO_2 microspheres, the size distribution was assumed to be similar to that given in Figure 14 from Reference 1. All the particles are assumed to fall free, with negligible size attenuation due to subsequent aerodynamic heating. The latter assumption is considered reasonable in light of studies of PuO_2 particle burnup which have been performed for Union Carbide Corporation (Oak Ridge National Laboratory) by Hittman Associates (Subcontract No. 2433).

In the process of deceleration prior to fallout the smallest particles will slow down first and the heaviest particles last. Hence, at the start of fallout the smallest particles are displaced behind the largest particles by a distance of perhaps several miles or more. As fallout occurs in the direction of the prevailing wind, the smallest particles, which have the lowest fallout rates, will tend to catch up with and move ahead of the larger particles which have already reached the ground. The net displacement of the smallest particles in front of the largest in the direction of the prevailing wind is then less than would be the case if all the particles started to fallout at the same time.

Fallout was assumed to start from an altitude of 200,000 feet. Since winds may vary widely in direction and magnitude at different altitudes, accurate prediction of downwind fallout is not possible. A reasonable indication of the fallout pattern to be expected following an atomic bomb detonation

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can be obtained by assuming a mean velocity of 15 mph (Reference 2). This figure was also used to study PuO_2 microsphere fallout.

The vertical particle velocity during fallout can be expressed by:

$$V = 0.35\Theta\, d^2 \rho \tag{C-1}$$

where: V = the fallout velocity in ft/hr
d = the particle diameter in microns
ρ = the particle density in gm/cc
Θ = ≤ 1 is a correction factor for non-Stokesian flow.

Using a PuO_2 density of 9.7 gm/cc, Equation (C-1) becomes

$$V = 3.4\,\overline{\Theta}\, d^2 \tag{C-2}$$

The correction factor Θ is a function of the particle Reynolds number Re, defined as

$$Re = \frac{V_s d}{v} = \frac{3.4\, d^3}{v} \tag{C-3}$$

where: V_s = the fallout velocity for $\Theta = 1$
v = the kinematic viscosity of air

Since v is a function of altitude, the correction factor varies with both size and altitude. Values of Θ were obtained from p. 1020 of "Chemical Engineering Handbook", Perry, J. H., (ed), 3rd edition, McGraw-Hill, N. Y., 1950. In Table C-1 Θ was given as a function of altitude for three representative particle sizes, and the mean value of Θ for an altitude of 200,000 feet is shown.

The correction factor is seen to be greatest for the largest particle. The average fallout velocity is obtained by using $\overline{\Theta}$ in Equation (C-2) for the appropriate diameter particle. The fallout time is then obtained from

$$t_{(hr)} = \frac{h\ (ft)}{V\ (ft/hr)} = \frac{200,000}{3.4\,\overline{\Theta}\, d^2} \tag{C-4}$$

and the horizontal displacement S in a 15 mph downwind is

$$S\ (mi) = 15\ t \tag{C-5}$$

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Table C-1

CORRECTION FACTOR θ NON-STOKESIAN FLOW DURING FALLOUT

	Particle Diameter - microns		
Altitude	100	160	220
200	1	1	1
180	1	1	1
160	1	1	1
140	1	1	0.97
120	1	0.98	0.90
100	1	0.90	0.78
80	0.92	0.78	0.66
60	0.82	0.66	0.54
40	0.72	0.54	0.42
20	0.63	0.46	0.34
0	0.57	0.40	0.30
$\bar{\theta}$	0.92	0.83	0.75

The fallout velocity, fallout time, and horizontal displacement are shown in Table C-2 for three particle sizes.

Table C-2

DISPERSION CHARACTERISTICS OF PuO_2 MICROSPHERES IN 15 MPH WIND

Diameter (microns)	Mean Fallout Velocity (ft/hr)	Fallout Time (hours)	Horizontal Displacement (miles)
100	3.11×10^4	6.44	96.6
160	7.20×10^4	2.78	41.7
220	12.32×10^4	1.62	24.3

The 100 and 220 micron particles are separated after deposition by 96.6 - 24.3 = 72.3 miles. Since these figures are representative of the minimum and maximum particle diameters, dispersion of the particles over a distance of about 70 miles in the direction of the wind is indicated. The dispersion distance varies directly as the initial altitude and the mean wind velocity. The extent of lateral dispersion is estimated to be about 1/10 that in the prevailing wind direction on the basis of atomic bomb fallout data, or about 7 miles. The

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area over which the microspheres are deposited is then about 500 square miles, with an average spacing of about 7.5 feet. The dispersion is by no means uniform, and can be estimated from the fraction of total particles in each size range. More important is the concentration of radioactivity, which is proportional to the relative volume of fuel at each location.

Table C-3 shows the ratio of maximum to average particle concentration and maximum to average radioactivity concentration and their relative downwind locations.

Table C-3

RELATIVE PARTICLE AND RADIOACTIVITY CONCENTRATIONS

Item	Downwind Locations
Maximum Particle Concentration	52 miles
Average Particle Concentration	38 miles, 69 miles
Maximum Radioactivity Concentration	42 miles
Average Radioactivity Concentration	38 miles, 67 miles

The maximum concentration of particles or radioactivity is seen to be about 2.8 times the average value. The concentration is in excess of the average value over a distance of 69 - 38 = 31 miles in the downwind direction.

2. Dose Rate From Dispersed Microspheres

The dose rate can be estimated on the assumption that the deposited particles represent an infinite plane source. The radiation flux at 3 to 6 feet from such a source is approximately equal to the source strength. PuO_2 emits both neutron and gamma rays. The estimated average neutron source strength is 1.175×10^{-5} n/cm^2-sec, of $\sim$1 Mev neutrons; the estimated average gamma source strength is 1.34×10^{-3} Mev/cm^2-sec of 0.15 Mev gammas and 5.97×10^{-5} Mev/cm^2-sec of 1 Mev gammas (Reference 3). Using a neutron conversion factor of 10 n/cm^2-sec = 1 mr/hr and a gamma conversion factor of 580 Mev/cm^2-sec = 1 mr/hr (Reference 4), the estimated average neutron and gamma doses are:

Average Neutron Dose	1.18×10^{-6} mr/hr
Average Gamma Dose	2.33×10^{-6} mr/hr
Total Average Radiation Dose	3.5×10^{-6} mr/hr
Total Maximum Radiation Dose	10^{-5} mr/hr

[redacted]

The significance of these numbers becomes apparent in comparison with the minimum dose rate produced from natural sources of radioactivity, which is on the order of 10^{-2} mr/hr (Reference 2). Thus, the maximum dose rate from dispersed microspheres is calculated to be 1000 times less than that from natural background radiation. Even allowing for unfavorable meteorological conditions, variations in the release altitude, and localized hot spots, the external radiation from deposited PuO_2 microspheres does not appear to represent a significant hazard.

The situation would, of course, be quite different if a substantial fraction of the microspheres were bonded together as the result of self-sintering. Then a significant external radiation shield would exist within about 50 feet of the deposited PuO_2. Dose rates would be roughly 200 f $(3/D)^2$ mr/hr, where f is the fraction of the original fuel loading which lands intact as a unit and D is the separation distance in feet. For example, if a single spherical fuel capsule (out of a total of 19) were to reenter intact, the dose rate at 3 feet would be about 11 mr/hr.

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REFERENCES

1 AEC SCO 5-025-(09), "Proposal for a 75 Watt(e) Power Supply - SNAP-25", CRD.

Glasstone, S. (ed.), "The Effects of Nuclear Weapons," AEC, Washington, June 1957.

AEC SCO 4-147, "A Radioisotope Thermoelectric Power Supply for the Surveyor Lunar Roving Vehicle", Astro-Electronics Division, RCA, Princeton, N. J., October 1964, CRD.

4. Rockwell, T. (ed.), Reactor Shielding Design Manual, McGraw-Hill, New York, 1956.

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APPENDIX D

COMPUTATION OF RANGE, MANEUVER TIME AND STAGNATION POINT HEATING RATE FOR LIFTING BODY - SEMI-CONTROLLED REENTRY

Nomenclature

Symbol	Description	Units
a	parameter = $1/2\ L/D \sin \phi$	--
A	reference area	ft^2
C_D	drag coefficient = $\frac{D}{1/2\ \rho A V^2}$	--
C_L	lift coefficient = $\frac{L}{1/2\ \rho A V^2}$	--
D	drag force	lb
g	acceleration of gravity	ft/sec^2
h	altitude	ft
H	effective heat absorption	Btu/lb
L	lift force	lb
L/D	lift to drag ratio	--
m	mass	slugs
$\dot{q}$	heating rate	Btu/ft^2-sec
Q	total convective heat absorbed	Btu
$\overline{Q}$	dimensionless heat absorbed	--
R	radius of curvature or range	ft, stat. miles
R_E	radius of the earth	ft
t	time	sec
V	velocity	ft/sec
W	weight	lb
X	dimensionless velocity = V/V_i or longitudinal coordinate	ft
y	lateral coordinate	ft
γ	flight path angle	radians or degrees
ρ	density	$slugs/ft^3$
ϕ	roll angle	radians or degrees
ψ	heading angle	radians or degrees

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Nomenclature (Cont'd)

Subscripts

c	circular or cooling system
i	initial
s	stagnation
x	longitudinal direction
y	lateral direction
max	maximum
opt	optimum

A. Range

The equations of motion for a banked, equilibrium glide are:

$$m\dot{V} = -D - mg \sin\gamma \tag{D-1}$$

$$mV\dot{\gamma} = L \cos \phi + \frac{mV^2 \cos\gamma}{R_E} - mg \cos\gamma \tag{D-2}$$

$$mV\dot{\psi} = \frac{L \sin \phi}{\cos\gamma} \tag{D-3}$$

For small flight path angles, $\gamma \approx 0$, $\dot{\gamma} = 0$, $\cos\gamma = 1$, $\sin\gamma = 0$, hence,

$$\frac{\dot{V}}{g} = -\frac{D}{W} \tag{D-4}$$

$$L \cos \phi = mg - \frac{mV^2}{R_E} \tag{D-5}$$

$$mV\dot{\psi} = L \sin \phi \tag{D-6}$$

The heading angle can be determined by integrating (D-3)

$$\psi = \int_{V_i}^{V} \dot{\psi} \, dt = \int_{V_i}^{V} \frac{\dot{\psi}}{\dot{V}} \, dV \tag{D-7}$$

[redacted]

which, after substituting (D-4) into (D-7), gives

$$\psi = -\int_{V_i}^{V} \frac{L}{D} \sin \phi \, \frac{dV}{V} \qquad \text{(D-8)}$$

For constant L/D and ϕ, (D-8) can be integrated to give

$$\psi = \frac{L}{D} \sin \phi \ln \frac{V_i}{V} \qquad \text{(D-9)}$$

The longitudinal range is given by

$$R_X = \int_{V_i}^{V} V \cos \psi \, d\, t = \int_{V_i}^{V} V \cos \psi \, \frac{dV}{\dot{V}} \qquad \text{(D-10)}$$

Substitution of $V_c^{\,2} = g\, R_E$ into (D-5) and combining (D-4), (D-5), (D-9) and (D-10) produces

$$\frac{R_X}{R_E} = \int_{0}^{\frac{V_i}{V_c}} \frac{L}{D} \cos \phi \left[\frac{\frac{V}{V_c}}{1 - \frac{V^2}{V_c^{\,2}}} \right] \cos \left(\frac{L}{D} \sin \phi \ln \frac{V_i}{V} \right) d \left(\frac{V}{V_c} \right) \qquad \text{(D-11)}$$

Maximum longitudinal range occurs when $\phi = 0$. Letting $X = V/V_c$

$$\left(\frac{R_X}{R_E} \right)_{max} = \frac{L}{D} \int_{0}^{X_i} \frac{X\, d\, X}{1 - X^2} = -1/2 \, \frac{L}{D} \ln \left(1 - \frac{V_i^{\,2}}{V_c^{\,2}} \right) \qquad \text{(D-12)}$$

Similarly, the expression for lateral range is

$$\frac{R_y}{R_E} = \int_{0}^{\frac{V_i}{V_c}} \frac{L}{D} \cos \phi \left[\frac{\frac{V}{V_c}}{1 - \frac{V^2}{V_c^{\,2}}} \right] \sin \left(\frac{L}{D} \sin \phi \ln \frac{V_i}{V_c} \right) d \left(\frac{V}{V_c} \right) \qquad \text{(D-13)}$$

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which for constants L/D and ϕ, and for $X_i = 1$ results in

$$\frac{R_y}{R_E} = \frac{L}{D} \cos \phi \int_o^1 \frac{X}{1 - X^2} \sin (- \frac{L}{D} \sin \phi \ln X \, d X \qquad \text{(D-14)}$$

This equation is integrated by transforming variables. Let

$y = -\ln X^2$

$a = 1/2 \frac{L}{D} \sin \phi$

Thus,

$$\frac{R_y}{R_E} = 1/2 \frac{L}{D} \cos \phi \int_o^{\infty} \frac{e^{-y} \sin (a\, y)}{1 - e^{-y}} \, dy \qquad \text{or,}$$

(D-15)

$$\frac{R_y}{R_E} = 1/2 \frac{L}{D} \cos \phi \int_o^{\infty} e^{-y} \sin (a\, y) \sum_{n = o}^{\infty} e \quad dy$$

Interchanging the order of integration,

$$\frac{R_y}{R_E} = 1/2 \frac{L}{D} \cos \phi \sum_{n = o}^{\infty} \int_o^{\infty} e^{- ny} \text{ in } (a\, y) \, dy \qquad \text{(D-16)}$$

$$= \cot \phi \sum_{n = 1}^{\infty} \frac{a^2}{a^2 + n^2}$$

In Reference (3), the series is approximated for $0 < a < 2$ to give the result

$$\frac{R_y}{R_E} = \frac{(\frac{L}{D})^2 \sin 2 \phi}{5.2 \left[1 + 0.106 \, (L/D)^2 \sin^2 \phi\right]}^{1/2} \qquad \text{(D-17)}$$

[redacted]

with

$$\left(\frac{R_y}{R_E}\right)_{max} = \frac{\left(\frac{L}{D}\right)^2}{5.2\sqrt{1 + 0.106\left(\frac{L}{D}\right)^2}} \qquad (D\text{-}18)$$

and the roll angle for maximum lateral range is given by

$$\cot^2 \phi_{opt} = 1 + 0.106\left(\frac{L}{D}\right)^2 \qquad (D\text{-}19)$$

Since (D-11) cannot be integrated when $V_i = V_c$, the initial condition is taken as an altitude of 400,000 feet. The initial velocity is then determined from

$$\frac{D}{W} = 1/2\ \rho_i\ A\ V_i^2\ \frac{C_D}{W} = \frac{1/2\ \rho_i\ V_i^2\ \left(\frac{L}{D}\right)}{\frac{W}{C_D A}} \qquad (D\text{-}20)$$

Substituting (D-20) into (D-5) gives

$$X_i = \left[\frac{1/2\ \rho_i\ V_c^2\ \left(\frac{L}{D}\cos\phi\right)}{\frac{W}{C_D A}} + 1\right]^{-1/2} \qquad (D\text{-}21)$$

Equations (D-12), (D-17), (D-18), and (D-21) were used directly to obtain the maximum longitudinal and lateral ranges and optimum roll angle for L/D's ranging from 0.5 to 3.0 and ballistic coefficients of 10, 100, and 500.

In order to obtain the longitudinal range at optimum roll angle, it was necessary to integrate Equation (D-11) numerically. In the process of the integration, it was found that in order to maintain sufficient accuracy for $X > 0.9$, the step sizes required were so small that hand integration was nearly impossible. To alleviate this problem, some simplifying assumptions had to be made.

It was assumed that because the density is so low at the initial altitude, the vehicle cannot develop a significant side force until it has descended lower into the atmosphere, hence the effect of side force on longitudinal range is small. For convenience in the calculations, this assumption was used for $X > 0.9$, and the longitudinal range covered in this portion of the trajectory was computed from Equation (D-12) after replacing L/D with L/D cos ϕ and integrating from $X = 0.9$ to $X = X_i$. In other words, it was assumed that for $X > 0.9$, the longitudinal range covered was the same as

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if the vehicle had an L/D equal to L/D cos ϕ and no side force. The longitudinal range covered in the remainder of the trajectory, $0 < X < 0.9$, was computed from Equation (D-11).

In order to check the assumption involved, Equation (D-16) was integrated from $X = X_i$ to $X = 0.9$ and the lateral range was computed and compared to the total lateral range and the longitudinal range for $X > 0.9$. The results for L/D's of 0.5 and 3.0 are shown in Table I, and give an indication of the amount of error involved.

Table I

Lateral Range for Various Ballistic Coefficients

L/D	Percent Total Lateral Range Covered for $X > 0.9$	Ratio of Lateral Range to Longitudinal Range for $W/C_DA = 10$	$W/C_DA = 100$	$W/C_DA = 500$
0.5	8.9	0.0039	0.0028	0.0024
3.0	14.9	0.0357	0.0218	0.0186

B. Maneuver Time

The results in Table I indicate that the lateral maneuvering capability of the vehicle is limited above $X > 0.9$, hence, the maneuvering time has been defined as the time required for descent from $X = 0.9$ to impact. These times are available directly from Chapman's machine calculations (Reference 1) which give time to impact as a function of L/D. Since Chapman's results have been computed for planar trajectories only (no lateral maneuvering) the L/D's in Reference 1 correspond to L/D cos ϕ in this analysis.

C. Stagnation Point Heating Rate

An expression for the maximum stagnation point heating rate has been derived in Reference 3 as

$$(\dot{q}_s)_{max} = 8.025 \left(\frac{W}{C_L AR} \right)^{1/2} \qquad \text{(D-22)}$$

which can be rearranged to provide an expression involving both L/D and W/C_DA.

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$$(\dot{q}_s)_{max}\sqrt{R} = \frac{8.025\left(\frac{W}{C_D A}\right)^{1/2}}{(L/D)^{1/2}}$$

This relation is valid for laminar flow and also assumes that the enthalpy at the surface stagnation point is much less than the enthalpy at the outer edge of the boundary layer.

D. Total Heat Absorbed During Reentry

Integrated heating rates are also available as a function of dimensionless total laminar heat absorbed, $\overline{Q}$, up to $X = 0.999$. Chapman suggests using

$$Q = K_2 \sqrt{\frac{m}{C_D AR}} \; \overline{Q}$$

where $K_2 = 15,900$ has been chosen such that the results agree with previous investigations.

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APPENDIX E

RELIABILITY OF ELECTRONIC COMPONENTS IN A NUCLEAR ENVIRONMENT

To approximate the sensitivity of the typical sensory and control system circuitry which would be inherent in the lifting body concept, a study was undertaken to evaluate the degradation of the most critical components.

Isotopic power sources for use in spacecraft require shielding or a degree of isolation to prevent radiation damage to instrument systems associated with the mission. For the total dosage received by an instrumented spacecraft, one must also consider the contribution of the space radiation environment.

Allowable radiation doses vary widely with the type of radiation. Maximum allowable doses of 10^7 rads of gamma radiation, 10^4 rads of fast neutrons (approximately $10^{12}\ \frac{\text{neutron)}}{\text{cm}^2}$ and 10^4 rads of solar protons have been chosen on the basis that these doses probably would not cause significant radiation damage in typical transitorized instrument systems, (Reference 40). Recent investigation by G. E. Spacecraft Department has revealed that surface effects on transistors can be induced by gamma exposure doses on the order of 10^4 roentgens (Figure E, taken from Reference 41). The 2N708 transistor used in the graph is commonly used in digital processing systems found in telemetry circuitry.

The power supply that must be shielded was chosen typically as a 2.5 kw(e)(16.7 kwt) isotopic dynamic system. Isotopes considered for this system were Po^{210} and Pu^{238}. Graphs of gamma exposure dose rate in units of milliroentgens/hr versus shield thickness for lead and uranium were obtained for 20 kw sources of Po^{210} and Pu^{238} (Figures E-1 and E-2). From these graphs, the graph of degradation of current gain versus time for the two sources was constructed(Figure E-3).

For a 10 percent degradation in current gain and a mission lifetime of 10,000 hours, it can be seen from Figure E-3 that no shielding is required for the 20,000 watt Pu^{238} source, and approximately 1 cm. of depleted uranium is required for the Po^{210} for the same power output. However, with the Pu^{238}, one must also consider the neutrons emitted by spontaneous fission. Assuming a disc source without shielding, the flux at a distance of 100 cm. was calculated to be $1.07 \times 10^3 \frac{\text{neutrons}}{\text{cm}^2\text{-sec}}$. For a mission lifetime

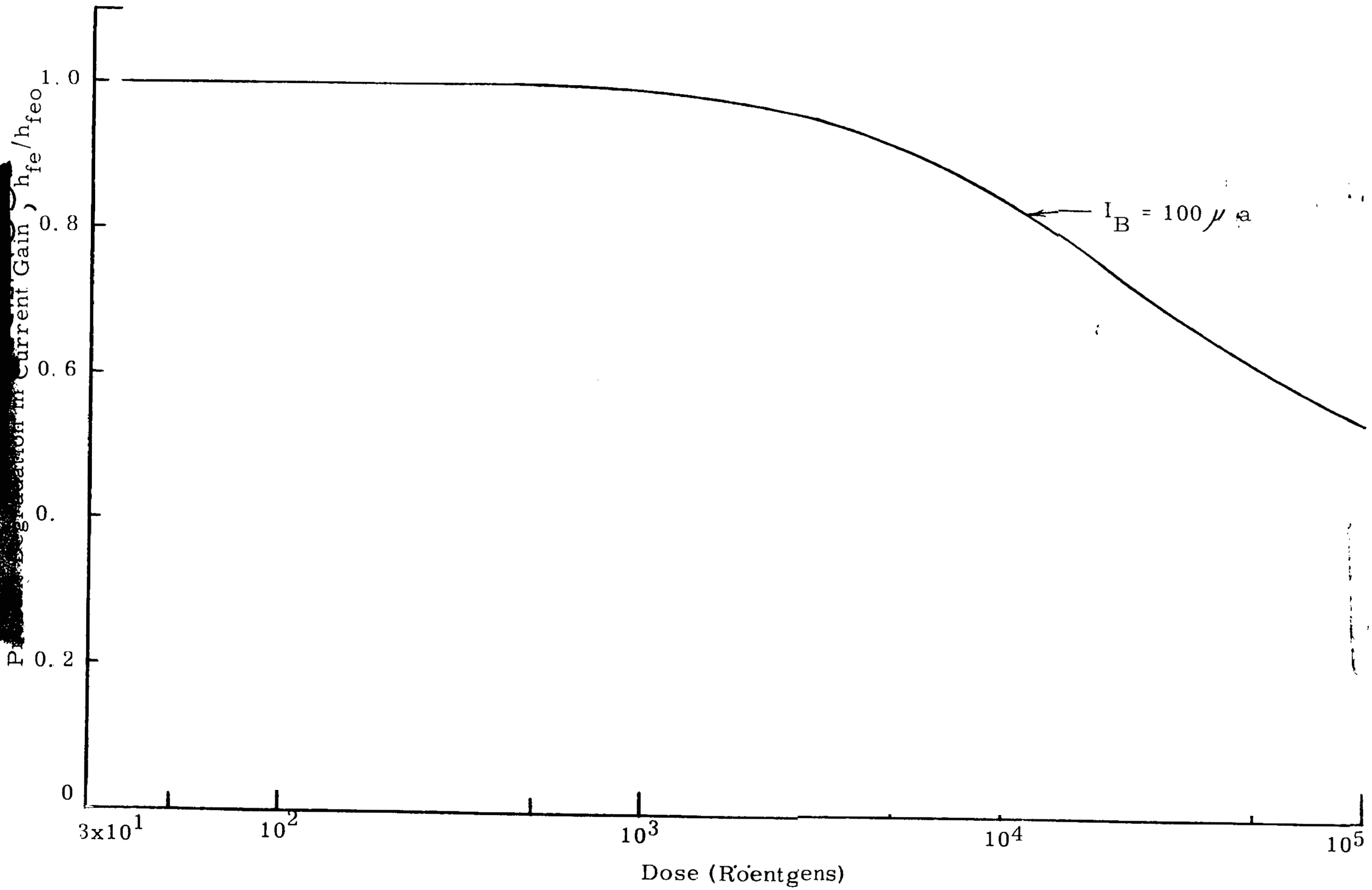

Figure E. Degradation of Common Emitter Current Gain Due to Surface Effects - 2N708

[redacted]

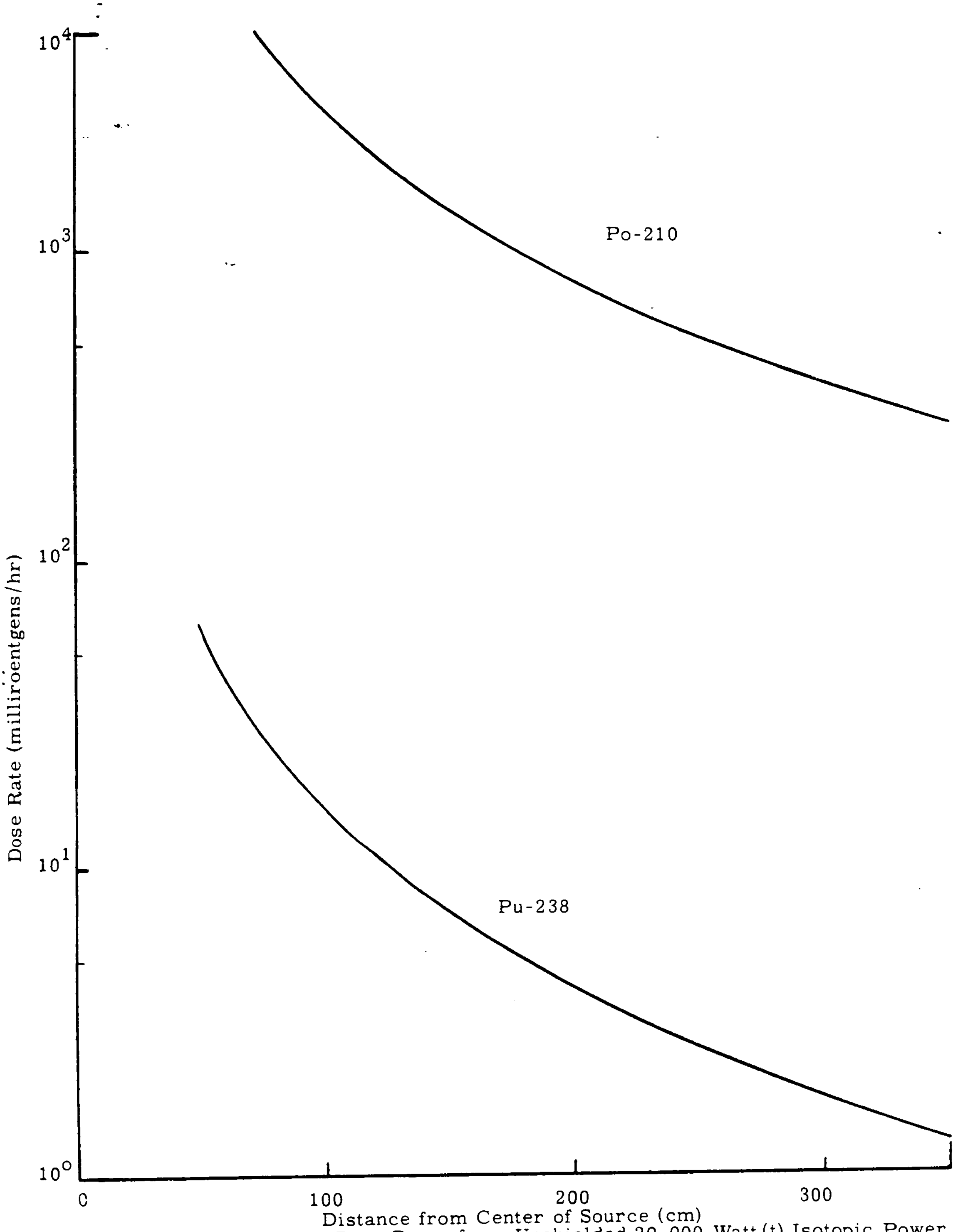

Figure E-1. Gamma Dose Rates from Unshielded 20,000 Watt(t) Isotopic Power Sources of Po-210 and Pu-238

[redacted]

[redacted]

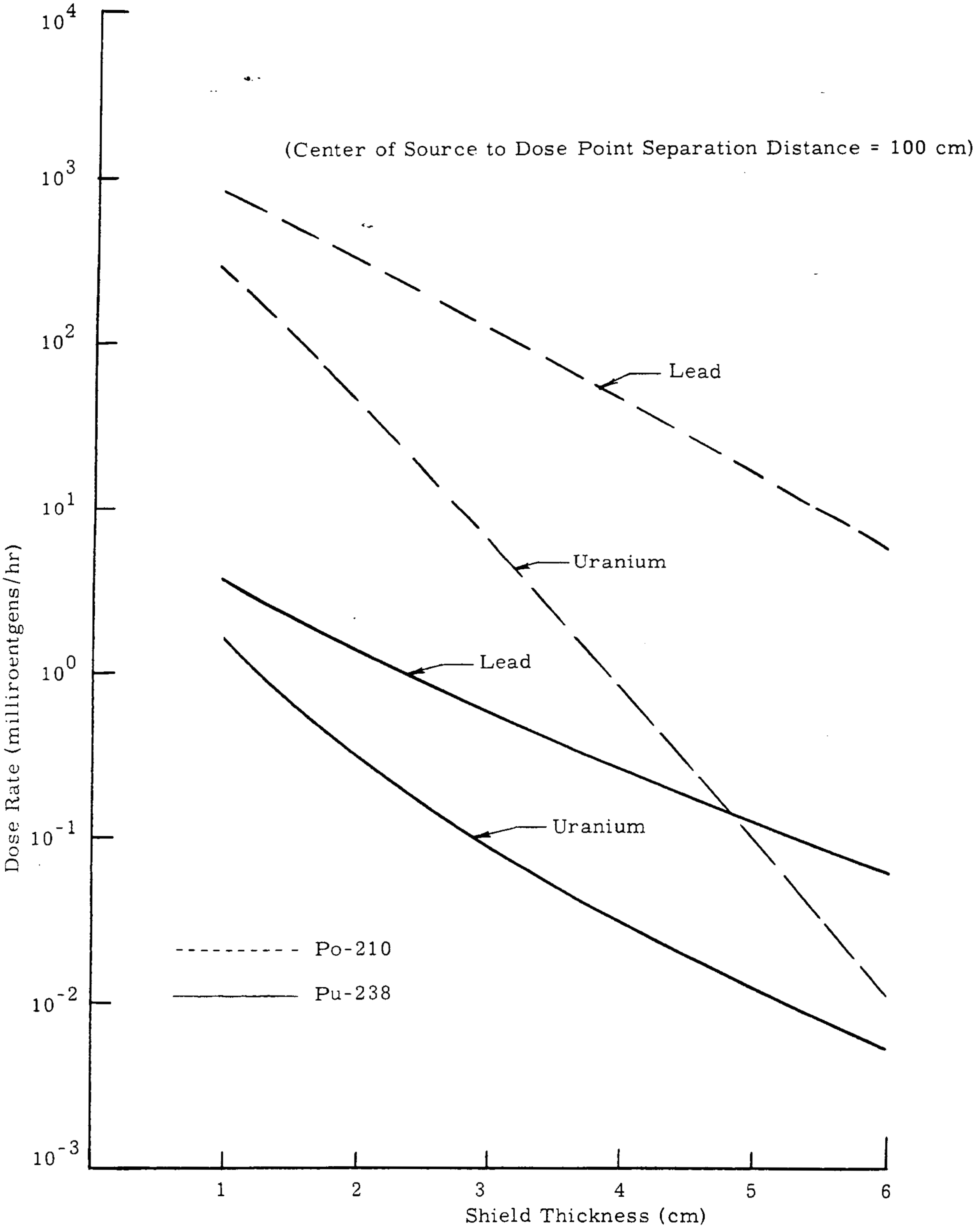

Figure E-2. Gamma Dose Rates from 20,000 Watt (t) Isotopic Power Sources of Po-210 and Pu-238 Shielded by Lead and Uranium

[redacted]

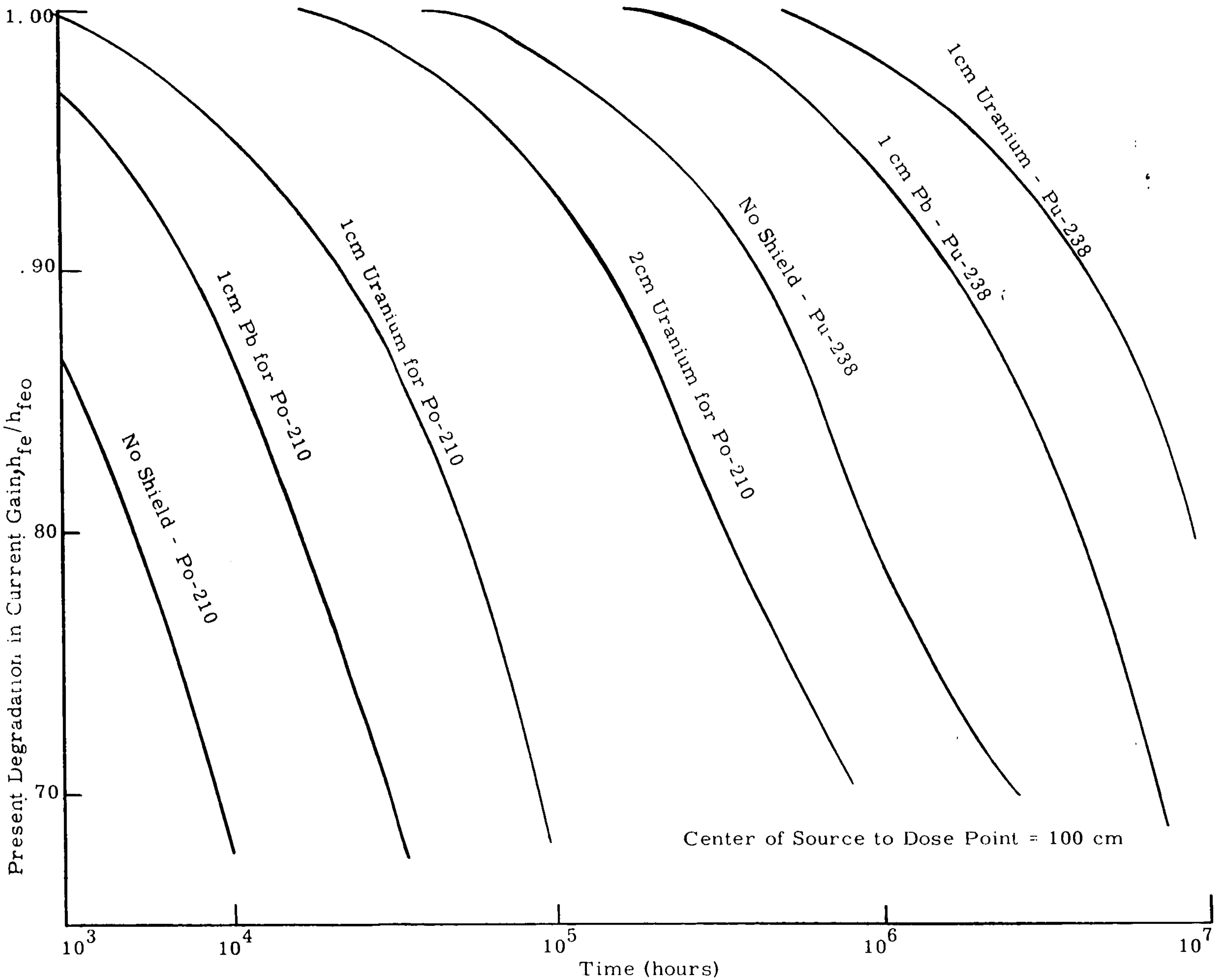

Figure Degradation of Current Gain - Versus Time for the 2N708 Transistor in the Field of a 2.5 Kw(e) Radioisotope Space APU

[redacted]

of 10,000 hours, the integrated flux is $3.85 \times 10^{10} \frac{\text{neutrons}}{\text{cm}^2}$. This value is approximately two orders of magnitude less than the maximum integrated flux given by Reference 40 and thus may be assumed to result in negligible degradation. Neutrons may also be emitted from both Po^{210} and Pu^{238} sources if they are mixed in a matrix or contain low atomic number impurities such as oxygen or beryllium that undergo an (α, n) reaction.

The determination of the space radiation environment is difficult because of the variation in radiation exposure with various orbits. Also, uncertainties exist in the residual electrons injected into the radiation belt by the July 9, 1962 high altitude nuclear test, "Starfish". Present estimates of the radiation dose that the Nimbus B satellite would receive in a 600 N. mile polar orbit are in the range of 10^4 to 10^5 rads/year for an effective spacecraft shielding of 1 to 2 $\frac{\text{cm}^2}{\text{gm}}$ of aluminum. For a 750 N. mile orbit, the doses would be 2 to 3 times higher.

Thus, it can be seen that the space radiation is significant when compared to the radiation from the isotopic power source and its effect must be considered in the design of the lifting body control and sensory equipment.

In general Figure E-3 shows no severe shielding problem for and Pu^{238} sources in a reentry body which can return the heat block of a 2.5 kwe nuclear power plant.

[redacted]

[redacted]

APPENDIX F

EFFECT OF BALLISTIC COEFFICIENT ON ORBITAL LIFETIME

Considering any given ballistic coefficient as

$$\frac{W}{C_D A} = C_B \qquad (1)$$

The weight is a function of the cube of the average radius of the debris

$$W = \frac{4}{3} \rho_m \pi (\bar{r})^3 \qquad (2)$$

while the drag area is a function of square of the average radius of debris

$$A = \pi (\bar{r})^2 \qquad (3)$$

Therefore in terms of the average radius

$$\frac{W}{C_D A} = \frac{4}{3} \frac{\rho_m (\bar{r})}{C_D} \qquad (4)$$

Since the density of the material and its drag coefficient in molecule flow as encountered under orbital conditions will remain constant as the size of the object is reduced; debris lifetime can be expressed approximately as a function of radius reduction in the following manner:

orbital lifetime

$$t_L = \left(\frac{W}{C_D A}\right) \left[\frac{(h_2 - h_1)}{\sqrt{g_o R}\ \rho_1 R_E \ln \left(\frac{\rho_1}{\rho_2}\right)} \times \left(\frac{\rho_1}{\rho_2} - 1\right) \right] \qquad (5)$$

where h_2 = initial altitude

h_1 = 400,000 ft reentry alt.

g_o = 32.2 ft/sec^2

R = Alt. of Sat. from center of Earth

ρ_1 = Density at h_1

ρ_2 = Density at h_2

[redacted]

[redacted]

Therefore, for the same decay conditions, i. e., average orbital altitude and reentry altitude;

$$\frac{t_{L1}}{t_{L2}} = \frac{\left(\frac{W}{C_D A}\right)_1}{\left(\frac{W}{C_D A_2}\right)} = \frac{r_1}{r_2} \tag{6}$$

which is the ratio of the radius of the debris to the radius of the original orbiting body.

Therefore, Figure 79 can be simply modified viz the radius ratio of original to final debris sizes. Hence, if an explosion could reduce a given device to one thousand pieces of debris:

$$\frac{t_{L1}}{t_{L2}} = \frac{r_1}{r_2} = (n)^{1/3} \tag{7}$$

Therefore, the orbital lifetime is reduced by a factor equivalent to the cube root of the number of particles. Hence in our example

$$\frac{t_{L1}}{t_{L2}} = 10 \tag{8}$$

[redacted]

[redacted]

APPENDIX F. 2

EFFECTS OF FRACTURE ON INCREASE OF SURFACE AREA

Consider a simple sphere

$$A_s = 4\pi r^2 \qquad (9)$$

$$W = \rho \frac{4}{3} \pi r^3 \qquad (10)$$

if the sphere is broken into n smaller units of average radius r_2 then

$$\frac{m_1}{m_2} = \frac{r_1^3}{n\, r_2^3} = 1 \qquad \text{hence} \qquad \frac{r_1}{r_2} = n^{1/3}$$

Taking the ratio of surface areas

$$\frac{A_{s1}}{A_{s2}} = \left(\frac{r_1}{r_2}\right)^2 = n^{2/3} \qquad (11)$$

Effects of fracture on Equilibrium Temperature

Again considering a simple radiation cooled sphere

$$\dot{q} = P_D\, m = \sigma \epsilon\, T_s^4\, A_s \qquad (12)$$

Where

$\dot{q}$ = heat flux generated by fuel

P_D = power density watts/#

m = mass #

σ = Stephan Boltzmann Constant

ϵ = emissivity

T_s = surface temp.

A_s = surface area

hence, from (12)

$$T_s = \left[\frac{P_D m}{\sigma \epsilon A_s}\right]^{1/4} \qquad \text{or} \qquad \frac{T_{s1}}{T_{s2}} \simeq \left[\frac{A_2}{A_1}\, \frac{\epsilon_2}{\epsilon_1}\, \frac{m_1}{m_2}\right]^{1/4} \qquad (13)$$

[redacted]

[redacted]

Since $m_2 = nm_1$ and $\epsilon_2 = \epsilon_1$

$$\frac{T_{s1}}{T_{s2}} \simeq \left[n^{1/3}\right]^{1/4} \qquad (14)$$

Equations (11) and (14) representing the change in surface area and temperature are plotted in Figure 79. Considering an approximation of evaporation rates from the Langmuir equation

$$\dot{W} = \frac{P_v}{17.14}\rho\sqrt{\frac{m}{T}} = \frac{gm}{cm^2 \cdot sec} \qquad (15)$$

for any given spherical particle

$$m = \frac{4}{3}\pi r^3 \rho$$

$$\frac{dm}{dt} = 4\pi r^2 \rho \frac{dr}{dt} \qquad (16)$$

hence

$$\rho\frac{dr}{dt} = \frac{dm}{dt}\left(4\pi r^2\right)^{-1} = \dot{W} \text{ from Langmuir equation.} \qquad (17)$$

Then the time for any given spherical particle to evaporate completely in vacuum can be given as

$$-\int_{r_o}^{o} dr = \frac{\dot{W}}{\rho}\int_{o}^{t} dt \qquad (18)$$

$$t = \frac{r_o \rho_m}{\dot{W}} \text{ where } \rho_m = \text{density of fuel} \qquad (19)$$

where r_o is the particle radius and can be expressed as

where $\bar{r}$ = Av initial device radius (20)
N = # of debris

Hence,

$$t_E = \frac{\bar{r}\,\rho_m}{n^{1/3}\,\dot{W}} \qquad (21)$$

[redacted]

[redacted]

Considering an initial isotope fuel block assembly of $\bar{r} \cong \frac{1}{2}$ ft = 15. 3 cm for Pu^{238}, Sr^{90}, and Po^{210} metals. Table F. 2-1 can be compiled.

Table F. 2-1

Material	m	N	T°K	W	T_2/T_1
Pu-238	20	1	913	240×10^{-7}	-1
		10			-. 83
		10^2			-. 70
		10^3			-. 58
		10^4			-. 47
					-. 38
Sr-90	2. 62	1	1040	1.45×10^{-5}	
		10			
		10^2			
		10^3			
		10^4			
Po-210	9. 3	1	980	2.24×10^{-6}	
		10			
		10^2			
		10^3			
		10^4			

[redacted]

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F.2-4

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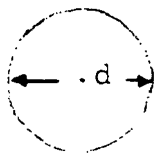
APPENDIX G

FUEL FORM EVAPORATION IN SPACE ENVIRONMENTS

Given a sphere of fuel of diameter "d", as shown below, then:



a) Heat Generation

$$\dot{q} = P_D m = \epsilon \sigma T^4 A_s$$

b) Mass:

$$m = \frac{4}{3}\pi \rho r^3$$

c) Surface area:

$$A_s = 4\pi r^2$$

then also

c) $$r = \frac{3\epsilon \sigma T^4}{P_D \rho}$$

d) Evaporation rate from b):

$$dm/dt/A_s = \dot{W} = \rho \frac{dr}{dt}$$

e) Evaporation rate from Langmuir Equation

$$\dot{W} = \frac{Pv}{c\sqrt{Nak}}\sqrt{\frac{M}{T}} \quad \text{where} \quad Pv = f(T)$$

Therefore, combining equation d) and e),

f) $$\frac{dr}{dt} = \frac{P v}{c \rho \sqrt{Nak}}\sqrt{\frac{M}{T}}$$

but

g) $$\frac{dr}{dt} = \frac{12\epsilon \sigma T^3}{P_D \rho}\frac{dT}{dt}$$ from equation c)

hence,

h) $$\frac{12\epsilon \sigma T^3}{P_D \rho}\frac{dT}{dt} = \frac{Pv}{c\rho \sqrt{Nak}}\sqrt{\frac{M}{T}}$$

whose solution yields,

i) $$\int_{Ti}^{Tf} \frac{T^{7/2}}{P v}\, dT = \frac{P_D \sqrt{M} t}{12\epsilon \sigma (2\pi Nak)^{1/2}}$$

[redacted]

[redacted]

Where:

$\dot{q}$ = Heat flux generated by nuclear decay in particle (watt)

P = Power density $\frac{(\text{watts})}{(\text{gm})}$

m = mass (gm)

ϵ = Emissivity

σ = Stephan Boltzmann Const. $\left(\frac{\text{erg}}{\text{cm}^2\ {}^{\circ}\text{K}^4\ \text{sec}}\right)$

T = Temperature (${}^{\circ}$K)

A_s = Surface area (cm^2)

r = Radius (cm)

P_v = vapor pressure (mm Hg)

N_a = Avogadro's No (mole^{-1})

k = Boltzmann Const. $\left(\frac{\text{erg}}{{}^{\circ}\text{K}}\right)$

M = Molecular wt (gm/mole)

$\dot{W}$ = Evaporation rate $\left(\frac{\text{gm}}{\text{cm}^2\text{-sec}}\right)$

ρ = Density

Solution of the temperature-time equation for any given particle number representing portions of some original mass will yield enough data to compute mass loss as a function of time.

Solutions for strontium-90, plutonium-238 and polonium-210 were carried out as follows:

Solution to:

$$\int_{T_i}^{T_f} \frac{T^{7/2}}{P_v}\, dT = \frac{P_D \sqrt{M}\ t}{12 \epsilon \sigma \sqrt{2 \pi N a k}}$$

Step (1) Determine temperature range of interest for a given mass size of the particular material.

[redacted]

[redacted]

(a) From a) $P_D m = \epsilon \sigma T^4 A_s$, we may solve for T_i and obtain

$$T_i = \sqrt[4]{\frac{P_D r_o \rho}{3 \epsilon \sigma}}$$

(b) This value of T_i varies as a function of radius. when the initial mass is broken into a number of smaller particles, the radius varies according to:

$$r_1 = \frac{r_o}{\sqrt[3]{n}}$$

r_o initial radius

r_1 new radius

n number of particles

Step (2) Determine vapor pressures in temperature range of interest. Vapor pressure equations vary with each material but are of the general form $\log_{10} P = \frac{A}{T} + B$

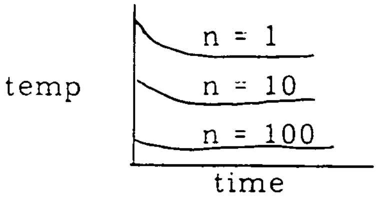
where A and B are constants

Step (3) Since $\int_{T_i}^{T_f} \frac{T^{7/2}}{Pv(T)} dT$ is not readily integrable, a numerical solution was conducted by placing the integral equal to a series

$$\int_{T_i}^{T_f} \theta \, dT = \frac{1}{2} \sum_{i}^{f} \left[\theta_i + \theta_{(i+1)} \right] \Delta T_i$$

where $\theta = \frac{T^{7/2}}{Pv}$

Step (4) On the right hand side of the differential equation $\frac{P_D \sqrt{M}}{12 \epsilon \sigma \sqrt{2\pi} \, Nak}$ is a constant for the particular material. Time is found by dividing this number into the value of the integral over a particular temperature range. This data is then plotted on a temperature versus time curve for the various numbers of particles.



[redacted]

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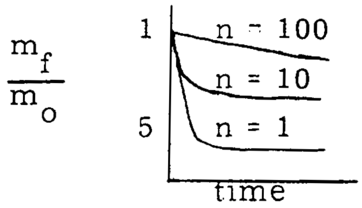
Step (5) The mass loss ratio, $^{mf}/mo$, is found by solving $P_D m = \epsilon \sigma T^4 A_s$ for m as a function of r and T. The ratio becomes:

$$\frac{mf}{mo} = \left(\frac{T_f}{T_o}\right)^4 \left(\frac{r_f}{r_o}\right)^2$$

The value of r is found by

$$r = \frac{3 \epsilon \sigma T^4}{P_D \rho}$$

Step (6) Mass loss ratio may now be plotted as a function of time by using the above equations and the temperature time plot for various numbers of particles.



Evaporation Characteristics of Sr, Pu, and Po

$$\int_{T_i}^{T_f} \frac{T^{7/2}}{P_v(T)}\, dT = \frac{P_D \sqrt{M}}{12\, \epsilon\, \sigma\, 2} \; \frac{t}{N_a k}$$

I. Sr-90

A. Properties

1) P_D = .95 watts/gm

2) 1657°K boiling point

3) ρ = 2.63 gm/cm^3

B. Vapor pressure given by $\log_{10} P = \frac{-7300}{T(^\circ K)} + 7.30$

C. Selecting 15.24 cm as initial material radius, T_i limits are:

[redacted]

[redacted]

1)	@n = 1	$T_i = 1220^\circ K$	$P(T) \approx 17$ mm
2)	@n = 10	$T_i = 1000^\circ K$	$P(T) \approx 1$ mm
3)	@n = 100	$T_i = 828^\circ K$	
4)	@n = 1000	$T_i = 688^\circ K$	

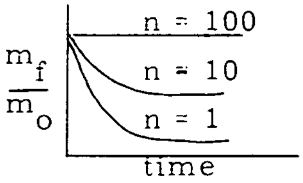
D. Remarks

1) In this temperature range, the Sr is molten.

2) Time/temperature curve shows the greatest rate of temperature loss for the material left in one piece, decreasing to practically zero for the material broken into 100 and 1000 pieces.

3) The same general remarks as above are true for the mass loss/time curve. m_f/m_o decreases at a much greater rate for material left whole than for the material broken into 10 or 100 pieces.

4) If the mass loss curve is carried out for a sufficiently long time, each of the cases, n = 1, n = 10, n = 100, would eventually reach the same point of m_f/m_o with n = 1 being first; n = 10, second; etc.



5) Depending on length of time considered, it appears that, in order to destroy an amount of molten Sr, it is more advantageous to leave the material in one piece to take advantage of the higher mass loss rate. If a very long time period, several thousand seconds, is considered any of the modes would be acceptable.

II. Pu-238

A. Properties

1) $P_D = .55$ watts/gm

2) $\approx 2400^\circ K$ boiling point

3) $\rho = 19.86\ gm/cm^3$

[redacted]

[redacted]

B. Vapor pressure given by $\log_{10} P \text{ (mmHg)} = \frac{-17,587}{T(^{\circ}K)} + 7.895$

C. Selecting 15.24 cm as initial material radius T_i limits are:

1)	@n = 1	$T_i = 1770^{\circ}K$	$Pv(T) \approx 3.5 \times 10^{-3}$ mm
2)	@n = 10	$T_i = 1470^{\circ}K$	$Pv(T) \approx 2.2 \times 10^{-5}$ mm
3)	@n = 100	$T_i = 1210^{\circ}K$	

D. Remarks

1) Plutonium is in a molten state in this temperature range.

2) Vapor pressures of plutonium are considerably lower than those of strontium (by a factor of $10^4 - 10^5$) thus affecting the magnitude of the integral $\int \frac{T^{7/2}}{Pv} dT$.

3) Because of the greater size of the integral, the time interval between temperature changes is longer by an order of $10^4 - 10^5$. This affects the time-temperature curve in that the rate of temperature change is negligible for at least 10^4 seconds for a unit mass.

4) Since mass loss rate is directly related to temperature change, there is no appreciable mass loss until 10^4 seconds.

5) Temperature loss rate is even lower for the mass broken into a number of particles thereby affecting the mass loss ratio in the same way.

6) Vapor pressure seems to have the greatest effect on evaporation mass loss rate for plutonium.

7) If carried out over a long enough time interval ($10^8 - 10^{10}$ seconds) the same characteristics as those observed with strontium with regard to effect of particle number will be observed, i. e., each n-case will eventually reach the same m_f/m_o.

[redacted]

8) As was the case with the strontium, it appears that leaving the mass in one piece is more advantageous than breaking it into many particles because of the higher mass loss rate. However, the extremely long time interval necessary to achieve appreciable temperature and mass loss, regardless of particle number, is a significant consideration.

III. Po-210

A. Properties

1) P_D - 144 watts/gm
2) 1235°K boiling point
3) ρ=9.29 gm/cm^3

B. Vapor pressure given by $\log_{10} P \text{ (mmHg)} = \frac{-5377}{T(^{\circ}K)} + 7.2345$

C. Selecting 15.24 cm as initial material radius, T_i limits are:

1) @ n = 1 $T_i = 6000^{\circ}K$
2) @ n = 10 $T_i = 4950^{\circ}K$
3) @ n = 100 $T_i = 4100^{\circ}K$

D. Remarks

1) Each of the above temperatures is well over the boiling point of polonium which means that the material is gaseous upon release, therefore, there is no need to carry out a mass loss due to evaporation calculation.

2) Even if the temperatures were below the boiling point, the vapor pressure equation indicates extremely high vapor pressures (of the order of 10^5 - 10^7 **mmHg**). High vapor pressure causes a high temperature and mass loss rate.

3) The above remarks indicate the easy destruction of polonium fuel.

[redacted]

[illegible]

[illegible]

[redacted]

[redacted]

APPENDIX H

SPACE VACUUM EFFECTS ON FUEL CAPSULE DESIGN

Fuel containment structures or other system components that are exposed to high vacuums in space must retain their strength and integrity if eventual impact after reentry is expected. For example, if intact reentry and fuel containment upon impact are to be achieved, excessive losses of material from the structure must be prevented, or the structure must be overdesigned to account for the inherent losses. Therefore, it is evident that when metals or alloys are used in space, their evaporation characteristics must be thoroughly examined in order to eliminate premature failure. The evaporation rate is sensitive to temperature, the vapor pressure of the metal, and naturally, the vacuum to which it is exposed. Depending upon the mission, the material may be exposed to widely varying vacuum conditions, as shown by Table H-1.

Table H-1

Characteristics of Space Environment (Reference H-1)

Altitude	Pressure (mm Hg)	Concentration molecules atoms or ions cm^3	Composition
Sea level	760	2.5×10^{19}	78% N_2, 21% O_2, 1% A
19 miles	9	4×10^{17}	N_2, O_2, A
125 miles	10^{-6}	10^{10}	N_2, O, O_2, O^+
500 miles	10^{-9}	10^{6}	O, He, O^+, H
4000 miles	10^{-13}	10^{3}	H^+, H, He^+
Above 14000 miles	$< 10^{-13}$	10^{1} - 10^{2}	85% H^+, 15% He^{++}

[redacted]

[redacted]

Vacuum effects have long been studied and evaluated on metals, the basis of which has been the work of Langmuir. It is difficult, if not impossible, to determine vacuum effects upon an alloy by the Langmuir approach because it is based upon the properties of a single element not the properties of the alloy. Therefore, to determine how a typical fuel capsule material might react in space, some experimental data was obtained. The work of Bourgette at Oak Ridge National Laboraty (Reference 2) was of particular interest in that several Iron, Nickel, and Cobalt-base alloys in the 760 to 900°C range and in a vacuum range of 5×10^{-7} to 5×10^{-9} mm Hg were studied. Both the temperature and the vacuum test ranges are typical of the operating conditions of a SNAP device encountered in earth orbital applications. Therefore, to illustrate space vacuum effects and to establish the behavior of typical fuel capsule materials, parts of Bourgette's work will be presented below.

Evaporation losses from Haynes-25, 316 and 416 stainless steel, INOR-8 and Inconel are illustrated in Figure H-1. The curves show that the evaporation rates become practically constant after approximately 50 hours. Their analysis showed that grain boundaries beneath the surface exhibited severe material losses, subsurface voids, grain boundary grooving and grain growth. Also, in multicomponent alloys containing high vapor pressure alloying components and interstitial impurities, the grain boundaries became enriched in these elements and impurities, thus producing a degree of disorder at the grain boundary and promoting greater evaporation rates when compared to the grain themselves. The process involved in mass losses is clearly illustrated in Figure H-2, the resultant effect upon the material being the loss of its usefulness in the design.

The effects of high vacuum may be reduced by preoxidizing the material, if desirable for the particular use. For example, the experimental data presented in Table H-2 shows that preoxidizing in air slows down the evaporation process appreciably. Another attempt at minimizing the effect of evaporation was made by preoxidizing in wet hydrogen. This proved to result in evaporation rates between those for no preoxidation and those for air preoxidation.

Even with preoxidizing in air, the evaporation rates are not negligible for long duration missions, particularly for those missions which would subject the material to higher vacuum, e.g., lunar, solar, and deep space.

To illustrate what these evaporation rates mean in terms of fuel containment structure design, an analysis of a typical fuel capsule was performed.

Haynes-25 Fuel Capsule Evaporation

Cylindrical Capsule : diameter = 1 in. (mean)
length = 8 in.
wall thickness = 0.1 in.

Evaporation rate = 6.65×10^{-3} mg/cm^2-hr @ 982°C

[redacted]

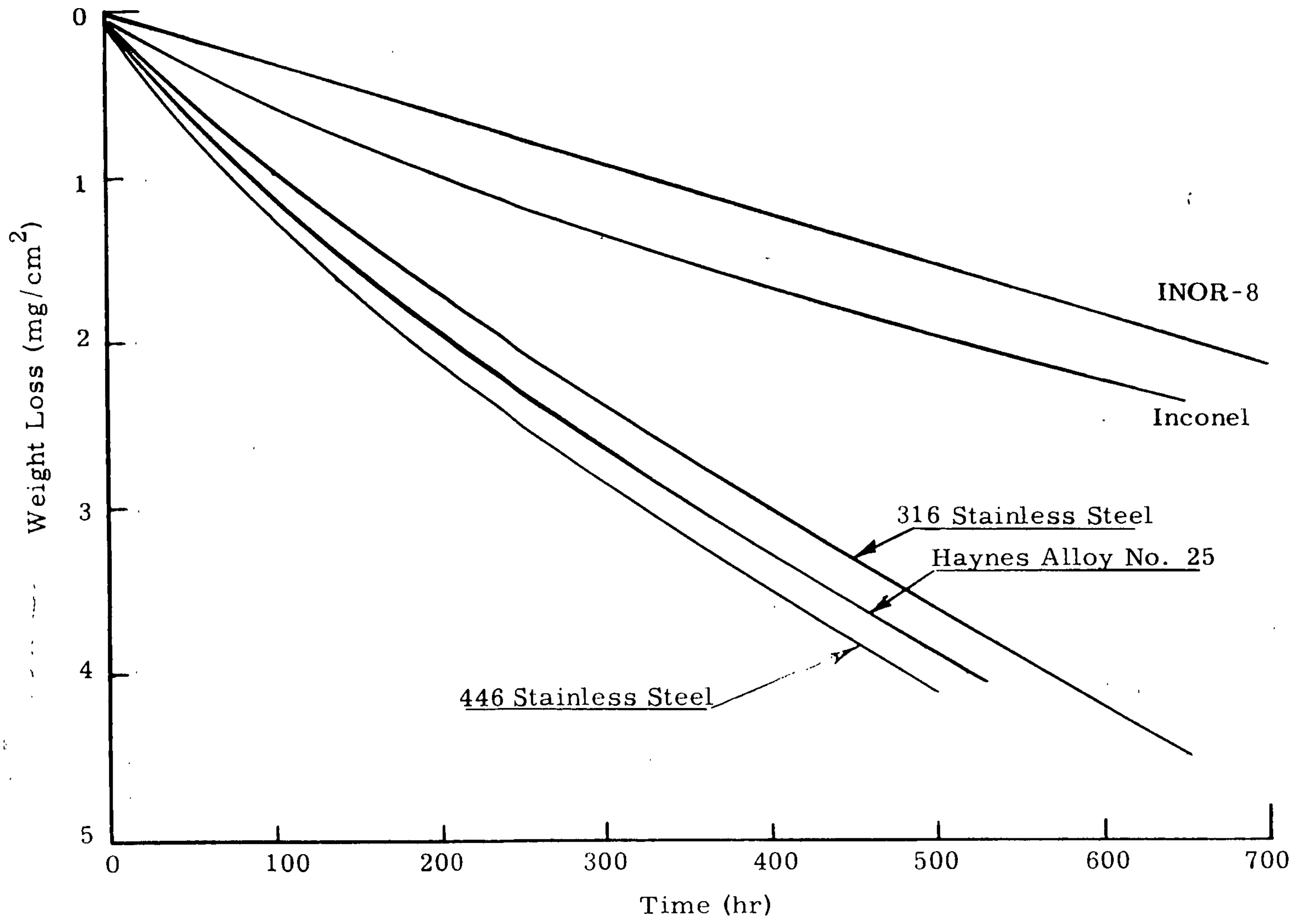
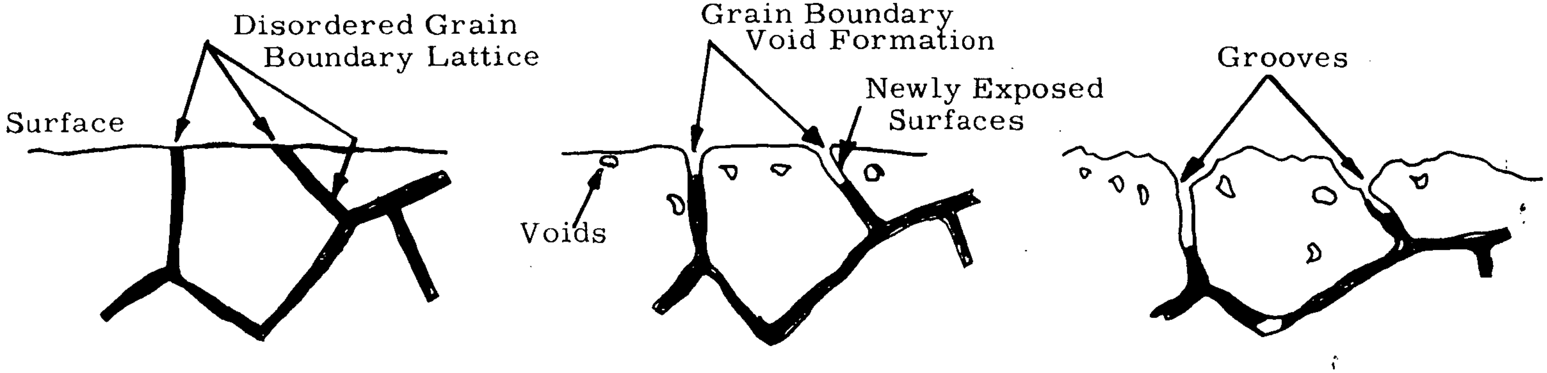


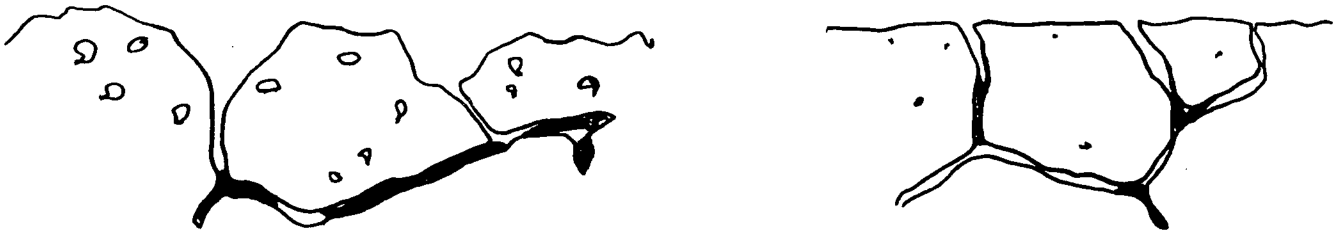
Figure H-1 Evaporation Losses of Iron-, Nickel-, and Cobalt-Base Alloys at 872 and 982°C and Approximately 5 x 10^{-7} to 5 x 10^{-9} torr.



(a) As Polished Surface

(b) Beginning of Grain Boundary Evaporation and Boid Formation

(c) Beginning of Surface Roughing and Formation of Grain Boundary Grooves



(d) Continued Widening of Grooves, Void Formation, and Grain Boundary Evaporation

(e) After Long Times, Production of Smooth Surfaces and Nearly Total Grain Boundary Evaporation with Void Coalescence Accompanied by Specimen Thinning

Figure H-2. Evaporation Sequence of an Alloy Exposed to High Vacuum.

[redacted]

Table H-2

Evaporation Rates of Common Space Applicable Materials at 5×10^{-7} to 5×10^{-8} Torr

	No Preoxidation				Oxidized in Air	
	872°C		982°C		982°C	
Alloy	Initial	Final	Initial	Final	Initial	Final
INOR-8	1.54	1.54	30.8	30.8	8.5	25.0
Inconel	7.57	2.89	59.7	33.5	36.3	30.5
Type 316 Stainless Steel	30.0	6.59	114.0	66.4	1.94	6.65
Haynes Alloy No. 25	58.5	5.66	140.0	69.2	66.5	3.34

Notes:

1) Initial: Maximum rate during first 50 hours of test.

2) Final: Rate during last 50 hours of test.

[redacted]

$$\text{Time for complete Evaporation} = \frac{\text{amount of material (gm)}}{\text{area exposed}(\text{cm}^2) \times \text{rate of loss}\left(\frac{\text{gm}}{\text{cm}^2\text{-hr}}\right)}$$

$$= \frac{147.8 \text{ gm}}{6.38 \text{ cm}^2 \times 6.65 \times 10^{-6} \frac{\text{gm}}{\text{cm}^2\text{-hr}}}$$

$$= 3.47 \times 10^5 \text{ hrs} \quad \text{or}$$

$$= 39.6 \text{ years}$$

Obviously, the capsule would fail before the elapse of 39.6 years, since this time period represents the time for complete loss of material. The question that arises is: "How long can a fuel capsule be subjected to a space environment and still satisfy safety requirements such as containment upon earth impact?" This is of vital importance for any system designed for intact reentry and containment which is used in short lived earth orbits, the effect being of greater concern to missions utilizing long-lived isotopes, such as Pu-238 (half-life = 89.6 years). For example, the fuel capsule would be completely consumed via evaporation for missions with orbital altitude above those shown below:

Fuel Capsule Ballistic Coefficient (lb/ft^2)	Orbital Lifetime (yrs)	Orbital Altitude (n. mi.)
10	40	450
20	40	420
50	40	380

This illustrates that if containment for several half-lives (consider Pu-238) is to be achieved, considerable overdesign of the capsule is required. The significance of this study is that the effects of high vacuum must be thoroughly evaluated in designing a fuel containment structure or other system components that might fail, negating nuclear safety. Specifically, items include:

(1) Rate of evaporation - operating temperature, vapor pressure of elements in the alloy or mission environment

(2) Surface and subsurface effects - particular grain boundary and internal voids.

(3) Effect on strength - tensile and impact strength.

(4) Exposure time - consider at a minimum mission duration time.

[redacted]

REFERENCES

1. Jaffe, L. D., and Rittenhouse, T. B., "Evaporation Effects on Materials in Space", Materials for Missiles and Spacecraft, ed. E. R. Parker, McGraw-Hill Book Co., Inc., p. 253.

2. Bourgette, D. T., "Evaporation of Iron-Nickel, and Cobalt-Base Alloys a5 760 to 980°C in High Vacuums", Oak Ridge National Laboratory, ORNL-3677.

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